T. O. 1F-100D-3

TECHNICAL MANUAL

STRUCTURAL REPAIR

USAF SERIES

F-100D AND F

AIRCRAFT

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LIST OF EFFECTIVE PAGES

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4-60A Added	19 Dec 58
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4-62A thru 4-62C Added	6 Sep 57
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INTRODUCTION

Note

The information and illustrations contained in this handbook apply to both F-100D and F-100F Airplanes except as noted.

The Handbook of Structural Repair Instructions is designed to guide personnel in handling damaged airplanes, in evaluating the damage, and in accomplishing the necessary repairs. Effective use of the information given will reduce to a minimum the time an airplane is out of commission. In order to ensure effective use of the handbook, responsible structural repair personnel should become generally familiar with its contents before an actual need arises to use it. Making an accurate evaluation of the damage and accomplishing the appropriate repairs is often a complex and difficult task.

The information given is intended to aid and direct these efforts. Sound judgment in applying the information to the job at hand is the responsibility of repair personnel.

Evaluation of the damage serves a twofold purpose. First, the nature and extent of damage will determine the type of facility necessary for its repair. Airplanes damaged beyond the repair capabilities of the base where the damage occurred will have to be transported to a facility capable of making the repairs. Transit time and distances will be cut by scheduling the airplanes to the proper repair bases. Second, by knowing the condition of the airplanes and the parts damaged, the time and effort required to get replacement parts will be reduced to a minimum.

Repairs outlined provide structural maintenance coverage for the airplane. The repairs are not based on any certain extent of damage, since the scope of possible damage is so broad. It is impractical to outline procedures because, in many cases, it will be necessary to alter the repair to suit the specific situation. The material requirements, rivet specifications, etc, are the minimum allowable. For repairs of important primary structure, when excessive alteration of the repair would be required, the structural repair officer should be consulted.

The repair figures in each section are preceded by a group repair index for the section. In this index, the group is broken down into its major components, and reference is made to the repair indexes for each component. Component repair indexes and accompanying keys show the structural breakdown of the component and give the North American drawing number, descriptive nomenclature, material and gage, and repair figure numbers for each item. The repair figures listed follow the component repair index except when the repair may be used on more than one component. In this case, it is shown in one place only. Suitable reference is made in all appropriate keys.

HOW TO FIND INFORMATION

1. Look in the Table of Contents at the front of the manual.

2. Turn to the page number of the desired section to find the Table of Contents for the section.

3. To find a repair for a damaged component, turn to the group repair index (listed in the Table of Contents) and find the figure number of the repair index for that component. Item numbers on the component repair index correspond with item numbers in the key. Applicable repairs are found in the right-hand column of the key.

4. To select a typical repair, refer to the instructions in the first part of Section X.

5. If you are looking for one specific item or procedure, refer to the alphabetical index in the back of the manual.

SUPPLEMENTARY INFORMATION

For supplementary information on operation, maintenance, parts listing, weight and balance, and inspection of the F-100D and F-100F Airplanes, refer to the following publications:

Index of Technical Publications	T.O. 1F-100C(I)-01	T.O. 1F-100C(I)-01
Flight Manual	T.O. 1F-100D(I)-1	T.O. 1F-100D(I)-1
General Airplane	T.O. 1F-100D(I)-2-1	T.O. 1F-100F(I)-2-1
Fuel and Utility Systems	T.O. 1F-100D(I)-2-2	T.O. 1F-100F(I)-2-2
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Illustrated Parts Breakdown	T.O. 1F-100D(I)-4	T.O. 1F-100F(I)-4
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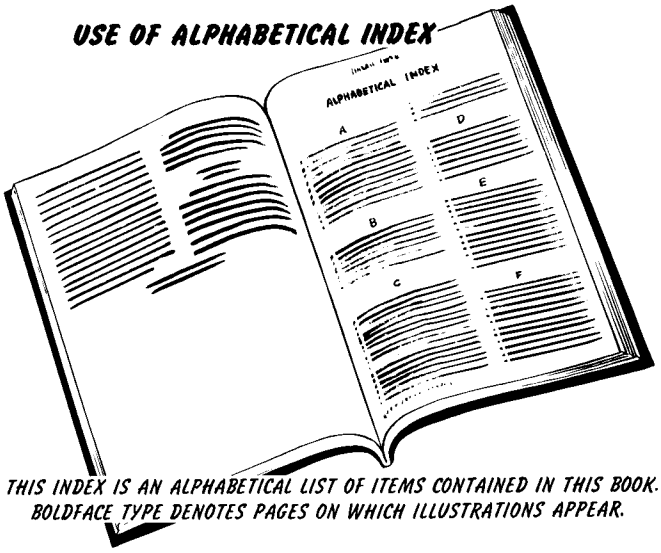
Basic Weight Check List and Loading Data	T.O. 1F-100D-5	T.O. 1F-100F-5
Technical Manual of Inspection Requirements	T.O. 1F-100C(I)-6	T.O. 1F-100C(I)-6

Throughout this manual where supplementary information on operation, maintenance, etc, is required, the T.O. referenced is for F-100D Airplanes; however, if the airplane being repaired is an F-100F, use the corresponding F-100F T.O. Also, if the airplane has been modernized by the "High Wire Project," refer to a "Project High Wire" T.O. for supplementary information. "Project High Wire" T.O.'s are identified by an (I) in the T.O. number, e.g., T.O. 1F-100D(I)-2-1, and T.O. 1F-100F(I)-2-1, etc.

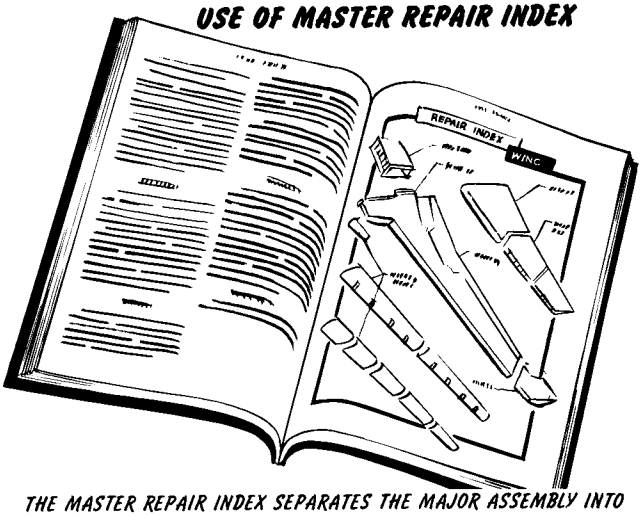
USE OF HANDBOOK...

USE OF ALPHABETICAL INDEX



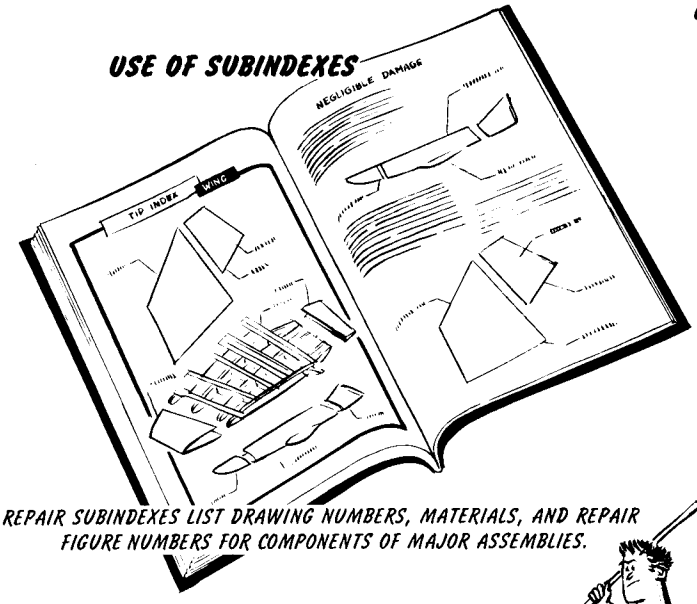
THIS INDEX IS AN ALPHABETICAL LIST OF ITEMS CONTAINED IN THIS BOOK. BOLDFACE TYPE DENOTES PAGES ON WHICH ILLUSTRATIONS APPEAR.

USE OF MASTER REPAIR INDEX

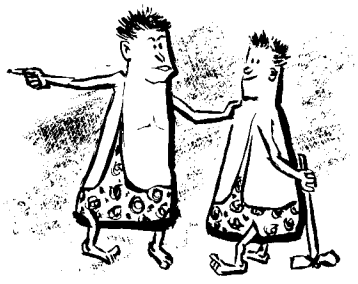


THE MASTER REPAIR INDEX SEPARATES THE MAJOR ASSEMBLY INTO COMPONENTS AND REFERS TO SUBINDEXES FOR THESE COMPONENTS.

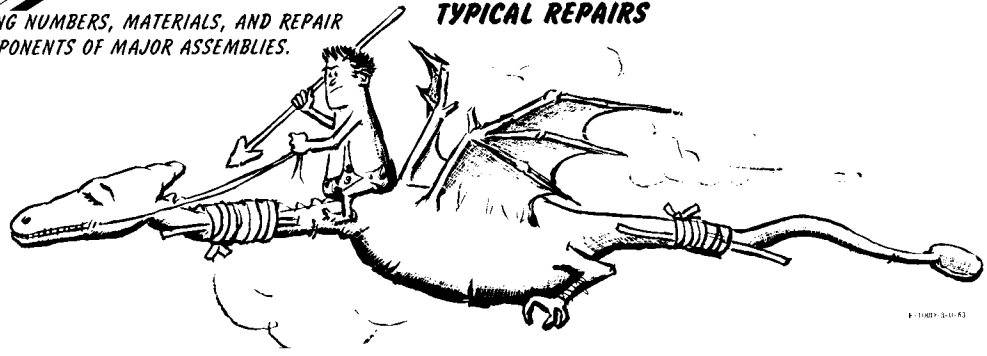
USE OF SUBINDEXES



REPAIR SUBINDEXES LIST DRAWING NUMBERS, MATERIALS, AND REPAIR FIGURE NUMBERS FOR COMPONENTS OF MAJOR ASSEMBLIES.

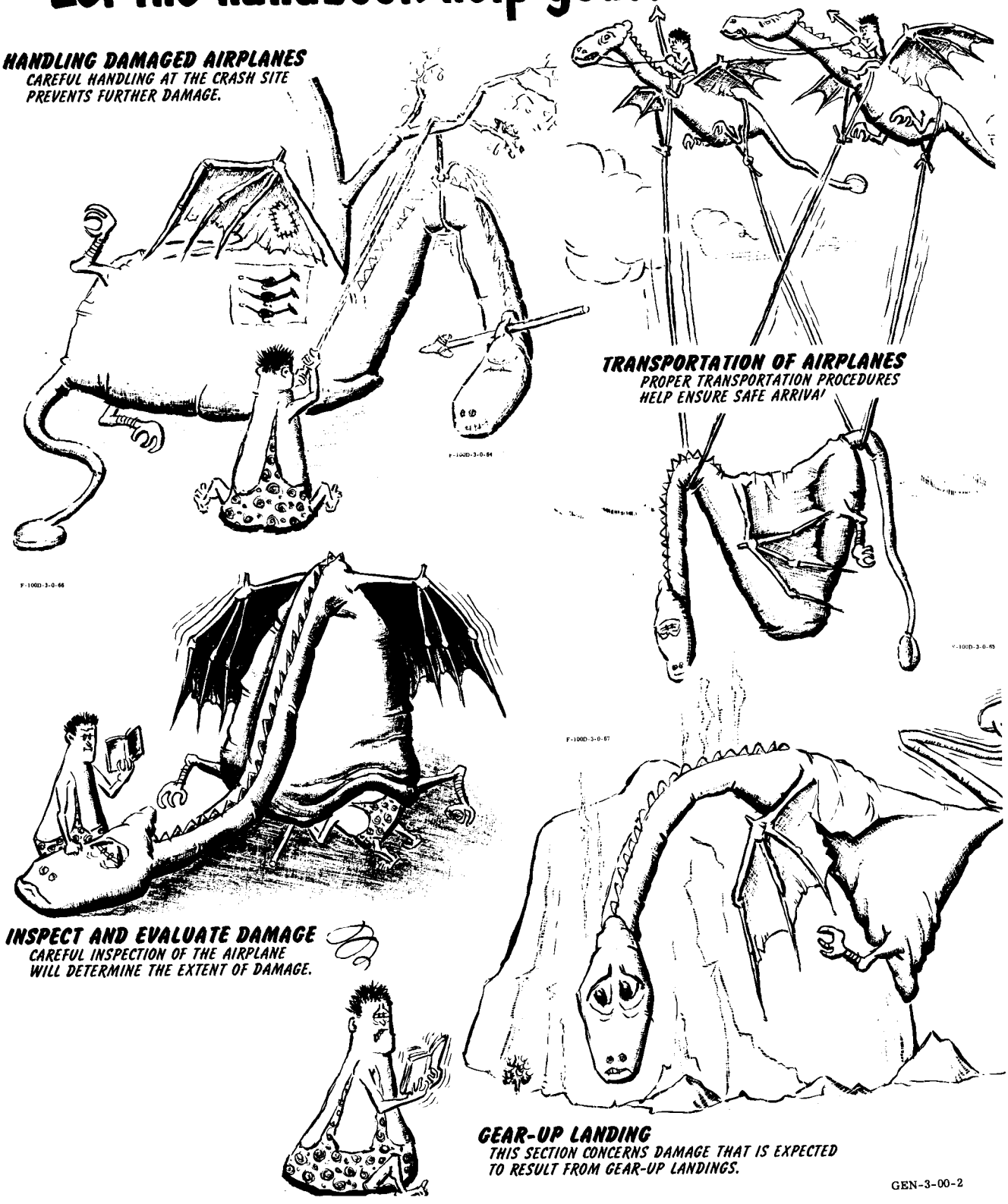


TYPICAL REPAIRS



TYPICAL REPAIRS ARE DESIGNED FOR WIDE AND VARIED APPLICATION THROUGHOUT THE AIRPLANE.

Let the handbook help you...



HANDLING DAMAGED AIRPLANES
CAREFUL HANDLING AT THE CRASH SITE PREVENTS FURTHER DAMAGE.

TRANSPORTATION OF AIRPLANES
PROPER TRANSPORTATION PROCEDURES HELP ENSURE SAFE ARRIVAL

INSPECT AND EVALUATE DAMAGE
CAREFUL INSPECTION OF THE AIRPLANE WILL DETERMINE THE EXTENT OF DAMAGE.

GEAR-UP LANDING
THIS SECTION CONCERNS DAMAGE THAT IS EXPECTED TO RESULT FROM GEAR-UP LANDINGS.

GEN-3-00-2

HOW TO REVISE THIS HANDBOOK

The airplanes you are working with do not remain static in design. Improvements and design changes are taking place through the entire life of the airplane.

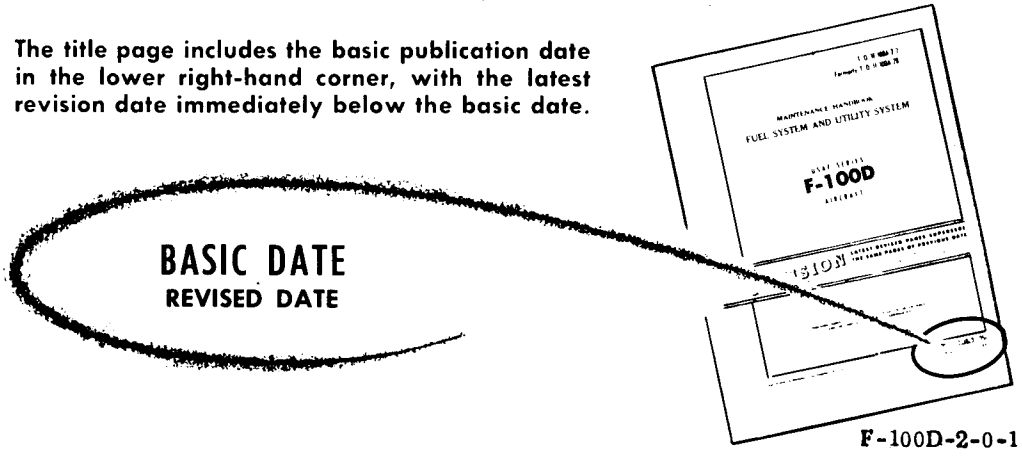
To reflect these changes, this handbook is constantly being revised. It is either revised or reissued every 90 days. The following steps will make sure you have the latest handbook and the proper method of keeping it revised.

STEP 1 **Make sure you have the latest issue of the book, including all outstanding revisions. To check this, refer to T.O. 0-1-1, Numerical Index of Technical Publications. Find Model F-100D Series listing of Technical Orders. Here you will find the basic issue date for this book, the date of the latest revision, and the T.O. number and title of the book. The dates given should be identical to the dates appearing on the title page of your book. If they are not, your book is not up-to-date.**

STEP 2 **Notify your Engineering Officer if your book is not up-to-date. He can obtain the latest issue of your handbook by ordering.**

STEP 3 **When you receive revisions for your handbook, be sure you insert them in your book as soon as possible and properly.**

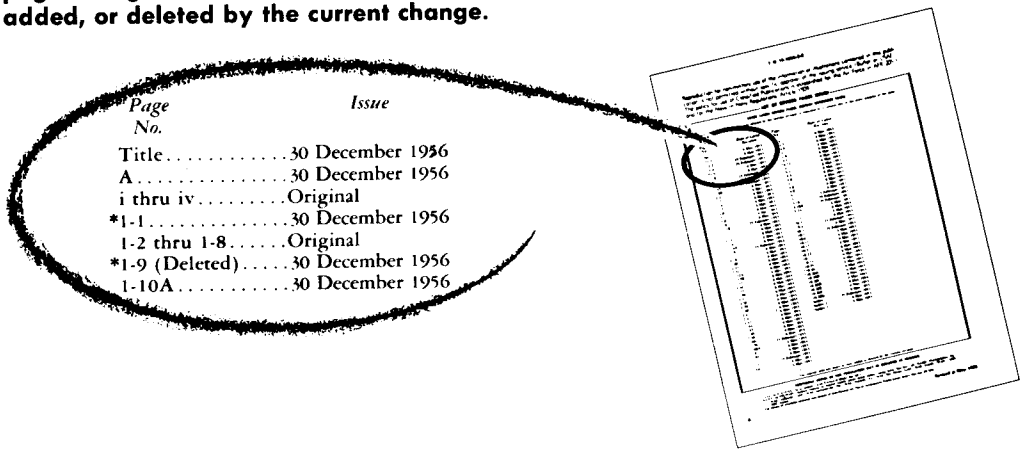
1 The title page includes the basic publication date in the lower right-hand corner, with the latest revision date immediately below the basic date.



F-100D-2-0-1

STEP 3 (CONTINUED)

2 **The "A" page shows what the changed status of the manual should be since the last revision or basic issue of the manual was made. All pages are listed, whether they be original, changed, added, or deleted. The dates of changed, added, or deleted pages are given in the column headed "Issue." The asterisk indicates pages changed, added, or deleted by the current change.**



Page No.	*Issue*
Title	30 December 1956
A	30 December 1956
i thru iv	Original
*1-1	30 December 1956
1-2 thru 1-8	Original
*1-9 (Deleted)	30 December 1956
1-10A	30 December 1956

3 **Each changed page carries the changed date on the lower right or left corner.**

4 **Replace all old and like-numbered pages in your manual with the new changed pages. Insert all "A" and "B" pages in alphabetical order following page with same basic page number.**

Example: Pages 3-14A, 3-14B, etc, should follow page 3-14.

5 **Text changes or additions on each new changed page are indicated by a black vertical line placed opposite the text lines changed. Read lines of text so marked, and compare them with same lines on the old page. This will give you a simple and quick method to determine a change in information. The remainder of the new page will be identical to the page it is replacing.**

6 **Check bottom spacing of each new page for other change notices. Sometimes it may be necessary to delete whole pages of information. These pages are not replaced with new pages. In this case, you will find such a notation on the changed page which precedes or follows the pages requiring deletion. For example, a notation on page 3-14 may say that pages 3-15 through 3-20 are deleted. If any illustrations appeared on the pages being deleted, the figure numbers of the deleted illustrations are also given in the deletion notation. Remove these pages from your manual as they contain obsolete information.**

7 **When a complete set of pages are issued, this means the entire manual has been revised. A note appears on the new title page to that effect. In this case, remove the old pages from your manual and insert the new pages.** F-100D-2-0-2C

HOW T.C.T.O.'s AFFECT MAINTENANCE MANUALS

The following information is to help acquaint you with how Time Compliance Technical Orders (T.C.T.O.'s) affect the Maintenance Manual. T.C.T.O.'s are normally written for airplane improvement, and some of them affect the Repair Manual. T.C.T.O.'s that do not affect manual procedures are not mentioned. When one is issued that changes or modifies a system or part, this information together with the old information is included in the manual, and is coded for the airplanes affected.

Both groups of information are kept in the manual until the T.C.T.O. is rescinded. The automatic rescission date for Routine Action Category 1 T.C.T.O.'s is as follows: (1) for all safety T.C.T.O.'s and all T.C.T.O.'s in categories 1, 21, and 31—one year; (2) for all other T.C.T.O.'s—2 years. By the rescission date, all airplanes should be changed to conform to the T.C.T.O. When the T.C.T.O. is rescinded, the information considered the older information, or information not conforming to the T.C.T.O., is usually removed. This leaves only the latest information in the manual. The coding for the newer information or for information conforming to the T.C.T.O., is also removed.

T.C.T.O. IDENTIFICATION

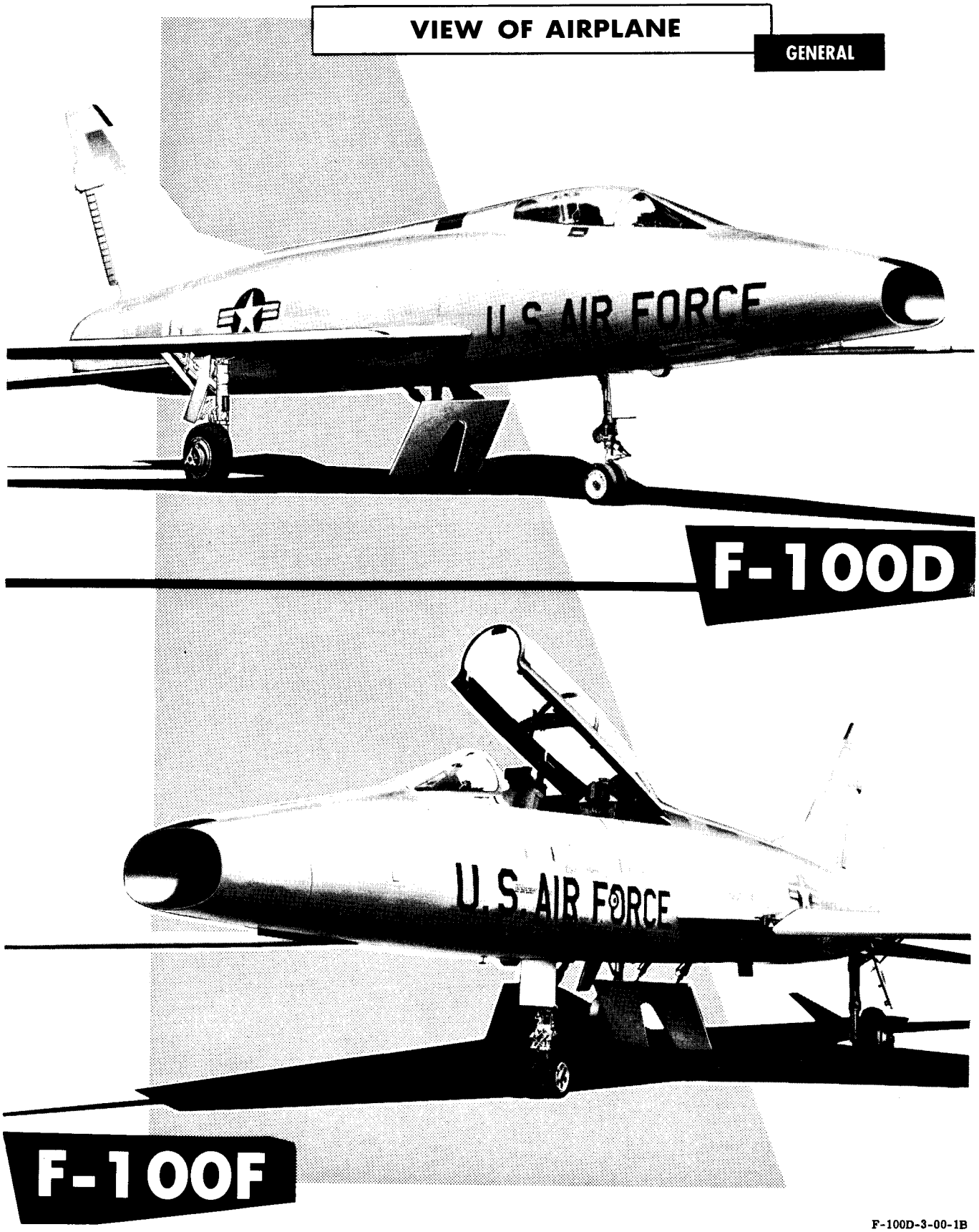
Pertinent repair instructions resulting from the following T.C.T.O.'s and applicable supplements thereto are included in this manual. A T.C.T.O. supplement is not listed unless it was necessary to code the information and/or procedures in this manual because of a change caused by the supplement.

This is not a complete T.O. listing and does not include rescinded T.C.T.O.'s. Refer to T.O. 0-1-1-4 for complete listing of T.C.T.O.'s for these airplanes.

T.O. NUMBER	SUBJECT
1F-100-1053	Modification of Fuselage Upper Longeron—F-100 Series Aircraft

VIEW OF AIRPLANE

GENERAL



F-100D-3-00-1B

View of Airplane

Section I

GENERAL

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TYPE OF CONSTRUCTION.

The North American F-100D Super Sabre is an operational, supersonic airplane with fighter-bomber capabilities. Its most distinctive external feature is the 45-degree sweepback of wing and empennage. Other design features include a thin lip engine air inlet duct,

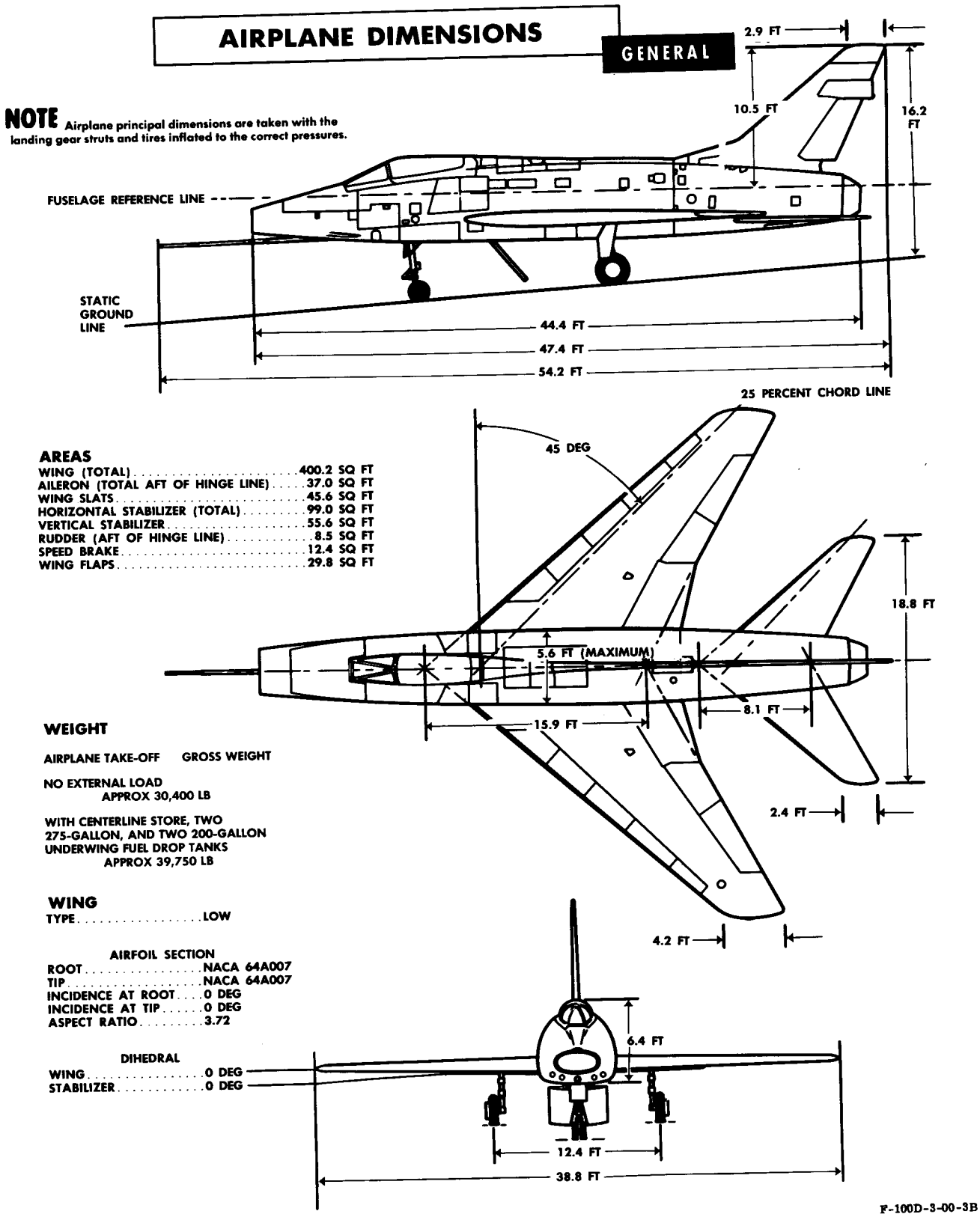
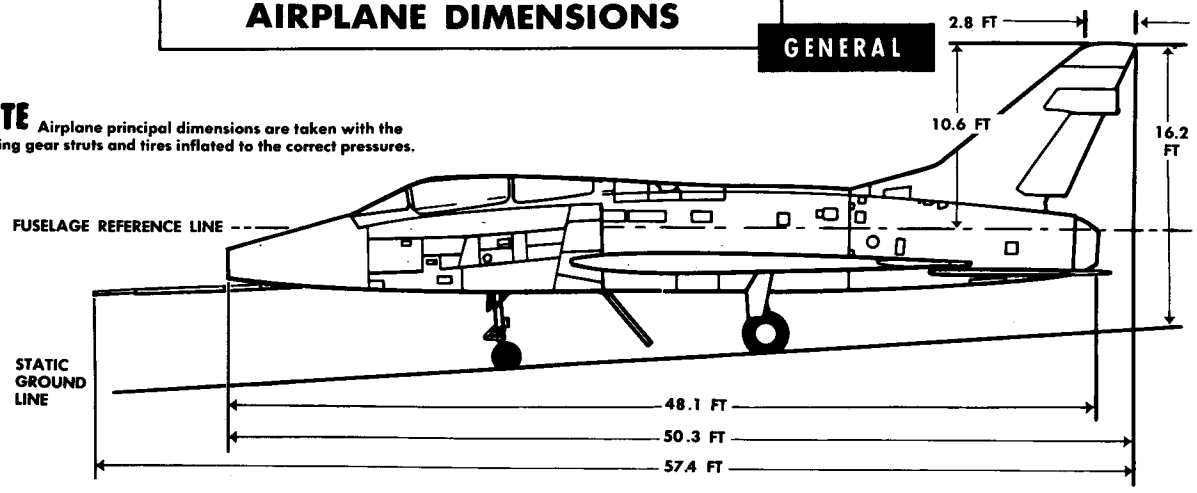


Figure 1-1. Airplane Dimensions—F-100D Airplane

AIRPLANE DIMENSIONS

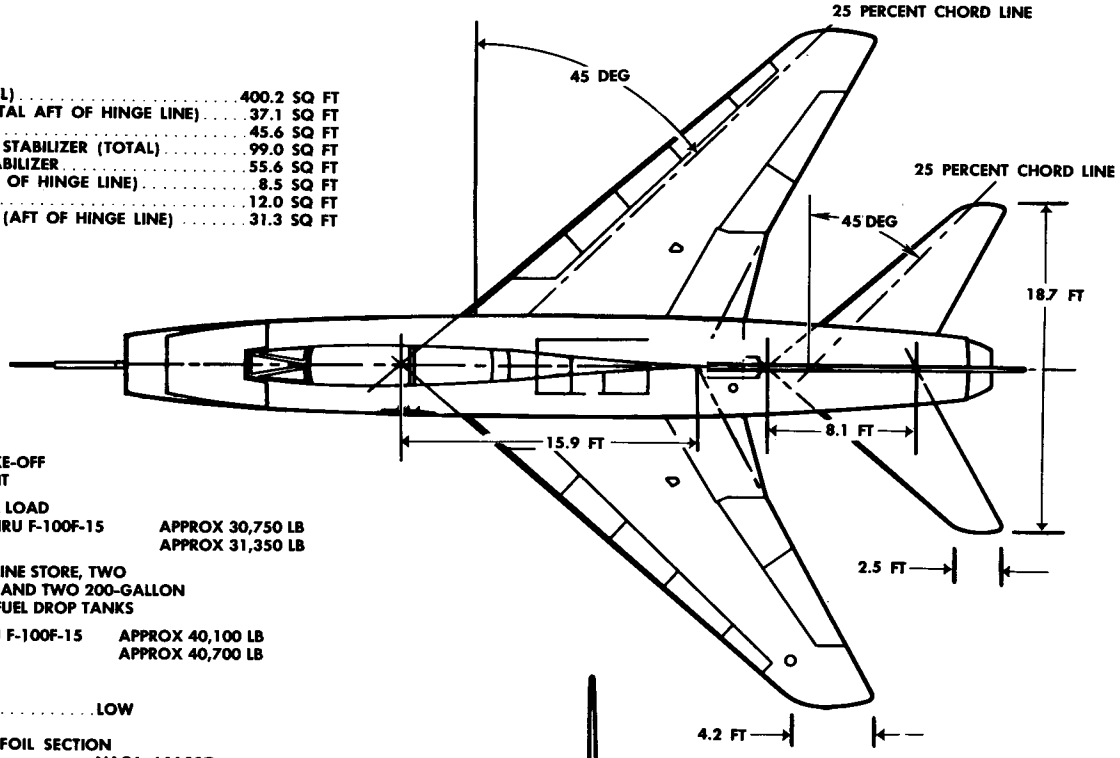
GENERAL

NOTE Airplane principal dimensions are taken with the landing gear struts and tires inflated to the correct pressures.



AREAS

WING (TOTAL)	400.2 SQ FT
AILERON (TOTAL AFT OF HINGE LINE)	37.1 SQ FT
WING SLATS	45.6 SQ FT
HORIZONTAL STABILIZER (TOTAL)	99.0 SQ FT
VERTICAL STABILIZER	55.6 SQ FT
RUDDER (AFT OF HINGE LINE)	8.5 SQ FT
SPEED BRAKE	12.0 SQ FT
WING FLAPS (AFT OF HINGE LINE)	31.3 SQ FT



WEIGHT

AIRPLANE TAKE-OFF GROSS WEIGHT

NO EXTERNAL LOAD	
F-100F-1 THRU F-100F-15	APPROX 30,750 LB
F-100F-20	APPROX 31,350 LB
WITH CENTERLINE STORE, TWO 275-GALLON, AND TWO 200-GALLON UNDERWING FUEL DROP TANKS	
F-100F-1 THRU F-100F-15	APPROX 40,100 LB
F-100F-20	APPROX 40,700 LB

WING

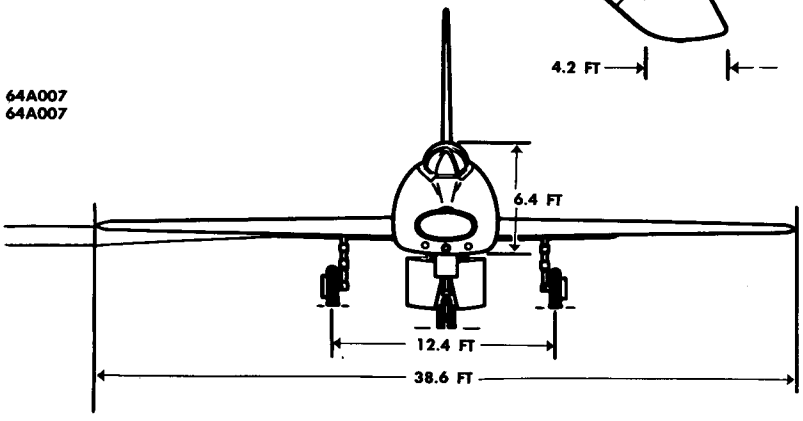
TYPE . . . . . LOW

AIRFOIL SECTION

ROOT	NACA 64A007
TIP	NACA 64A007
INCIDENCE AT ROOT	0 DEG
INCIDENCE AT TIP	0 DEG
ASPECT RATIO	3.72

DIHEDRAL

WING	0 DEG
STABILIZER	0 DEG



F-100F-3-00-1A

Figure 1-1A. Airplane Dimensions—F-100F Airplane

LEFT WING UPPER SURFACE

GENERAL

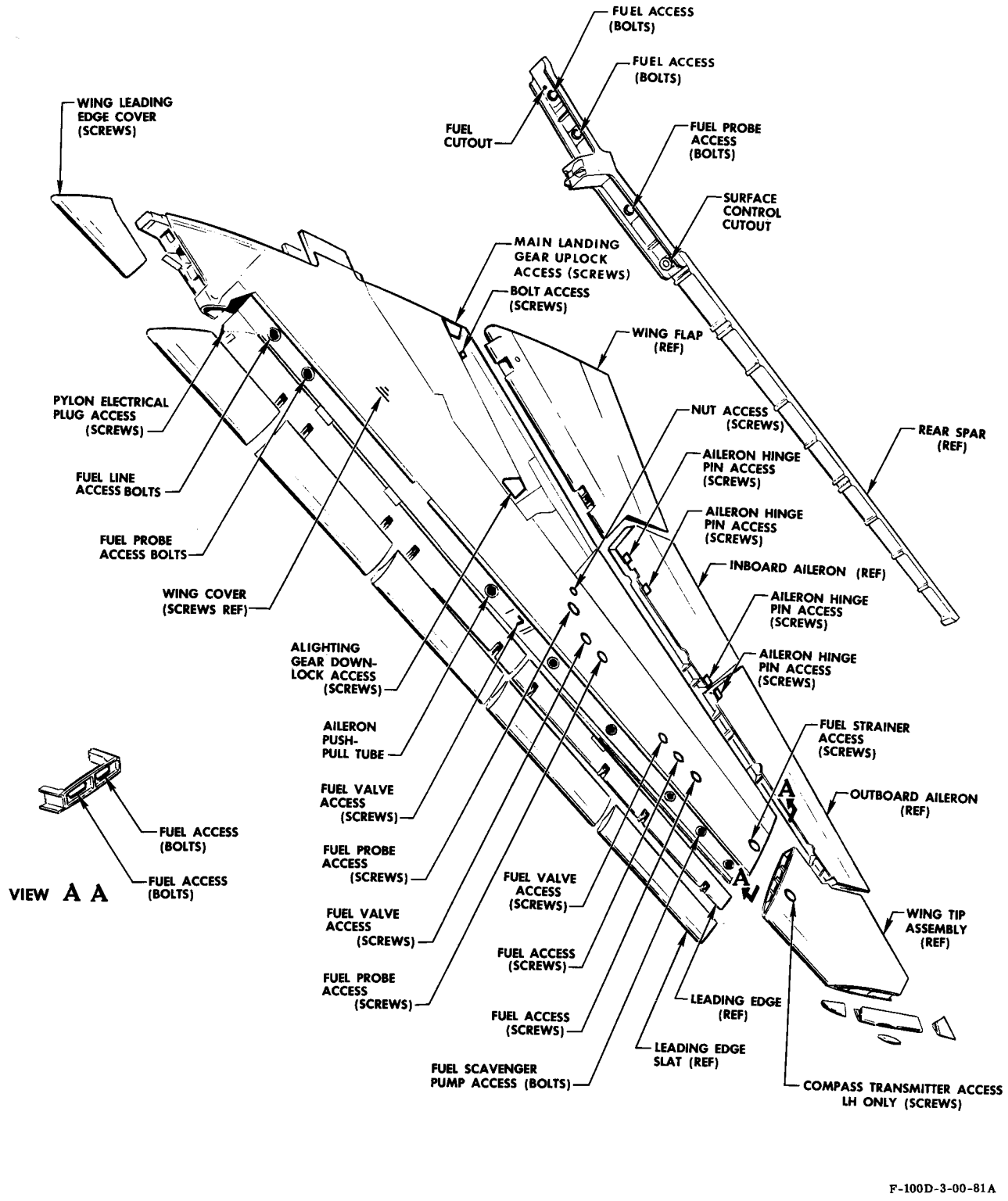


Figure 1-2. General Access Provisions—Wing (Sheet 1 of 2)

RIGHT WING LOWER SURFACE GENERAL

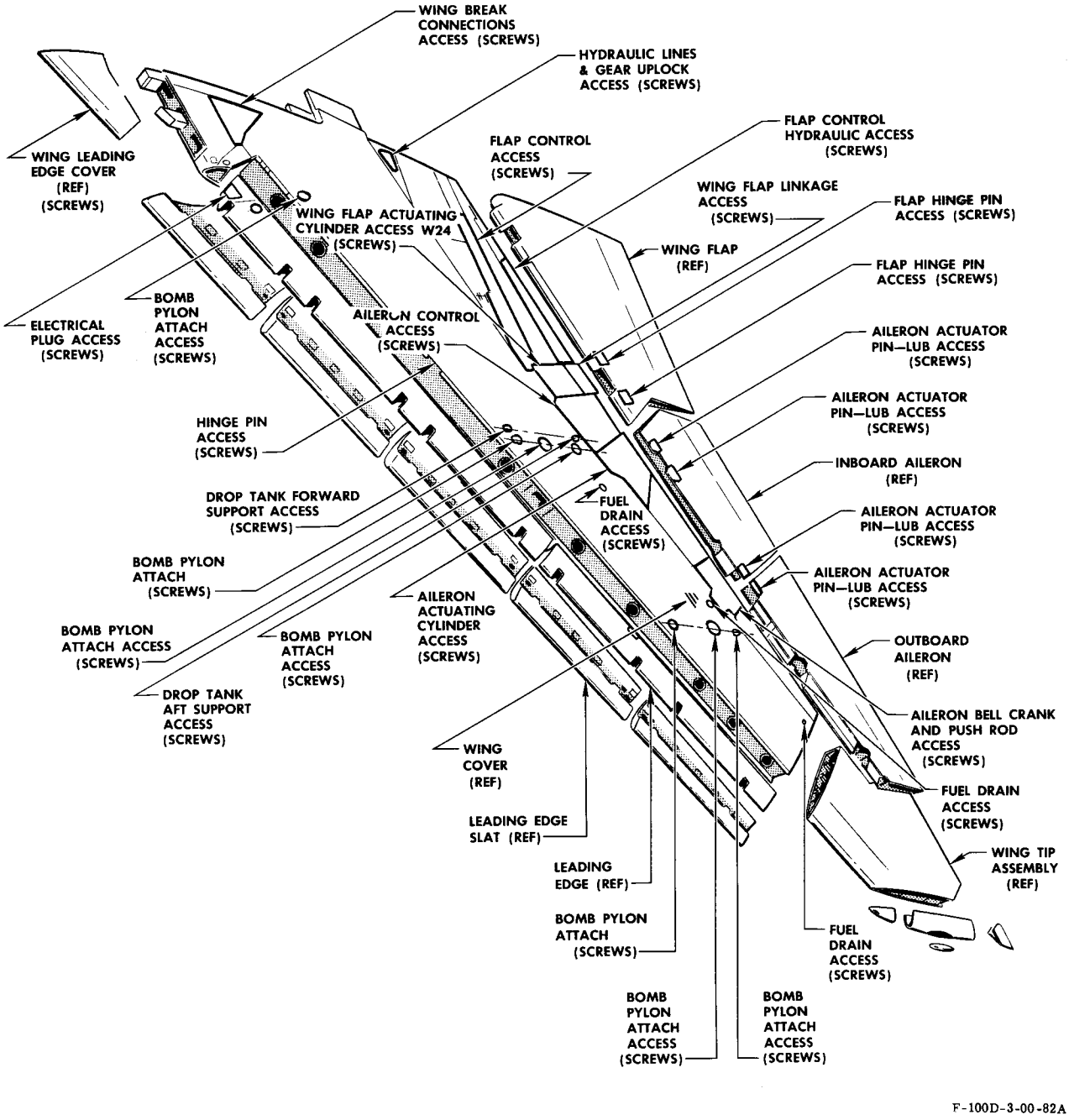
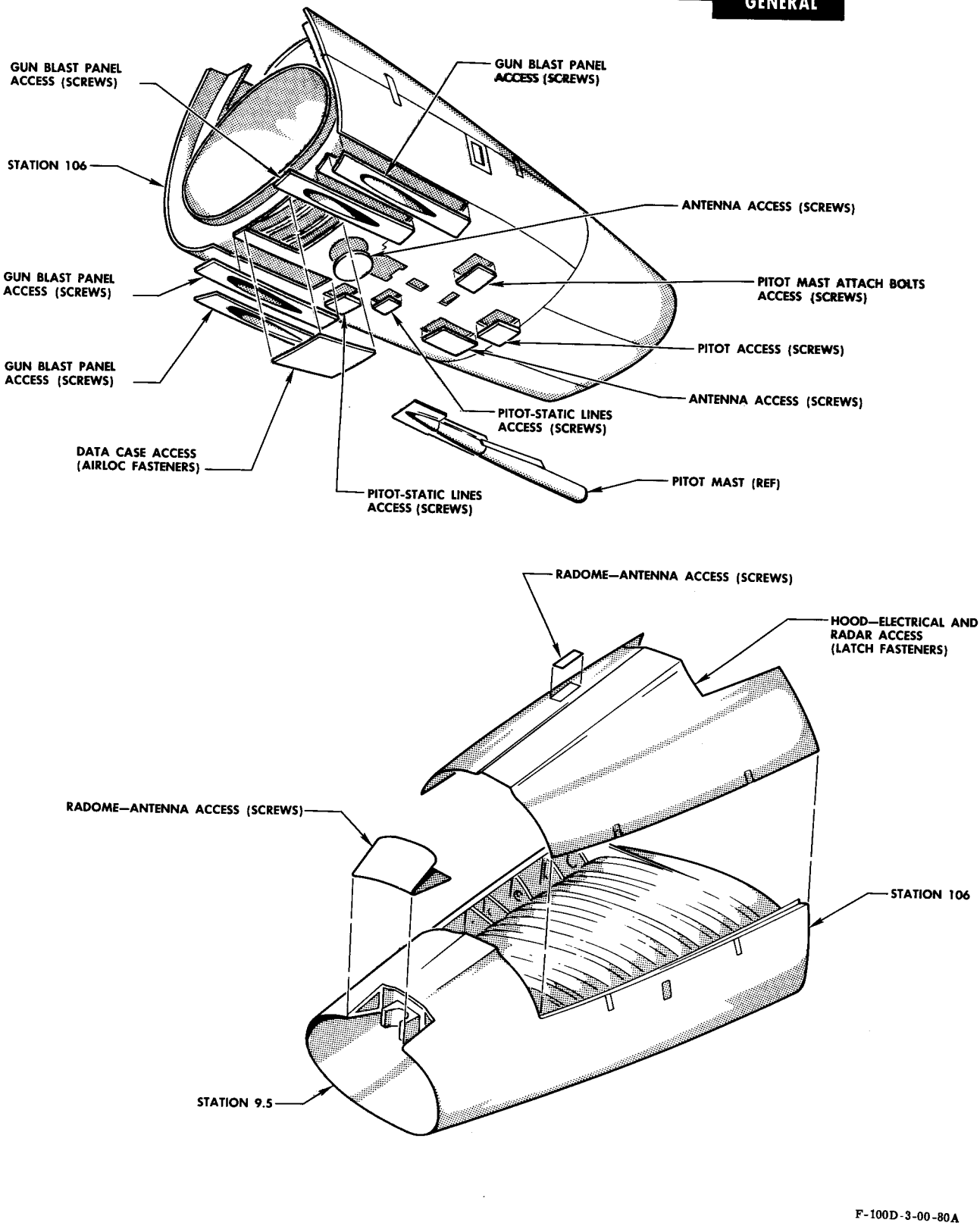


Figure 1-2. General Access Provisions—Wing (Sheet 2 of 2)

ACCESS PROVISIONS—STA 9.5 TO 106

GENERAL



F-100D-3-00-80A

Figure 1-2A. General Access Provisions—Forward Fuselage—F-100D Airplanes

ACCESS PROVISIONS—STA -25.5 TO 71

GENERAL

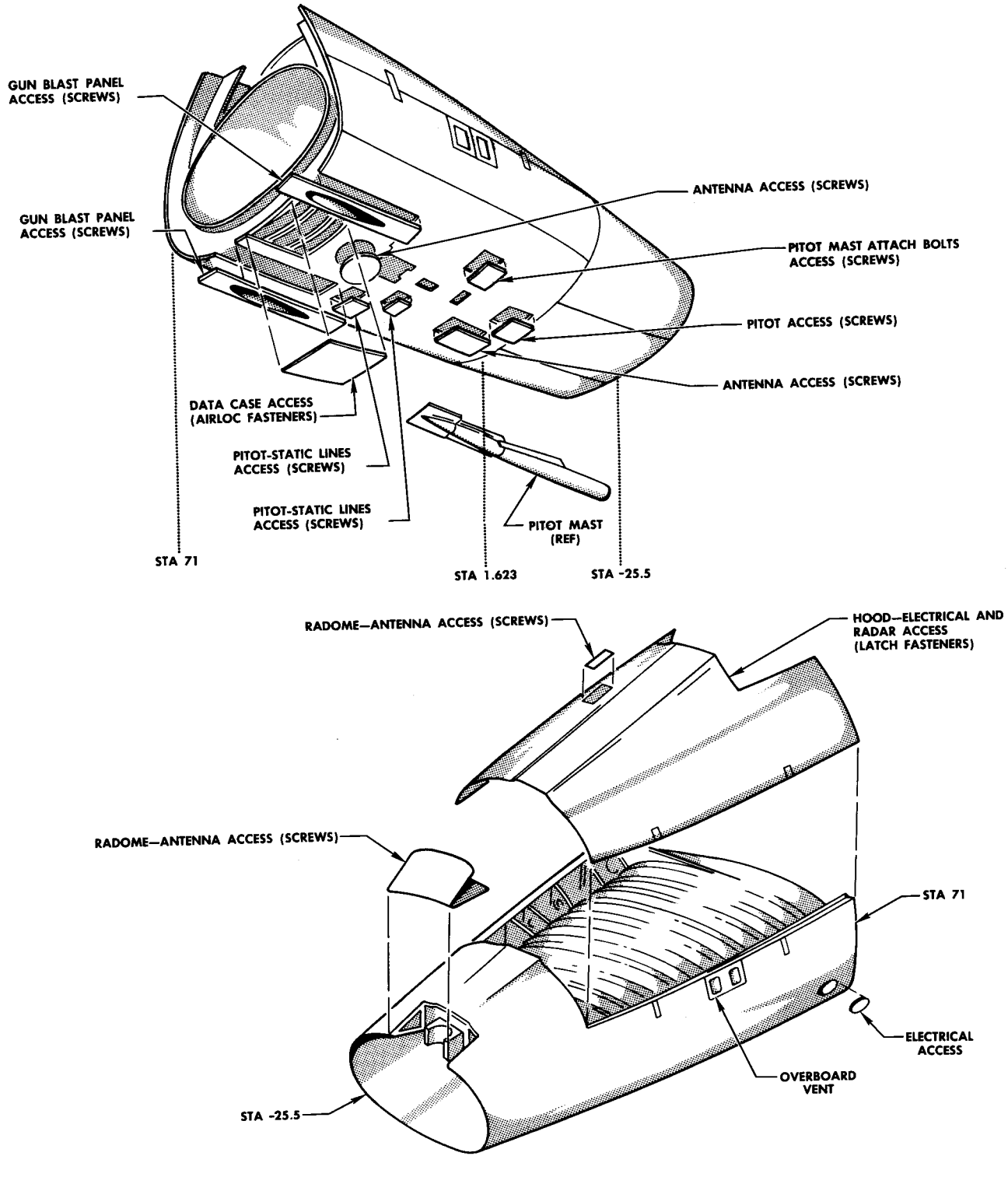
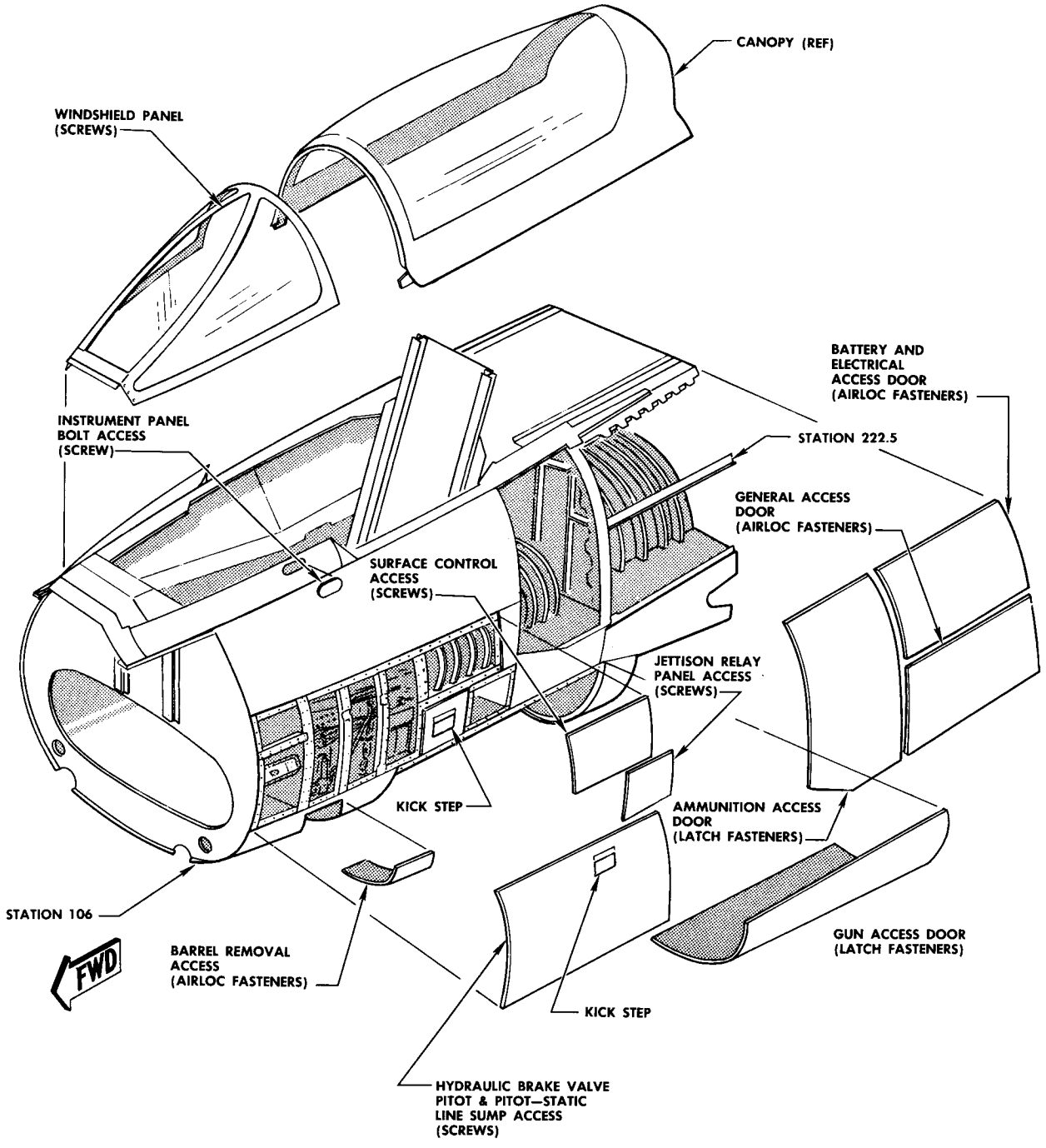


Figure 1-2B. General Access Provisions—Forward Fuselage—F-100F Airplanes

ACCESS PROVISIONS—STA 106 TO 222.5

GENERAL

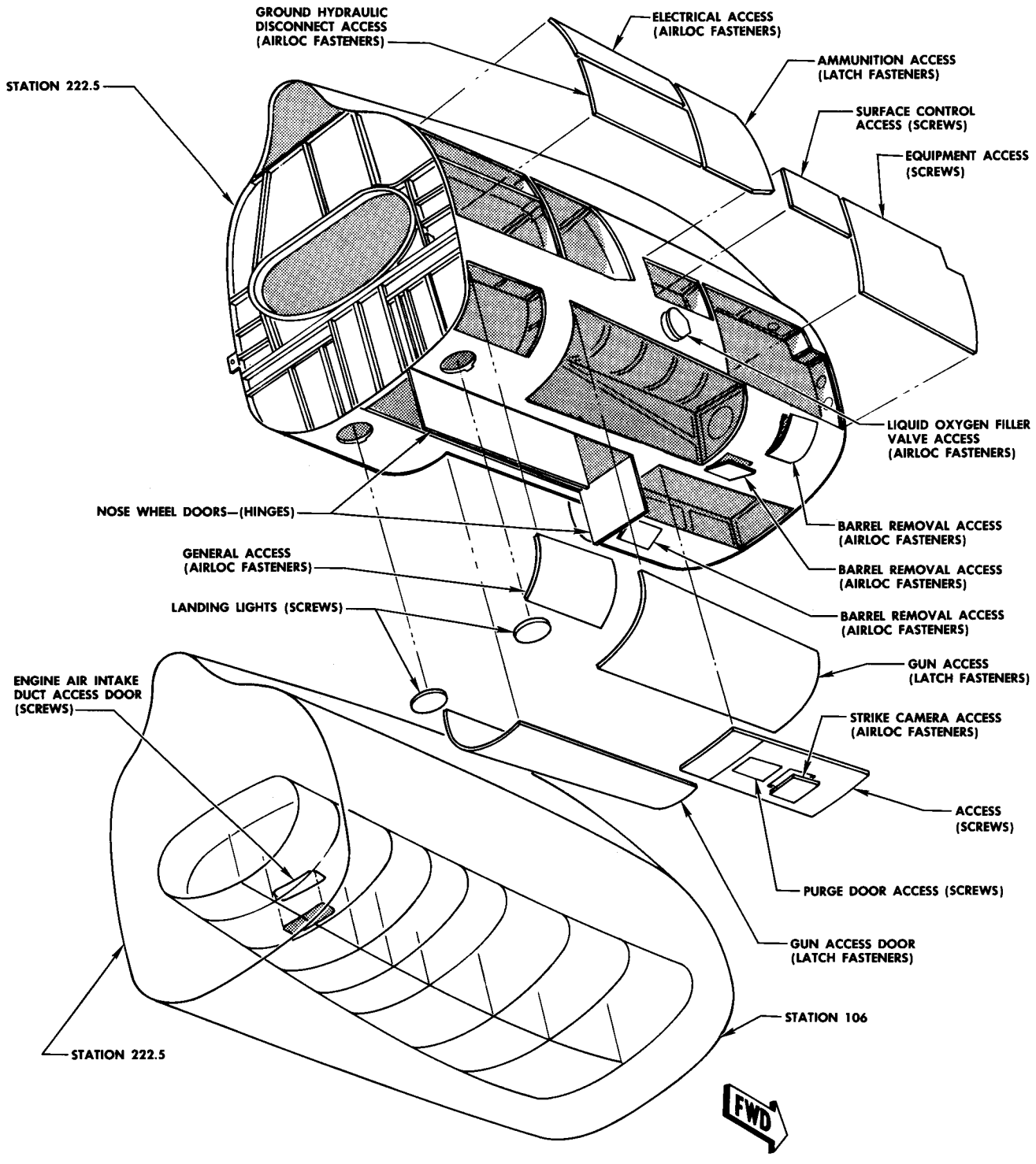


F-100D-3-00-114A

Figure 1-2C. General Access Provisions—Forward Intermediate Fuselage—F-100D Airplanes

ACCESS PROVISIONS—STA 106 TO 222.5

GENERAL

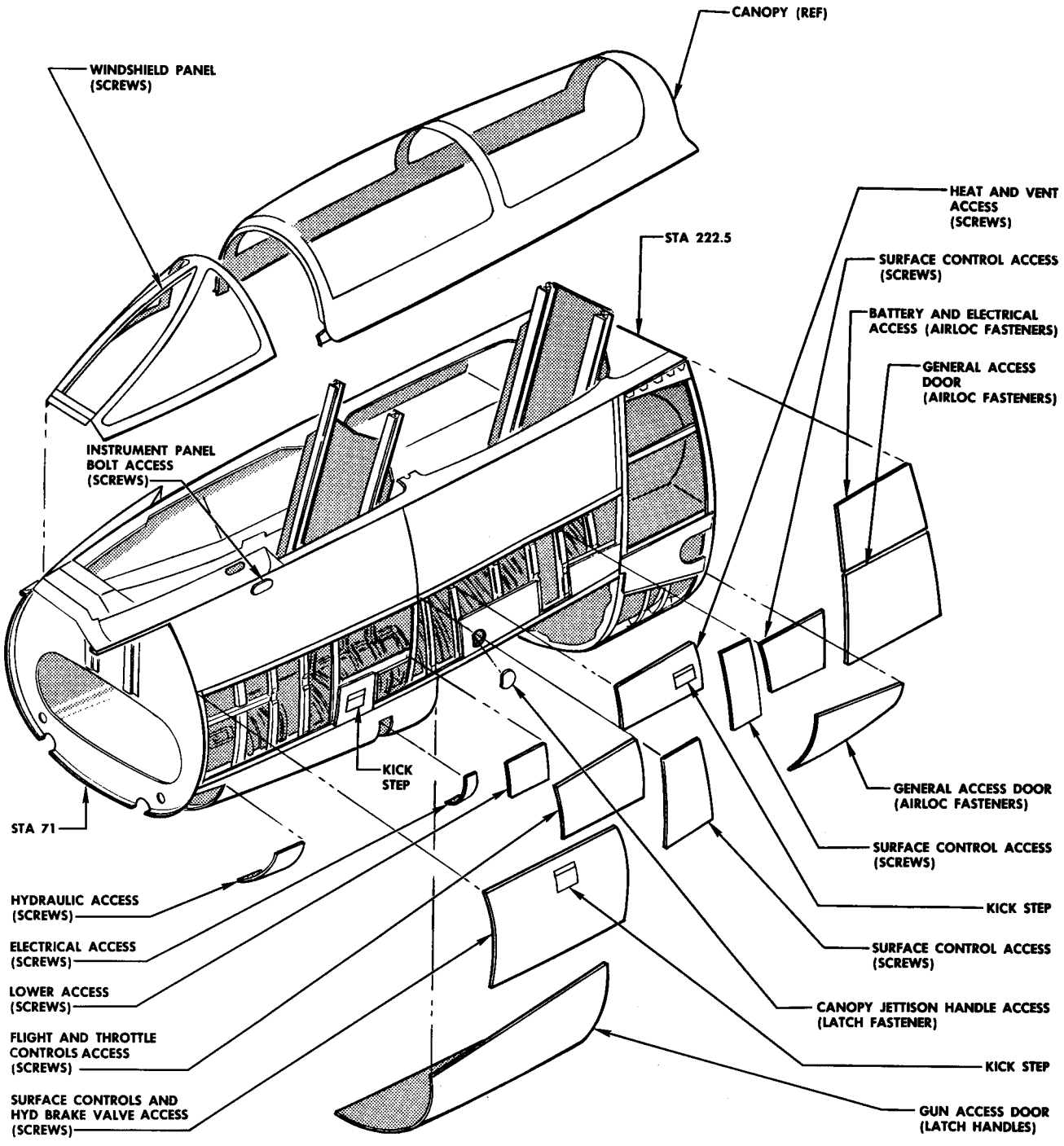


F-100D-3-00-77A

Figure 1-2C-1. General Access Provisions—Forward Intermediate Fuselage—F-100F Airplanes

ACCESS PROVISIONS—STA 71 TO 222.5

GENERAL

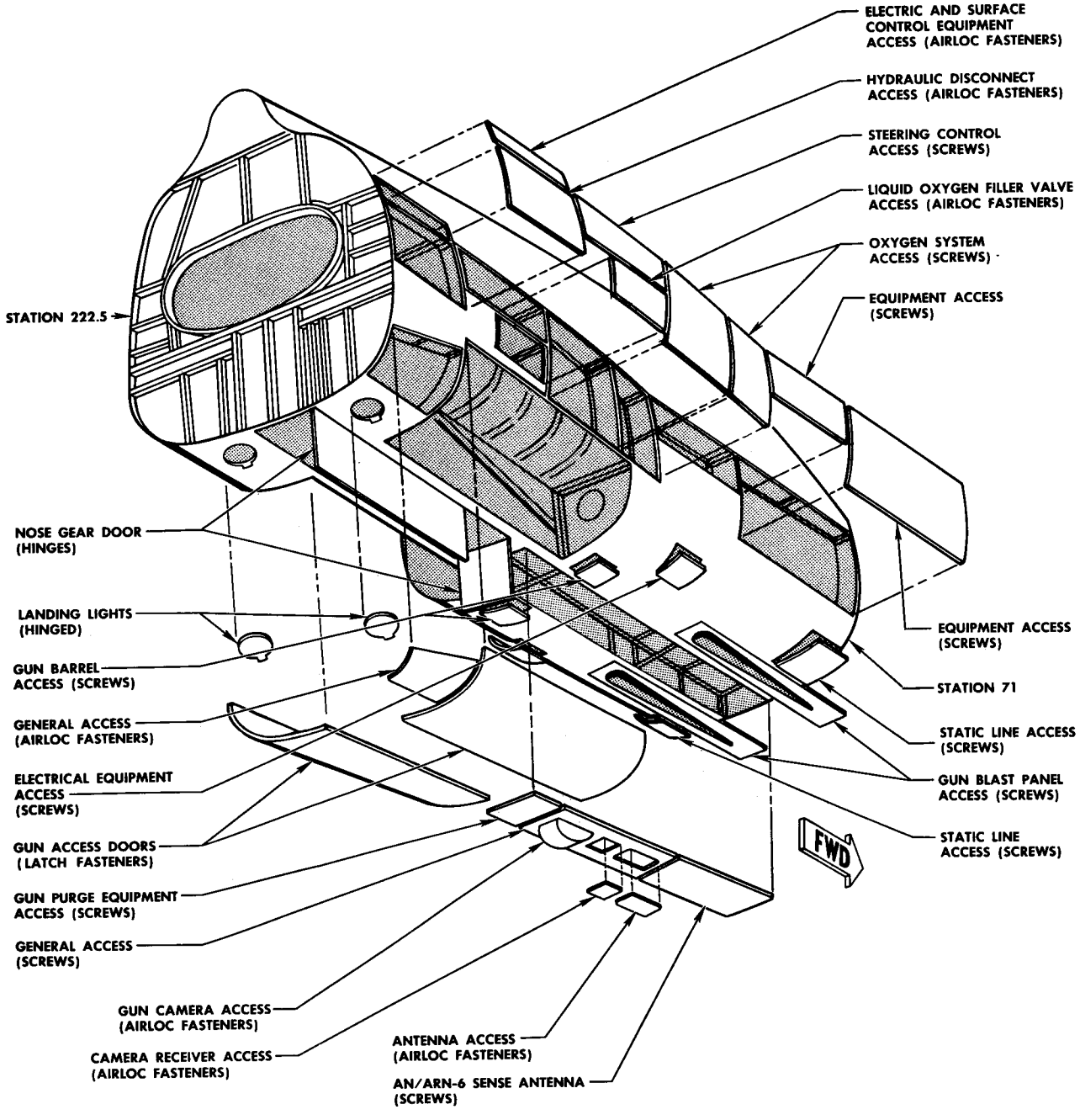


F-100D-3-00-126

Figure 1-2D. General Access Provisions—Forward Intermediate Fuselage—F-100F Airplanes (Sheet 1 of 2)

ACCESS PROVISIONS—STA 71 TO 222.5

GENERAL



F-100D-3-00-127

Figure 1-2D. General Access Provisions—Forward Intermediate Fuselage—F-100F Airplanes (Sheet 2 of 2)

ACCESS PROVISIONS-STA 222.5 TO 389

GENERAL

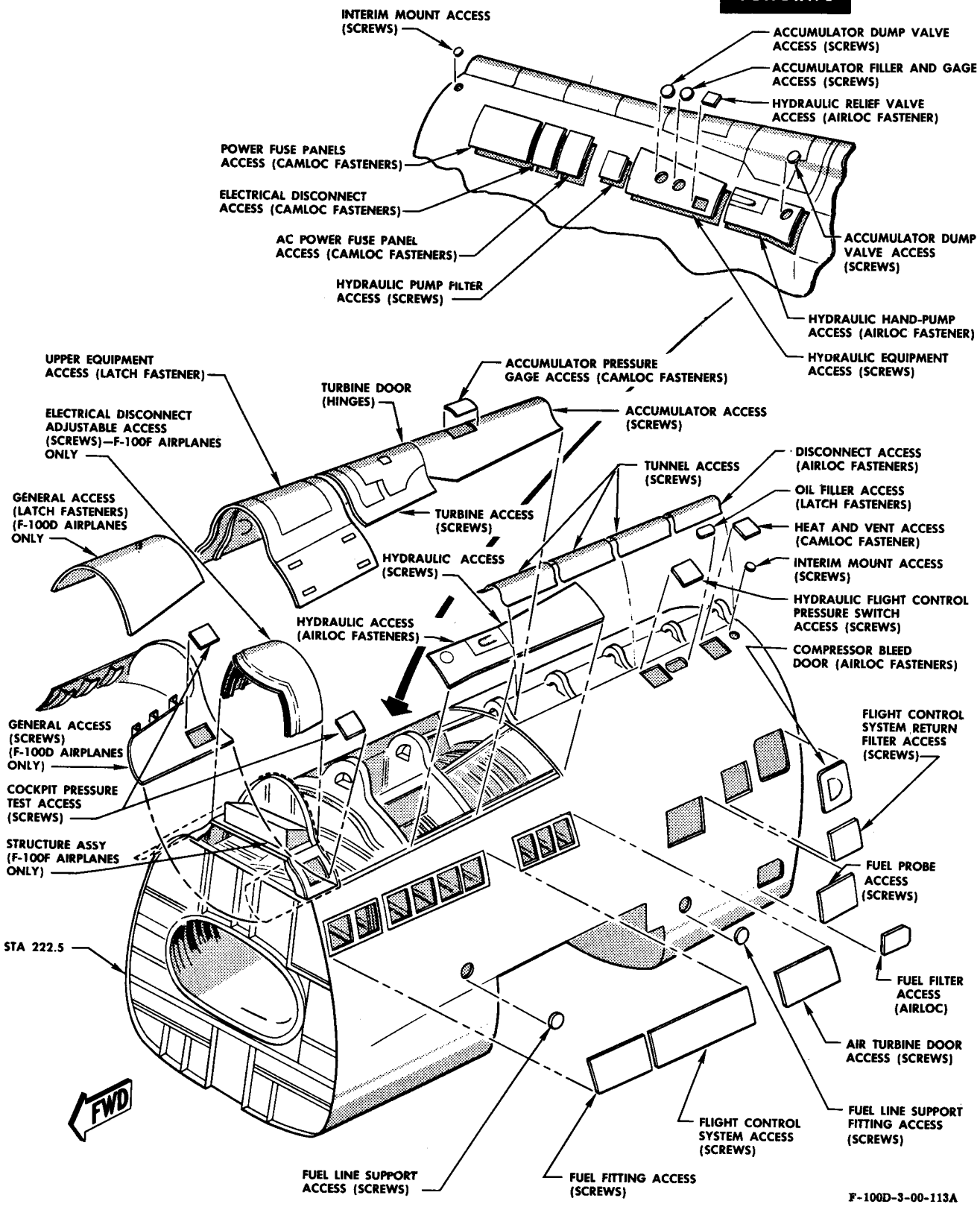


Figure 1-2E. General Access Provisions—Aft Intermediate Fuselage (Sheet 1 of 2)

ACCESS PROVISIONS—STA 222.5 TO STA 389

GENERAL

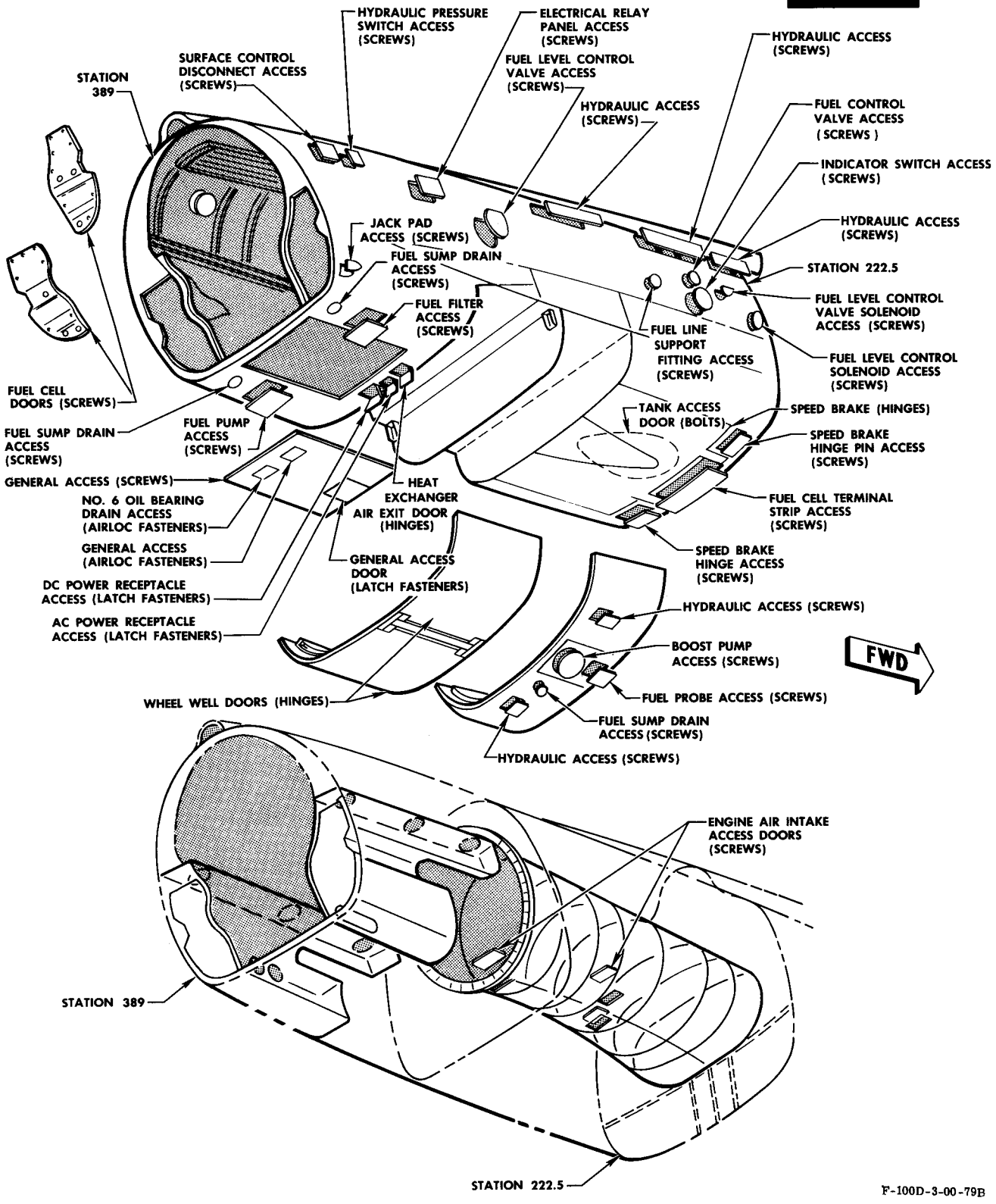


Figure 1-2E. General Access Provisions—Aft Intermediate Fuselage (Sheet 2 of 2)

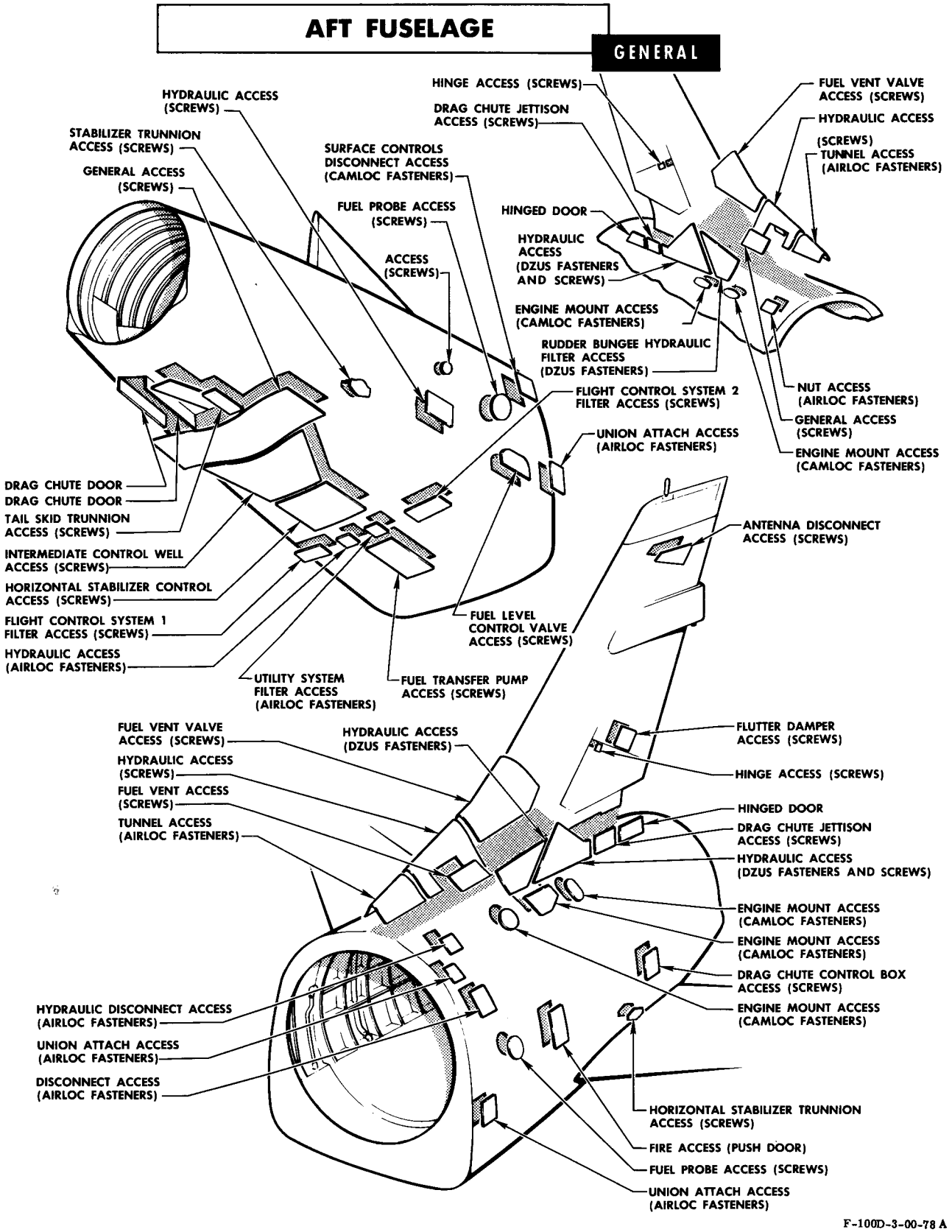
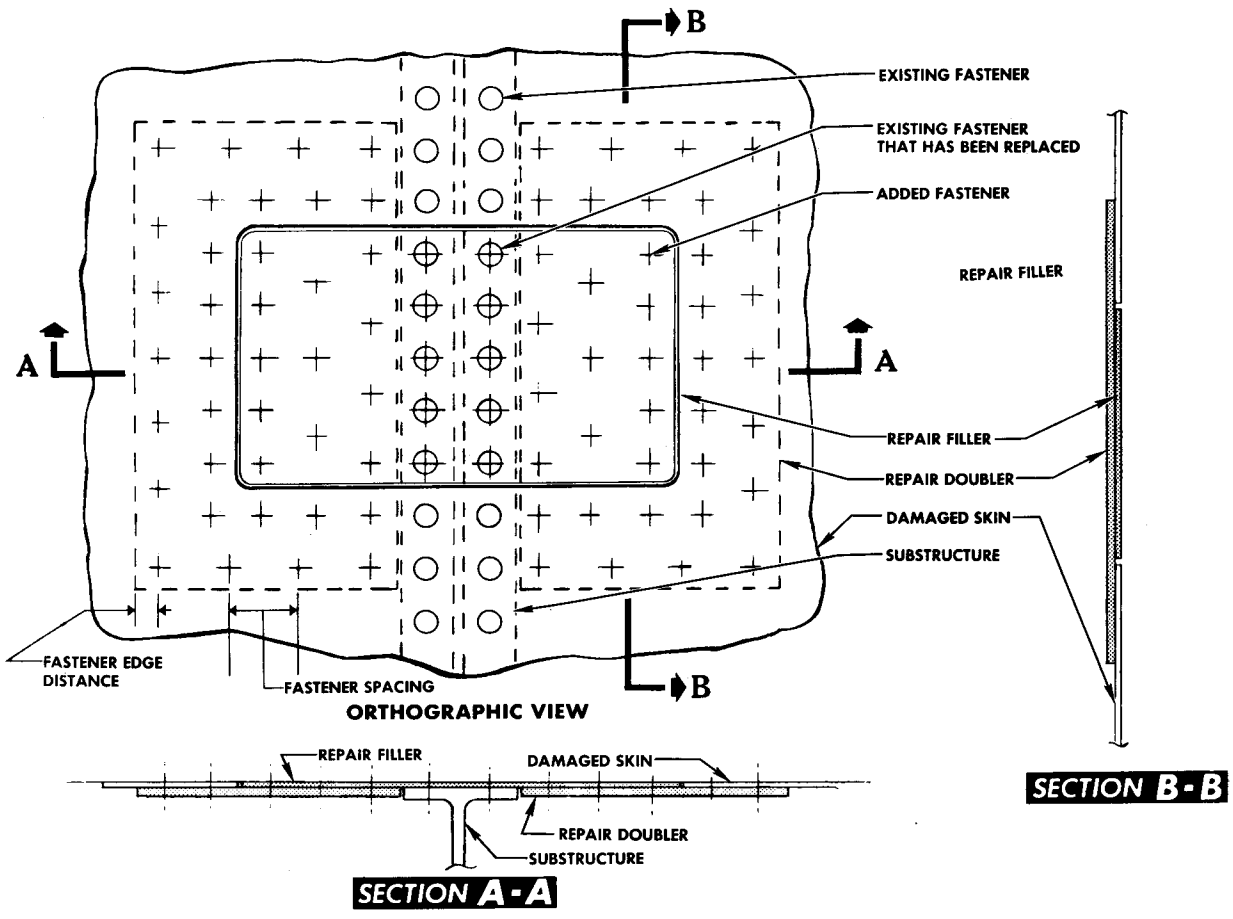
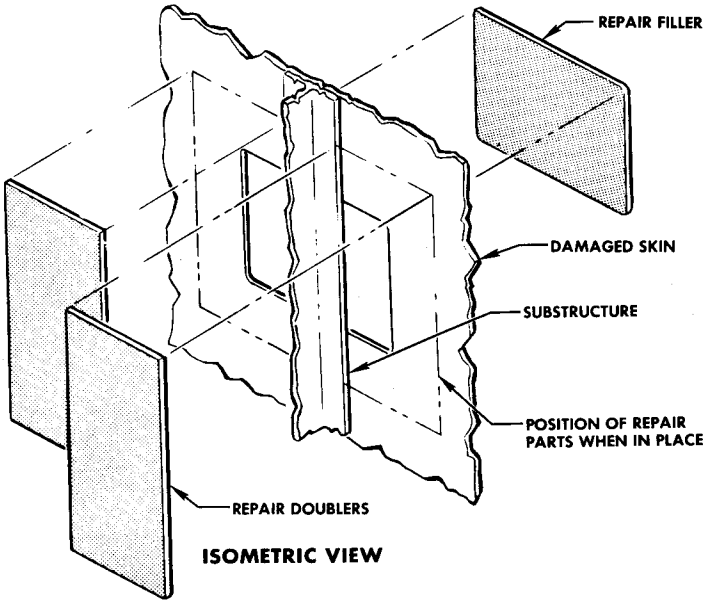


Figure 1-2F. General Access Provisions—Aft Section

REPAIR DRAWING EXPLANATION [redacted]





TYPICAL FASTENER SYMBOLS USED ON ALL REPAIR DRAWING

- ⊕ Indicates added fastener used to accomplish repair
- ⊕ Indicates an original fastener location. Fastener removed and replaced with an added fastener
- ○ Indicates an original fastener through surrounding skin and structure

Any additional types of fasteners required will be coded on the drawing.

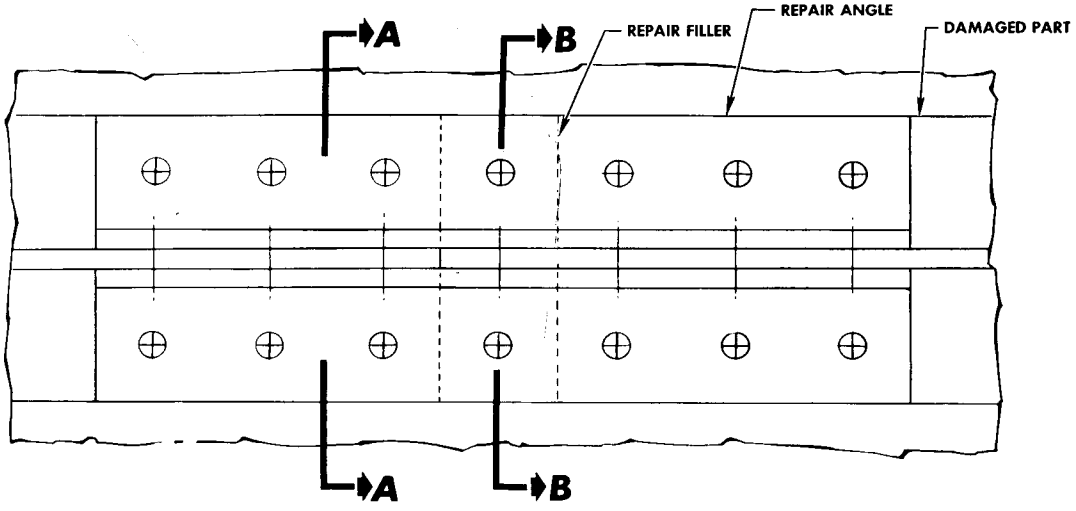
Fastener size, types, spacing, and edge distance are called out on the orthographic view only.

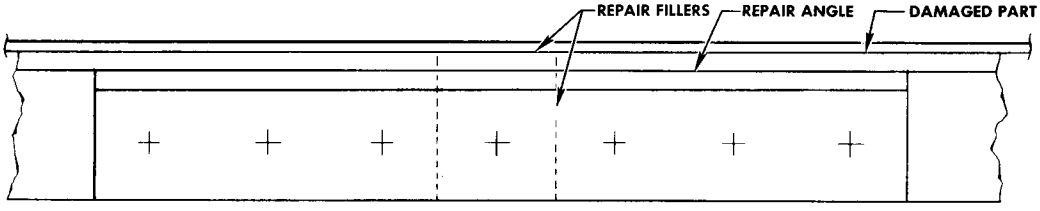
GEN-3-00-87

Figure 1-2G. Repair Drawing Explanation (Sheet 1 of 2)

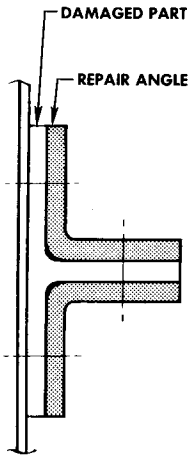
REPAIR DRAWING EXPLANATION

GENERAL

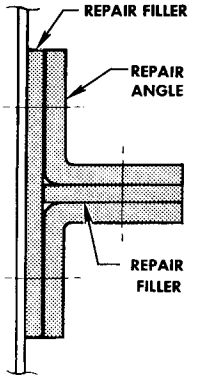




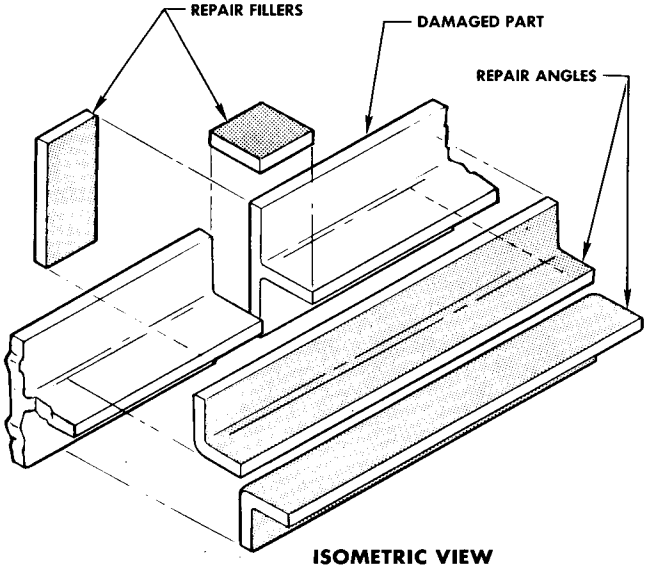
ORTHOGRAPHIC VIEW



SECTION A-A



SECTION B-B



ISOMETRIC VIEW

Section views show *only* that which can be seen from the indicated section cut—nothing beyond. Section views would appear as a thin section of the part removed for viewing at the place indicated by the section lines.

Section arrows indicate the direction in which a person would appear to be looking when viewing the section cut.

The letters next to the section arrows indicate or label section view details.

Dotted lines indicate a hidden part or structure that cannot be seen from the side from which the part is being viewed.

Phantom lines are used either to indicate the position of the repair part when in place or as guide lines to show how repair parts are exploded from isometric view.

Shading is used for repair parts on the isometric and section views only.

Repair part material and gage are called out on the isometric view.

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Figure 1-2G. Repair Drawing Explanation (Sheet 2 of 2)

and a correct plate fuselage which is designed aerodynamically to assist in exceeding the speed of sound. It is powered by a single Pratt & Whitney J57 series turbojet engine with an afterburner. The wing consists of a center section panel and two outer panels. The center section contains fuel cells. The outboard panels have integral-type fuel tanks. Provisions have been made for carrying various combinations of droppable fuel tanks, bombs, or special stores under the wings. The airplane may be refueled in flight by the probe-and-drogue method. The wings have automatically operated leading edge slats and are fitted with flaps which extend from aileron to fuselage. The empennage consists of a low-positioned, all-movable horizontal stabilizer, a fixed vertical stabilizer, and a rudder. Longitudinal control of the airplane is provided by movement of the entire stabilizer. The fuselage is a semimonocoque structure with a fuselage rear section attachment point at fuselage station 389. Access doors in the fuselage make most areas accessible for repair and maintenance. A fully retractable, tricycle-type landing gear is used. In the retracted position, the gear is completely covered and enclosed by fairing doors. Details of the structure for each part of the airplane are covered in the applicable section of the handbook.

The F-100F Airplane has the same features as the F-100D Airplane with the exception of its longer fuselage. The forward intermediate section was lengthened by 35 inches and the cockpit was lengthened to facilitate the installation of another seat and controls for use as a pilot trainer. The basic armament installation on the F-100F Airplane that is different from the F-100D Airplane is the two 20 mm automatic guns mounted in compartments in the lower surface of the forward section. One gun is mounted on each side and outboard of the nose wheel well, while the F-100D has four guns, two mounted on each side of the nose wheel well.

AIRPLANE DIMENSIONS.

For F-100D Airplanes, see figure 1-1. For F-100F Airplanes, see figure 1-1A.

AIRPLANE BLOCK AND SERIAL NUMBERS.

The airplane block and serial numbers are stenciled on the left side of the fuselage, below the windshield side panel, at station 116 on F-100D Airplanes, and at station 81 on F-100F Airplanes.

BLOCK NUMBER	AF SERIAL NUMBER
F-100D-1	AF54-2121 through -2132
F-100D-5	AF54-2133 through -2151
F-100D-10	AF54-2152 through -2221
F-100D-15	AF54-2222 through -2303
F-100D-20	AF55-3502 through -3601
F-100D-25	AF55-3602 through -3701
F-100D-30	AF55-3702 through -3814
F-100D-35	AF55-2734 through -2743
F-100D-40	AF55-2744 through -2783
F-100D-45	AF55-2784 through -2863
F-100D-50	AF55-2864 through -2908
F-100D-55	AF55-2909 through -2954
F-100D-60	AF56-2903
	AF56-2905 through -2946
	AF56-2948 through -2962
F-100D-65	AF56-2963 through -3022
F-100D-70	AF56-3023 through -3142
F-100D-75	AF56-3143 through -3198
F-100D-80	AF56-3351 through -3378
F-100D-85	AF56-3379 through -3463
F-100D-90	AF56-3199 through -3346
F-100F-1	AF56-3725 through -3739
F-100F-5	AF56-3740 through -3769
F-100F-10	AF56-3770 through -3919
F-100F-15	AF56-3920 through -4019
	AF58-6975 through -6983
	AF59-2558 through -2563
F-100F-20	AF58-1205 through -1233

GENERAL ACCESS PROVISIONS.

See figures 1-2 through 1-2F.

MAJOR ASSEMBLIES—EXPLODED VIEW.

For F-100D Airplanes, see figure 1-3.
For F-100F Airplanes, see figure 1-3A.

LOCATING AIRPLANE STATIONS.

The extensive use of milled skins in the F-100D and F-100F Airplanes has reduced the number of fastener rows normally relied on for locating airplane stations. Figures 1-4 through 1-4C show the stations represented by ribs or frames. The following text aided by the view shown in figure 1-5 explains the measuring systems used for each airframe component. The recognizable

points shown in the airplane station drawings, plus a knowledge of the measuring systems used, provide the information required for an accurate location of any point on the airplane.

FUSELAGE STATIONS.

The fuselage is divided horizontally along the Zf plane by units referred to as water planes, vertically along the Xf plane by buttock planes, and longitudinally along the Yf plane by fuselage station planes. The Yf plane is used as a fuselage reference plane. Water plane zero, which falls on the fuselage reference plane, and station zero are arbitrarily defined by design. Buttock plane zero is defined as the vertical plane of symmetry. Water planes below the fuselage reference plane are negative (−); those above it are positive (+). Stations forward of station zero are negative (−); those aft of it are positive (+). Buttock planes to the left of buttock plane zero are positive (+); those to the right of it are negative (−). Fuselage stations are normal (at right angles) to the fuselage reference plane and to the plane of symmetry. They are measured in increments of inches forward and aft of station zero. Fuselage canted stations are diagonal to the fuselage reference plane. They are designated by their point of intersection with the fuselage reference plane.

WING STATIONS.

The wing is broken up into subsections for basic reference data. These are wing stations, canted wing stations, aileron stations, leading edge stations, and slat stations. The wing stations are a continuation of the fuselage buttock planes. The wing vertical plane, Xw, is normal (at right angle) to the fuselage reference plane. Wing station zero is where the 25 per cent plane, projected, intersects the fuselage buttock plane zero. The canted wing stations are diagonal to the vertical wing reference plane. The design of the structure adjacent to the canted ribs determine their positions in relation to the vertical wing reference plane. The aileron stations are normal (at right angles) to the aileron hinge centerline. The aileron hinge centerline is the 75 percent plane of the wing. Leading edge stations are normal (at right angles) to the leading edge mold line, which is considered the zero percent plane. Slat stations are normal (at right angles) to the slat hinge centerline.

AILERON AND FLAP STATIONS.

Because the aileron stations are normal (at right angles) to their hinge centerline, they may be measured along their hinge centerline from the inboard skin trim lines.

Flap stations are normal (at right angles) to their hinge centerline. Station zero is where the flap hinge centerline, projected, would intersect fuselage buttock plane zero.

VERTICAL STABILIZER STATIONS.

Vertical stabilizer stations are generally normal (at right angles) to the rudder hinge centerline. The rudder hinge centerline is the 74.85 percent (Zh) plane. The root rib and the closing rib at the point of tip attachment are canted.

HORIZONTAL STABILIZER STATIONS.

The horizontal stabilizer stations in general are normal (at right angles) to the 46.43 percent plane. Horizontal stabilizer station zero is where the 46.43 percent plane, projected, intersects the fuselage buttock plane zero. The root rib and the tip attachment rib are canted.

REPAIR CAPABILITY CATEGORIES.

Repairs may be divided into two categories: those which would require extensive disassembly of the airplane to gain access to the damaged area, and those for which the damage is readily accessible. Also, repairs may be categorized by the extent of damage, and by the materials available for the repair. The ability of an organization to repair an airplane will depend on the manpower, tools, and materials available. The repairs in this handbook are divided into the two categories. Category "A" repairs are minor (squadron or base shop) repairs; category "B" repairs are major (depot) repairs. It is assumed that at depot level both category "A" and "B" repairs would be performed. It is also assumed that a squadron or base shop, with adequate facilities and manpower would perform certain category "B" repairs.

TYPES OF REPAIR AND CLASSIFICATION OF DAMAGE.

The repairs in this handbook are broken down into four basic types: negligible damage repairs, permanent repairs, temporary repairs, and one-time flight repairs.

ONE-TIME FLIGHT DAMAGE.

One-time flight damage is damage that can be left as is or which requires only minor repairs to allow the

EXPLODED VIEW OF AIRPLANE

GENERAL

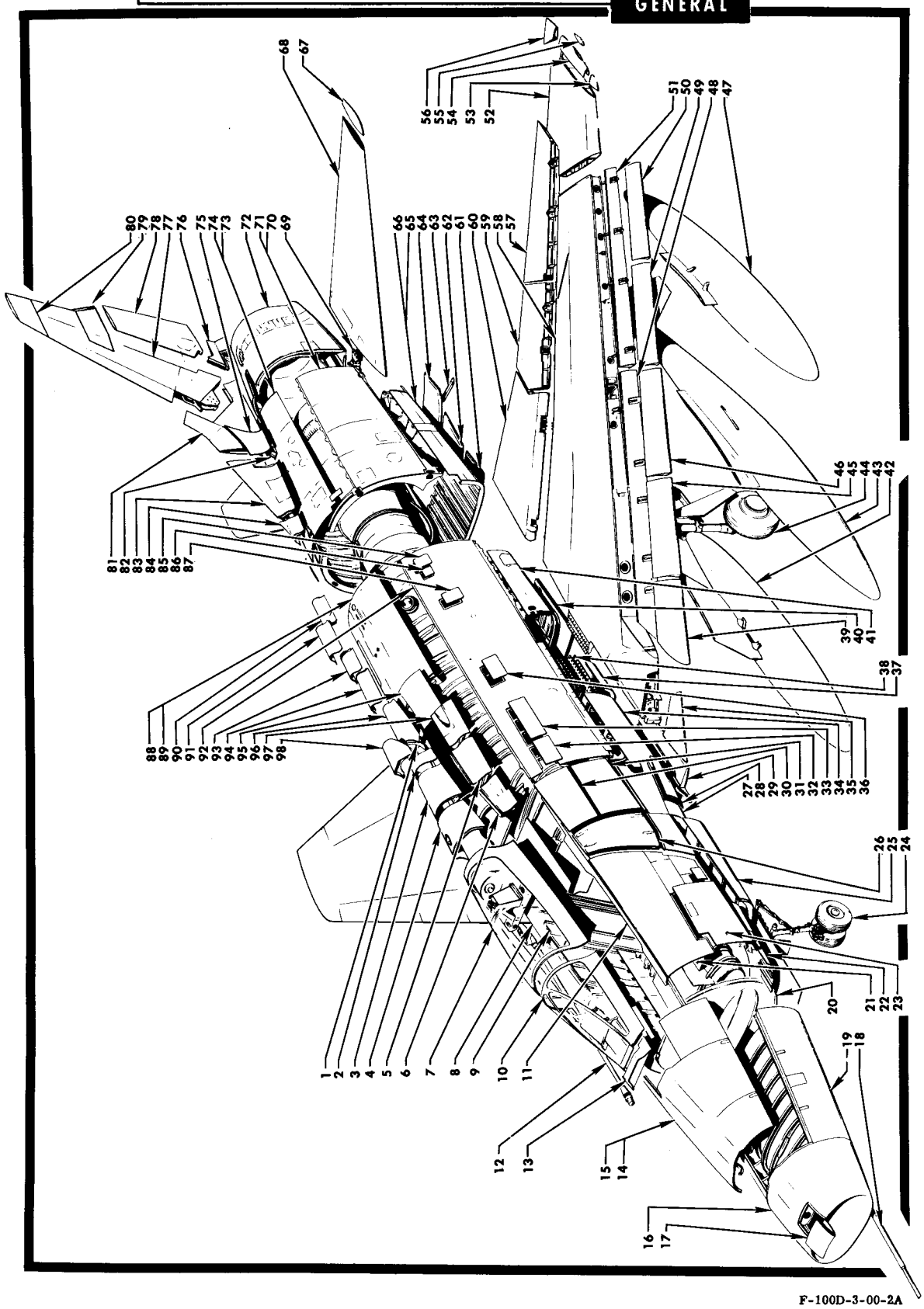


Figure 1-3. Major Assemblies—Exploded View

ITEM NO.	DRAWING NO.	DESCRIPTION	ITEM NO.	DRAWING NO.	DESCRIPTION
1	192-310143	Fairing Assy	51	223-14006	Leading Edge Assy
2	192-310120	Fairing Assy	52	223-14017	Structure Assy
3	192-310101	Door Assy	53	192-12016	Panel Assy
4	223-31061	Door Assy	54	192-12012	Panel Assy
5	192-310225	Door Assy	55	192-12879	Fairing
6	223-31214	Frame Assy	56	192-12023	Fairing
7	223-31802	Canopy Assy	57	223-16003	Aileron Assy
8	223-31158	Deck	58	223-14002	Structure Assy
9	223-53013	Seat Rails	59	223-16002	Aileron Assy
10	223-31805	Windshield Frame Assy	60	223-18002	Flap Assy
11	223-31112	Longeron Assy	61	223-31036	Door Assy
12	192-48775	Probe Instl	62	223-31037	Door Assy
13	180-53341	Fairing Assy	63	223-34081	Skid Instl
14	192-31002	Hood Instl	64	192-310307	Door Assy
15	223-31003	Hood Assy	65	223-31391	Frame Assy
16	192-31004	Nose Assy	66	223-31506	Frame Assy Bottom
17	192-31037	Radome	67	192-21311	Stab Tip
18	192-51275	Boom Assy	68	192-21001	Stabilizer Assy
19	223-31103	Frame Assy		192-21002	Hor Stab Frame Assy
20	223-31106	Bhd Assy	69	180-21403	Beam Assy
21	223-31153	Web Assy	70	192-31509	Side Frame Assy
22	223-31113	Frame Assy	71	192-314175	Side Frame Assy
23	223-31034	Door Assy	72	192-315175	Frame Assy
24	223-34011	Nose Gear Instl	73	192-31508	Top Frame Assy
25	223-31013	Door Assy	74	192-31504	Top Frame Assy
26	223-31014	Door Assy	75	223-25003	Door
27	223-31107	Frame Assy	76	192-25025	Fillet Assy
28	223-31019	Door Assy	77	217-23002	Frame Assy
29	192-39002	Speed Brake	78	192-24001	Rudder Assy
30	223-31015	Door Assy	79	192-54608	Elect Instl
31	223-31016	Door Assy	80	192-23805	Door
32	192-31073	Door	81	217-25002	Fillet Instl
33	192-310191	Door	82	192-31459	Fitting Assy
34	192-31404	Frame Assy	83	223-31407	Fin Assy
35	223-14004	Cover Assy	84	192-31025	Dorsal Fin Access
36	192-31074	Door	85	180-42031	Door
37	192-31076	Door Assy	86	192-310282	Door
38	192-13002	Str Assy	87	192-31081	Door
39	192-17101	Slat Assy	88	192-31403	Frame Assy
40	217-31036	Door Assy	89	192-31320	Frame Assy
41	192-31405	Frame Assy	90	192-31010	Tunnel Assy
42		200 Gal AF Tank	91	192-31009	Tunnel Assy
43		275 Gal NAA Tank	92	192-31318	Side Panel Assy
44	223-33011	Main Landing Gear Instl		192-31402	Side Panel Assy
45	192-17202	Slat Assy		192-31319	Side Panel Assy
46	192-17203	Slat Assy	93	192-31008	Tunnel Assy
47		200 Gal AF Tank	94	192-31007	Tunnel Assy
48	223-14005	Leading Edge Assy	95	192-31067	Door Assy
49	192-17204	Slat Assy	96	192-31068	Door Assy
50	192-17205	Slat Assy	97	192-31066	Door Assy
			98	192-31064	Door Assy

Key to Figure 1-3

EXPLODED VIEW OF AIRPLANE

GENERAL

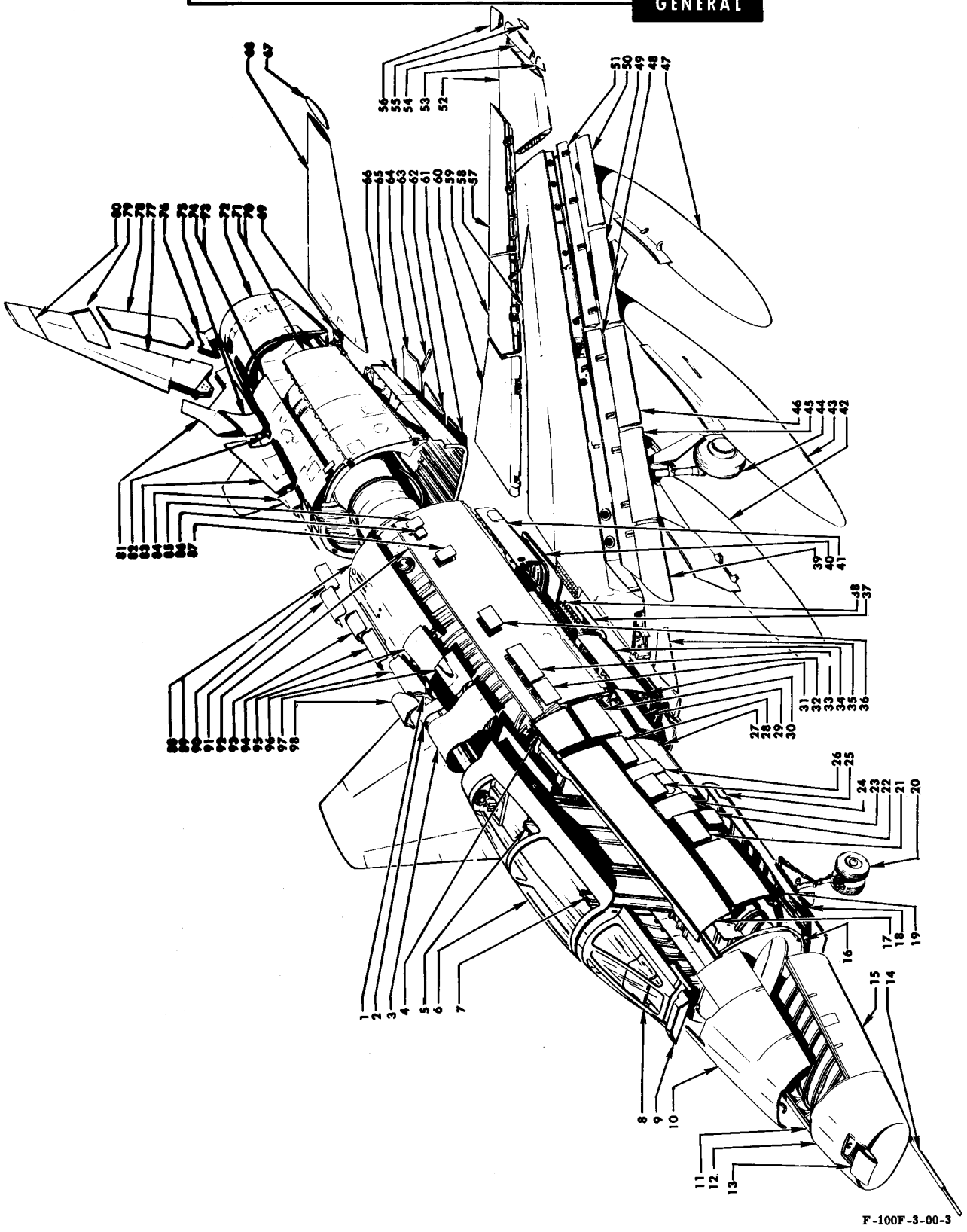


Figure 1-3A. Major Assemblies—Exploded View—F-100F Airplanes

ITEM NO.	DRAWING NO.	DESCRIPTION
1	192-310143	Fairing Assy
2	192-310120	Fairing Assy
3	192-310101	Door Assy
4	243-31015	Door Assy
5	243-53014	Aft Rails
6	243-31802	Canopy Assy
7	243-53013	Fwd Rails
8	223-31805	Windshield Frame
9	180-53341	Fairing Assy
10	192-31002	Hood Instl
11	223-31103	Frame Assy
12	192-31004	Nose Assy
13	217-31042	Radome Assy
14	192-51276	Boom Assy
15	223-31103	Frame Assy
16	223-31106	Bhd Assy
17	243-31105	Frame Assy
18	223-31113	Frame Assy
19	243-31034	Door Assy
20	223-34011	Aux Flight Gear
21	243-31017	Door Assy
22	243-31095	Door Assy
23	243-31014	Door Assy
24	243-31010	Door Assy
25	243-31030	Door Assy
26	243-31047	Door Assy
27	243-31013	Door Assy
28	243-31023	Door Assy
29	192-39002	Speed Brake
30	243-31014	Door Assy
31	243-31016	Door Assy
32	192-31073	Door Assy
33	192-310191	Door Assy
34	192-31404	Frame Assy
35	223-14004	Cover Assy
36	192-31074	Door Assy
37	192-31076	Door Assy
38	192-13002	Str Assy
39	192-17101	Slat Assy
40	217-31036	Door Assy
41	192-31405	Frame Assy
42		200 Gal AF Tank
43		275 Gal NAA Tank
44	223-33011	Main Flight Gear
45	192-17202	Slat Assy
46	192-17203	Slat Assy
47		200 Gal AF Tank
48	223-14005	Leading Edge Assy
49	192-17204	Slat Assy
50	192-17205	Slat Assy
51	223-14006	Leading Edge Assy
52	223-14011	Str Assy
53	192-12016	Panel Assy
54	192-12012	Panel Assy
55	192-12879	Fairing
56	192-12023	Fairing
57	223-16003	Ail Assy
58	223-14002	Struct Assy
59	223-16002	Ail Assy
60	223-18002	Flap Assy
61	223-31036	Door Assy
62	223-31037	Door Assy
63	223-34081	Skid Instl
64	192-310307	Door Assy
65	223-31391	Frame Assy
66	223-31506	Frame Assy Bottom
67	192-21311	Hor Stab Tip
68	192-21001	Hor Stab Assy Complete
	192-21002	Hor Stab Assy Frame
69	180-21403	Beam Assy
70	192-31509	Side Frame Assy
71	192-314175	Side Frame Assy
72	192-315175	Frame Assy
73	192-31508	Top Frame Assy
74	192-31504	Top Frame Assy
75	223-25003	Door Assy
76	192-25025	Fillet Assy
77	217-23002	Frame Assy
78	192-24001	Rudder Assy
79	192-54608	Elec Instl
80	192-23805	Door Assy
81	217-25002	Fillet Instl
82	192-31459	Fitting Assy
83	223-31407	Fin Assy
84	192-31025	Dorsal Fin Access
85	180-42031	Door Assy
86	192-310282	Door Assy
87	192-31081	Door Assy
88	192-31403	Frame Assy
89	192-31320	Frame Assy
90	192-31010	Tunnel Assy
91	192-31009	Tunnel Assy
92	192-31318	Side Panel Assy
	192-31402	Side Panel Assy
	192-31319	Side Panel Assy
93	192-31008	Tunnel Assy
94	192-31007	Tunnel Assy
95	192-31067	Door Assy
96	192-31068	Door Assy
97	192-31066	Door Assy
98	192-31064	Door Assy

Key to Figure 1-3A

BLUEPRINT NUMBERING

GENERAL

The blueprint and part numbering systems of the airplane are broken down so that all parts within the basic components of the airplane will be numbered within the same series of numbers.

Prefix numbers are used to indicate under which contract the part was originally designed. Basic part numbers assigned to a part or blueprint are determined by the component for which the part has been designed, e.g., flap, 18 000 Series; aileron, 16 000 Series. Dash numbers are used to identify detail parts used in an assembly and to indicate right and left parts. Left parts have an odd-numbered dash number, and right parts are identified with an even-numbered dash number.

For example, a part number of 223-14006-3 would indicate that the part had been designed under the 223 contract, that it was designed for the wing, and that it was for the left assembly.

This illustration shows the basic airplane components and the number series assigned them.

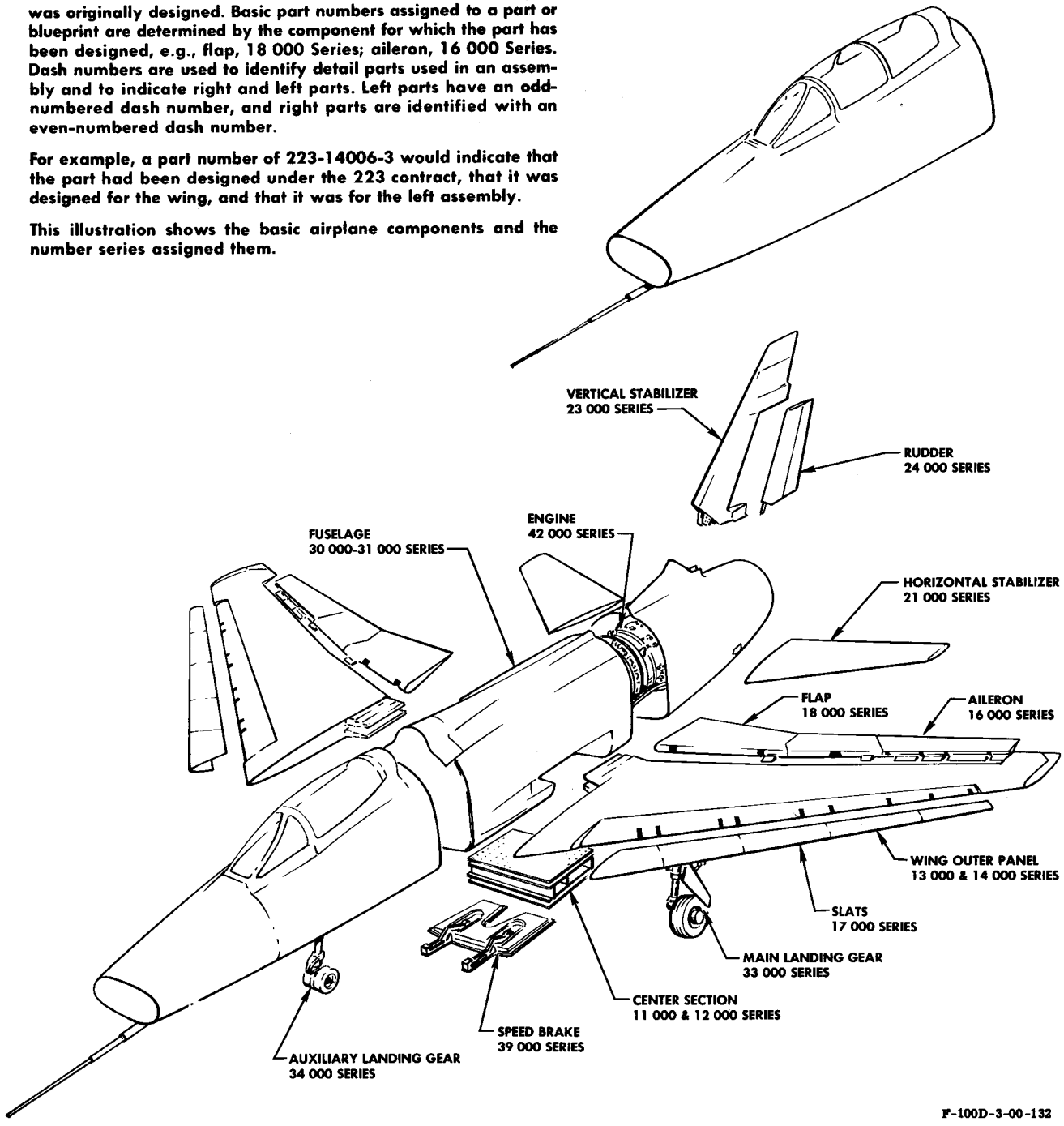
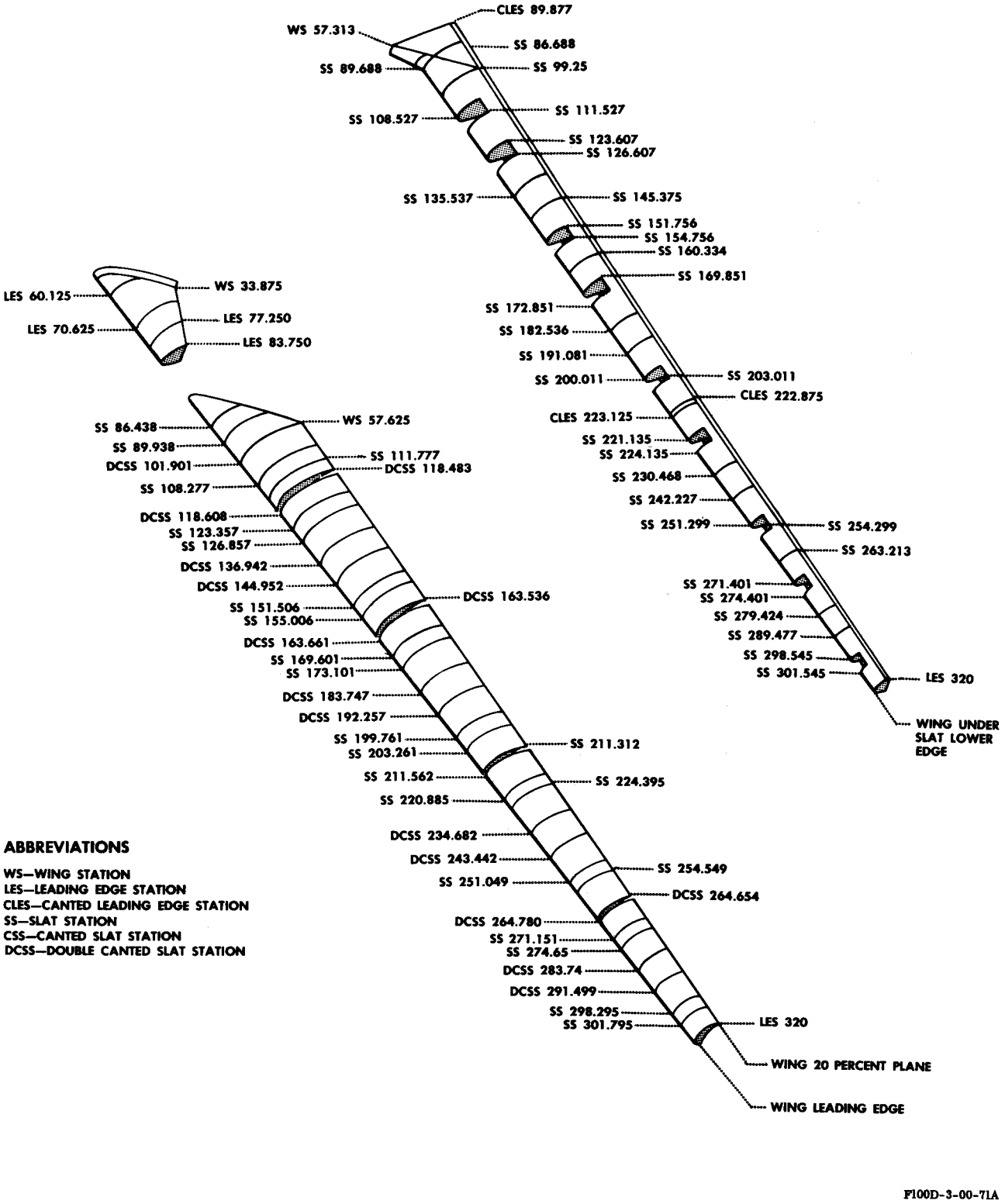


Figure 1-3B. Structural Drawing Numbering System

AIRPLANE STATIONS

GENERAL



ABBREVIATIONS

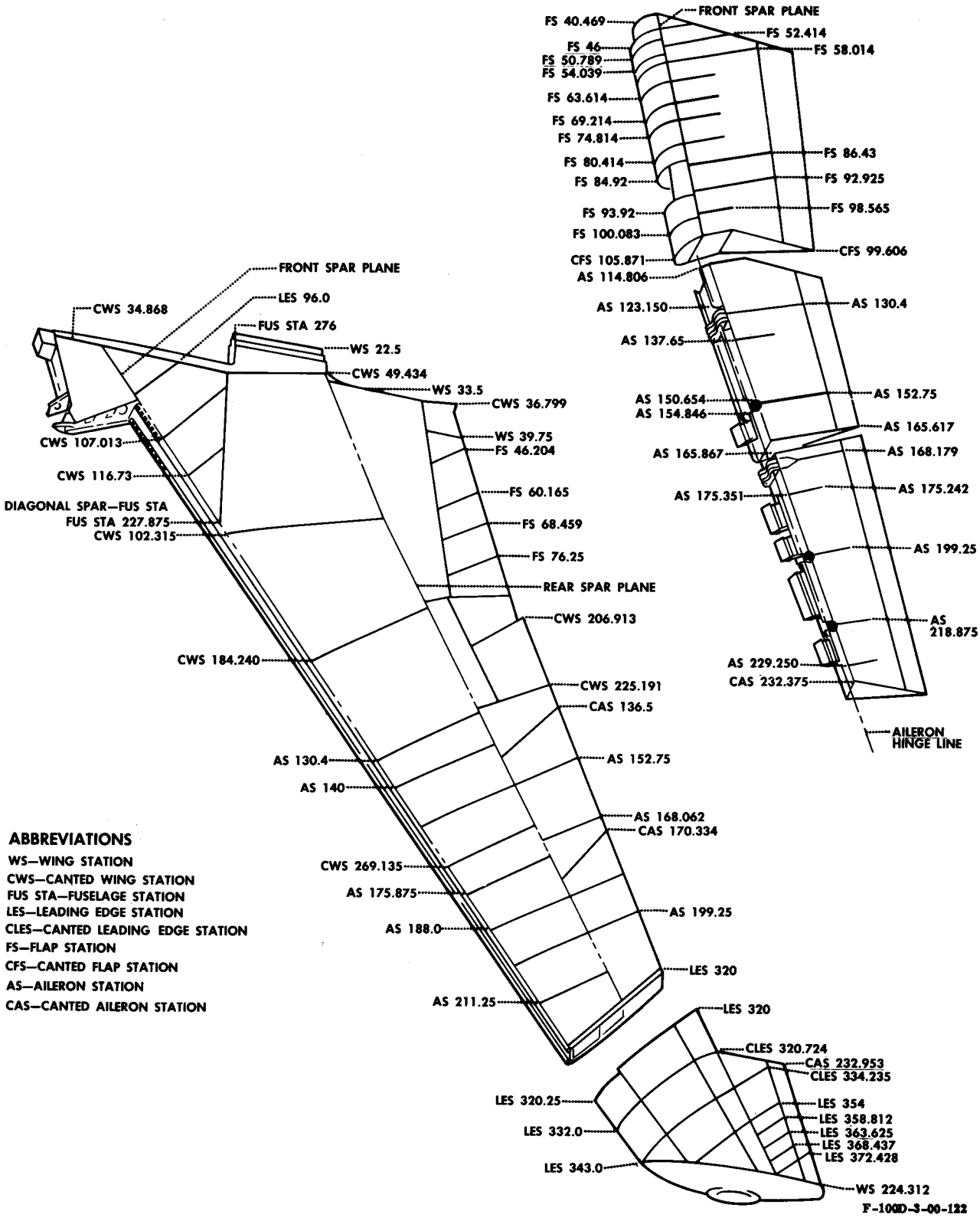
WS—WING STATION
LES—LEADING EDGE STATION
CLES—CANTED LEADING EDGE STATION
SS—SLAT STATION
CSS—CANTED SLAT STATION
DCSS—DOUBLE CANTED SLAT STATION

F100D-3-00-71A

Figure 1-4. Airplane Stations—Wing (Sheet 1 of 2)

AIRPLANE STATIONS

GENERAL



ABBREVIATIONS

WS—WING STATION
CWS—CANTED WING STATION
FUS STA—FUSELAGE STATION
LES—LEADING EDGE STATION
CLES—CANTED LEADING EDGE STATION
FS—FLAP STATION
CFS—CANTED FLAP STATION
AS—AILERON STATION
CAS—CANTED AILERON STATION

Figure 1-4. Airplane Stations—Wing (Sheet 2 of 2)

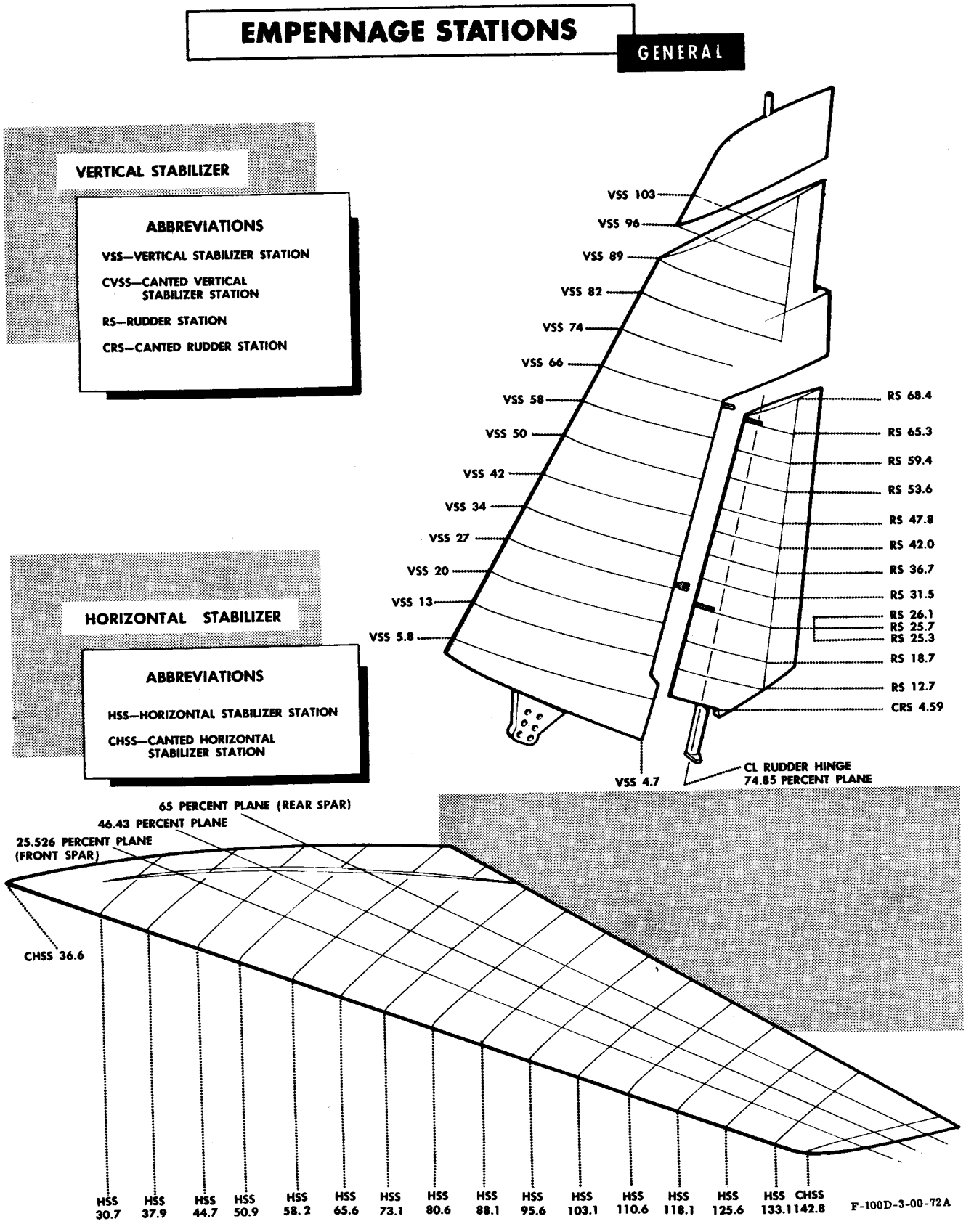


Figure 1-4A. Airplane Stations—Empennage

FUSELAGE STATIONS

GENERAL

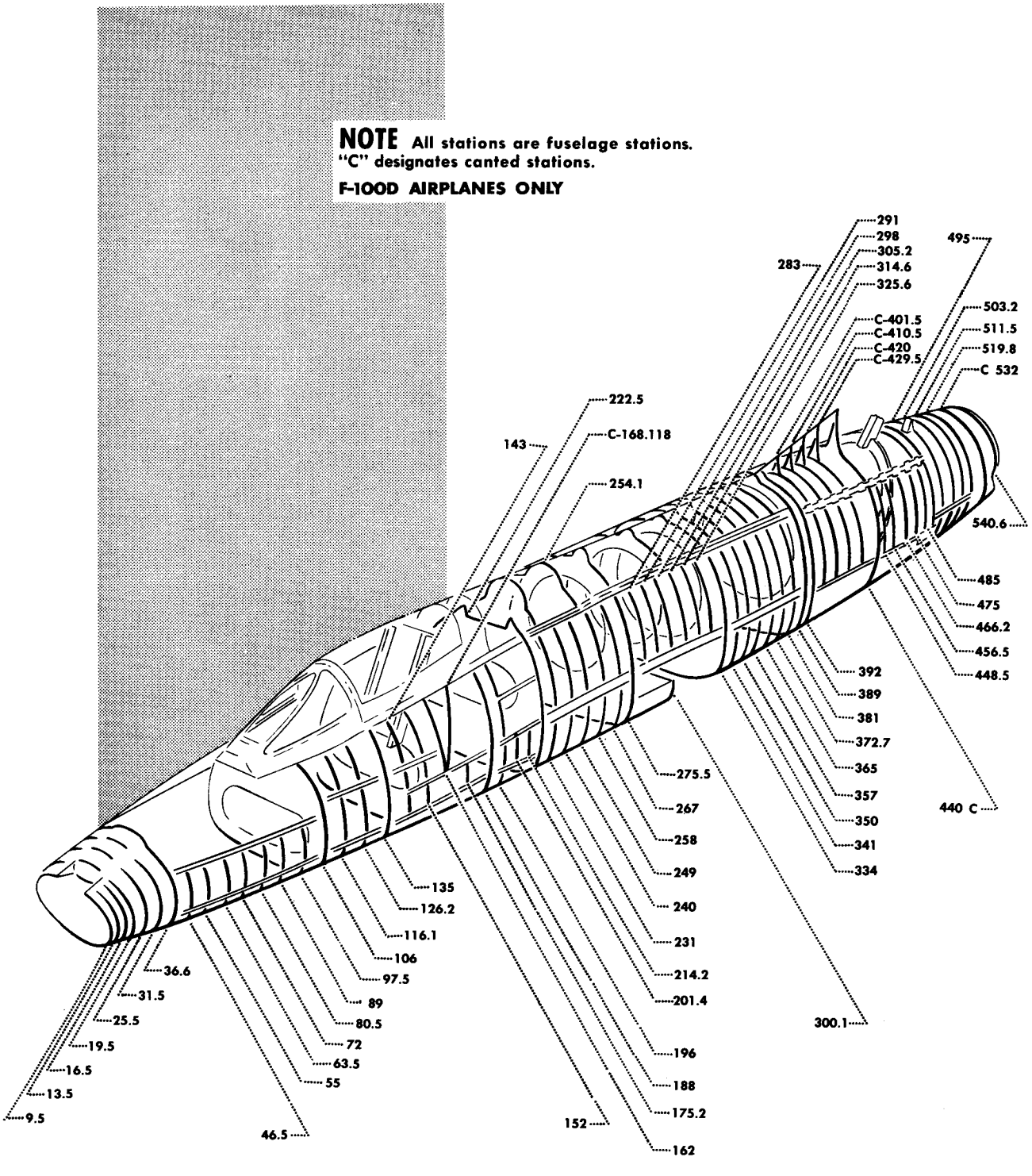
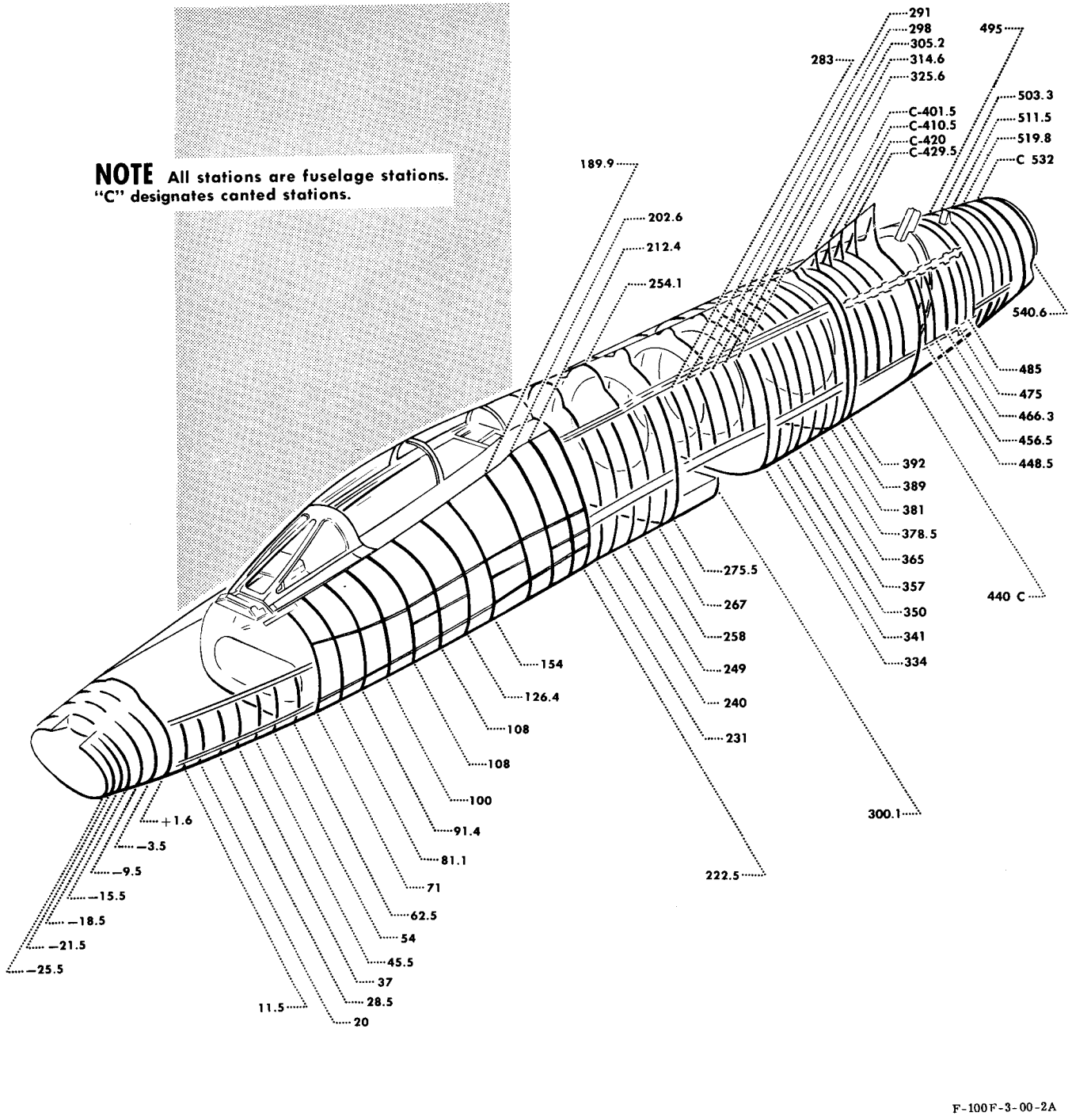


Figure 1-4B. Airplane Stations—Fuselage—F-100D Airplanes

FUSELAGE STATIONS

GENERAL



F-100F-3-00-2A

Figure 1-4C. Airplane Stations—Fuselage—F-100F Airplanes

LOCATING AIRPLANE STATIONS

GENERAL

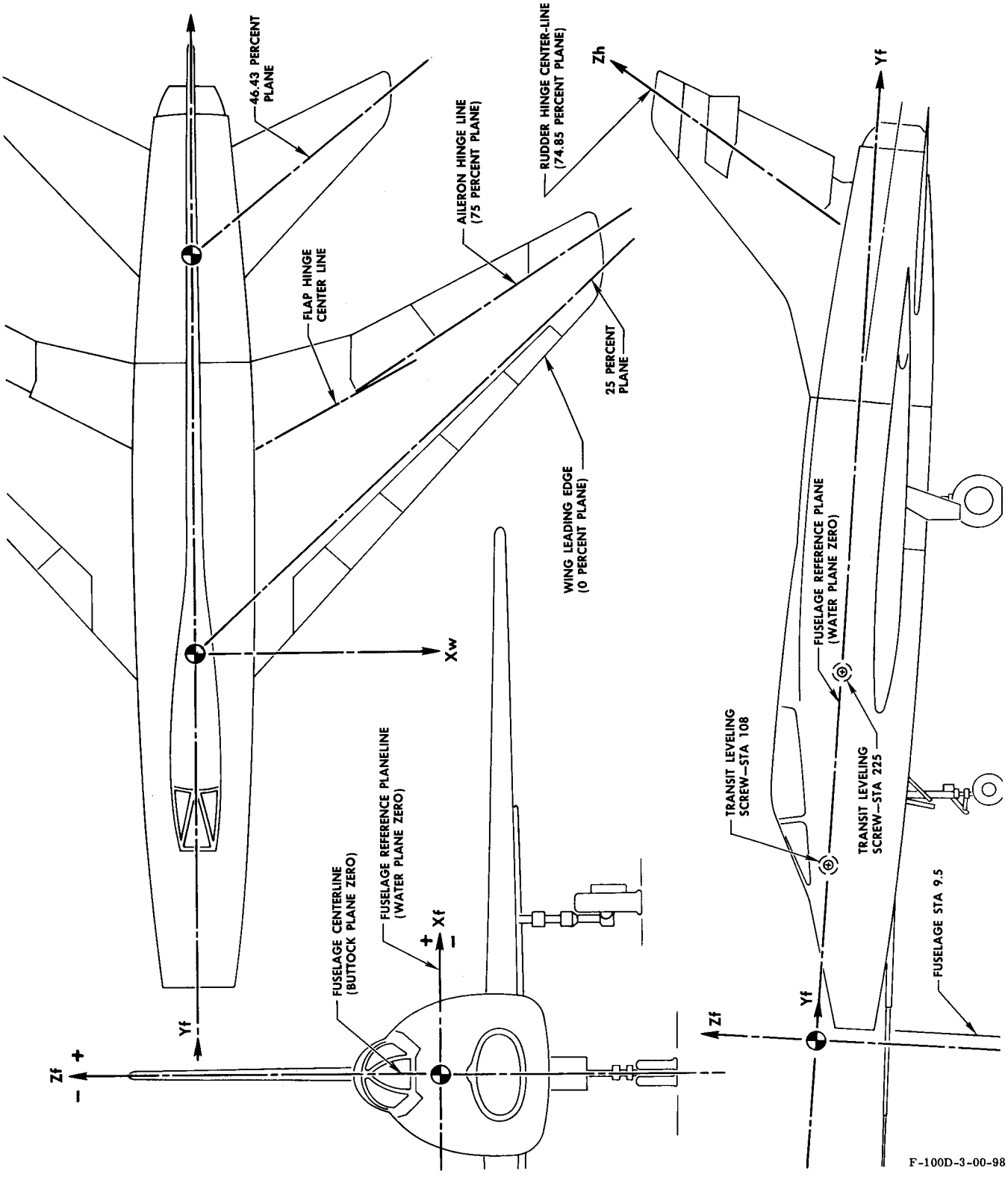


Figure 1-5. Locating Airplane Stations

airplane to be ferried to an overhaul base. Flight restrictions will be placed on the airplane as follows:

Maximum airspeed	300 knots
Maximum G-load	+3G, −1G

A placard will be placed in the cockpit and shall read as follows:

WARNING

FLIGHT OF THIS AIRPLANE IS RESTRICTED TO A MAXIMUM AIRSPEED OF 300 KNOTS AND A MAXIMUM G-LOAD OF +3G, −1G.

Each section of the handbook contains a one-time flight diagram which indicates the area of the airplane that may be damaged but that will not prevent the airplane from being flown to a repair base. Repair to these areas is not required for ferry flight. One-time flight repairs are included in the handbook and will be used as indicated on the individual repair.

NEGLIGIBLE DAMAGE.

Negligible damage is that damage which can be permitted to exist as is, or which may be corrected by a simple procedure, without restricting flight. In most cases, a corrective action must be taken to keep the damage from spreading. Frequent inspection of areas with negligible and minor damage is necessary to ensure that the damage does not spread. Some examples of negligible damage are:

1. Dents. Minor dents with no damage to structure which may be left as is, and large dents after being bumped back to mold line dimension, provided there are no sharp dents or wrinkles.

2. Scratches and nicks. A damage which does not go through the protective coating of the material, or through the cladding on alclad sheet, is not considered a scratch. To determine whether the damage has gone through the cladding, thoroughly clean the surface with methyl ethyl ketone and apply a 10% solution of sodium hydroxide to the damage area. If the damage has gone through the cladding and into parent material, a black or dark brown discoloration will appear. This solution should not remain on the area for longer than two minutes because of its corrosive action. The solution must be thoroughly washed from the area with water.

Scratches or nicks are not considered negligible until after they have been reworked by blending or other prescribed methods. Scratches or nicks may reduce the cross sectional area of a material and produce localized stress concentration. This condition can lead to fatigue cracks and possible failure of the component. (See figure 1-6.)

Blending of scratches or nicks is the process of smoothing out the damage area. To blend properly, a fine grit (400 or finer) abrasive cloth or paper, or aluminum wool should be used. Do not use crocus cloth or steel wool since it tends to cause corrosion. Scratches or nicks should be smoothed out with the abrasive to remove all damage. The reworked area must be kept as small as possible. (See figure 1-7.)

In certain highly stressed areas, it is necessary to use a dye penetrant check after blending to reveal any cracks which may be undetected. (Refer to "Penetrant Method of Inspection.")

After blending, the damaged area must be chemical film treated (refer to "Corrosion Rework by Chemical Film Treatments") and painted.

Limitations regarding the length and depth of scratches and nicks vary in different locations throughout the airplane. These limitations are shown on the illustrations of negligible damage for the various areas.

3. Corrosion damage. Corrosion damage which does not exceed the prescribed limits for scratches and nicks may be considered negligible if the corrosion is cleaned out and the area chemical film treated (refer to "Corrosion Rework by Chemical Film Treatments") and painted.

4. Cracks. This type of damage usually originates at edges, holes or points where concentrated loads are applied or abrupt changes occur in the cross-sectional area. Limitations regarding length of crack, distances from other damages, specific locations, etc, vary in

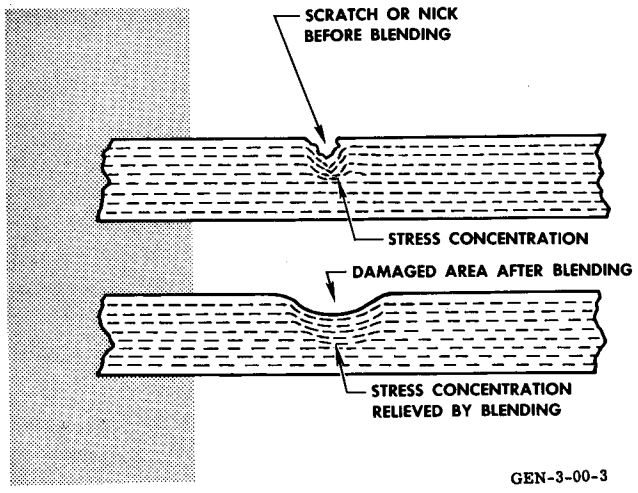
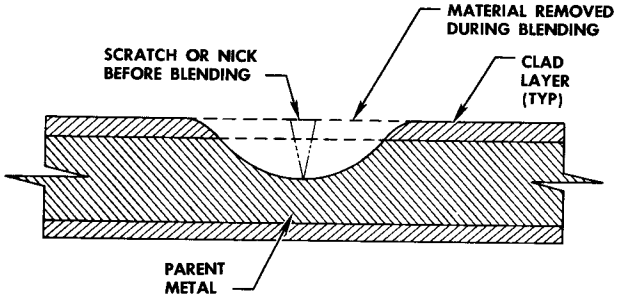


Figure 1-6. Stress Concentration at Nicks and Scratches



NOTE Blending scratch or nick in machined bar or extrusion may be accomplished in same manner.

GEN-3-00-4

Figure 1-7. Blending Scratches

different locations throughout the airplane. These limitations are shown on the illustrations of negligible damage for the various areas. All cracks will be stop-drilled at each end.

5. Holes. Cleaned-up holes that do not reduce the cross-sectional area of a material enough to introduce prohibitive stress, or which do not interfere with the function of a component (including pressurized areas and fuel-tight areas) may be considered negligible. Limitations regarding size of holes, distances from other damages, etc, vary in different locations throughout the airplane. These limitations are shown on the illustrations of negligible damage for the various areas. All holes in external skins must be filled. Aerodynamic skin filler may be used in very small holes. For filling larger holes, refer to figure 10-37.

PERMANENT REPAIR.

Permanent Repairs are those which put the airplane, both structurally and aerodynamically, back to its original configuration. Permanent repairs may be made by one of the following methods:

1. Patching the damage. Repairing damage by patching is to completely encircle or bridge the damaged area of a component with splice material of the same type as the original. Holes and cracks which exceed the prescribed limitations for negligible damage in a given area may be repaired by this method. Filler plates may be used for bearing surfaces or for returning the part to its original contour when necessary.

2. Repairing damage by insertion. Damage repairable by insertion is that damage which can be repaired by splicing in a section of material to replace the damaged material. The section used for insertion should be identical in shape and type to the damaged part. The inserted material is fastened into place by a splice. The splice connection to the original structure provides for load transfer between the original structure and the inserted section.

3. Repair of damage by replacement of parts. The part is generally replaced when a repair cannot be made by practical means. When replacing a damaged component or part, care should be taken to avoid damage to adjacent or attaching structure.

TEMPORARY REPAIR.

Temporary repairs are repairs that return the airplane to its original configuration structurally, but do not fully restore it aerodynamically. External (scab) patches, protruding head fasteners, etc, are temporary repairs. (Refer to "Aerodynamic Smoothness.") In spite of the undesirable effects of nonflush (temporary) repairs, they are sometimes necessary to make airplanes available for flight in the shortest possible time. They are *temporary repairs* however, and they should be replaced with permanent (flush) repairs as soon as possible.

When the wings, vertical stabilizer, or control surfaces have been repaired in any manner which could possibly affect flight characteristics, a test flight should be made to determine what flight restrictions are necessary.

AERODYNAMIC SMOOTHNESS.

Modern airplane specifications require that the airplane exterior surfaces be of extremely smooth construction to obtain design performance. The structural repairman must keep this in mind when making repairs or changes. Figure 1-8 shows the areas that are most critical aerodynamically; however, smoothness of the entire airplane surface area is a critical requirement from the performance standpoint. The seriousness of this condition is proportional to the airflow disturbance created.

The prime considerations of aerodynamic smoothness are parasite drag (drag caused by skin friction and nonlifting surfaces of an airplane) and the aerodynamic load distribution on the surface. Any repair or change which increases parasite drag will reduce the top speed of the airplane. Also, combat range will be reduced, since more power, and thus more fuel, is required to maintain cruising speed.

Any change which disturbs the smooth flow of air over a critical surface will change the load on that surface, as well as increase the drag. This effect is greatly increased at transonic and supersonic speeds because of shock wave formation in these speed regions. Since, in most cases, a shock wave results in a radical change of the load distribution, and an increase in drag, a patch that will cause no difficulty (disturbance) at low speed may become critical when a shock wave is produced by this patch in the transonic and supersonic speed

AERODYNAMICALLY CRITICAL AREAS

GENERAL

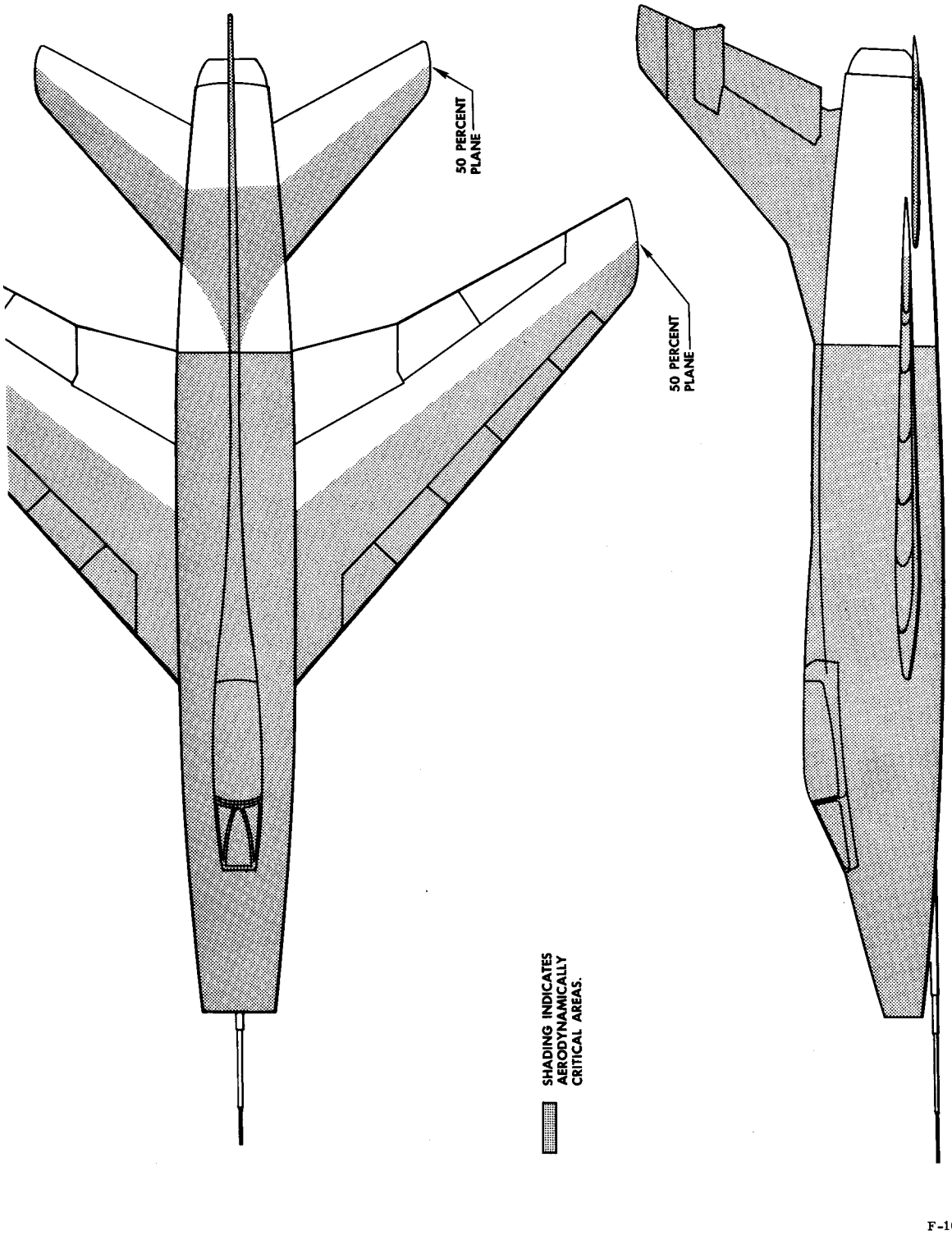


Figure 1-8. Aerodynamically Critical Areas

regions. If this flow change occurs on the wing, a wing-heavy condition proportional to the amount of disturbance will result. If this condition is on the vertical stabilizer, a yawed flight condition will result. These conditions became increasingly serious as control effectiveness drops off at high Mach numbers. Any change in the airflow around the control surfaces is critical since the control effectiveness, particularly at high Mach numbers, may be affected.

Another problem that may arise at the high speeds is buffeting. Buffeting can be caused by airflow separation around an abrupt change, such as a nonflush patch, in the mold line. Nonflush repairs are authorized as *temporary repairs* only unless otherwise noted on the repair. (Refer to "Temporary Repairs.")

CAUSES AND TYPES OF DAMAGE.

There are many causes and degrees of damage which may occur to an airplane. However, for purposes of description some of the causes are:

1. Collision or impact damage. This type of damage varies from minor damage, such as a small hole or dent, to extensive damage such as torn, crushed, or burned skin and structural members, and misalignment of the airplane. Damage such as this can occur at any point on the airplane.

2. Stress damage. This type of damage generally consists of loosened, sheared, or popped rivets, wrinkled skin or webs, and cracked or deformed structural members. Damage such as this is caused by abnormal stresses imposed upon the members.

3. Fatigue damage. This type of damage generally consists of small cracks which are caused by vibration, oil cans, and variable loads imposed on skin, fittings, and load bearing members where fittings are attached. Damage of this type occurs more frequently as the total operation time of the airplane accumulates.

4. Corrosion damage. This type of damage may occur where dissimilar metals come in contact with each other or where certain alloys or steel are exposed to moisture. Corrosion damage may be detected on aluminum alloys by the presence of white powdery spots or deposits, and pitting of the metal. Rust is a form of corrosion on steel surfaces.

5. Fire damage. This may or may not be the result of fire after collision or impact damage, landing accidents, or fire in flight.

CRASH HANDLING AND SHIPPING.

The purpose of this information is to guide personnel who handle crash damaged airplanes. It is the responsibility of the crash crews to make sure that the airplane is properly handled at the crash site. Every attempt must be made to keep crash damage to a minimum. A good knowledge of the airplane structure and the use of crash equipment will be an aid in airplane handling. It is recommended that crash handling equipment, of the type described in this handbook, be obtained and that personnel be trained in the techniques of using it.

All items suggested for handling damaged airplanes may be obtained through regular procurement sources. Tools and equipment which are to be fabricated locally are made from commercial materials.

SAFETY PRECAUTIONS BEFORE CRASH HANDLING.

The aim at all times in crash handling damaged airplanes is to work with the maximum personnel safety that circumstances permit. However, many different conditions affect the degree of safety attainable. These conditions include time available, structural condition and attitude of the airplane, equipment available, locale of the accident, and armament carried.

PREFERRED SAFETY CONDITIONS.

The preferred condition of the airplane (and the condition which must be strived for) is when it is completely inerted. To achieve this state, the following must be done:

- Battery disconnected (or removed).
- Seat and canopy explosive charges removed.
- All ammunition and external stores removed.
- Fuel drained and the system nitrogen purged.
- Oxygen system drained.

Refer to the applicable Systems Maintenance Manuals (T.O. 1F-100D-2 series and T.O. 1F-100F-2 series) for procedures in accomplishing all preceding operations. All handling equipment should be either approved for the specific airplane or approved for the class of aircraft being handled.

ALTERNATE SAFETY CONDITIONS.

Circumstances often dictate a less than satisfactory degree of crash handling safety, particularly during the early stages of the operation. The following discussion deals broadly with those conditions.

TIME AVAILABLE. Time can be extremely important, as when a runway must be cleared for emergency landings. One method of removal under such conditions is shown in figure 1-8A. Here a crash truck is the towing vehicle. A bed of foam provided by the crash truck supplies a wetting and smothering agent to minimize the possibility of fire and to reduce friction. Even under such emergency conditions, the battery should be disconnected, seat and canopy ejection units safetied, and personnel should be clear of the airplane. This removal method should be used to clear the runway for emergency flight operations only. Subsequent moving should be accomplished by conventional methods.

STRUCTURAL CONDITION AND ATTITUDE OF THE AIRPLANE. The extent of structural damage and the position of the airplane are often determining factors in how safe the airplane can be made before removal operations are begun. All sources of potential danger, including electricity, fuel, armament, and oxygen should be neutralized as much as possible under the conditions. However, a desirable degree of safety may have to be accomplished in steps. It may be necessary to lift the airplane to gain access to areas, and to cut into deformed structure to reach or to free hazardous items for removal. In these instances, the minimum number of qualified personnel should be employed.

EQUIPMENT AVAILABLE. Proper airplane lifting and transporting equipment is necessary for maximum handling safety. The condition or attitude of the airplane, however, does not always permit use of the equipment as intended. Also, it may be necessary to work with equipment not specifically designed for the airplane. In this case it is particularly important to make sure that all equipment used is capable of lifting and carrying the loads. When the lifting and supporting points provided cannot be used, the selection of substitute points should be made with care (lifting points for their ability to carry the load, and supporting points which will not collapse or otherwise allow the load to shift).

LOCATION OF ACCIDENT. The geographical location and climatic conditions in which the crash occurred can seriously affect crash handling procedures. Substitute equipment may be required because of distance, terrain, or for other reasons. Such equipment calls for unusual handling procedures. However, the airplane must be safetied to the maximum extent possible, as discussed in previous paragraphs.

ARMAMENT. The explosives carried and the condition they are in can cause the most serious hazard in crash handling. Before any airplane is moved, the armament condition must be determined. When the structural condition or attitude of the airplane does not permit complete dearming, all movement of the airplane before dearming should be done with the advice of qualified armament specialists and in strict compliance with all armament directives.

EMERGENCY REMOVAL OF AIRPLANE FROM RUNWAY.

The emergency removal method shown in figure 1-8A should be used only when a runway must be cleared in the shortest possible time. In all other instances, methods which will not create additional damage should be employed.

Towing an all-gear-up airplane with a cable installed, as shown, will add damage, but is considered the least damaging method of attachment when an airplane is in this condition.

DAMAGED WHEEL TOWING DOLLY.

This dolly is a self-contained unit that can be used to quickly move an airplane with damaged main gear wheels and/or tires, from the runway. This versatile dolly can be rapidly towed to the damaged airplane, may be operated by two men to lift the airplane without the aid of other jacks or equipment, and can be used on both F-100 and F-86 Series Airplanes. One dolly will fit either right or left gear and may be used singly or in pairs. The large wheel area makes it possible to tow the airplane over rough ground. Another advantage of this dolly is that it can be made from scrap stock and surplus wheels and tires. For fabrication details, see figure 1-8B.

LIFTING AIRPLANE WITH PNEUMATIC LIFTING BAGS.

Airplanes that have made wheels-up landings on soft or swampy terrain may be lifted with pneumatic lifting bags (Part No. 44D5158, Type F-1, Specification No. 94-40683). This equipment may also be used for lifting belly-landed airplanes from runways in a minimum amount of time. The bags are similar in construction to bladder-type fuel cells. Each bag is capable of lifting a maximum weight of 24,000 pounds to a height of 6 feet. Operating pressure is provided by a gasoline engine-driven blower, which supplies 5 psi pressure at a volume of 40 cubic feet per minute. Each bag is provided with a tarpaulin, which serves the dual purposes of protecting the bag during storage and transportation, and acting as a base upon which to set the bag during the lifting operation.

MOORING AIRPLANE.

The airplane must be secured to prevent its moving forward, backward, or to either side while being lifted. (See figure 1-9.) Mooring must be provided at the tail to prevent the airplane from nosing over when bags are inflated. Ropes are tied from the aft mooring point to ground stakes. The two ropes running forward and out are secured at one end to the wing mooring points and at the other end to ground stakes or heavy trucks.

If for some reason the regular mooring points cannot be used, the aft tie-down may be made with soft rope around the aft fuselage, forward of the horizontal stabilizer. Cable or wire must not be used. The wing mooring lines may be tied into the slat tracks. These are alternate methods of mooring only and should not be used unless it is impractical to use the regular mooring points.

EMERGENCY REMOVAL

GENERAL

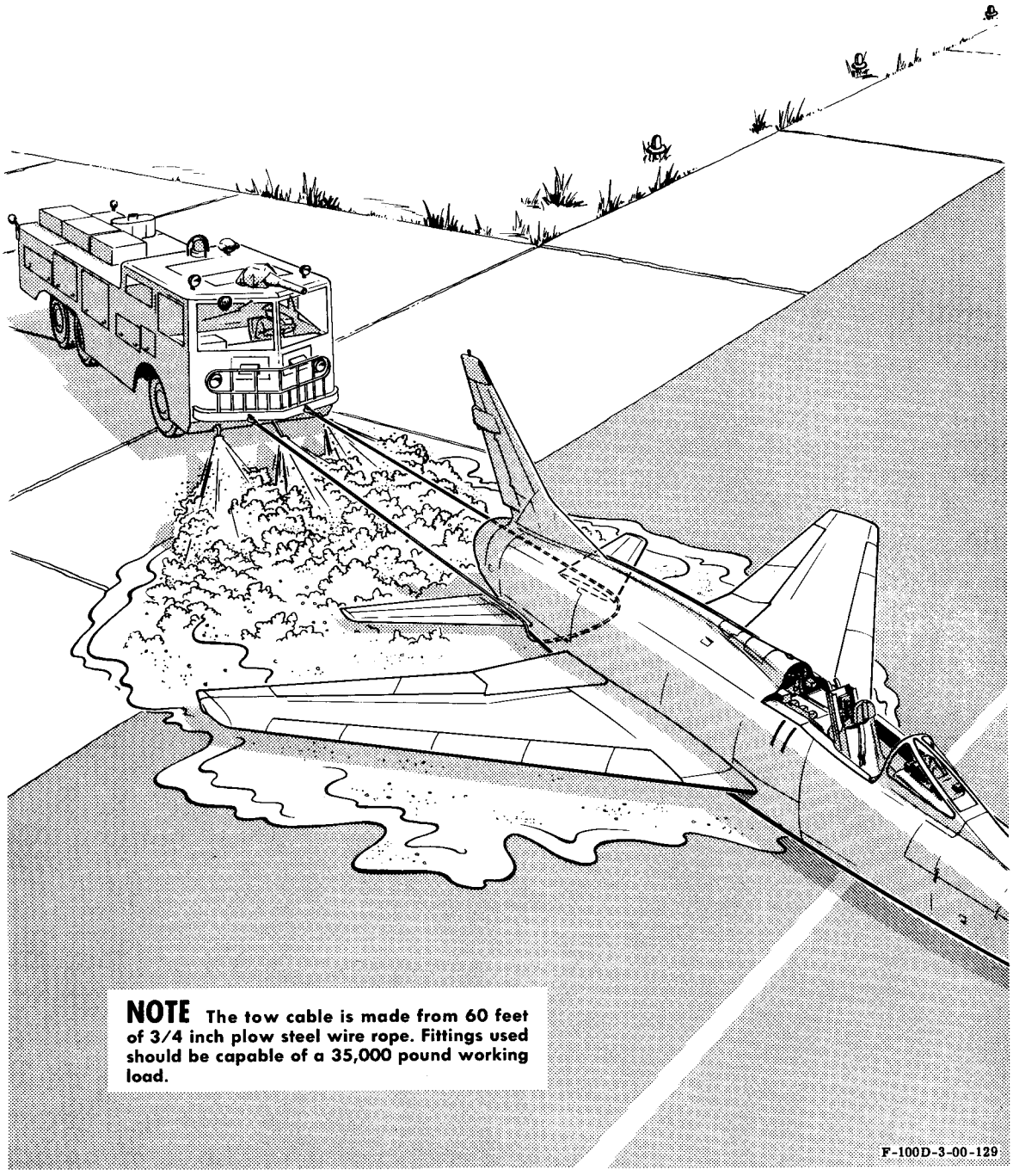


Figure 1-8A. Emergency Removal of Airplane From Runway

DAMAGED WHEEL TOWING DOLLY

GENERAL

Details of the towing dolly components are found on the sheets listed below:

ITEMS	SHEET NO.
1,2,3	2
4,5,6	3
7,8,9,10,11,12	4
13,14,15	5
16,17	6
18,19	7
20	5- or 10-ton hydraulic jack

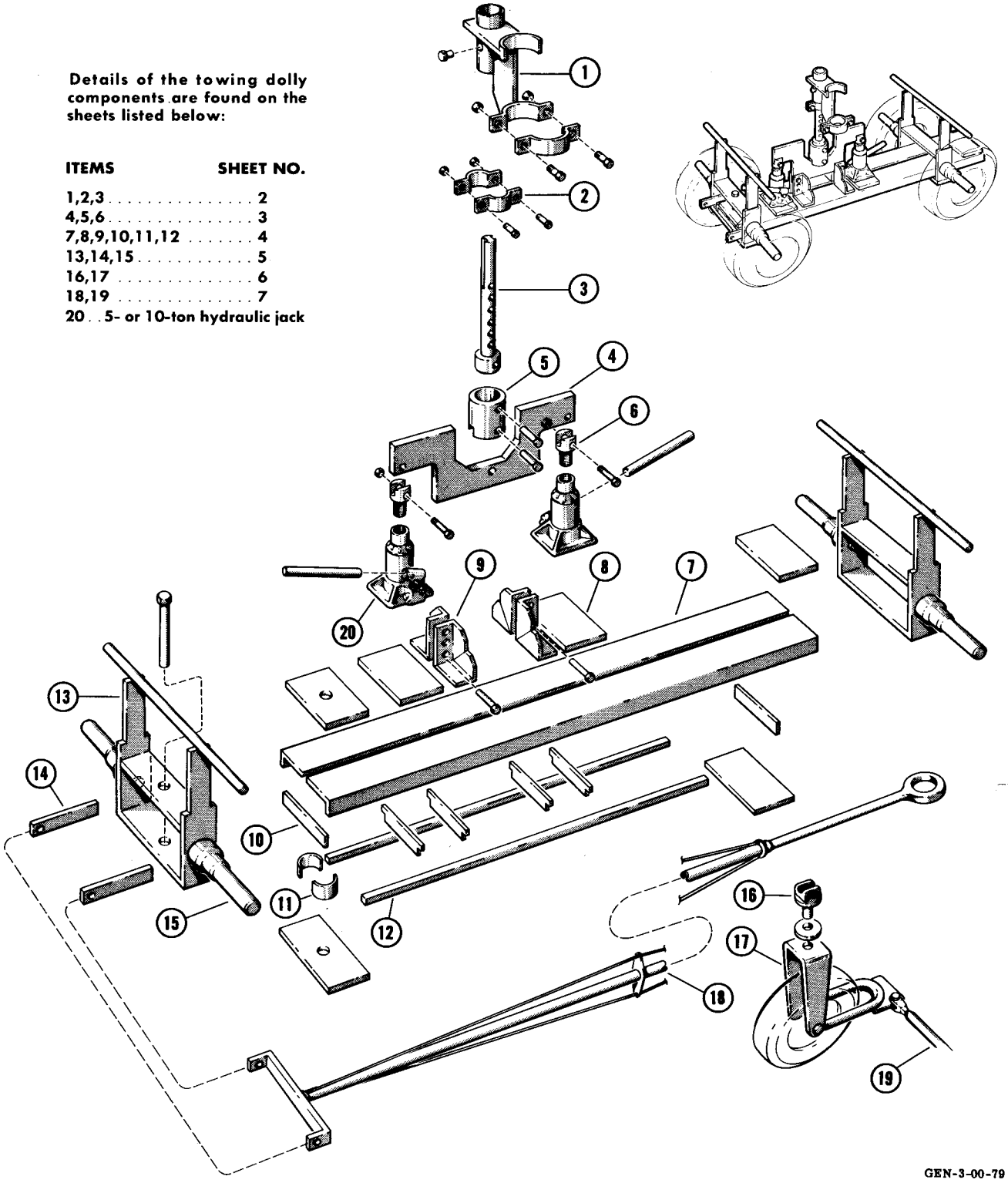
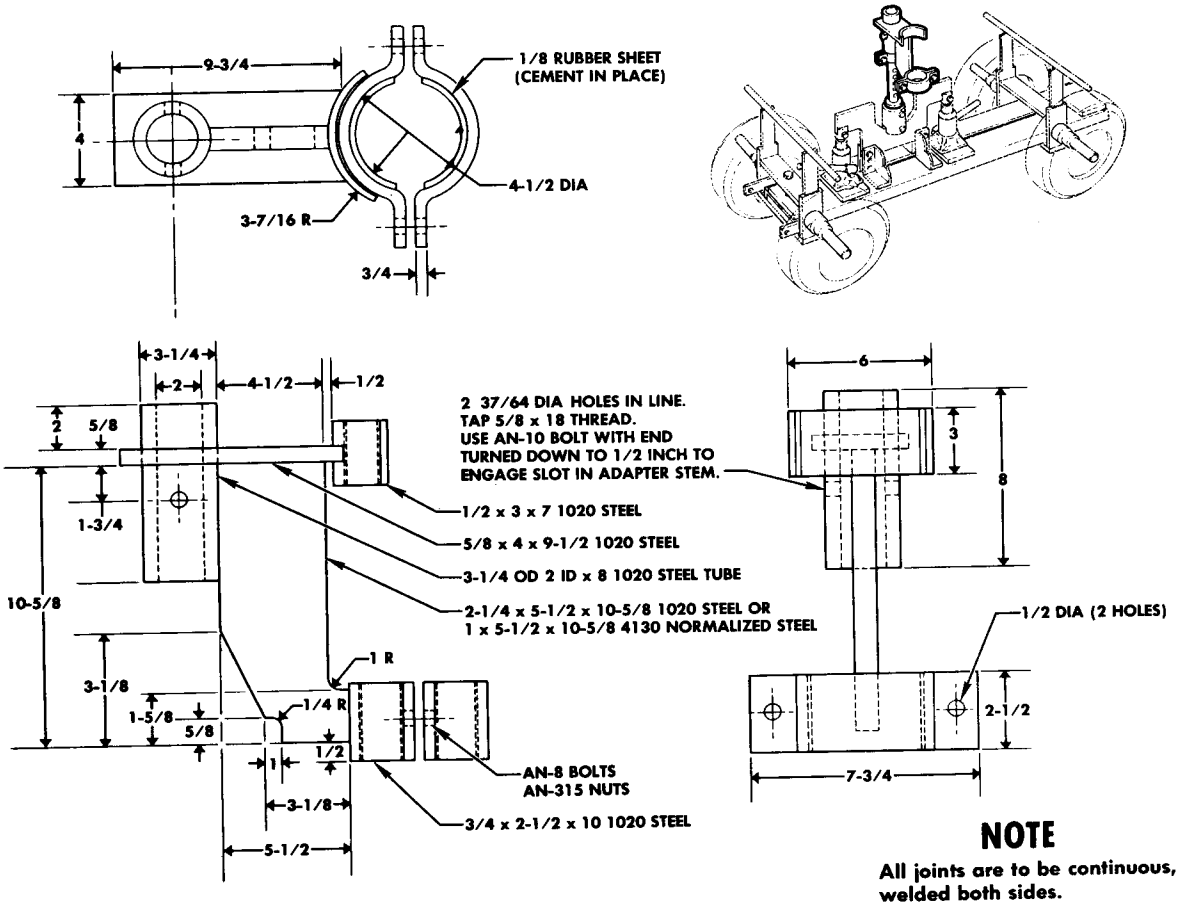


Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 1 of 8)

DAMAGED WHEEL TOWING DOLLY

GENERAL

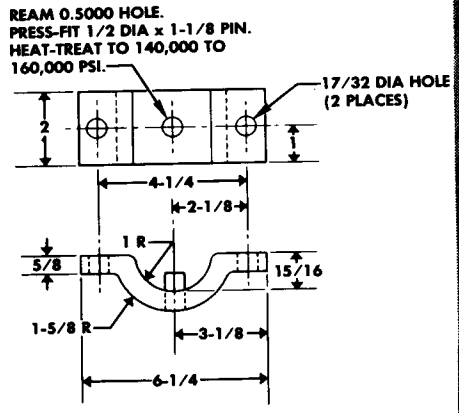


NOTE

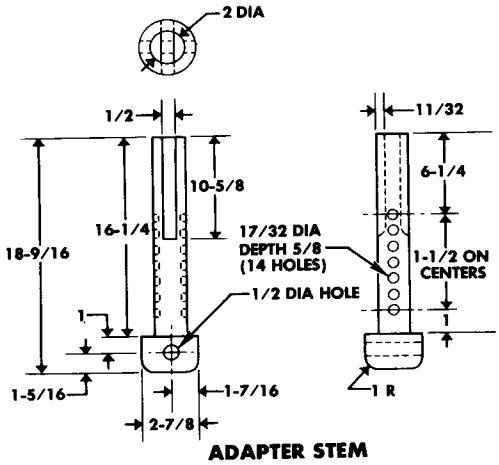
All joints are to be continuous, welded both sides.

JACKING ADAPTER ASSEMBLY

1



2 **ADAPTER STEM ADJUSTING CLAMP**
1020 STEEL (2 REQD)



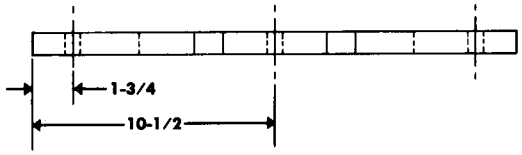
3 **ADAPTER STEM**
4130 STEEL

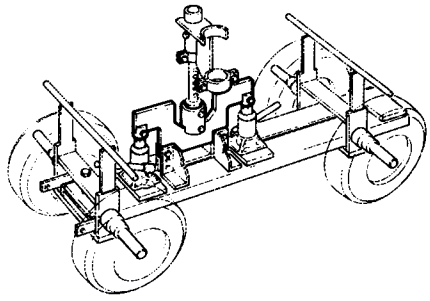
GEN-3-00-80

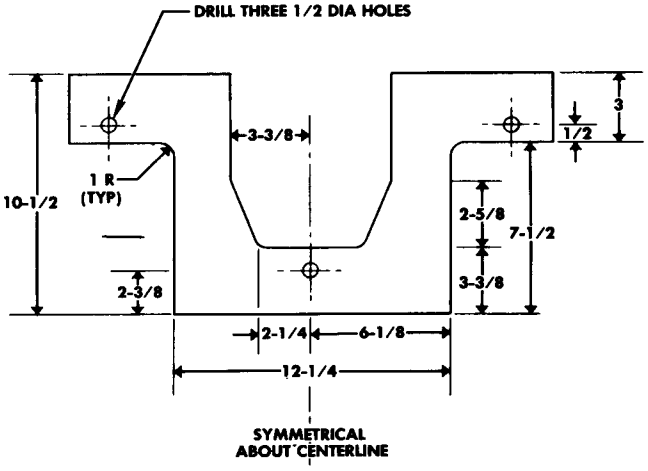
Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 2 of 8)

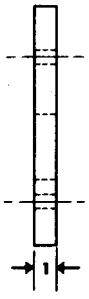
DAMAGED WHEEL TOWING DOLLY

GENERAL



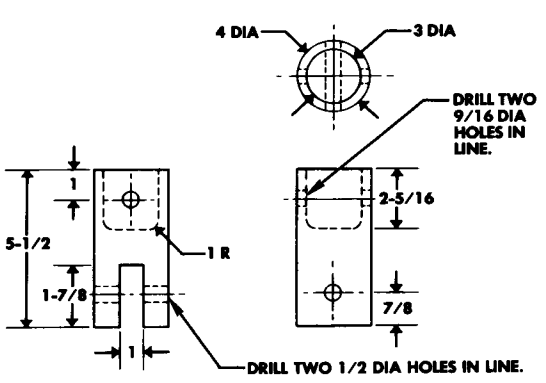






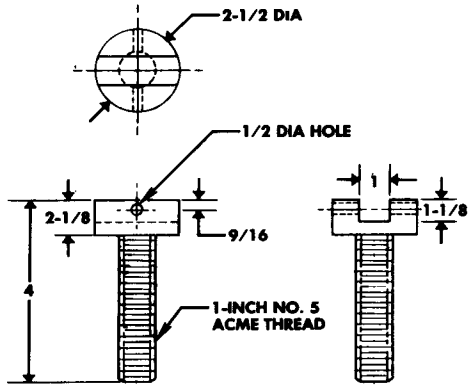
4

YOKE
1020 STEEL



ADAPTER PILLOW
1020 STEEL

5



JACK HEAD REPLACEMENT
4130 STEEL

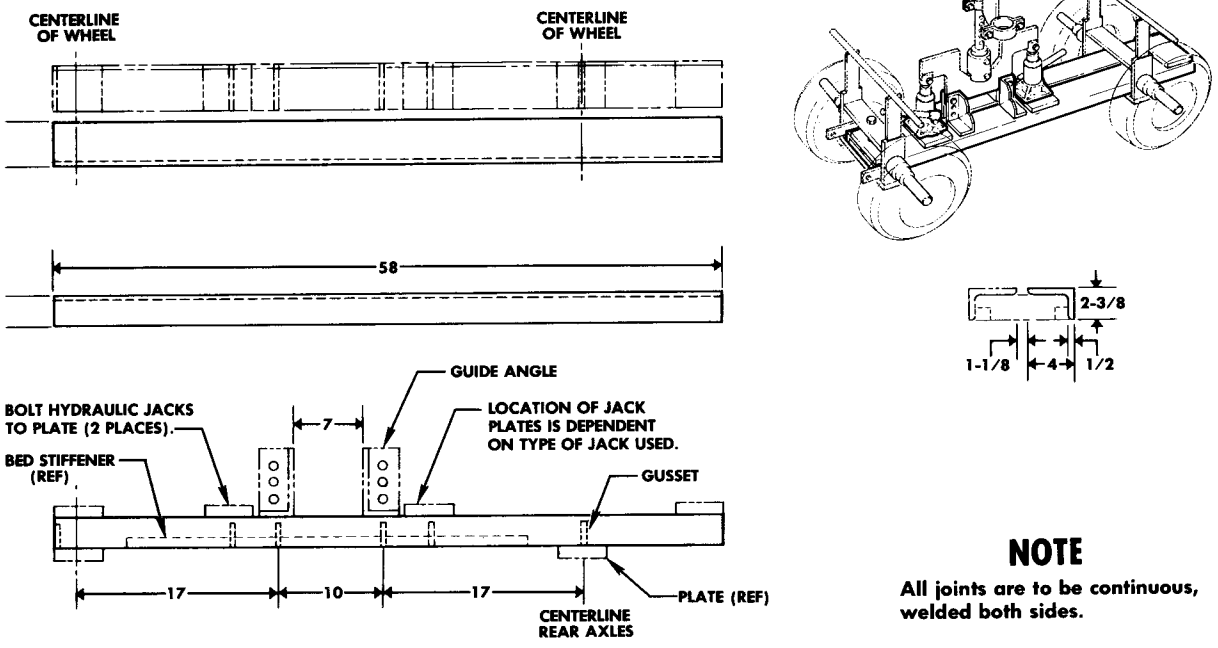
6

GEN-3-00-81

Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 3 of 8)

DAMAGED WHEEL TOWING DOLLY

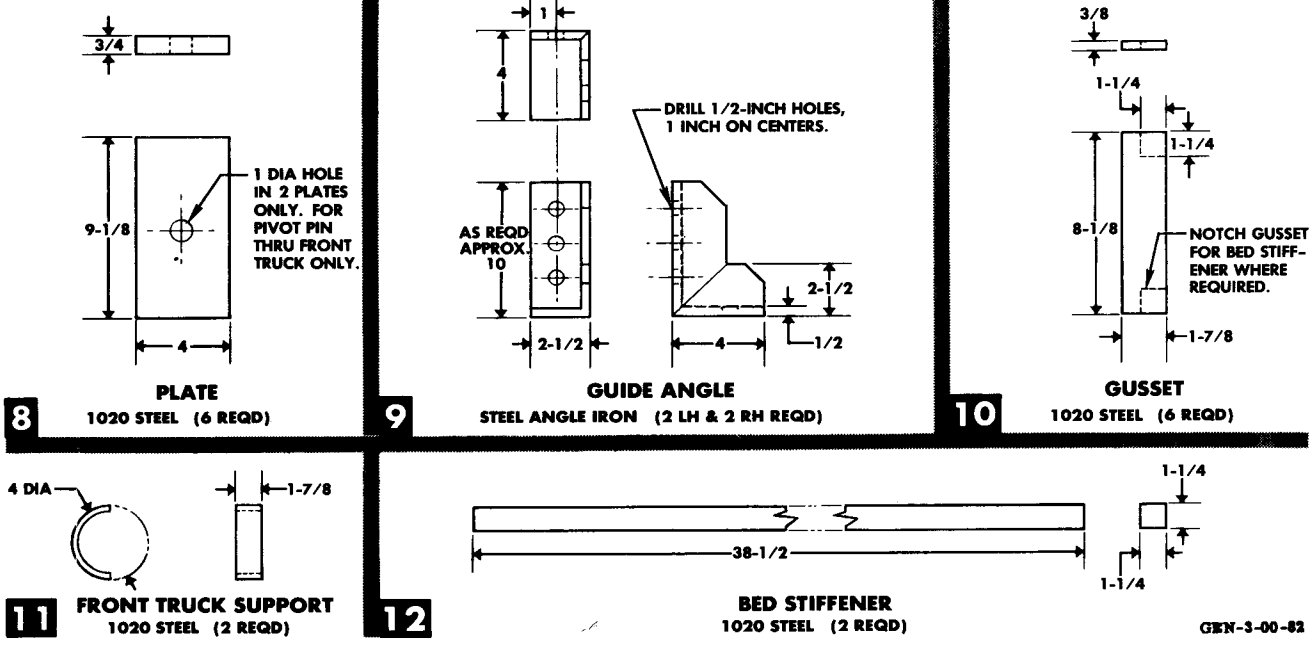
GENERAL



NOTE

All joints are to be continuous, welded both sides.

7 BED ASSEMBLY
STEEL ANGLE IRON



8 PLATE
1020 STEEL (6 REQD)

9 GUIDE ANGLE
STEEL ANGLE IRON (2 LH & 2 RH REQD)

10 GUSSET
1020 STEEL (6 REQD)

11 FRONT TRUCK SUPPORT
1020 STEEL (2 REQD)

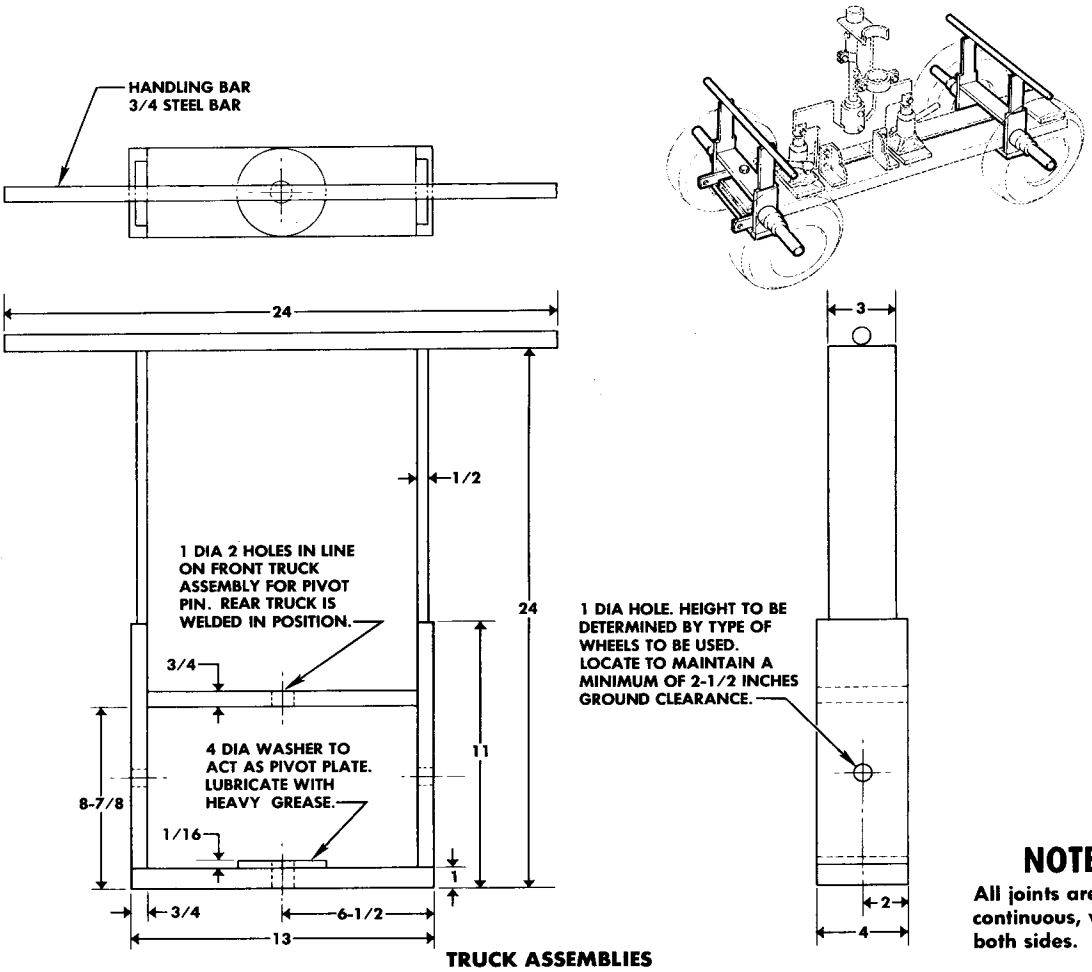
12 BED STIFFENER
1020 STEEL (2 REQD)

GEN-3-00-82

Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 4 of 8)

DAMAGED WHEEL TOWING DOLLY

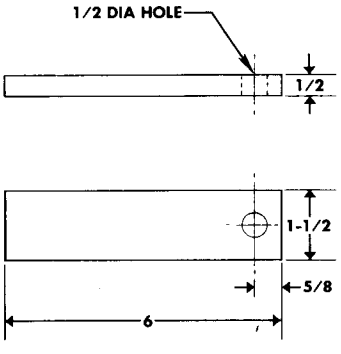
GENERAL



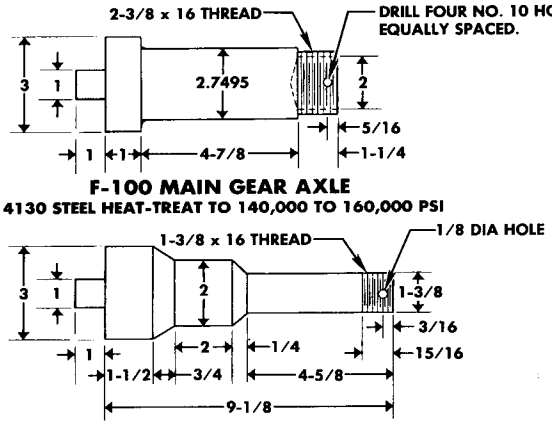
NOTE

All joints are to be continuous, welded both sides.

13 **TRUCK ASSEMBLIES**
1020 STEEL



14 **TOW BAR ATTACHMENT**
1020 STEEL



F-100 MAIN GEAR AXLE
4130 STEEL HEAT-TREAT TO 140,000 TO 160,000 PSI

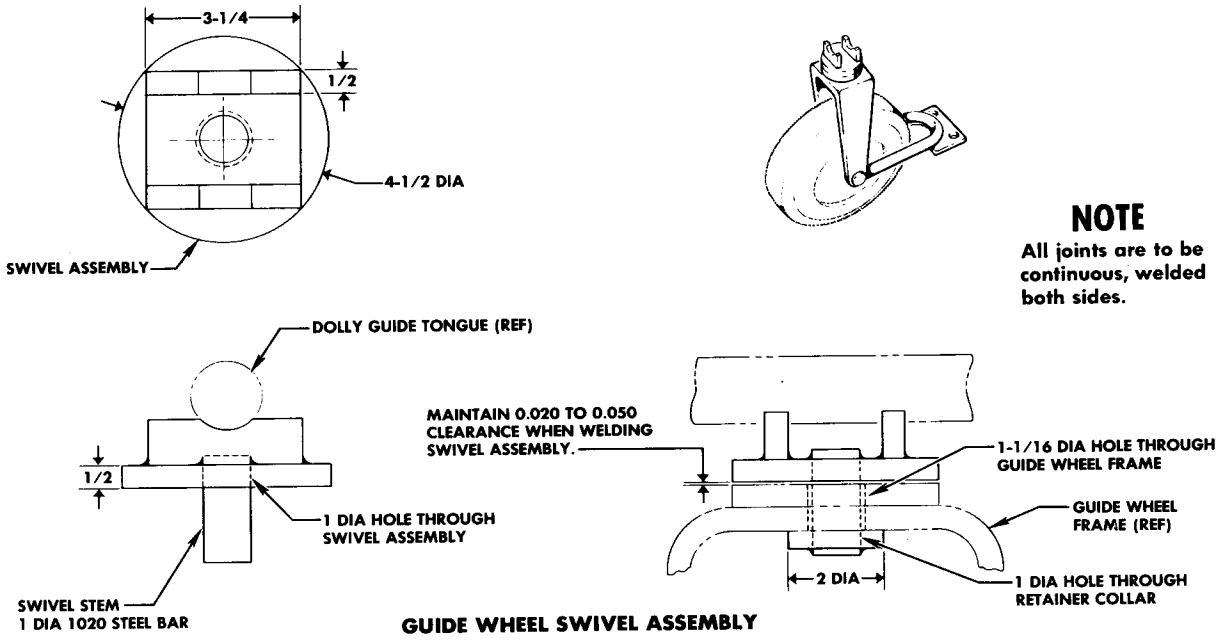
15 **T-33 NOSE GEAR AXLE**
4130 STEEL HEAT-TREAT TO 140,000 TO 160,000 PSI

GEN-3-00-83

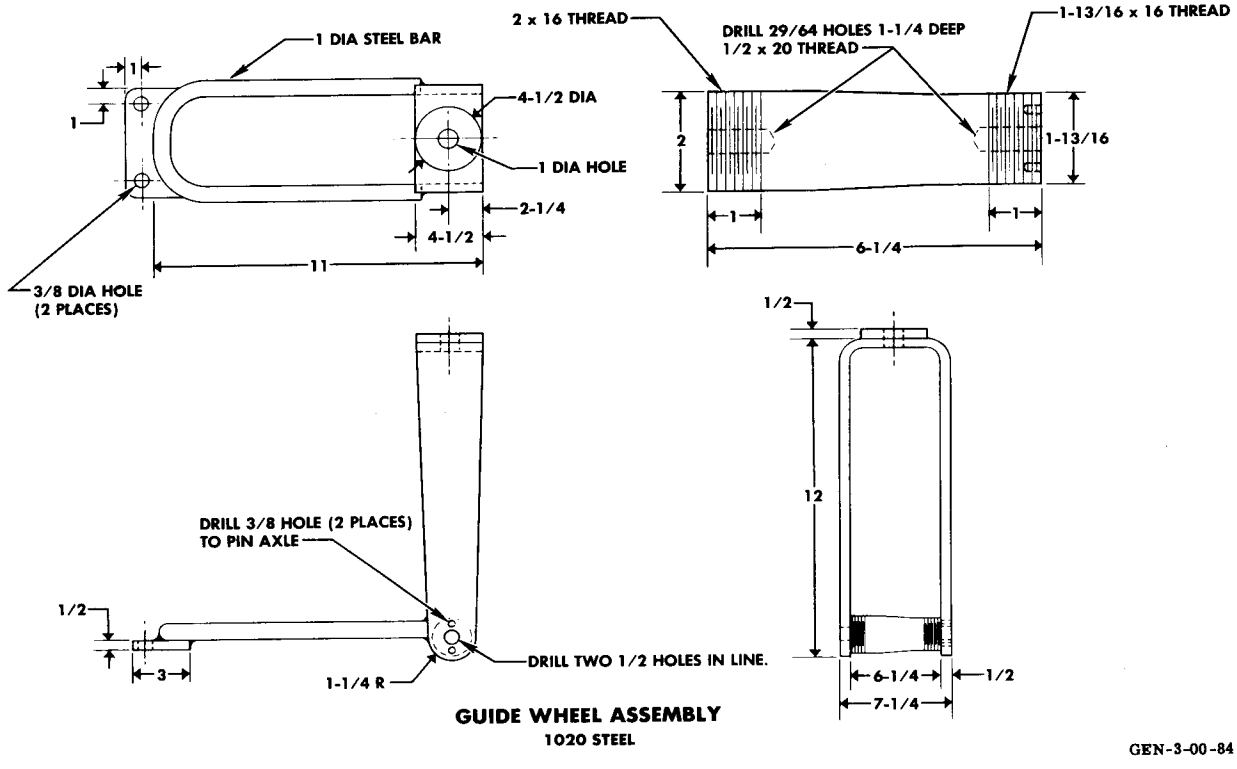
Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 5 of 8)

DAMAGED WHEEL TOWING DOLLY

GENERAL



16



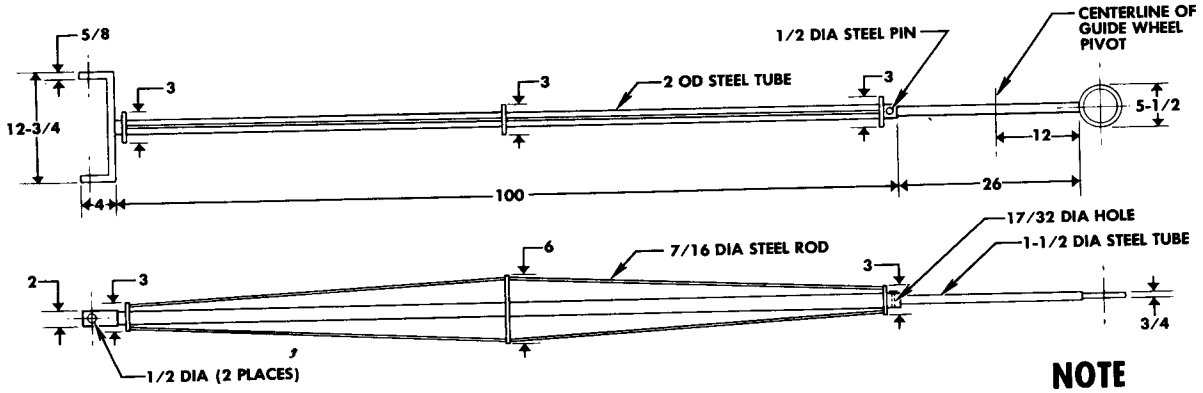
17

GEN-3-00-84

Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 6 of 8)

DAMAGED WHEEL TOWING DOLLY

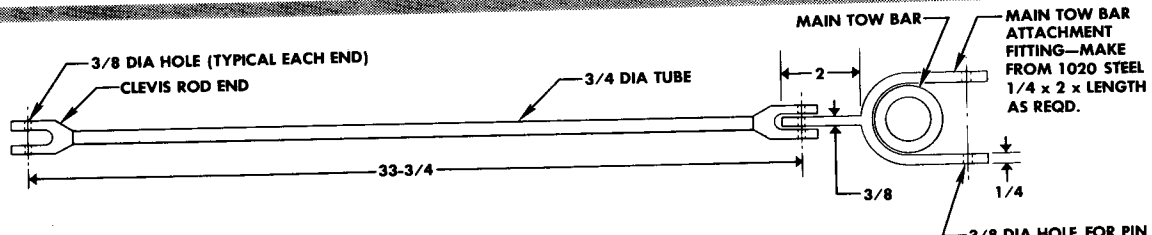
GENERAL



NOTE

All joints are to be continuous, welded both sides.

18 GUIDE WHEEL JOCKEY BAR



19 TOW BAR ASSEMBLY
1020 STEEL

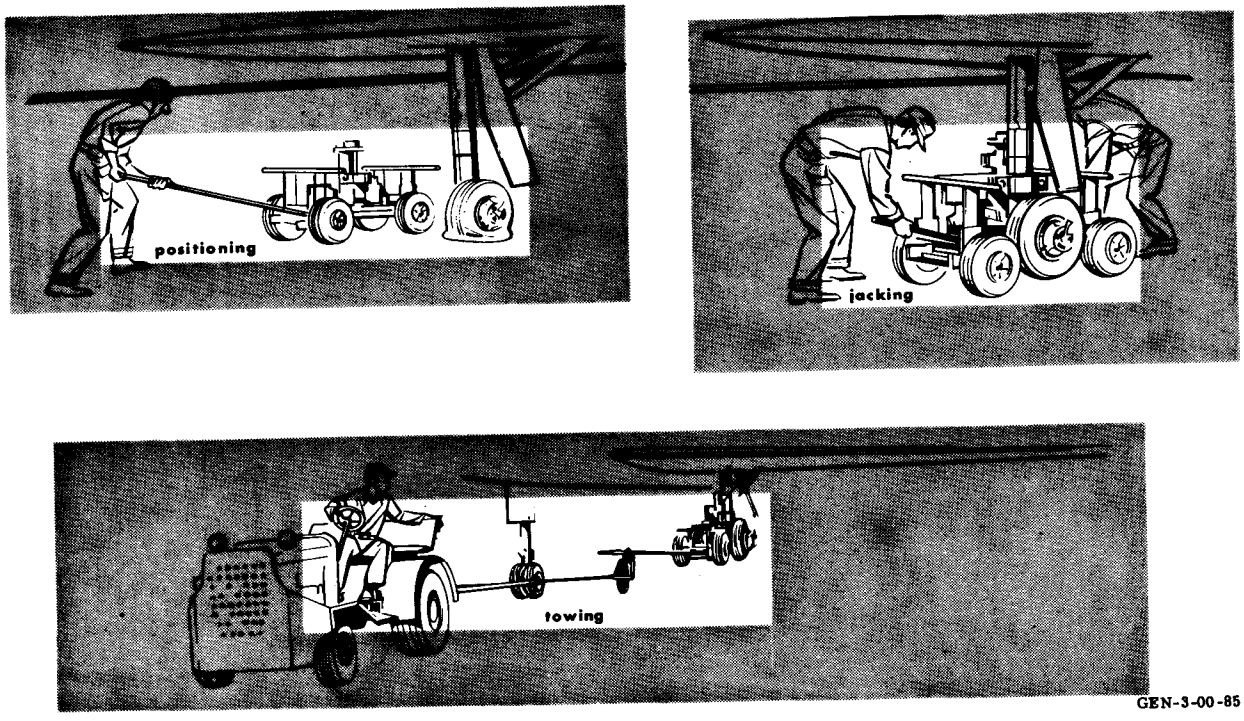


Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 7 of 8)

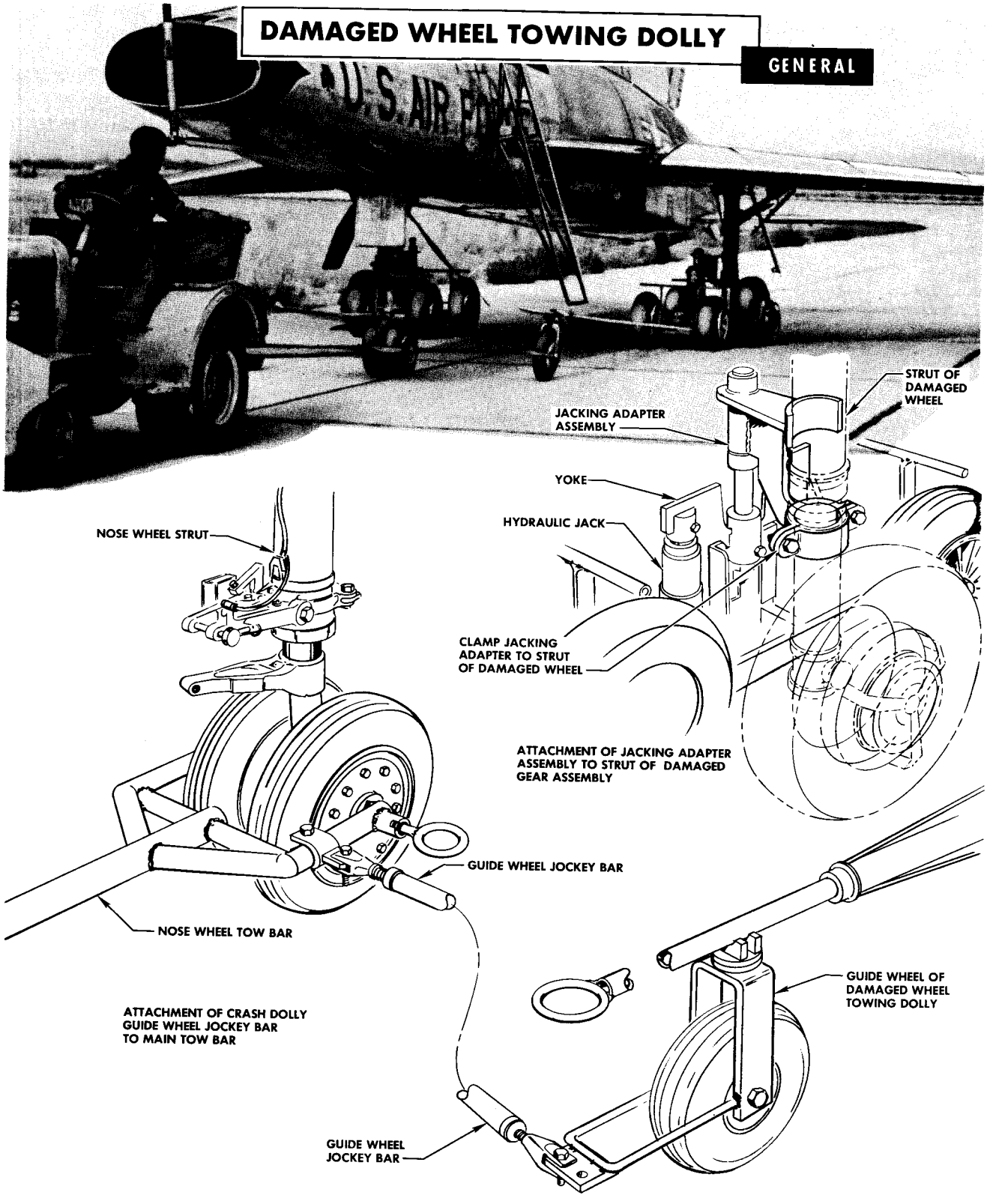


Figure 1-8B. Damaged Wheel Towing Dolly (Sheet 8 of 8)

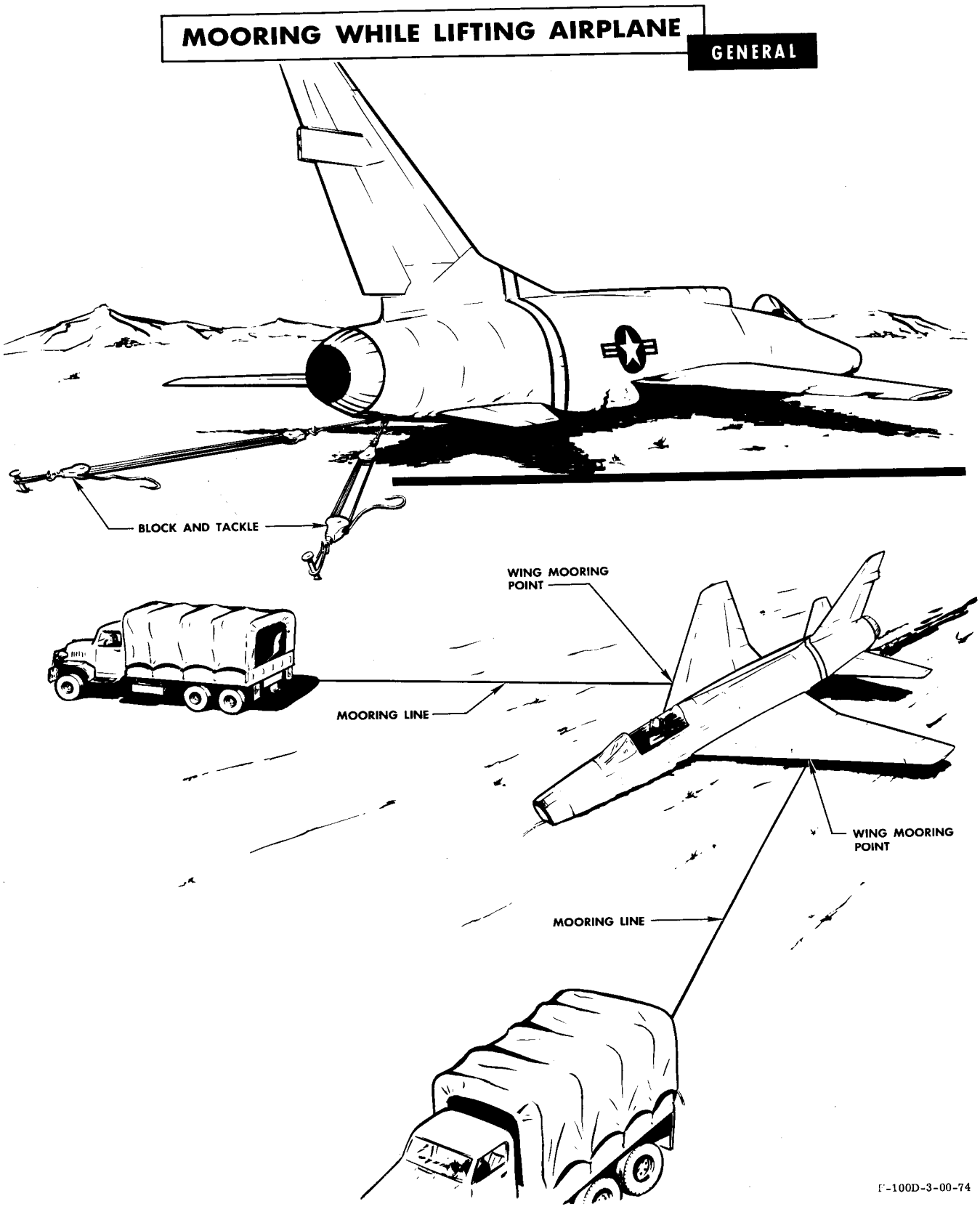


Figure 1-9. Lifting Airplane With Pneumatic Bags (Sheet 1 of 3)

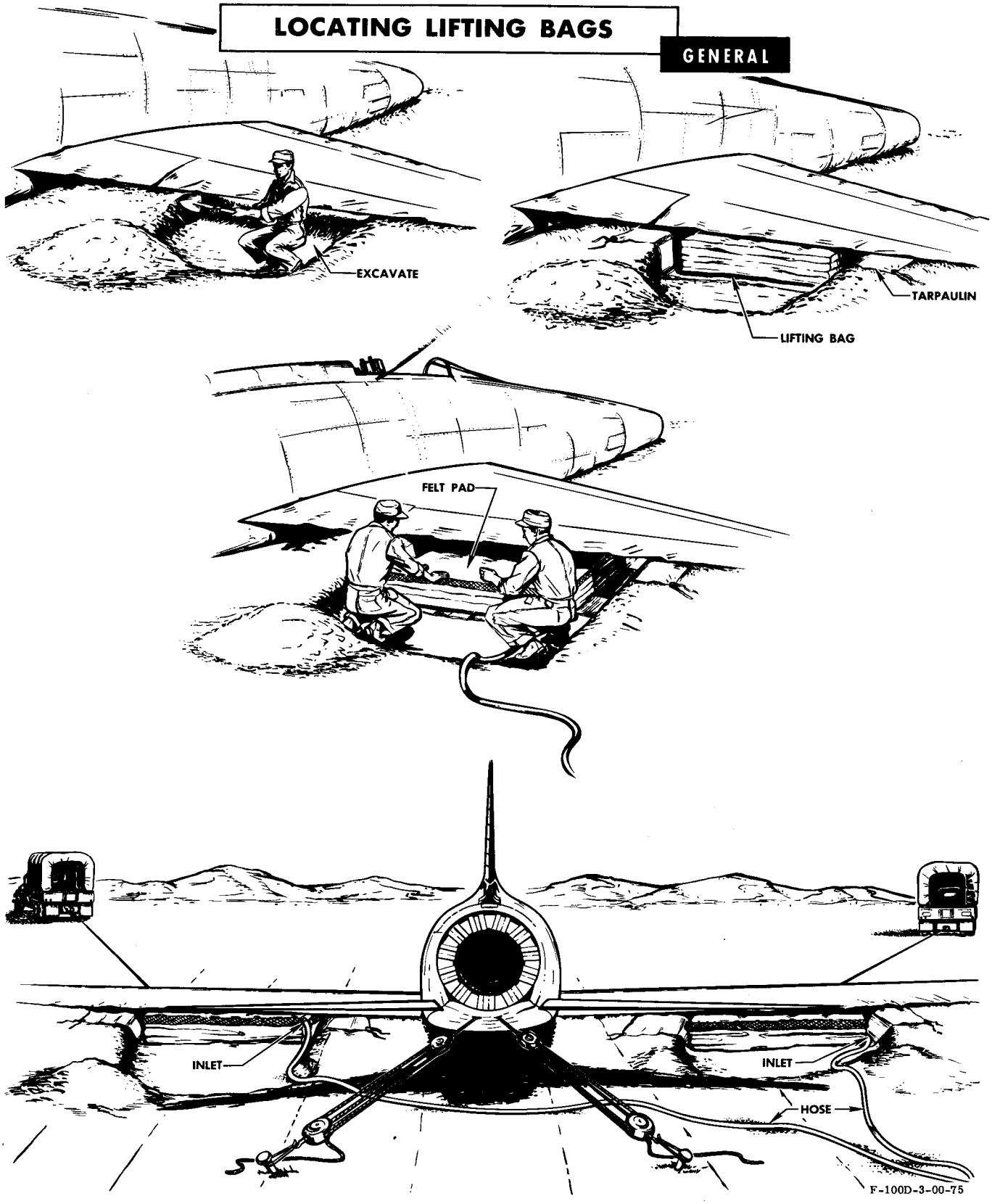


Figure 1-9. Lifting Airplane With Pneumatic Bags (Sheet 2 of 3)

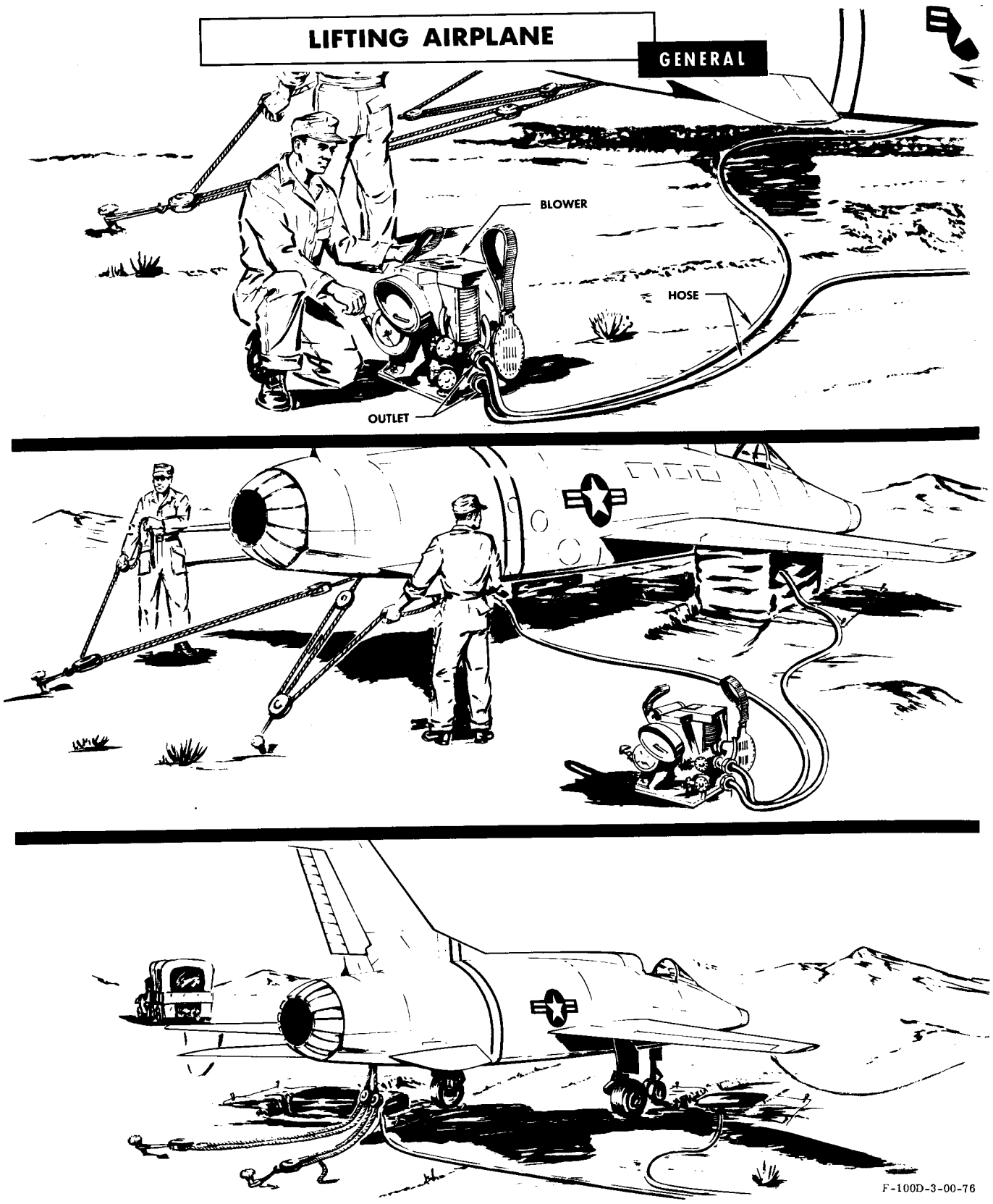


Figure 1-9. Lifting Airplane With Pneumatic Bags (Sheet 3 of 3)

Lines running forward must have enough slack to permit an upward motion of the airplane without tightening the lines. If available, block and tackle or chain falls should be used in the aft mooring lines. This provides a means of letting out the lines while the airplane is being lifted.

POSITIONING LIFTING BAGS.

In cases where the airplane is down in soft terrain, it may be necessary to dig out under the wings in order to get the bags in place. The tarpaulin in which the bag is wrapped is first spread out under the airplane and then the bag positioned upon it.

Bags must be positioned so as to allow as much surface as possible to come in contact with the wing, and also permit the gear to be dropped, or a dolly or truck to be rolled under the airplane after it is lifted. Before positioning the bag, check the underside of the wing for sharp or rough projections that might damage the bag. If any sharpness or roughness is found, the felt pad, provided in the lifting kit, should be placed between the bag and the wing to protect the surface of the bag.

LIFTING AIRPLANE.

To inflate the bags, the two outlet sleeves (stenciled with the word "OUTLET" directly below each sleeve) must be securely fastened. Two fabric straps with buckles are provided for securing each outlet.

Attach hoses securely to the inlet connection of the bag and the outlet connection of the blower. These connections are one-inch pipe-threaded fittings. The inlet connection of the bag has the word "INLET" stenciled directly above it.

CAUTION

Before starting blower, check mooring lines and be sure bags are located for greatest possible surface contact between bag and underside of wing.

Bags should be inflated together, unless the terrain is such that it would cause shifting of the airplane. As the bags are inflated, slack off on the mooring lines to permit the airplane to rise evenly.

CAUTION

Keep enough tension on the lines to prevent airplane from shifting.

When the airplane has reached sufficient height, it must be supported on its own gear, or by jacks or crash dollies. Do not try to support the airplane on bags for any period of time. The bags do not have adequate pressure-holding qualities to permit them to be used as permanent supports.

With airplane safely supported, bags may be deflated by opening outlet sleeves and collapsing the bags. (Refer to applicable Technical Order on pneumatic airplane-lifting bags for inspection and storage.)

REMOVING AIRPLANE FROM CRASH SITE.

When the airplane has been raised to sufficient height, an attempt should be made to lower the gear. If the gear is serviceable, the airplane may be towed from the crash site in the usual manner. (Refer to applicable Technical Order for towing procedures.) If the gear is damaged, it will be necessary to load the airplane on a truck or dolly.

To load the airplane on a truck, sand bags are used as cribbing. The sand bags are placed under the wing at the heavy structural section near the root. Care must be taken to place bags so that no airplane weight will be supported on the light structure of the wing leading or trailing edge. The airplane must be secured to the truck bed with rope or cable. Tie-down should be from the airplane mooring points if possible. If other securing points are used, care must be taken not to damage structure.

CAUTION

Wire, cable, chain, etc, must not be used where it will come in contact with airplane structure.

TRANSPORTING AIRPLANE.

A damaged airplane that is to be transported by air, rail, or truck should be packed and handled in the same manner as new equipment. Proper crating and padding is essential. It is recommended that when possible the airplane be shipped by air. Rail and truck transportation is satisfactory with proper loading procedures.

Note

All airplanes are shipped with engines and all loose equipment removed. Fuel tanks must be empty and purged. The airplane and its equipment will be preserved in accordance with current directives.

An airplane shipped by rail will be subjected to high fore-and-aft loads. Adequate cradling and tie-down are necessary. By breaking the airplane down into its major components (fuselage, wing outer panels, vertical stabilizer, and horizontal stabilizer), each section may be crated and loaded. The shipping cradles are designed with heavy bases so that the components may be stored in them until work is to be done. Any part of the cradle coming in contact with the airplane structure must have one inch of padding. Movable surfaces, such as ailerons or rudder, should be secured so that they cannot move. All supports are placed at heavy structural members. Each component is secured within its own cradle, so that it cannot shift or slide from the supporting points.

The fuselage presents the greatest packing and shipping problem because of its bulk and weight. Packing it in two separate units is the easiest method of handling. The supporting cradles should be well padded and fit snugly. If the bottom of the fuselage is damaged, it may be necessary to change the supports to fit the situation. Much of the fore-and-aft load of the fuselage forward section is carried by the fuselage rear section attach fittings. Care should be taken in the construction of the aft cradle support to ensure a good fit. Cables from the fore-and-aft jack-pad fittings carry some of the load. If the forward jack-pad fitting cannot be used, an alternate strong point, such as the nose gear trunnion, should be selected for tie-down. Securing the fuselage with steel banding or cable that is not tied into some fixed point on the fuselage is undesirable. Load shifting will cause the band or cable to cut or chafe the airplane structure.

The individually packed components may be arranged on the railroad car in any convenient manner. Any hoisting equipment capable of lifting the gross weights may be used. There are fork-lift slots on the base of each cradle. The cradles are secured to the car with bolts (⅝-inch minimum) through the base and the car floor. One bolt for about every 5 feet of base skid is required. Bolts are equally spaced along the length of the skids. The end bolts should be about one foot from the skid end. When the cradle consists of three base skids, only the two outside skids need to be bolted.

SHIPPING CRADLES.

See figures 1-10 through 1-10C.

STRUCTURAL ILLUSTRATION DOCUMENT NA56-1281 (F-100D) OR NA57-557 (F-100F).

The purpose of this document is to aid in the identification of damaged parts. The document will be issued through the prime AMA (Air Materiel Area) and will be used when replacement parts are to be ordered. Detailed instructions for use of the document are contained in Air Force Manual 67-1 (Volume I, Section 7, paragraph 15).

Figure 1-11 is the master perspective of the F-100D document, and figure 1-11A is of the F-100F document. It shows the general breakdown of the airplane. The airplane is broken down into its major production components. Several views of each component are shown so that the structure may be seen more clearly. Structural illustrations are not engineering drawings and should be used for reference only.

When an airplane is damaged, a structural illustration document will be issued. All pertinent airplane data will be entered in the space provided on the title page. Damaged parts will be shaded in with red pencil on the illustration which shows the damaged area. Any other information which would aid in the identification of parts should also be included. All additional information will become a part of the structural illustration document, and will be submitted with the documents to the supply source. Replacement parts will be identified and issued through regular supply channels.

Care must be taken when preparing the document to order parts. The replacement parts will be issued according to the information supplied. A complete report will be very helpful in the procurement of replacement parts.

FORCES ON AIRPLANE IN FLIGHT.

The five major forces working on an airplane in normal flight are:

- Lift—the force which supports the airplane in the air.
- Gravity—the downward pull of the earth, or the complete weight of the airplane.
- Thrust—the force which moves the airplane forward through the air.
- Drag—the resisting force which tends to hold the airplane back.
- Inertia—the characteristic of any physical body which tends to resist any change in speed or change in direction of motion from a straight line. (See figure 1-12.)

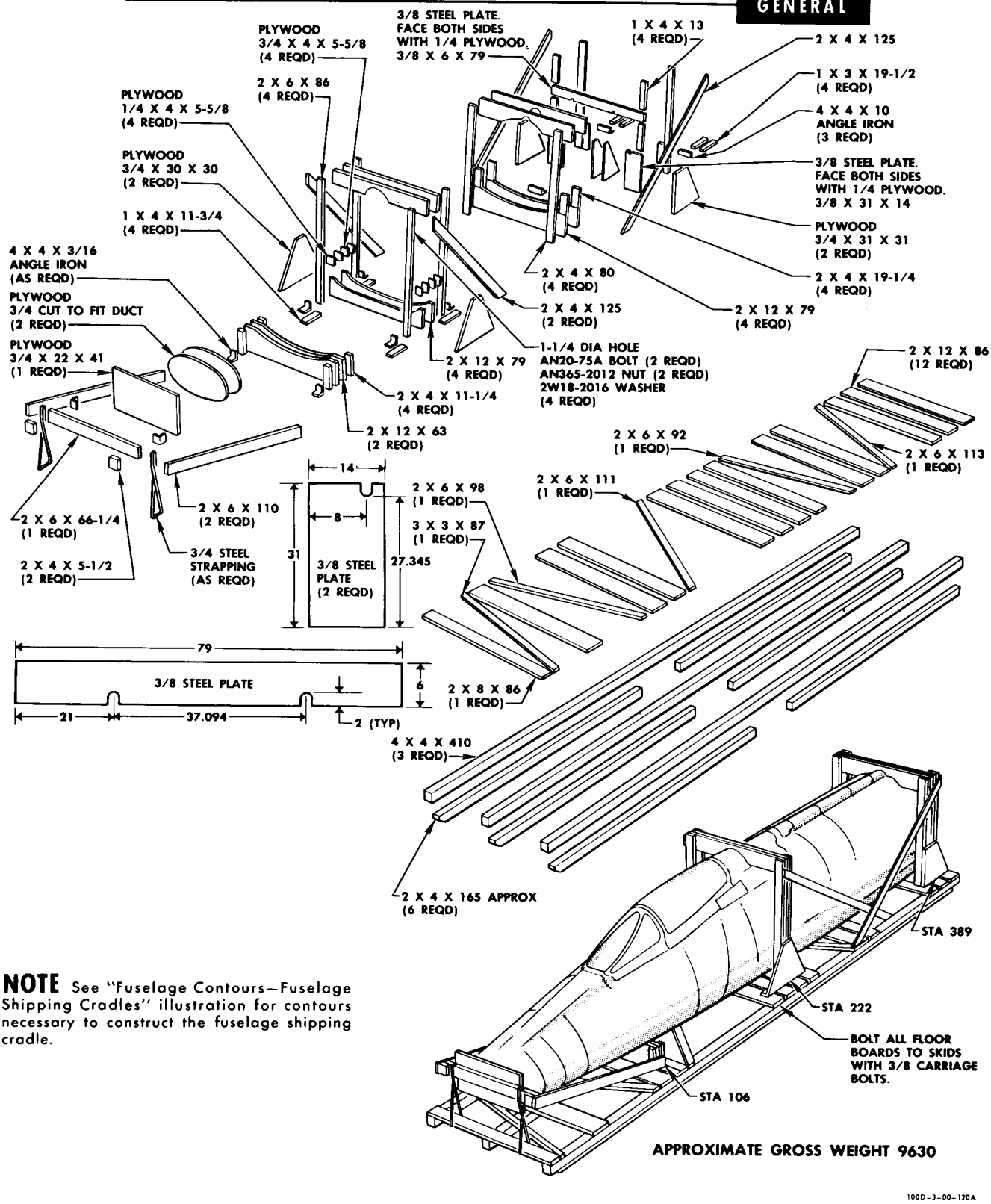
These forces act not only on the airplane as a whole, but may also be applied separately to individual parts of an airplane.

STRESSES AND STRAINS.

External forces acting on the airplane in flight set up stresses within the structure. Stresses are the intensity of the internal forces which are set up in a member to resist the external forces, and are measured in pounds per square inch. The intensity at any one point is the amount of the applied force acting at that point, divided by the cross-sectional area over

FORWARD FUSELAGE SHIPPING CRADLE

GENERAL

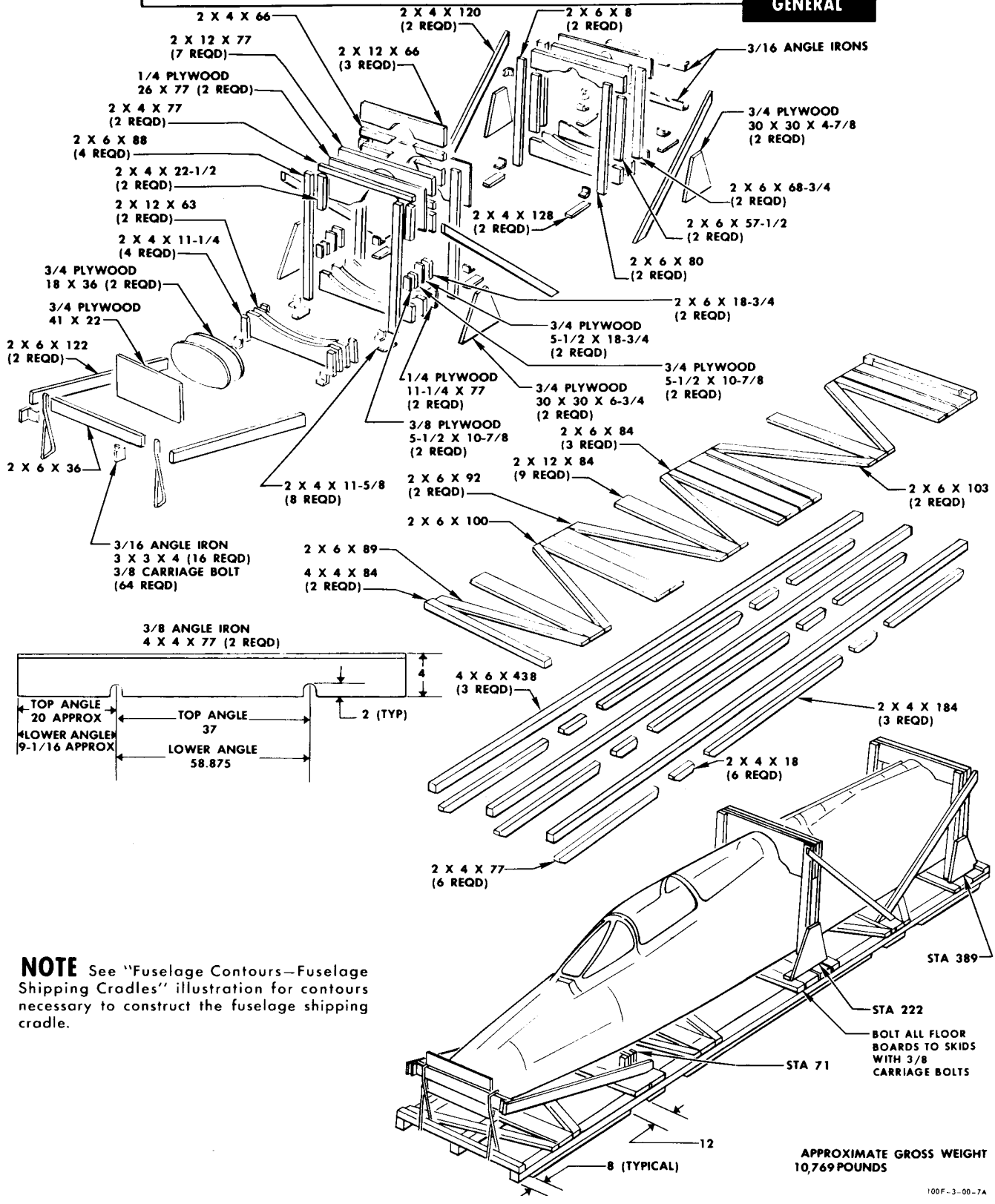


NOTE See "Fuselage Contours—Fuselage Shipping Cradles" illustration for contours necessary to construct the fuselage shipping cradle.

Figure 1-10. Shipping Cradle—Forward Fuselage—F-100D Airplanes

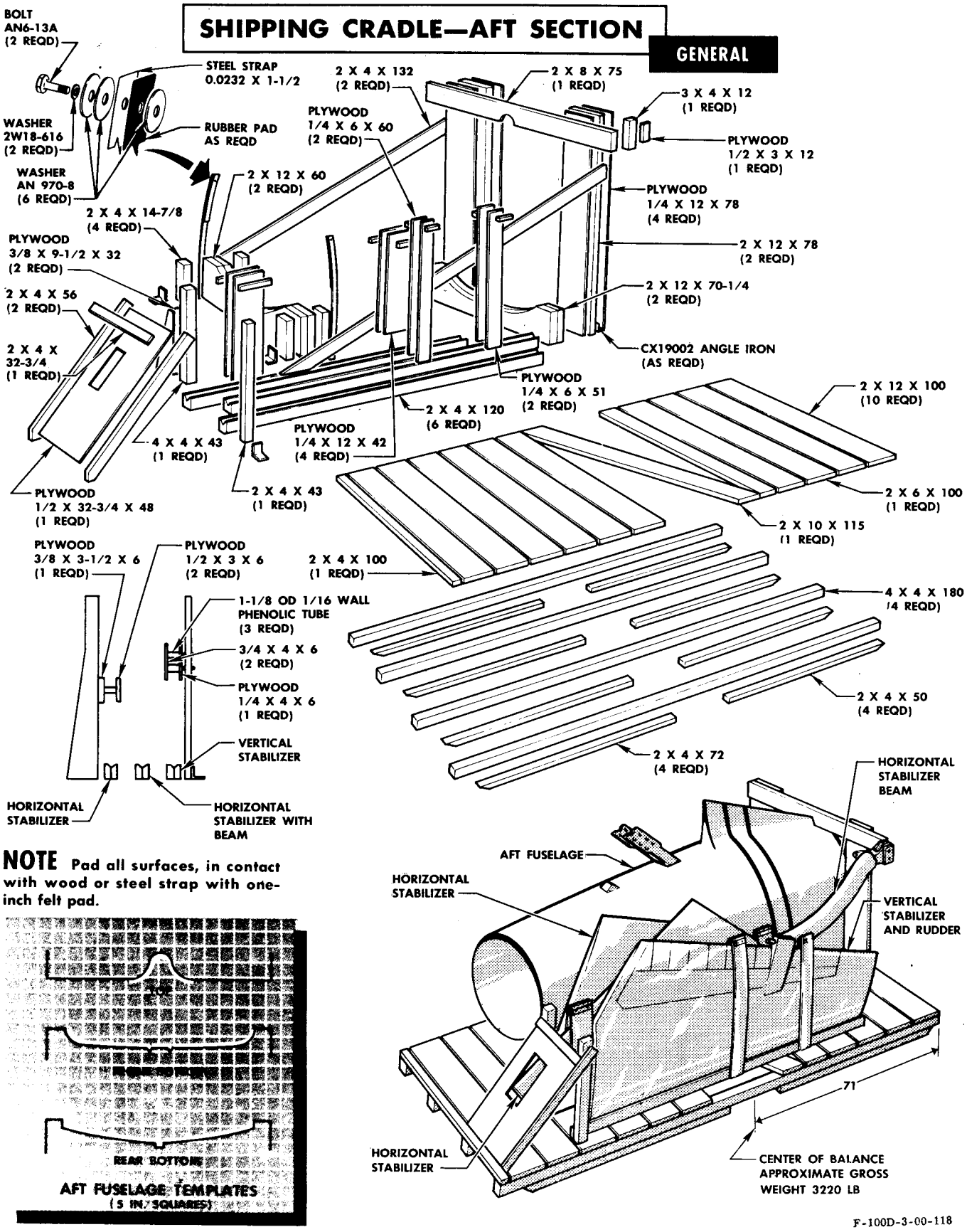
FORWARD FUSELAGE SHIPPING CRADLE

GENERAL



NOTE See "Fuselage Contours—Fuselage Shipping Cradles" illustration for contours necessary to construct the fuselage shipping cradle.

Figure 1-10A. Shipping Cradle—Forward Fuselage—F-100F Airplanes

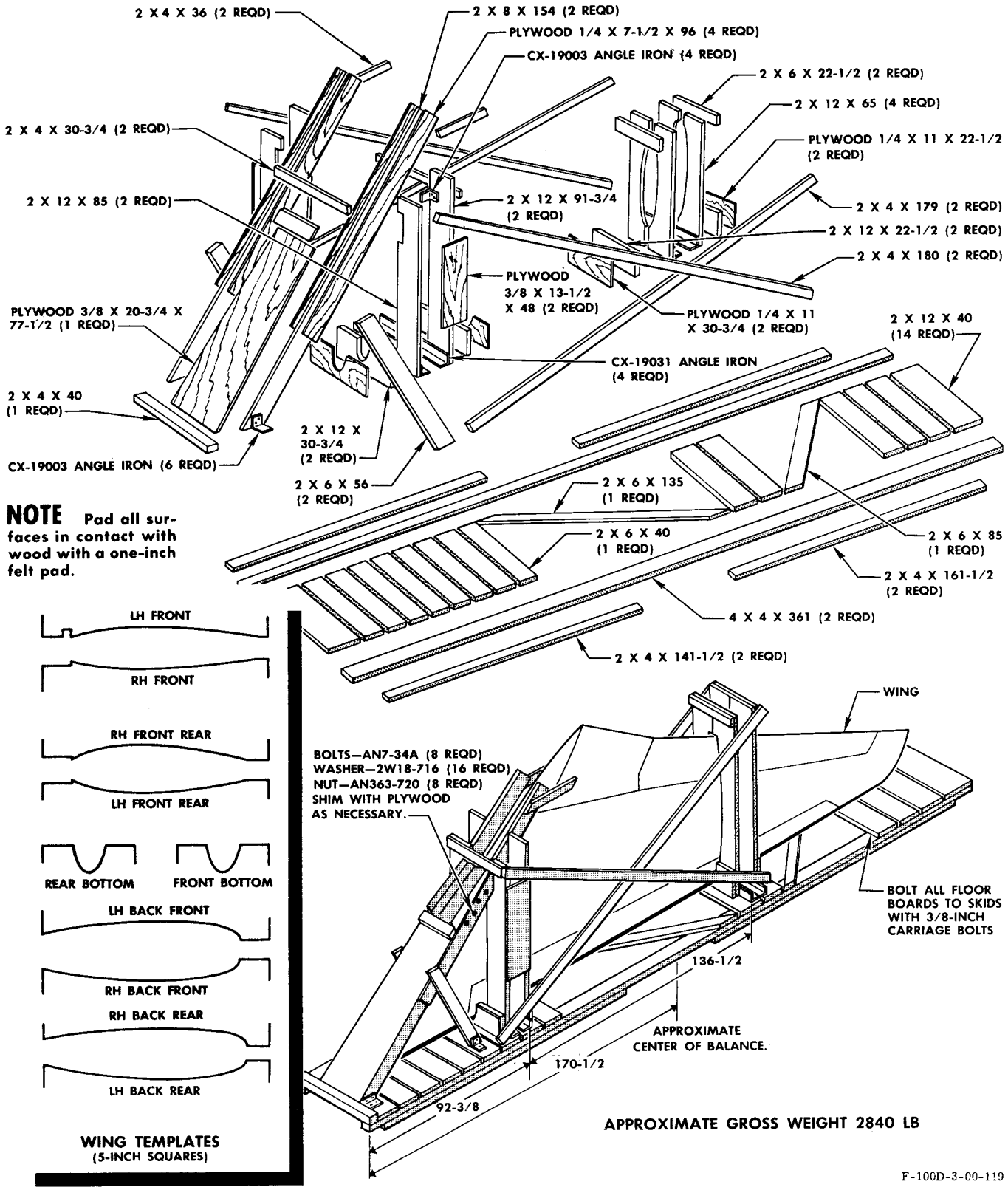


NOTE Pad all surfaces, in contact with wood or steel strap with one-inch felt pad.

Figure 1-10B. Shipping Cradle—Aft Section

SHIPPING CRADLE—WING

GENERAL



NOTE Pad all surfaces in contact with wood with a one-inch felt pad.

Figure 1-10C. Shipping Cradle—Wing

SHIPPING CRADLE-FUSELAGE

GENERAL

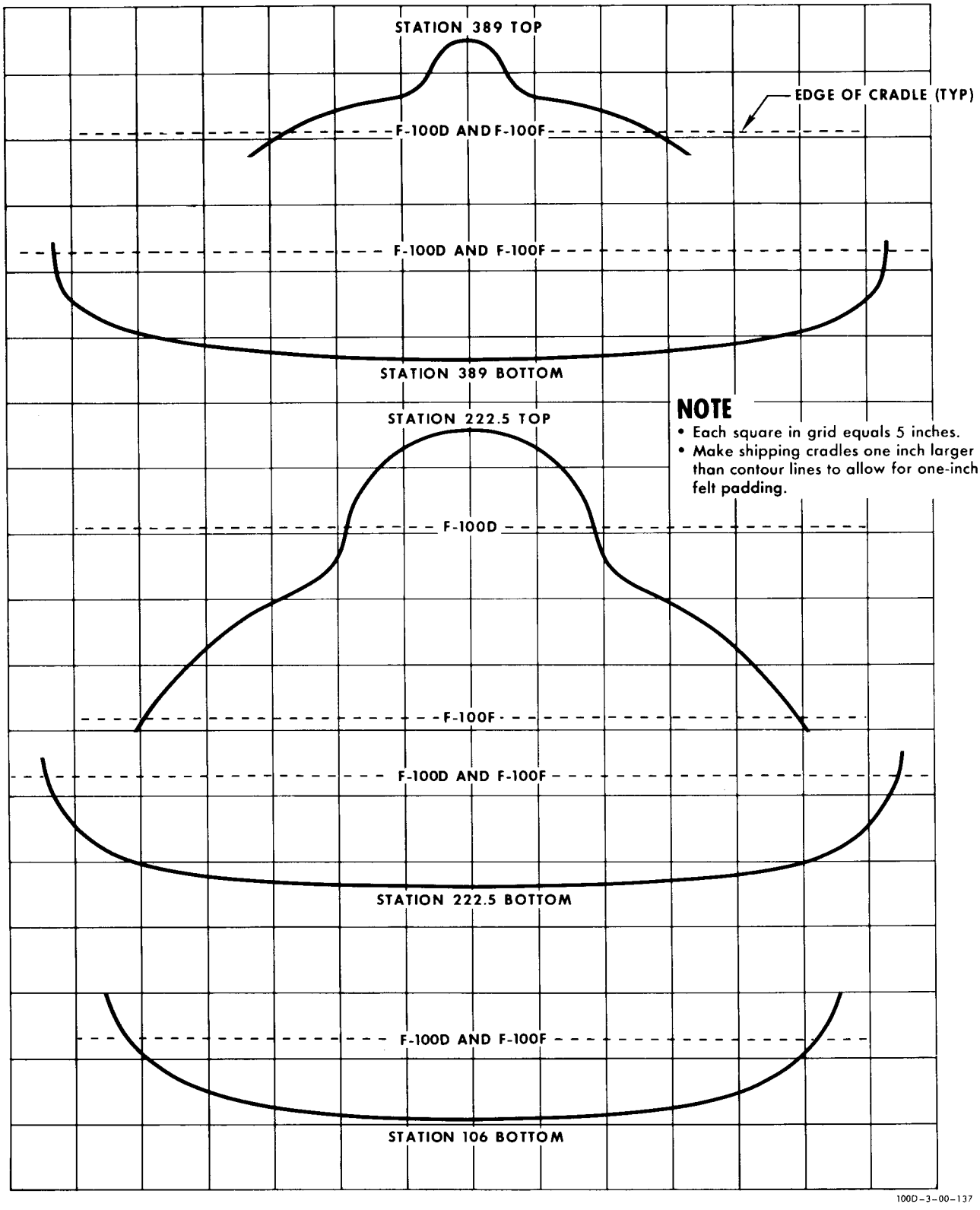


Figure 1-10C-1. Fuselage Contours—Fuselage Shipping Cradles

SHIPPING CRADLE-WING

GENERAL

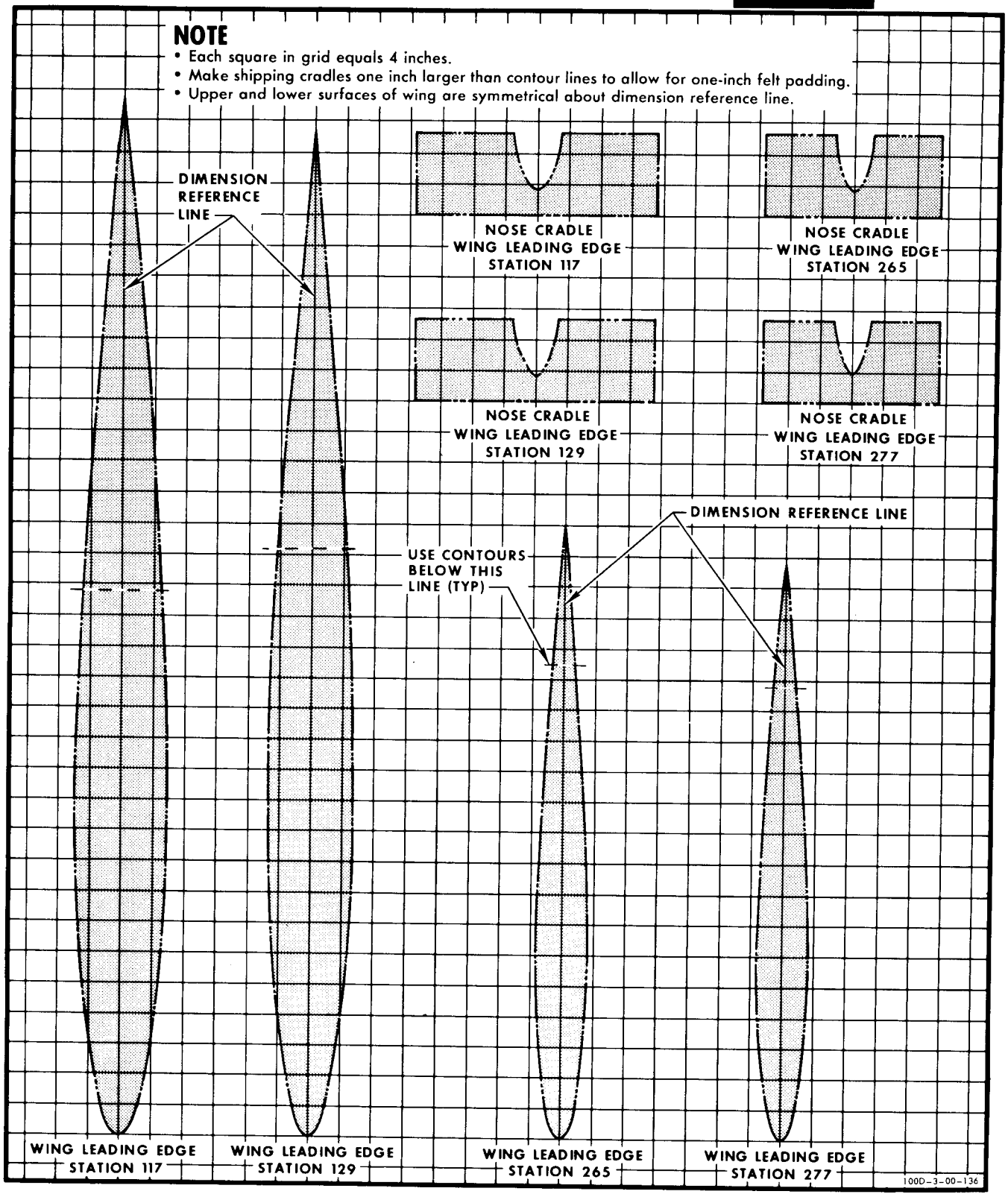


Figure 1-10C-2. Wing Contours—Wing Shipping Cradle

WEIGHTS AND CENTER OF BALANCE

GENERAL

NOTE The weights and center-of- balance figures are approximate, and may differ slightly on different airplanes of the same series.

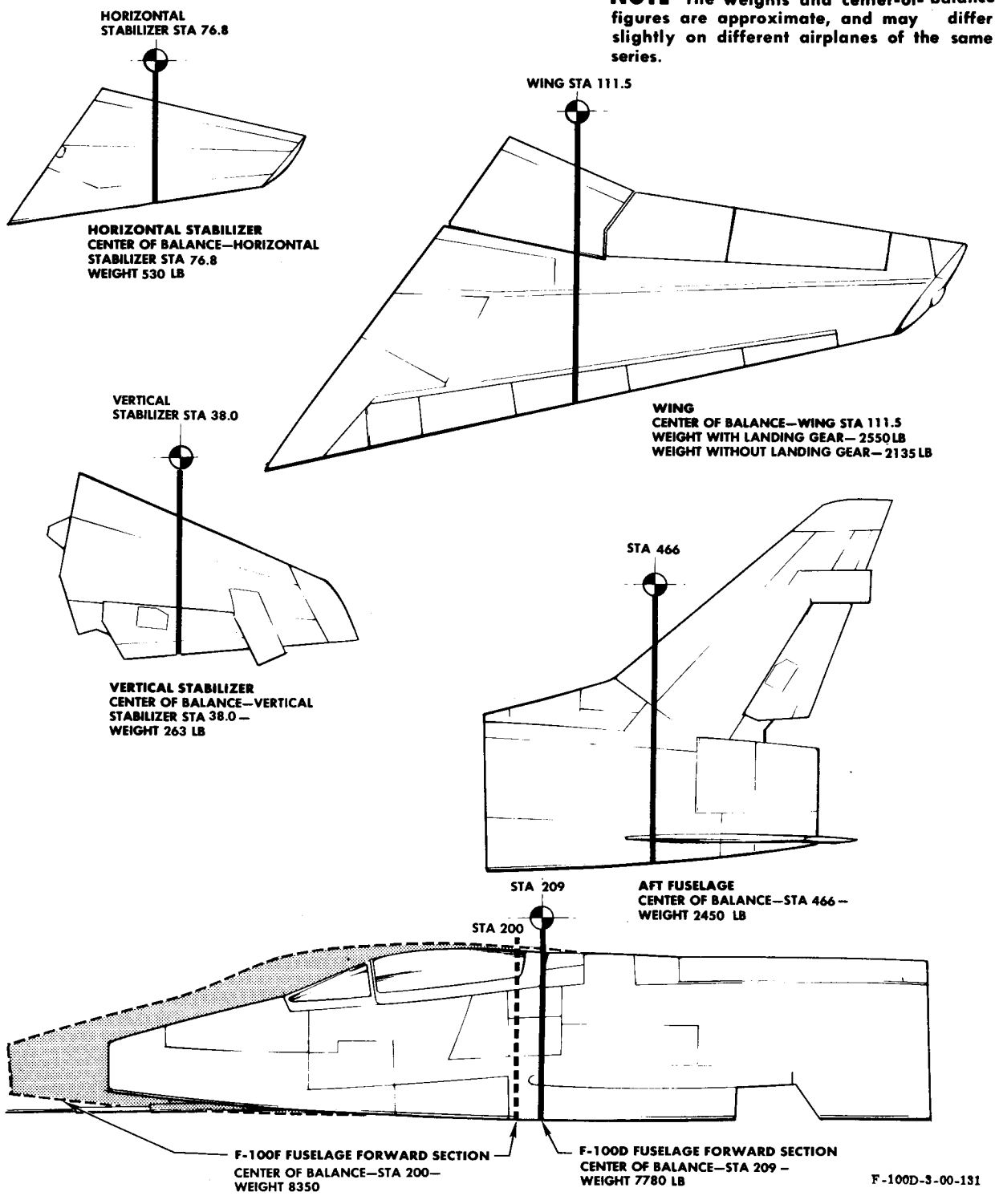


Figure 1-10D. Weights and Center of Balance of Airplane Components

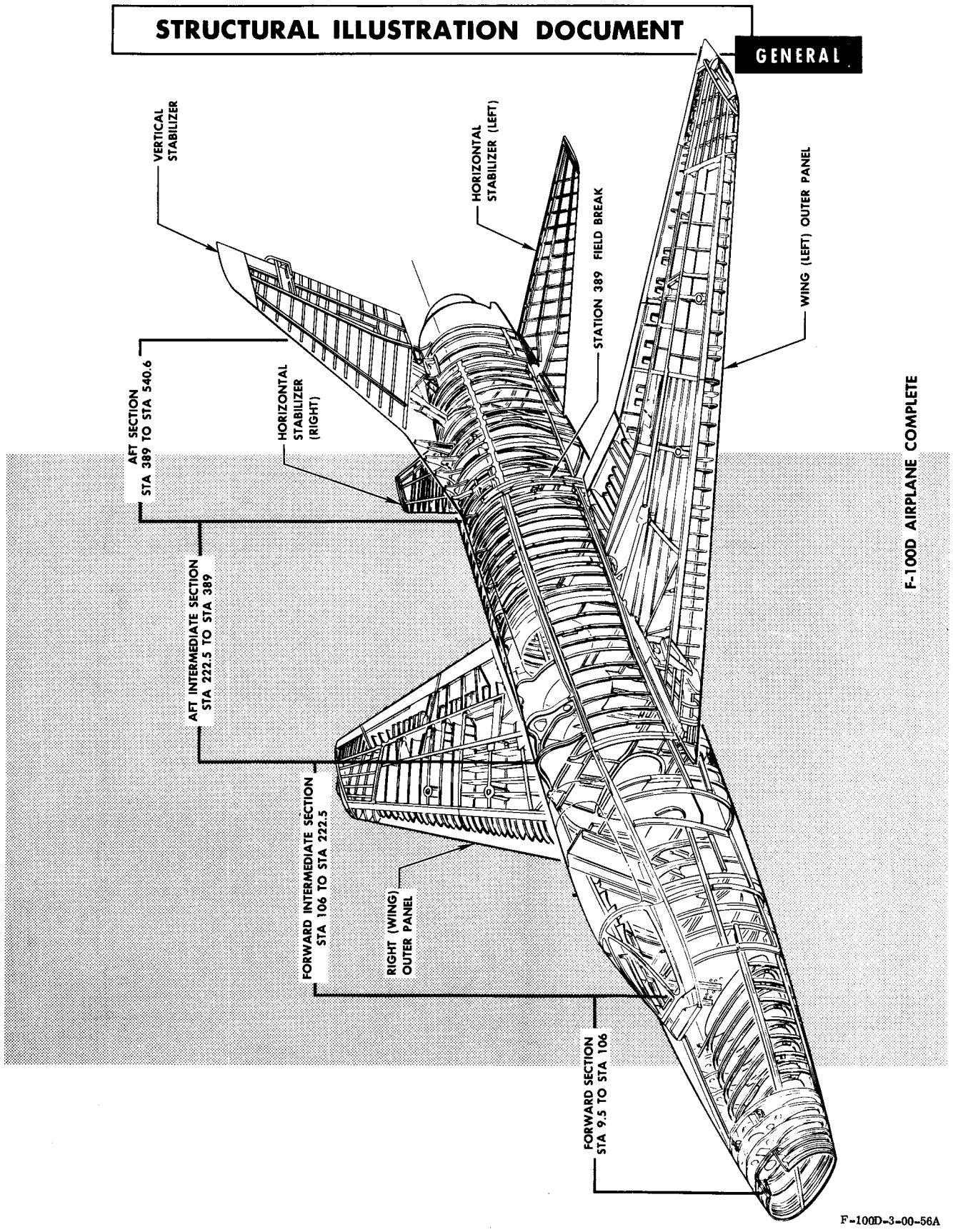


Figure 1-11. Structural Illustration Document—F-100D Airplanes

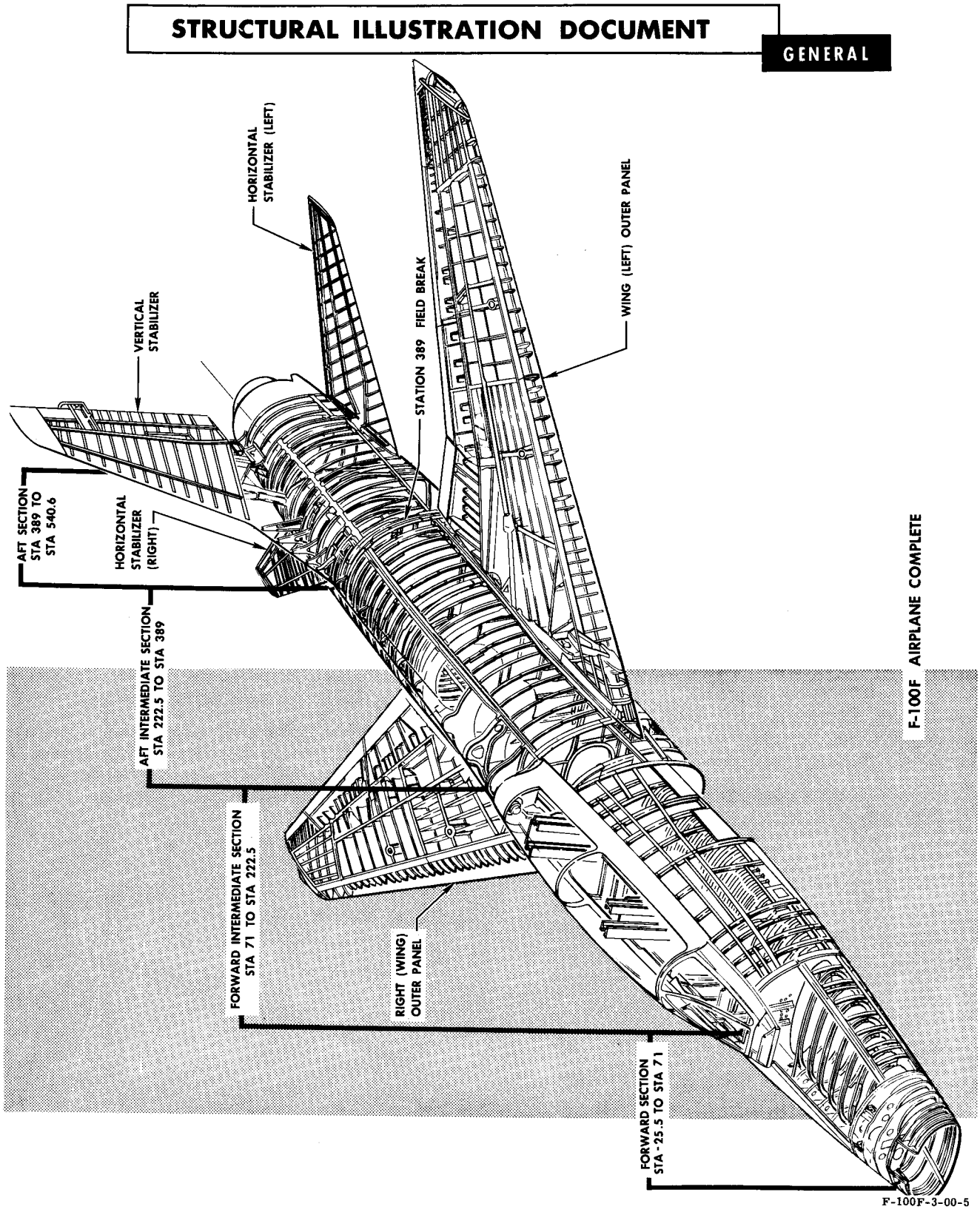


Figure 1-11A. Structural Illustration Document—F-100F Airplanes

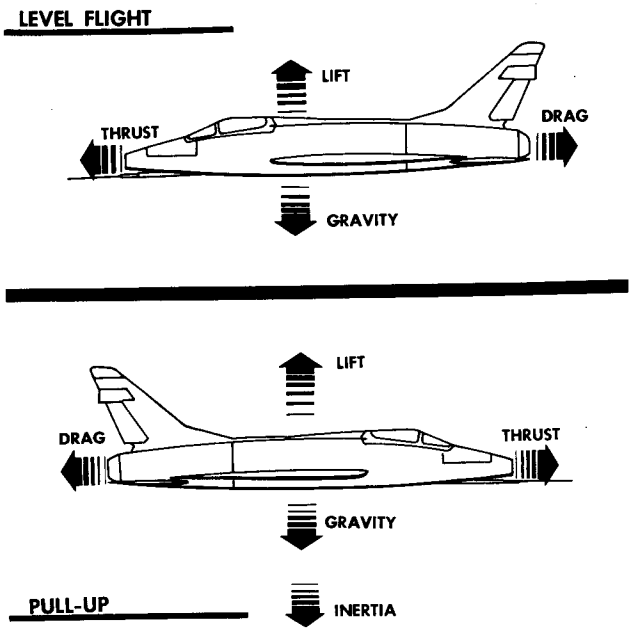


Figure 1-12. Forces on Airplane in Flight

which the force acts. Where the stress is uniform, the entire force may be divided by the entire cross sectional area.

$$\left(\text{Stress} = \frac{\text{Applied force}}{\text{Cross-sectional area}}\right)$$

Strain is the deformation of a member resulting from the stress within it. Strain is measured as a change in length, thickness, or movement per unit length.

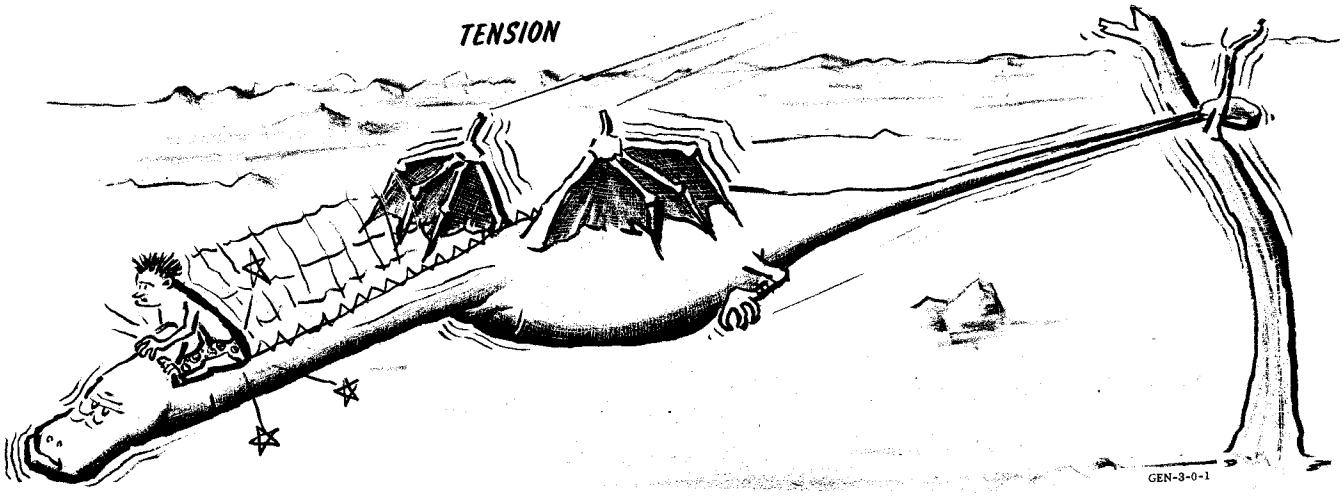
There are five types of stresses to be considered when making a repair; tension, compression, shear, torsion, and bending.

Tension, compression and shear are basic stresses, while torsion is a type of shear, and bending is a combination of tension and compression. Tension occurs when an applied force tends to stretch a member. Compression occurs when the force tends to shorten the member. Shear stress within a member results from a force tending to move or slide one portion of the member past an adjacent portion of the member. Torsion shear stress exists within a member when a torque or twisting moment acting upon it tends to turn or twist one portion of the member in relation to an adjacent portion in a parallel plane. Bending stress results when a force or bending moment acting upon a member tends to bend one portion of the member so that it is no longer parallel to the adjacent portion. The resistance to bending is a combination of tension and compression, although in most aircraft structures shear is also present at the same time as the bending.

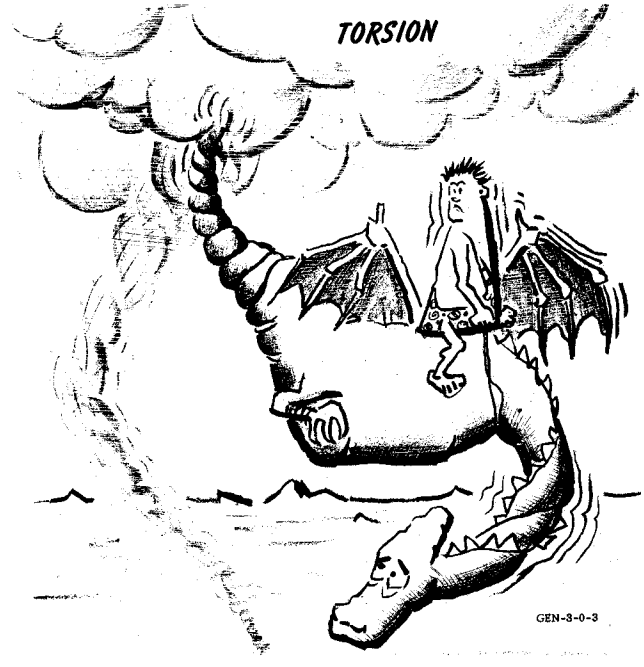
STRENGTH OF MEMBERS.

The strength of a member in tension is determined on the basis of its cross-sectional area and material only. Thus drilling a hole in the member decreases the cross-sectional area and reduces its ability to withstand tensile forces. To install a rivet or bolt in the hole will not materially affect the ability of the member to withstand tensile forces, since the rivet or bolt will not carry the tension loads across the hole.

The strength of a member in compression is determined on the basis of its cross-sectional area, material, and shape. Drilling a hole in a member reduces its ability to withstand compressive loads. However, if a plug,



such as a rivet or bolt of equivalent or stronger material is fitted tightly into the drilled hole, the rivet or bolt will carry compressive loads across the hole.



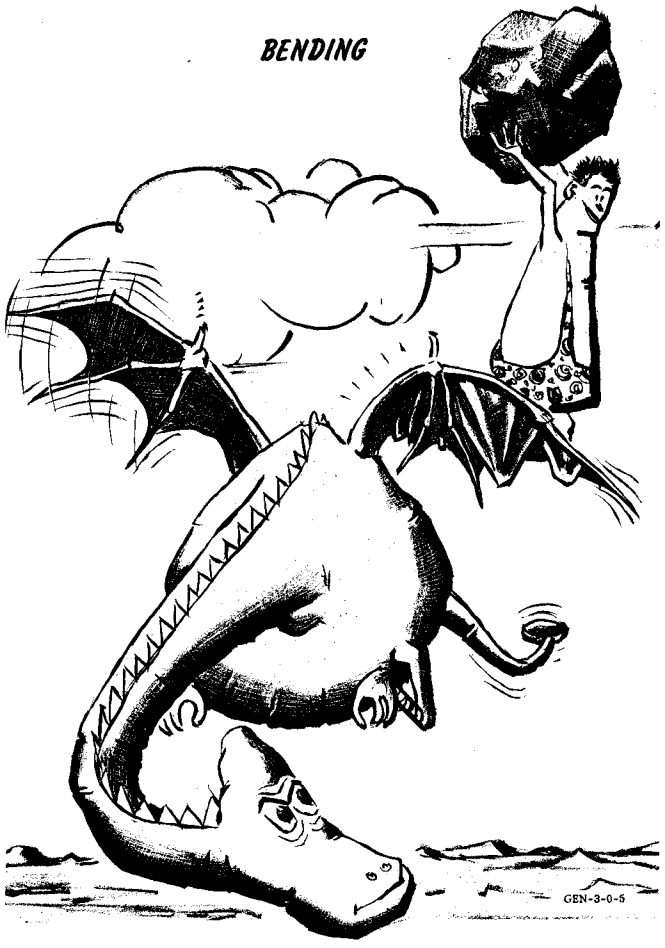
The member will now carry about the same compression loads as it would have, had the hole not been drilled.

The strength of a member in shear is determined primarily on the basis of its cross-sectional area and material. Shape has less effect on shear strength if the member is supported properly to eliminate bending or buckling.

The strength of a member in torsion is determined to a large degree by its shape as well as its cross-sectional area and material.

The strength of a member in bending is greatly dependent upon the shape of the member, the cross-sectional area, and the material. Resistance to bending involves a combination of stresses.

The shape, cross-sectional area, and material used in the structural members of airplanes are usually selected to produce as strong and as light a structure as is possible within aerodynamic and production requirements. Any member, when repaired, should be returned as nearly as possible to its original shape and strength. Any loss of strength because of a change in shape must be made up by an increase in cross-sectional area.



EFFECT OF STRAIN IN A MEMBER.

When a force is applied to a member, stresses are set up within the member. These stresses resist the force but result in a strain or deformation of the member. Normally, when the force is removed, the member will return to its original shape. This is because of the elasticity of the member material. Figure 1-13 shows how stress resulting in strain affects a material. Material loaded to less than the stress corresponding to point "A" on the figure will have a strain shown by line OA. When the stress is removed, the strain will follow this line back to point "O." If a material is stressed to a degree that the proportional or elastic limit is exceeded (point "A") when the stress is removed, the strain will diminish along a line parallel to OA starting at the highest point reached. The material will not return to its original shape because a certain amount of the strain or deformation will remain after the stress is removed. In this instance, it is said that a permanent set has been taken by the material. This does not necessarily mean failure. In most cases, the member will still be able to withstand reapplication of the same force without further yielding. A small predetermined amount of permanent set is expected and permissible.

The yield stress "B" is defined as that stress level, which when removed, will leave a permanent set equal to the maximum permissible set, point "D." Upon reloading, the strain will follow the line DB, if "B" is not exceeded. Any stress higher than "B" will follow the line BC up to the value of stress imposed and then return along a new line to a permanent set "E," which is greater than permitted.

If the stress is allowed to reach point "C," the part will break.

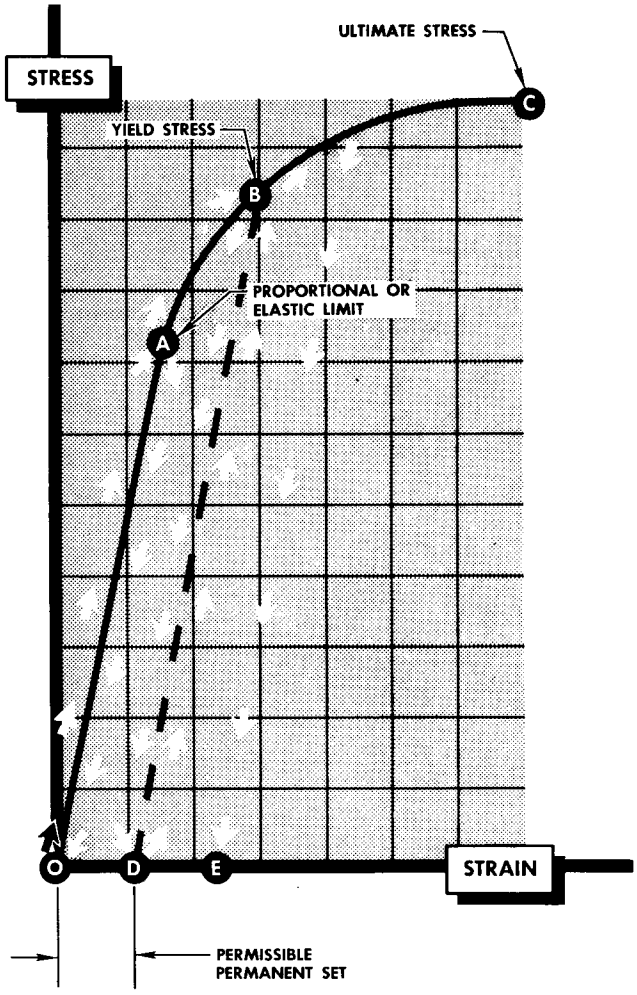


Figure 1-13. Stress-Strain Curve

LOCATION OF DAMAGE IN A MEMBER.

When a force which causes a member to bend is applied, (figure 1-14), the bending stress causes a shortening or compressive stress to act upon one half of the beam and a stretching or tensile stress to act on the other half. These stresses act opposite one another on the two sides of the centerline (neutral axis) of the beam. Shear stresses occur along the member; they are greatest at the neutral axis and diminish to zero at the edges of a member. When there are bending stresses within a member, these stresses are greatest in the parts of the member most distant from its center. For this reason, ribs, spars etc, are built with the greater part of their cross-sectional area at the outer edges. These reinforced edges are called flanges or caps. Because of the concentration of stresses within the flanges or caps, they can stand only a very limited amount of damage. Care should be exercised when choosing rivet sizes and locating holes in these flanges or caps. Holes drilled by the manufacturer are generally a reliable guide to use.

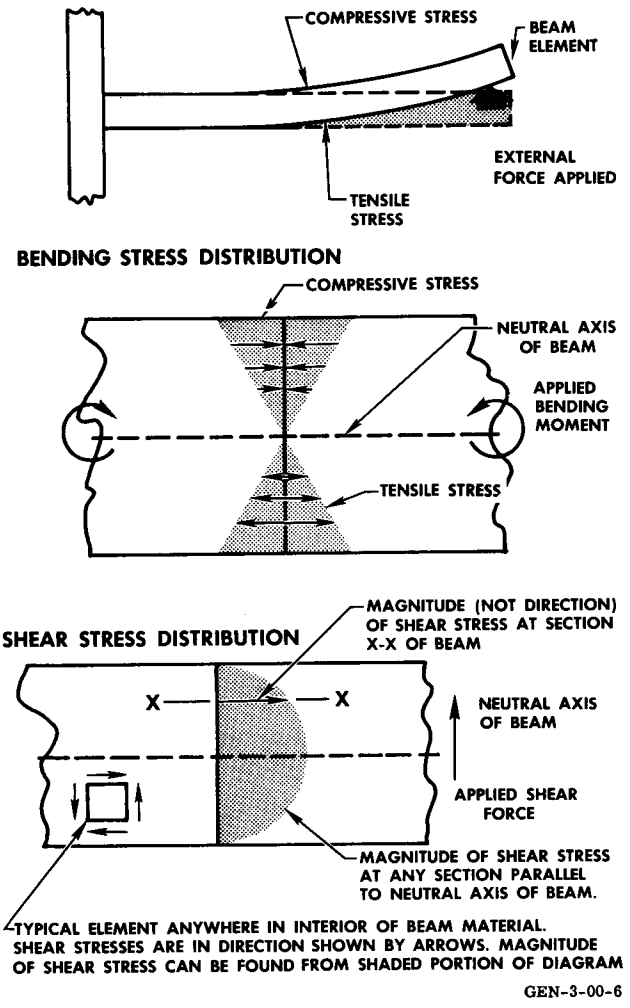


Figure 1-14. Simple Beam Elements

INVESTIGATING DAMAGE.

The primary structure of an airplane is designed to resist variable forces imposed upon it in operation. The structure must distribute these forces to the principle structural strength members of the wing and fuselage. When a member has been damaged, a complete and thorough inspection must be made. This inspection should include the damaged structural member and the adjacent or attaching members. It is possible that the damaging force might have been transmitted over the surrounding structure causing secondary damage in areas other than the primary damage area. Should the secondary damage not be discovered, forces distributed through the structure during normal airplane operation may cause this structure to fail. Careful investigation must be made of all structure across which the forces must travel, and parts must be checked for possible misalignment.

PRELIMINARY EXTERNAL INSPECTION.

A great deal may be learned about the condition of the internal structure of an airplane by a careful examination of the skin on the wings, stabilizers, and fuselage. Some indications might be buckle patterns on surface skin of stabilizers and along the sides, top and bottom of fuselage. Loose, tipped, or sheared fasteners, and joggles or bulges in the skin at structural joints or where heavy fittings are attached, indicate that a thorough inspection of the underneath structure should be made. Inspect fairings and other non-structural parts for tears, cracks, buckles, and misalignment. Any unusual appearance of these parts is reason for a close inspection of the structures they cover.

Check also the fit of doors, covers, canopy, etc. Anything unusual about their fit or alignment may indicate a warped or sprung structure, and will be cause for inspection of that structure.

When possible, actuate control surfaces, flaps, speed brake, etc. Observe these units closely throughout their travel. They should operate freely and without binding or chafing. In the zero or closed positions, check the units for proper alignment with adjacent members.

DETAILED INSPECTION.

Make a careful visual inspection of all areas adjacent to the obvious damage. Inspect forgings and other machined parts for cracks and for evidence of having been moved. This can often be done by observing the paint around the edges, by tipped bolt or rivet heads, or by a slight space at the edge of faying surfaces. When possible, remove two bolts from a suspected fitting. Examine bolts for evidence of shearing or bending. Check bolt holes for elongation.

Inspect supporting structures for buckling, warpage, and joggles. Check for sheared, loose, or tipped fasteners. Alignment with adjacent members is also important. Using a straight edge, check known flat areas such as spar webs, frame webs, and bulkheads. Any deviation should be thoroughly investigated.

When an airplane has been damaged, or damage is suspected, thoroughly examine by means of penetrant, radiography, or magnetic particle all areas where major components are joined, or where heavy weights are supported. The principal areas of this nature, with their supporting structures, are as follows:

- Wing center section to outer wing panel
- Wing to fuselage attachment points
- Fuselage rear section attachment point
- Horizontal stabilizer attachment point

- Vertical stabilizer attachment point
- Rudder hinge
- Aileron and rudder balance weight attachment point
- Landing gear trunnion & braces
- Engine mount

Continue the inspection throughout the disassembly and repair of the airplane. During this time, indications of unsuspected damage are often revealed. When new parts being installed do not fit properly, find out why. It may be because of damage which has been overlooked.

After movable members have been installed, move them through their full range of travel as soon as possible. Check closely for freedom of movement, proper clearances and alignment. It is often helpful to install a control surface or other mechanism on a temporary basis while the structural repair is in progress. It will aid in the alignment of parts being repaired or replaced and may also reveal unnoticed damage.

SKIN BUCKLING.

The mere presence of wrinkled or buckled skin is not positive evidence of an overloaded structure. It is not uncommon to find permanently buckled skin on an airplane which has never been overloaded. These buckles may appear on new airplanes because of manufacturing tolerances, or on others merely as evidence of age. In many examples of present day design, the vital importance of saving weight has led to types of structure in which the skin is designed to buckle, even to a large degree, under normal loading within the established flight limits of the airplane. Under perfect circumstances, these buckles disappear each time the loads are removed. It is therefore important to remember that buckled structure is not necessarily weakened structure. However, it is also important to remember that, although buckles themselves do not generally detract from the strength of the structure, their presence may be an external indication of trouble elsewhere. Unfortunately, since no hard and fast rules can be laid down, it must be left to experience and judgement to determine when an internal inspection should be made. For example, the sudden appearance of buckles after any one flight, where there were no buckles before, justifies an examination. Buckles that cross rows of rivets should be investigated. These are often a sign of tension failure in the rivets straddling the buckle, or a crippling failure in the flange of the internal member. Unusually severe buckles or sharp creases in the skin often lead to cracks after repeated flexing and, therefore, should be investigated.

Figure 1-15 shows certain typical skin buckling patterns and a few of their common causes.

Figure 1-16 shows the resultant inertia loading when certain external loads are applied to various points on the airplanes. The initial loads shown are typical of those frequently occurring in airplane accidents or in high-G pullouts. Initial load, or point of contact, is represented by a broken arrow. The directions and magnitudes of the resultant inertia loadings are indicated by solid arrows within shaded areas.

When the load pattern charts are used as a guide in determining which areas should be more carefully inspected, it must be remembered that each illustration represents but one given load. In any given accident, there may be secondary loadings which would result in compound load patterns. An example of this might be when an airplane makes a hard two-wheel landing, runs off the runway during runout, and is spun around by contact with another object or because one wheel sinks into a soft surface.

The load pattern charts show only a few of the possible loading conditions. However, they do show how an external load applied to one area is transferred throughout the airframe. A careful analysis of the type of accident or unusual maneuver, and the probable resultant loadings, is of great importance in a thorough inspection.

DAMAGE RESULTING FROM EXCESSIVE LOADS IN FLIGHT OR LANDING.

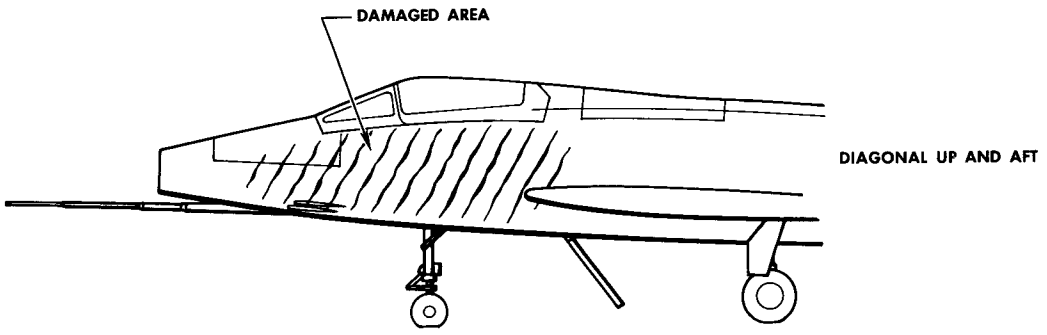
An airplane is designed with respect to a great many conditions which it is expected to encounter in its service life. It is therefore able to withstand the loads resulting from a great variety of actual flight and landing conditions. When an airplane is subjected to an extremely hard landing or over-G flight maneuvers, the structure may have loads imposed upon it which exceed the design limitations. It is not practical to say that an airplane will be damaged everytime it suffers a hard landing or is subjected to an extreme flight maneuver. It is not possible to indicate specific parts of an airplane which may be visibly damaged everytime an airplane exceeds its design limitations. It is essential, however, that an inspection be made for damage which may have occurred.

In general, permanent deformation of an airplane structure is significant primarily on those parts carrying compression loads or where looseness of joints result. Permanent wrinkles in shear-carrying panels such as wing skins, fuselage skins, bulkheads, beam webs, etc, are evidence that there may be more serious damage in the surrounding structure. Overall permanent deformation of the airplane structure (misalignment of major components such as a twisted or

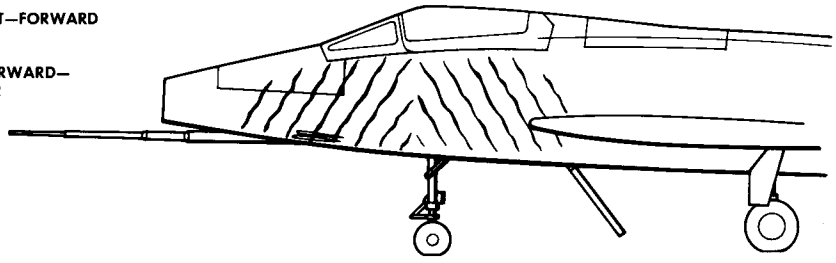
SKIN BUCKLES

GENERAL

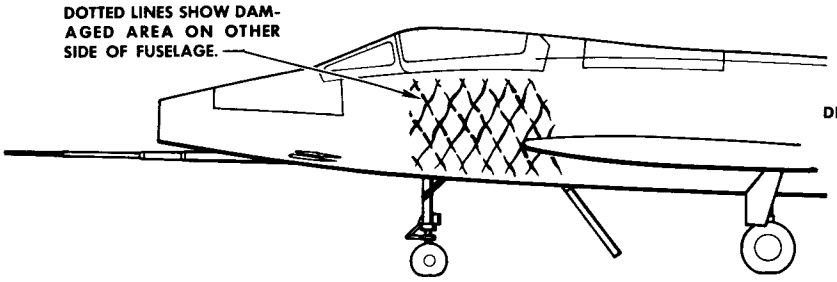
FORWARD FUSELAGE



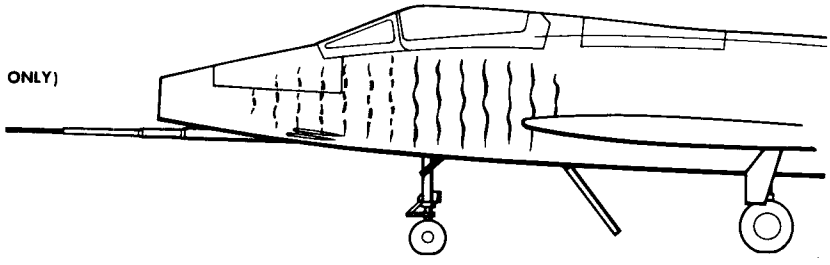
1 HIGH-G PULL-OUT OR HARD NOSE-UP LANDING



2 HARD THREE-WHEEL LANDING OR NOSE-WHEEL-FIRST LANDING



3 LANDING WITH HIGH SIDE DRIFT, IMPROPER TAXIING, OR TOWING WITH TOO HIGH NOSE GEAR SIDE LOAD.



4 LANDING WITH HIGH SIDE DRIFT, IMPROPER TAXIING, OR TOWING WITH TOO HIGH NOSE GEAR SIDE LOAD.

F-100D-3-00-83B

Figure 1-15. Skin Buckling Patterns (Sheet 1 of 2)

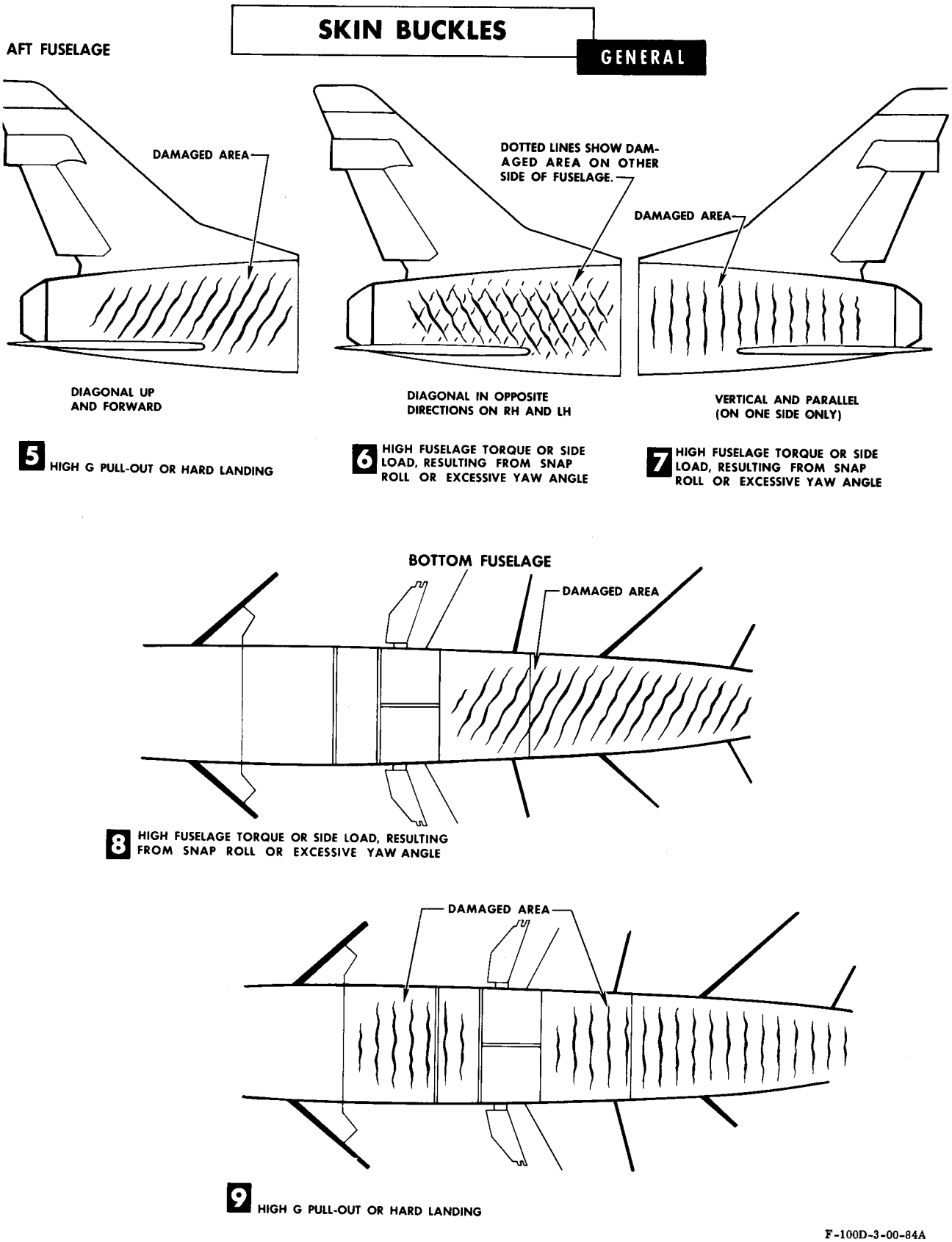


Figure 1-15. Skin Buckling Patterns (Sheet 2 of 2)

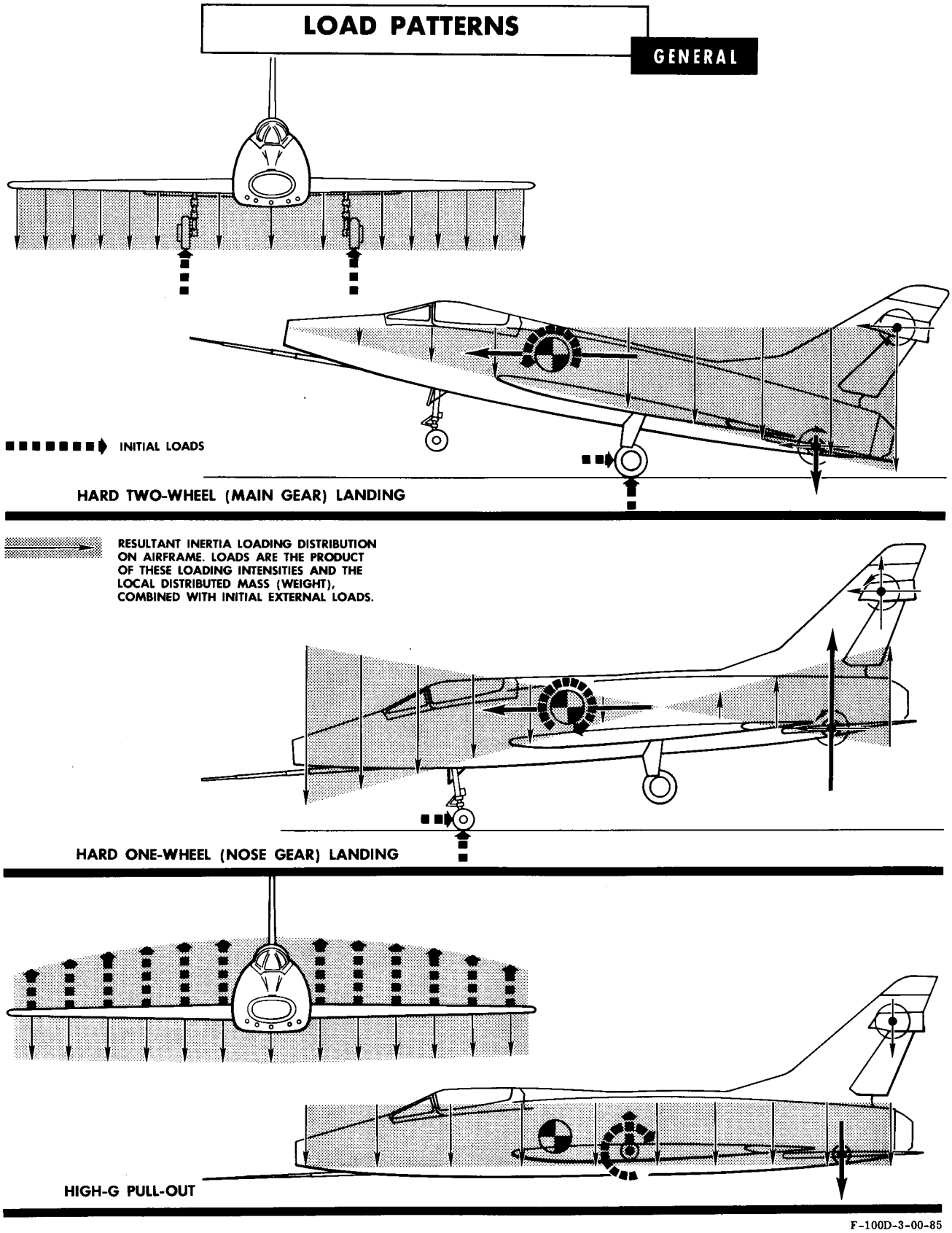
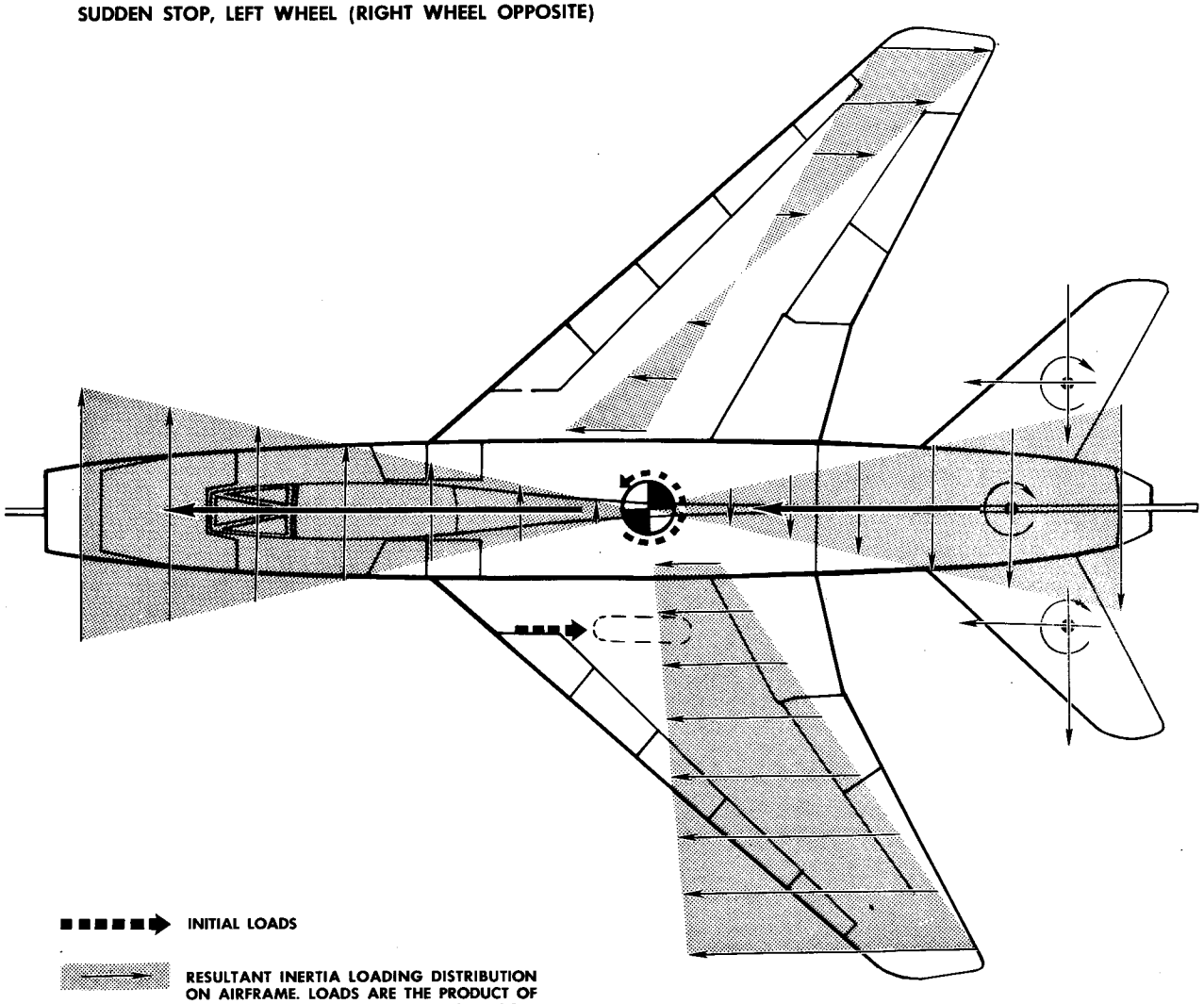


Figure 1-16. Load Patterns (Sheet 1 of 3)

LOAD PATTERNS

GENERAL

SUDDEN STOP, LEFT WHEEL (RIGHT WHEEL OPPOSITE)



INITIAL LOADS

RESULTANT INERTIA LOADING DISTRIBUTION ON AIRFRAME. LOADS ARE THE PRODUCT OF THESE LOADING INTENSITIES AND THE LOCAL DISTRIBUTED MASS (WEIGHT), COMBINED WITH THE INITIAL EXTERNAL LOADS.

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Figure 1-16. Load Patterns (Sheet 2 of 3)

LOAD PATTERNS

GENERAL

LEFT WING HITS OBJECT (RIGHT WING OPPOSITE)

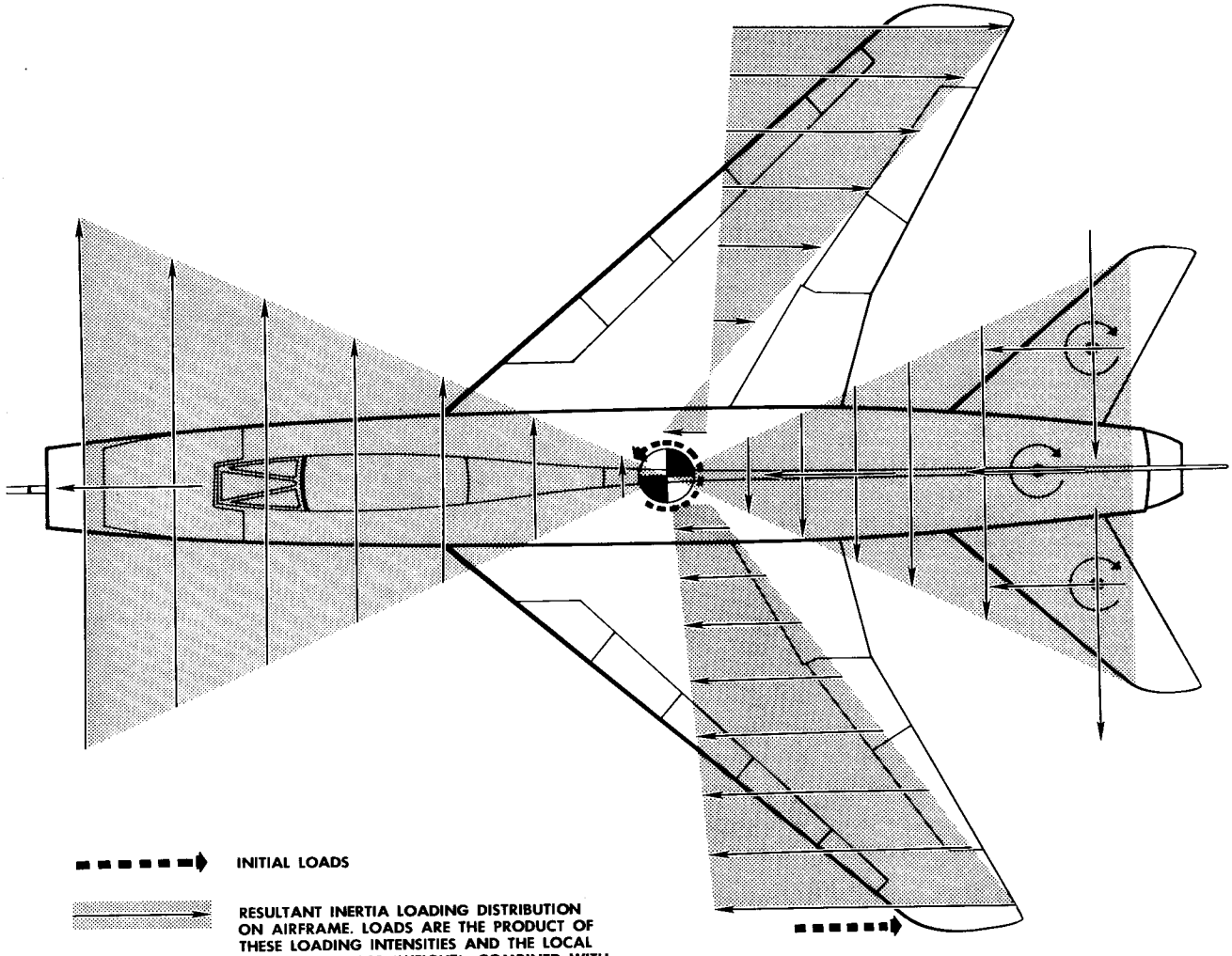

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Figure 1-16. Load Patterns (Sheet 3 of 3)

bent wing, tail, or main fuselage) usually does not mean the structure is unsound unless the deformation is so great that flight characteristics are affected. Compression buckles (sharp waves or kinks in stringers, longerons, frame or rib flanges, landing gear struts, control rods, etc,) are structural failures. These failures may not immediately prevent the structure from carrying part of its design load. However, such failures may become progressively worse and eventually cause complete failure of the structure. Long out-of-contour waves in the wing or tail surfaces, in any sheet stringer structure, or in longerons may indicate failure due to excessive loads or damage to supporting structure. This, too, may speed up total failure under additional high loading.

Excessive loads may cause enough permanent deformation in joints to cause the joints to loosen up. This is most serious in rotating joints. Loose joints which are subject to fluctuating loads or rapid application of intermittent loading may wear rapidly. In a short time the available load carrying material surrounding the joint may be reduced to such an extent that failure will occur at low-level loads.

Permanent structural deformation is rarely so serious that it reduces structural strength enough to prevent operation under restricted conditions. The real danger is the probability of rapid progressive deterioration under repeated high loading, especially if the deformation is in the form of compression buckles and waves or loose joints.

The most serious reduction in strength of a structure which has been overloaded is due to actual fracture or partial fracture of components. Inspection of such structure should be concentrated on finding fractures or partial fractures appearing as cracked skin, sheared or partially sheared rivets, cracked or broken lugs, fractured threaded parts, cracked or broken spars or longerons, etc.

EXCESSIVE LOAD FACTOR CRITICAL AREAS.

When an airplane has been subjected to excessive loads, particular areas of the structure are more critical than others and should receive special attention.

Blocked "particular attention" areas of figures 1-17 through 1-19A must be inspected by means of X-ray, penetrant, magnetic particle, or visual methods. Hidden areas should be inspected by X-ray equipment. Steel bolts should be Magnafluxed. Accessible boltholes should be penetrant-inspected. Skins require only visual inspection except where noted. Other accessible structures should be penetrant-inspected.

WING DAMAGE RESULTING FROM EXCESSIVE LOADS IN FLIGHT OR LANDING.

When an airplane has been subjected to excessive load factors, the wings may suffer extensive damage. The wing tip should be inspected for sheared spar fasteners, buckled covers, and indications of failures at skin splices. The main wing skins must be carefully checked for evidence of column failure, tension failure and cracks. Internally, the trunnion rib, which not only supports the main gear but also absorbs wing box bending loads, may fail. The fuselage attach, or root rib, attaches the wing to the fuselage. Over-G loads could shear the rib web and fail the caps in compression or tension. The components in the wing trailing edge which should be checked are the flap support structure, flap hinge fittings, and aileron hinge fittings. Most of the leading edge air load is applied on slats.

The upper and lower skin of the slats should be checked for buckles. (See figure 1-17.) Check slat in closed position with respect to wing contour. If the slat is out of contour, inspect the attachment of slat to slat track to determine whether the penetrating type serrations have slipped. If the serrations have slipped, the serrated bushings and all parts which the serrations penetrate should be replaced. If it is determined that the serrations have not slipped, examine the slat tracks for permanent deformation. This may be done by measuring the depth of the slot in the slat tracks. The variation in slot depth for any particular track should not exceed 0.015 inch along the length of the slot.

If the serrations have not slipped, and the slot in the track is not out of tolerance, the fittings which attach the slat to the tracks should be inspected. Particular attention should be given to the fasteners that attach these fittings to the slat ribs. Because slat loading is high, the support forgings and the hinge joint which attaches them to the main wing structure are critical.

Tank ejection causes critical shock loading on the wing. If this happens while the wing is being subjected to over-G loads, shear failure or cracks might occur in the spar cap. Upper skin buckling or lower skin tension failure or cracks could also result. Failure or cracks in the ribs that support center posts for stores are possible.

Because the aileron control hydraulic system is provided with relief valves which should protect it from being overstressed, it requires only a general inspection.

The wing flaps are also affected when the airplane is subjected to overload. The most likely places for significant damage on the flaps are at the hinges and the flap actuating linkage. When inspecting these points, the holes in the hinges, actuating links, and attachment points should be carefully checked for elongation. The wing flap should also be inspected for evidence of compression buckles, especially in the skin land at the main spar.

The areas and suggested methods of inspection are as follows:

- Visually inspect wing tip box for buckled skin and sheared spar fasteners.
- Penetrant-inspect wing tip box skin splice attach point for cracks.
- Visually inspect outer panel box skins for buckles, tears, and fuel leaks.
- X-ray-inspect outer wing panel box skins and internal structure for cracks.
- X-ray-inspect trunnion rib for cracks or other evidence of failure.
- Visually inspect center section box for skin buckles and damaged attach fitting holes.
- Penetrant-inspect center section box front and rear spars for cracks.
- Penetrant-inspect wing root rib for cracks on web and cap.
- Magnaflux-inspect wing attach fitting bolts for cracks.
- Penetrant-inspect wing attach fitting boltholes for cracks.
- Penetrant-inspect wing front and rear spar webs shown in blocked areas of figure 1-17 for cracks.
- Visually inspect leading edge slats shown in blocked areas of figure 1-17 for loose attach fittings.
- Magnaflux-inspect bolts and penetrant-inspect slat boltholes and slat tracks on leading edge slats shown in blocked areas of figure 1-17 for cracks.

EMPENNAGE DAMAGE RESULTING FROM EXCESSIVE LOADS IN FLIGHT OR LANDING.

When the empennage has been subjected to loads in excess of those for which it was designed, the main structural boxes are the parts most likely to be damaged. (See figure 1-18.) The bolts that attach the vertical stabilizer in the fuselage and those that attach the horizontal stabilizer panels to the horizontal stabilizer center section beam should be inspected for looseness. This may be done by visually inspecting the bolts while applying loads at the tip, parallel to the chord line and at right angles to the surface. If the bolts appear loose in this visual inspection, they should be removed and the holes should be measured accurately. If holes are elongated they may be reworked as outlined in figures 3-18 and 3-32.

The areas and suggested methods of inspection are as follows:

- Visually inspect vertical stabilizer skins for buckles, fractures, and loose fasteners.
- Magnaflux-inspect and penetrant-inspect boltholes of vertical stabilizer fuselage attach fittings for cracks.
- Visually inspect horizontal stabilizer skins for buckles, fractures, and loose fasteners.
- Magnaflux-inspect bolts and penetrant-inspect boltholes of horizontal stabilizer fuselage fittings for cracks.
- Magnaflux-inspect bolts and penetrant-inspect boltholes for cracks of rudder attach fittings to vertical fin.

FUSELAGE DAMAGE RESULTING FROM EXCESSIVE LOADS IN FLIGHT OR LANDING.

When the fuselage is inspected, particular attention should be paid to the riveted splice and splice fittings. (See figure 1-19, view "A" for F-100D Airplanes or figure 1-19A, view "A" for F-100F Airplanes.) The most critical area in the forward intermediate section for side bending conditions is along the station 222.5 frame. (See figures 1-19, view "B" or 1-19A, view "B.") Skin riveting at the forward edge of this frame is especially critical. Examination should be made for tilted fasteners and skin tears. View "C" in figures 1-19 or 1-19A shows the most critical point in the aft intermediate section. Skin should be checked for buckles and rivets checked for indications of failure. The structural doors in this section must also be inspected for damage.

Check for damaged fasteners and possible cracks in the corners of cutouts in the basic skin. View "D" in figures 1-19 or 1-19A shows the location of the field break fittings (fuselage rear section attachment fittings). These fittings and their attaching structure are critical items and should be thoroughly inspected. An indication of overload would be loss of torque value in the disconnect nut and in the studs. If all or most of the torque is lost, the bolts should be removed and checked, and the holes should be checked for elongation. The titanium side panel skins in the aft section should be inspected for cracks originating from, or between, dimpled rivets.

The areas and suggested methods of inspection are as follows:

- Visually inspect fuselage skins and doors for buckles, fractures, and loose and failed fasteners.
- Magnaflux-inspect bolts and penetrant-inspect boltholes of fuselage attach points of views "B," "C," and "D" on figures 1-19 and 1-19A for cracks.
- X-ray-inspect fuselage skin splice of view "A" on figures 1-19 and 1-19A for cracks.

LANDING GEAR DAMAGE RESULTING FROM EXCESSIVE LOADS IN LANDING.

Because of the close tolerances and highly stressed condition of the landing gear structural components, excessive landing loads can cause critical damage.

WING CRITICAL AREAS

GENERAL

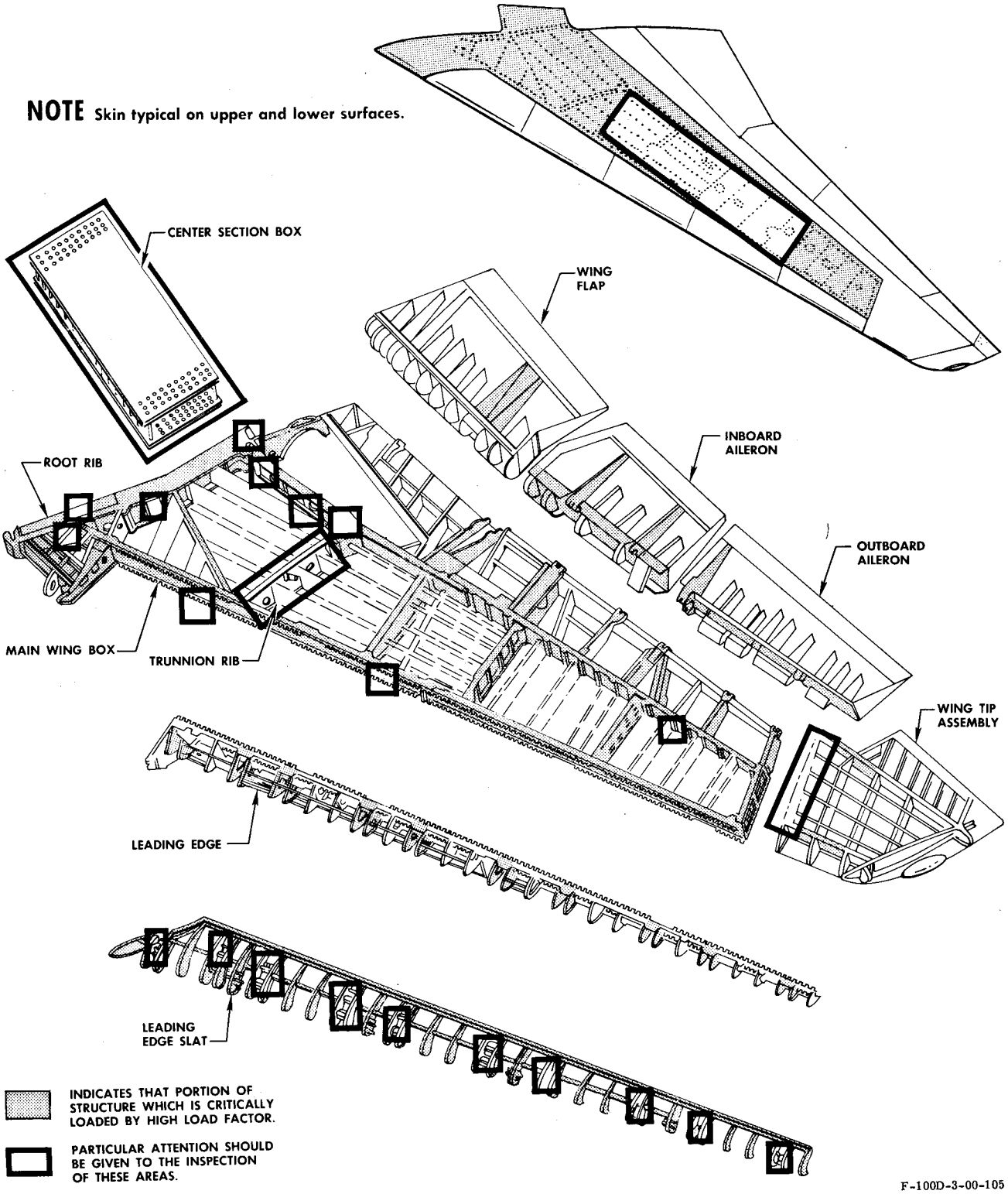


Figure 1-17. Excessive Load Factor Critical Areas—Wing

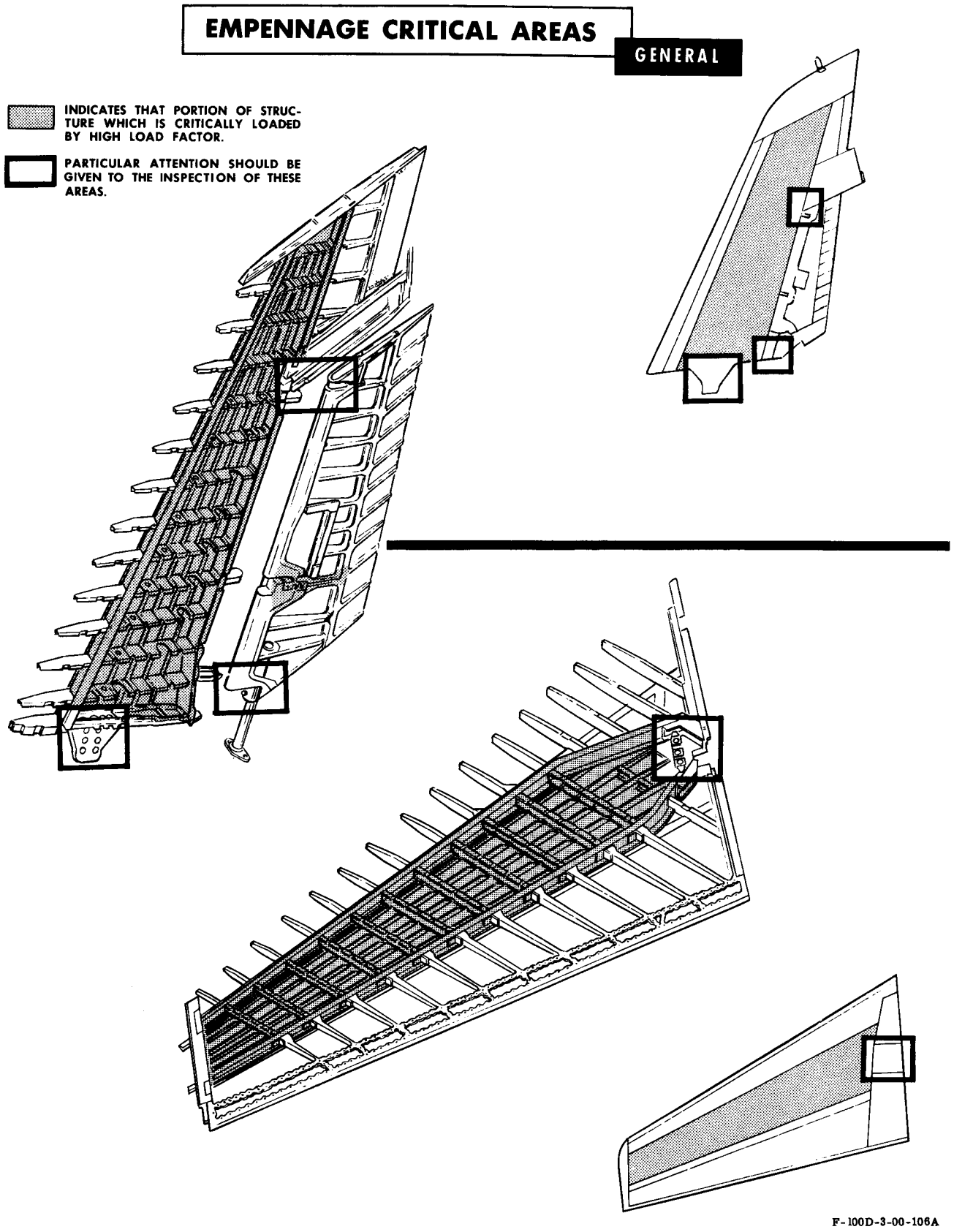


Figure 1-18. Excessive Load Factor—Critical Areas—Empennage

The areas and suggested methods of inspection are as follows:

- Magnaflux-inspect nose gear inner and outer cylinder, shimmy damper attach bolt, drag brace, and drag brace trunnion for cracks.
- Magnaflux-inspect main gear inner and outer cylinder, upper and lower torque link, and torque link bolts for cracks.

DAMAGE RESULTING FROM SEVERE ENGINE STALL.

After a severe engine stall, check following for damage: (1) entire intake duct for deformation of skins and frames, particularly with respect to cracks in duct skin; (2) engine tail pipe for deformation due to inward pressure differentials; and (3) left forward engine mount retaining screw for possible tension failure. This part is critical at the aft end of the acme thread.

DAMAGE AFTER SEVERE TURBULENT OR HARD IN-FLIGHT REFUELING HOOKUPS ARE ENCOUNTERED.

In case in-flight refueling has taken place in severe turbulence or if a hard hookup has been encountered, the following checks for damage should be accomplished as soon as possible.

- Check for misaligned or bent probe.
- Check for cracks in area of the four upper support bolts on in-flight refueling probe support fitting on airplanes not changed by T.O. 1F-100-960.
- Check wing spar in area of spar attaching fitting for cracks and breaks (using Zyglo dye penetrant or equivalent) on those airplanes not changed by T.O. 1F-100-960.
- On airplanes changed by T.O. 1F-100-960, check in-flight refueling probe for the following:
  (1) Alignment of fitting and spar cap (shim as needed)
  (2) Attaching bolts for proper torque (500 to 750 inch-pounds, working from center to outer edge)

DAMAGE RESULTING FROM ABNORMAL DRAG CHUTE DEPLOYMENT.

If the drag chute is deployed in any manner contrary to the instructions in Flight Manual, T.O. 1F-100D-1, a thorough inspection of the aft fuselage structure should be made. The rigging of the drag chute mechanism (192-73704) should be checked. The drag chute hook (192-73732) and the trunnion (192-737129) should be carefully checked for evidence of yielding or cracking. If evidence of structural damage is found in either of these parts, all the items on the following list should be inspected for damage.

PART	TYPE OF FAILURE	LOCATION
192-73729 bolts	Tension & shear	Joining 192-73704 & 192-73742
192-31486 intercostal	Bending	Fus sta 495 to 511
Frame fus sta 511 192-315172	Bending due up-load	Top & side
Fitting 192-73733	Bending	Top c
Fasteners thru skin	Shear and/or tension	Fus sta 495 to 511 Top c to WL-25
192-315106 fitting	Bending	Aft leg
¼-inch bolts attaching 192-315106 fitting to 192-31486 intercostal	Shear	Fus sta 498
192-73742 support angles	Compression	Fus sta 511
5/32 monel rivets thru 192-315204 splice angle	Shear	Fus sta 495 lower shoulder
192-315204 splice angle	Tension	Fus sta 495 lower shoulder
192-73704 mechanism assembly	General check of all parts affected by drag chute cable load	Fus sta 511

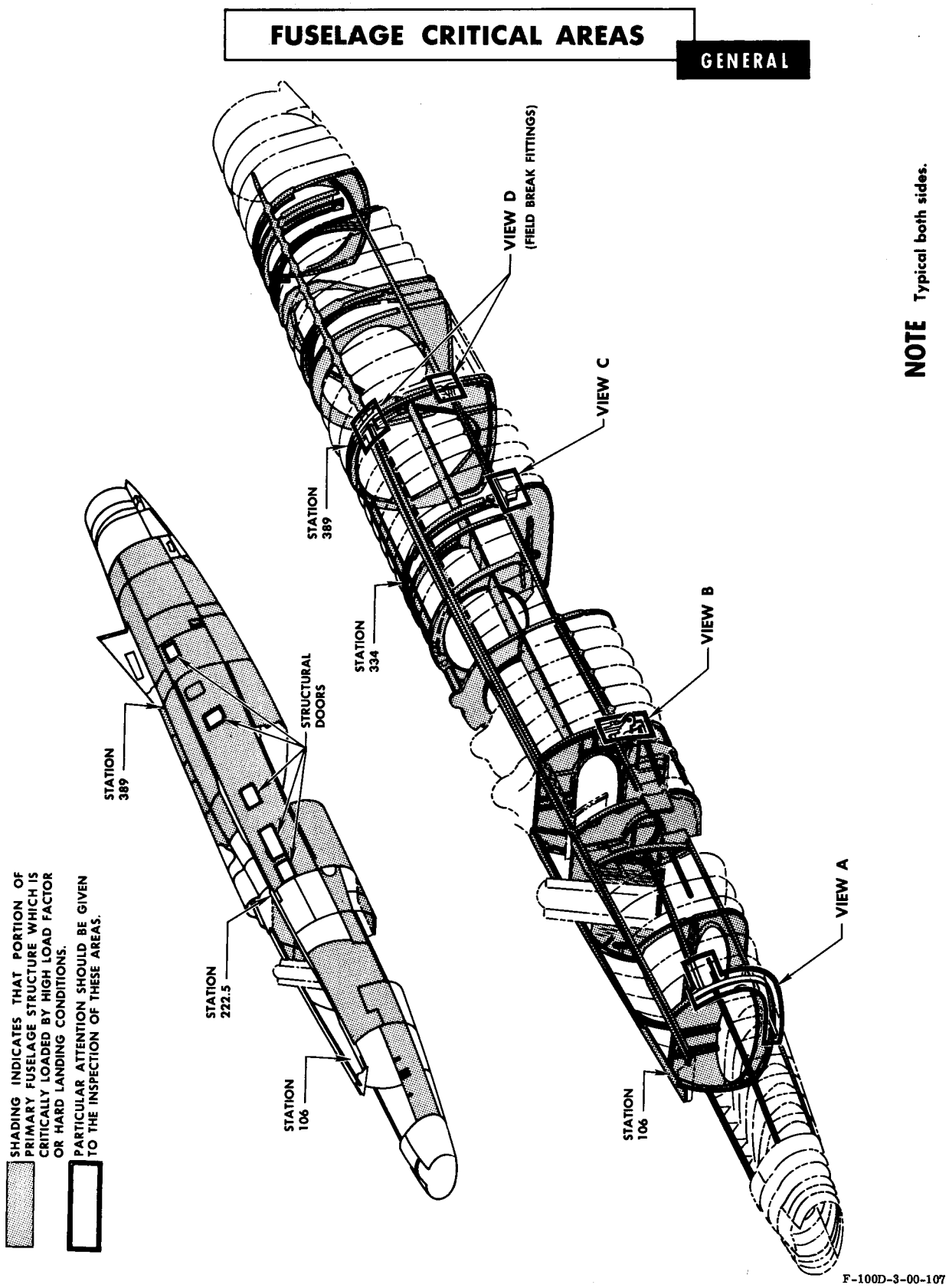


Figure 1-19. Excessive Load Factor Critical Areas—Fuselage—F-100D Airplanes

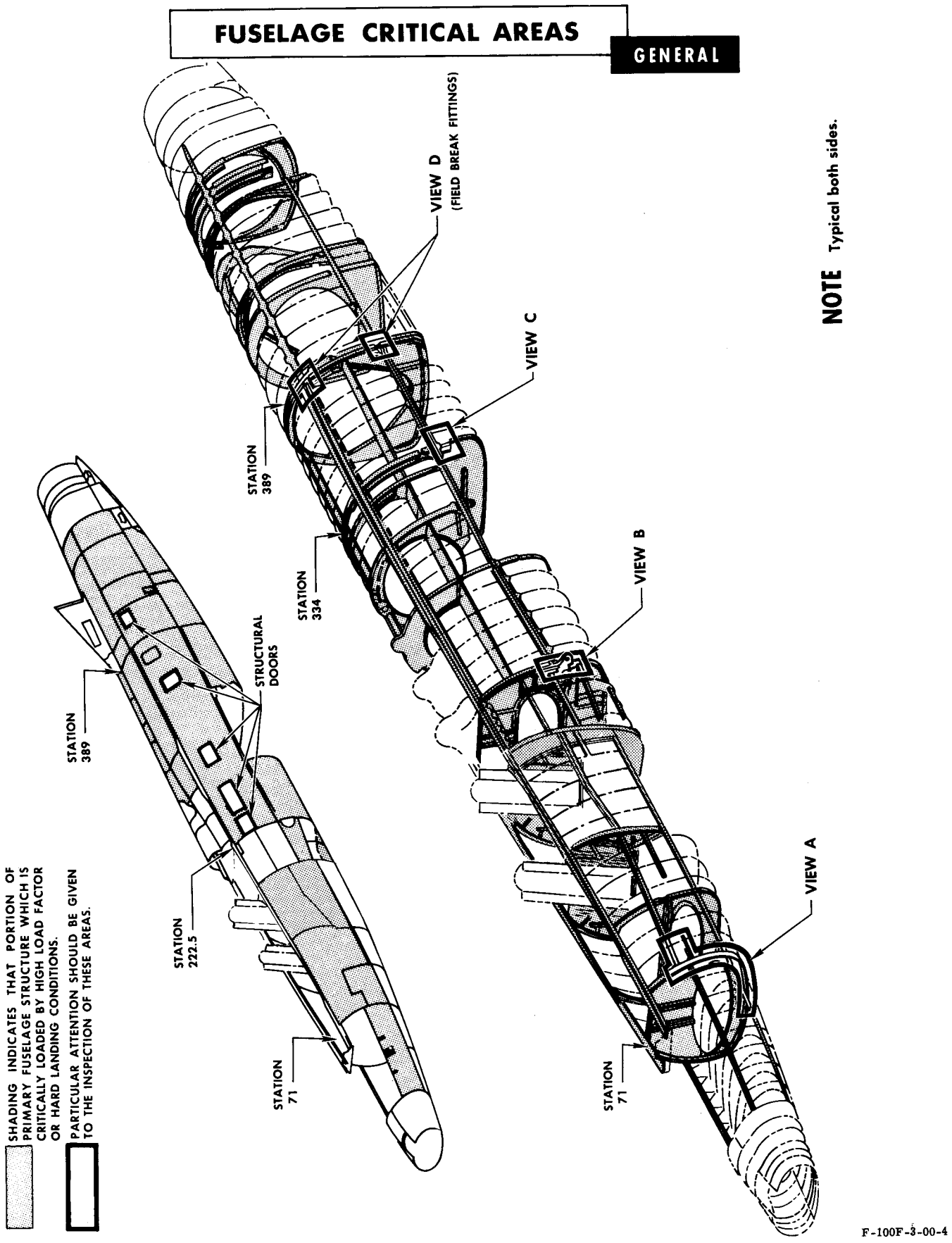


Figure 1-19A. Excessive Load Factor Critical Areas—Fuselage—F-100F Airplanes

FLIGHT-TESTING AFTER CHECKING FOR DAMAGE RESULTING FROM EXCESSIVE LOADS IN FLIGHT OR LANDING.

When adequate inspection and functional checks reveal the structure to be sound and the operation of units to be normal, a test flight should be made. The test flight should consist of a normal take-off and climb to a 10,000-foot altitude. With directional and lateral trim in the takeoff position and the wings carefully held at a level attitude, the airspeed should be increased in 50-knot increments from 250 to 550 knots, maintaining a 10,000-foot altitude. The ball widths off-center should be noted from the turn and bank indicator at each 50-knot increment. If the maximum ball widths off-center at all noted speed increments are ¾ or less, the airplane may be considered acceptable directionally. The airplane is acceptable laterally if, over the speed range tested, the maximum lateral stick force to hold the wings level is three pounds or less.

If the results of the structural inspection and test flight are satisfactory, the airplane may be considered acceptable for flight status. If the test flight indicates the airplane is either laterally or directionally out of the tolerances stated above, a complete alignment check should be made. The results of the alignment check and the test flight should be forwarded to the maintenance officer for engineering evaluation.

AIRPLANE ALIGNMENT.

LEVELING AIRPLANE.

There are times when an airplane must be placed in a level attitude for various reasons relative to repair. Leveling may be done with a spirit level or a transit. When accurate leveling is required, the transit method of leveling is recommended.

LEVELING AIRPLANE WITH SPIRIT LEVEL.

Support airplane on jacks. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.) Adjust jacks until airplane is level laterally as indicated by a spirit level resting on the leveling lugs mounted on the right main gear wheel well web.

Longitudinal leveling may be done using the leveling lugs on the nose gear wheel well right hand web. Recheck lateral leveling after leveling longitudinally.

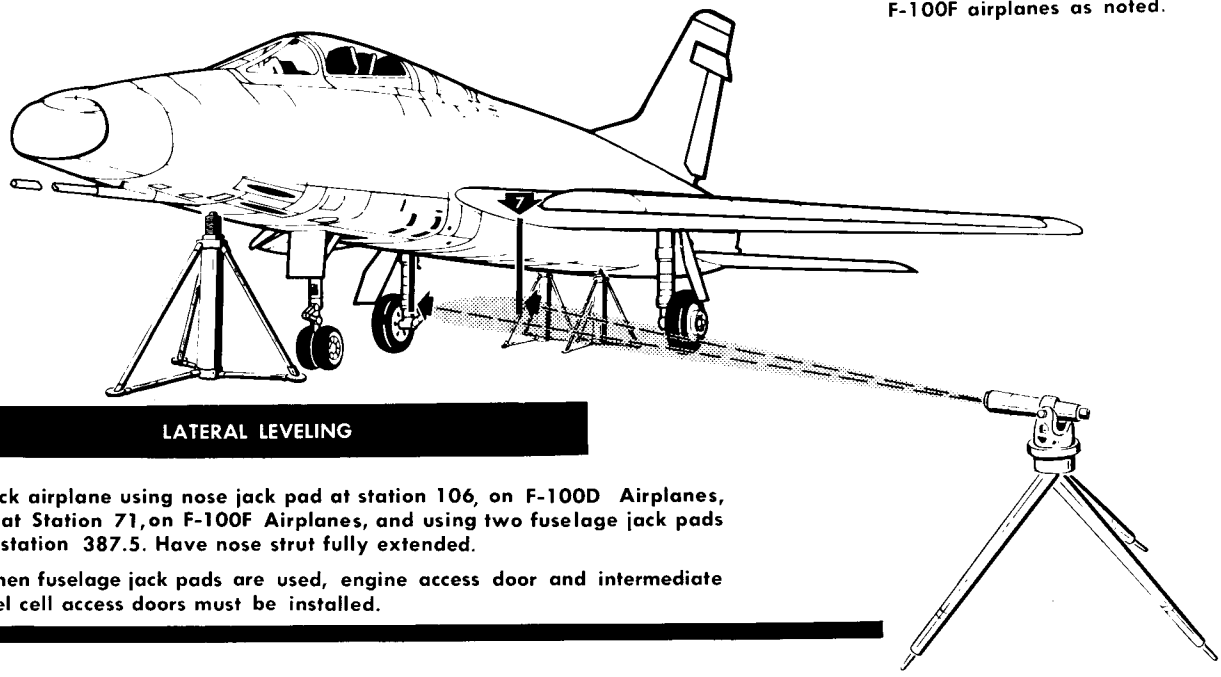
TRANSIT LEVELING AIRPLANE.

Preliminary leveling with a spirit level may or may not be done prior to transit leveling. Because of their location, the longitudinal leveling points on the left side of the fuselage are quite difficult to use for leveling. It is much more convenient, and just as reliable, to use the forward and aft plumb-bob points. (See figure 1-20.) These plumb-bob points are used in the leveling description that follows. Using fuselage jack pads, jack the airplane as outlined in "F-100D General Airplane," T.O. 1F-100D-2-1. To level laterally, position transit at a point forward and to the left of the nose of the fuselage. Level the transit at a position where there is a clear view from under the airplane. Drop scales from lateral check points on underside of wing. (See figure 1-21.) Raise or lower fuselage jacks until readings on both scales are identical. For longitudinal leveling, position

LEVELING AIRPLANE WITH TRANSIT

GENERAL

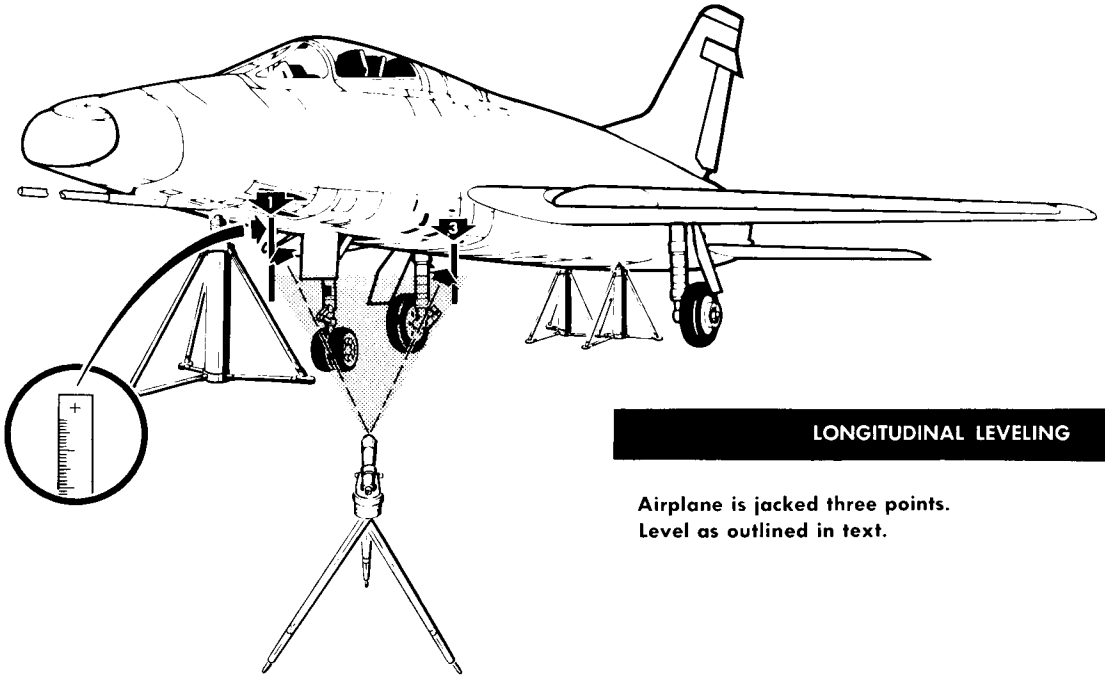
NOTE This is Applicable for F-100D and F-100F airplanes as noted.



LATERAL LEVELING

Jack airplane using nose jack pad at station 106, on F-100D Airplanes, or at Station 71, on F-100F Airplanes, and using two fuselage jack pads at station 387.5. Have nose strut fully extended.

When fuselage jack pads are used, engine access door and intermediate fuel cell access doors must be installed.



LONGITUDINAL LEVELING

Airplane is jacked three points.
Level as outlined in text.

F-100D-3-00-99B

Figure 1-20. Leveling Airplane With Transit

transit on left side of airplane. Level the transit at a position where there is a clear view from under the airplane. Drop scale from forward plumb-bob point and another from the aft plumb-bob point. Raise or lower nose jack until reading at a scale dropped from aft plumb-bob point is 2.151 inches less than reading at scale dropped from forward plumb-bob point. Recheck lateral leveling after leveling longitudinally.

ALIGNMENT CHECK.

Generally the only time an alignment check of the airplane will be necessary is when the airplane does not meet standard outlined in "Flight-testing After Checking for Damage Resulting from Excessive Loads in Flight or Landing."

The alignment check is made with external stores removed and fuel tanks empty. Airplane must be supported on fuselage jacks and in a level attitude. Jack airplane high enough to level fuselage longitudinally with landing gear completely extended and clear of the ground. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

A surveyor's transit is used for accurate leveling of the airplane and for the alignment check. To obtain correct information, both transit and airplane must be leveled carefully. (Refer to "Leveling Airplane.") The transit must not be disturbed after it has been leveled for any phase of the check. The airplane must remain undisturbed throughout alignment check.

Note

Personnel will not be permitted on the airplane while it is on jacks and the alignment check is in progress. All reference points must be reached from work stands.

SYMMETRY CHECK.

With the airplane supported on jacks, and leveled, drop plumb bob from check points and mark locations on the floor, which is a reference plane in this case. The measurements are then taken from point to point with a tape. The horizontal stabilizer must be set at zero degree incidence. This can be done with the rigging fixture. Dimensions must fall within tolerances shown in figure 1-22 and be symmetrical within one-half inch.

ELEVATION CHECK.

To check elevation dimensions, with the airplane supported on jacks and leveled, set transit forward and to left of nose. (See figure 1-23.) The telescope must be low enough to have a clear, level sight under the airplane. Dimension 3 is the zero or reference dimension. Drop a scale from point "B" and take reading. Drop a scale or tape from other check points, as shown in figure 1-23, and take readings. To obtain proper check dimensions, subtract dimensions 3 from all other readings.

Note

For readings on empennage, horizontal stabilizer must be set at zero degree incidence. (Use rigging fixture.)

AFT FUSELAGE REFERENCE DIMENSIONS.

To check alignment of the fuselage rear section and empennage, support fuselage rear section on jacks at two aft jack points (station 495) and forward jack point (station 428). (See figure 1-24.)

Note

The fuselage rear section must be raised high enough so that a clear sight may be taken underneath the airplane with a transit.

Set up transit to one side, in line with the fuselage rear section attachment point and about halfway up the fuselage. To position fuselage rear section attachment fitting, swing transit through its vertical axis and take reading on scales. Raise or lower aft end of fuselage rear section until readings are identical. Lower transit so that a clear sight may be taken under fuselage. Drop scales from center of lower fuselage rear section attachment fittings. Take readings on scales. Raise or lower either aft jack until readings are identical. Reading taken on scale at lower fuselage rear section attachment fitting is the zero or reference dimension. For proper dimensions, subtract this reading from other readings taken.

Note

For readings on empennage, horizontal stabilizer must be at zero degrees incidence. (Use rigging fixture.)

When the alignment of a spare fuselage rear section is doubtful, it is recommended that it be installed on the airplane it is to be used on and that the alignment be checked in relation to the airplane.

HORIZONTAL STABILIZER ALIGNMENT CHECK.

To determine whether the horizontal stabilizer structure has exceeded yield conditions, an alignment check should be made as indicated by figure 1-24A. Since the level of the ground plane is important to alignment accuracy, the use of a spirit level is recommended in determining a suitable alignment location.

REFUEL PROBE ALIGNMENT.

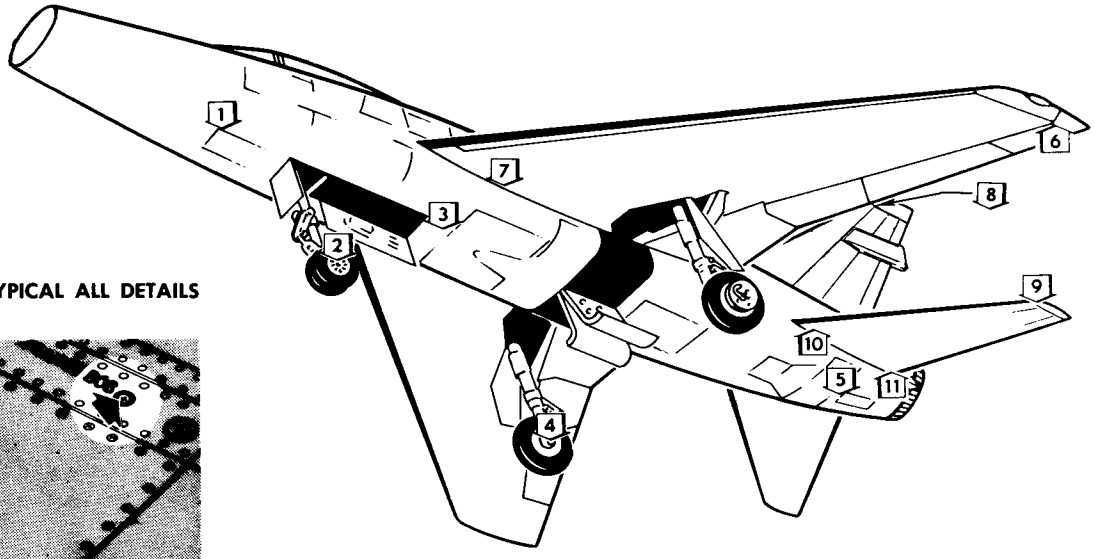
See figure 1-24B.

AIRPLANE REFERENCE DIMENSION CHECK.

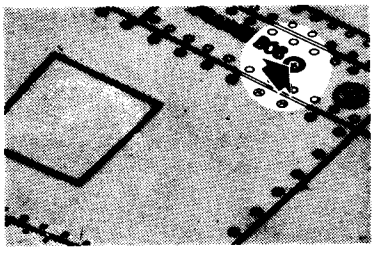
Meaningful tolerances for reference dimensions are virtually impossible to state. They are design dimensions. Jig tolerances are held to a minimum, consistent with practical and economical production methods.

ALIGNMENT CHECKING POINTS

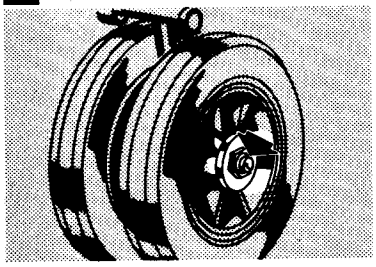
GENERAL



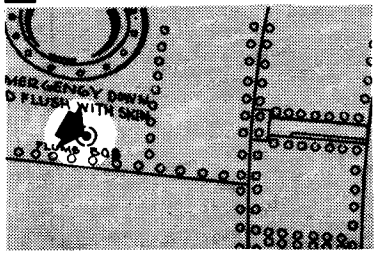
FWD TYPICAL ALL DETAILS



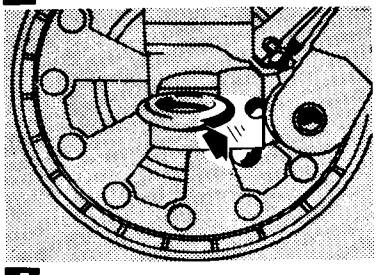
1 FWD PLUMB BOB POINT



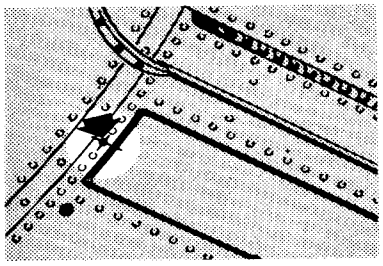
2 NOSE GEAR



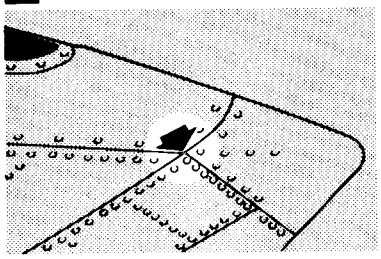
3 AFT PLUMB BOB POINT



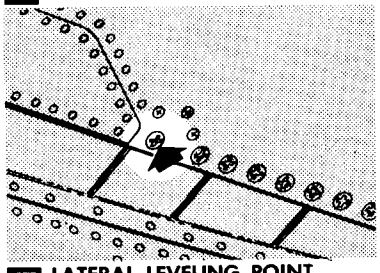
4 MAIN GEAR



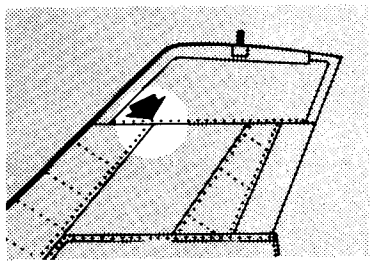
5 AFT FUSELAGE CHECK POINT



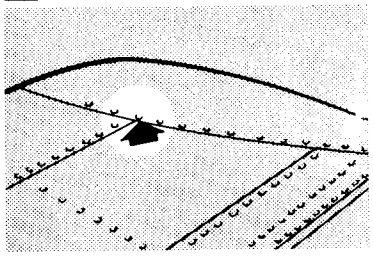
6 WING TIP



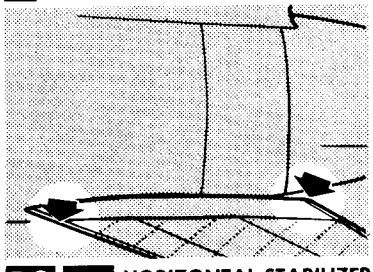
7 LATERAL LEVELING POINT (TYPICAL BOTH SIDES)



8 VERTICAL STABILIZER



9 HORIZONTAL STABILIZER



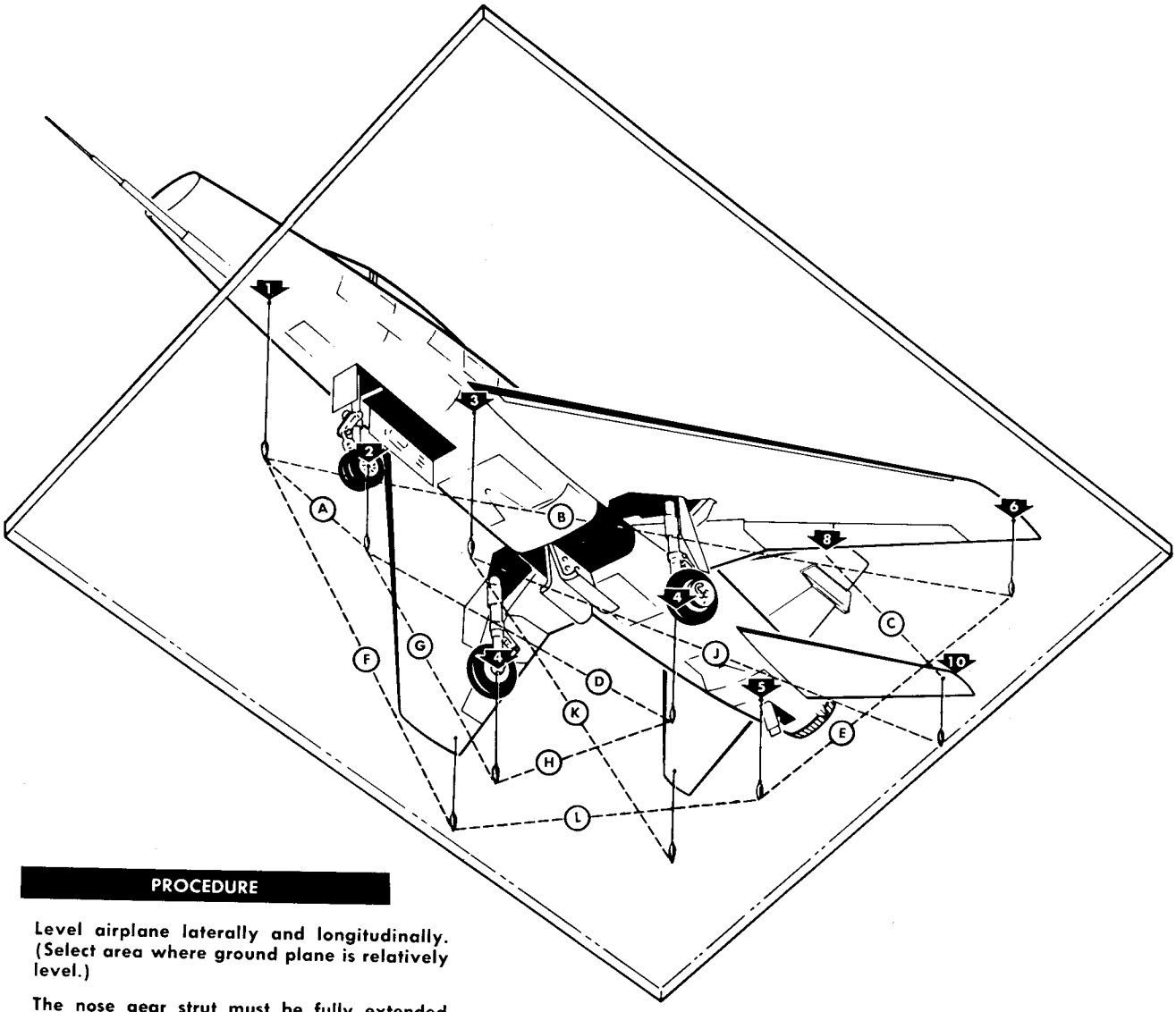
10 11 HORIZONTAL STABILIZER

F-100D-3-00-6D

Figure 1-21. Alignment Checking Points

SYMMETRY CHECK

GENERAL



PROCEDURE

Level airplane laterally and longitudinally. (Select area where ground plane is relatively level.)

The nose gear strut must be fully extended and the nose wheel must be parallel with the centerline of the fuselage.

Drop plumb bobs from check points and measure horizontal distance on ground plane.

The airplane must be symmetrical within 1/2 inch.

NOTE Dimensions are checked with horizontal stabilizer set at 0-degree incidence. (Install horizontal stabilizer checking fixture on airplane; then align horizontal stabilizer leading edge centerline with 0-degree increment on fixture.)

DIMENSIONS

A = 4 FT 1/8 IN. (±1/2 IN.)
B = 34 FT 5 IN. (±2 IN.)
C = 14 FT 8-13/16 IN. (±1 IN.)
D = 15 FT 3-15/16 IN. (±3/4 IN.)
E = 18 FT 10-1/2 IN. (±1 IN.)
F = 35 FT 3-1/32 IN. (±2 IN.)
G = 15 FT 8-13/32 IN. (±3/4 IN.)
H = 10 FT 6-13/16 (±1) IN.
J = 28 FT 11-15/32 IN. (±1 IN.)
K = 29 FT 4-1/16 IN. (±1 IN.)
L = 18 FT 10-1/2 IN. (±1 IN.)

F-100D-3-00-100B

Figure 1-22. Symmetry Check

ELEVATION CHECK

GENERAL

NOTE For readings on the empennage, the horizontal stabilizer must be at 0-degree incidence. (Use rigging fixture.)

NOTE Applicable for F-100D and F-100F airplanes as noted.

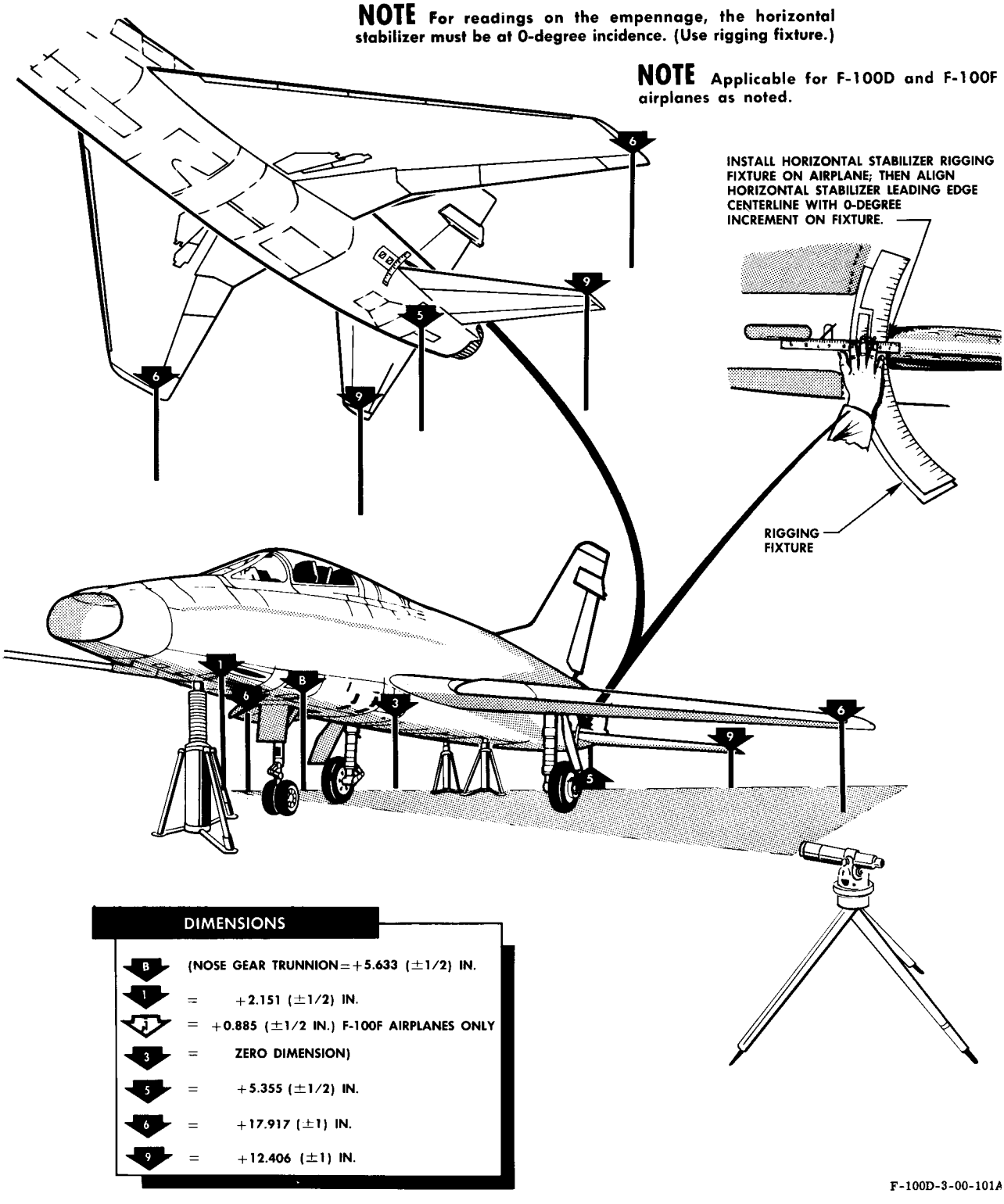


Figure 1-23. Elevation Check

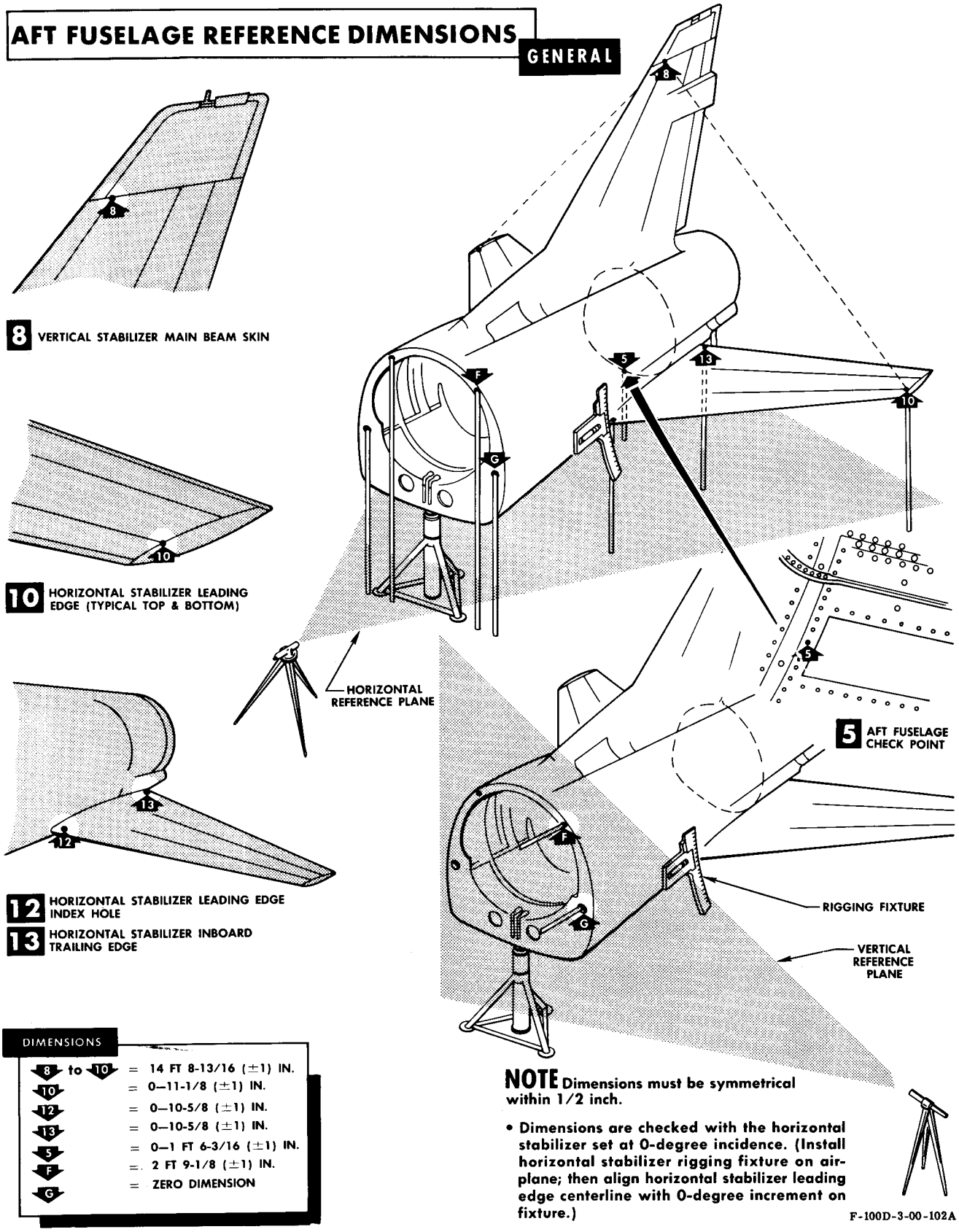
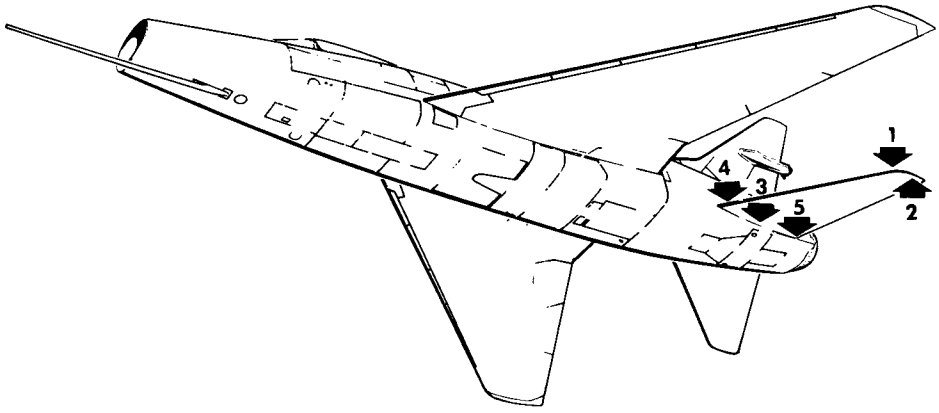
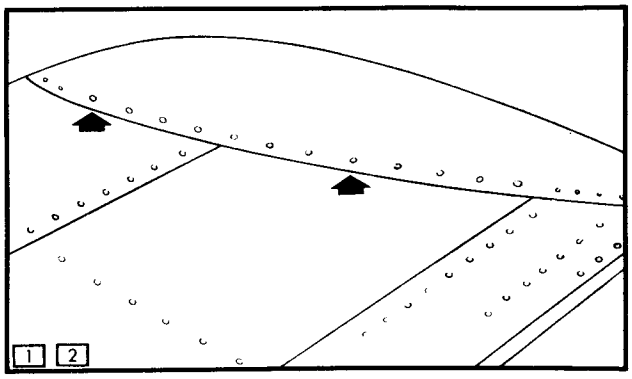


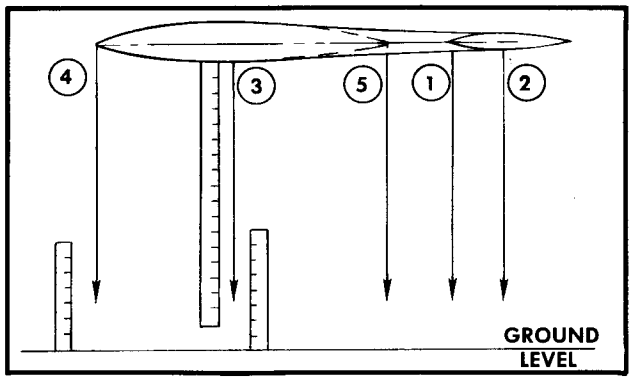
Figure 1-24. Aft Fuselage Reference Dimensions

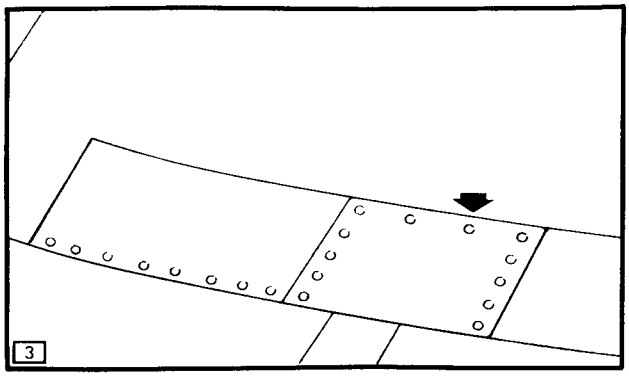
HORIZONTAL STABILIZER ALIGNMENT CHECK

GENERAL

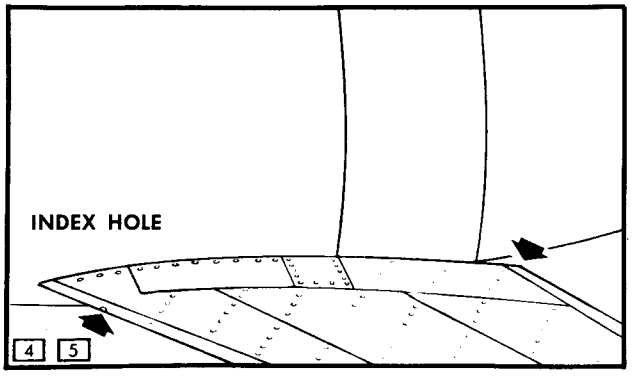








DIMENSIONS
L_1-L_2=5/32 (±5/16) IN.
L_2-L_3=1-7/16 (±1-7/8) IN.



PROCEDURE

Before making alignment check, select area where ground plane is relatively level.

Drop plumb bob lines of equal length from point 3 of left and right stabilizer and adjust airplane laterally (by jacking one main gear) until the difference in distances from ground to plumb bob points is no greater than 1/8 inch (L_{3L}−L_{3R}=0 to 1/8 inch).

Drop plumb bob lines of equal length from points 4 and 5. Level stabilizer with ground by adjusting stabilizer until the difference in distances from ground to plumb bob points is no greater than 1/8 inch (L_4−L_5=0 to 1/8 inch).

Measure distances to ground from points 1, 2, and 3 by totaling scale readings of plumb bob points to stabilizer surface and ground (L_1, L_2, and L_3).

NOTE If the dimensions shown are exceeded, the structure will have to be referred to the prime AMA for disposition.

F-100D-3-00-133

Figure 1-24A. Horizontal Stabilizer Alignment Check

REFUEL PROBE ALIGNMENT

GENERAL

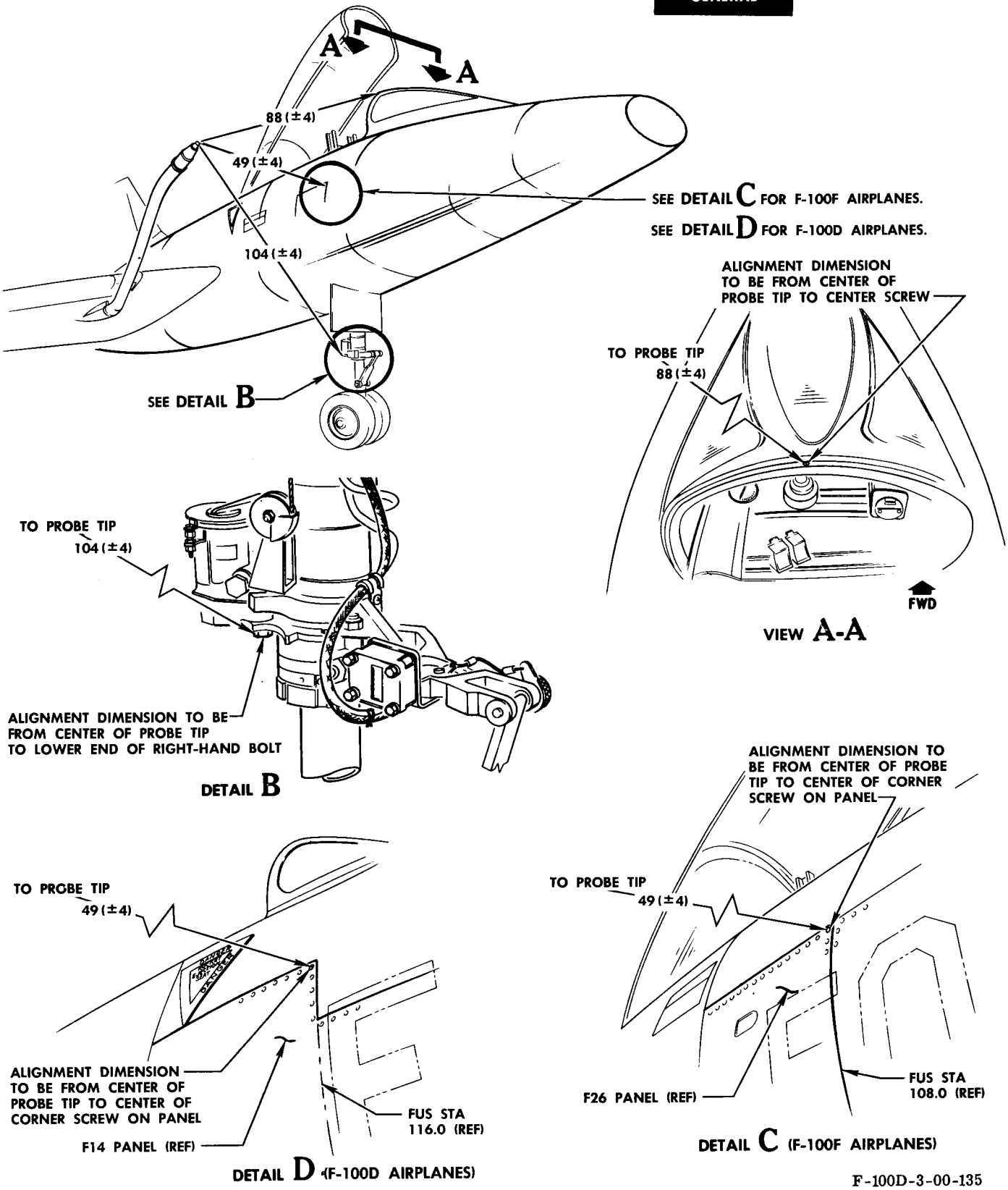


Figure 1-24B. Refuel Probe Alignment

However, an airframe is not an absolutely rigid structure. It is flexible and springy. It is capable of withstanding tremendous forces but its members must deflect under the influence of these loads. Because of this flexibility, the normal loads imposed on the structure during flight and landings can cause minute shifting, or slipping, or the structural members. Each airplane, after considerable flying, will have settled into an individual pattern (most natural for it). Some dimensions may have been changed considerably from design dimensions and jib tolerances, but, under normal conditions, this does not necessarily mean that damage has occurred. Many airplanes develop pronounced buckles or wrinkles in some portion of the airframe. This condition is generally caused by the shifting and settling of the structure. It is a normal and expected condition, provided there is no record of unusually severe loads in the history of the airplane. Tolerances can have value in determining if an airframe component has been overstressed only if actual dimensions (measured after considerable flight time) are known, since these dimensions would be the true and normal ones for that component.

FORWARD FUSELAGE REFERENCE DIMENSIONS.

To level the fuselage forward section with wing and fuselage rear section removed, support fuselage on forward and aft jack points. Locate transit behind and slightly below the fuselage rear section lower attachment fittings. With scales on the lower attachment fittings, adjust side jacks until readings are equal. Move transit to the left side of the fuselage, and adjust it a little below the longitudinal leveling points. With scales on longitudinal leveling points, adjust fuselage forward section jack until readings are equal.

Vertical dimensions of the fuselage forward section reference points are taken from the aft fuselage section (station 389) attachment fittings. These will be the zero or reference dimensions. To obtain proper dimensions, subtract reference reading. (See figure 1-25 for F-100D Airplanes and figure 1-25A for F-100F Airplanes.)

WING AND CENTER SECTION REFERENCE DIMENSIONS.

To check alignment of wing and center section with fuselage removed, support it on outboard wing jacks and a jack at the center section front spar. (See figure 1-26.) Set transit forward of, and about in line with, centerline of wing assembly. With the attaching bolt placed in rear wing-to-fuselage attachment fitting of right and left wing outer panels, place a scale vertically on each bolt. Adjust outer wing jacks until scale readings are identical. Chordwise leveling is done in a similar manner. With the inboard leading edge access cover removed, insert attaching bolt in the forward wing-to-fuselage attachment fitting. Set transit in position to one side and just forward of wing leading edge assembly. Place a scale vertically on the forward and aft wing-to-fuselage attaching bolts. Adjust jack at the wing center section front spar until the scale on the rear attaching bolt reads 2.062 inches more than the scale at the forward attaching bolt.

The zero or reference dimension readings are taken at the rear wing-to-fuselage attachment fittings. Subtract this dimension from all other readings taken to get proper dimensions.

WING ALIGNMENT.

In-service wing alignment should not be taken before a visual inspection of the wing has been made. The inspection should be thorough and should reveal no major structural damage. (Refer to "Damage Resulting From Excessive Loads in Flight or Landing," in this section.) Wings which are found to be structurally damaged or which are not within the tolerances specified in the following in-service alignment must be referred to the Weapons Engineering Division of the controlling AMA for individual disposition.

The following steps outline the procedure for (1) setting up the wing and (2) measuring the vertical distance from the wing station points to a reference plane. The reference plane is a line of sight located at any selected distance above the wing. The wing measuring points are located as follows:

- Points 1 and 2 are on the forward and aft wing attach fittings.
- Points 3, 4, 5, and 6 are points where a line through the center of the fasteners on the rear spar would intersect with a line through the center of the fasteners of the ribs or formers of the stations shown in figure 1-26.
- Point 7 is a point at the center of the wing fastener inboard and aft of the intersection of the front spar plane and leading edge station 320 as shown on figure 1-26.
- Points 8, 9, 10, and 11 are points where a line through the center of the fasteners on the front spar would intersect with a line through the center of the fasteners of the ribs or formers of the stations shown in figure 1-26.

1. To set up the wing in the alignment position, proceed as follows:

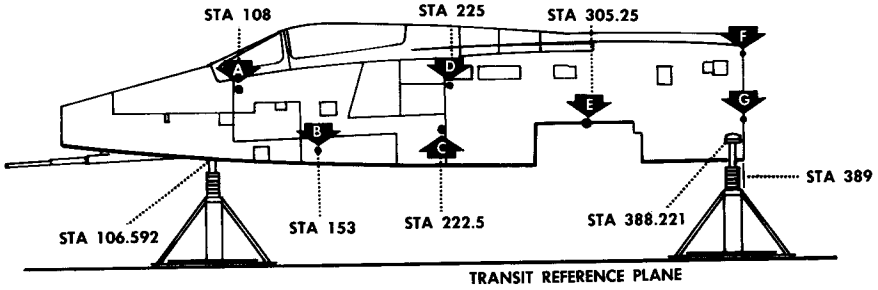
a. Place wing on four adjustable post jacks, locating jacks below points 1, 2, 6, and 7.

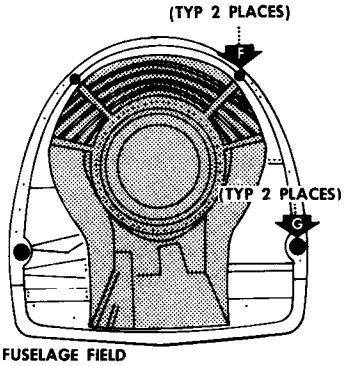
b. Set up an optical level to have a clear line of sight approximately 6 to 8 inches over reference points. This line of sight is to be the reference plane.

c. Determine dimension from reference plane to point 1 on wing. This dimension will hereafter be called dimension "X."

FORWARD FUSELAGE REFERENCE DIMENSIONS

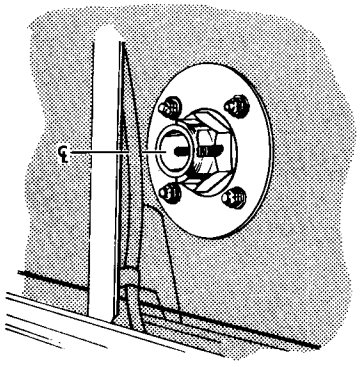
GENERAL



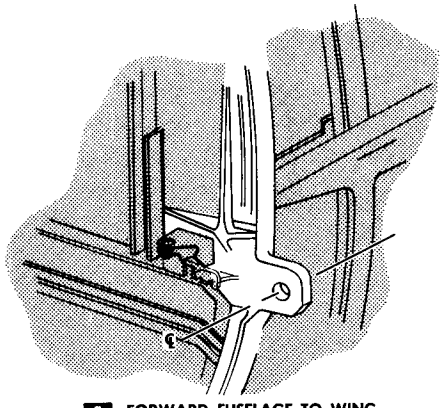


FUSELAGE FIELD BREAK—STATION 389

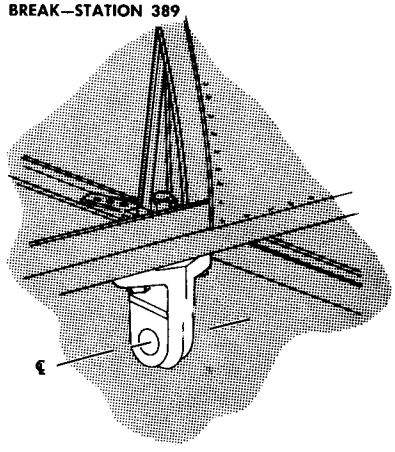
Level fuselage as directed in text. Transit is first positioned to the rear of the fuselage for lateral leveling, then to the left side for longitudinal leveling.



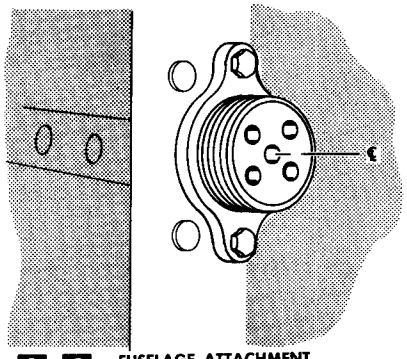
B NOSE GEAR TRUNNION AT STATION 153 WATER PLANE -36.25



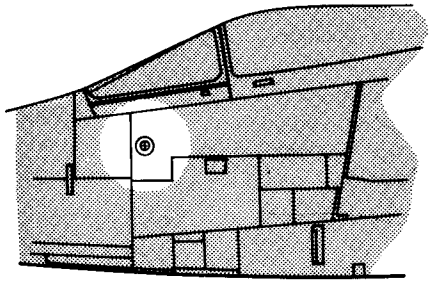
C FORWARD FUSELAGE TO WING ATTACHMENT AT STATION 222.5 WATER PLANE -23.0—BUTTOCK PLANE 35.25



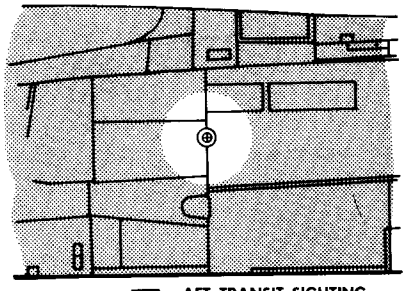
E AFT FUSELAGE TO WING ATTACHMENT AT STATION 305.25 WATER PLANE -21.0 BUTTOCK PLANE 31.187



F G FUSELAGE ATTACHMENT FITTING (TYP 4 PLACES) FUSELAGE STATION 389



A FORWARD TRANSIT SIGHTING POINT AT STATION 108



D AFT TRANSIT SIGHTING POINT AT STATION 225

DIMENSIONS—INCHES	
A = + 18.344	E = − 2.656
B = − 17.906	F = + 33.137
C = − 4.656	G = ZERO DIMENSION
D = + 18.344	

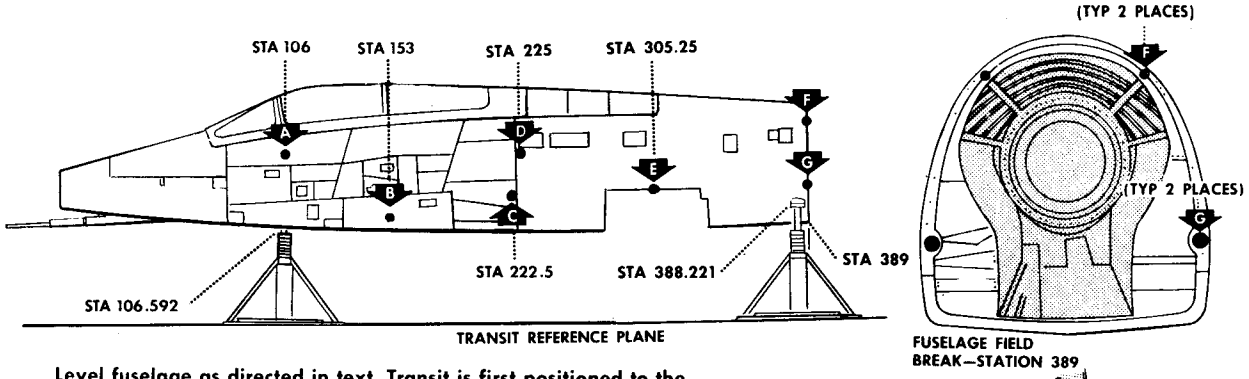
NOTE All dimensions are typical except "A" and "D."

F-100D-3-00-103A

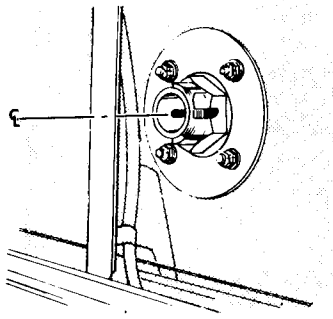
Figure 1-25. Forward Fuselage Reference Dimensions—F-100D Airplanes

FORWARD FUSELAGE REFERENCE DIMENSIONS

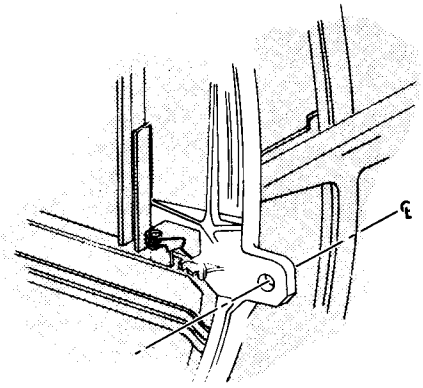
GENERAL



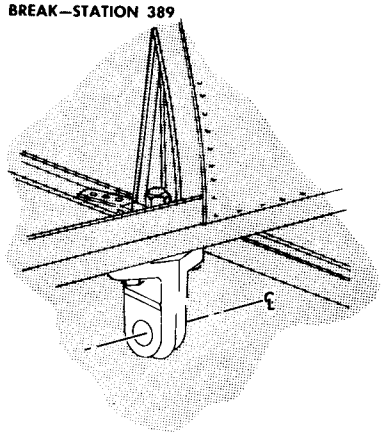
Level fuselage as directed in text. Transit is first positioned to the rear of the fuselage for lateral leveling, then to the left side for longitudinal leveling.



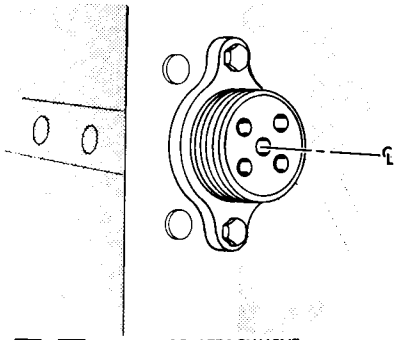
B NOSE GEAR TRUNNION AT STATION 153 WATER PLANE -36.25



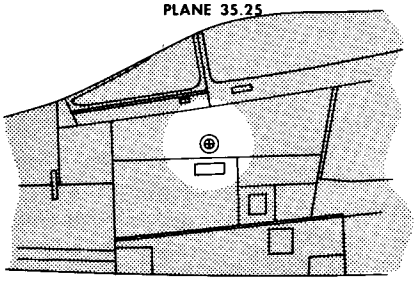
C FORWARD FUSELAGE TO WING ATTACHMENT AT STATION 222.5 WATER PLANE -23.0—BUTTOCK PLANE 35.25



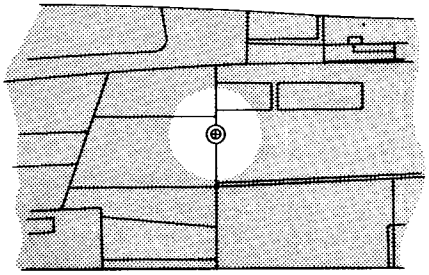
E AFT FUSELAGE TO WING ATTACHMENT AT STATION 305.25 WATER PLANE -21.0 BUTTOCK PLANE 31.187



F G FUSELAGE ATTACHMENT FITTING (TYP 4 PLACES) FUSELAGE STATION 389



A FORWARD TRANSIT SIGHTING POINT AT STATION 106 FUSELAGE REFERENCE PLANE



D AFT TRANSIT SIGHT POINT AT STATION 225 FUSELAGE REFERENCE PLANE

DIMENSIONS—INCHES	
A = + 18.344	E = − 2.656
B = − 17.906	F = + 33.137
C = − 4.656	G = ZERO DIMENSION
D = + 18.344	

NOTE The dimensions are taken from the centerlines.

F-100F-3-00-6

Figure 1-25A. Forward Fuselage Reference Dimensions—F-100F Airplanes

d. Adjust jack under point 2 until dimension between reference plane and point 2 is exactly dimension "X" minus 2.062.

e. Recheck dimension "X" at point 1. Any movement of any of the jacks will cause this dimension to change.

f. Adjust jack under point 7 so that dimension between point 7 and reference plane is exactly dimension "X" minus 1.619.

g. Recheck and readjust if necessary to maintain correct dimensions at points 1, 2, and 7. When all dimensions are correct, the wing is in the proper alignment position.

2. With wing in correct alignment position, measure distance from reference plane to remaining check points (3, 4, 5, 6, 8, 9, 10, and 11). They must be within specified tolerance of the algebraic difference of x minus the reference point dimension given on figure 1-26. (See page 1-69 for tolerance chart.)

If all points are found to be within tolerances, the wing is assumed to be correctly aligned. If the wing is found to be over tolerance, it must be referred to the Weapons Engineering Division for disposition.

CORROSION.

Corrosion can be defined as the deterioration of a metal by direct chemical or electrochemical reaction with its environment. The conditions necessary for corrosion are the same for all metals and alloys, differing in degree, not in kind. Corrosion is not only a surface condition, but is sometimes directed along grain boundaries or other lines of least resistance.

There are two general types of corrosion: one is direct chemical attack, where the reaction is similar to that of an acid on a metal and where its environment can be likened to the acid; the other is electrochemical attack, where anodic and cathodic changes between two different metals occur in the presence of its environment, where the action is similar to that of a battery with an electrolyte present, and where its environment can be likened to the electrolyte.

Both types of corrosion depend on an atmospheric environment containing moisture; the degree or rate of corrosion is dependent on the temperature of the atmosphere and the quantity of impurities present, such as dust, salt, smoke, etc. Other variables that affect the degree of corrosion are material section size, type of material, quality of material heat-treatment, and quality of corrosion preventive maintenance. The various forms of corrosion are evident by their distinctive appearance and are commonly known as uniform etch, pitting, intergranular, exfoliation, and galvanic.

UNIFORM ETCH CORROSION.

Uniform etch corrosion may be recognized by a uniform dull appearance of a polished surface. If the surface is allowed to corrode further, it becomes rough and frosty or icy in appearance. This type of corrosion is the result of direct chemical attack.

PITTING CORROSION.

Pitting corrosion may be recognized by a white or gray powdery deposit blotching the surface of aluminum or magnesium alloys and a brown or red powdery deposit on steel. When the deposit is cleaned away, tiny pits or holes can be seen on the surface. This type of corrosion is the result of direct chemical attack or electrochemical attack between the metal and foreign materials on the surface of the metal in the presence of moisture.

INTERGRANULAR CORROSION.

Intergranular corrosion may be recognized by a streaky or discolored appearance of aluminum alloy surfaces. Alloys consist of quantities of individual grains, each separated from the other with clearly defined grain boundaries which chemically differ from the element within the grain center. The adjacent grains of different elements can react with each other as anode and cathode in the presence of moisture to produce rapid selective electromechanical attack at the grain boundaries. 2024, 7075, and 7178 aluminum alloys are the most susceptible to intergranular corrosion, since their alloy constituents are ideal for electromechanical attack.

EXFOLIATION CORROSION.

Exfoliation corrosion is intergranular, predominates in aluminum alloys, and may be recognized by a blistering or actual separation of metal around or near rivet or bolt holes. Electrochemical attack originates in the previously unprotected surface of the rivet or bolt hole and proceeds along grain boundaries until the metal can no longer contain the corrosive products, resulting in blistering of the outer surface.

GALVANIC CORROSION.

Galvanic corrosion may be recognized around fittings, lap joints, rivet connections, etc, by a build-up of powdery white, gray, or red corrosive products. Galvanic corrosion is the direct result of mating two dissimilar metals that are far enough apart in the galvanic series of metals and alloys. Of the two mated metals, electrochemical attack will occur on the metal closer to the anodic end of the series.

GALVANIC SERIES OF METALS AND ALLOYS

Corroded End (Anodic)

Magnesium and magnesium alloys
Zinc
Clad 7075
Clad 6061
5052
Clad 2024
6061-T6
7075-T6
7178
Cadmium
2017-T4
2024-T4
2014-T6
Steel and iron
Cast iron
Chromium
18-8 stainless steel (CRES)
Lead
Tin
Nickel
Brass
Copper
Bronze
Titanium
Monel
Silver
Gold
Platinum

Procted End (Cathodic)

CONSEQUENCES OF CORROSION.

Corrosive deterioration combined with service loading of an airplane can have serious effects on the structural integrity or service life of primary structural members. Stress corrosion cracking and corrosion fatigue are the results of the effects of corrosive deterioration. Loss of effective material, concentration of stress around the affected area, and acceleration of corrosion as a result of internal stresses are the causes for premature failure of a structural member. Loss of effective material will increase the stress on a member carrying the same load as previously carried. Abrupt changes in cross section, such as surface irregularities caused by corrosion, will increase stress in the affected area. Internal stresses, as induced by cold deformation of material, press and shrink fits, unequal cooling from high temperature, etc, will not only accelerate corrosion, but in certain instances will add to the applied stresses, thereby increasing the probability of cracks and failure. Since stress and corrosion are interactive when combined, cracking may occur at as low a stress as the yield strength of the material. The combination of stress and corrosion may also cause fatigue failure long before the normal service life of a material by diminishing the number of allowable stress cycles.

CORROSION CONTROL.

The necessity for corrosion control is never felt more strongly than when corrosive deterioration has gone as far as to require a major structural repair. The time and expense lost in airplane down time, in procurement of materials, and in personnel, not to mention engineering man-hours required to provide an adequate repair, are prohibitive. In certain instances, relatively minor damage in a critical or inaccessible area becomes a major repair. Of all the factors governing the probability and degree of corrosion, preventive maintenance by field personnel is the only substantial control over corrosion. Since the prime requirement for corrosive conditions to exist is a high-temperature, moisture-filled atmosphere (high humidity), field maintenance depots can be considered forewarned as to the necessity for increased corrosion surveillance and preventive maintenance schedules. If other factors besides high humidity are known regarding the area, such as concentration of industrial smoke or chemicals or a salt-water atmosphere, further vigilance is necessary. Preventive maintenance as a means of controlling corrosion is normally accomplished by a step-by-step procedure.

First, the airplane must be thoroughly cleaned to rid it of all atmospheric and induced impurities, such as salt spray, dirt, gun blast, urine spray, or fuel exhaust deposits. Second, an inspection should be made for the various forms of corrosion, externally and where possible internally. Third, if corrosion is found, the areas should be cleaned of all corrosive products and treated in the manner prescribed for the individual metals.

PREVENTIVE MAINTENANCE.

CLEANING.

Airplane cleaning serves the purpose of eliminating foreign deposits, as previously discussed, and preparing the surfaces for a final corrosion inspection. Either of two types of cleaners may be used: alkaline or solvent. These cleaners may be utilized by means of hand-cleaning, tank-cleaning, or spray-washing, whichever is most suitable. Use of a 20-gallon portable spray cleaner with nozzle attachment (Federal Stock No. 4940-567-9558) is recommended for spray-cleaning.

ALKALINE CLEANING. Before cleaning, all natural entrapments and exposed lubricated parts should be masked, using masking tape such as Federal Stock No. 7510-266-6710. Lubricated parts may be flap and control actuators, bearings, wheels, brakes, etc. Natural

entrapments may be louvers, scoops, wheel wells, etc. In addition, main and nose gear tires should be protected by enclosing them in a vinyl tarpaulin (Federal Stock No. 8340-262-3679).

1. Mix one part alkaline water-base cleaner (Specification MIL-C-25769) by diluting with three to seven parts of water, depending on the amount and degree of foreign matter adherence.

2. Use soft brush (Specification MIL-B-5612) to completely clean surface, including tight places around laps and fasteners. Environmental temperature will govern increments of area to be cleaned before alkaline cleaner will dry. If at all possible, cleaning should be accomplished in shaded areas when temperatures are above 80°F. Allowing alkaline cleaner to dry before rinsing causes stains and leaves residue on surface. The cleaner must not be allowed to dry on plastics, plate glass, and painted surfaces.

3. Rinse surface thoroughly with clean water.

Note

In PACAF areas where clean, salt-free water may be a rationed commodity, strict control of rinse water will be necessary. If this is not possible, a solvent-cleaning method should be used.

SOLVENT CLEANING. When conditions (weather, lack of equipment, etc) prevent use of an alkaline cleaner, then dry-cleaning solvent (Federal Specification P-S-661) or kerosene solvent (Federal Specification VV-K-211C) should be used.

WARNING

In areas of oxygen storage and transfer system, do *not* use solvent. This material and oxygen (liquid or gas) are highly flammable or explosive.

1. Apply solvent with a wiping cloth, sponge, or soft brush (Specification MIL-B-5612) to area being cleaned.

2. Allow solvent to remain on surface for about 30 seconds; then agitate area, using cloth, sponge, or brush.

3. Wipe surface thoroughly with clean, dry cloth. Surface should be dried as much as possible by wiping. Blowers or low-pressure air may be used to speed evaporation in areas such as engines, bays, etc.

ACID BRIGHTENING FOR CLAD ALUMINUM ALLOYS ONLY. Before application of acid, all surfaces and areas to be protected (refer to "Alkaline Cleaning") should be masked, using tape (Federal Stock No. 7510-266-6710), and main and nose gear tires should be enclosed in a vinyl plastic tarpaulin. If spray-cleaning equipment is to be used, make sure equipment is thoroughly flushed clean.

1. Mix one part metal conditioner and brightener (Specification MIL-C-25378) with one part water. (This dilution ratio should not be varied for brightening.)

WARNING

Keep acid brightener from contacting magnesium, plate glass, and non-corrosive-resistant metals. This can best be done by masking areas; however, if contact is made, flushing immediately with water will prevent damage to material.

2. Treatment of surface should be limited to small areas for each application to prevent drying before rinsing and should not be accomplished when environmental temperatures are below 50°F or above 100°F. Apply acid brightener with spray equipment or brush, being sure to use face shield.

CAUTION

If brightener gets into eyes, immediately flush them thoroughly with water and report to dispensary.

3. Agitate acid brightener by scrubbing with brush (Specification MIL-B-5612). Depending on weather conditions (hot, cold, or windy) and the amount of foreign deposits present, allow about 5 to 12 minutes from application before rinsing.

4. Rinse acid brightener from surface, starting from lower surface and working up; this permits water to flow over previously brightened areas. The procedure should then be reversed, rinsing from the upper surface to the lower. All traces of acid brightener should be removed when the rinse water stops bubbling or foaming at the surface.

5. Wipe rinsed surface dry with clean, dry cloth.

Note

Acid brightening should not be performed more than four times a year. To ensure that this limit is maintained, an entry on DD Form 781-2 as to the date and the activity accomplishing the brightening should be made.

CORROSION REMOVAL.

If at any time there is doubt as to what type of metal is being examined or treated, a metal spot check should be performed, using inspection kit No. 4920-831-5932. The use of this kit employs two methods of identification: primary classification of metals and chemical spot analysis. Detailed instructions on the use of the kit may

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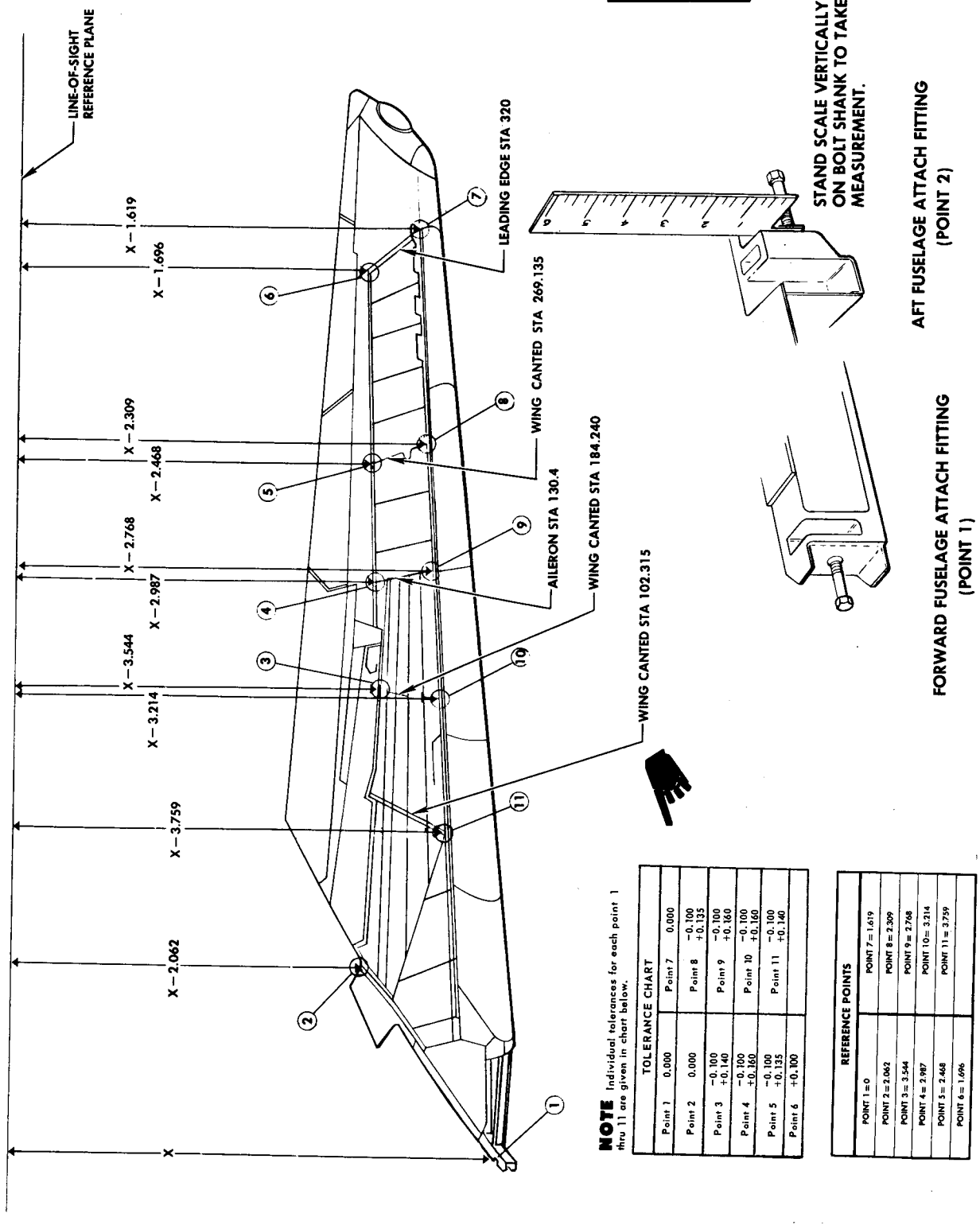
WING ALIGNMENT

GENERAL



NOTE Individual tolerances for each point 1 thru 11 are given in chart below.

TOLERANCE CHART			
Point 1	0.000	Point 7	0.000
Point 2	0.000	Point 8	−0.100 +0.135
Point 3	−0.100 +0.140	Point 9	−0.100 +0.160
Point 4	−0.100 +0.160	Point 10	−0.100 +0.160
Point 5	−0.100 +0.135	Point 11	−0.100 +0.140
Point 6	+0.100		

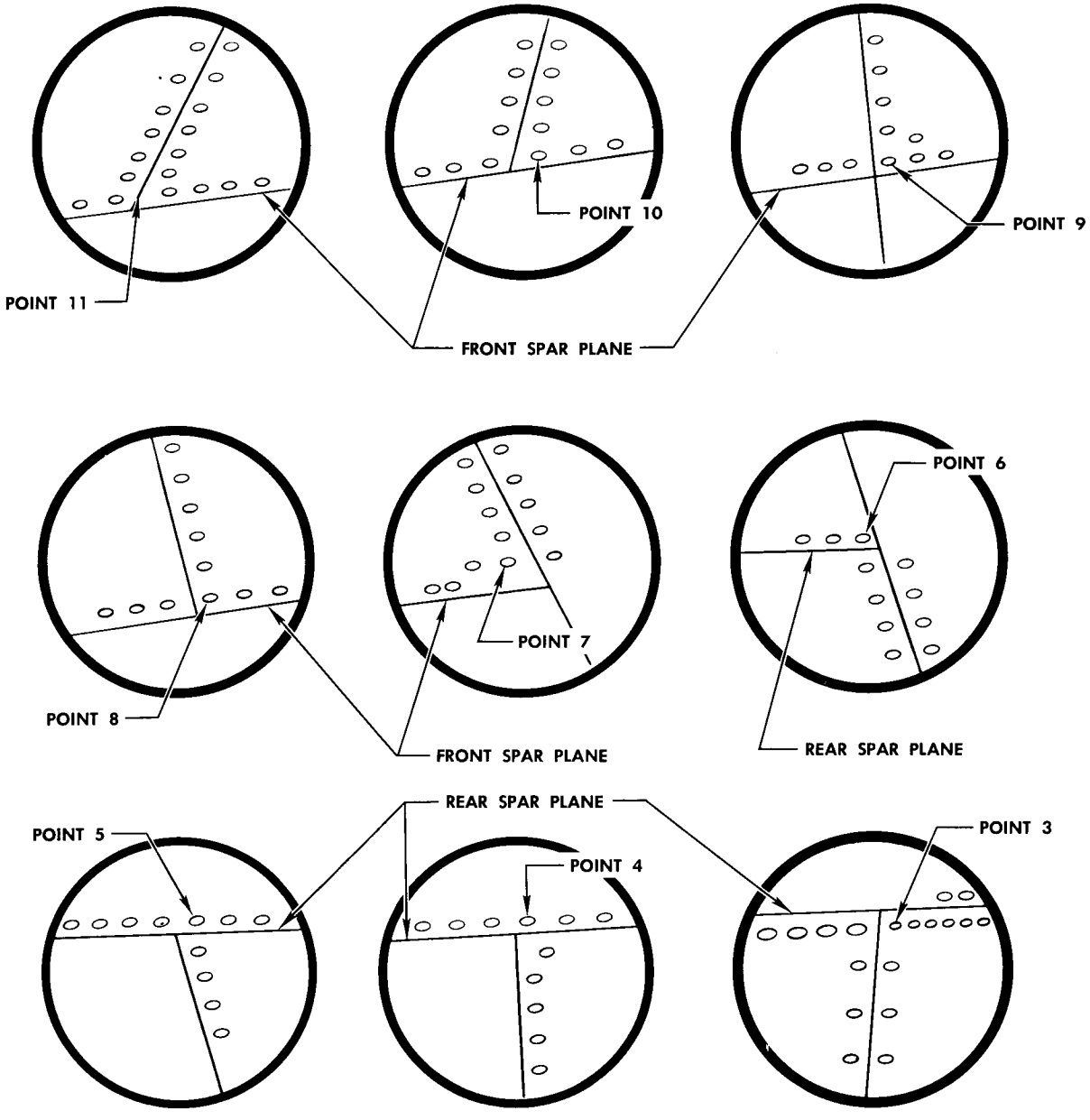
REFERENCE POINTS	
POINT 1 = 0	POINT 7 = 1.619
POINT 2 = 2.062	POINT 8 = 2.309
POINT 3 = 3.544	POINT 9 = 2.768
POINT 4 = 2.987	POINT 10 = 3.214
POINT 5 = 2.468	POINT 11 = 3.759
POINT 6 = 1.696	

F-100D-3-00-130A

Figure 1-26. Wing Alignment (Sheet 1 of 2)

WING ALIGNMENT

GENERAL



REFERENCE POINT LOCATIONS

F-100D-3-00-134

Figure 1-26. Wing Alignment (Sheet 2 of 2)

be found in Corrosion Control and Treatment of Aircraft and Missiles, T.O. 1-1-2, dated 15 May 1961 and subsequent. Certain protective coatings make identification of metals absolutely mandatory.

Certain metals limit the method of corrosion removal. Depending on the metal involved, mechanical methods that may be used are hand scraping, rotary filing, sandpapering, or abrading with aluminum or steel wool. Chemical methods that may be used are applications of metal conditioner and brightener (Specification MIL-C-25378), chromic acid (Federal Specification O-C-303), or metal conditioner and rust remover (Specification MIL-M-10578B).

Note

- If corrosion is discovered in a painted area, all paint must be removed from the area before a corrosion removal method can be used. (Refer to paint stripping procedures.)
- If more than 5 percent of the metal thickness is removed in a critical structural area, the damage must be evaluated. If a repair is decided to be necessary and this repair manual cannot provide a solution, a structural engineer from the prime AMA must be consulted.

REMOVAL OF CORROSION FROM CLAD ALUMINUM ALLOYS. The only time corrosion should be removed mechanically from clad aluminum alloys is when the metal is attacked by exfoliation corrosion; otherwise, the following chemical removal will apply.

1. Mask off nearby plated steel and magnesium, as well as cracks and overlapping surfaces.

2. Apply thickened (undiluted) metal conditioner and brightener (Specification MIL-C-25378) to corroded area, using brush or clean cloth. A wet rag should be handy to immediately remove acid splashes.

CAUTION

Rubber gloves are required while working with acid solutions. If acid contacts skin or eyes, immediately flush thoroughly with fresh water and report to dispensary.

3. As an aid to this chemical surface treatment, lightly agitate surface with aluminum wool (Specification MIL-A-4864), being careful not to remove too much clad material. The acid should not be left on longer than 20 minutes in one application.

4. Sponge off acid material and waste with a damp cloth rinsed frequently with clean water. After corrosion is removed, the surface should be uniformly bright.

5. Finally rinse with clean water and dry with compressed air or clean, dry cloth.

6. If clad material is heavily attacked by exfoliation corrosion, use hand scraper, rotary file, or abrasive wheel to remove corrosion. (Refer to "Removal of Corrosion From Bare Aluminum Alloys.")

REMOVAL OF CORROSION FROM BARE ALUMINUM ALLOYS. Light pitting corrosion can be removed, using the chemical removal method for clad aluminum alloys; however, the following mechanical method is recommended.

1. Remove corrosion products with a hand scraper, rotary file, or abrasive wheel.

2. After all visible corrosion has been removed, blend scraped area into surrounding area.

3. Finally, polish until surface is smooth, using No. 280 grit abrasive paper (Federal Stock No. 5350-721-8119) and then No. 400 grit abrasive paper (Federal Stock No. 5350-721-8118).

REMOVAL OF CORROSION FROM MAGNESIUM ALLOYS. Except when there is light pitting corrosion, the mechanical method (same as used for bare aluminum alloys) is recommended for magnesium. The chemical removal method is as follows:

1. Mask off nearby operating mechanisms, aluminum alloys, plated steel, and cracks.

2. Prepare chromic acid solution (Federal Specification O-C-303) by diluting 10 percent by weight in water. When practical, apply solution hot.

3. Apply solution sparingly to corroded surface with a brush.

4. Allow acid to stand for 5 minutes. Agitating the surface with a brush (nonmetallic bristles) may be necessary to help loosen the corrosive deposits.

5. Thoroughly rinse with clean water.

6. Dry surface with a clean, dry cloth.

REMOVAL OF CORROSION FROM STEEL AND STAINLESS STEEL. A combination of chemical and mechanical means is recommended for the removal of corrosion from steel. However, when there is danger of chemicals becoming trapped and when rinsing is difficult, only mechanical means of removal should be used as indicated in steps 2 and 3.

1. Protect nearby plated areas, copper-bronze alloys, aluminum, braided flexible lines, and operating mechanisms with masking tape.

2. Remove as much corrosion as possible by hand-brushing, scraping, or rotary filing.

3. Loosen corrosion further by agitating surface with steel wool or No. 400 grit abrasive paper. Remove loose particles by wiping with cloth or using low-pressure air.

4. Mix one part by volume of metal conditioner and rust remover (Specification MIL-M-10578B) with one part water in an acid-resistant steel or earthenware container.

WARNING

Proper personnel protection should be used when working with acids. Rubber gloves must be worn; the use of a face mask is dependent on ventilation conditions. Protective clothing at all times is advisable.

5. Apply solution to corroded area with brush and allow one minute for solution to take effect. Thoroughly rinse surface with water (preferably hot) and wipe dry with clean, lint-free cloth.

WARNING

In areas of oxygen storage and transfer system, do *not* use this rust remover. This material and oxygen (liquid or gas) are flammable or explosive.

REMOVAL OF CORROSION FROM COPPER AND COPPER ALLOYS. Mechanical removal of corrosion is recommended on copper and copper alloys where chemical removal would be impractical because of rinsing difficulties or danger of trapping chemicals.

1. Since copper is normally masked during general cleaning operations, remove oil, grease, and dirt from corroded area by wiping with a clean cloth dampened with dichloromethane (Specification MIL-D-6998) or trichloroethylene (Specification MIL-T-7003).

2. Remove corrosion by abrading with brass wire brush or No. 400 grit abrasive paper.

3. Remove loose particles with pressurized air or bristle brush. If part has been removed and is threaded, wipe threaded area with fingerprint remover (Specification MIL-C-15074), applied with a clean cloth.

4. For chemical removal, use the same method given in steps 4 and 5 of "Removal of Corrosion From Steel and Stainless Steel."

CORROSION PROTECTION.

Protection from further corrosion is accomplished by treating the affected area chemically. The surface treatment not only provides protection but provides the proper surface for a paint finish, if required.

There are two types of protective chemical surface treatments: one for bare and clad aluminum alloys and one for magnesium and magnesium alloys. Stainless steel and titanium require no protective treatment; steel requires only a paint finish.

CAUTION

Both chemical surface treatments consist of acid solutions and require proper protective covering. Personnel must wear rubber gloves, goggles, and protective covering. If acid contacts skin or eyes, flush immediately with water and report to dispensary.

The combination of protective treatment and final paint finish gives the surface the best possible protection. However, unless the airplane is painted or will be painted, a final paint finish should not be considered except in special cases, such as steel landing gear and their components.

PROTECTIVE TREATMENT FOR ALUMINUM ALLOYS. Surface treatment consists of applying chemical films for aluminum alloys (Specification MIL-C-5541) to the affected area. Most solutions conforming to this specification leave a stain which may be considered objectionable in appearance, but in no way does the stain detract from the protective value of the treatment. Solutions of Specification MIL-C-5541 must be mixed according to the instructions on its container. Use of a rubber pail (Federal Stock No. 7240-125-9060) is recommended for mixing.

Note

Clean material in accordance with instructions contained in T.O. 1-1-2, before application of Specification MIL-C-5541 solution.

1. Apply Specification MIL-C-5541 solution to affected surface by using a nylon brush or a clean, lint-free cloth. Keep surface wet with solution until a dark yellow or orange stain is evident. Keep solution confined as much as possible to affected area, off operating mechanisms and plated steel, and away from cracks and crevices.

2. Remove excess solution by blotting with a soft, clean cloth. Then rinse with clean water.

3. Allow surface to air-dry or blot dry with clean, lint-free cloth. To speed drying, a low-pressure air stream may be directed onto the surface.

4. Mix fresh Specification MIL-C-5541 solution when solution darkens or becomes contaminated. Keep solution in closed polyethylene container when not in use. Brushes and cloths should not be left in the solution.

CORROSION PROTECTION FOR MAGNESIUM ALLOYS. Surface protection for magnesium alloys consists of a chrome pickle treatment conforming to Specification MIL-M-3171, Type I, or a Dow Process 19. The chrome pickle treatment consists of two types of solution, depending on the type of magnesium treated.

1. For magnesium alloys other than magnesium alloy castings, mix the following quantities of chemicals according to Specification MIL-M-3171:

- Sodium dichromate ($NaCr_2O_7 2H_2O$), 1½ pounds
- Concentrated nitric acid (HNO_3, specific gravity 1.42), 1½ pints
- Water to make one gallon of solution

2. For magnesium sand, permanent mold, or die castings, mix the following quantities of chemicals according to Specification MIL-M-3171:

- Sodium dichromate ($NaCr_2O_72H_2O$), 1½ pounds
- Concentrated nitric acid (HNO_3, specific gravity 1.42), one to 1½ pints
- Sodium, potassium, or ammonium acid fluoride ($NaHF_2$,KHF_2,NH_4HF_2), 2 ounces
- Water to make one gallon of solution

Note

Mix solution in a rubber pail or polyethlene or glass container to avoid contamination of the solution by the container.

3. Apply applicable solution to affected area with a soft-bristled brush or a clean, lint-free cloth. Prevent area from drying too rapidly by reapplying solution for approximately one to 3 minutes. Application temperature must be between 65°F and 90°F. Application time will vary; a brassy or dark brown finish color should be obtained in no less than 30 seconds and no more than 3 minutes. Overextending the time of application will allow powdery deposits to form, thus seriously affecting paint adhesion.

4. Immediately rinse treated area with clean running water after obtaining desired finish color. Rinsing may also be accomplished by wiping the area with wet, lint-free cloth frequently rinsed in clean water. Surface drying should be accomplished by normal air or by directing low-pressure air onto the surface.

5. The Dow Process 19 may also be used for magnesium castings. Mix the following quantities of chemicals in one gallon of water:

- Chromic acid (CrO_3), 1⅓ ounces
- Calcium sulfate ($CaSO_4$), one ounce

Slowly add chromic acid and then calcium sulfate while stirring. Stir for 15 minutes to ensure that chemicals are dissolved.

6. Apply this solution to affected area with a swab and allow to remain for one minute. Add more solution if necessary to maintain wetness during the time interval.

7. Thoroughly rinse with clean water and allow to dry.

FINAL PROTECTIVE PAINT FINISH.

In case a final paint finish is required, application of the finish must be made within 2 hours after the chemical surface treatment is dry. The surface to be painted must be thoroughly clean and free from contamination. A stained surface resulting from a chemical surface treatment is not considered contamination. Attempts to remove this stain destroy the benefits of the surface treatment. Because of the limited use of paint finishes on prime aircraft, this manual covers only quantities and identification of material requirements for final paint finishes. Details regarding the use of final paint finishes can be found in T.O. 1-1-2, 1-1-636, and 1F-100-2-1.

CORROSION TREATMENT.

PAINT REMOVAL.

If the airplane is painted in an area where corrosion is detected, the paint should be completely removed. The method of paint removal will depend on the ease of accessibility to the corroded area.

Note

For complete airplane paint removal procedures, refer to T.O. 42A2-1-1.

PAINT STRIPPING IN ACCESSIBLE AREAS.

1. Apply solvent-type paint and lacquer remover (Specification MIL-R-25134) with a long-handled brush (Specification MIL-B-5612).

CAUTION

This paint remover is toxic and should be used in a well-ventilated area.

2. Let stripper remain on surface for approximately 5 to 15 minutes to allow paint finish to soften and wrinkle.

3. Wash surface with water under high pressure if possible; otherwise, paint should be removed using a clean, damp cloth.

4. Repeat these steps as necessary to completely remove paint.

PAINT STRIPPING IN CONFINED AREAS.

1. Apply ethyl acetate (Federal Specification TT-E-751) or lacquer thinner (Federal Specification TT-T-266) to corroded area, using a soft-bristled brush or clean cloth.

2. Agitate surface until paint is thoroughly loosened.

3. Wipe surface clean and allow to dry.

CORROSION TREATMENT OF STEEL AND ALUMINUM LANDING GEAR.

Care should be taken in all phases of corrosion treatment of landing gear if the treatment is to be performed on assembled gears. Since landing gear usually consist of a combination of steel, aluminum, and/or magnesium parts besides cracks and crevices, appropriate masking tape applications are necessary to protect individual parts and moving surfaces. The extent of corrosion will determine whether the gear should be treated on the airplane or removed and disassembled. If the gear is heavily corroded, the most effective and economical means of treatment is accomplished after removal and disassembly.

CLEANING, CORROSION REMOVAL, AND CORROSION PROTECTION.

Cleaning, corrosion removal, and corrosion protection are accomplished as indicated in previous applicable paragraphs and dependent on the type of metal treated. Since there is no surface treatment for steel and copper, a final paint finish will serve as corrosion protection except when the parts are inside the gear and in contact with hydraulic fluid.

FINAL PROTECTIVE PAINT FINISH.

1. External steel and aluminum are given a final protective paint finish consisting of the following:
   - One coat pretreatment wash primer (Specification MIL-C-15328)
   - Two coats zinc-chromate primer (Specification MIL-P-8585)
   - Two coats gray enamel (Specification MIL-E-7729) or two coats aluminized lacquer (Specification MIL-L-7178)

2. External copper and magnesium are given a final protective paint finish consisting of the following:
   - One coat pretreatment wash primer (Specification MIL-C-15328)
   - One coat vinyl zinc-chromate primer (Specification MIL-P-15930)
   - Two coats gray vinyl topcoat (Specification MIL-P-15936)

CORROSION TREATMENT OF ROCKET BLAST AND EXHAUST AREAS.

CLEANING.

Rocket blast and exhaust areas should be cleaned as indicated by the alkaline cleaning method, except the solution should be mixed in the following proportions: one part alkaline water-base cleaner (Specification MIL-C-25769) diluted with one to three parts water.

PAINT REMOVAL.

Stripping of paint should be accomplished as given in "Paint Stripping in Accessible Areas" or "Paint Stripping in Confined Areas."

CORROSION REMOVAL AND PROTECTION.

Depending on the metal involved in the rocket blast or exhaust area, use the appropriate corrosion removal and protection methods given under "Corrosion Control." If the metal involved is steel, a final paint finish is the only corrosion protection necessary.

FINAL PROTECTIVE PAINT FINISH FOR ALUMINUM OR STEEL.

For areas subjected to temperatures up to 300°F, aluminum and steel are given a final protective paint finish consisting of the following:
   - One coat pretreatment wash primer (Specification MIL-C-15328)
   - Two coats zinc-chromate primer (Specification MIL-P-8585)
   - Two coats aluminized lacquer (Specification MIL-L-7178)

For areas subjected to temperatures of 300°F to 400°F, aluminum and steel are given a final protective paint finish consisting of the following:
   - One coat pretreatment wash primer (Specification MIL-C-15328)
   - Two coats zinc-chromate primer (Specification MIL-P-8585)
   - Three coats aluminized lacquer (Specification MIL-L-7178)
   - One coat aluminized varnish (Specification MIL-V-6894)

CORROSION TREATMENT OF BATTERY COMPARTMENT.

REMOVING BATTERY ALKALI OR ACID.

1. Wash affected areas with plenty of white vinegar (Federal Specification Z-V-401B) or a 5 percent solution of water and acetic acid (Federal Specification O-A-76C), using brush dipped in the solution. Care must be taken not to splash acid solution onto surrounding surfaces.

WARNING

If alkali or acid gets into eyes or on hands, immediately flush with water and report to dispensary.

2. Rinse affected areas with plenty of clean water to dilute and remove sediment. Care must be taken not to splash water onto surrounding areas.

3. Treat affected areas with plenty of 20 percent solution of water and sodium bicarbonate (Federal Specification O-S-576B), applied with brush or cloth. Finally, allow solution to remain on surface for 5 minutes.

4. Remove sodium bicarbonate solution by rinsing with clean water and wipe dry.

CORROSION REMOVAL.

Corrosive deposits should be removed by the method governed by the type of metal involved as indicated under "Corrosion Control."

FINAL PROTECTIVE PAINT FINISH.

The affected surfaces should be clean, dry, and free from contaminants before final paint finish is applied. The paint finish should consist of the following:

- One coat pretreatment wash primer (Specification MIL-C-15328)
- Two coats zinc-chromate primer (Specification MIL-P-8585)
- Two-mil coat of alkali and acid-resistant coating (Specification MIL-C-7439)

CORROSION TREATMENT OF CONTROL CABLES.

1. Move surface controls to reveal concealed areas of cable.

2. Clean cable with cloth, using dry-cleaning solvent (Federal Specification P-S-661, Type II).

WARNING

In areas of oxygen storage and transfer system, do *not* use solvent. This material and oxygen are highly explosive.

3. If cable looks rusted, relieve tension and inspect thoroughly.

4. Replace internally rusted cables.

5. If cable shows no evidence of rust, reapply corrosion-preventive compound (Specification MIL-C-16173, Grade 4), and adjust cable to proper tension.

CORROSION TREATMENT FOR MATED SURFACES OF TWO DISSIMILAR METALS.

LAP JOINTS.

All dissimilar lap joint surfaces other than magnesium should be treated with two coats of zinc-chromate primer (Specification MIL-P-8585) on each surface. The first coat should be allowed to dry before the second coat is applied.

Where magnesium is one of the dissimilar metals, the metals should be separated by an approved barrier tape such as Federal Stock No. 8030-200-9179. The tape must extend not less than ¼ inch beyond the joint edges. (See figure 1-27.)

RIVETED AND BOLTED JOINTS.

All dissimilar metal applications involving bolt or rivet hole and fasteners, other than when magnesium is involved, should be treated as follows:

1. Clean hole surface, using an appropriate piece of No. 280 and then No. 400 abrasive grit paper. Clean outer surface surrounding hole, using rotary abrading wheel (Federal Stock No. 3460-528-3106). Wipe surface, using cloth slightly dampened with clean water, and then dry. If grit paper has not removed hole surface corrosion or if too much metal surface has been removed, replace part.

2. If material removed is within limits and the metal is aluminum, carefully treat hole surface and surrounding area with chemical films for aluminum alloys (Specification MIL-C-5541), as indicated under "Corrosion Control."

3. Apply one coat of zinc-chromate primer (Specification MIL-P-8585), corrosion-preventive petrolatum (Specification MIL-C-11796, Class 3), or corrosion-preventive compound (Specification MIL-C-16173, Grade I) to surface of hole with brush or swab.

4. In case a protective paint finish is required, apply the following to outer surface:

- One coat wash primer (Specification MIL-C-8514), 0.2 to 0.3 mil thick
- One coat zinc-chromate primer (Specification MIL-P-7962), 0.3 to 0.4 mil thick
- Two coats acrylic nitrocellulose lacquer (Specification MIL-L-19537), 0.4 to 0.5 mil per coat

Note

Use Specification MIL-P-7962 zinc-chromate primer with Federal Specification TT-T-266 lacquer thinner and Specification MIL-L-19537 acrylic nitrocellulose lacquer with Specification MIL-T-19544 lacquer thinner in the following proportions:

- Three parts by volume of Federal Specification TT-T-266 to two parts by volume of Specification MIL-P-7962.
- One part by volume of Specification MIL-T-19544 to one part by volume of Specification MIL-L-19537. (For pigmentation, add 12 ounces of Federal Specification TT-A-468, Type II, Class A, to every gallon of Specification MIL-L-19537.)

Allow zinc-chromate primer to dry at least one hour between coats and allow acrylic lacquer to dry at least 45 minutes between coats.

Dissimilar metal applications involving bolt or rivet hole and fastener, where the hole surface metal is magnesium, should be treated in the following manner:

1. Clean hole surface with an appropriate piece of No. 280 and then No. 400 grit abrasive paper. Clean outer surface surrounding hole with rotary abrading wheel (Federal Stock No. 3460-528-3106). Wipe surface with cloth slightly dampened with clean water, and then dry. If grit paper has not removed hole surface corrosion or if too much metal surface has been removed, replace part.

2. If material removed is within limits and metal is magnesium, carefully treat hole surface and surrounding area according to Specification MIL-M-3171 and as indicated under "Corrosion Control."

3. Apply sealing compound (Specification MIL-S-8802 or MIL-S-7502) to surface of hole as indicated on compound container. After rivet has been bucked or bolt has been tightened, clean excess material from area with damp cloth.

4. In case a protective paint finish is required, apply the following to outer surface:

- One coat wash primer (Specification MIL-C-8514), 0.2 to 0.3 mil thick
- Two coats zinc-chromate primer (Specification MIL-P-7962), 0.3 to 0.4 mil per coat
- Three coats acrylic nitrocellulose lacquer (Specification MIL-L-19537), 0.4 to 0.5 mil per coat

Note

Since Specifications MIL-S-7502 and MIL-S-8802 sealing compounds have a maximum application time (pot life) of 2 hours, judgment should be used on the quantity of compound to be mixed.

CORROSION LIMITS AND TREATMENT FOR DRAG CHUTE BEAM SPLICE AND VERTICAL STABILIZER REAR BEAM FITTINGS.

Corrosion removal limits and corrosion treatment for Part No. 180-23414 vertical stabilizer rear beam fitting and Part No. 192-315106-3, -4 drag chute beam splices are as follows:

1. Corrosion Removal Limits:

- Corrosion may be removed to a depth of 0.080 inches.
- For a part which has corrosion on both sides of a leg or flange, the total amount of material removed cannot exceed 0.080 inches.
- If enough material is removed from the surfaces which affect installation of the parts, the resulting fit would be loose. In this case, shim as required to restore proper fit. Additional material may be removed, maintaining prescribed limits, to allow for proper shim installation.

2. Corrosion Treatment:

- Clean parts in accordance with T.O. 1-1-1.
- Remove corrosion in accordance with T.O. 1F-100D-3, page 1-56 "Removal of Corrosion from Steel and Stainless Steel", using limits given above.
- Give part two coats of TT-P-28 (Heat Ren 1200), 8010-664-7463.

AREAS REQUIRING SPECIAL CORROSION INSPECTION.

The following table gives special areas to inspect for corrosion.

AREA	FIGURE NO.	ITEM NO.	MATERIAL	MATING ITEM NO.	FIGURE NO.	MATERIAL
Leading edge and slats	2-20	50	CRES rib assy	51	2-20	2024-T42 TE extr
				100, 101	2-20	2024-T4 clad skin
Wing box	2-24	38	7075-T6 skin		2-7	F260, F312, F375, and 69509 bolts & rivets
	2-24	33, 36	7075-T6 skin			MS20007 and MS20008 bolts
Wing trailing edge	2-44	2, 3	CRES rib assy	28	2-44	7075-T6 clad skin
		10	4140 steel hinge assy	25	2-44	7075-T6 skin
Wing flap	2-47	24	4340 steel fitting	3, 4	2-47	2024-T4 clad web 7075-T6 spar
Wing tip	2-58	18	CRES aft tip assy		2-58	7075-T4 spar
	2-58	26, 27	7075-T6 skin			NAS screws
Horizontal stabilizer	3-7	1	Magnesium TE beam	58	3-7	2024-T3 clad TE skin
	3-7	50	2024-T3 clad LE skin	49	3-7	CRES LE doubler skin
	3-7	52	7075-T6 skin	Zone 1, 2, 3	3-38	F200 and F312 Jo-bolts
	3-7	53	4340 steel beam	55	3-7	2024-T4 fairing cover
Vertical stabilizer	3-19	32	7075-T6 skin	Zone 1, 2, 3	3-38	F260 and F312 Jo-bolts
Rudder	3-33	18	356-T6 lower LE beam	21	3-33	2024-T3 Lower skin
Fuselage forward section	4-5	45	CRES skin	42, 46	4-5	2024-T4 Clad skin
	4-5	45	CRES skin	11, 15, 19, 23, 27	4-5	7075-T6 clad Lower frames

AREA	FIGURE NO.	ITEM NO.	MATERIAL	MATING ITEM NO.	FIGURE NO.	MATERIAL
Fuselage intermediate section	4-13	27	CRES canted beam	60, 61, 66, 68, 69	4-13	7075-T6 clad skin
Fuselage aft section	4-17C	5	7075-T6 skin	1, 4	4-17C	Titanium skin
	4-17C	6	7075-T6 skin	4, 7, 8	4-17C	Titanium skin
	4-17C	17	7075-T6 skin	4, 7	4-17C	Titanium skin
	4-17C	18	7075-T6 skin	7, 19	4-17C	Titanium skin
	4-17C	20	7075-T6 skin	19	4-17C	Titanium skin
	4-17C	21	7075-T6 skin	19	4-17C	Titanium skin
	4-17C	22	2024-T3 skin	9, 23	4-17	Titanium skin
Fuselage aft section	4-38	Detail A	4340 steel fuselage—vertical stabilizer fitting	32	3-19	7075-T6 skin
Fuselage aft section	4-18	16	7075-T6 clad skin	1, 4	4-18	Titanium skin
	4-18	20	7075-T6 clad skin	4, 5, 23	4-18	Titanium skin
	4-18	21	7075-T6 clad skin	23	4-18	Titanium skin
	4-18	22	7075-T6 clad skin	23	4-18	Titanium skin
	4-18	24	7075-T6 clad skin	7, 23	4-18	Titanium skin
	4-18	25	7075-T6 clad skin	7	4-18	Titanium skin
	4-18	26	2024-T4 clad skin	9	4-18	Titanium skin
	4-18	27	2024-T4 clad drag chute door	9, 15	4-18	Titanium skin
	4-18	39	6061-O shroud	34, 35, 36, 37, 38	4-18	Titanium angles, channels, and stiffeners
Fuselage aft section	4-18A	7	2024-T42 clad door	5, 20, 21	4-18A	CRES skin
	4-18A	16	7075-T6 clad skin	1, 4 5	4-18A 4-18A	Titanium skin CRES skin
	4-18A	17	7075-T6 clad skin	20, 21	4-18A	CRES skin
	4-18A	19	7075-T6 clad skin	21	4-18A	CRES skin
	4-18A	22	7-75-T6 door	21	4-18A	CRES skin
Fuselage aft section	4-18A	23	2024-T4 clad door	9	4-18A	Titanium skin
	4-18A	24	2024-T4 clad skin	9, 15	4-18A	Titanium skin

In case corrosion is detected, materials should be treated as indicated under "Corrosion Control" and in "Lap Joints" and "Riveted and Bolted Joints."

For additional information, refer to T.O. 1-1-1 and 1-1-2, dated 15 May 1961 and subsequent.

CORROSION CONTROL MATERIALS.

FEDERAL STOCK NO.	NOMENCLATURE	SPECIFICATION	SOURCE	APPLICATION
	CLEANING MATERIALS AND EQUIPMENT			
7930-543-4552	Alkaline water base cleaner	MIL-C-25769	AF stock	Cleaning
6810-290-0053	Dry-cleaning solvent	Fed. Spec P-S-661B	Local purchase	Cleaning
7920-281-7009	Nonmetallic brush	MIL-B-5612	Local purchase	Cleaning
9140-242-6750	Kerosene	Fed. Spec VV-K-211C	Local purchase	Cleaning
6810-285-4318	Trichloroethylene	MIL-T-7003	AF stock	Cleaning copper
6810-223-2737	Dichloromethane	MIL-D-6998	AF stock	Cleaning copper
6810-281-2762	Methyl ethyl ketone	Fed. Spec TT-M-261	AF stock	Cleaning metal
8010-160-5787	Lacquer thinner	Fed. Spec TT-T-266	Local purchase	Cleaning metal
4940-567-9558	Portable air pressure spray cleaner, 20-gallon	—	AF stock	Applying cleaner or acid
6850-527-2426	Metal conditioner and brightener	MIL-C-25378	Local purchase	Cleaning clad aluminum alloys
7510-266-6710	Masking tape	—	Local purchase	Protecting surface while cleaning
8340-262-3679	Vinyl plastic tarpaulin	—	AF stock	Protecting landing gear tires and wheels

FEDERAL STOCK NO.	NOMENCLATURE	SPECIFICATION	SOURCE	APPLICATION
	CORROSION REMOVERS AND EQUIPMENT			
6850-527-2426	Metal conditioner and brightener	MIL-C-25378	Local purchase	Removing corrosion from aluminum
5350-286-4851	Aluminum wool	MIL-A-4864	Local purchase	Removing corrosion from aluminum
7920 N/L	Aluminum brush	—	Local purchase	Removing corrosion from aluminum
7920 N/L	Stainless steel brush	—	Local purchase	Removing corrosion from steel
5350-721-8119	No. 280 emery cloth	—	Local purchase	Removing corrosion from bare aluminum and magnesium
5350-721-8118	No. 400 emery cloth	—	Local purchase	Removing corrosion from bare aluminum and magnesium
5350-186-8812	No. 500 emery cloth	—	Local purchase	Removing corrosion from bare aluminum and magnesium
6810-587-5775	Chromic acid	Fed. Spec O-C-303	Local purchase	Removing corrosion from magnesium
5350-242-4405	Steel wool	—	Local purchase	Removing corrosion from steel
6850-201-1218	Phosphoric acid	MIL-M-10578B	Local purchase	Removing corrosion from steel
7920-633-9911	Sponge cellulose	—	Local purchase	Rinsing
7240-125-9060	Rubber pail, 3-gallon	—	Local purchase	Used with acid
3460-528-3106	1½-inch abrading wheel (soft)	—	AF stock	Removing corrosion
3460-528-3105	2-inch abrading wheel (soft)	—	AF stock	Removing corrosion
3460-528-3102	2-inch abrading wheel (tough)	—	AF stock	Removing corrosion
	PAINT REMOVERS			
8010-165-5535	Paint remover	MIL-R-25134	AF stock	Removing paint
8010-165-5536	Paint remover	MIL-R-25134	AF stock	Removing paint
6810-209-8091	Ethyl acetate	Fed. Spec TT-E-751	AF stock	Removing paint
6810-209-8093	Ethyl acetate	Fed. Spec TT-E-751	AF stock	Removing paint
7920-281-7009	Fiber brush	MIL-B-5612	Local purchase	Removing paint
	CORROSION-PROTECTIVE TREATMENT			
8030-099-1052	Chemical films for aluminum alloys	MIL-C-5541	Local purchase	Protecting aluminum
6810-281-2686	Sodium dichromate	Fed. Spec O-S-595a	Local purchase	Protecting magnesium
6810-N/L	Sodium acid fluoride	—	Local purchase	Protecting magnesium
6810-N/L	Potassium acid fluoride	—	Local purchase	Protecting magnesium
6810-N/L	Ammonium acid fluoride	—	Local purchase	Protecting magnesium
8030-231-2354	Corrosion-preventive petrolatum	MIL-C-11796A	Local purchase	Protecting metal
8030-526-1605	Corrosion-preventive compound	MIL-C-16173B	Local purchase	Protecting metal
6810-271-1410	Nitric acid (specific gravity 1.42)	—	Local purchase	Protecting magnesium

FEDERAL STOCK NO.	NOMENCLATURE	SPECIFICATION	SOURCE	APPLICATION
8030-200-9178	Zinc-chromate barrier tape	—	AF stock	Protecting dissimilar metals
8030-200-9179	Zinc-chromate barrier tape	—	AF stock	Protecting dissimilar metals
		PAINTS AND FINISHES		
8010-246-0872	Zinc-chromate primer	MIL-P-8585A	AF stock	Protecting metals
8010-165-8626	Zinc-chromate primer	MIL-P-8585A	AF stock	Protecting metals
8010-597-7864	Gray enamel	MIL-E-7729	AF stock	Additional protection
8010-241-8406	Aluminized lacquer	MIL-L-7178	AF stock	Additional protection
8010-168-8810	Aluminized varnish	MIL-V-6894	AF stock	Additional protection
8010-221-2777	Fuel-resistant lacquer	MIL-L-6047	AF stock	Protecting inside fuel tanks
8010-221-2773	Alkali- and acid-resistant lacquer	Fed. Spec TT-L-54	AF stock	Protecting battery compartments
8030-506-4030	Elastomeric coating	MIL-C-7439	Local purchase	Protecting battery compartments
8010-664-4966	Wash primer	MIL-C-15328A	AF stock	Pretreat coating
8010-634-7324	Vinyl topcoat	MIL-P-15936	AF stock	Special area finish
8010-664-0018	Vinyl primer	MIL-P-15930A	AF stock	Special area finish
8030-597-0892	Wash primer	MIL-C-8514	AF stock	Pretreat coating
8010-526-2523	Zinc-chromate primer	MIL-P-7962	AF stock	Protecting metals
8010-527-1508	Acrylic nitrocellulose lacquer	MIL-L-19537	AF stock	Final paint finish
8010-527-2897	Lacquer thinner	MIL-T-19544	AF stock	Thin lacquer
8010-687-4019	Aluminum pigment	Fed. Spec TT-A-468	AF stock	Pigmenting clear lacquer
		PERSONNEL EQUIPMENT		
8405-265-6058	Vinyl-coated fiber glass apron, Type C1-A	—	AF stock	Protecting personnel
8405-268-8003	Rubberized coveralls	—	AF stock	Protecting personnel
8405-268-8049	Wet-weather trousers	—	AF stock	Protecting personnel
8430-147-1022	Boots, size 9	—	AF stock	Protecting personnel
8430-160-8587	Boots, size 7	—	AF stock	Protecting personnel
8415-634-6522	Oil- and chemical-resistant gloves	—	AF stock	Protecting personnel
4240-278-9646	Plastic safety goggles	—	AF stock	Protecting personnel
		MISCELLANEOUS MATERIALS AND EQUIPMENT		
8030-723-2746	Sealing compound	MIL-S-8802	AF stock	Coating lap surfaces and holes
8030-270-8144	Sealing compound	MIL-S-7502	AF stock	Coating lap surfaces and holes
8030-243-3285	Antiseize compound	MIL-T-5544	AF stock	Coating bolt threads
4920-831-5932	Metal identification kit USAF Dwg 61B25042	—	AF stock	Identifying metals and surface finishes
8950-223-5849	White vinegar	Fed. Spec Z-V-401B	Local purchase	Neutralizing alkali
6810-275-1215	Acetic acid	Fed. Spec O-A-76C	Local purchase	Neutralizing alkali
6810-237-2914	Sodium bicarbonate	Fed. Spec O-S-576B	AF stock	Neutralizing acid

CORROSION BARRIERS.

CORROSION- AND ABRASION-RESISTANT PLASTIC COATING.

This is a material which may serve both as a corrosion barrier and as an abrasion resistant coating. The material may either be sprayed or brushed. The method used is determined by ease of application. It is furnished in a kit consisting of two parts, a base material, and curing agent. The plastic material may be applied over metal pretreatments such as anodizing, chemical film treatment for steel, cadmium plating, and protective treatments for magnesium.

Note

Zinc-chromate primed surfaces must be stripped of the primer before the plastic material is applied.

MATERIAL	SOURCE
Coating—plastic, corrosion- and abrasion-resistant, Everlube 1329 (black), Everlube 1329A (aluminized)	Everlube Corp
Tape—vinyl No. 473	Minnesota Mining & Mfg Co
Naphtha	Federal Spec TT-N-95
Toluol	Federal Spec TT-T-548
Methyl isobutyl ketone	Federal Spec TT-M-268
Butyl Cellosolve	Commercial
Wash primer	Specification MIL-C-8514

Refer to Appendix I for sources of commercial products.

PREPARATION FOR APPLICATION OF CORROSION- AND ABRASION-RESISTANT PLASTIC COATING.

All metal surfaces to which the coating is to be applied must have oil, grease, dirt, etc, removed. Wipe surfaces with a cloth saturated in solvent and then wipe off solvent with a clean dry cloth. Spray one thin wet coat of wash primer onto the cleaned surface, and allow it to dry one to four hours before applying the plastic coating.

MIXING CORROSION- AND ABRASION-RESISTANT PLASTIC COATING.

Immediately before use, the base material and the curing agent should be mixed together and thoroughly stirred. The mixed material has a pot life of about 4 hours. Therefore, only the amount which is to be used should be prepared. Generally, the properly mixed material is of the correct consistency for spray or brush application. If additional thinning is necessary, add a small amount (1 to 5 percent) of the following mixture.

45 percent by volume—Methyl isobutyl ketone
5 per cent by volume—Butyl Cellosolve
50 percent by volume—Toluol (Federal Spec TT-T-548)

APPLICATION OF CORROSION- AND ABRASION-RESISTANT PLASTIC COATING.

The material may be applied by brushing or spraying successive coats, allowing an adequate drying time between coats to prevent sagging. Normally, two passes over the surface with a spray gun will provide a 0.003 inch thick film. A film 0.003 to 0.005 inch thick is adequate for abrasion and corrosion protection in most cases.

The applied material may be completely cured by air drying seven days at room temperature, or by baking. Air-cured films will be dry to handle in 6 to 16 hours. Films which are to be baked must be allowed to air dry at least one hour, then baked for 10 minutes at 250°F.

STRIPPING PROCEDURE FOR CORROSION- AND ABRASION-RESISTANT PLASTIC COATING.

Refinishing the plastic coating that has been subjected to high abrasion is at times necessary. The section between the flap rubbing strip and fuselage skin represents an area of extreme wear; therefore, removal of the old coating is recommended before refinishing.

Materials required for the stripping procedure are as follows:

Remover—LAC1981C noncresylic, Type I (for spray application)
Remover—713C, Type II (for brush-on application)

Refer to Appendix I for sources of commercial products.

1. Mask off all plastic (both transparent and nontransparent), rubber, and fabric surfaces. Mask off all external and aerodynamically smoothed joints. Make certain that all access doors are tightly closed; mask off completely all holes of any sort where there is a possibility of remover getting on interior surfaces or other areas not to be stripped. This is particularly important when remover is to be spray-applied. The remover must not get on any type of lubricated bearings, continuous hinges, electrical equipment, or wiring.

2. Thoroughly stir or agitate remover.

3. Apply a heavy coat of remover with a nonatomizing spray nozzle and a simple pressure tank, or by brushing. For efficient removal, a wet film of remover should be kept on the surface being stripped. This may require additional applications of the remover. Progressive areas should be treated and kept wet, and sufficient time should be allowed for stripping action, which is usually 10 to 20 minutes. In no case should the spray be highly atomized. Avoid spattering and run-down of remover.

3. Remove loosened coating with a stiff-bristle brush, or equivalent, or by pressure-spraying with water. For most efficient removal, start hosing at the bottom and work up. Small quantities of water tend to set up the stripper.

4. Repeat steps 2 and 3 where coating has not been completely removed.

5. Rinse area thoroughly with clean water, preferably by water spray. Be sure all remover is removed, particularly along joint lines, around rivets, and in recesses.

6. Remove all masking materials. Dry areas by wiping them with clean cloth or by air blasts.

CAUTION

Rubber or synthetic rubber gloves, eyeglass-type goggles, apron, and rubber boots should be worn during stripping operations. Avoid excessive breathing of the vapors. If any of the remover accidentally contacts skin, wash off immediately with a diluted solution of alcohol; then follow with an application of glycerin or petrolatum (vaseline). Treat eyes with pure water only.

VINYL TAPE AS A CORROSION BARRIER.

The use of vinyl tape No. 473 is recommended as a corrosion barrier between magnesium and other metals. Tape is to be used in addition to the paint coats required. Tape should not be used between press-fitted surfaces, nor as a corrosion barrier between dissimilar metals on mechanical, hydraulic, or electrical assemblies where the tape could cause faulty operation of the part. When one part is removable, the tape should be applied to the fixed member.

SURFACE PREPARATION FOR VINYL TAPE AS A CORROSION BARRIER.

All sharp corners or edges of mating surfaces must be smooth to prevent cutting through the tape.

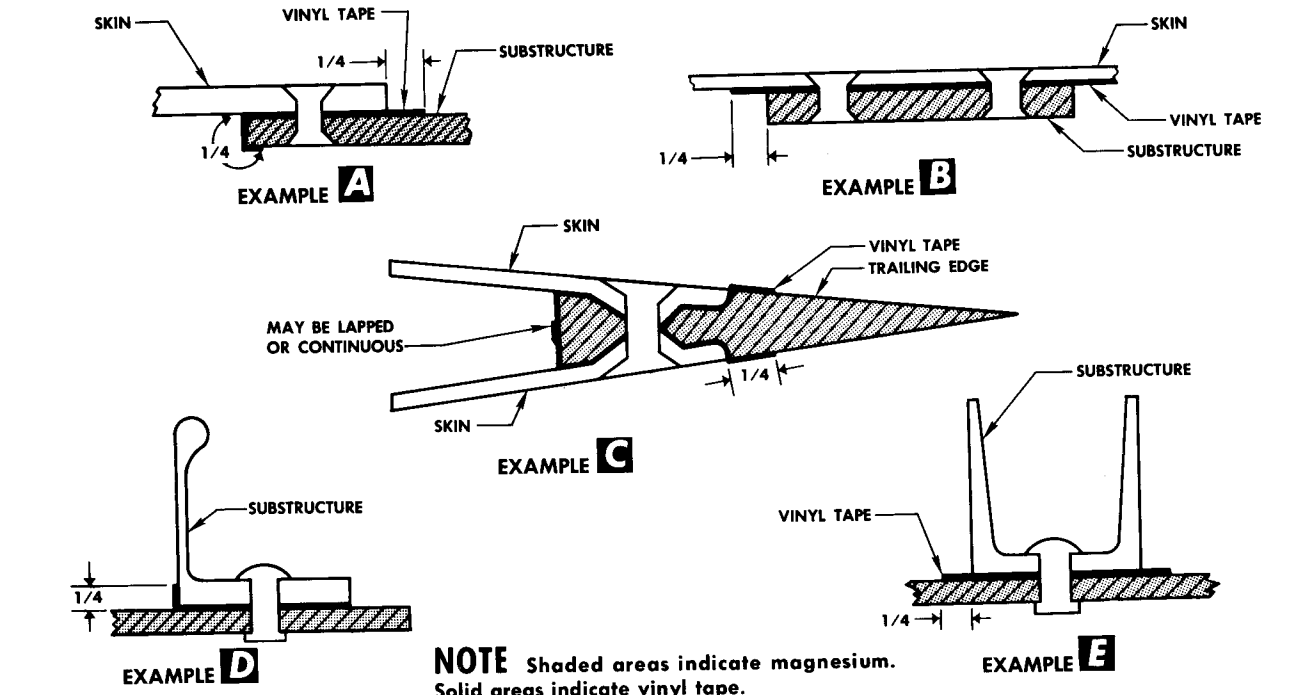
All operations on the mating surfaces such as drilling, dimpling, chemical treating, etc, must be complete. Immediately before application of tape, all mating surfaces must be carefully wiped with a clean cloth saturated in naphtha (Federal Specification TT-N-95) or other suitable solvent.

APPLICATION OF VINYL TAPE AS A CORROSION BARRIER.

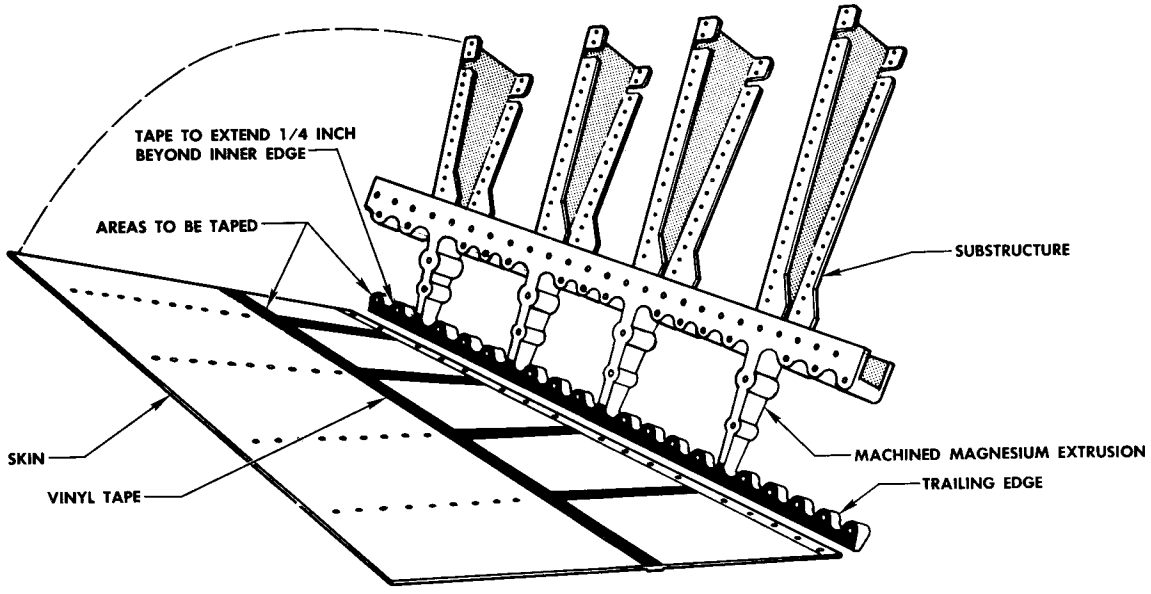
If magnesium is one of the dissimilar metals, the tape should be applied carefully to the magnesium surface. If this is not convenient, the tape may be applied to the other metal. The tape should be wide enough and long enough to extend ¼-inch beyond all edges of the mating surfaces. (See figure 1-27.) The portion of the tape extending beyond the edges of the mating surfaces should be folded as shown in details "A" and "D" of figure 1-27. On simple extrusions the tape should be wide enough to cover the entire interior surface of the extrusion and extend ¼-inch beyond the mating surfaces. On machined magnesium extrusions such as the trailing edge beam of the horizontal and vertical stabilizers, the tape shall be applied to the extrusion and the attaching skins. Extreme precautions should be taken to see that all tape covering air vent and water drain holes is removed. The tape

VINYL TAPE CORROSION BARRIER

GENERAL



NOTE Shaded areas indicate magnesium.
Solid areas indicate vinyl tape.



NOTE
Drain and vent holes not covered.

GEN-3-00-8

Figure 1-27. Corrosion Barriers

should be applied in one continuous length. If splicing the tape is necessary, the laps shall be at least 1/4-inch wide but not more than 5/16-inch wide. Lapping of tape over any attaching hole must be avoided.

Before any fasteners are installed, the tape over the attaching holes should be pierced with a sharp awl or similar tool. During this operation, be careful not to tear the tape beyond the limits of the hole. When placing the parts into position for fastening or attaching, extreme care should be taken to prevent damage to the tape. Corrosion protection is provided only if the tape is applied to form a continuous unbroken film.

SEALING COMPOUND AS A CORROSION BARRIER.

Sealing compound (PR-1221) as a corrosion barrier may be used where the use of vinyl tape is impracticable.

Note

Sealing compound shall not be used as a corrosion barrier on movable control surfaces. Tape only shall be used on such surfaces. When sealing compound is used as a corrosion barrier, it shall be used in addition to the paint coats required.

Sealing compound should not be used between press-fitted surfaces, or on mechanical, hydraulic, or electrical assemblies where the sealing compound could cause faulty operation of the part.

SURFACE PREPARATION FOR SEALING COMPOUND AS A CORROSION BARRIER.

All sharp corners or edges of mating surfaces must be smooth to avoid damage to the film of sealing compound. All operations, such as drilling, dimpling, chemical treatments, etc, must be complete. Immediately the sealing compound is applied, all mating surfaces must be carefully wiped with a clean cloth saturated in naphtha (Federal Specification TT-N-95) or other suitable solvent. In primed areas, primer adhesion must be tested by scraping it with the thumb nail. If the primer can be removed in this manner, the area to which the sealing compound is to be applied must be stripped.

APPLICATION OF SEALING COMPOUND AS A CORROSION BARRIER.

Sealing compound may be applied with a brush, spatula, or special applicators. When attachment of the metals receiving the film of sealing compound is made by bolts, screws, or quick-disconnect fasteners, the sealing compound must be applied to the stationary part. The removable part including fasteners must be coated with petrolatum (AN-P-51) to prevent the sealing compound from sticking.

Immediately after the sealing compound is applied, assemble the part and let cure for 24 hours. Remove excess petrolatum from the exterior of the part with a cloth saturated in naphtha. After the sealing compound has cured, the part should be painted if required.

FIRE DAMAGE.

CORROSIVE EFFECTS OF AIRCRAFT FIRE-EXTINGUISHING AGENTS AND METHODS OF REMOVAL.

CORROSIVE EFFECTS AND REMOVAL OF DRY CHEMICAL POWDER.

This type of extinguisher uses a specially prepared grade of sodium bicarbonate as the extinguishing agent. Heat causes the bicarbonate to decompose, releasing carbon dioxide gas and water vapor to smother the fire. A powdery residue of sodium carbonate (soda ash) will usually remain on and around the heat-affected area.

Dry sodium bicarbonate is nonabrasive and noncorrosive. In water solution, it will cause mild corrosion of most metals by either electrolytic or chemical means. However, its rate of corrosion is so slow that it can be cleaned up with little chance of damaging delicate parts. The decomposition product, sodium carbonate, will cause little damage if kept dry, but in solution with water is strongly caustic and will rapidly attack many common aircraft materials. For this reason, when water is used to clean up after a fire, thorough rinsing is absolutely necessary to eliminate any pockets of strong carbonate solution from the airplane. Care should be taken to prevent sodium carbonate dust from entering the eyes, nose, or mouth during the following cleanup procedure.

In areas of the airplane which are free of electrical equipment, or where it is hermetically sealed, such as in wheel wells, blow any powder residue away with compressed air. Wash the area thoroughly with fresh water and then dry with clean rags and compressed air. When nonsealed electrical and electronic equipment is present, the decision to use water should be determined by how readily this equipment can be removed or dried. (Large bundles of wire will have to be loosened and blown dry.) If it is decided that water would be harmful to any of the airplane equip-

ment, a satisfactory removal can be accomplished by a combination of brushing, vacuum cleaning, and careful blowing with compressed air.

When the powdery residue sticks to hydraulic oil or grease, it should be removed with the solvent approved for removing the oil or grease alone.

CORROSIVE EFFECTS AND REMOVAL OF MECHANICAL FOAM.

This fire-extinguishing agent is harmless to all materials not damageable by water alone. It is normally used on major fires that will damage the airplane extensively. The foam may be removed with a high-velocity jet of fresh water and scrubbing with a fibre brush. Dry the airplane thoroughly with clean rags and compressed air. Remove all affected, nonsealed electrical and electronic components immediately. Dry and prepare them for overhaul in accordance with current directives.

CORROSIVE EFFECTS AND REMOVAL OF OTHER EXTINGUISHING AGENTS.

Carbon dioxide and carbon tetrachloride evaporate rapidly and normally require no cleanup. Chlorobromomethane is highly corrosive. It evaporates rapidly, but if any of the substance is trapped in pockets it must be cleaned out.

FIRE DAMAGE EVALUATION.

Three methods are used in checking airplane structures after a fire: visual inspection for buckles, canning and primer discoloration; hardness testing of questionable parts with portable or stationary hardness tester; and tension testing of specimens taken from areas of doubtful valves.

In visually inspecting an airplane, the search for buckles or canning is primarily concerned with the possibly weakened condition of primary internal structures. This is especially important when the fire has been in the wing area. It is possible that an area unaffected by the fire directly may have been bent enough to receive a permanent set.

Large (or thick) aluminum parts, such as forgings, are not as quickly affected by heat as are thinner and smaller components. However, when heat has been great enough to injure sheet and extruded parts, all heavy fittings in the area should be removed and hardness tested. Similar parts from unaffected areas may be used for a comparison. Figure 1-28 shows the heat effect on three 7075-T6 forgings taken from a fire area and compared with similar parts from the opposite side of the airplane. Tests were made with a Riehle portable hardness tester.

It is often necessary to remove sample disks from sheet metal parts in order to test them on a bench-type hardness tester. Disks for this purpose should be removed with a one-inch hole saw. If the part is to be retained in use, make repairs as directed in the applicable section of this handbook.

Access hole covers in the fire area are excellent samples for hardness sampling. They also provide an easily replaceable part for destructive testing.

When tension bars from sheet metal parts are taken for laboratory testing, the blanks removed should measure 1 X 7 inches. However, pieces measuring 1 X 5 inches are acceptable. They should be cut with the long edges in line with the grain whenever possible. When parts or samples are sent to a laboratory for analysis, information as to the type and duration of the fire should be included.

HARDNESS TEST OF 7075-T6 WING FORGING — GENERAL

LOCATION	MATERIAL	* ROCKWELL HARDNESS
LH (NORMAL) RH (HEATED)	LARGE FORGING LARGE FORGING	B89 B86
LH (NORMAL) RH (HEADED)	MEDIUM FORGING MEDIUM FORGING	B89 B82
LH (NORMAL) RH (HEATED)	SMALL FORGING SMALL FORGING	B89 B59

*Using a Riehle portable hardness tester

GEN-3-00-9

Figure 1-28. Hardness Tests—7075-T6 Wing Forgings Of One Fire-damaged Airplane

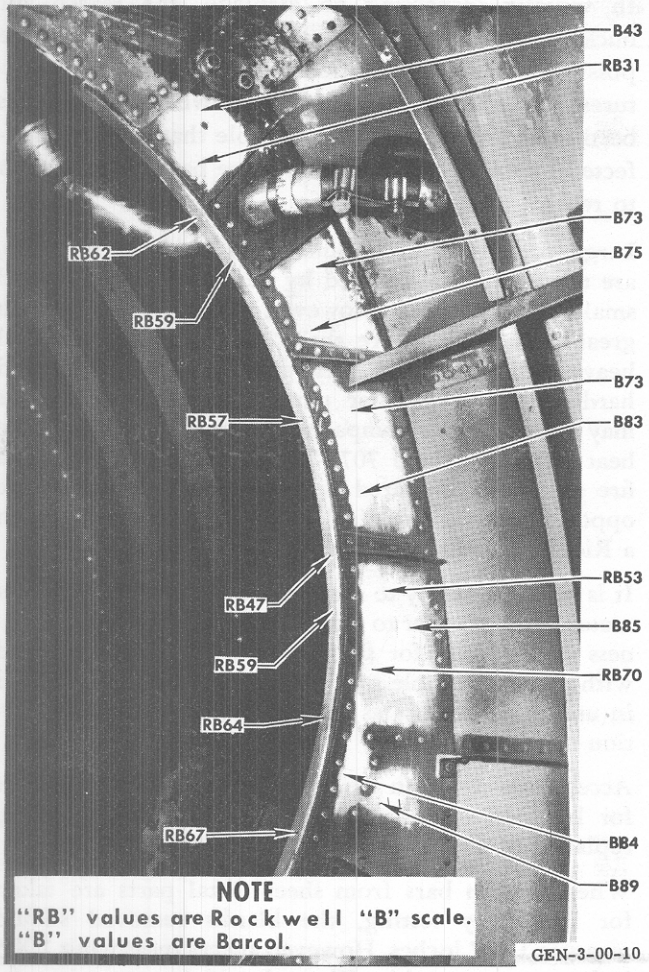


Figure 1-29. Hardness Tests Of Fire-damaged Frame

PRIMER DISCOLORATION EVALUATION.

Determining the extent of fire damage by zinc chromate primer comparison can be reliable when used carefully. A clean sample of the primer known to be unaffected by heat is compared with chromate in the heated area. The sample should be from the same general area of the structure in question. The suspected area must be free of dirt, oil, and smoke.

Untinted chromate primer is a slightly greenish yellow. When exposed to about 300 degrees of heat for a few minutes, it turns a light tan color. Temperatures which discolor chromate primers will change the physical properties of 2024-T4 and 7075-T6. The 7075 which is already in the T6, or artificially aged, condition will be softened and weakened. The 2024-T4 will be artificially aged and hardened, until the temperature reaches 500 degrees; then, with increased heat, this metal also softens and weakens. When the temperature of 2024-T4 is raised to 300 to 500 degrees for a short period under uncontrolled conditions, as is the case in an airplane fire, the inherent corrosion-resisting properties are seriously affected.

Both 2024-T4 and 7075-T6 are of questionable value and should be subjected to more accurate methods of testing when the slightest discoloration of the primer is evident.

Reliable limits of correlation between hardness tests and tensile tests have been established for 7075-T6, using hardness testers. Material which gives readings of 84 to 94 in the Rockwell "B" scale is normal. Material which falls below the minimum should be replaced, or specimens should be removed from the areas with the lowest Rockwell readings and sent to a laboratory for tension tests.

Hardness test readings are not accurate indications of heat damage on 2024-T4 because of the corrosion factor. A quick heating to 300 degrees may not raise the hardness number above normal but will affect the corrosion resistance. Therefore, if chromate primer discoloration indicates heat has affected the material, it should be replaced or a specimen removed and sent to a laboratory for tension and corrosion-susceptibility tests.

If a decision is made to retain in use a section of 2024-T4 which has undergone slight mechanical change due to fire, every effort should be made to protect the metal. Painted areas should be stripped, treated with phosphoric cleaner, Iridite No. 14, and repainted. Surfaces not normally painted should be treated with phosphoric cleaner.

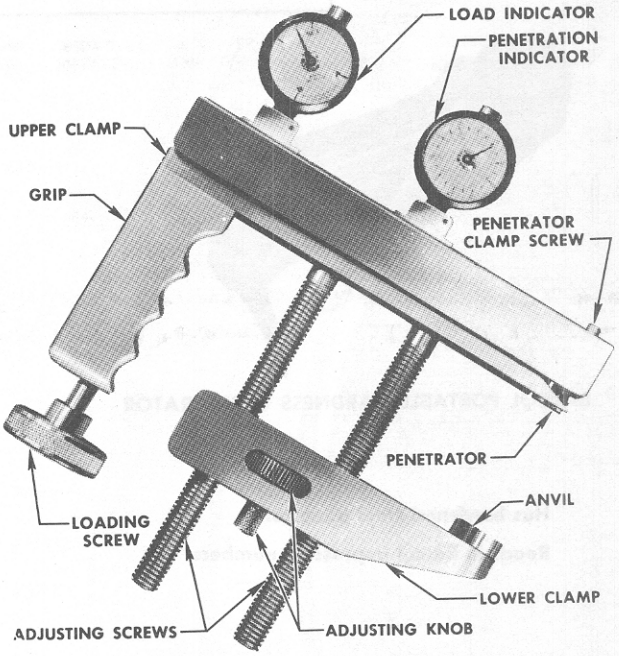
PORTABLE HARDNESS TESTING OF ALUMINUM ALLOYS.

Material to be tested must be free of paint and foreign material. Anodizing, which is hard, and cladding, which is soft, must be removed at the point of penetrator contact.

Portable harness testers (figure 1-30) are of two types. The first grips the material like a clamp. The second is pressed onto the material from one side. The Riehle tester is of the first type; the Barcol and Ernst testers, the second. The Riehle portable hardness tester is supplied with a diamond penetrator and a 1/16-inch ball penetrator and reads in Rockwell "C," "A," "B," and "F" scales. The standard anvils (flat, "V," and convex) may be supplemented with special-purpose anvils of local manufacture. For use over flanges on relatively thin material, anvils up to 4 inches long have been successfully used. One-inch diameter, cold-rolled steel rod is a satisfactory material for anvils. Care must be exercised in machining to ensure that the face of the anvil will be perpendicular to the penetrator, and that the face of the anvil and shoulders on which it rests are parallel.

RIEHLE PORTABLE HARDNESS TESTER

GENERAL



1 Select penetrator to be used. Use diamond penetrator for hardened or heat-treated steel. Use ball penetrator for nonferrous materials, or materials softer than C-20.

2 Select proper anvil to suit nature of piece.

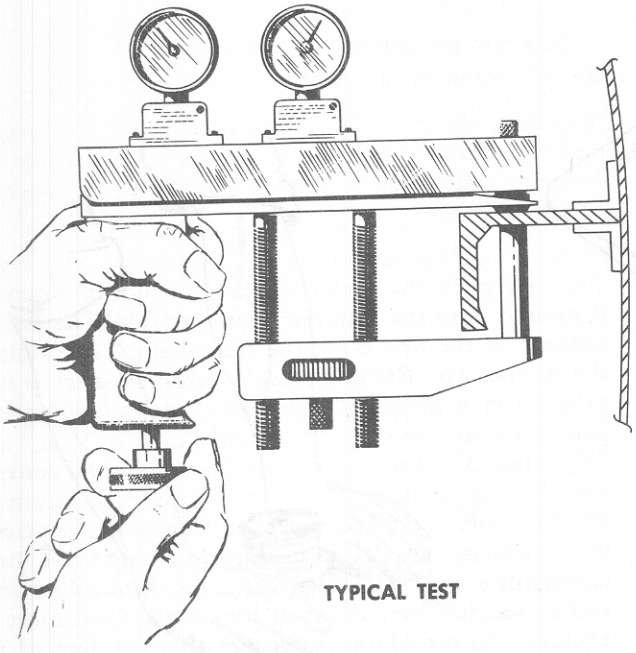
3 Make sure loading screw is backed off sufficiently so that penetrator does not project beyond upper clamp.

4 Clamp work firmly between upper clamp and anvil by means of adjusting knob on lower clamp. DO NOT FORCE.

5 Check zero setting of load indicator. Rotate bezel to bring pointer over small black dot for zero load.

6 Apply 10 kg minor load (pointer to SET) by means of loading screw.

Caution Hold instrument steady to avoid twisting motion of penetrator on the work.



TYPICAL TEST

7 Check zero setting of penetration indicator. Rotate bezel to bring pointer to 0 on the black scale.

8 Apply major load [normally, "C" scale (150 kg) or "A" scale (60 kg) for diamond, or "B" scale (100 kg) for ball].

9 Take off major load—reduce to minor load (pointer back to SET).

10 Hardness is indicated on penetration indicator. Read black figures when using diamond penetrator. Read red figures when using ball penetrator.

11 Release load completely by backing off loading screw.

12 Turn adjusting knob to left to loosen lower clamp. For more detailed information covering the use, operation, and maintenance, refer to separate instruction folder supplied with this instrument.

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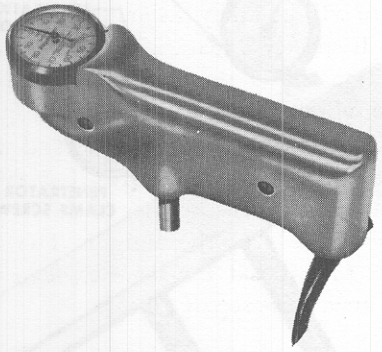
Figure 1-30. Portable Hardness Testers (Sheet 1 of 2)

PORTABLE HARDNESS TESTERS

GENERAL

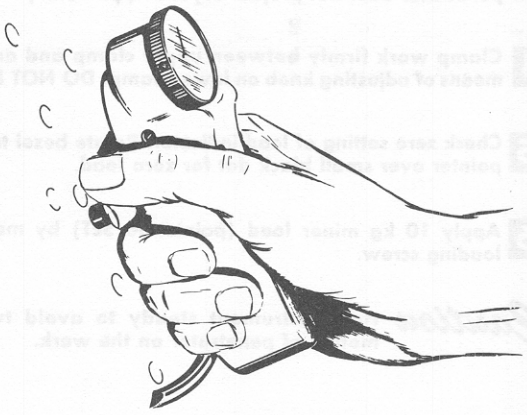


BARCOL PORTABLE HARDNESS COMPARATOR

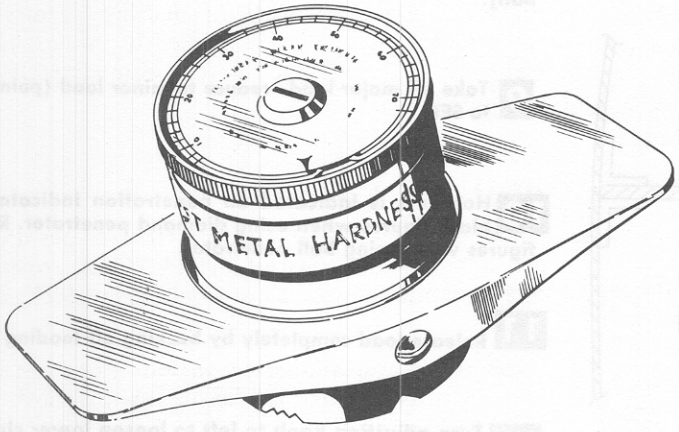
Has hardened-steel penetrator.

Reads in Barcol impress or numbers.

Material must be solidly supported from behind. DO NOT ALLOW TESTER TO SLIDE ON MATERIAL.



ERNST PORTABLE HARDNESS TESTER



Has diamond-tipped penetrator.

Reads in Rockwell or Brinell scales.

Material must be solidly supported from behind. Press down with a steady, even force.

GEN-3-00-13

Figure 1-30. Portable Hardness Testers (Sheet 2 of 2)

Three test blocks are supplied with the tester and should be used frequently to check the performance of the instrument. It is advisable to run a test on a hardness block between each series of tests, and at any time when hardness readings are not normal.

The Barcol hardness comparator is a small, lightweight instrument, suitable for comparative checks within a given sheet of metal or on an object formed from one sheet of metal. It should be used in conjunction with a bench-type tester or with a Riehle tester or equivalent. Edge readings are taken with the Rockwell, and the same area is checked with the Barcol. The remainder of the sheet, or part, may be tested by using the lowest Barcol reading from the sample area as a minimum value.

The Barcol comparator is not recommended for reliable testing of 7075-T6, since the -T6 hardness value is in the upper range of the tester, however, it can be used to advantage, in the inspection of 7075-T6 structure which is suspected of damage due to heat. Tests are taken of the part or sheet in question and, at the point of lowest Barcol reading, a one-inch disk is cut and tested on a bench-type tester. (Riehle or equivalent may be used if bench tester is not available.) The Barcol must be used carefully to avoid any bending of the fine steel penetrator. It must rest firmly on the material, and the pressure applied must be in line with the penetrator.

Frequent tests to ensure that the tester is reliable should be made on the test block provided.

The Ernst portable hardness tester is a small, versatile instrument which requires access to only one side of the material tested. It has a diamond-point penetrator and reads in Rockwell or Brinell scales, depending on the model. To ensure accurate readings, the handgrips must be pressed downward with a steady, even force until the fluid column has stopped moving. The final position of the fluid shows the hardness value directly on the scale.

As with other portable testers of this type, the material being tested must be very smooth and firmly backed up so there will be no tendency to sag under the load applied on the tester. The test block supplied with each tester should be used frequently to check the performance.

Figure 1-31 shows a tabular correlation of primer color, Rockwell harness (clad removed), and ultimate tensile strength of 7075-T6 samples taken from a fire-damaged airplane. It should be noted that tests were made on a relatively small number of specimens, and the limits on the scatter band shown are not necessarily those which could be found with a greater number of samples.

SHEET METAL THICKNESS CHANGE.

The services and aircraft industry have effected a new thickness standard for aircraft sheet metal materials. During the transition period from mill standard thicknesses and gage numbers to the new standard for all sheet metals there will be many instances when the material call-out will specify a material not in stock. The table in figure 1-32 shows the old material thickness and the new one which has replaced it. Reading across the chart, the thicknesses, except those marked with an asterisk, may be used interchangeably. If a thickness is marked with an asterisk, the heavier of the two materials may be substituted for the lighter. The lighter material may be substituted for the heavier only on approval by the structural repair officer.

Examples:

Material call-out	Steel, 4130, 0.040
Substitute	Steel, 4130, 0.042
Material call-out	Steel, 4130, 0.042
Substitute (on approval by the structural repair officer)	Steel, 4130, 0.040

The primary structural materials used in the construction of the major portion of the airplane are 2024-T and 7075-T in the clad state. 2024-T is a standard, general-purpose alloy. 7075-T is a high-strength aluminum which has physical properties that are, in certain applications, superior to 2024-T. It is used only where 2024-T is not suitable. It is more brittle than 2024-T and presents a greater problem in forming. These aluminum alloys, used in the primary structure, are heat-treatable and are in a treated condition as indicated by the "T." Materials used in making repairs and in replacement parts may be formed in the soft condition but must be heat-treated before being installed in the airplane.

ALUMINUM ALLOYS.

There are four basic aluminum alloy shapes used in the construction and repair of aircraft. These are extrusions or roll formed sections, bar stock, sheet stock, and plate stock.

Sheet stock is available in various gages up to 0.250 inch in thickness. It may be obtained in both the clad and the bare state. Gages heavier than 0.250 are plate stock. Bar stock is a material from which parts may be machined. Shapes made from bar stock are generally not quite as strong as extruded shapes of the same alloy. Therefore the cross-sectional area of a repair part machined from bar stock should be about 10 per cent greater than that of the original part to compensate for this lower strength factor. Extrusions are die-formed shapes that can be made to any specific cross-sectional area. (Refer to Section VIII.)

COLOR-HARDNESS-STRENGTH CORRELATIONS

GENERAL

SAMPLES FROM ONE FIRE-DAMAGED AIRPLANE

	ROCKWELL HARDNESS	PRIMER COLOR	ULTIMATE TENSILE STRENGTH	YIELD STRENGTH	ELONGATION PERCENT IN 2 INCHES
1	B38-40	BLACK	43,000 43,400	23,100 17,950	11 12
2	B38-40	BLACK	47,700 47,100	24,000 24,600	11 12
3	B52-56	BLACK	53,800 54,200	27,500 27,100	16.5 16
4	B-57	BLACK	54,000 54,300	27,500 28,600	14.5 15
5	B61-64	BROWN	56,100 56,500	41,700 43,300	6.5 7.5
6	B78-80 B68-72	TAN BROWN	64,500 61,600	51,600 47,700	6 6
7	B80-82	YELLOW (NORMAL)	74,800 74,100	63,400 65,400	13.5 12
Minimum values for 7075-T6 Federal Spec QQ-A-287			72,000	62,000	8

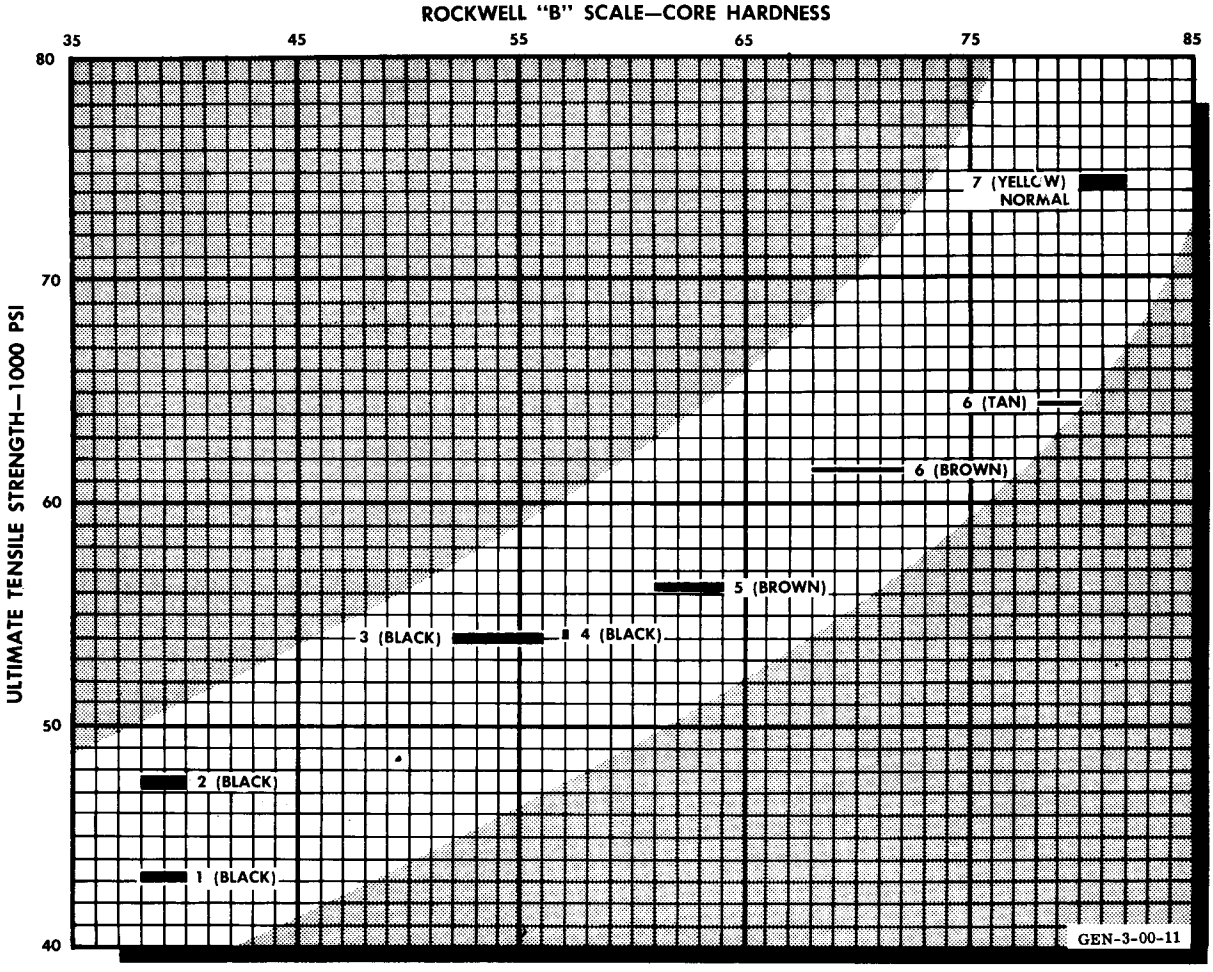


Figure 1-31. Color-Hardness-Strength Correlations—7075-T6

STANDARD SHEET METAL THICKNESS

GENERAL

ALUMINUM AND MAGNESIUM		STEEL	
OLD STANDARD	NEW STANDARD	OLD STANDARD	NEW STANDARD
	0.006	0.010	0.010
	0.008	0.012	0.012
0.010	0.010	0.016	0.016
0.012	0.012	0.018	0.018
0.016	0.016	0.020	0.020
	0.018	0.025	0.025
0.020	0.020	0.028	0.028
	0.022	0.032	0.032
0.025	0.025	0.035	0.036
	0.028	0.036	0.036
0.032	0.032	0.040	0.040
	0.036	0.042*	0.040
0.040	0.040	0.045	0.045
	0.045	0.050	0.050
0.051	0.050	0.060	0.060
	0.056	0.063	0.063
0.064	0.063	0.065*	0.063
0.072	0.071	0.071	0.071
0.081	0.080	0.078	0.080
0.091	0.090	0.080	0.080
0.102	0.100	0.083*	0.080
	0.112	0.090	0.090
0.125	0.125	0.093*	0.090
	0.140	0.094*	0.090
0.156*	0.160	0.100	0.100
	0.180	0.109	0.112
0.188	0.190	0.112	0.112
	0.200	0.125	0.125
	0.244	0.156	0.160
		0.187	0.190
		0.188	0.190

NOTE Reading across the chart, thicknesses may be used interchangeably except as noted. GEN-3-00-18

*Reading across the chart, the thicker material may be substitutea for the thinner material. Substitution of the thinner material for the heavier material must be approved by the structural repair officer.

Figure 1-32. Standard Sheet Metal Thickness

DESIGNATION AND PROPERTIES OF ALUMINUM ALLOYS.

The aluminum industry has changed the alloy designations for all wrought aluminum alloys. The new designation is a four digit number whose last two digits usually conform to the old standard alloy number. The letter "S" is eliminated in the new designations. For example; the designation for aluminum alloy formerly known as 75S is now 7075. The temper designations have not been changed. For example; material marked 7075-0 is the same as material previously marked 75S-0. The basic temper designation is a letter. When the temper must be subdivided, a number follows the letter, for example; T4. The alloy number and the temper designation are separated by a dash, (2024-T4).

The general system of designation is as follows:

ALLOY DESIGNATIONS

FORMER DESIGNATION	ALLOY NO.	DESCRIPTION
2S	1100	Used where forming is severe—weldable—not heat-treatable—can be work-hardened.
3S	3003	Forms nearly as well as 1100—weldable—not heat-treatable—can be work-hardened.
52S	5052	More formable than 2024—less formable than 3003—weldable—not heat-treatable—can be work-hardened.
61S	6061	Weldable and heat-treatable—may be spot-welded or seam-welded in the heat-treated condition—may also be fusion-welded if the resultant annealed condition bordering the weld is acceptable.
24S	2024	Heat-treatable—may be spot- or seam-welded in heat-treated condition, but not fusion-welded. Can be formed in the heat-treated condition.
75S	7075	Heat-treatable—may be spot- or seam-welded in heat-treated condition, but not fusion-welded—does not form as readily as 2024 in heat-treated condition.

ALUMINUM ALLOY TEMPER DESIGNATIONS

LETTER	INDICATES	DESCRIPTION
F	As fabricated	Applies to products that acquire some temper qualities in the shaping processes but are not later heat-treated or intentionally work-hardened.
O	Annealed	Applies to the softest temper of wrought alloy products.
H	Strain Hardened	Applies to products that have their strength increased by work-hardening. The "H" is always followed by a number of two or more digits which subdivides this temper designation. The first digit indicates specific combination of basic operations. The final digit or digits indicates the degree of hardness.

Subdivisions of "H" Temper

H1—Work-hardened only. The number following this designation indicates the degree of work-hardening.

H2—Work-hardened and then partially annealed. The number following this indicates the degree of work-hardening remaining after partial annealing.

H3—Work-hardened and then stabilized. The number following this designation indicates the degree of work-hardening remaining after having been work-hardened a specific amount and then stabilized.

Breakdown of Hardness

¼ hard indicated by 2	¾ hard indicated by 6
½ hard indicated by 4	Full hard indicated by 8

The digit "3" would indicate a degree of hardness between ¼ hard and ½ hard, etc.

W	The unstable condition following solution heat-treatment	This designation, because of natural aging, is specific only when the period of aging is indicated, for example 2024-W ½ hr, etc.
T	Treated to produce a stable temper (other)	This designation applies to alloys when treated thermally to produce stable tempers with or without supplementary strain hardening. The "T" is followed by a number of one or two digits which indicate a specific combination of operations.

LETTER INDICATES	DESCRIPTION
	Subdivision of "T" Temper
	T2—Annealed (applies to cast products only).
	T3—Solution heat-treated and then cold-worked. This designation applies to products that are cold-worked after heat-treating chiefly to increase their strength; *however*, it also applies to alloy sheets as they come from the mill since the flattening process after heat-treating is recognized as cold-working, and does increase the strength of the material.
	T4—Solution heat-treated and then naturally aged to a substantially stable condition. This designation applies when the alloy is not cold-worked in any manner after heat-treatment. When the user heat-treats an alloy, the flattening process does not follow. Therefore sheet stock received from the mill in the "T3" condition which is annealed and re-heat-treated by the user becomes "T4." Unless specifically noted, "T4" can be substituted for "T3."
	T5—Artificially aged only. This designation applies to products that are artificially aged without prior solution heat-treatment.
	T6—Solution heat-treated and then artificially aged. This designation applies to products that are not cold-worked after heat treatment and any flattening, even at the mill, is not recognized as cold-working.
	T7—Solution heat-treated and then stabilized. This designation applies to products in which the temperature and time conditions for stabilizing are such that the aging is carried beyond that required to produce maximum strength and hardness in order to control growth and distortion resulting from casting or quenching strain.
	T8—Solution heat-treated, cold-worked and then artificially aged. This designation applies when cold-working is done for improving strength and the flattening process at the mill is recognized as cold-working.
	T9—Solution heat-treated, artificially aged, and then cold-worked.
	T10—Artificially aged and then cold-worked.

TITANIUM.

GENERAL CHARACTERISTICS.

The appearance of titanium is very similar to that of stainless steel. Both are a silver gray in color, although titanium is a little duller and slightly less lustrous than stainless steel. Titanium is about 40 percent lighter than stainless steel and about 60 percent heavier than aluminum. There are several methods of identifying titanium. One method is to touch it against a grinding wheel; this will produce white traces ending in brilliant white bursts. Another method is to moisten the suspected material and rub it across a piece of glass. If the material in question is titanium, it will leave gray-white marks on the glass. Titanium has a coefficient of expansion which is equal to only about half that of stainless steel. In other words, it does not expand as much as other metals with an increase in temperature. Like stainless steel, titanium is a very poor conductor of heat. Because of its low thermal conductivity, titanium is extremely sensitive to hot spots when it is unevenly heated. The corrosion-resistant properties of titanium are excellent, being, in most applications, comparable to, or better than, stainless steel. Titanium's resistance to salt water and marine atmosphere is ranked with platinum, surpassing even that of stainless steel.

FORMING.

Since nearly all repairs made on titanium parts will consist of formed titanium angles, the following information on methods and procedure is helpful. After determining the sizes of the material required to make the repair parts, mark the trim lines on the titanium sheet with a soft black pencil. Do not use any type of grease pencil to mark titanium. Methods of trimming titanium include shearing, sawing, nibbling, or blanking. Shearing or blanking has a tendency to leave minute cracks in

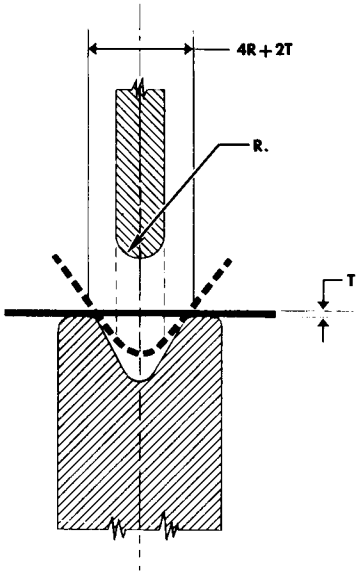


Figure 1-32A. Brake-forming Titanium Angles

the sheared edge, especially if the shear blades or the blanking die is slightly dull. Sawing or nibbling leaves an edge that will require considerable smoothing and polishing. Before any forming is attempted, all edges must be smooth and free of nicks that would cause cracks. The initial forming of titanium angles can be done with a power brake, hand brake, punch press, or hydropress. The recommended radii of the male forming punch are as follows: For bend radii up to 3/32 inch, the forming punch radius would be the same as the part being formed; for bend radii from 3/32 to 3/16 inch, the forming punch radius should be 1/64 less than the part being formed; and for bend radius over 3/16, the forming punch radius should be 1/32 less than the part being formed. The recommended span across the female forming die is four times the bend radius plus two times the material thickness. The suggested bend radius for forming titanium parts up to 0.070 inch in thickness is four times the material thickness, and five times the material thickness for parts over 0.070 inch in thickness.

COLD FORMING.

When the radius of the part being formed is large enough or the material gage is thin enough, cold forming is acceptable. Titanium is subject to excessive spring-back when cold-formed; therefore, stress relieving will be necessary to maintain the desired shape. If the radius is small or the degree of bend is extreme, stress relieving part way through the forming operation would be advisable.

ELEVATED TEMPERATURE (HOT) FORMING.

Hot forming of titanium is advantageous because it is possible to produce smaller bend radii, less spring-back, and increased elongation. Although any degree of heating is beneficial, a forming temperature of 1050°F should not be exceeded. Hot forming substantially increases the galling tendencies of titanium. One method of reducing the galling is to sandwich the titanium between sheets of other types of metal, such as 1010 steel or aluminum, and form the titanium. It is mandatory that some type of lubricant be used when hot-forming titanium. Waxes, molybdenum disulfide, or phosphate coatings are preferable to greases or other petroleum-based lubricants.

CONTOURING.

There are several methods of contouring angles after they have been initially formed to the desired cross-section. (See figure 1-32A.)

STRETCH FORMING.

Stretch forming (figure 1-32B) can be used for either mild or severe contours. Stretch form dies can be made from Kirksite, aluminum alloy, mild steel, or high-density plastic-impregnated wood. Each end of the angle to be contoured is inserted into the stretcher jaws and clamped securely. Tension is then applied to the angle through the hydraulic cylinders to which the stretcher jaws are attached. At the same time tension is applied, the stretcher jaws are slowly pulled back and around the stretch form die which has been fastened to the stationary stretcher bed. When this method of contouring is used, additional material on each end of the preformed angle must be allowed. This is to allow for the material inserted in the stretcher jaws.

SIMULATED STRETCH FORMING.

The method of stretch forming as shown in figure 1-32C induces excessive stretch in the part and should be limited to very mild contouring. This type of stretching can be done with an arbor press, hydraulic press, or a slow moving punch press, or power brake. Secure riser blocks (of sufficient height to prevent the male forming die from bottoming) to press bed. Clamp angle to be contoured to riser blocks with one leg upright and against a supporting angle plate. Secure male forming punch to press ram. Allow one material thickness between forming punch and supporting angle plate. Slowly lower forming punch against titanium angle and apply pressure until the desired contour is achieved.

STRETCH FORMING

GENERAL

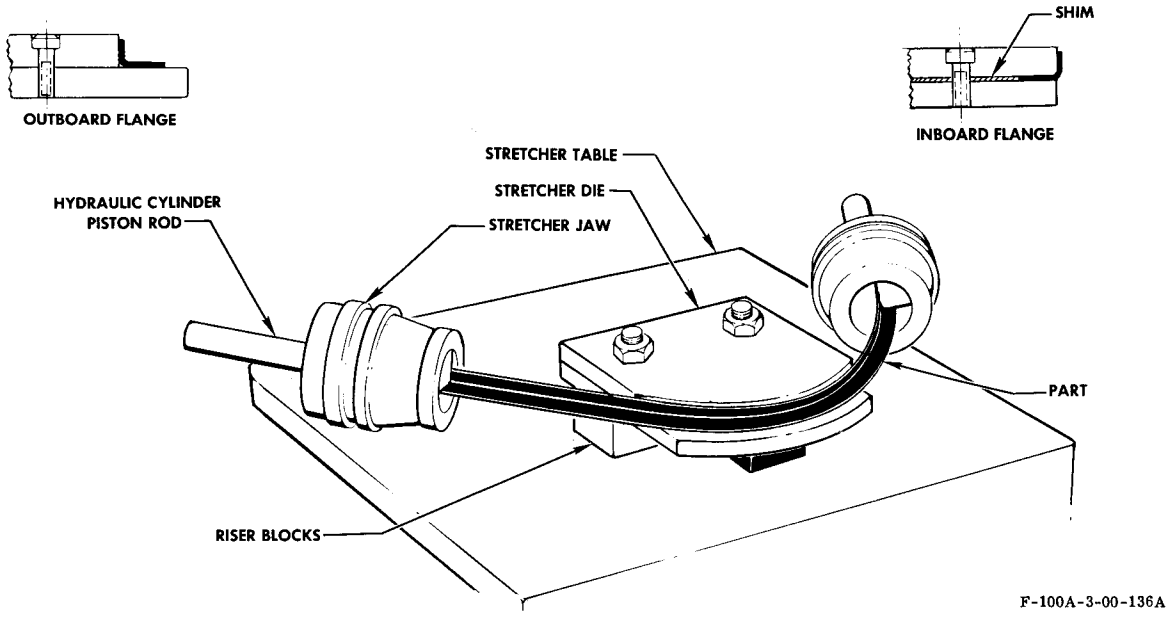


Figure 1-32B. Stretch-forming Titanium Angles

SIMULATED STRETCH FORMING

GENERAL

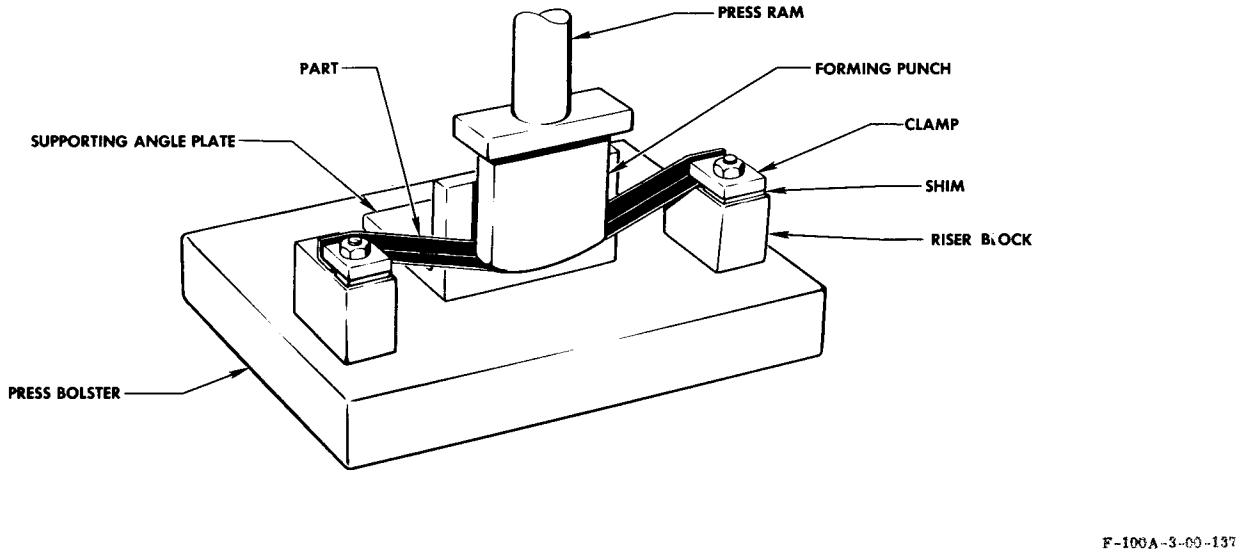


Figure 1-32C. Simulated Stretch-forming Titanium Angles

PRESS OR IMPACT FORMING

GENERAL

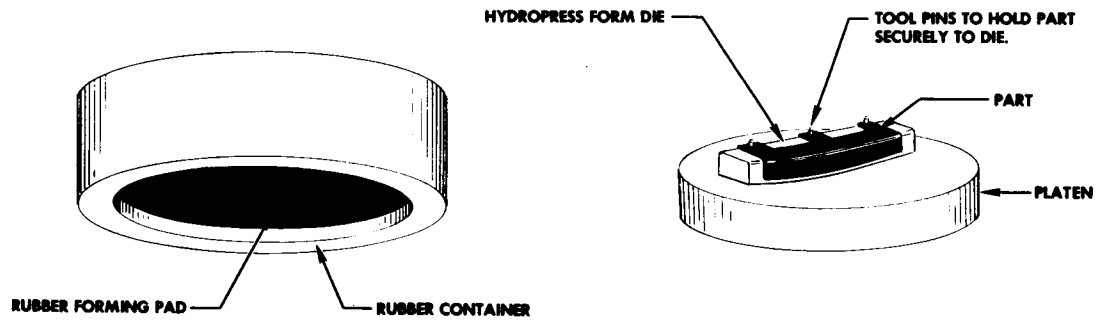


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Figure 1-32D. Hydropress or Impact-trapped Forming of Titanium Angles

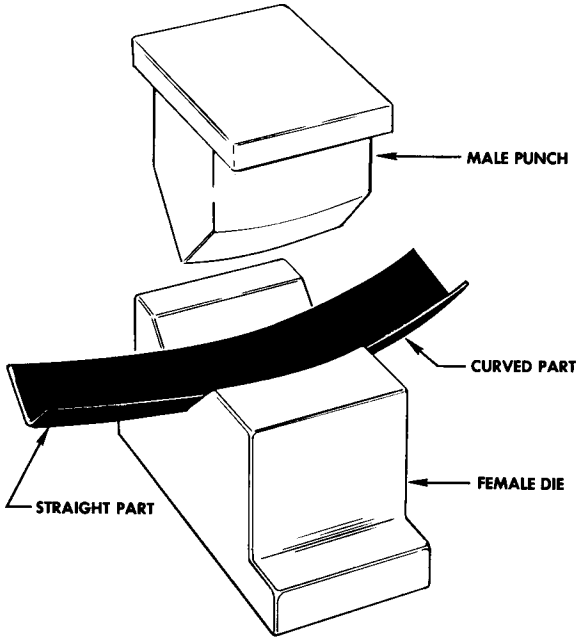

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Figure 1-32E. Progressive "V" Die-forming of Titanium Angles

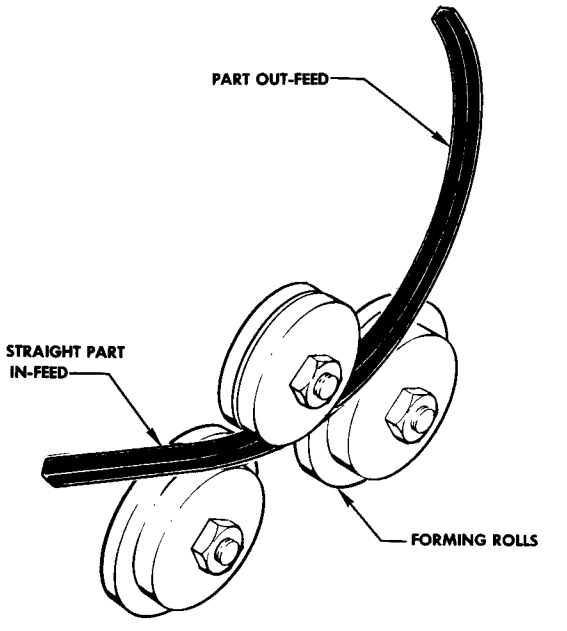

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Figure 1-32F. Roll-forming Titanium Angles

HYDROPRESS OR IMPACT-TRAPPED RUBBER FORMING.

The method of forming titanium shown in figure 1-32D is the only method shown where a part may be formed and contoured in the same operation. The tools necessary are mild steel form blocks, a heavily constructed rubber container to hold the rubber forming pad, and a lower plate or platen. The platen must fit snugly into the rubber container to ensure that the rubber is trapped inside and not allowed to squeeze out when pressure is applied. Fasten rubber container to ram or movable part of machine and locate platen directly underneath on stationary bed. Place form block and material to be formed approximately in the center of the platen and slowly bring rubber forming pad down over it. As pressure is applied, rubber flows around form block, forcing material against it. Power equipment required to do this type of forming would be either a punch press, power brake, hydropress, or drop hammer.

PROGRESSIVE "V" DIE FORMING.

The progressive "V" die-forming method as shown in figure 1-32E may be used to shape angles of any desired length to any required contour. Fasten male forming die to ram and female forming die to stationary bed of forming equipment. A punch press or power brake is best suited for this type of forming. The forming dies must be aligned so when the ram is moved up and down, the dies will mate perfectly. Feed preformed angle through forming dies as ram is actuated progressively, forming short sections of angle with each stroke. By increasing or decreasing the ram travel, the degree of contour may be controlled.

ROLL FORMING.

Roll forming with a standard three-roll Buffalo-type roll as shown in figure 1-32F can be used to shape angles with mild contours. The machine is set up with a series of rolls of various widths and spacers on each spindle to accommodate the legs of the angle being contoured. The amount of contour of the finished part depends upon the pressure applied by the center roll. Severe contours should not be attempted by this method, as too much pressure will cause the legs of the angle to wrinkle.

SHRINK FORMING.

Preformed angles with a slight or gradual contour may be contoured with a foot or power shrinker. (See figure 1-32G.) To avoid marking the part being contoured with the shrinker jaws, it is advisable to sandwich the titanium between two pieces of thin-gage, sand-blasted aluminum sheet. The sand-blasted side is placed against the titanium part. A medium-grit emery cloth may be used as a substitute for the sand-blasted aluminum. Place grit side of emery cloth against part being contoured.

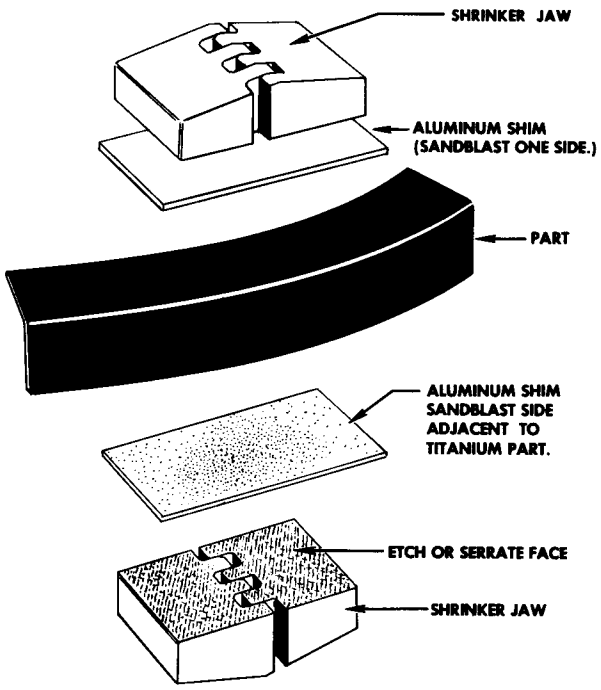


Figure 1-32G. Shrink-forming Titanium Angles

Joggling of titanium is done most easily by using elevated temperatures. Joggling should not be attempted adjacent to or intersecting previously formed areas without annealing the part first.

Bend radii must be maintained within the minimum limits described.

Bench work or hand finishing of titanium should be held to a minimum. Where bench work is necessary, every precaution should be taken to prevent the titanium from becoming marked or scratched. Stress relieving will be necessary after extensive bench work.

HOT SIZING.

Hot sizing of titanium is necessary to remove stress relief warpage or to make minor changes in the shape of the

part to conform to necessary tolerances. The part to be heat sized is placed into mated steel sizing dies that perfectly match the radius, contour, and degree required of the finished part. The sizing dies are next placed into a heat sizing fixture and are securely wedged in place. An example of a universal hot sizing fixture is illustrated in figure 1-32H. By using a series of different sizes of wedges and blocks, this type of fixture can be made to accommodate various sizes of sizing dies. The hot sizing fixture is then placed in either an electric or fuel fired furnace of the muffle type at a temperature of between 800 and 1050°F for 120 minutes. The time at temperature will not start until the surface temperature of the part being sized has reached the desired temperature. Fuel-fired furnaces where the flame can come in contact with the sizing fixture are not acceptable. Furnaces used must be automatically controlled by recording pyrometers or potentiometers. The furnaces must be capable of maintaining temperatures within ±25°F. Portable pyrometric equipment can be used to maintain constant control of the surface temperatures of the part being sized.

To permit adequate inspection, finished parts may be cleaned by careful use of motor-driven wire brushes, fine abrasive cloth or paper, and rubbing compounds.

DIMPLING.

The drilling of holes for dimpling determines to a large degree whether the finished assembly will be crack-free or not. Slotted or oversize holes will cause various misalignment problems because the dimpled skin will not seat properly. Rivets driven into such holes will usually result in cracked dimples. Always use the correct size of pilot drill for screws, Hi-Shear rivets, and blind rivets before dimpling. All holes to be dimpled should be deburred on both the top and bottom side of the sheet. Check on excessively slotted, rough, or misaligned holes before dimpling. Titanium *cannot* be redimpled; therefore, the only solution may be to drill out to the next larger size. Titanium sheets must be hot-dimpled. Either stationary or portable dimpling equipment may be used. To ensure good quality, all dimpling should be done at 750°F (±50°F). Low temperatures or low coining preload may cause either radial or circumferential cracks. In all events, these defects will be visible on the surface. Cracks may occur at the crest of the dimple if marks, scratches, or notches are present on the surfaces of the material.

Assembly of titanium requires much closer tolerances than either aluminum or stainless steel. Mismatched parts with over 0.010 gap should be shimmed with titanium shims. The center-to-end method of riveting should be employed when titanium is riveted. The first rivet should be driven as near the center of the fastener row as possible, and succeeding rivets should be progressively installed from the center out on each side of the central rivet.

TITANIUM COLOR CHANGE AND WARPAGE DUE TO FIRE DAMAGE.

Temperature changes bring out a wide range of colors in titanium material. The color of the metal changes on a time and temperature curve, and in the following order: original metallic, straw, purple, blue, greenish gray with a powdery cast, and just before complete destruction, tan with a powdery cast. Both greenish gray and tan with a powdery cast indicate that repairs or replacement is needed. The purple and blue colors are commonly found at normal operating temperatures and are not considered harmful.

Warping alone is no proof that the material has been overheated, as warping is caused by uneven temperatures over the surfaces of the metal and stress relieving of the structure. These uneven temperatures could be well below those detrimental to titanium.

If titanium warpage is minor and no greenish gray or tan color with a powdery cast can be detected, an attempt to straighten the metal may be tried. After straightening, the part should be thoroughly inspected for cracks. If no cracks are found, the part should be serviceable. Buckles or warps that cannot be satisfactorily straightened may be repaired by using the repairs and the fastener information in Section IV.

DRILLING AND REAMING.

The important factor in drilling titanium, as in drilling other hard materials, is that sufficient power and pressure must be applied to keep the drill cutting until the hole is completed. If the drill is allowed to ride in the partially drilled hole, the area will work-harden and further drilling will be extremely difficult. High-speed drills of the type used for corrosion-resistant steels should be used. The short fluted 135-degree split point drill has proved to be the most satisfactory type. An example of this type of drill is shown in figure 1-32J. Recommended drill speeds are also given in this figure.

High-speed, spiral-fluted reamers may be used at cutting speeds at about half those recommended for drills.

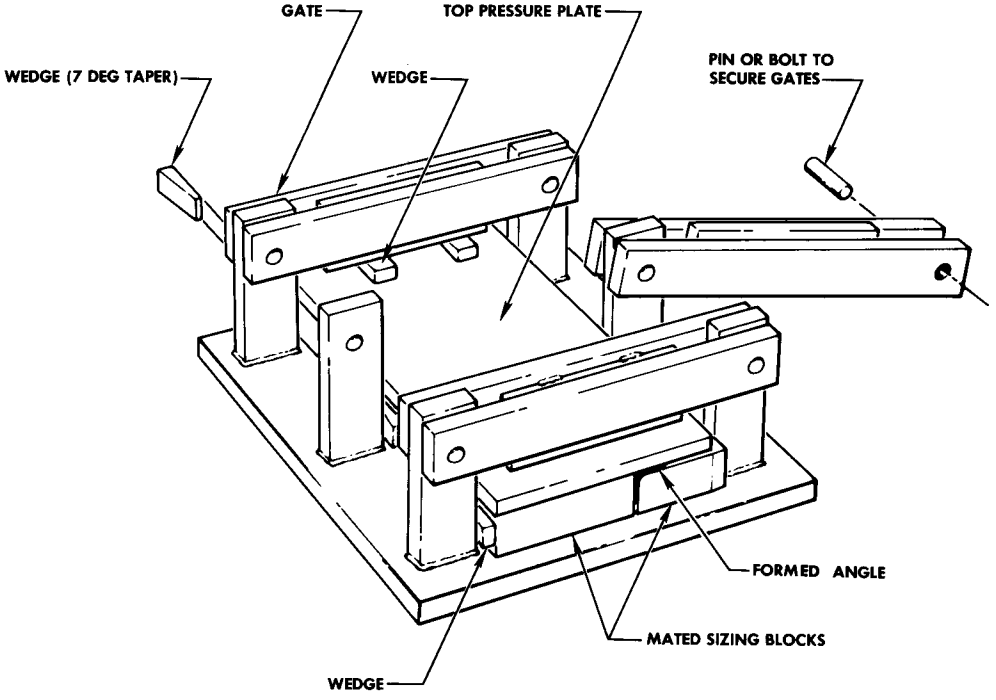
Broken tools such as drills and taps can be loosened from titanium for easier removal by submerging parts in a saturated solution of ferric ammonium sulfate or nitric acid, which do not appreciably affect titanium.

SCRATCHED TITANIUM.

Titanium is very notch-sensitive. All possible means of avoiding scratches should be taken. Scratches can lead to cracks and part failure. Long, deep scratches should be blended out. Fine sandpaper, or emery cloth of 120 grit or finer, may be used to blend out scratches.

HEAT SIZING

GENERAL



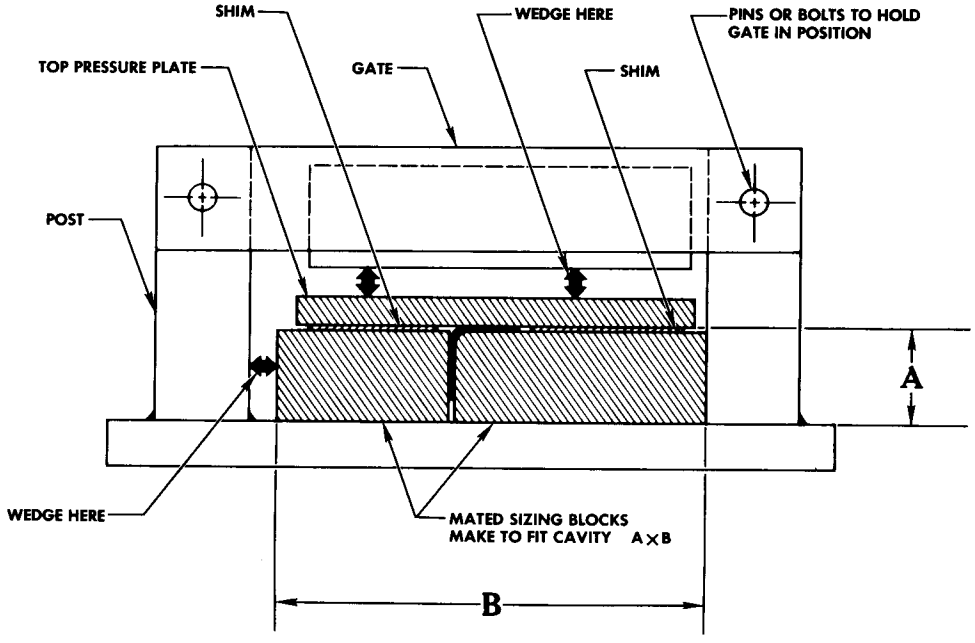


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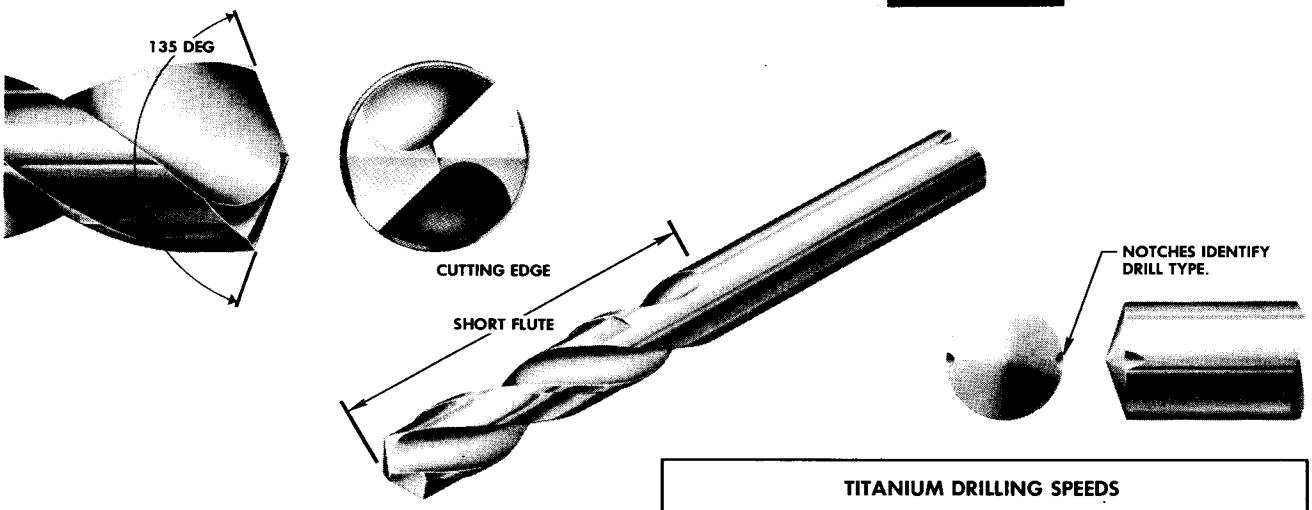
Figure 1-32H. Heat-sizing Fixture for Titanium Angles

DRILL FOR TITANIUM

GENERAL



TITANIUM DRILLING SPEEDS			
MOTOR SPEED (RPM)	FRACTIONAL DRILL SIZE	NUMBERED DRILL SIZE	LETTERED DRILL SIZE
1000	3/32–3/16	40–13	
750	13/64–1/4	12–1	A–F
500	17/64–1/2		G–Z

NOTE

Refer to titanium drilling procedure.

GEN-3-00-59

Figure 1-32J. Drills for Titanium

ASSEMBLING.

Do not force or spring oversize parts or splices into place when assembling new titanium parts or repairing titanium parts in use. Work parts until they fit properly. Shim if necessary. Flanges that have been pulled up in a nested splice or to a mold line contour have stresses introduced which may result in cracks or rupture.

CUTTING.

Titanium is very susceptible to cracking if rough areas are present; therefore, all cut and trimmed edges must be filed free of burrs, grooves, scratches, and sharp corners before fabrication is started. Cutting of titanium is similar to that of stainless steel. It work-hardens readily if light pressures are applied and the cutting tool is allowed to ride. Grinding removes metal slowly and a poor surface finish with heavy burring results. Titanium clogs and wears grinding wheels rapidly. Conventional saw equipment can be used; however, titanium is more apt than stainless steel to wear and clog saw blades.

WARNING

Fire precautions are necessary. Titanium burns similarly to magnesium and is explosive when suspended in air as dust particles.

CORROSION-RESISTANT STEEL.

Like titanium, corrosion-resistant steel is used in the fuselage rear section because of the high operating temperatures in this section. Its use is intended for applications requiring high heat-resistance values as well as high corrosion-resistance properties. Following are listed three of the most frequently used types of corrosion-resistant steel and their properties.

1. Specification MIL-S-5059, Comp 301, annealed and ½ hard. This material is not to be heat-treated, fusion-welded, or brazed, but may be spot-welded or seam-welded. In the annealed condition, this is a nonmagnetic material, and forms very easily. In the ½-hard condition, it is slightly magnetic and is considerably harder to form. Maximum service temperature is 700°F.

2. Specification MIL-S-6721, Comp 321. This material should be used when it is not feasible to use Specification MIL-S-5059 steel or when a part requires fusion welding. Specification MIL-S-6721, Comp 321, steel forms very easily and is nonmagnetic. It may not be heat-treated, but it can be spot-welded, seam-welded, fusion-welded, and silver-brazed. Specification MIL-S-6721, Comp 321, steel should be used where temperatures are to exceed 700°F.

3. Specification MIL-S-25043, 17-7PH. This material may be used where higher strength or better formability is required than can be obtained from Specification MIL-S-5059, ½-hard, steel and in all cases must be heat-treated

to 180,000 to 200,000 psi in accordance with Specification MIL-H-6875 after forming. This material may be spot-welded or seam-welded after heat-treatment, or fusion-welded before heat-treatment, but it must not be silver-brazed or copper-brazed. Specification MIL-S-25043, 17-7PH, steel is a magnetic material with a maximum service temperature up to 55°F.

CHROME MOLYBDENUM STEEL.

Chrome molybdenum steel is a low-alloy steel used in limited quantities in the construction and repair of aircraft. Following are listed three of the most commonly used types of chrome molybdenum steel and their properties.

1. 4130. This material may be copper-brazed or fusion-welded, but not spot-welded or seam-welded; it also may be silver-brazed if the part is not to be heat-treated after silver brazing. 4130 steel should not be surface-hardened, but it may be heat-treated to 160,000 to 180,000 psi.

2. 4140. This material may be fusion-welded or copper-brazed, but not spot-welded, seam-welded, or silver-brazed. This material should not be surface-hardened, but it may be heat-treated to 180,000 to 200,000 psi.

3. 4340. This material may be fusion-welded or copper-brazed, but not spot-welded, seam-welded, or silver-brazed. This material is not to be surface-hardened, but may be heat-treated to 200,000 to 220,000 psi.

A low-alloy steel to meet repair requirements can be selected from the following chart, using full hardening thickness and the strength range required. 4140 steel may be used in place of 4130 steel, and 4340 steel may be used in place of either 4130 or 4140 steel.

THICKNESS FOR FULL HARDENING (INCHES)	TENSILE STRENGTH REQD (PSI)				
	125,000 THRU 140,000	140,000 THRU 160,000	160,000 THRU 180,000	180,000 THRU 200,000	200,000 THRU 220,000
0.000 thru 0.052	4130	4130	4130	4130	4340
0.251 thru 0.500	4130	4130	4130	4140	4340
0.501 thru 0.750	4130	4130	4140	4140	4340
0.751 thru 1.000	4140	4140	4140	4140	4340
1.001 thru 1.250	4140	4140	4140	4340	4340
1.251 thru 1.500	4140	4340	4340	4340	4340
1.501 thru 3.000	4340	4340	4340	4340	4340
3.001 thru 3.500	4340	4340	4340	4340	—
3.501 thru 4.000	4340	4340	4340	—	—

FLIGHT CONTROL SURFACE STATIC BALANCE.

By nature of the construction and location of their hinge points, movable control surfaces are tail-heavy until balanced. Lead weights, added to the leading

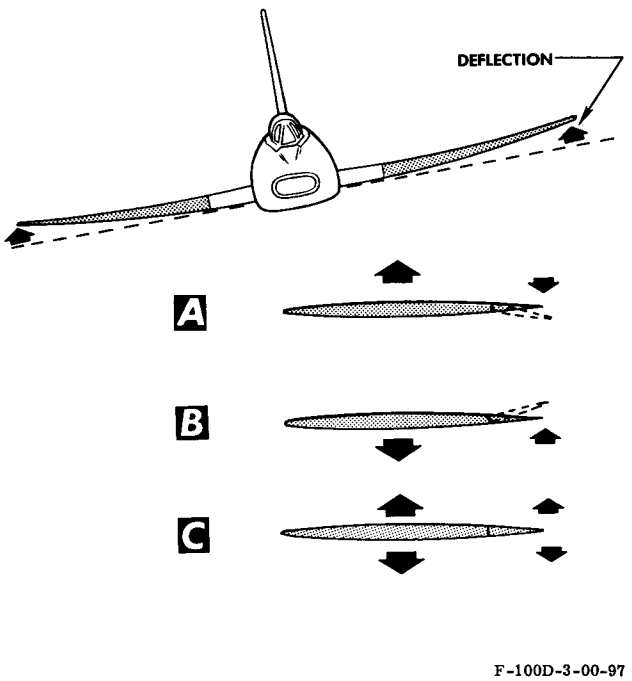


Figure 1-33. How Wing Deflection Affects Control Surface

edge in certain locations, provide this balance. Static balance is carefully considered in the design of an airplane to prevent flutter. When in flight, the wings of an airplane are deflected as the various loads are applied. (See figure 1-33.) This deflection of the wing is greatest at the tip and diminishes toward the root. As the wing is deflected upward, tail-heaviness in an unbalanced control surface will cause the control surface to move downward. Changing the flow of air over the wing in this manner would cause the wing to move upward even more. In the same way, if the wing is deflected downward, tail-heaviness would cause the control surface to move up, and tend to cause the wing to move down even further. A balanced control surface has enough weight added forward of the hinge line to offset the natural-tail heaviness of the surface. When a wing with a balanced surface is deflected upward or downward, the surface itself moves upward or downward in the same plane with the wing. After making repairs to a control surface, rebalancing may be necessary. This can be determined by testing the balance of the surface. (Refer to "Methods of Checking Static Balance.") Since the safety of the airplane is involved, it is most essential that the tolerance values be held. The terms used in regard to static balance are defined as follows:

Moment—The tendency that a force (such as weight) to produce rotation about an axis (centerline of control surface hinge) because of its position with respect to the axis. For example; one pound of weight 10 inches, horizontally, from an axis produces ten inch-pounds of moment (1 pound X 10 inches = 10 inch-pounds) about the axis.

Static Balance—The numerical value of the moment resulting from the weight of the surface about the hinge line (with surface horizontal). Balance is usually expressed in inch-pounds of moment.

Static Unbalance – Underbalance or Tail-Heaviness – The moment that causes the trailing edge of a surface, which is suspended on its hinge points in a fixture, to rotate downward after having placed the surface at neutral. The center of gravity of the surface is aft of the hinge line. To balance an underbalanced surface at neutral, it would be necessary to place a quantity of weight forward of the hings line which would bring the trailing edge up. The product of this weight, and its measured distance from the hinge line is the moment in inch-pounds by which the surface is underbalanced. (See figure 1-34.)

Static Overbalance or Nose-heaviness—The moment that causes the leading edge of a surface, which is suspended on its hinge points in a fixture, to move downward from neutral. The center of gravity of the surface is forward of the hinge line. A quantity of weight placed on the surface aft of the hinge line would be required to balance an overbalanced surface at neutral. The surface would be overbalanced by the moment found when multiplying this weight by its measured distance from the hinge lines. (See figure 1-35.)

Complete Static Balance—The condition where the surface balances at neutral without having to place a weight anywhere on the surface. The center of gravity is on the hinge line. (See figure 1-36.)

Statically Unstable Surface—A surface in which the center of gravity is above the hinge line. When an

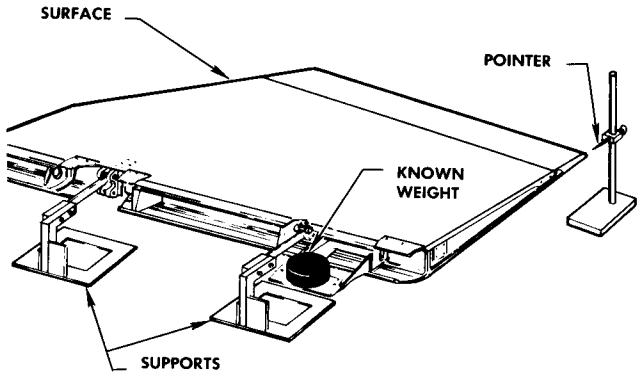


Figure 1-34. Static Underbalance

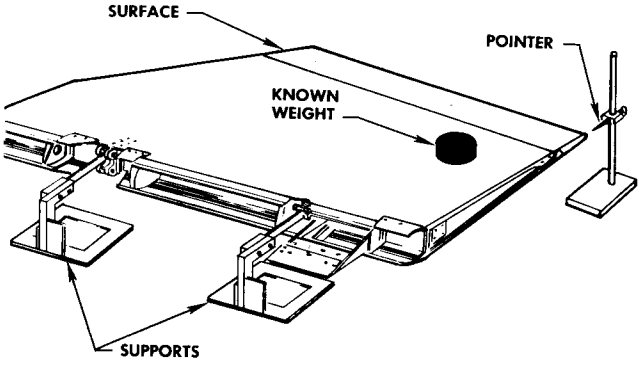


Figure 1-35. Static Overbalance

unstable surface is suspended on its hinge points in a fixture, it will rotate as far as the fixture will permit. Statically Stable Surface—A surface in which the center of gravity is below the hinge line. When a stable surface is suspended on its hinge points in a fixture, it will assume a nose or tail down attitude in proportion to its overbalance or underbalance.

METHODS OF CHECKING FLIGHT CONTROL SURFACE STATIC BALANCE.

There are different methods of checking static balance. Manufacturers and repair depots might use a production method. Elsewhere, an alternate (field) method might be used. The end result of either method is the same.

The production method requires a fixture (figure 1-37) which duplicates the hinge brackets of an airplane. The fixture has a means of indicating neutral. Test weights are used to determine balance.

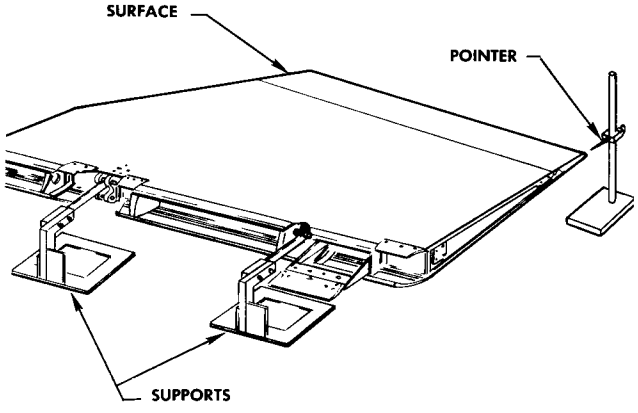


Figure 1-36. Complete Static Balance

The alternate method is relatively simple and can be used at smaller bases. The equipment necessary to balance a surface is a set of support stands, a reference pointer, a beam scale and a reaction block or a weight of known value. (See figure 1-38.) The support stands must be arranged so that they are absolutely level. Mount the surface in the support stands and measure from center of hinge line to table to establish neutral. Set pointer at neutral. If surface trailing edge aligns with pointer, it is at neutral and in complete static balance. If the trailing edge of the surface does not align with pointer the moment of underbalance or overbalance may be found as follows:

To find underbalance moment, place reaction block on scale pan and adjust height of reaction block until tip is equal to neutral. Set scale and reaction block under trailing edge of surface. (See figure 1-39.) The surface, being trailing-edge heavy, will bear on the scale. Balance scale beam and take scale reading. Measure the distance from hinge line to point of contact of reaction block. This is the moment arm. The underbalance moment is equal to the scale reading minus weight of reaction block, times the measured distance (moment arm). If the underbalance moment falls within the tolerance established for the surface, it need not be re-balanced. For example: A surface has a tolerance of 118 to 138 inch-pounds underbalance.

1. Subtract weight of reaction block (1.0 pounds) from scale reading (6.5 pounds). The result is 5.5 pounds. (6.5 – 1.0=5.5.)

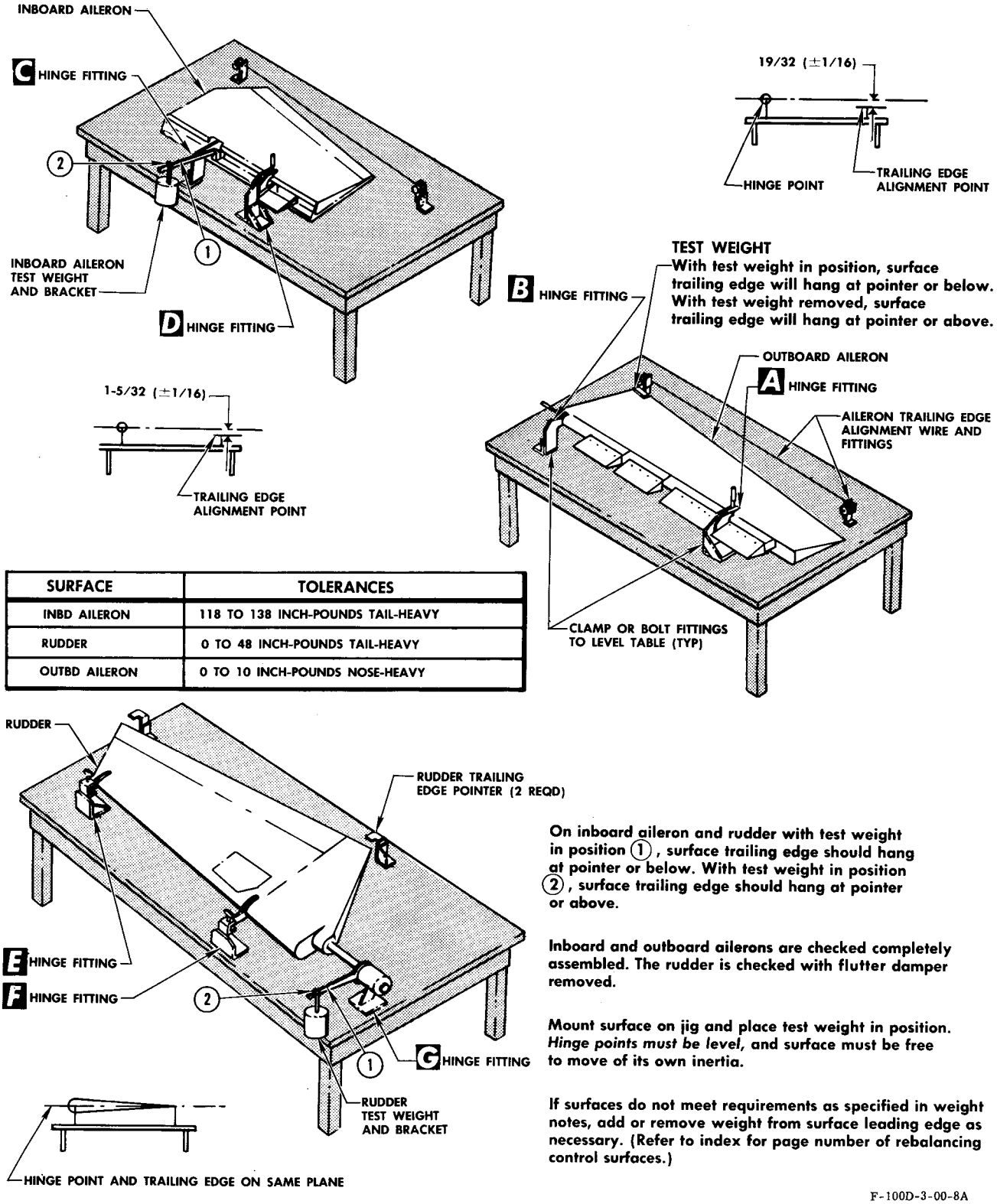
2. Multiply the result from step 1 (5.5 pounds) by the moment arm (22 inches). The product, 121.0 inch-pounds (22 X 5.5 = 121.0), is the underbalance moment.

The surface is within tolerance. Had the surface been out of tolerance, it would have been necessary to add weight at the leading edge. Should a scale not be available, a weight of known value may be used. With surface suspended on support, place weight on surface forward of hinge line. Slide weight toward leading edge until trailing edge is in neutral. The distance from the centerline of the weight to the hinge line (moment arm), times the value of the weight will be the underbalance moment in inch-pounds.

Overbalance may be found in much the same manner as underbalance, except that the scale would be placed under the leading edge or the known weight would be used aft of the hinge line. Because of the shape of the leading edge, using a weight of known value is much more convenient than using a scale. Overbalance would be corrected by removing weight from the leading edge.

PRODUCTION METHOD FIXTURES

GENERAL



SURFACE	TOLERANCES
INBD AILERON	118 TO 138 INCH-POUNDS TAIL-HEAVY
RUDDER	0 TO 48 INCH-POUNDS TAIL-HEAVY
OUTBD AILERON	0 TO 10 INCH-POUNDS NOSE-HEAVY

On inboard aileron and rudder with test weight in position (1), surface trailing edge should hang at pointer or below. With test weight in position (2), surface trailing edge should hang at pointer or above.

Inboard and outboard ailerons are checked completely assembled. The rudder is checked with flutter damper removed.

Mount surface on jig and place test weight in position. *Hinge points must be level,* and surface must be free to move of its own inertia.

If surfaces do not meet requirements as specified in weight notes, add or remove weight from surface leading edge as necessary. (Refer to index for page number of rebalancing control surfaces.)

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Figure 1-37. Static Balance Check—Production Method (Sheet 1 of 7)

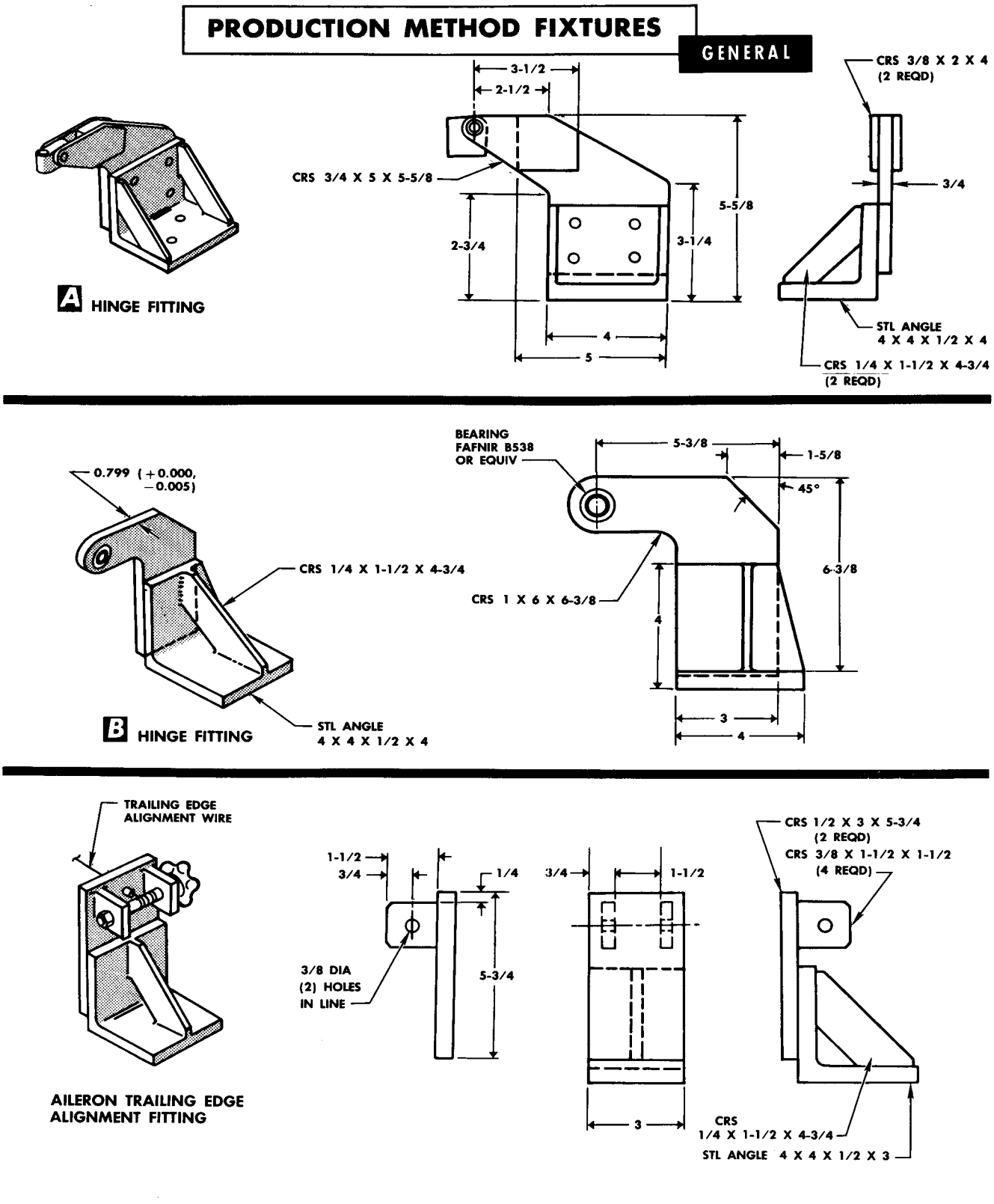
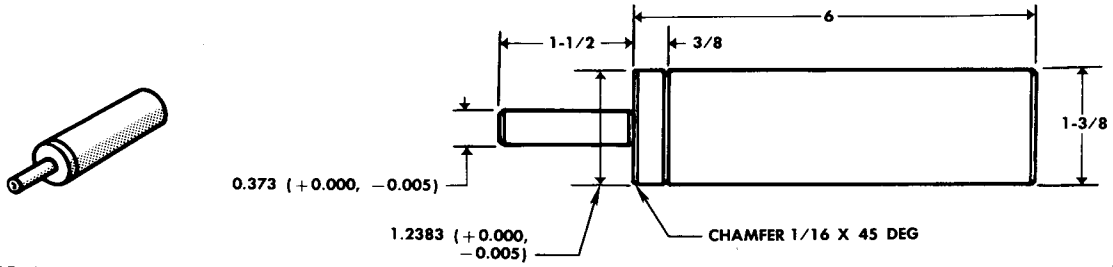


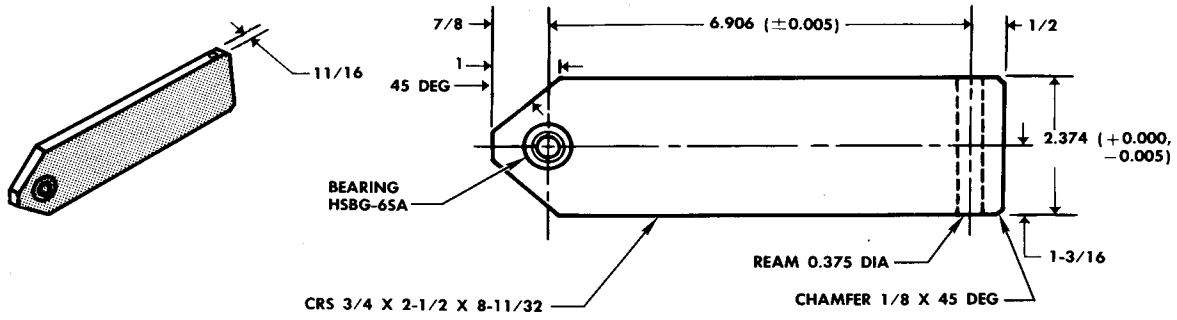
Figure 1-37. Static Balance Check—Production Method (Sheet 2 of 7)

PRODUCTION METHOD FIXTURES

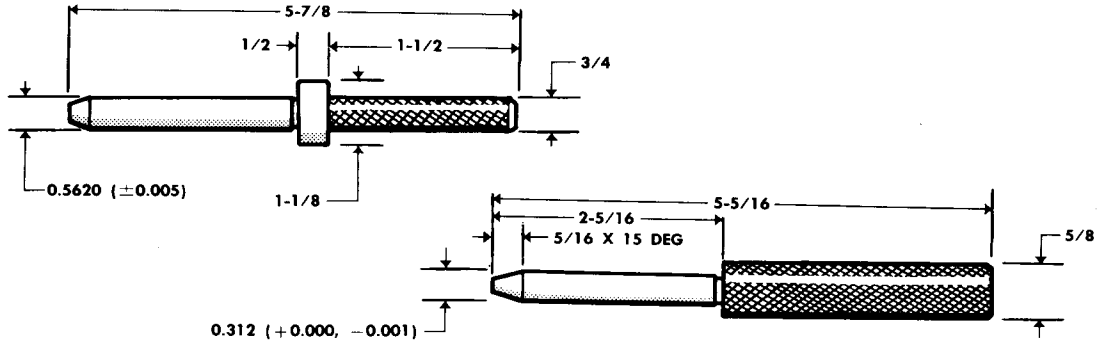
GENERAL



RUDDER—CENTER HINGE PIN



ADAPTER—RUDDER CENTER HINGE



AILERON HINGE PINS

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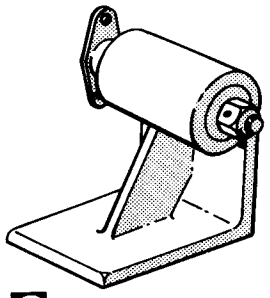
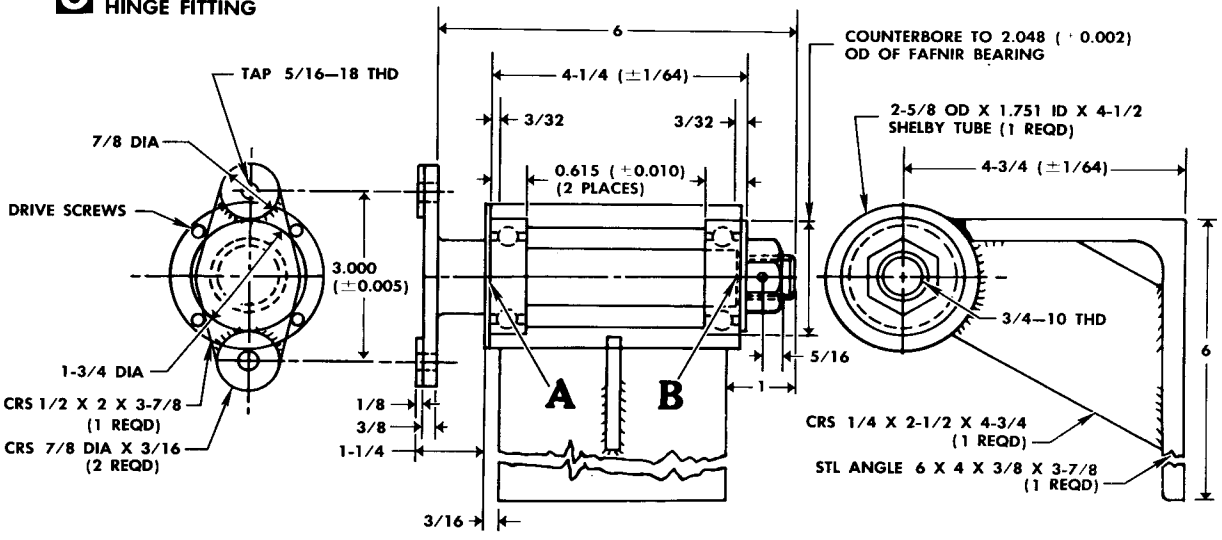
Figure 1-37. Static Balance Check—Production Method (Sheet 3 of 7)

PRODUCTION METHOD FIXTURES

GENERAL



G HINGE FITTING



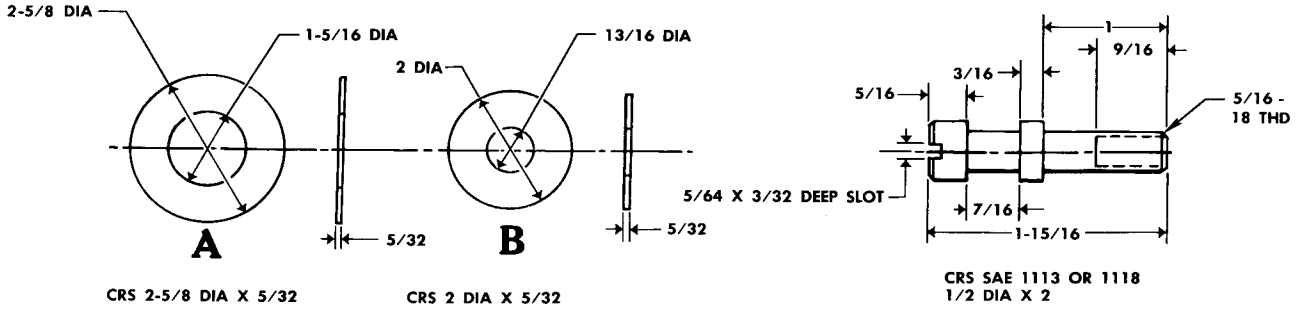


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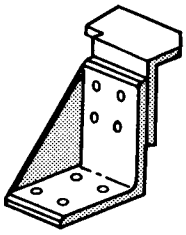
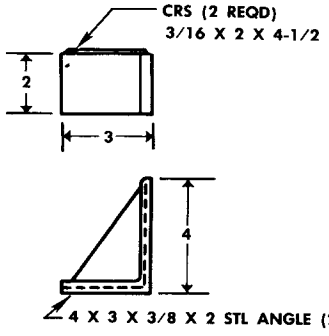
Figure 1-37. Static Balance Check—Production Method (Sheet 4 of 7)

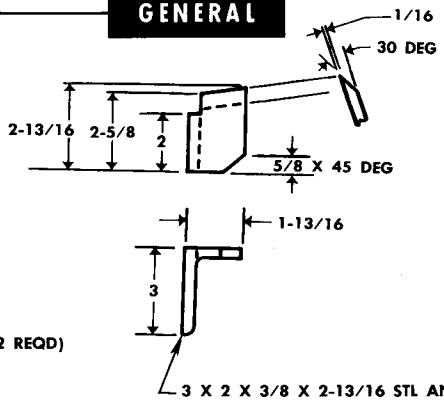
PRODUCTION METHOD FIXTURES

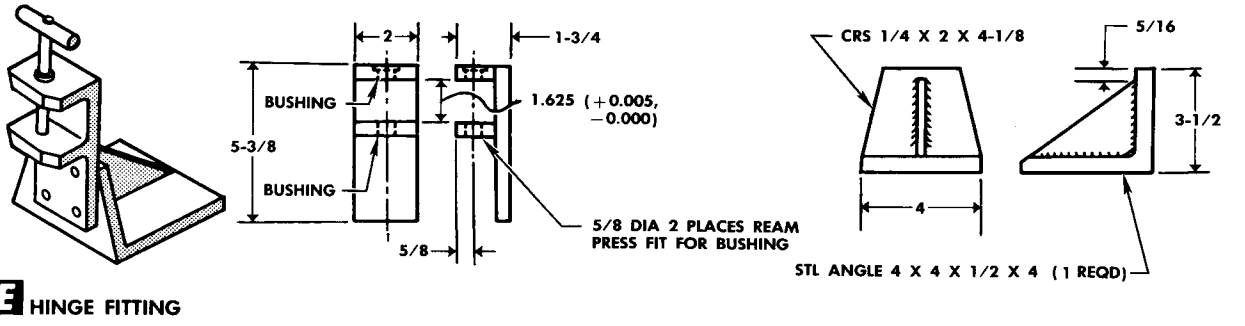
GENERAL



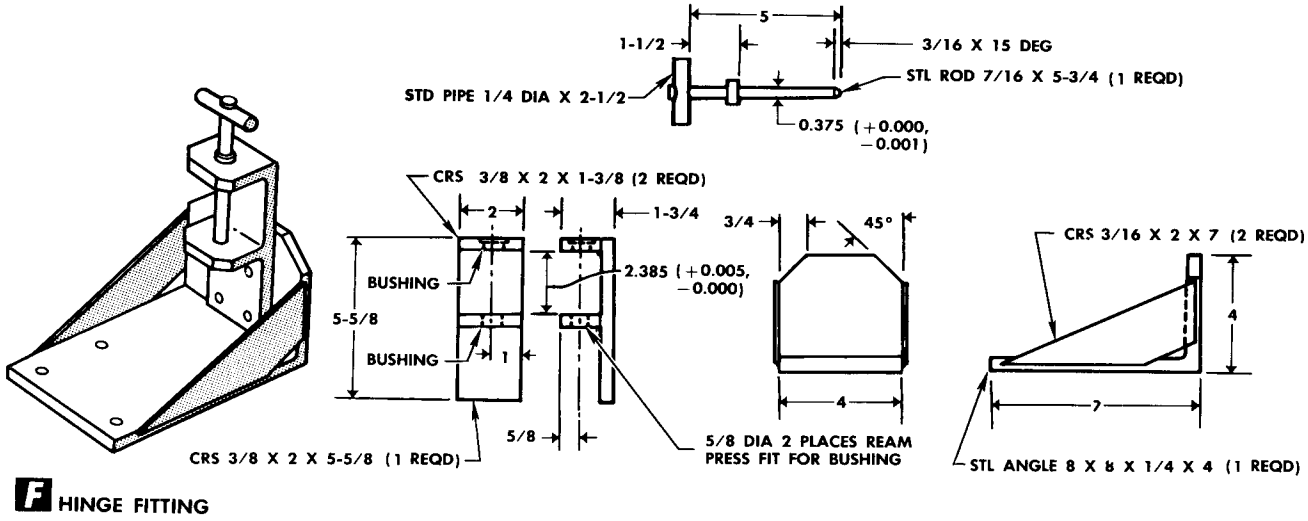
RUDDER
TRAILING EDGE
POINTER (2 REQD)







E HINGE FITTING



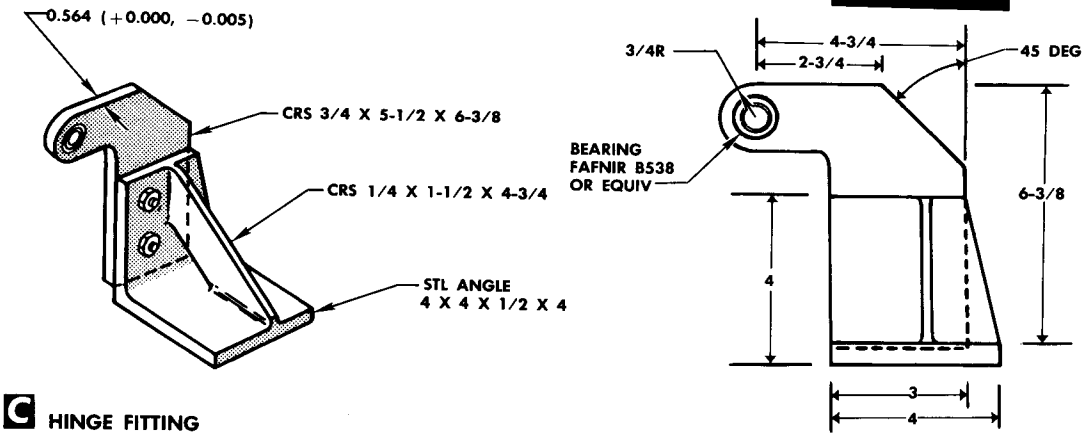
F HINGE FITTING

F-100D-3-00-14A

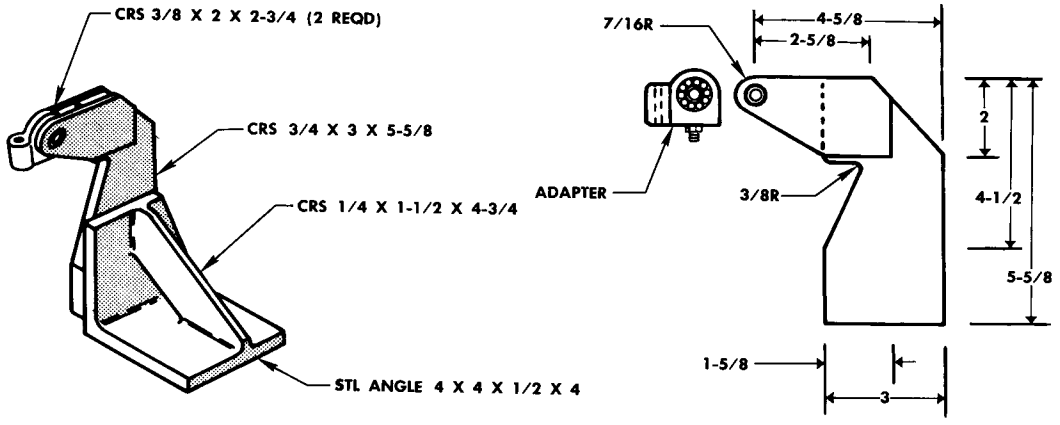
Figure 1-37. Static Balance Check—Production Method (Sheet 5 of 7)

PRODUCTION METHOD FIXTURES

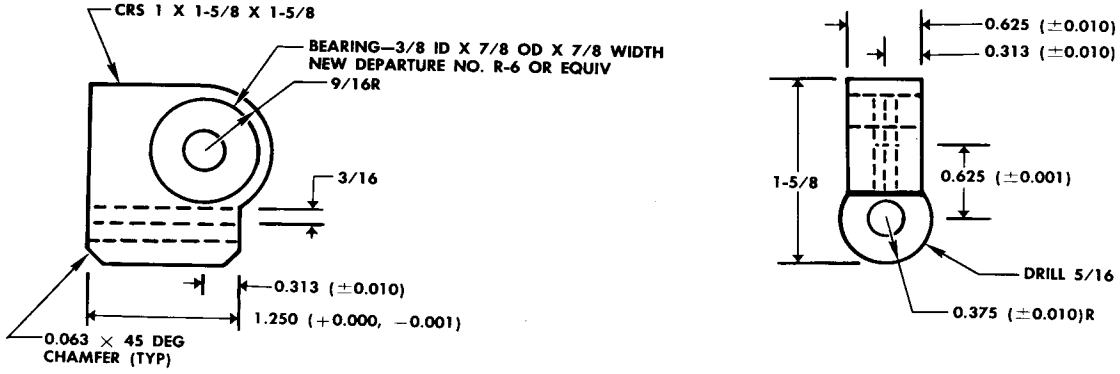
GENERAL



C HINGE FITTING



D HINGE FITTING



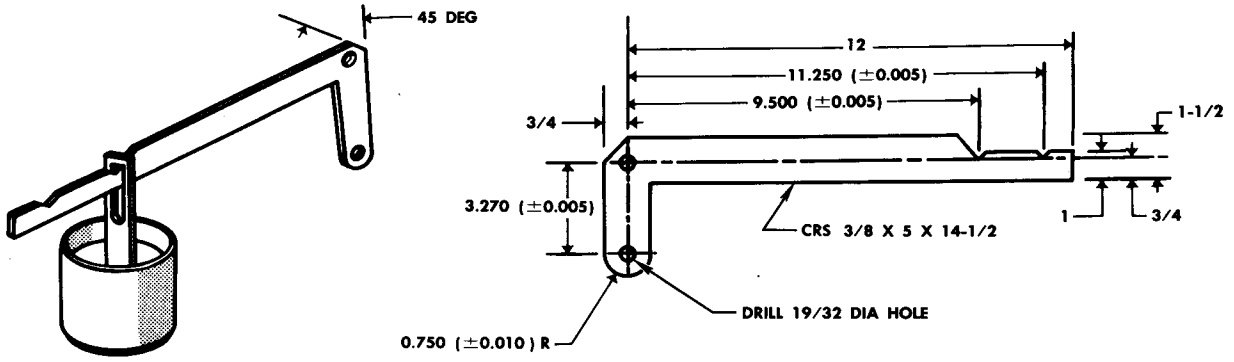
ADAPTER
Use with outboard hinge fitting on both inboard and outboard ailerons.

F-100D-3-00-16A

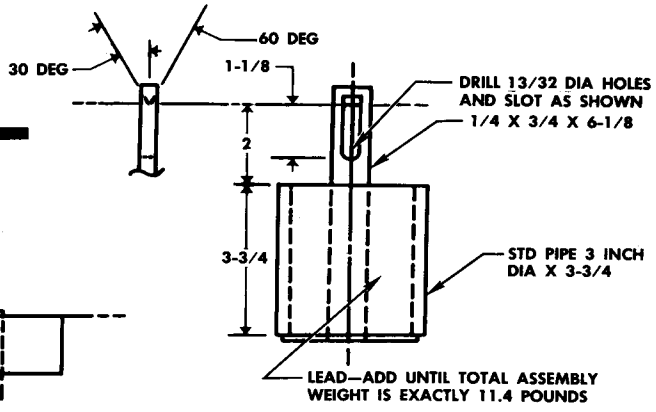
Figure 1-37. Static Balance Check—Production Method (Sheet 6 of 7)

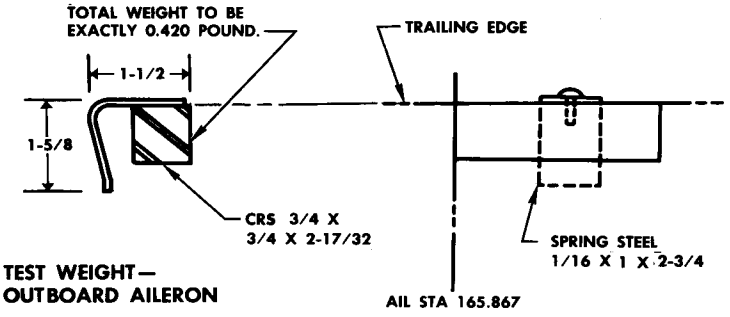
PRODUCTION METHOD FIXTURES

GENERAL

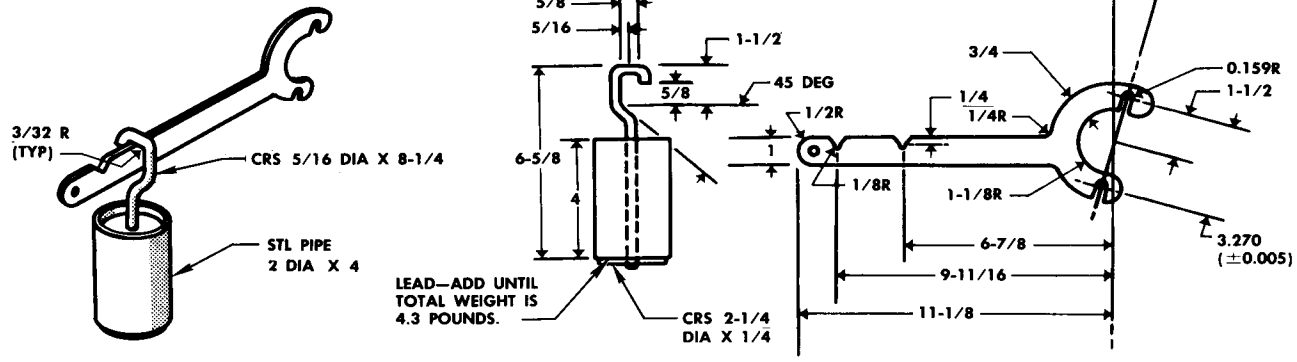


TEST WEIGHT AND BRACKET—INBOARD AILERON





TEST WEIGHT—OUTBOARD AILERON



TEST WEIGHT AND BRACKET—RUDDER

F-100D-3-00-15A

Figure 1-37. Static Balance Check—Production Method (Sheet 7 of 7)

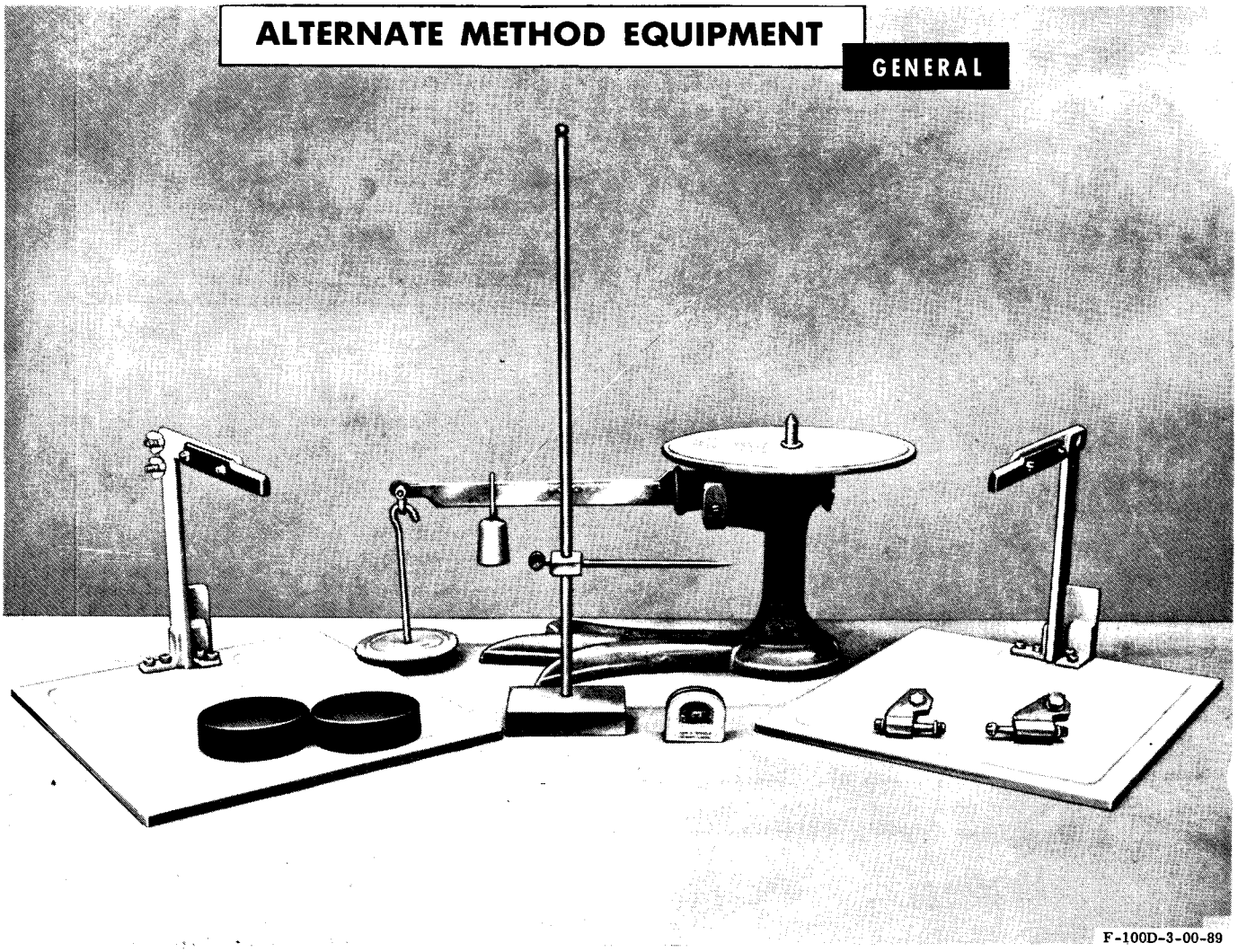


Figure 1-38. Static Balance Check—Alternate Method Equipment

When checking balance of a surface, it is essential that the surface be supported so that it has the same freedom of movement and range of travel that it has on the airplane. The area should be free of air currents.

Note

Checking static balance while the surface is mounted on the airplane is not recommended.

When rebalancing, to compensate for a repair, balance weight should be added as nearly as possible at the station of repair.

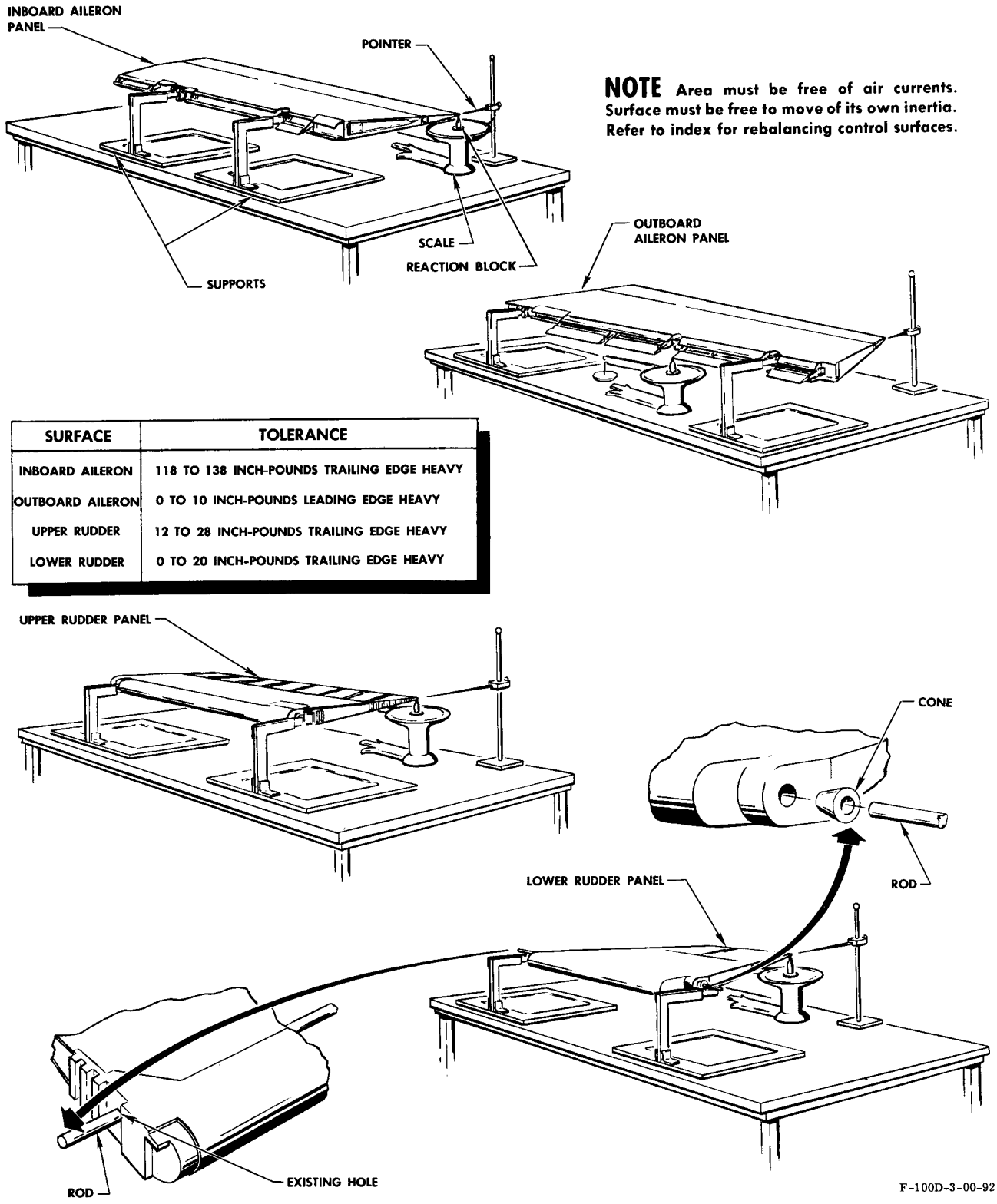
BALANCING DUAL FLIGHT CONTROL SURFACES.

To eliminate any possibility of binding as a result of deflection of the fixed surfaces, the rudder and ailerons are in two sections. When static balance is being checked by the field method, it is necessary to balance each panel of a dual surface separately. All surfaces must be balanced completely assembled. Suspend aileron panel on knife-edge supports using hinge link assembly on the outboard hinge. The hinge link must be installed in the proper position. (See figure 1-39, sheet 2 of 2.)

The upper rudder panel is balanced with the flutter damper removed. It may be suspended from knife-edge supports as shown in figure 1-39. The lower rudder panel is balanced with the torque tube removed for convenience. To make the use of knife-edge supports easier, remove rudder torque tube from lower rudder panel. Make a centering cone of metal or wood to fit opening in leading-edge beam where torque tube was removed. Drill hole through center of cone. Drilled hole should be same size as hole in top rib of panel at hinge line. Obtain a rod long enough to extend through the centering cone, leading edge and top rib. The rod, resting on the knife-edge supports, will suspend the panel for balancing.

STATIC BALANCE CHECK—ALTERNATE METHOD

GENERAL



NOTE Area must be free of air currents. Surface must be free to move of its own inertia. Refer to index for rebalancing control surfaces.

SURFACE	TOLERANCE
INBOARD AILERON	118 TO 138 INCH-POUNDS TRAILING EDGE HEAVY
OUTBOARD AILERON	0 TO 10 INCH-POUNDS LEADING EDGE HEAVY
UPPER RUDDER	12 TO 28 INCH-POUNDS TRAILING EDGE HEAVY
LOWER RUDDER	0 TO 20 INCH-POUNDS TRAILING EDGE HEAVY

Figure 1-39. Static Balance Check—Alternate Method (Sheet 1 of 2)

STATIC BALANCE CHECK—ALTERNATE METHOD

GENERAL

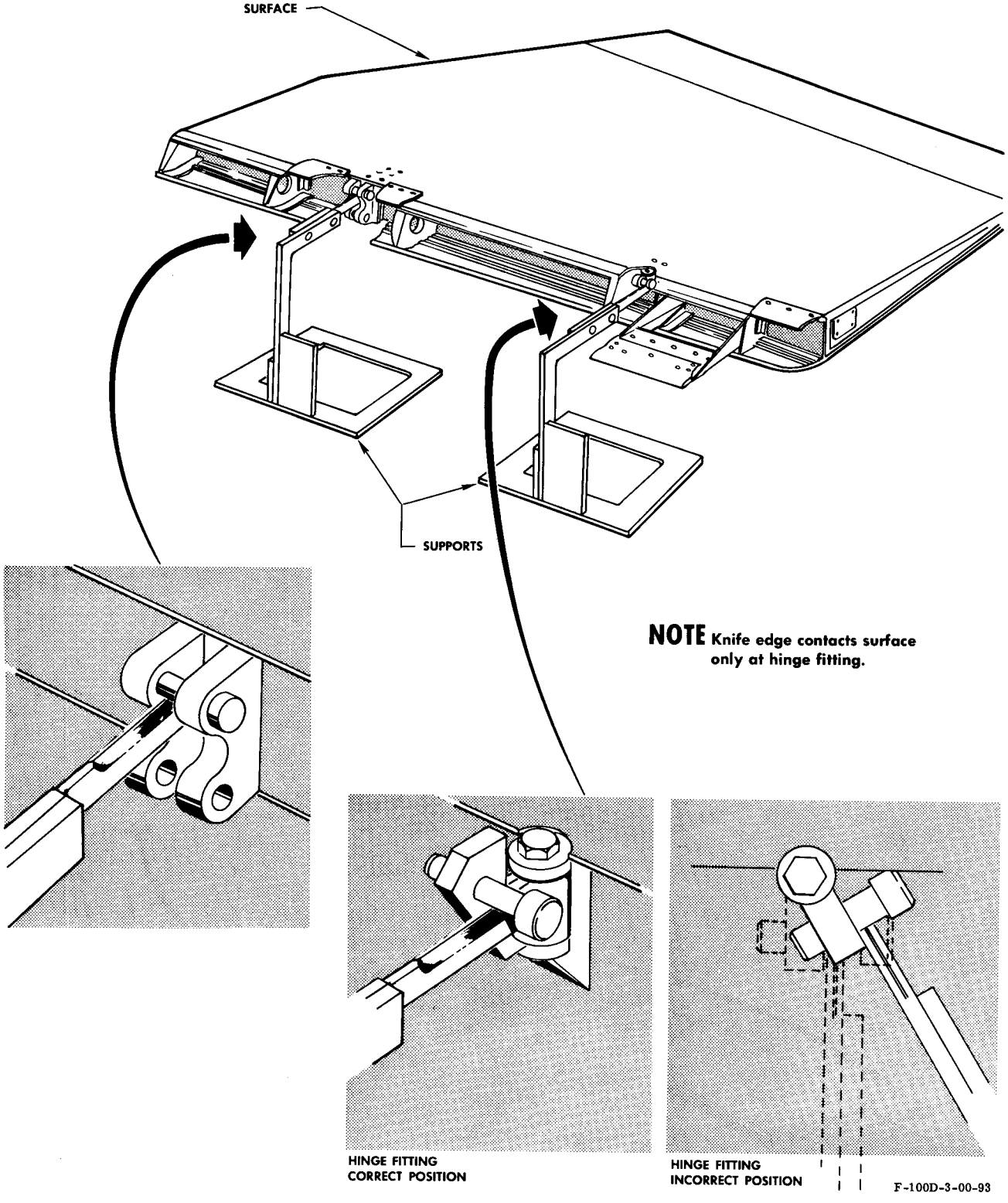


Figure 1-39. Static Balance Check—Alternate Method (Sheet 2 of 2)

REBALANCING FLIGHT CONTROL SURFACES

The inboard ailerons each have one lead balance weight at the outboard end. There are four lead weights along the span of the outboard aileron. Each aileron is rebalanced by the addition of lead washers (2W16) aft of the lead casting.

A static unbalance of 5 inch-pounds additional tail heaviness beyond the specified limits given in figure 1-39 is permissible on either the inboard or the outboard aileron or divided between the two. If all or part of the 5 inch-pounds is used on one aileron, the adjoining one must be checked to make sure that the total of 5 inch-pounds for both is not exceeded.

Balance weights are installed in the upper rudder panel at the lower end of the leading-edge beam. (See figure 1-40.) The lower rudder panel has no balance weights.

SEALS AND SEALANTS.

Many areas in an airplane must be sealed in order for the airplane to operate effectively and to provide comfort and safety for the pilot. (See figure 1-41.)

TYPES OF SEALANTS.

Two basic types of sealants used are curing and noncuring. The noncuring sealant is used exclusively in the integral fuel tank of the wing. (Refer to "Wing Group," Section II.) The curing sealants used vary in characteristics. Each sealant is chosen to do a specific job in the area to which it is applied.

To allow removal or replacement, parts such as castings, electrical equipment, hydraulic lines, bolts, etc, are sealed with rubber-type gaskets. Rubber seals, such as those used on doors, canopy, etc, should be replaced when deteriorated. Refer to Illustrated Parts Breakdown, T.O. 1F-100D-4, for nomenclature and part numbers.

SEALING PRESSURIZED AREAS.

The fuselage structure and seals in the pressurized cockpit section (figure 1-42) are designed to withstand a maximum pressure differential of 10 psi. However, the cockpit pressure regulator will normally prevent the pressure differential from exceeding 5 psi maximum in operation. Should the pressure regulator fail, a safety valve limits maximum pressure differential in the cockpit to 5.55 psi for F-100D Airplanes and 5.75 psi for F-100F Airplanes. Allowable leakage from the cockpit area is 50 cfm on new F-100D Airplanes and 65 cfm on new F-100F Airplanes. (This includes the 5 cfm leakage through the cockpit pressure regulator when it is off for ground pressurization tests.) On airplanes which have been in service for a considerable length of time, the maximum leakage rate is 80 cfm for F-100D Airplanes and 108 cfm for F-100F Airplanes before repairs are necessary. This must be considered when any damage to the pressure seals in the cockpit section is repaired. Complete pressure-testing procedures, together with minimum pressure requirements and necessary precautionary measures, are outlined in T.O. 1F-100D-2-2. The pressure sealing should be done from the pressurized side of the surface being sealed. Sealing tapes must not be used between closely united surfaces, such as riveted structural components, in primary structure. Care should be taken to make sure that all areas are sealed before assembly operations make the area inaccessible.

SEALING PRESSURIZED AREAS—COCKPIT PRESSURE-SEALING REPAIR PRECAUTIONS.

Before making any cabin repairs, all personnel concerned should be thoroughly acquainted with the following pressurization precautions:

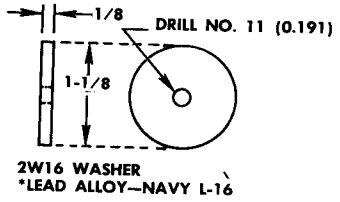
1. With personnel inside, the cockpit must never be pressurized to a higher pressure than has been previously established during the testing procedure with the cockpit empty.
2. It is recommended that only personnel who have been examined and approved by medical authority be allowed in the cockpit while it is under pressure.
3. It is essential that no one who has recently had a cold or has had his sinuses impaired in any way be allowed in the cockpit while it is under pressure.
4. An experienced operator must be present at the pressurization equipment control panel at all times while the cockpit is under pressure.
5. While the cockpit is under pressure, all personnel should stay clear of the canopy.
6. While the cockpit is being pressurized, pressure change should *never* exceed 1000 feet per minute altitude change, as indicated on a rate-of-climb indicator.

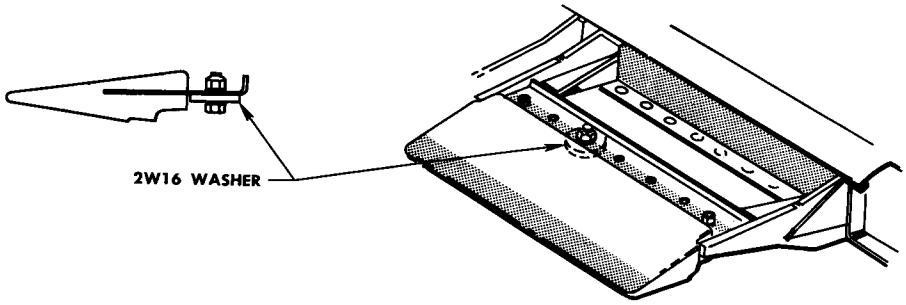
SEALING PRESSURIZED AREAS—DETECTING LEAKS WITH COCKPIT UNDER PRESSURE.

1. With the cockpit depressurized, let mechanic selected to do repair work enter.
2. Pressurize cockpit to required test pressure, or to maximum pressure attainable below required test pressure, using a rate of pressure increase not exceeding ½ psi per minute (a rate of descent of 1000 feet per minute on a vertical velocity indicator).
3. With cockpit pressurized, on outside of airplane locate leaks with a castile soap solution. Isolate leaks on inside by running a stethoscope or a piece of rubber tubing along seams in leakage area, listening for a change in sound when tube is passed over a leak.
4. Repair leaks until it is not possible to detect any change in sound when stethoscope or rubber tubing is passed over location, or until someone on outside indicates that leak has been stopped.

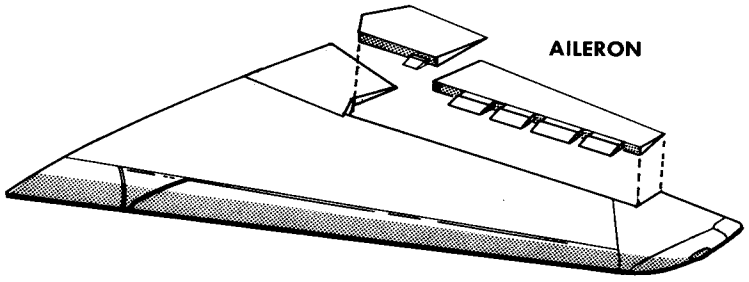
REBALANCING CONTROL SURFACES

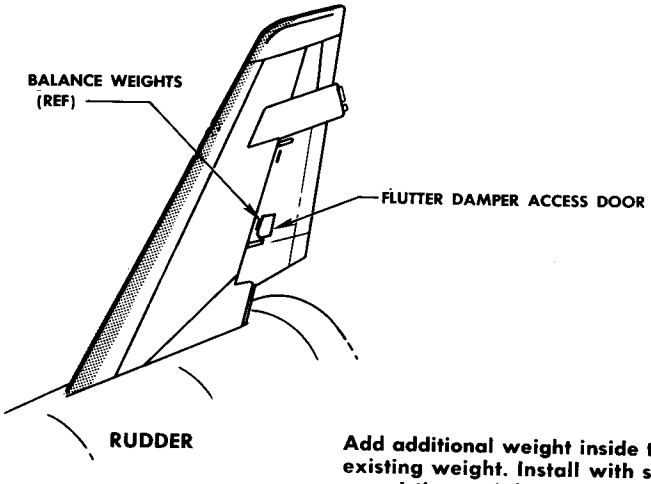
GENERAL



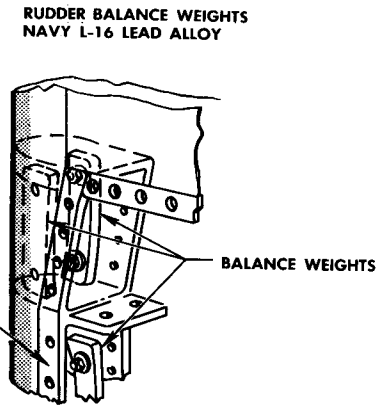


Balance ailerons by adding 2W16 washers* singly at each hole, working consecutively from outboard to inboard end of aileron, or by removing 2W16 washers, working from inboard to outboard end of aileron. On the outboard aileron, the maximum is one 2W16 washer per bolt on all weights. On the inboard aileron the maximum is three 2W16 washers per bolt.





Add additional weight inside this bay opposite existing weight. Install with same size screws as existing weight is installed with.



The upper rudder panel may be balanced by adding weight at lower end of rudder in case of tail heaviness or removing part of the existing weight in case of nose heaviness. Access is through the flutter damper access door.

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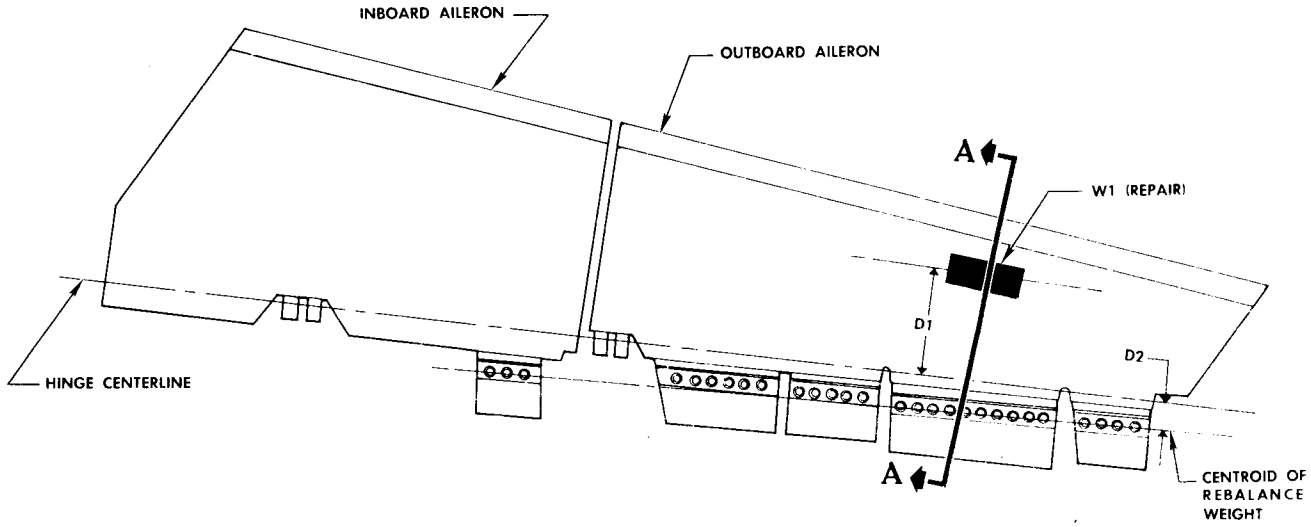
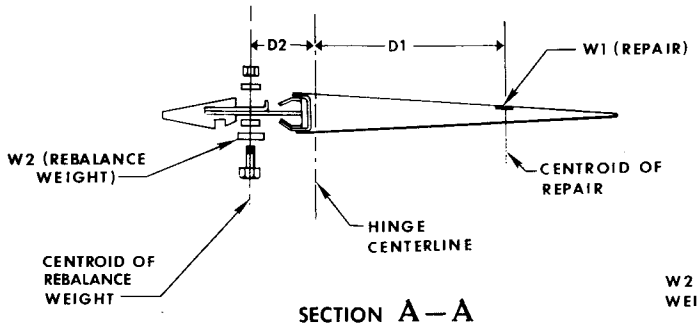
Figure 1-40. Rebalancing Flight Control Surfaces

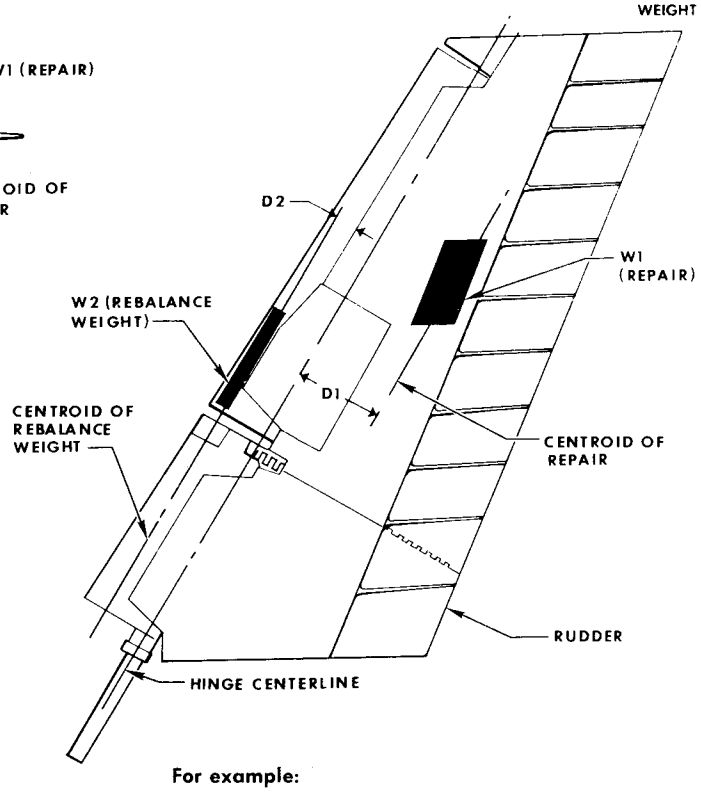
MECHANICAL BALANCE CHECK

GENERAL





SECTION A—A



D1 represents the distance in inches from center of gravity (centroid) of repair to the hinge centerline (fulcrum).

D2 represents the distance in inches from center of gravity (centroid) of rebalancing weight to hinge centerline (fulcrum).

W1 represents the net weight of repair materials (gross weight of added repair materials minus the weight of damaged material removed).

W2 represents the required weight necessary to rebalance control surface.

To determine effect of repair on control surfaces, multiply W1 (weight) by D1 (distance).

For example:

W1 = 1 pound
D1 = 8 inches

The effect of repair on balance would be:

1 pound × 8 inches = 8 inch-pounds underbalance.

When effect of repair causes underbalance, use following equation to determine weight necessary to rebalance.

For example:

W2 (weight necessary to balance effect of repair) $= \frac{W1 \times D1}{D2}$

W1 = 1 pound
D1 = 8 inches
D2 = 4 inches

$W2 = \frac{1 \text{ pound} \times 8 \text{ inches}}{4 \text{ inches}} = 2$ pounds required to rebalance control.

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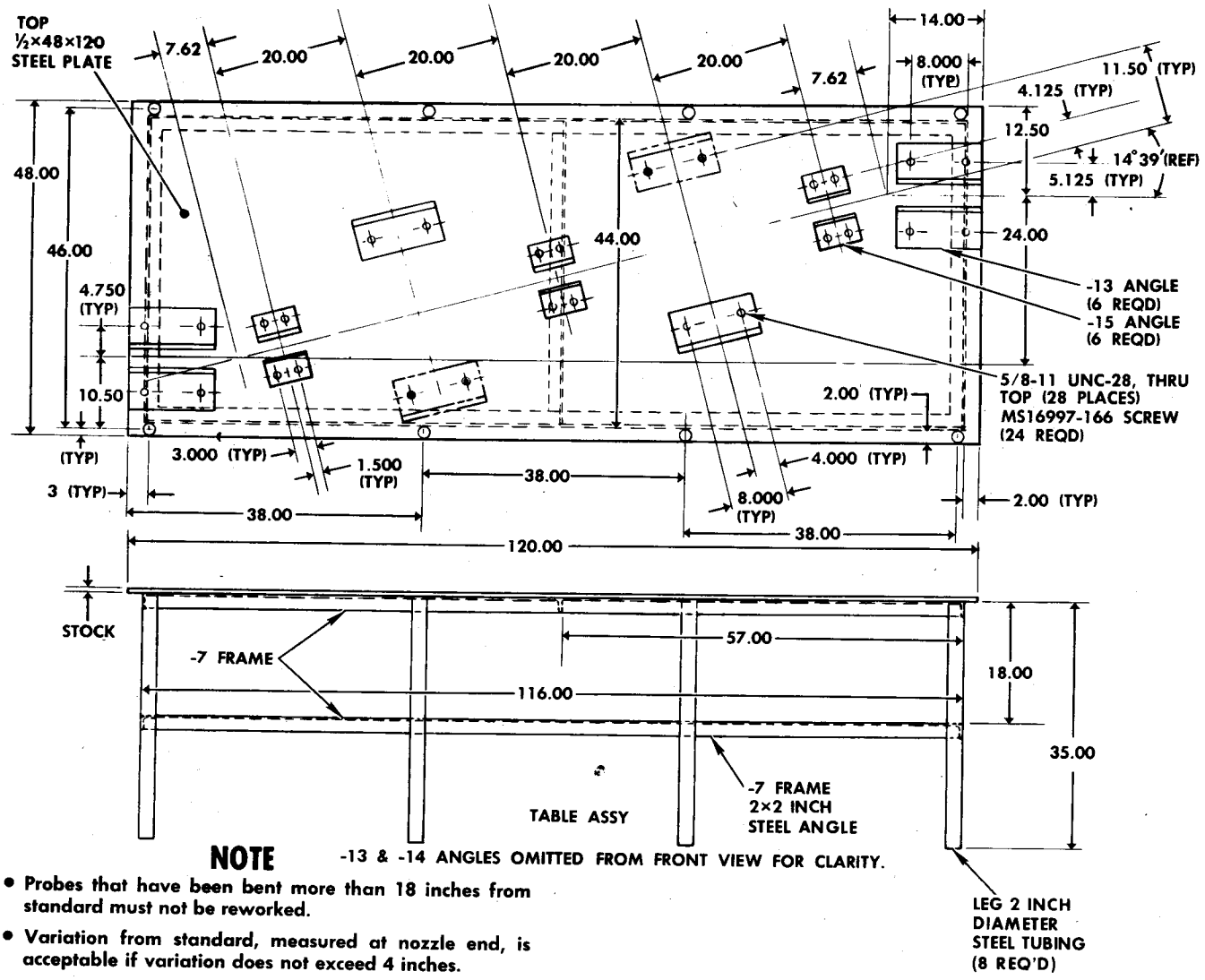
Figure 1-40A. Mathematical Balance Check



NOTE -13 & -14 ANGLES OMITTED FROM FRONT VIEW FOR CLARITY.

- Probes that have been bent more than 18 inches from standard must not be reworked.
- Variation from standard, measured at nozzle end, is acceptable if variation does not exceed 4 inches.
- Probes must not be straightened more than three times. Straightened probes should be identified by metal impression stamping on the butt end as follows:
  1st straightening — STR-1
  2nd straightening — STR-12
  3rd straightening — STR-123
- Probes must be straightened by use of approved tube bending equipment.
- After straightening, probes must be inspected by Zyglo or an equal inspection method.

Rework criteria:

a. Dents greater than 1/32 inch in depth at the nozzle end or 1/16 inch in depth at the butt end are cause for rejection.

b. Scratches or gouges exceeding 1/64 inch in depth at the nozzle end or 1/32 inch at the butt end of the probe are cause for rejection. Scratches and gouges of lesser dimensions may be sanded out.

c. Flat areas are acceptable if the probe variation, 90 degrees apart, at any point, does not exceed 1/4 inch.

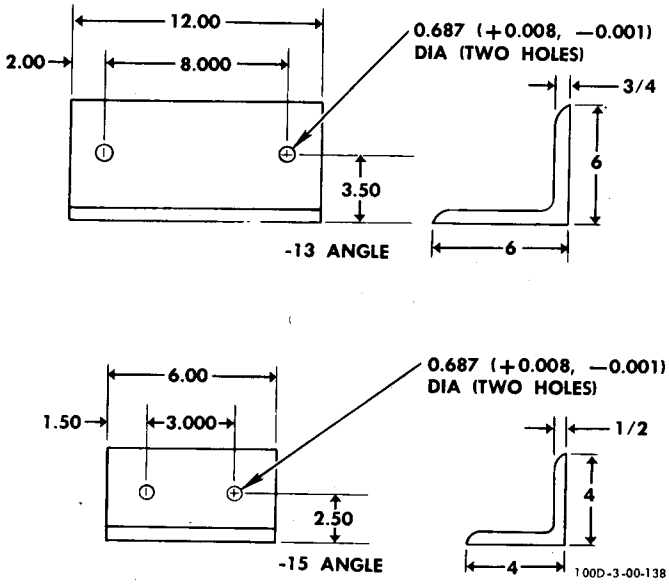


Figure 1-40B. Realigning In-flight Refueling Probe

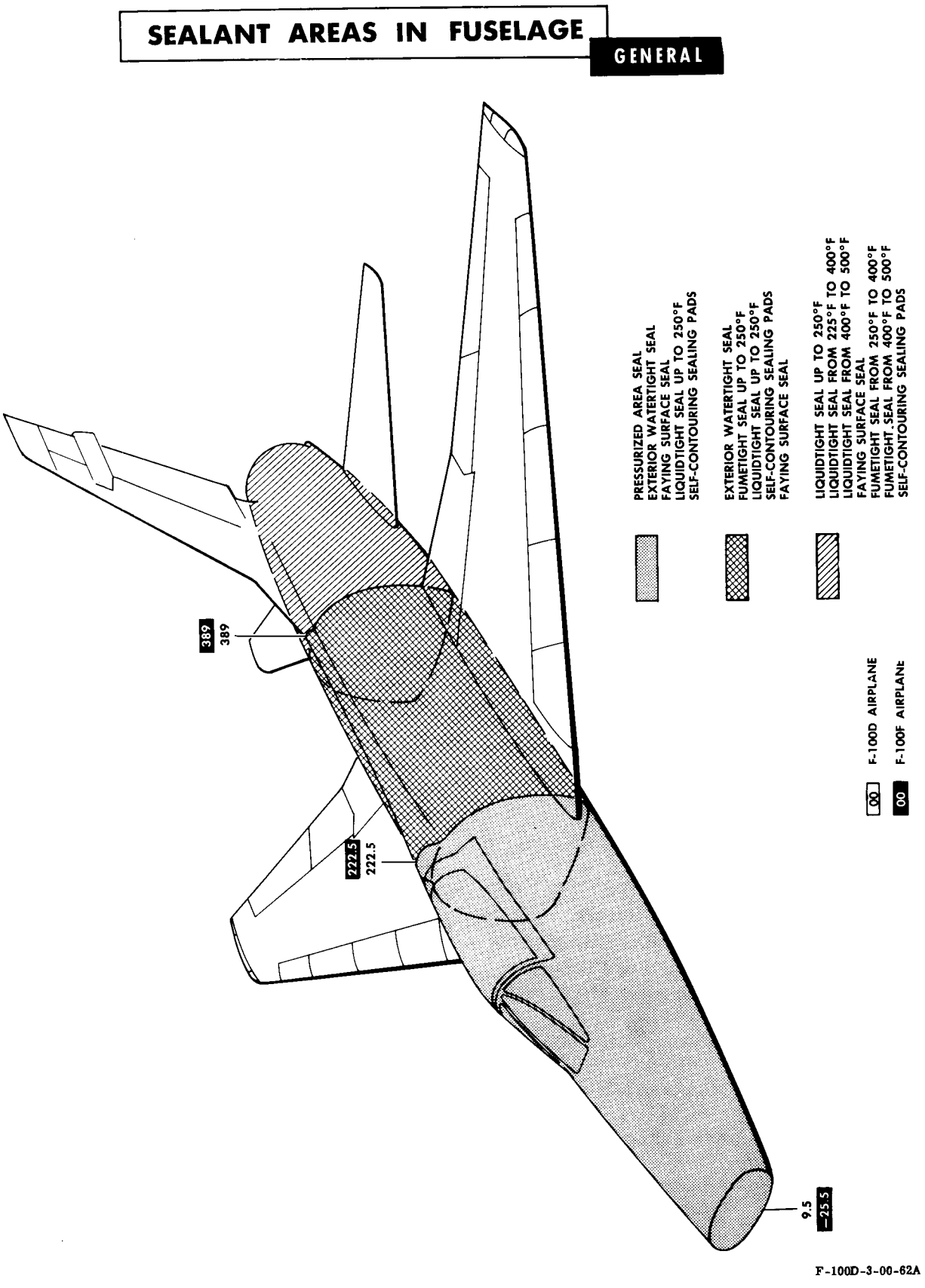


Figure 1-41. Areas Where Sealants Are Used

PRESSURIZED COCKPIT SECTION

GENERAL

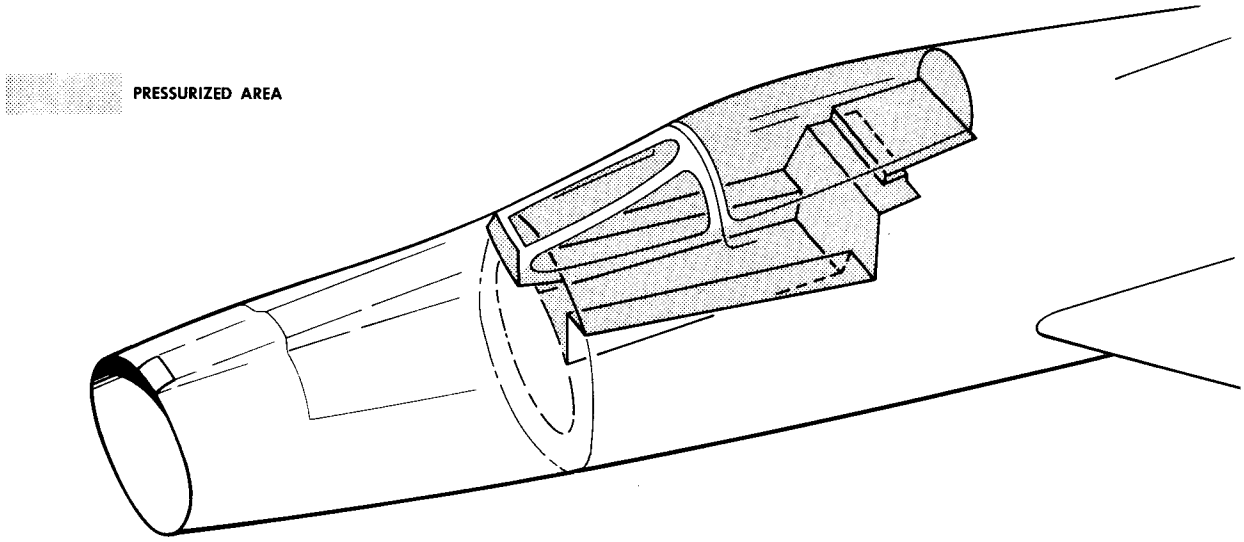


Figure 1-42. Pressurized Section

5. After repairs have been completed under pressure, depressurize cockpit at a rate of descent not to exceed 1000 feet per minute.

Air-tight seals are provided for all joggles, seams, butt and lap joints, holes, slots, rivets, etc, in the pressurized area.

SEALING PRESSURIZED AREAS—PRESSURE-SEALING MATERIALS.

MATERIAL	SOURCE
Naphtha	Specification MIL-N-15178
Toluol	Federal Spec TT-T-548
Sealing compound, general purpose	Specification MIL-S-7502, Class B-4
Pro-Seal 414, corrugation sealant	Coast Pro-Seal & Mfg Co
EC-612P	Minnesota Mining and Mfg Co
Cleaning compound, De-seal A or equivalent	Kelite Products, Inc
Coating, Vulcabond V-36	American Latex Products Corp
Coating, A56B	B. F. Goodrich Co
Tape, general purpose, pressure sensitive, Type II, Grade C	Specification JAN-P-127

(Refer to index for source of commercial products.)

SEALING PRESSURIZED AREAS—MIXING AND HANDLING OF SEALING COMPOUND MIL-S-7502.

The base compound and accelerator should be mixed together in the proportions recommended by the manufacturer. It is important that the two parts be blended until the accelerator is uniformly distributed throughout the mixture (there should be no evidence of streaks or discoloration). A fillet made from sealant which is not mixed thoroughly will have areas which do not cure completely.

At room temperature (about 77°F), the sealing compound must be used within three hours total time after having been mixed with the accelerator or removed from cold storage. After addition of the accelerator, the sealing compound may be stored at 0°F to −10°F for 72 hours.

SEALING PRESSURIZED AREAS—CLEANING SURFACES TO BE SEALED.

The surface to which the sealant is to be applied must be absolutely clean. Remove all foreign material with air hose, vacuum cleaner, or clean, dry cloth. Moisten clean cloth with naphtha or solvent and wipe all surfaces to be sealed. Surface should be wiped dry before solvent evaporates. Always pour solvent onto cloth to prevent contamination of solvent supply. It is important that clean cloths be used. Surface is completely

clean when, after wiping the surface, the cloth is not soiled.

Before applying sealing compound to any painted surface, check to see if paint is adhering to surface satisfactorily. If paint is removed easily when scratched with thumbnail, it is not satisfactory and should be removed before applying sealing compound.

Scrub paint to be removed with a cloth soaked with wash thinner, (Federal Specification P-S-661). When paint is dissolved, clean with cloth wet with wash thinner or other suitable solvent. Wipe dry with clean cloth.

SEALING PRESSURIZED AREAS— SEALING PROCEDURE.

Seal joggles first by forcing compound under the joggle from either side. If joggle is too small to seal from the side, it may be sealed by drilling a No. 40 hole in joggle as shown in figure 1-43. Apply compound through hole until it comes out from both sides of joggle.

CAUTION

When it is necessary to drill joggles, be sure necessary steps are taken to prevent damage to structure which is underneath the joggle.

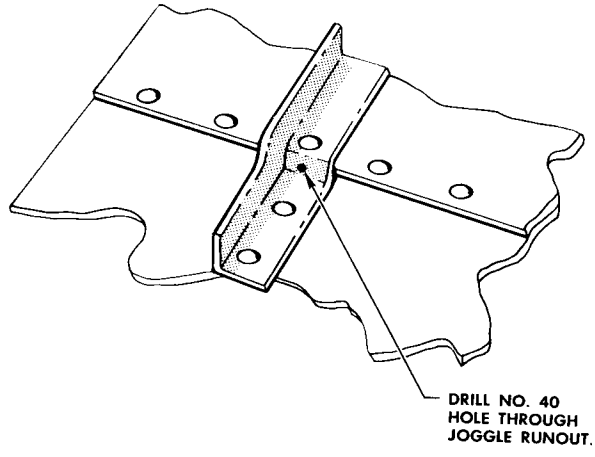


Figure 1-43. Sealing Joggle at Drilled Hole

Seal holes and slots up to ⅛-inch in diameter by filling with sealing compound and building up a bead at least ⅛-inch high. (See figure 1-44.) Holes and slots greater than ⅛-inch in diameter, but not larger than ¼-inch are sealed by applying a layer of pressure-sensitive tape over the holes on the pressure side (tape to extend ½-inch beyond edge of hole in all directions) and then a layer of sealing compound which extends at least ⅛-inch beyond edge of the tape in all directions. Tape is used under the sealing compound to prevent pressure from causing the compound to be blown through the hole. Where pressure-sensitive tape cannot be used, fill hole completely with EC-612P sealing compound, forming a fillet on the pressure side. Then apply a layer of general purpose sealing compound over the EC-612P. (See figure 1-45.) Holes greater than ¼-inch in diameter must be filled with a rivet, balsa wood plug, or a cap to reduce the size of the opening below ⅛ inch. Seal hole with layer of sealing compound which extends at least ¼-inch beyond edge of hole in all directions.

Slots and joggles greater than ¼ inch must be closed with sheet metal similar to that used in surrounding structure.

After all holes and joggles have been sealed, the seams and joints are sealed. With a pressure applicator or spatula, apply bead of sealing compound carefully along seams and joints so there are no bridges or gaps in the bead.

Seal rivets and bolts as necessary. Brush a coat of Pro-Seal No. 414 or general purpose sealing compound on all bolts and rivets in pressurized area.

Note

General purpose sealing compound may be thinned to brush consistency by mixing one part toluol to two parts of sealing compound by volume.

CAUTION

Toluol and the solvent used in Pro-Seal No. 414 are highly inflammable and toxic. Safety precautions must be observed while using these materials to prevent fire. If used in a closed area, forced ventilation must be provided for personnel.

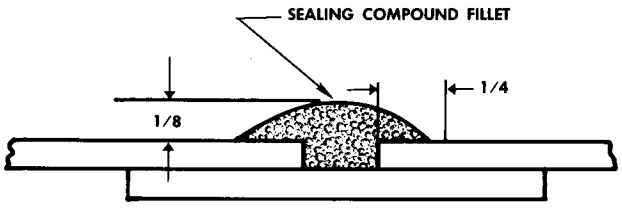
If it is necessary to apply sealant to any rubber parts, the rubber must first be cleaned with toluol.

Note

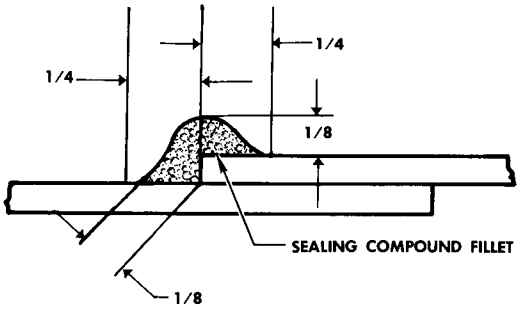
Toluol will remove B. F. Goodrich A-56B coating. If this coating is required, apply two brush coats of A-56B to exposed rubber.

TYPICAL PRESSURE SEALING

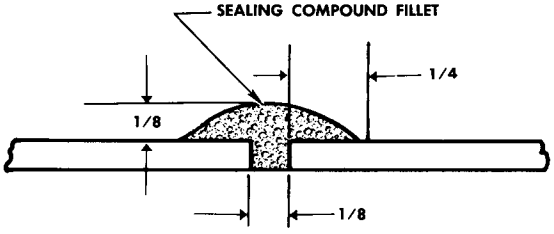
GENERAL



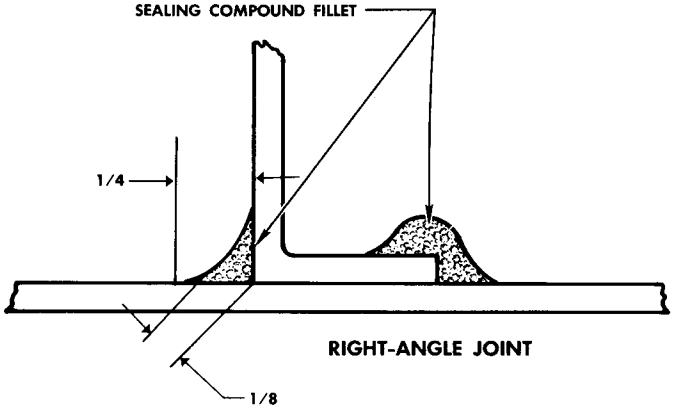
BUTT JOINT



LAP JOINT



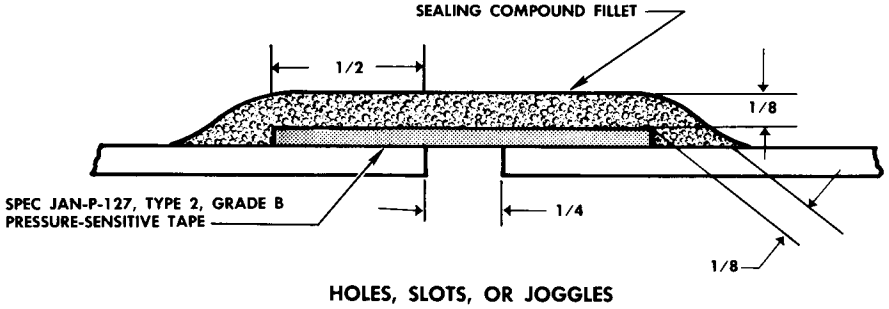
HOLES, SLOTS, OR JOGGLES



RIGHT-ANGLE JOINT

Caution

Before making repairs, read text on cockpit pressure sealing. (Refer to table of contents for page number.)



HOLES, SLOTS, OR JOGGLES

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Figure 1-44. Typical Pressure Sealing

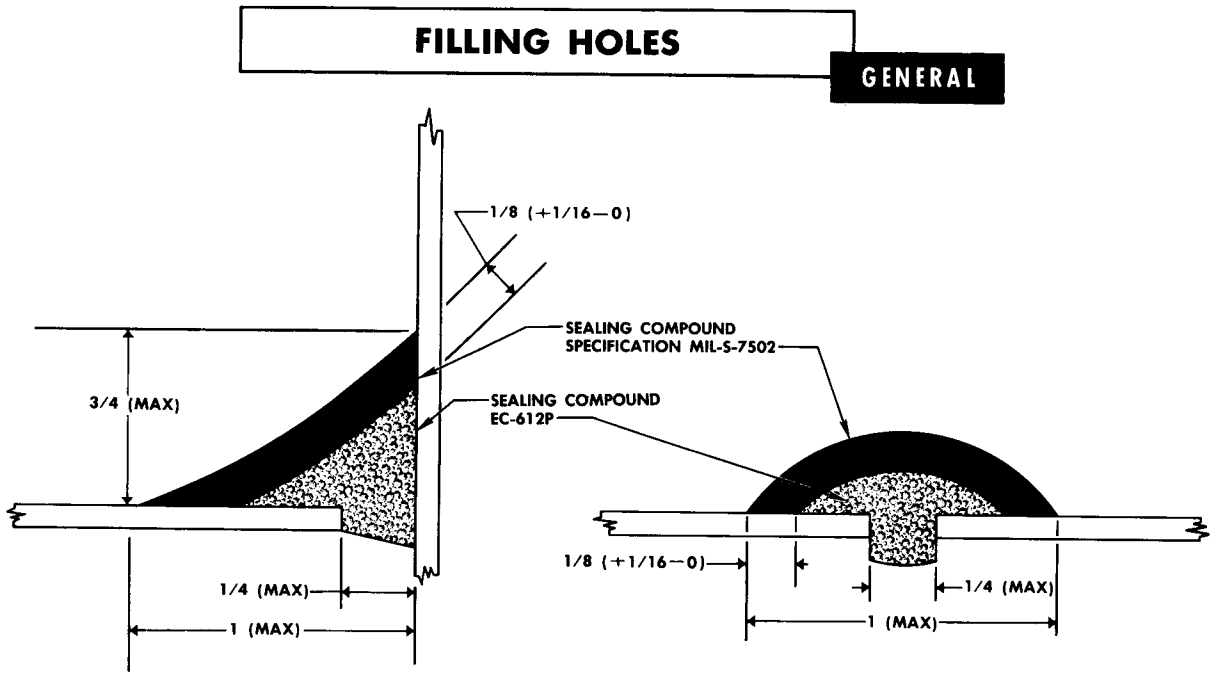

Figure 1-45. Filling Holes Where Tape Cannot Be Used

Apply coating of Vulcabond V-36. When Vulcabond V-36 is no longer tacky, the sealant may be applied.

All equipment used, and any excess sealant must be cleaned up within three hours after exposure to room temperature. Use naphtha, toluol, or Deseal A for cleaning.

LIQUID-TIGHT SEALING.

There are some areas in the airplane where leakage of fluids such as hydraulic oil, fuel, water, etc, must be avoided, or controlled drainage provided. Since these areas are in different locations throughout the airplane, a wide range of operating temperatures exist between these various areas. For this reason, a sealant which may be used satisfactorily in one area, may not be satisfactory in another. Liquid-tight sealants are provided for three temperature ranges:

- Temperatures up to 250°F
- Temperatures from 225°F to 400°F
- Temperatures from 400°F to 500°F

All areas where liquid-tight sealing is required must be completely sealed.

LIQUID-TIGHT SEALING—TEMPERATURES UP TO 250°F.

Use the same materials and procedures as outlined under "Sealing Pressurized Areas."

LIQUID TIGHT SEALING—TEMPERATURES FROM 225°F TO 400°F.

This sealing process is used in areas where temperatures may be as high as 400°F in normal operation, with occasional temperatures up to 500°F for short periods.

LIQUID-TIGHT SEALING MATERIALS—TEMPERATURES FROM 225°F TO 400°F.

MATERIAL	SOURCE
Solvent, cleaning	Federal Spec P-S-661
Pro-Seal high-temperature sealing compound No. 714	Coast Pro-Seal & Mfg Co
Pro-Seal catalyst No. 714A	Coast Pro-Seal & Mfg Co

(Refer to index for source of commercial products.)

MIXING AND HANDLING SEALING COMPOUND FOR LIQUID-TIGHT SEALS—TEMPERATURES FROM 225°F TO 400°F.

Thoroughly mix 1½ parts catalyst (curing agent) by weight to each 100 parts of base compound by weight. After having been mixed, the sealant has a working life of from 1 to 2 hours in an open container when

exposed to room temperature. It may be stored at 5°F for a period of two weeks. If no low-temperature storage is available, it may be kept for 24 hours at room temperature if it is kept in a tightly closed container.

LIQUID-TIGHT SEALING PROCEDURES—TEMPERATURES FROM 225°F TO 400°F.

Procedures for the use of this sealant are the same as those outlined under "Sealing of Pressurized Areas." However, balsa wood plugs may not be used for sealing at these temperatures.

LIQUID-TIGHT SEALING—TEMPERATURES FROM 400°F TO 500°F.

In areas where liquid-tight sealing is to be used and the continuous operating temperatures may reach 500°F, the following sealing process is used.

LIQUID-TIGHT SEALING MATERIALS—TEMPERATURES FROM 400°F TO 500°F.

MATERIAL	SOURCE
High-temperature sealing compound, EC-1548	Minnesota Mining and Mfg Co
Sealing tape, unfused Teflon No. 547 (one-inch wide by 5 mils thick)	Minnesota Mining and Mfg Co
Coating compound, No. C-136	American Latex Products, Inc
Cement, No. EC-613B	Minnesota Mining and Mfg Co
Cleaning solvent	Federal Spec P-S-661

(Refer to index for source of commercial products.)

HANDLING SEALANT FOR LIQUID-TIGHT SEALS—TEMPERATURES FROM 400°F TO 500°F.

Stir the high-temperature sealing compound thoroughly before use. If it appears caked or lumpy it should be discarded. The compound cannot be thinned successfully with water or solvent of any kind. The container should be kept tightly covered when not being used. Freezing will damage the compound. The high temperature sealing compound must not be disturbed while it is hardening during the 24 hours following application. It should not be applied until the job has progressed to a point where the compound may be allowed this undisturbed period. Hot water may be used to clean equipment.

SURFACE PREPARATION FOR LIQUID-TIGHT SEALS—TEMPERATURES FROM 400°F TO 500°F.

Refer to "Sealing Pressurized Areas—Cleaning Surfaces to be Sealed."

SEALING PROCEDURE FOR LIQUID-TIGHT SEALS—TEMPERATURES FROM 400°F TO 500°F.

For sealing of this nature, faying surfaces in riveted assemblies are sealed by a single thickness of sealing tape. To form a gasket for bolted assemblies, use 3 layers of sealing tape. The tape may be held in place during assembly with EC-613B cement. Apply tape while cement is wet. Sealing of holes, joggles, slots, etc, is done with high temperature sealing compound in the same manner as outlined under "Sealing Pressurized Areas—Sealing Procedures." However, balsa wood plugs are not to be used for sealing at these temperatures. Twenty-four hours after application of the high temperature sealant, apply 2 brush coats of Stabond C-136 coating. Allow three hours drying time between each coat.

Note

Do not allow this sealant to be subjected to sharp blows or movement during or after curing.

FUME-TIGHT SEALING.

There are some areas in the airplane where excessive leakage of air, fumes, smoke, etc, may be dangerous. Therefore, these areas must be sealed. Since these areas are in different locations in the airplane, there is a wide range of temperatures between these areas during airplane operation. A sealant which may be used satisfactorily in one area may be unsatisfactory in another area.

Fume-tight sealants are provided for three temperature ranges:

- Temperatures up to 250°F
- Temperatures from 250°F to 400°F
- Temperatures from 400°F to 500°F

FUME-TIGHT SEALING—TEMPERATURES UP TO 250°F.

The same materials and procedures apply as outlined in "Sealing Pressurized Areas," except that for this particular sealing process only, small leaks, such as might be found at rivets, seams, and joints, need not be sealed.

FUME-TIGHT SEALING—TEMPERATURES FROM 250°F TO 400°F.

This process is included for areas in which a fume-tight seal is required and where temperatures normally range from 250°F to 400°F, but may at times reach a maximum of 500°F.

The same materials and procedures are used as those outlined under "Liquid-tight Sealing—Temperatures From 225°F to 400°F."

FUME-TIGHT SEALING—TEMPERATURES FROM 400° TO 500°F.

Fume-tight sealing in areas where normal operating temperatures exceed 400°F is done using same materials and procedures as outlined under "Liquid-tight Sealing—Temperatures from 400°F to 500°F."

SEALING FAYING SURFACES.

Faying surfaces may be sealed using general purpose sealing compound (refer to "Sealing Pressurized Areas") or Pro-Seal 714 (refer to "Liquid-tight Sealing —Temperatures From 225°F to 400°F.") The type sealant used is determined by the normal operating temperature of the area to which it will be applied.

Fit components of structure to be sealed together and drill rivet and bolt holes as required. Disassemble components, clean thoroughly and check paint. (Refer to "Sealing of Pressurized Areas.") Apply a smooth, even coat of the correct sealant, about 1/32-inch thick, to the faying surface area of one component using either a spatula or pressure applicator. As soon as possible after application of the sealing compound, rivet or bolt the components of the structure together, squeezing out the excess compound.

EXTERIOR WATER-TIGHT SEALING.

In some areas of the fuselage it is necessary to protect against the possibilty of moisture entering the interior compartments. The exterior water-tight sealant is used at the windshield, canopy, electrical compartment, etc. It is applied to the exterior surfaces of aircraft structures.

Applicable Materials—Water-Tight Sealing

MATERIAL	SOURCE
Solvent, cleaning	Federal Spec P-S-661
Toluol	Federal Spec TT-T-548
Naphtha	Specification MIL-N-15178
Methyl ethyl ketone	Federal Spec TT-M-261
Sealing Compound PR-341	Products Research Co
Accelerator (Curing Agent) PR-341A	Products Research Co
Cleaning Compound LAR1634	Turco Products Co
Deseal A	Kelite Products, Inc

(Refer to index for source of commercial products.)

Surface Preparation.

Paint and primer coats must be removed from the area to be sealed using wash thinner and cloths or methyl ethyl ketone and a stiff bristle brush. Paint must be removed for a minimum of ½ inch on each side of skin gaps, and skin gaps must be thoroughly clean.

All surfaces to be sealed must be thoroughly cleaned. (Refer to "Sealing Pressurized Areas—Cleaning Surfaces to be Sealed.")

MIXING AND HANDLING SEALING COMPOUND FOR EXTERIOR WATER-TIGHT SEALING.

Prepare the sealing compound for use by mixing ten parts by weight of PR-341 base compound with one part by weight of PR-341A accelerator (curing agent). The materials should be mixed until no particles of accelerator are evident in the base compound.

At room temperature (about 77°F), the sealing compound must be used wiithin three hours total time after having been mixed with the accelerator or removed from cold storage.

Note

After addition of the accelerator, the sealing compound may be stored at 0°F to −10°F for 72 hours.

SEALING PROCEDURE FOR EXTERIOR WATER-TIGHT SEALING.

Apply masking tape to skin on each side of skin gap to be sealed leaving 1/16 (±1/32) inch of skin exposed on each side of gap.

Note

Tape should be applied immediately after cleaning the area with solvent. Do not allow fingers to touch cleaned surface as tape is applied.

Apply the sealant with a pressure applicator or clean wooden spatula. Work sealant into, and along, the seam. After the seam has been sealed, remove excess compound and masking tape. Tape must be removed before compound starts to set. After removing tape,

fair out edges of sealing compound. This may be done by moistening the finger with water and drawing the moistened finger along the edge of the compound. Clean up excessive compound and equipment within four hours using methyl ethyl ketone, Toluol or Deseal A.

SELF-CONTOURING SEALING PADS.

Self-contouring sealing pads (figure 1-46) are applied to stationary structure at removable doors or between sections of a structure which may normally be separated.

The surface preparation, materials applicable, and mixing and handling methods covered under "Exterior Water-Tight Sealing," this section, also apply to this process. The sealing procedure too, is the same, with the following exceptions.

1. The mixed sealant is applied to the cleaned stationary part of the assembly to be sealed.

2. Apply petrolatum (Federal Specification AN-P-51) to the removable part, to screws, and quick-disconnect fasteners. This prevents the sealing compound from sticking to the removable part and fasteners.

3. Assemble the parts being sealed and allow to remain in position for about eight hours, or until the compound has cured.

4. After the sealant has cured, disassemble the parts and trim off excess sealant with a sharp knife. Remove all traces of petrolatum with naphtha.

Note

Tooling holes and gaps may be backed up with general purpose pressure-sensitive tape, Specification JAN-P-127 Type II, Grade C. The tape must be removed after the sealant has cured.

HONEYCOMB SANDWICH PANELS.

Honeycomb sandwich panel construction is used in two basic types of airplane structure: that type of structure which must carry or transmit loads (primary structure), e.g., speed brakes; and that type used in fairing or streamlining (secondary structure). Construction of both types is generally the same, consisting of a core made of aluminum honeycomb and sheet-metal face sheets bonded (cemented on each side). A very high-strength, light structure can be built in this way. The structural or load-carrying type repair requires the replacement of damaged material so that the load can be transmitted through the repair. Minor damage may be considered nonstructural, and filling the damage only is all that is required. No loads will be carried through this type of repair.

MATERIALS FOR HONEYCOMB SANDWICH PANEL REPAIR.

MATERIAL	SOURCE
Adhesive, EC-1469	Minnesota Mining & Mfg Co
Epoxy filler, Corefil 615	Bloomingdale Rubber Co
Curing agent U	Shell Chemical Corp
Aerodynamic filler, PR-341 and PR-341 accelerator	Products Research Co
Toluene	Federal Spec TT-T-548
Methyl ethyl ketone (MEK)	Federal Spec TT-M-261
Sulphuric acid, specific gravity 1.84	Commercial
Sodium dichromate, technical grade	Commercial
Talc, white powder, technical grade	Commercial
Honeycomb core $\frac{1}{8} \times 0.002$	Specification MIL-C-7438
Balsa wood	Specification MIL-S-7998
Sandpaper, 320 grit, wet or dry	Minnesota Mining & Mfg Co
Cellophane sheet	Commercial
Glass wool	Commercial
Naphtha	Federal Spec TT-N-95
Wash primer	Specification MIL-C-15328A
Zinc-chromate primer	Specification MIL-P-8585
Aluminized lacquer	Specification MIL-L-7178
Sealant injection gun	AF Stock No. 4940-345-1178
Clean dry rags	Commercial
250-watt infrared heat lamp	Westinghouse Electric Corp

SURFACE PREPARATION FOR ADHESIVE BONDING.

The strength of the bond depends largely upon the cleanness and proper preparation of the bonding surfaces. The importance of proper surface preparation cannot be overemphasized.

PREPARATION OF METAL SURFACES.

1. Remove all dirt, grease, and similar surface contamination with toluene. Apply toluene to bonding surfaces with clean rags. To prevent contamination of toluene, pour toluene onto rags or surfaces to be cleaned rather than dipping rags into toluene container. *Use only clean toluene in clean cans.* Do *not* permit toluene to air-dry on surfaces, but wipe dry with clean, dry rags.

2. Lightly sand bonding surfaces with clean 320-grit sandpaper and wipe off loose particles with clean rags.

3. Apply the following acid cleaner to bonding surfaces for 20 to 25 minutes at a solution temperature of at

SELF-CONTOURING SEALING PADS

GENERAL

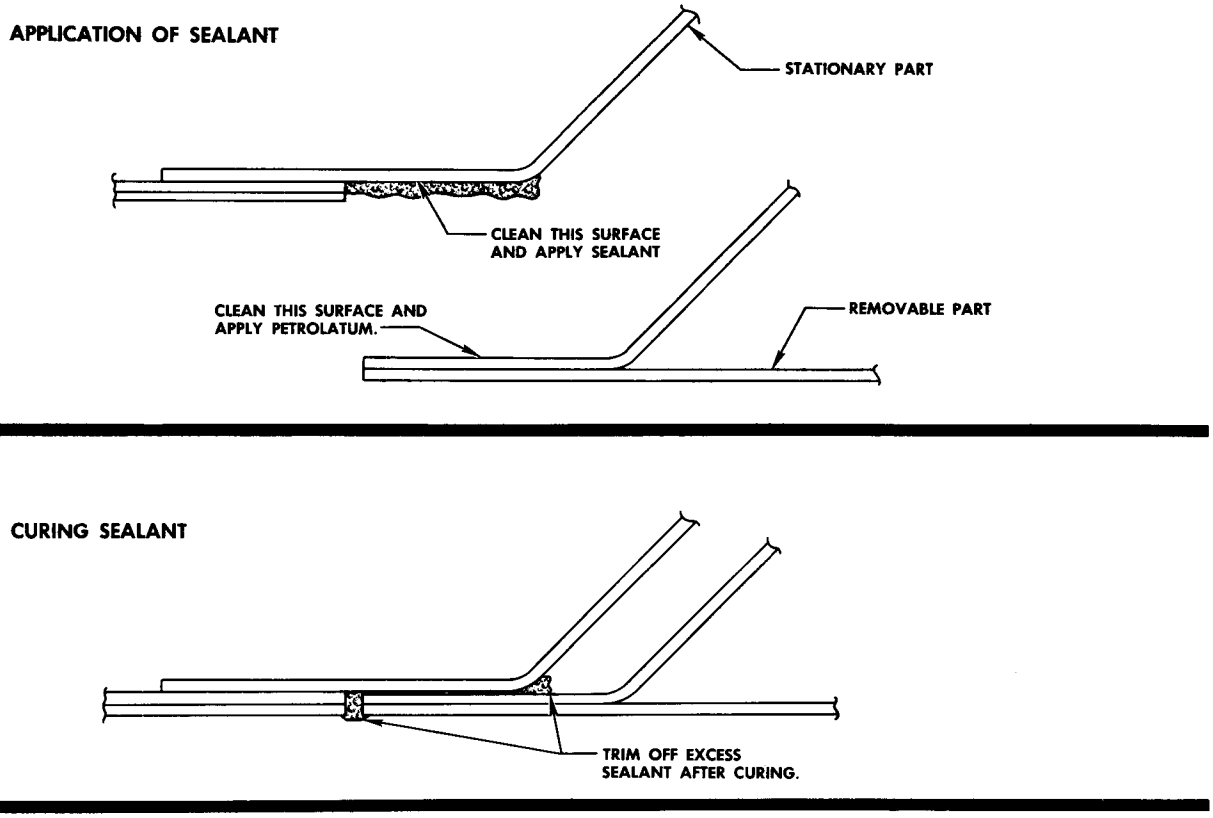

Figure 1-46. Self-contouring Sealing Pads

least 70°F. Apply cleaner by immersion or swabbing with glass wool.

Sulphuric acid, specific gravity 1.84.....55 parts by weight
Sodium dichromate .....10 parts by weight
Water .....80 parts by weight

4. Rinse thoroughly with water and check surface for water-break. If water on surface tends to form drops, repeat cleaning and rinsing procedures until a water-break-free surface is obtained.

5. Dry surfaces thoroughly. Do not heat over 150°F.

Note

When applying acid cleaner to vertical surfaces or areas near cracks or holes, add sufficient talc to cleaner to form a thick paste which will not run or sag. *Do not permit acid cleaner to run into areas that cannot be flushed with water.*

CAUTION

Use rubber gloves and adequate eye protection when handling acid cleaner.

PREPARATION OF HONEYCOMB CORE.

If a vapor degreaser is available, suspend core in vapor zone until condensation ceases. Otherwise, flush core with clean toluene.

PREPARATION OF BALSA WOOD.

Lightly sand bonding surfaces with 320-grit sandpaper and wipe off all loose particles.

MACHINING HONEYCOMB CORE.

The honeycomb core must be cut to the desired thickness by sawing, milling, or other similar procedure, so that the core-to-face bond is continuous over 100 percent of the bond area. The operations of cutting, forming, and shaping must not cause buckling or crippling of the cell walls. The core must be cut sufficiently oversize in length and width dimensions to allow a slight compression of the core cells next to the edges and splice joints; the ribbon direction must be parallel to that of the original part being repaired. The core must be machined without lubricants and kept free of all oil and other contaminants during the machining operation.

Note

When noncompressible materials, such as metal, micarta, or wood, are being used as fillers to replace sections of honeycomb, the filler material should be trimmed to provide a clearance of 0.003 to 0.006 inch between all surfaces to be bonded. This clearance is necessary to provide for a sufficient amount of adhesive.

MIXING ADHESIVE AND FILLER MATERIAL.

The adhesive and fillers used for these repairs are two-part materials which begin to cure or harden as soon as the two components are mixed. Do not mix more material than can be used within the pot life of the mixture. The ratios of the materials and the pot life at room temperature are as follows:

MATERIAL	HARDENER	POT LIFE
EC-1469 adhesive, 100 parts	Curing agent U, 40 parts	20 to 30 minutes
Corefil 615 filler, 100 parts	Curing agent U, 20 parts	20 to 30 minutes
PR-341 filler, 100 parts	PR-341 accelerator, 10 parts	3 to 4 hours

Mix hardener thoroughly into adhesive or filler, using a spatula or putty knife. Observe the following precautions:

1. Mix in small quantities, one quart or less.
2. Ventilate mixing area and avoid breathing vapors.
3. Keep curing agent U and adhesive off hands and clothing.
4. In case of skin contact with curing agent U or mixed adhesive, wash immediately with soap and water or denatured alcohol.

NEGLIGIBLE DAMAGE.

Dents. Dents are considered negligible when occurring no closer than 12 inches from other damage or dents. The allowable depth of a dent in relation to the length of the dent is as follows:

LENGTH	DEPTH
½ to 1 inch	0.050 inch
1 to 2 inches	0.075 inch

Check all dents for delamination. (Refer to "Delamination Test of Honeycomb Panels.") Refer to "Aerodynamic Skin Filler" for filling dents.

NONSTRUCTURAL REPAIRS.

The restrictions imposed on nonstructural repairs are as follows: the minimum distance between damage is 12 inches; the maximum hole size is one inch in diameter. These restrictions are firm and must be adhered to, unless otherwise specified on the repair drawing. To repair nonstructural damage, proceed as follows:

1. Cut out damaged skin and core, using either a hole saw or a router and template.

2. Fill area flush with freshly mixed Corefil 615 filler. Avoid trapping any air in cavity.

3. Remove all excess filler from surrounding area with methyl ethyl ketone before filler hardens.

4. Allow repair to cure for at least 4 hours at a minimum temperature of 70°F. If ambient temperature is below 70°F, place a 250-watt heat lamp about 3 feet from repair. Do not permit temperature to exceed 180°F.

5. After repair has cured for required length of time, sand filler flush with adjacent skin areas, using 320-grit paper and being careful not to penetrate cladding.

STRUCTURAL REPAIRS.

The overlap requirements for skins or repair doublers when a structural repair is being made are as follows:

DAMAGE LENGTH	OVERLAP
Up to 2 inches	2 inches
2 to 4 inches	2½ inches
4 to 6 inches	2¾ inches
6 to 8 inches	3 inches

These overlap requirements must be used unless otherwise specified on the repair figure. To repair structural damage, proceed as follows:

1. Cut out damaged skin and core, using either a hole saw or a router and template.

2. Make replacement core from honeycomb core, balsa wood, or a drilled metal fillet block (as specified on the repair figure), to fit the exact area of removed core.

3. Trim and form replacement skins and doublers to match mold line of area being repaired.

4. Clean all surfaces to be bonded as specified in "Surface Preparation for Adhesive Bonding."

5. Apply a coating of freshly mixed EC-1469 adhesive, at least 1/32 inch thick, on all surfaces to be bonded. Fill all gaps between replacement core and existing core with EC-1469 adhesive. Assemble as quickly as possible. If repair cannot be completed within pot life of adhesive, remove all adhesive with toluene and repeat bonding procedure.

6. Cover repair area with cellophane and apply a pressure of from one to 10 psi by dead-weight loading or clamping.

7. Cure for a minimum of 4 hours at a minimum temperature of 70°F. If ambient temperature is below 70°F, place a 250-watt heat lamp about 3 feet from repair area. Do not permit temperatures to exceed 180°F during cure.

8. Remove pressure and sand excess adhesive flush with surrounding skin with 320-grit paper, being careful not to penetrate cladding.

Note

All structural repairs *must* be allowed to age-cure for a minimum of 7 days before being returned to service.

DELAMINATION TEST OF HONEYCOMB PANELS.

When tapped with a coin, the suspected delaminated area produces a dull tone if delaminated. If the area is not delaminated, a clear tone results. There is a slight difference in the tone produced by different filler materials—honeycomb, micarta, balsa wood, etc—but all have a clear ringing tone when struck if they are properly bonded. If a change in tone is detected, the area should be carefully sounded and the boundaries of the delamination marked with a grease pencil. With a little practice, the operator can outline the delaminated area with a high degree of accuracy.

REPAIRING DELAMINATED HONEYCOMB PANELS.

1. Depending upon the size of the delaminated area, drill two or more ⅛-inch diameter holes through delaminated skin near edge at opposite sides of delaminated area.

2. Inject freshly mixed EC-1469 adhesive into core through lower ⅛-inch hole until adhesive begins to flow out of higher hole.

3. Clean off excess adhesive with methyl ethyl ketone and cover filler holes with cellophane.

4. Apply light pressure against skin to force it down into contact with core.

5. Cure under pressure for a minimum of 4 hours at a minimum temperature of 70°F. If ambient temperature is below 70°F, place a 250-watt heat lamp about 3 feet from repair area. Do not permit temperatures to exceed 180°F during cure.

6. Remove pressure and sand excess filler flush with surrounding area, using 320-grit paper and being careful not to penetrate cladding.

Note

All delamination repairs *must* be allowed to age-cure for a minimum of 7 days before being returned to service.

AERODYNAMIC SKIN FILLERS.

Aerodynamic skin filler (PR-341) is a two part material: filler and accelerator. It is used extensively throughout the airplane for filling skin and door gaps. The cured material is silver-gray in color and feels like hard rubber. It may be sanded to blend with the surrounding surface and painted over. Filling a depression of less than 0.050 inch in the skin with PR-341 filler is not satisfactory; the filler would be too thin and would not stay in place.

To fill shallow dents or depressions less than 1/16 inch deep and less than 3 square inches in area, use Flex-Bond filler No. B-88 (FSN 8040-NI217115) or any of the following equivalent aerodynamic fillers, all listed under FSN 8030-693-2311: Preslite 108, EC-967, or Metlite 70-30. Each of these aerodynamic fillers will adhere in shallow dents and feather out well.

Either aerodynamic skin filler may be used where screw heads have been recessed 0.015 inch or more below the surface of the skin. The filler is applied in the same manner as for skin surfaces. A screwdriver of the proper type should be inserted into the screw recess to keep out filler and to locate the screw for future removal.

AERODYNAMIC SKIN FILLER MATERIALS.

MATERIAL	SOURCE
PR-341 filler	Products Research Co
Flex-Bond No. B-88 filler	Taylor and Art, Inc
Cleaner	Specification MIL-C-5410, Type I
Isopropyl alcohol	Commercial
Naphtha	Federal Spec TT-N-95
Wash primer	Specification MIL-C-15328A
Zinc chromate primer	Specification MIL-P-6889
Aluminized lacquer	Specification MIL-L-7178

Refer to index for source of commercial products.

MIXING PR-341 AERODYNAMIC SKIN FILLER.

Mix the base compound and accelerator together in proportions recommended by the manufacturer. It is extremely important that complete mixing of the accelerator and filler be accomplished. At room temperature (about 77°F), the filler should be used within three hours after accelerator has been added. The accelerated filler may be stored at 0°F to −10°F for a period of 72 hours if stored immediately after mixing.

APPLYING AERODYNAMIC SKIN FILLER.

1. Remove all primers or organic paint finishes to a distance of one inch along each side of the area to be filled.

2. After surfaces are clean, wipe the area to be filled with cleaner and brightener (Specification MIL-C-5410, Type I). Allow the cleaner to remain on the surface for two or three minutes, and then rinse it off with water. Water rinse may be followed by isopropyl alcohol rinse to speed drying. Allow all surfaces to dry thoroughly.

3. Force the filler into the damaged area with a knife or spatula. Smooth to desired contour by overfilling, and sanding down after material has cured.

4. The filler must be allowed a full 24 hours at room temperature to cure. Curing may be accelerated with heat lamps, but temperatures should not exceed 160°F.

RAM COIN DIMPLES

GENERAL

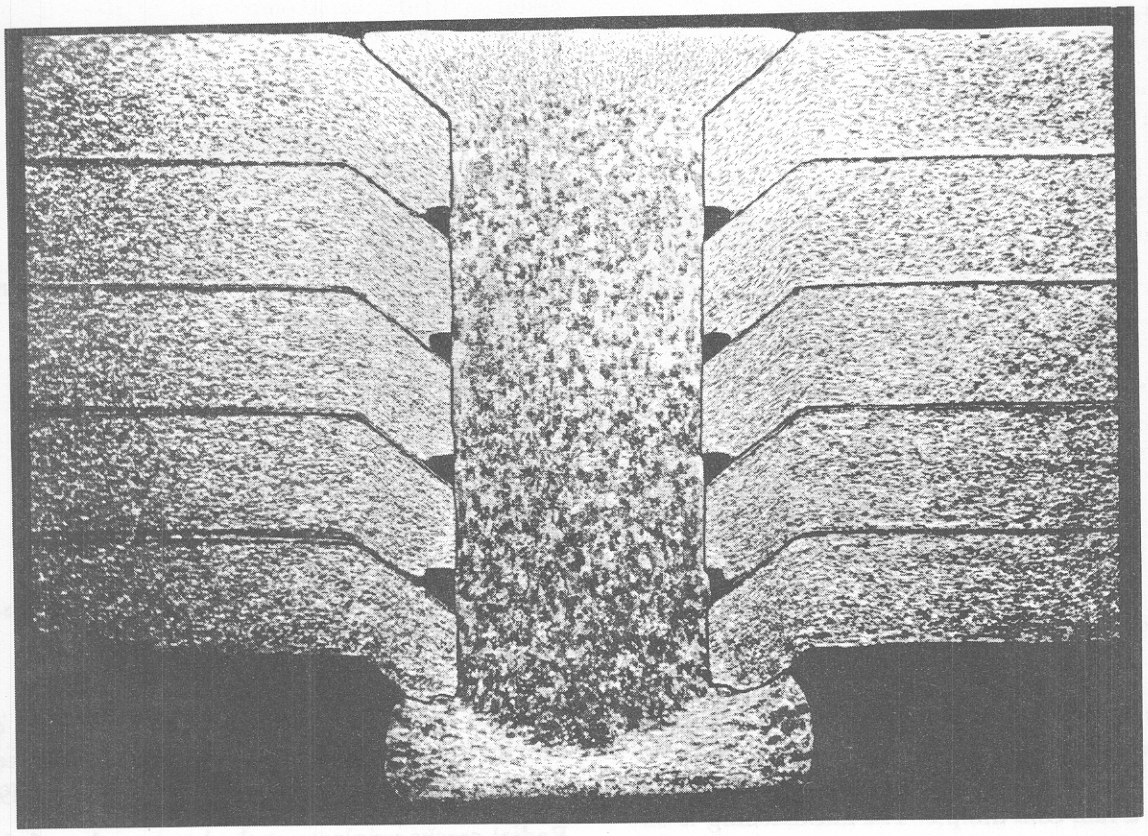

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Figure 1-47. Ram-coin Dimples Nested

5. After completion of cure, clean the affected area with diluent naphtha (Federal Specification TT-N-95). Then finish with one coat of wash primer (Specification MIL-C-15328), one coat of zinc-chromate primer (Specification MIL-P-6889), and two coats of aluminized lacquer (Specification MIL-L-7178). Apply these according to Specification MIL-C-8507, Specification MIL-P-6808, and Specification MIL-M-5055, respectively.

RAM-COIN DIMPLING.

DESCRIPTION OF RAM-COIN DIMPLING.

Ram-coin dimpling differs from all forms of radius dimpling in that the female die is fitted with a tube called a coining ram. The coining ram has a preset pressure which resists the downward pressure of the male die. It grips the lower side of the sheet close to the drilled hole and holds the material in closely against the pilot of the male die. The lower, outer edges are prevented from spreading out, thereby minimizing the chances of radial cracking. The upward pressure of the coining ram also forces the material into the radiuses of the dies, forming a dimple which closely matches the contours of the dies. Coin dimples nest (fit) closely into each other and into countersunk substructures. (See figure 1-47.)

WHY RAM-COIN DIMPLING IS USED.

All skin dimpling has been done by the ram-coin method. Dimpling of repair and replacement parts which are to nest into original dimples must be done with tooling which produces dimples of exactly the same contour and angle as the original. The strength of riveted joints in the repairs in this handbook presume the joints to be properly formed and mated and sound riveting practices employed. Dimples formed by radius dimpling have contours and angles which are different

RADIUS DIMPLE NESTED INTO RAM COIN DIMPLE

GENERAL

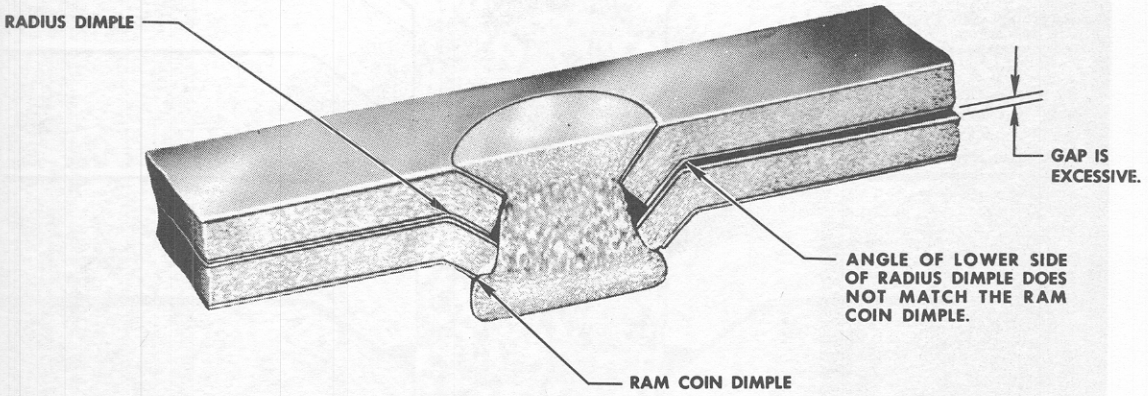

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Figure 1-48. Radius Dimple Nested Into Ram-coin Dimple

from those formed by the ram-coin method, therefore they will not test properly together. Figure 1-48 shows a sectional view of a riveted joint with a radius dimple nested into a ram-coin dimple. The mismatching is obvious. The misalignment caused by attempting to draw the oversized radius dimple into the smaller ram-coin dimple has caused a joggle in the rivet and the shop-formed point to be formed at one side.

Modified radius dimpling, into which ram-coin dimples will nest, may be used in substructures when space limitations do not permit the use of ram-coin tools. One radius dimple will not be nested into another radius dimple, and the top or outer skin will be ram-coin dimpled.

WHY RAM-COIN HEAT DIMPLING IS USED.

Dimpling with heated dies is necessary on 7075-T6 aluminum alloy, titanium, and titanium alloys in order to produce dependably sound dimples.

2024-T3 aluminum alloy can usually be successfully dimpled using cold dies for the smaller diameter head sizes. Heat should be applied to the dies when using the hand tool equipment, and also, when dimpling for such fasteners as screws, dzus, or other types requiring a deep draw formation. It is also desirable to apply heat if excessive warpage of the part occurs.

Dimples are subject to three types of failures during fabrication; radial cracks, circumferential tension cracks, and internal shear cracks. (See figure 1-49.)

Radial cracks originate at the bottom edge of the hole and extend along its radius. They are easily detected by visual inspection.

Circumferential tension cracks originate at the bend and extend around the circumference of the dimple. They are similar to cracks in straight bends when insufficient bend radius has been used.

Internal shear cracks are most common in the heavier gages of 7075-T6 aluminum alloy. They occur within the structure of the metal and in the case of clad material cannot be detected by visual inspection. The clad covering in this case is not broken. Internal shear cracks are just below the bend radius of the dimple and actually occur during the first one-third of the forming operation. Since this type of defect cannot be seen, proper machine setting and bend test strip results must be relied upon for ensurance of sound dimples.

All the cracking conditions described result from attempts to form relatively brittle materials. The heat from controlled heating of the dies mildly anneals the material in the dimple area and, in conjunction with the coining action, permits the necessary deformation.

DIMPLE DEFECTS

GENERAL

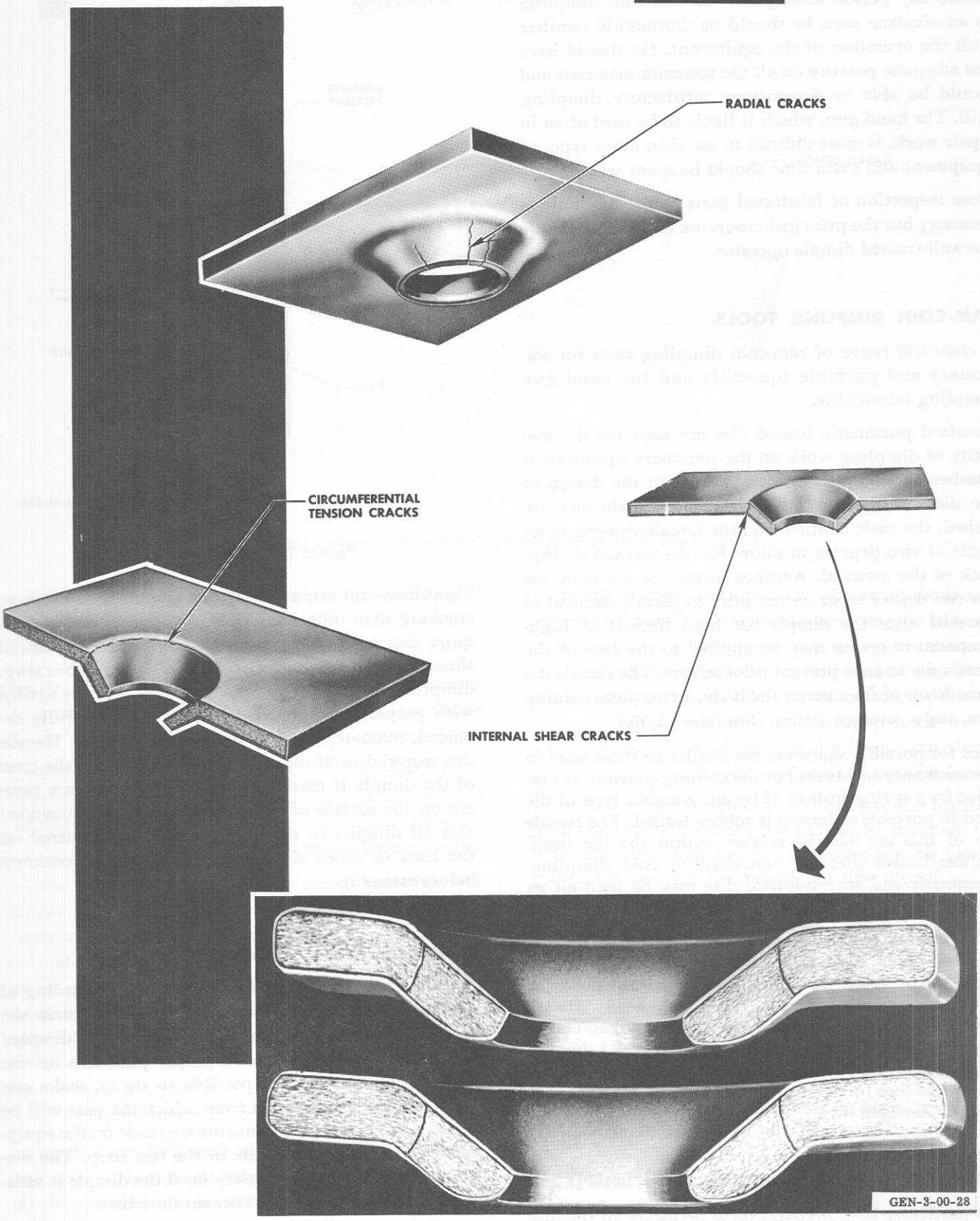

Figure 1-49. Dimple Defects

TRAINING RAM-COIN DIMPLE OPERATOR.

Before any person attempts to do ram-coin dimpling on an airplane part, he should be thoroughly familiar with the operation of the equipment. He should have had adequate practice on all the common materials and should be able to demonstrate satisfactory dimpling skill. The hand gun, which is likely to be used often in repair work, is more difficult to use than other types of equipment and extra time should be spent with it.

Close inspection of fabricated parts before assembly is necessary but the principal ensurance of a sound dimple is a well-trained dimple operator.

RAM-COIN DIMPLING TOOLS.

A complete range of ram-coin dimpling tools for stationary and portable squeezers and for hand gun dimpling is available.

Standard pneumatic loaded dies are used for the majority of dimpling work on the stationary squeezers. A number of features are incorporated in the design of the dies. The face of the male and female dies are dished, the male concave and the female convex, at an angle of two degrees to allow for the normal spring-back of the material. Another feature of the male die is a two degree taper on the pilot to permit removal of material after the dimple has been formed. A high-temperature grease may be applied to the face of the female die to help prevent pilot seizure. The female die is made up of four parts: the body, a two-piece coining ram, and a retainer spring. (See figure 1-50.)

Dies for portable squeezers are similar to those used in the stationary squeezers but the coining pressure is supplied by a spring instead of by air. Another type of die used in portable squeezers is rubber loaded. The female die of this set has the rubber within the die itself. Rubber-loaded dies are restricted to cold dimpling. Pneumatic and spring-loaded dies may be used either hot or cold.

All dimpling equipment should be handled with care. Control boxes, heater elements, and cords are easily damaged by rough handling. The dies must never be allowed to contact each other in the squeezer. Always have a clean sheet of metal between them when checking adjustments. Be sure the dies are clean and that the coining ram is free to move before starting to dimple. Check the dies for cracks, nicks, or dents. Do not try to rework damaged dies.

HOLE PREPARATION FOR RAM-COIN DIMPLES.

No particular hole preparation is necessary in the aluminum and steel alloys. A cleanly drilled hole is satisfactory.

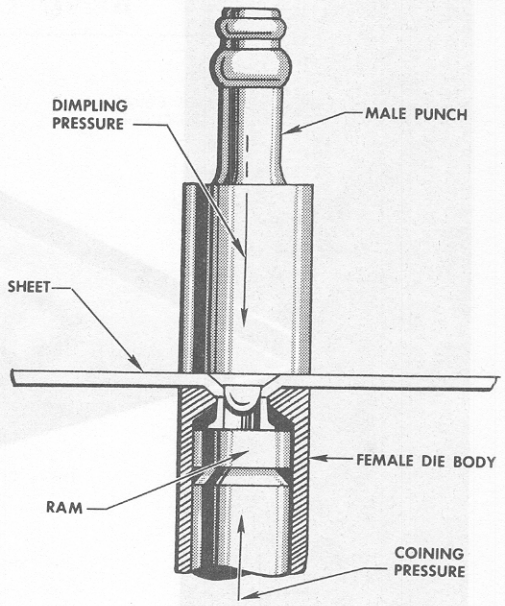

Figure 1-50. Ram-coin Dies

Titanium and titanium alloys are more subject to cracking than other materials in common use, and require that extra precautions be taken from the initial shearing to forming, fitting, drilling, deburring, dimpling, and final riveting. Titanium must be drilled with properly sharpened drills and then carefully deburred, removing as little material as possible. Handle this material carefully, as cracks may occur at the crest of the dimple if marks, scratches, or grooves are present on the surface of the material. It is recommended that all dimples in titanium material be deburred on the back or down side, and inspected for smoothness before assembly.

RAM-COIN DIMPLE BEND-TEST STRIPS.

A satisfactory bend test shall precede any dimpling of airplane parts made of 7075-T6 material. A strip the same thickness and temper as the sheet to be dimpled is selected and drilled to the proper pilot size for the dies to be used. (When possible to do so, make test strips from the same sheet from which the part will be made.) Preliminary adjustments are made on the equipment and a dimple is made in the test strip. The machine is readjusted as necessary until the dimple is satisfactory in appearance and fastener flushness.

A bend test is made by placing the strip in a vise across the center of the dimple, and bending it back and forth

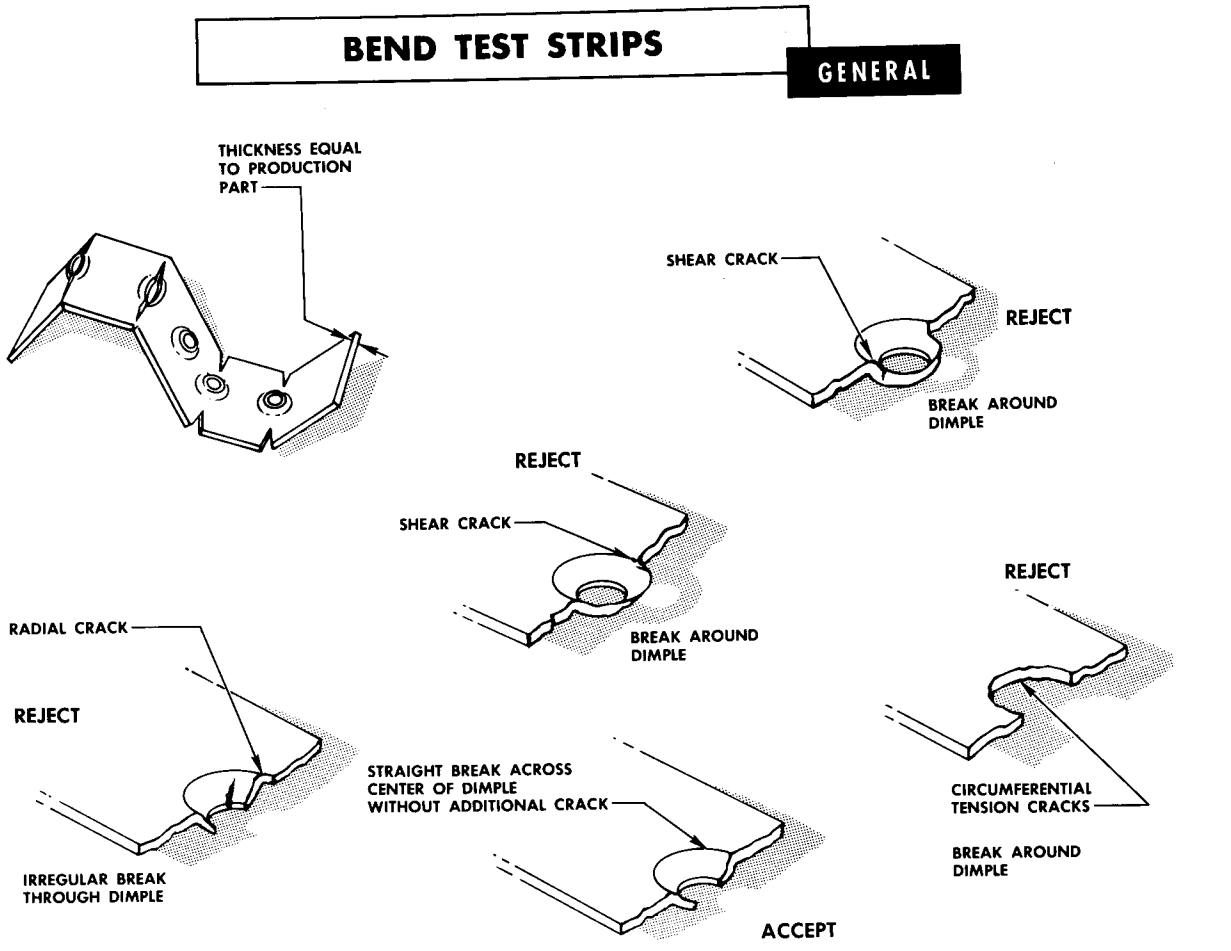

Figure 1-51. Bend-Test Strips

through the dimpled as well as the undimpled portion. Break at least two dimples in each direction. If a circumferential fracture is present, the break will usually follow the path of least resistance and break out a part of the dimple or the entire dimple. (See figure 1-51.) Bend tests for other materials are not required. However, the test strip should be used to obtain proper setup of the machine before dimpling of the airplane part is started. Critically examine the test dimples for appearance and flushness of sample fastener.

APPEARANCE AND REQUIREMENTS OF RAM-COIN DIMPLES.

There are four basic factors to be considered in determining ram-coin dimple quality:

1. Dimple formation and sharpness of break. Because of coining preload and dimpling pressure, a coin dimple may be formed with a sharp break from the sheet surface into the dimple. Definition should be checked with the corresponding fastener for flushness and to be certain that the correct size dies have been used.

2. Amount of coining. The coining ram preload pressure is the main factor in prevention of cracked dimples in aluminum alloy material. Too much preload will tend to extrude the coined area, and insufficient preload will not give the proper support to prevent cracking. Coining pressure is correct when there is a slight undercut of the flat on the bottom.

3. Sheet warpage. While it is practically impossible to completely eliminate sheet growth that results in warpage, it can be held to a minimum with correct dimpling and coining pressures. Because edge distance and hole layout pattern of the production part are different from those of the test strip, some pressure correction may be required on the production part.

4. Condition of the dimple. To determine whether it is free of cracks, the dimple should be carefully examined for any flaws that may result from damaged or

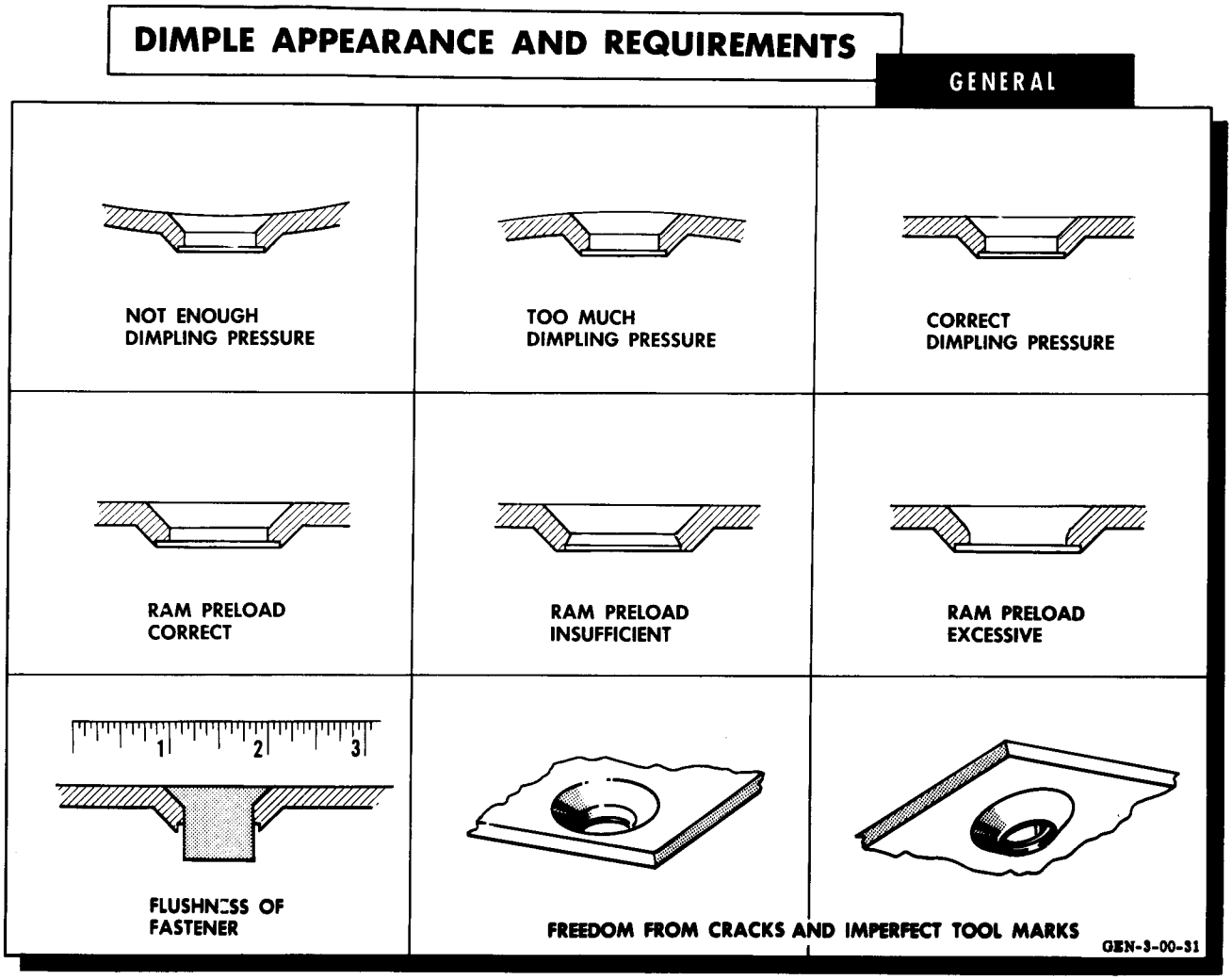

Figure 1-52. Dimple Appearance and Requirements

dirty dies, accumulated chips, inferior material, cracked coining rams, or other causes. Also, look for flat on the bottom (with slight undercut). (See figure 1-52.)

COUNTERSINKING SUBSTRUCTURES FOR RAM-COIN DIMPLES.

When a dimple nests into a subcountersink, there shall be a gap between the mating parts to ensure complete bearing of the dimple into the countersink. This gap shall be between a minimum of 0.0005 inch and a maximum of 0.0008 inch. (See figure 1-53.)

FASTENERS.

HOLES FOR BLIND FASTENERS.

Hole sizes for blind fasteners are critical because of the amount of expansion built into the fastener. Holes to be dimpled for blind fasteners shall be pilot drilled, dimpled, and redrilled to size. All holes should be checked with "go, no-go" gages before installing fasteners.

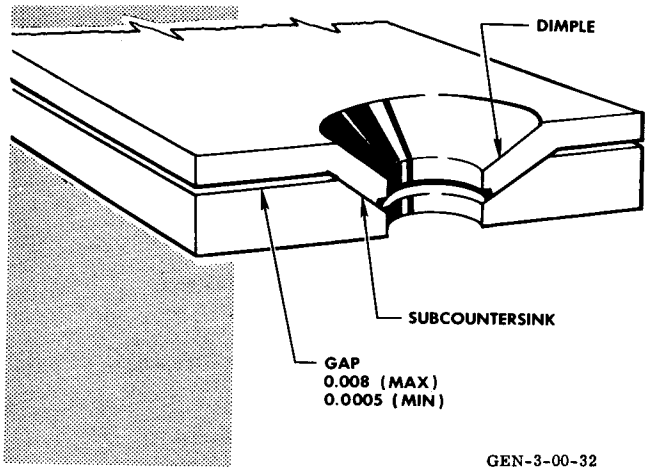

Figure 1-53. Dimple Into Subcountersink

FASTENER, DIE, & PILOT DRILL SIZES

GENERAL

SIZE & TYPE OF FASTENER	CODE NUMBER ON MALE & FEMALE DIES	PILOT DRILL SIZE
FLUSH-DRIVEN RIVETS 3/32 100 DEG 1/8 100 DEG 5/32 100 DEG 3/16 100 DEG 1/4 100 DEG	AN426-3 3/32 AN426-4 1/8 AN426-5 5/32 AN426-6 3/16 AN426-8 1/4	NO. 40 (0.097-0.100) NO. 30 (0.1290-0.1305) NO. 21 (0.160-0.162) NO. 11 (0.192-0.194) "F" (0.256-0.261)
FLUSH PULL RIVETS (CHERRY) 1/8 (CR-162 OR AF462-4) 5/32 (CR-162 OR AF462-5) 3/16 (CR-162 OR AF462-6)	AN426-4 1/8 AN426-5 5/32 AN426-6 3/16	NO. 30 (0.1290-0.1305) NO. 21 (0.160-0.162) NO. 11 (0.192-0.194)
EXPLOSIVE RIVETS 1/8 (DR-134-100 OR AF458-4) 5/32 (DR-173-100 OR AF458-5 3/16 (DR-204-100 OR AF458-6	AN426-4 1/8 AN426-5 5/32 AN426-6 3/16	NO. 30 (0.1290-0.1305) NO. 21 (0.160-0.162) NO. 11 (0.192-0.194)
HI-SHEAR RIVETS 3/16 100 DEG 1/4 100 DEG 5/16 100 DEG	2R2 OR 2R7-6 3/16 2R2 OR 2R7-8 1/4 2R2 OR 2R7-10 5/16	NO. 17 (0.172-0.176) NO. 1 (0.227-0.231) 9/32 (0.2802-0.2852)
FLUSH DEUTSCH DRIVE PINS 3/16 1/4 5/16	2R2 OR 2R7-6 3/16 2R2 OR 2R7-8 1/4 2R2 OR 2R7-10 5/16	NO. 17 (0.172-0.176) NO. 1 (0.227-0.231) 9/32 (0.2802-0.2852)
FLUSH SCREWS NO. 8 100 DEG NO. 10 100 DEG 1/4 100 DEG 5/16 100 DEG	NAS204 NO. 8 OR AN509 NAS205 NO. 10 OR AN509 NAS206 1/4 OR AN509 NAS207 5/16 OR AN509	NO. 29 (0.135-0.139) NO. 24 (0.151-0.155) NO. 5 (0.2045-0.2085) 1/4 (0.249-0.254)
FLUSH JO-BOLTS 3/16 1/4 5/16	NAS205 NO. 10 OR AN509 NAS206 1/4 OR AN509 NAS207 5/16 OR AN509	NO. 24 (0.151-0.155) NO. 5 (0.2045-0.2085) 1/4 (0.249-0.254)
FLUSH DZUS 5/16 (STUD) 5/16 (SUPPORT) 13/32 (STUD) 13/32 (SUPPORT)	FA-5 STUD 3/8 × 120 DEG FA-5 SPRING 3/8 × 110 DEG FA-6 1/2 STUD 1/2 × 120 DEG FA-6 1/2 SPRING 1/2 × 110 DEG	3/8 (0.374-0.379) 3/8 (0.374-0.379) 1/2 (0.499-0.504) 1/2 (0.499-0.504)
UNITED CARR AIRLOC NO. 5 NO. 98265	TJ403-100-303-3 FEMALE TJ403-100-308-7 MALE	9/32 (0.280-0.285)

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Figure 1-54. Fastener Die and Pilot-drill Sizes

HANDLING AND STORAGE OF BLIND FASTENERS.

The protective film on blind explosive and pull rivets should never be removed. On explosive rivets this film is necessary for proper heating, firing, and expanding of the rivet. Prevent dirt or grit from contacting pull rivets. A clean lubricated surface is necessary for proper stem installation and expansion.

Store explosive rivets in a cool, dry place, avoiding temperatures in excess of 100°F. Do not throw explosive rivets into trash containers. If rivets are not usable, place them in a wire basket and burn them.

Keep blind fasteners in boxes in which they are received, or in moisture-proof containers properly labeled and giving all information that appears on the manufacturer's box.

INSTALLATION OF BLIND FASTENERS.

The following blind fasteners are covered in detail in the "Aircraft Structural Hardware" manual, AN 01-1A-8: Cherry rivets, Huck rivets, Drill Lok-Rivets, Rivnuts, and explosive rivets.

In addition to the explosive rivets discussed in the reference manual are the sealed (noiseless) and high-temperature explosive rivets. The sealed explosive rivets have a plug in the cavity containing the chemical charge. They are used mostly in honeycomb sandwich construction where the blast from the open type rivet would damage the light expanded metal. High temperature explosive rivets are of both the open and closed types. They are made of nickel or nickel alloy and are used for joining stainless and titanium in areas where temperatures would be injurious to aluminum alloys.

WARNING

Sealed-type explosive rivets like the non-sealed type must not be driven where fuel or other inflammable vapors are likely to be present.

BLIND FASTENERS—DEUTSCH DRIVE PIN BLIND RIVET.

This fastener consists of two parts, either a flush or protruding brazier, a head sleeve of corrosion-resistant steel, and a hardened steel pin. (See figure 1-55.) The pin is driven into the sleeve after the rivet is firmly seated in the drilled or reamed hole. The action of the pin pushing out the restricting walls of the sleeve forms the upset, beginning with a swelling action in the second sheet and progressing into a definite shop-formed head on the blind side.

The protruding ridge on top of the sleeve head locks the pin in the rivet. The final strokes of the hammer form the locking material into the area provided by the chamfer on the drive pin head. After the pin is seated in the head access, a rotary action of the upsetting tool (hammer) smooths out the pin locking ring for improved flushness and locking action.

The Deutsch "SP" (short pin) rivet is the same as a standard Deutsch rivet except that the drive pin is 0.060 inch shorter. This type rivet is used in buried rivet application where clearance is restricted. "SP" rivets are available in 3/16-inch diameter and larger.

An "SP" rivet may be made by replacing the drive pin in a rivet of the correct grip length with a drive pin from a rivet one grip length shorter.

For removal of Deutsch rivets, see figure 1-56.

BLIND FASTENERS—JO-BOLTS.

Jo-bolts (figure 1-57) are blind fasteners of relatively high shear strength. They are used in many areas of the airplane where the installation of standard fasteners is difficult or impractical. Special tools are required for Jo-bolt installation. A wide range of grip lengths is available. Close-tolerance, reamed holes are required for good installation. Three types of Jo-bolts are available: protruding (hex) head, 100-degree countersunk head, and millable head. The millable-head fastener has a protruding (hex) head with a 100-degree taper on its lower side. After installation in a countersunk hole, the protruding part of the head is milled off.

For removal of Jo-bolts, see figure 1-58.

CAUTION

To prevent damage to the engine, do not use Jo-bolt fasteners in the engine air intake duct.

BLIND FASTENERS—CR762 RIVET.

The Cherry CR762 rivet is a wide-grip range, positive-hole-fill, high-clinch type blind fastener. The rivets may be installed with standard Cherry hand-powered or pneumatic guns. Modified controlled-stroke pulling heads for H80 Series and H90 Series are used. They are provided with replaceable spacer sleeves for positively controlling the length of pull. (Refer to "Table of Pulling Heads" for CR762 rivets.) When setting the rivet, the shank is expanded against the face of the back material. This results in a drawing action and clinches the two materials together. The pull is stopped at a predetermined

point without breaking the stem. When the rivet is set, the stem protrudes 1/16 inch above the rivet head and gives visual indication that the upset is properly formed, the hole is filled, and the rivet is properly set. The stems of installed rivets are trimmed with standard trimming equipment.

TABLE OF PULLING HEADS

Rivet Size (Inch)	Head Number
	H80 SERIES
1/8 dia	H 804U universal head H 804C countersunk head
5/32 dia	H 804U universal head H 804C countersunk head
3/16 dia	H 804U universal head H 804C countersunk head
	H90 SERIES
1/8 dia	H 904U universal head H 904C countersunk head
5/32 dia	H 904U universal head H 904C countersunk head
3/16 dia	H 904U universal head H 904C countersunk head
7/32 dia	H 904U universal head H 904C countersunk head
1/4 dia	H 904U universal head H 904C countersunk head
9/32 dia	H 904U universal head H 904C countersunk head

RIVET CALLOUT CHANGES.

BLIND RIVET CALLOUTS.

CURRENT*	OLD (AF)*	OLD (NAF)*	OLD (CHERRY)*
MS20600AD4-	AF463-4-	NAF1195-4B-	CR163-4-
MS20600B4-	AF463B4-	NAF1195-4B-	CR157-4-
MS20600AD5-	AF463-5-	NAF1195-5B-	CR163-5-
MS20600B5-	AF463B5-	NAF1195-5B-	CR157-5-
MS20600AD6-	AF463-6-	NAF1195-6B-	CR163-6-
MS20600B6-	AF463B6-	NAF1195-6B-	CR157-6-
MS20601AD4-	AF462-4-	NAF1195-4BC-	CR162-4-
MS20601B4-	AF462B4-	NAF1195-4BC-	CR156-4-
MS20601AD5-	AF462-5-	NAF1195-5BC-	CR162-5-
MS20601B5-	AF462B5-	NAF1195-5BC-	CR156-5-
MS20601AD6-	AD462-6-	NAF1195-6BC-	CR162-6-
MS20601B6-	AD462B6-	NAF1195-6BC-	CR156-6-

*Final dash number not shown

CONVENTIONAL RIVET CALLOUTS.

CURRENT	OLD
MS20426AD-	AN426AD-
MS20426DD-	AN426DD-

THREADED INSERTS—HELI-COIL.

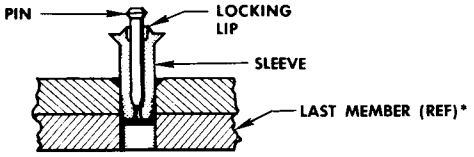
Heli-Coil inserts are made of 18.8 stainless steel (AMS-7245) helically coiled wire. The wire has a diamond shaped cross-section which accommodates internal and external threads at the same time. When assembled in a Heli-Coil tapped hole, the insert becomes a permanent standard internal thread with a class B tolerance. The same insert will produce other classes of fit depending upon tapped hole dimensions.

Heli-Coil inserts are available in two types; plain non-self-locking and self-locking mid-grip. Both types of inserts are made with a tang which is a portion of the bottom coil. The tang is offset to provide a driving member. The insert has a notch adjacent to the tang; this provides a break-off point for removing the tang after installation. (See figure 1-59.)

The mid-grip inserts, while not approved for general use, are approved for use in the F-100D and F-100F Airplanes. If a mid-grip type insert is removed, the same type should be reinstalled.

DEUTSCH RIVET INSTALLATION

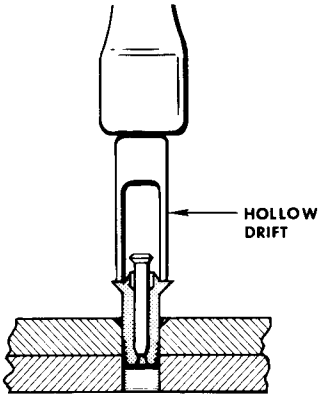
GENERAL



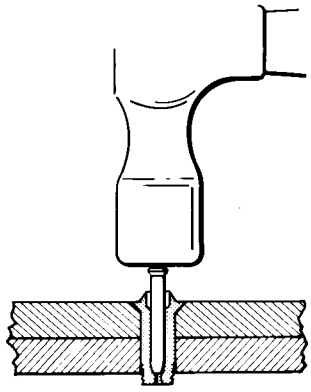
NOTE Total thickness of material being joined should be measured to determine correct length rivet to use. If measurement shows a borderline case between two different lengths, always use shorter length rivet. Dimpling tools for Hi-shear rivets may be used for Deutsch blind rivets.

1 Select proper length rivet (refer to note) and place in hole.

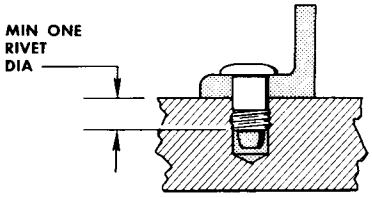
FASTENER SIZE—INCH	MIN THICKNESS OF LAST MEMBER*
3/16	0.125
1/4	0.188
5/16	0.250

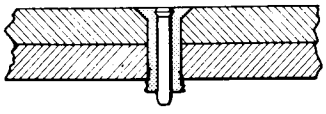


2 Use a hollow drift that will clear locking lip. Do not drive Deutsch drive pin blind rivets into tight holes by hammering on pin.



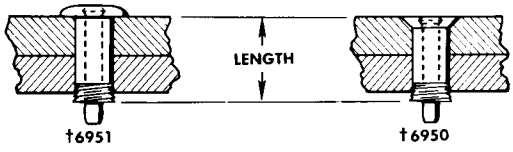
3 Rivets may be driven with air gun or hammer.





4 After pin has been seated in recess, smooth out the locking lip with a rotary or rolling action of the driving tool. If shaving is necessary for flushness, make certain that locking lip is not cut away to extent that lock is damaged or destroyed.

NOTE When rivet is installed in a buried hole, at least one times the rivet diameter (including locking rings) must extend into the hole.



CODING FOR SELECTION OF PROPER FASTENERS

6950 designates flush head.
6951 designates protruding brazier head.
First letter indicates sleeve material.
S—Steel
First dash number indicates rivet diameter in 1/32nds.
Second dash number indicates nominal grip in 1/16ths.

EXAMPLE:

6950-S-06-10 flush head—Cres—3/16 dia—5/8 nominal grip

FINAL HOLE SIZES

RIVET SIZE	DIAMETER	REAM
3/16	0.1915	0.191-0.193
1/4	0.260	0.259-0.262
5/16	0.316	0.315-0.318
3/8	0.386	0.385-0.388

LENGTH AND GRIP RANGE CHART

DASH NO.	NOMINAL GRIP	GRIP RANGE	LENGTH			
			3/16 DIA	1/4 DIA	5/16 DIA	3/8 DIA
-3	3/16	0.166—0.227	0.233			
-4	1/4	0.227-0.290	0.295	0.315	0.329	
-5	5/16	0.290-0.353	0.357	0.379	0.391	
-6	3/8	0.353-0.415	0.420	0.437	0.453	0.468
-7	7/16	0.415-0.478	0.482	0.500	0.516	0.530
-8	1/2	0.478-0.540	0.545	0.562	0.578	0.593
-9	9/16	0.540-0.603	0.607	0.625	0.641	0.656
-10	5/8	0.603-0.665	0.670	0.687	0.703	0.718
-11	11/16	0.665-0.728	0.732	0.749	0.766	0.780
-12	3/4	0.728-0.790	0.795	0.817	0.828	0.843
-13	13/16	0.790-0.853	0.857	0.877	0.891	0.905
-14	7/8	0.853-0.915	0.920	0.940	0.953	0.968
-15	15/16	0.915-0.978	0.982	1.000	1.016	1.030
-16	1.	0.978-1.040	1.045	1.062	1.078	1.093

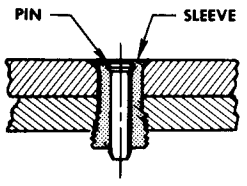
†The Deutsch Co. Refer to index for source of commercial products.

GEN-3-00-34

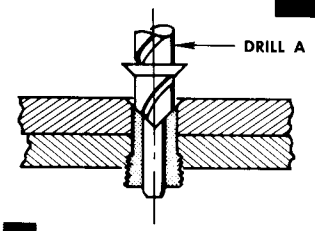
Figure 1-55. Installation of Deutsch Drive Pin Rivets

DEUTSCH RIVET REMOVAL

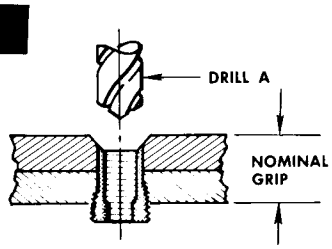
GENERAL



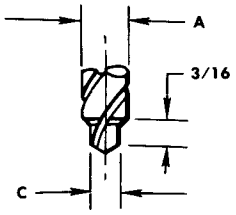
(6950)
FLUSH-HEAD RIVET



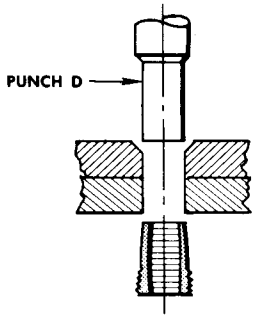
1 DRILL RIVET PIN HEAD OFF, USING DRILL SIZE A.



2 DRILL SLEEVE, USING DRILL SIZE A. DRILL TO NOMINAL GRIP DEPTH ONLY.



A PILOTED DRILL MAY BE USED.



3 DRIVE REMAINDER OF SLEEVE THROUGH.

DEUTSCH RIVET DASH NO.	DRILL SIZE A	PIN SIZE C	PUNCH SIZE D
-06 (3/16)	NO. 21 (0.1590)	0.1215	5/32
-08 (1/4)	NO. 1 (0.2280)	0.1625	7/32
-010 (5/16)	9/32 (0.2812)	0.1975	1/4
-012 (3/8)	NO. 5 (0.3480)	0.2425	11/32
-16 (1/2)	15/32 (0.4687)	0.3235	7/16

PROCEDURE FOR REMOVAL WHERE RIVET DOES NOT EXTEND THROUGH MATERIAL

1 Drill out rivet pin about half the depth, using recommended drill size.

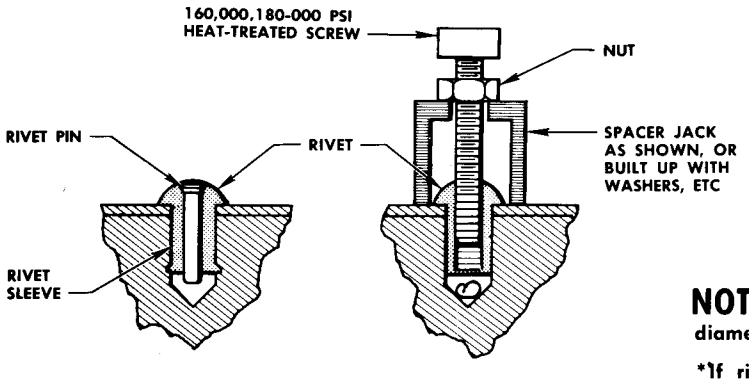
2 Tap sleeve, using recommended tap size.*

3 Drill out remainder of pin.

4 Insert Allen-head screw through spacer into tapped sleeve, and tighten nut.

5 Continue tightening nut until rivet sleeve is removed from work.

6 Hole is re-usable with same size rivet.



RIVET DIA (INCHES)	DRILL SIZE	SCREW AND TAP SIZE
3/16	29	8-32
1/4	19	10-32
5/16	3	1/4-28
3/8	I	5/16-24
7/16	Q	3/8-24
1/2	25/64	7/16-20

NOTE Use minimum of 1-1/2 times screw diameter thread engagement in rivet sleeve.

*If rivet sleeve rotates during tapping operation, use slightly larger drill for step 1, providing less cutting effort of tap.

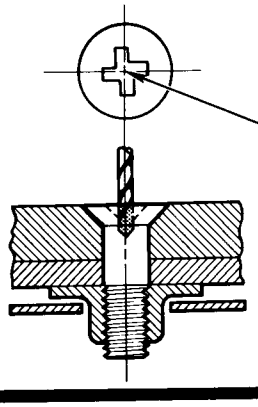
GEN-3-00-35

Figure 1-56. Removal of Deutsch Drive Pin Rivets

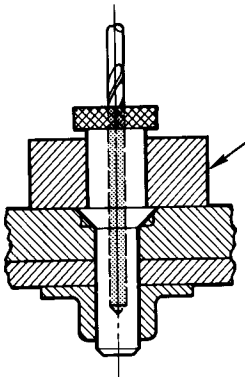
REMOVAL OF SCREW IN DAMAGED NUT PLATE

WING

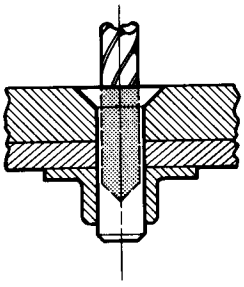
1 Start No. 30 hole at center of screw.



2 Use drill plate to ensure center drilling of screw. Drill to depth shown.

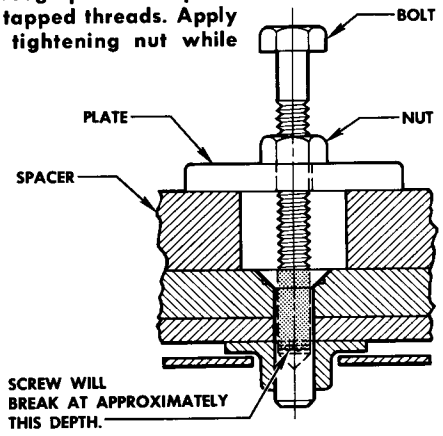


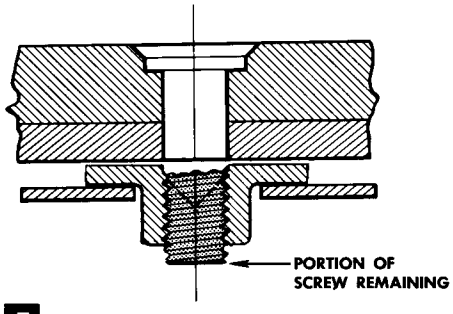
3 Drill hole to "A" diameter. Tap hole to depth as shown, using "B" size tap.



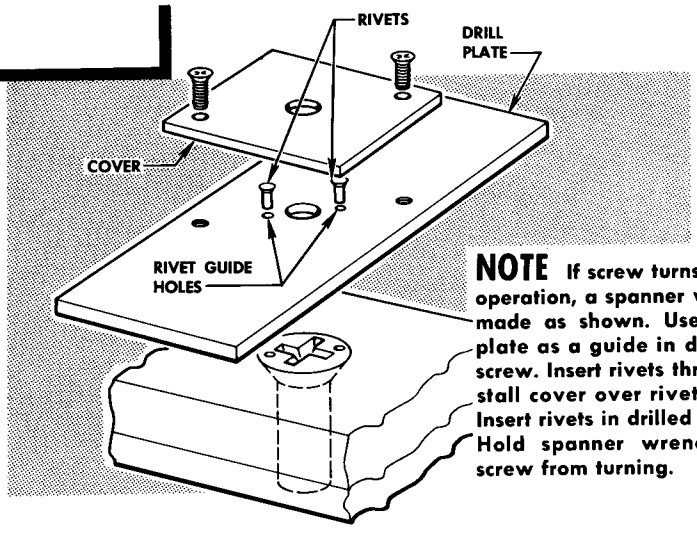
FASTENER SIZE	DRILL "A"	TAP "B"
1/4 IN.	No. 21 (0.159 IN.)	10—32
5/16 IN.	No. 3 (0.213 IN.)	1/4—28
3/8 IN.	No. 1 (0.272 IN.)	5/16—24

4 Install bolt through plate and spacer to bottom of tapped threads. Apply leverage by tightening nut while holding bolt.





5 After main portion of screw is removed, the remaining part may be drilled out or knocked out with the nut plate. Every effort should be made to remove any parts falling into fuel cell areas.



NOTE If screw turns during drilling operation, a spanner wrench may be made as shown. Use holes in drill plate as a guide in drilling holes in screw. Insert rivets through plate. Install cover over rivets with screws. Insert rivets in drilled holes in screw. Hold spanner wrench to prevent screw from turning.

F-100D-3-10-53

Figure 1-56A. Removal of Screw in Damaged Nut Plate

JO-BOLT INSTALLATION

GENERAL

PART NO.	DIAMETER	DRILL SIZE	REAM DIA	COUNTERSINK
200	No. 10	No. 15 (0.180)	0.199-0.202	100 deg x 0.375 to 0.390
260	1/4	D (0.246)	0.260-0.263	100 deg x 0.495 to 0.510

CODING FOR SELECTION OF PROPER FASTENERS

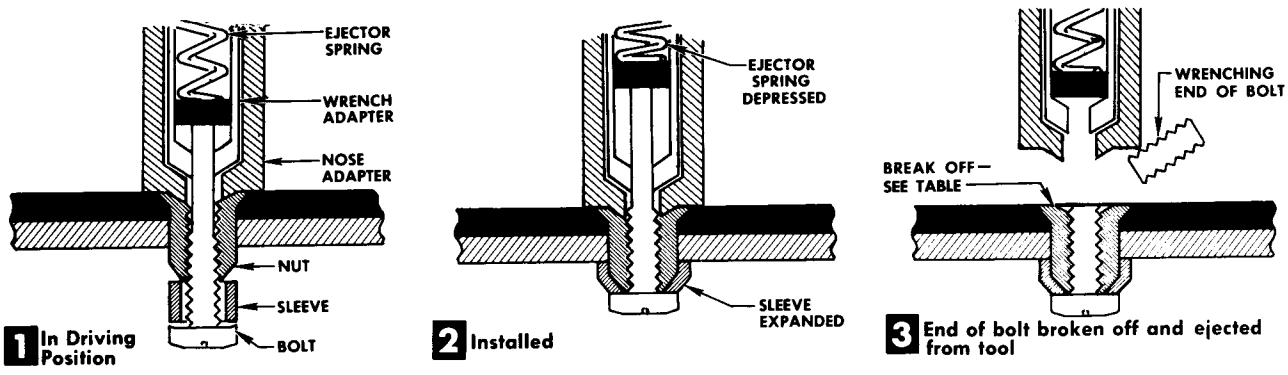
F indicates flush head.
P indicates protruding head.
260 indicates diameter in thousands.
Last dash number indicates nominal grip length in 1/16ths.
EXAMPLE: F260-4 flush head—0.260 dia—1/4 nominal grip

National Screw & Mfg Co of Calif.
Refer to index for source of commercial products.

HEAD STYLE	STEM BREAK-OFF TOLERANCES				
	5/32	3/16	1/4	5/16	3/8
FLUSH	+0.010 −0.068	+0.000 −0.078	+0.000 −0.078	+0.000 −0.083	+0.000 −0.083
PROTRUDING	+0.078 +0.000	+0.093 +0.015	+0.115 +0.037	+0.130 +0.047	+0.142 +0.059

TABLE OF GRIP LENGTHS

DASH NO.	2	3	4	5	6	7	8	9	10
GRIP LENGTH MIN	0.094	0.156	0.219	0.281	0.344	0.406	0.469	0.531	0.594
GRIP LENGTH MAX	0.156	0.219	0.281	0.344	0.406	0.469	0.531	0.594	0.656



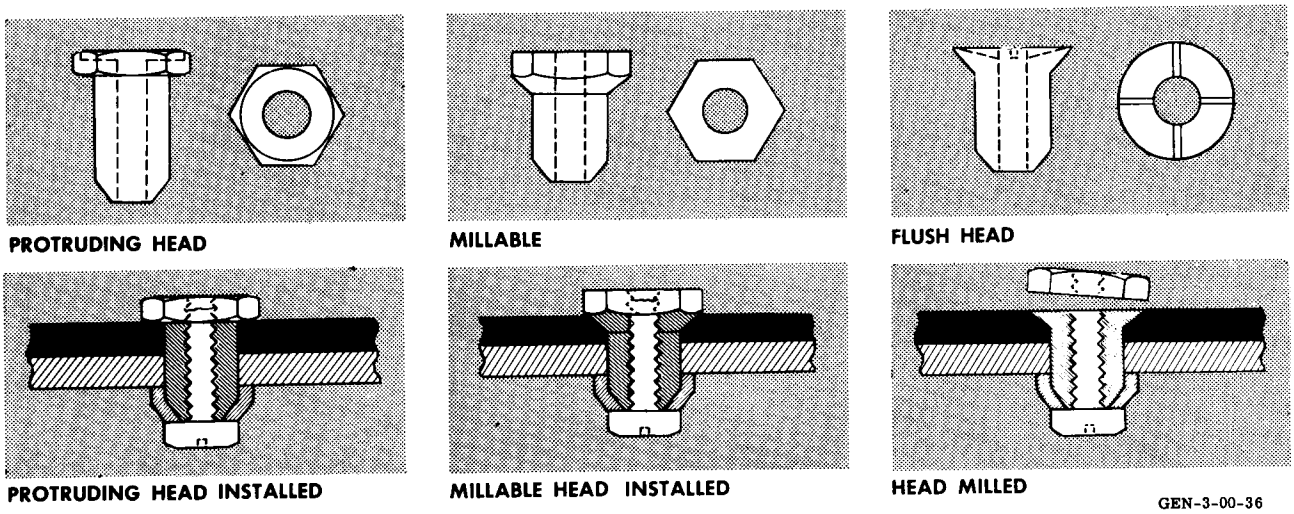
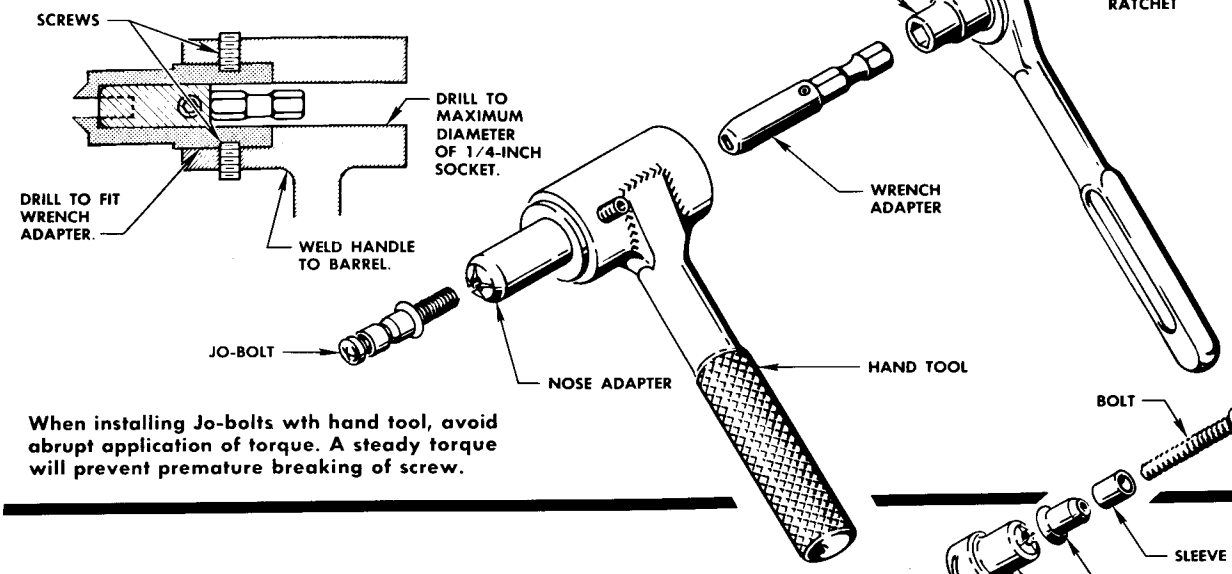


Figure 1-57. Installation of Jo-bolts (Sheet 1 of 2)

JO-BOLT INSTALLATION TOOLS

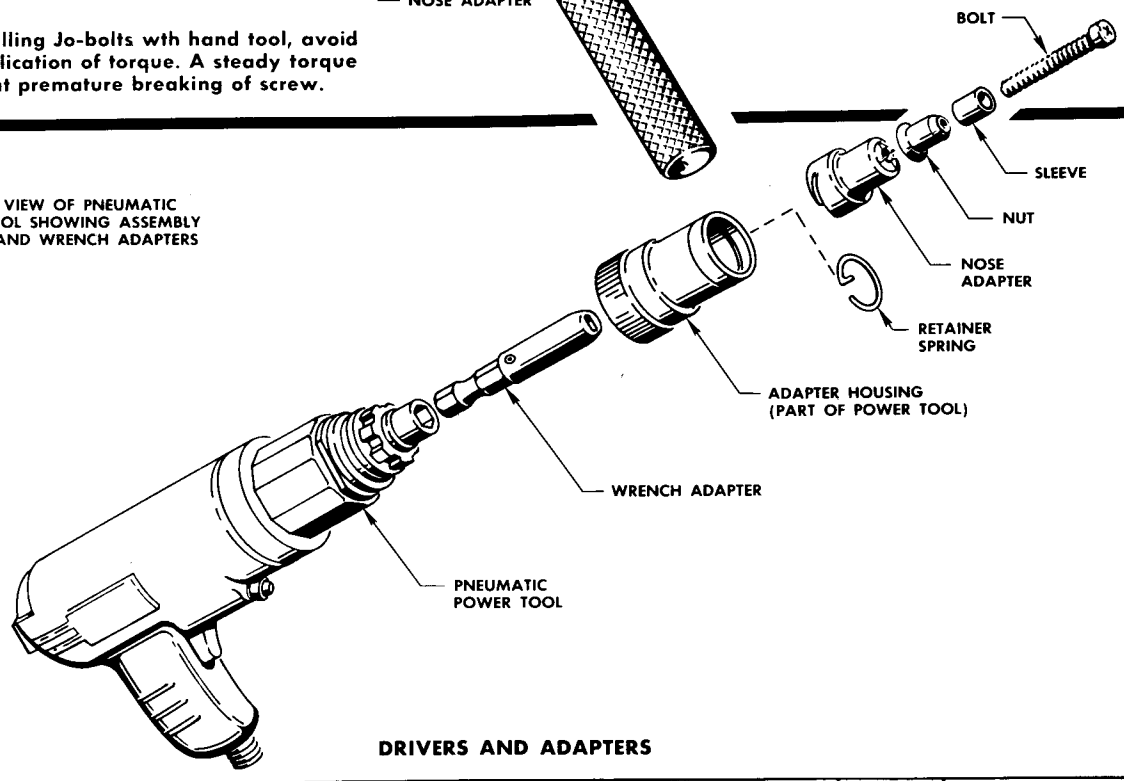
GENERAL

Hand tool may be locally fabricated. A stock nose adapter and wrench adapter will be used.



When installing Jo-bolts wth hand tool, avoid abrupt application of torque. A steady torque will prevent premature breaking of screw.

EXPLODED VIEW OF PNEUMATIC POWER TOOL SHOWING ASSEMBLY OF NOSE AND WRENCH ADAPTERS



DRIVERS AND ADAPTERS

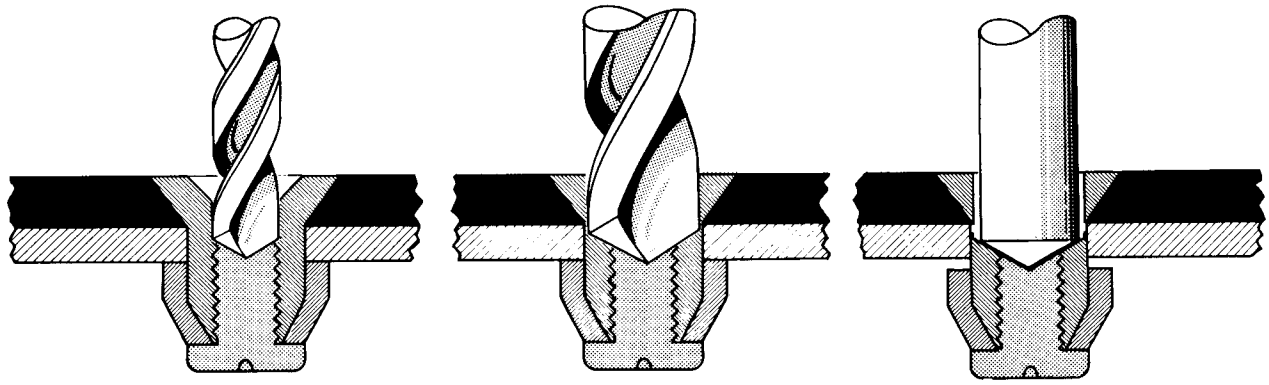
HAND TOOLS	PART NO.	FEDERAL STOCK NO.	POWER TOOLS	PART NO.	FEDERAL STOCK NO.
HAND DRIVER	HW 300	7900-273989-6	POWER DRIVER	302	8100-258233
HAND ADAPTER (F200)	HW 2000	4920-696-3340	POWER ADAPTER (F200)	PW 300	4920-696-3345
HAND ADAPTER (P200)	HW 2000P	4920-696-3341	POWER ADAPTER (P200)	PW 300P	4920-696-3346
HAND ADAPTER (F260)	HW 2600	4920-696-3342	POWER ADAPTER (F260)	PW 360	4920-696-3347
HAND ADAPTER (P260)	HW 2600P	4920-696-3343	POWER ADAPTER (P260)	PW 360P	4920-696-3348
HAND ADAPTER (F312)	HW 3120	4920-696-3344	POWER ADAPTER (F312)	PW 312	5120-568-0547

Lok-Fast Inc. Refer to index for source of commercial products. GEN-3-00-37A

Figure 1-57. Installation of Jo-bolts (Sheet 2 of 2)

JO-BOLT REMOVAL

GENERAL

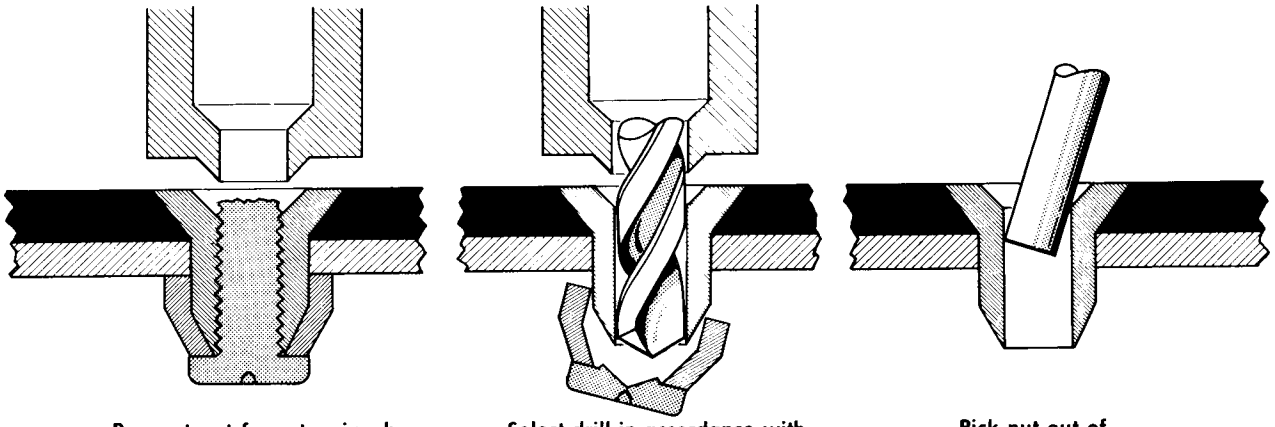


Select drill in accordance with Table I. Drill to below head-shank juncture.

Select drill in accordance with Table II. Drill to depth of pilot hole.

With hammer and nominal size punch, sever head and drive out shank and blind head.

REMOVAL WHEN BOLT TURNS IN HOLE WHILE DRILLING



Prevent nut from turning by engaging driving tool nose adapter. Hold nose adapter with hand tool handle or vise-grip pliers.

Select drill in accordance with Table I. Drill through shank, severing bolthead.

Pick nut out of hole with punch.

SELECT DRILL WITH 500 RPM MAXIMUM

	TABLE I	TABLE II
JO-BOLT SERIES	DRILL SIZE	DRILL SIZE
F- OR P-164	NO. 43 (0.089)	5/32 (0.1562)
F- OR P-200	NO. 35 (0.110)	12 (0.189)
F- OR P-260	NO. 24 (0.152)	D (0.246)
F- OR P-312	NO. 17 (0.173)	M (0.295)
F- OR P-375	NO. 5 (0.2055)	23/64 (0.3594)

GEN-3-00-38

Figure 1-58. Removal of Jo-bolts

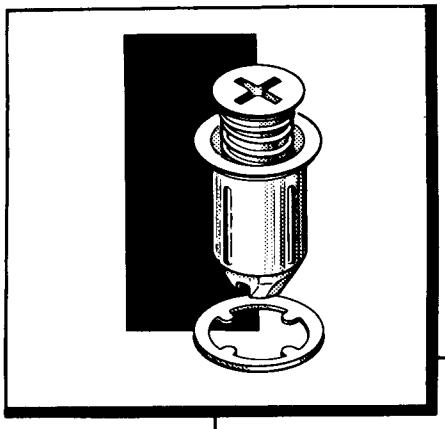
CAMLOC STRESSED PANEL FASTENER

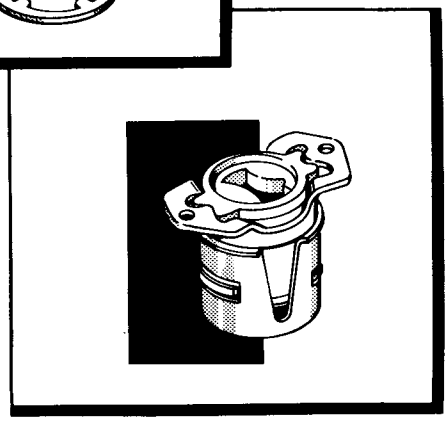
GENERAL

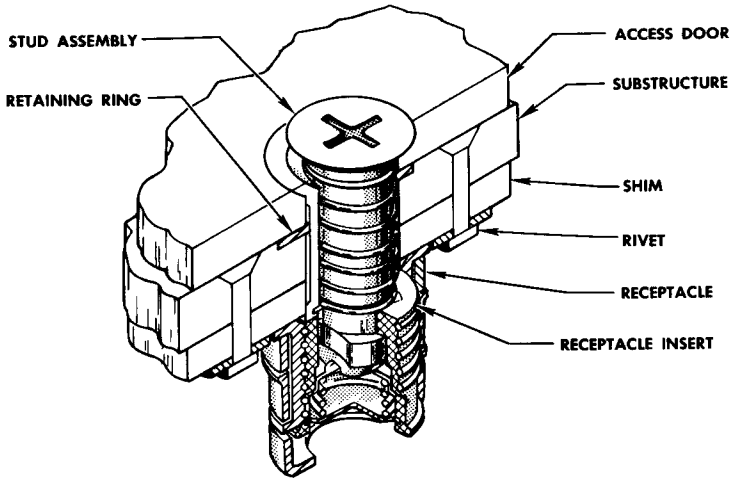
INSTALLATION AND REMOVAL

To engage stud assembly, push stud in and turn right one-quarter turn. If the stud does not rotate freely for the first quarter turn, it may not be engaged, and must be turned left to reset receptacle insert enough to permit free rotation for the first quarter turn to the right. After stud engagement, torque stud to about 40 inch-pounds.

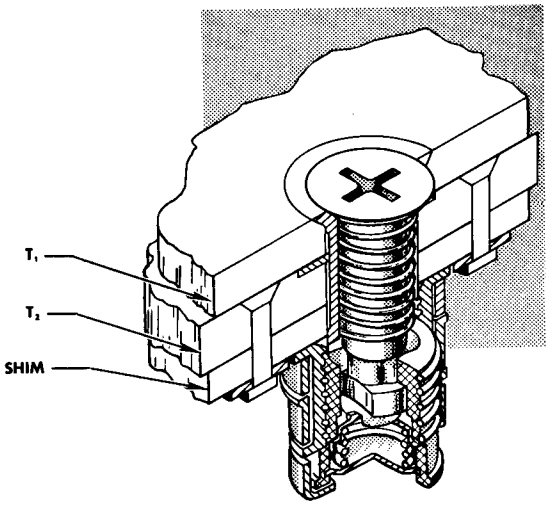
To disengage stud, rotate stud left at least one-half turn.







VIEW SHOWING CAMLOC STUD ENGAGED, LEAVING STUD HEAD SLIGHTLY ABOVE MOLD LINE



VIEW SHOWING CAMLOC IN INSTALLED POSITION

STRESSED PANEL FASTENER (SPF)

1. Hole size must be as follows:

   a. For countersunk applications, use drill W (0.3860 inch).

   b. For dimpled applications, predrill 19/32 inch, dimple, and redrill, using drill W (0.3860 inch).

2. Use retaining ring 459-2 for countersunk applications; use retaining ring 4S10-1 for dimpled applications.

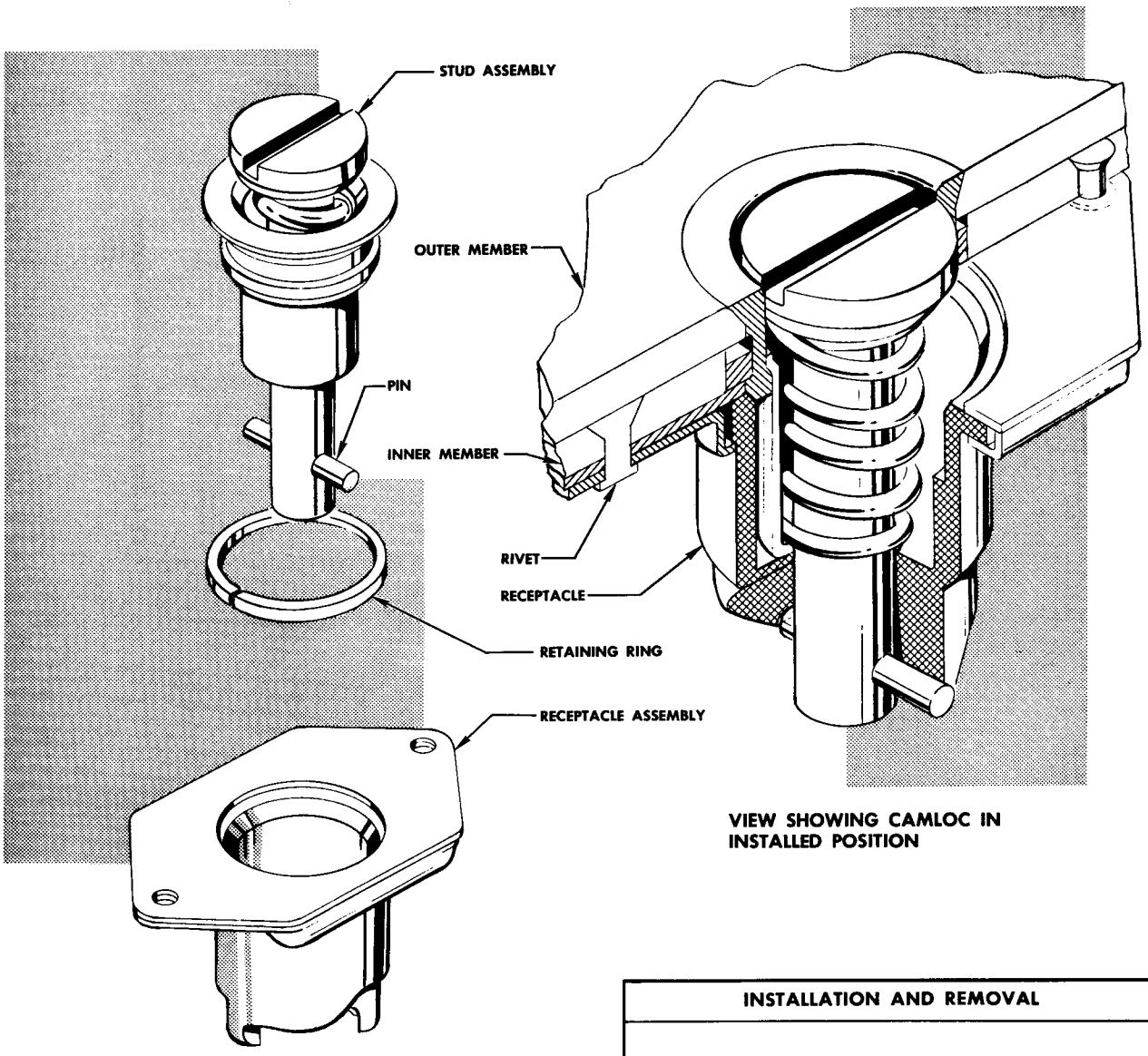
3. Shim is required only when $T_1 + T_2$ is less than 0.158 inch.

Gen. -3-00-55A

Figure 1-58A. Quick-acting Structural Fastener—Camloc (Sheet 1 of 2)

CAMLOC SHEAR FASTENER

GENERAL



VIEW SHOWING CAMLOC IN INSTALLED POSITION

INSTALLATION AND REMOVAL

To engage stud assembly, turn stud to align pin with slot in receptacle. Push in; then turn stud to the right about one-quarter turn.

To disengage stud assembly, push in; then turn stud about one-quarter turn to the left.

NOTE The hole size for shear panel fasteners using the 40G1 shear grommet must be 0.500 inch for the outer member and 0.594 inch for the inner member.

- The hole size for shear panel fasteners using the 40G9-1 grommet must be 9/16 inch for the outer and inner members.

Gen-3-00-56

Figure 1-58A. Quick-acting Structural Fastener—Camloc (Sheet 2 of 2)

CAMLOC FASTENER TABLES

GENERAL

TABLE NO. 1 SHEAR PANEL CAMLOC STUDS

PART NO.	GRIP (t_1+t_2)	LENGTH "L"	PART NO.	GRIP (t_1+t_2)	LENGTH
4002-1	0.051—0.080	0.650	4002-14	0.441—0.470	1.040
4002-2	0.080—0.110	0.680	4002-15	0.471—0.500	1.070
4002-3	0.111—0.140	0.710	4002-16	0.501—0.530	1.100
4002-4	0.141—0.170	0.740	4002-17	0.531—0.560	1.130
4002-5	0.171—0.200	0.770	4002-18	0.561—0.590	1.160
4002-6	0.201—0.230	0.800	4002-19	0.591—0.620	1.190
4002-7	0.231—0.260	0.830	4002-20	0.621—0.650	1.220
4002-8	0.261—0.290	0.860	4002-21	0.651—0.680	1.250
4002-9	0.291—0.320	0.890	4002-22	0.681—0.710	1.280
4002-10	0.321—0.350	0.920	4002-23	0.711—0.740	1.310
4002-11	0.351—0.380	0.950	4002-24	0.741—0.770	1.340
4002-12	0.381—0.410	0.980	4002-25	0.771—0.800	1.370
4002-13	0.411—0.440	1.010			

NOTE If floating-type receptacle (244-16) is used, use the next longer stud. (See table No. 1.)

- If ejection-type grommet (40G9-1) is used, use the next longer stud. (See table No.1.)
- Stud assemblies 4002-16 through 4002-25 are retained in the grommet by means of a 4002-SW washer. Stud assemblies 4002-2 through 4002-15 are self-retaining installation being made with Camloc 4-P3 pliers.

TABLE NO. 2

STRESSED PANEL CAMLOC STUDS	
PART NO.	GRIP (t_1+t_2)
(17)	0.058—0.157
4S7-4	0.158—0.257
4S7-6	0.258—0.357
4S7-8	0.358—0.457
4S7-10	0.458—0.557
4S7-12	0.558—0.657

NOTE Use either 4R19-14-102 or 4R19-16-102 shim under receptacle and use 4S7-4 stud when t_1+t_2 is less than 0.158 inch.

TABLE NO. 3 CAMLOC SHEAR GROMMET PART NUMBERS

PART NO.	OUTER SHEET THICKNESS	INNER SH MIN THICKNESS
40G1-5	0.051	0.072
40G1-5	0.064	0.064
40G1-6	0.072	0.064
40G1-6	0.081	0.051
40G1-8	0.091	0.064
40G1-8	0.102	0.051
40G1-10	0.112	0.064

NOTE The Camloc shear grommets used may be determined from table No. 6.

GEN-3-00-58

Figure 1-58B. Quick-acting Structural Fastener—Camloc Tables (Sheet 1 of 2)

CAMLOC FASTENER TABLES

GENERAL

TABLE NO. 4

FASTENER TYPE AND SIZE	HOLE LOCATION	MINIMUM THICKNESS FOR COUNTERSINKING	MAX THICKNESS FOR DIMPLING		
			ALLOY	CORR RES STEEL	
				ANNEALED	1/2 H
CAMLOC STUD 4002	OUTER MEMBER	0.051	AS NOTED *	AS NOTED *	AS NOTED *
CAMLOC STUD 4S7	OUTER MEMBER	0.090	0.090	0.063	0.053

NOTE The material thickness shown in the camloc table governs the countersinking and dimpling limitations.

*Dimpling is not permitted.

TABLE NO. 5

DIMENSION	STUD		RECEPTACLE		
———————	4002	4S7	214-16 OR 244-16	4R1-14	4R1-16
ANGLE *	140 DEG	100 DEG	———————	———————	———————
DIAMETER *	0.625	0.525	———————	———————	———————
SPACING	———————	———————	1.000	0.875	1.000

*These dimensions are without tolerance. The dimensional limits used must satisfy the flushness.

TABLE NO. 6

FASTENER	SPECIAL TOLERANCE	STANDARD TOLERANCE
	FOR CRITICAL ZONES	FOR OTHER ZONES
CAMLOC	0.005 ABOVE	0.010 ABOVE
CAMLOC	0.005 BELOW	0.010 BELOW

NOTE Flush-head studs must be flush with the external skin surface within the limits given in table No. 6.

Gen-3-00-57

Figure 1-58B. Quick-acting Structural Fastener—Camloc Tables (Sheet 2 of 2)

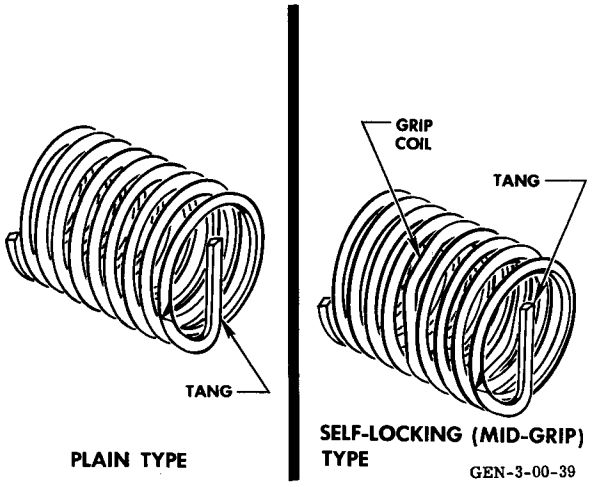


Figure 1-59. Heli-Coil Inserts

The mid-grip insert is dyed red for identification and has a specially formed grip coil, or coils, in the center which have a gripping effect on the engaging screw. (See figure 1-59.) For detailed information on installation and removal, refer to "General—Installation of Heli-Coil Inserts," T.O. 44H1-1-117. Although the mid-grip type insert is not covered in this Technical Order, the information is applicable for both types, except for some of the tools used. The mid-grip inserts require a different inserting and tang break-off tool than those used for plain inserts. The differences in basic tool numbers are noted in the table below.

NAME OF TOOL	BASIC TOOL NUMBERS	
	Plain Insert	Midgrip Insert
Prewinder inserting tool	528	3551
Prewinder inserting tool	535	3552
Tang break-off tool	1195	3580
Tang break-off tool	1196	3581

The size dash numbers are identical to those outlined in the Technical Order.

THREADED INSERTS—ROSAN.

Threaded inserts with the Rosan locking feature are available in several types. The type in most common use consists of an internally and externally threaded and an externally serrated insert with a matching internally and externally serrated ring. Variations include: plain holes, taper pipe threads, gasket seals, solid plugs, dowels, and studs. Design information for the internally threaded type is shown in figure 1-60. The insert is available either plain or with a slotted lower end for self-locking.

If a self-locking feature is used with an AN standard bolt, the insert shown in figure 1-60 shall be specified, rather than the longer types which are illustrated in the Rosan catalog. The insert is counterbored, as shown by the "M" dimension for No. 10 sizes and over, to prevent interference with the imperfect threads on the bolt, and to allow the threaded area of the bolt to penetrate through the locking device by at least a portion of the first thread past the chamfer. The letter "L" is omitted from the part number when the self-locking feature is not required. The insert and ring are made of carbon steel; the ring is casehardened. However, if the tensile strength of the material into which the ring is to be inserted is 125,000 psi or above, the counterbore diameter, "C," shown for aluminum alloy in figure 1-60, must be increased by 0.015 inch. The material code "SB" in columns "H," and "J," indicates either C1117 or B1113 low-carbon steel for the insert and B1113 low-carbon steel for the ring. The code "SB" supersedes "SH." However, identical material will be furnished for either call-out.

THREADED INSERTS—KEENSERT.

The Keensert is an internally-externally threaded bushing, manufactured from Type 303 stainless steel. The insert is positively locked against rotation by integral keys driven through the threads of the surrounding parent material. Keenserts are available in two types: the lightweight series and the heavy-duty series. The lightweight series, when installed in sand castings of 356-T6 aluminum alloy, will not pull out at tension loads exceeding the ultimate tensile strength for fasteners manufactured in accordance with Specification MIL-B-5812 (125,000 to 140,000 psi heat-treat ranges). The heavy-duty series, when installed in 356-T6 aluminum alloy sand castings, will not pull out at tension loads exceeding the ultimate tensile strength for threaded fasteners manufactured in accordance with Specification MIL-B-7838 (160,000 to 180,000 psi heat-treat ranges). Either insert, when installed in 2024-T42 aluminum alloy, will usually break the appropriate bolt before damaging either the insert or the parent material.

Keenserts are also available either plain (non-locking) or with a relieved and indented locking feature at the center of the insert. The indented locking feature produces a locking effect on an engaging threaded fastener and meets the torque requirements established by Specification MIL-N-25027. The locking-type inserts are treated with a dry nongalling lubricant identified by its dark gray color. The letter "L" is omitted from the part number when the locking feature is not required. Both types of inserts are counterbored to prevent interference with the imperfect threads of the bolt. The length of engagement of the bolt into the insert is not considered critical (except for penetration of the locking feature). The additional wall thickness provided by the increased minor diameter of the external threads transmits the tensile loads the full length of the insert regardless of the location of the end of the bolt within the insert.

QUICK-ACTING STRUCTURAL FASTENER—CAMLOC.

Quick-acting fasteners, (figure 1-58A) are of structural type and relatively high strength. They are used in some areas of the airplane for structural access panels and doors which are frequently removed. The Camloc fastener has three basic parts: the stud, the retaining ring, and the receptacle. The stud is held in the access panel or door with the retaining ring, and the receptacle is riveted to the substructure. There are four types of receptacles used with the Camloc fastener. Two types are used with the 4002 series shear-type fastener and two are used with the 4S7 series stressed-panel type fastener. Part numbers for the 4002 series are 214-16, which is a rigid-type receptacle, and 244-16, which is a floating-type receptacle. Part numbers 4R1-14 and 4R1-16 are used with the stressed-panel fastener, and are both floating-type receptacles.

For quick-acting Camloc fastener part numbers, tolerances, flushness, and dimpling information, see figure 1-58B.

SHOP PRACTICES.

LOCATING BLIND HOLES WITH HOLE FINDER.

The fastest and most accurate method of locating blind fastener holes is with the hole finder. They may be made of various materials and in special shapes and sizes when the conditions of the job do not permit the use of a hole finder such as is shown in figure 1-61. Although a temporary or one-time-use tool may be fabricated in a few minutes from any scrap of hardened aluminum alloy and a rivet of the desired size, it is urged that each sheet metal worker provide himself with a complete set of well made tools similar to the one shown in figure 1-61. A complete set will consist of a hole finder for each size sheet metal fastener used, usually 3/32 inch through 5/16 inch.

When blind holes are being located with a hole finder it is important that the material being marked or pilot-drilled does not move. It is usually advisable to fasten it to the under-structure with two temporary fasteners and move them as necessary as the marking or drilling progresses.

LOCATING BLIND HOLES WITH A SCALE.

See figure 1-61.

LOCATING SKIN TRIM LINES.

LOCATING SKIN TRIM LINES WITH SKIN SCRIBE.

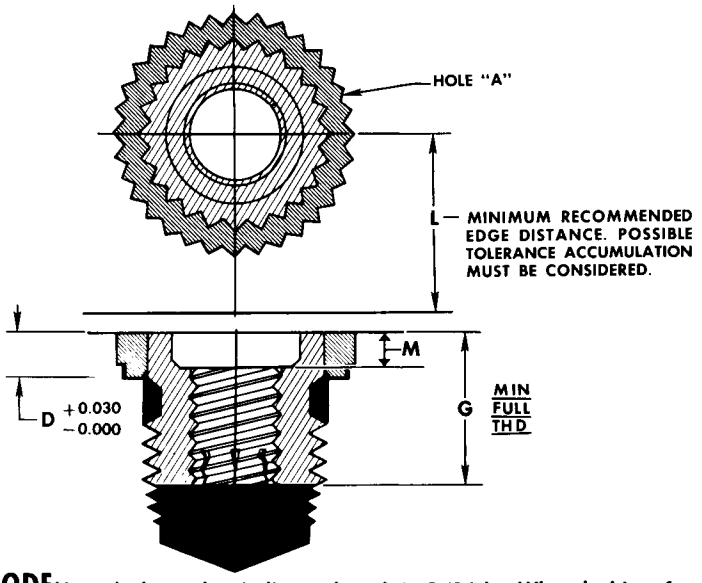
The skin scribe shown in figure 1-62 is made of any steel that is handy but a material that will take and hold a good scribing edge will last longer and give the most satisfactory results.

This tool is equally useful for marking the "cut lines" of irregular as well as straight-lined, flush patches, cover plates, doors, etc. The turned-down tip on the lower strap is bent as sharply as possible then the inside edge is rounded to about a 1/4 inch radius. It should be polished so that it will slide smoothly along the inside edge of the cutout being matched. The scribe point on the upper strap is set 1/32 inch outside the inner edge of the guide to allow for the desired skin gap.

The edges of cutout to be matched should be filed or sanded smooth. The oversized filler sheet should be

ROSAN INSERTS

GENERAL



1 Drill hole shown in column "A" to depth shown in column "B."

2 Counterbore diameter shown in column "C" (+0.003, −0.000 inch), to depth shown in column "D." Fillet radius 0.010 to 0.020 inch. Counterbore to be concentric with thread pitch diameter within 0.006-inch total indicator reading.

3 Countersink 90 degrees to full diameter of counterbore.

4 Tap for thread size shown in column "K" to depth shown in column "G."

5 Select applicable insert from column "H." Install from 0.010 to 0.020 inch below the surface.

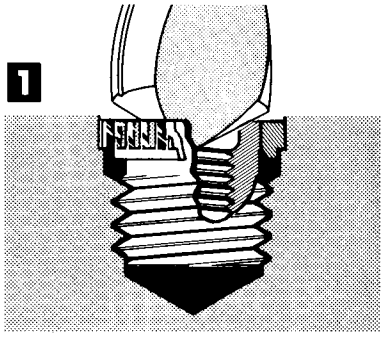
6 Select applicable lock ring from column "J." Install from flush to 0.010 inch below the surface.

CODE: Last dash number indicates length in 1/16ths. When locking feature is not required, omit letter "L" after part numbers. Example: RD206SB-8

INSERT INTERNAL THREAD	TAP DRILL A	DRILL DEPTH B	CBORE DIA C		CBORE DEPTH D	MIN FULL THD G	INSERT PART NUMBER H	LOCK RING PART NUMBER J	TAP THREAD K	MIN EDGE DISTANCE L	CBORE FOR AN BOLT M	REMOVAL DRILL X
			AL ALLOY	MAG								
4-40 NC-3B	25(0.149)	15/32	0.250	0.258	0.094	1/4	R102SB-4L	RL16SB-5	10-24 NC-3B	11/64	.....	11/64
6-32 NC-3B	16(0.177)	17/32	0.281	0.290	0.109	5/16	R103SB-5L	RL18SB-6	12-24 NC-3	13/64	.....	13/64
8-32 NC-3B	7(0.201)	9/16	0.328	0.337	0.109	3/8	R104SB-6L	RL21SB-6	1/4-20 UNC-3B	7/32	.....	15/64
10-32 NF-3B	F(0.257)	19/32	0.375	0.381	0.140	11/32	RD205SB-5.5L	RL24SB-7	5/16-18 UNC-3B	1/4	0.062	9/32
1/4-28-UNF-3B	5/16	21/32	0.438	0.448	0.156	13/32	RD206SB-6.5L	RL28SB-8	3/8-16 UNC-3B	5/16	0.080	11/32
5/16-24 UNF-3B	27/64	23/32	0.516	0.526	0.172	15/32	RAD207SB-7.5L	RL33SB-9	1/2-13 UNC-3B	3/8	0.095	27/64
3/8-24 UNF-3B	31/64	13/16	0.594	0.601	0.172	9/16	RD2098SB-9L	RL38SB-9	9/16-12 UNC-3B	7/16	0.100	31/64
7/16-20 UNF-3B	17/32	13/16	0.734	0.747	0.188	9/16	RD2089SB-9L	RL47SB-10	5/8-11 UNC-3B	1/2	0.100	19/32
1/2-20 UNF-3B	41/64	15/16	0.875	0.888	0.203	11/16	RD210SB-11L	RL56SB-11	3/4-10 UNC-3B	5/8	0.109	23/32

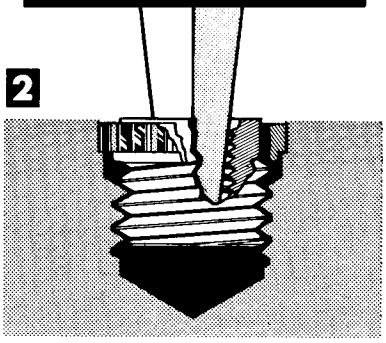
TO REMOVE INSERT

1



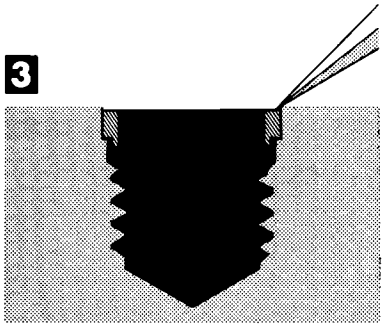
Drill out serrated collar of insert. Use drill same diameter as serrations. Drill to depth of lock ring only. Use drill at high speed. For correct drill sizes, see column "X" in table above.

2



Drive in square-type "ease-out" tool or similar square tool. Screw out insert.

3



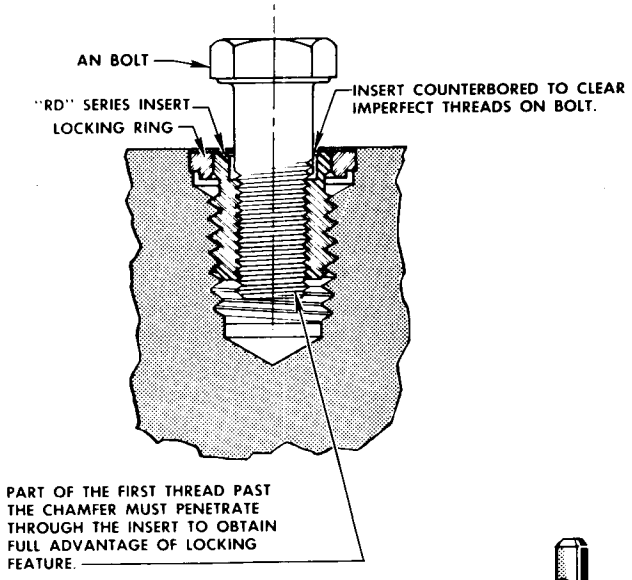
If lock ring fails to come out as insert is unscrewed, collapse remaining portion of ring with punch.

GEN-3-00-40

Figure 1-60. Rosan Inserts (Sheet 1 of 2)

ROSAN INSERTS

GENERAL



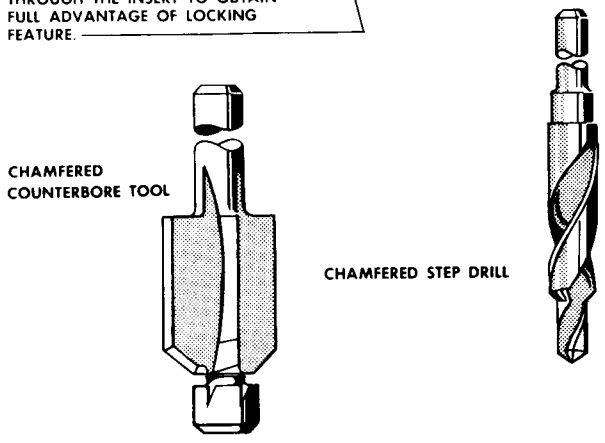
NOTE Sequences of operation for the installation of Rosan inserts.

1. Tap drill
2. Counterbore
3. Countersink.
4. Tap

Operations 1, 2, and 3 can be combined into one operation by using a chamfered step drill.

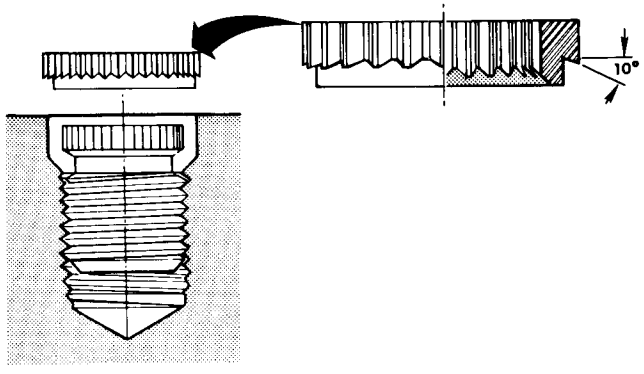
Operations 2 and 3 can be combined into one operation by using a chamfered counterbore tool with pilot.

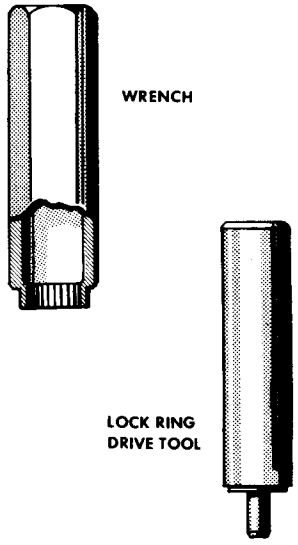
All operations may be accomplished with separate tools.



NOTE

Lock ring is designed to broach its own way into parent material. Ring should be driven straight with flat tool.





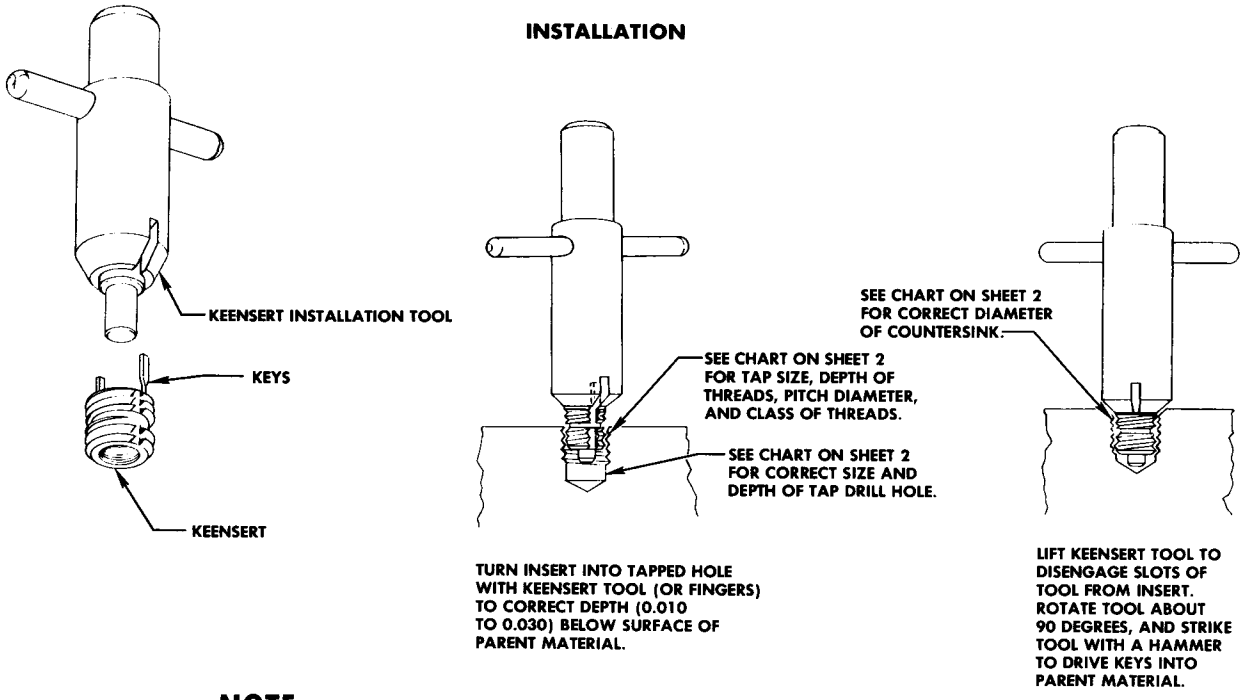
GEN-3-00-41

Figure 1-60. Rosan Inserts (Sheet 2 of 2)

KEENSERT THREADED INSERTS

GENERAL

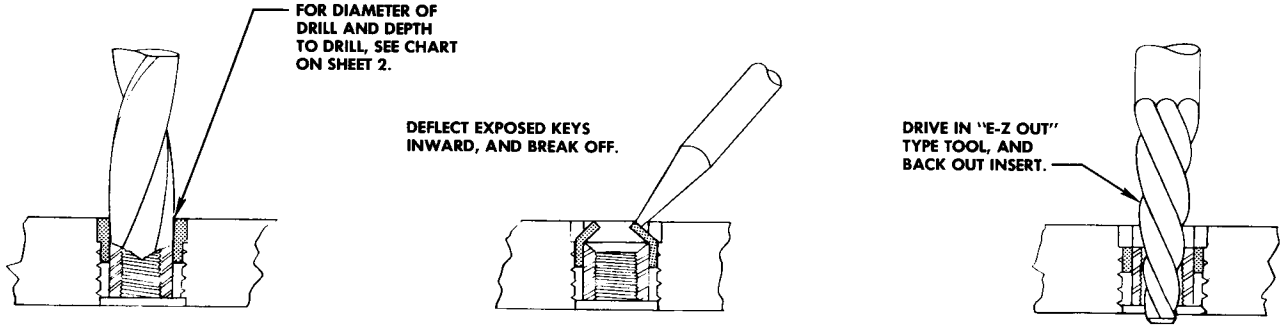
INSTALLATION



NOTE The tap drill used is larger than the standard tap drill, to clear the increased minor diameter of the external threads.

- The tolerance on the countersink is held to ±0.010 inch to automatically position insert 0.010 to 0.030 inch below the start of the first thread.
- If the proper Keensert tool is not available, the keys may be driven by a hammer.

REMOVAL



NOTE A new insert of the same size may be reinstalled in the original tapped hole if it is undamaged. The insert should be rotated sufficiently to eliminate the possibility of driving the keys into the original slots of the parent material.

GEN-3-00-66

Figure 1-60A. Keensert Threaded Inserts (Sheet 1 of 2)

KEENSERT THREADED INSERTS

GENERAL

INSERT INTERNAL THREAD	TAP DRILL SIZE		TAP DRILL DEPTH	COUNTERSINK DIA		EXTERNAL THREAD	
	LIGHT WEIGHT	HEAVY DUTY		LIGHT WEIGHT	HEAVY DUTY	LIGHT WEIGHT	HEAVY DUTY
# 4-40		#15 (0.1800)	13/32		0.220		# 12-24NC-3B
# 6-32		# 3 (0.2130)	13/32		0.254		1/4-20UNC-3B
# 8-32		I(0.2720)	15/32		0.323		5/16-18UNC-3B
#10-32	9/32	Q(0.3320)	15/32	0.320	0.385	5/16-18UNC-3B	3/8-16UNC-3B
1/4-28	11/32	X(0.3970)	19/32	0.380	0.447	3/8-16UNC-3B	7/16-14UNC-3B
5/16-24	Y(0.4040)	29/64	23/32	0.440	0.510	7/16-14UNC-3B	1/2-13UNC-3B
3/8-24	15/32	33/64	25/32	0.510	0.572	1/2-13UNC-3B	9/16-12UNC-3B
7/16-20		37/64	29/32		0.635		5/8-11UNC-3B

INSERT INTERNAL THREAD	FULL THREAD	EXTERNAL THREAD		KEENSERT NUMBER	
		LIGHT WEIGHT	HEAVY DUTY	LIGHT WEIGHT	HEAVY DUTY
# 4-40	9/32		0.1889 +0.0024 −0.0000		KNHL0440
# 6-32	9/32		0.2175 +0.0036 −0.0000		KNHL0632
# 8-32	11/32		0.2764 +0.0039 −0.0000		KNHL0832
# 10-32	11/32	0.2764 +0.0039 −0.0000	0.3344 +0.0043 −0.0000	KNL1032	KNHL1032
1/4-28	13/32	0.3344 +0.0043 −0.0000	0.3911 +0.0046 −0.0000	KNL428	KNHL428
5/16-24	1/2	0.3911 +0.0046 −0.0000	0.4500 +0.0048 −0.0000	KNL524	KNHL524
3/8-24	9/16	0.4500 +0.0048 −0.0000	0.5084 +0.0051 −0.0000	KNL624	KNHL624
7/16-20	11/16		0.5660 +0.0054 −0.0000		KNHL720

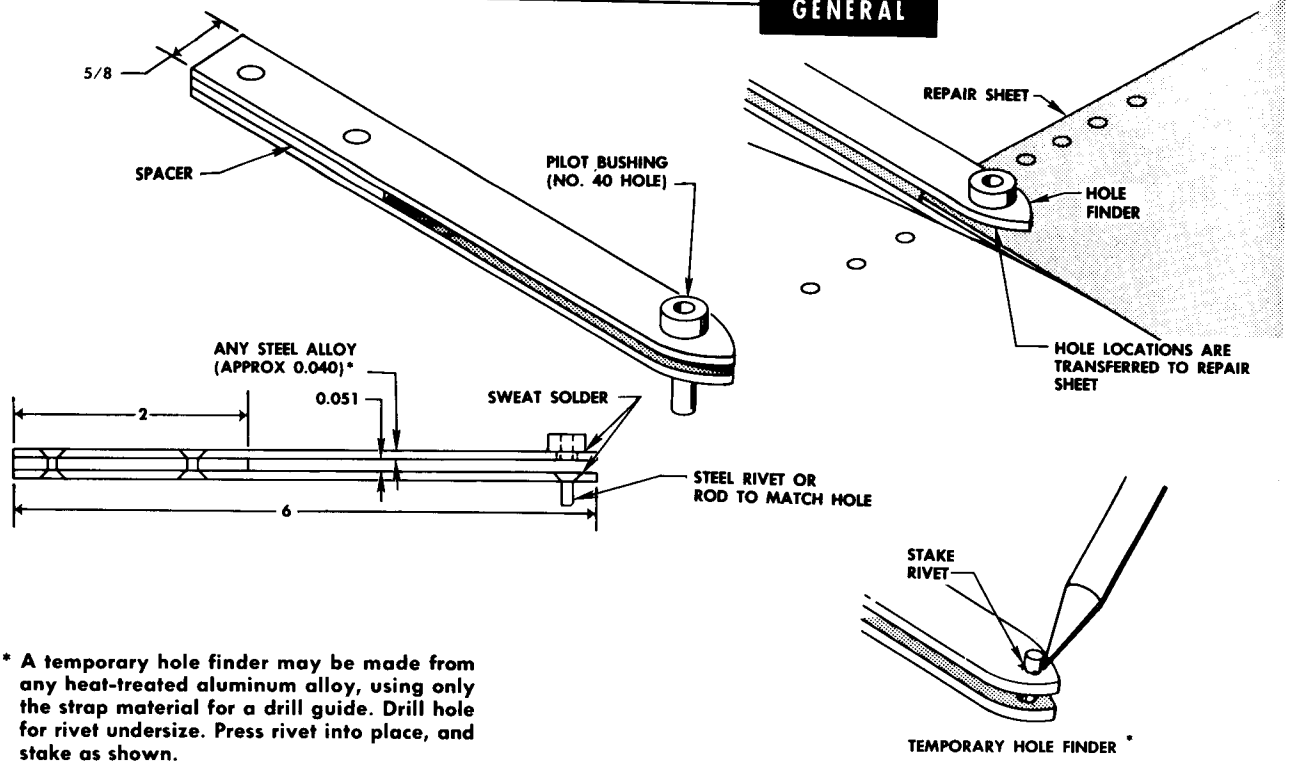
INSERT INTERNAL THREAD	KEENSERT TOOL NO.		INTERNAL THREAD SIZE	INTERNAL THREAD PITCH DIAMETER	SIZE OF DRILL FOR REMOVAL		DEPTH OF DRILL FOR REMOVAL	
	LIGHT WEIGHT	HEAVY DUTY			LIGHT WEIGHT	HEAVY DUTY	LIGHT WEIGHT	HEAVY DUTY
# 4-40		TH0440L	# 4-40NC-3B	0.0958 +0.0024 −0.0000		5/32		1/8
# 6-32		TH0632L	# 6-32NC-3B	0.1177 +0.0027 −0.0000		3/16		1/8
# 8-32		TH0832L	# 8-32NC-3B	0.1437 +0.0028 −0.0000		7/32		1/4
# 10-32	T1032L	TH1032L	#10-32NF-3B	0.1697 +0.0029 −0.0000	7/32	9/32	1/4	1/4
1/4-28	T428L	TH428L	1/4-28UNF-3B	0.2268 +0.0032 −0.0000	9/32	11/32	1/4	1/4
5/16-24	T524L	TH524L	5/16-24UNF-3B	0.2854 +0.0036 −0.0000	11/32	13/32	1/4	1/4
3/8-24	T624L	TH624L	3/8-24UNF-3B	0.3479 +0.0037 −0.0000	11/32	15/32	1/4	1/4
7/16-20		TH720L	7/16-20UNF-3B	0.4050 +0.0041 −0.0000		17/32		5/16

GEN-3-00-65

Figure 1-60A. Keensert Threaded Inserts (Sheet 2 of 2)

HOLE FINDER

GENERAL



* A temporary hole finder may be made from any heat-treated aluminum alloy, using only the strap material for a drill guide. Drill hole for rivet undersize. Press rivet into place, and stake as shown.

TWO METHODS WITH A SCALE

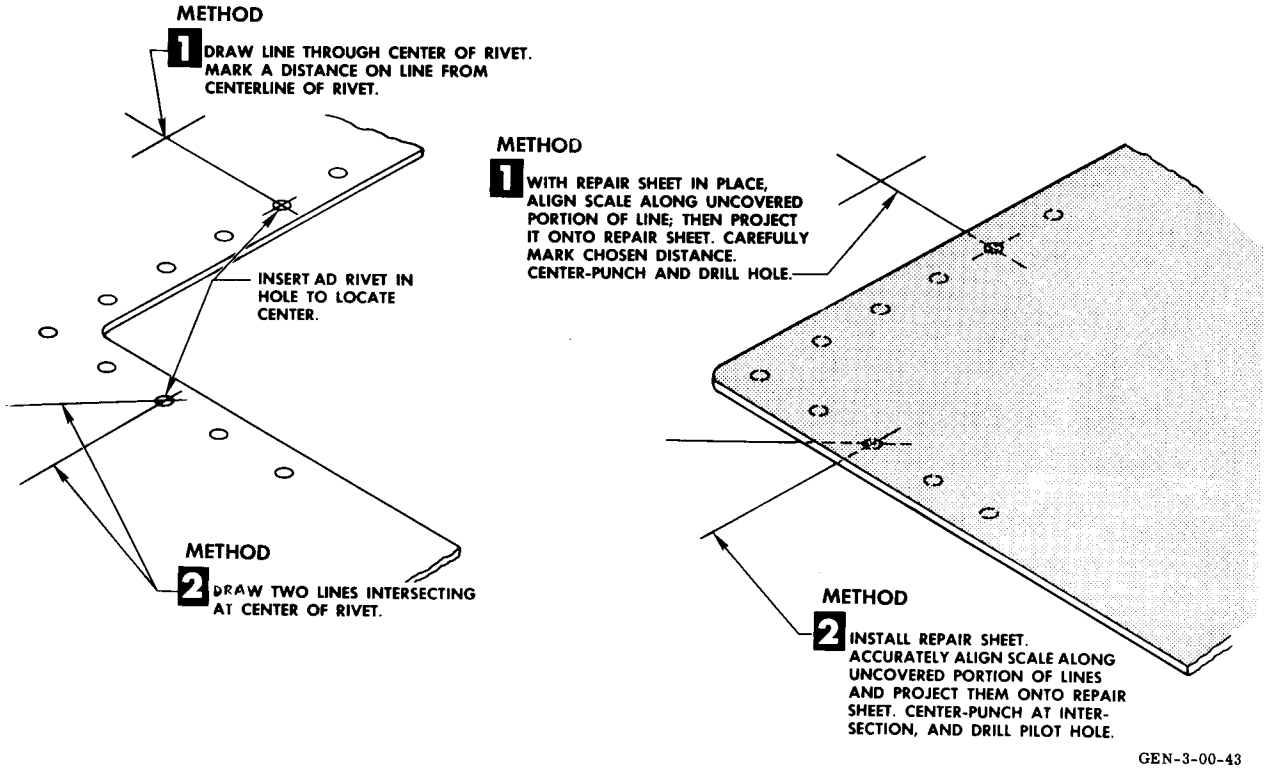
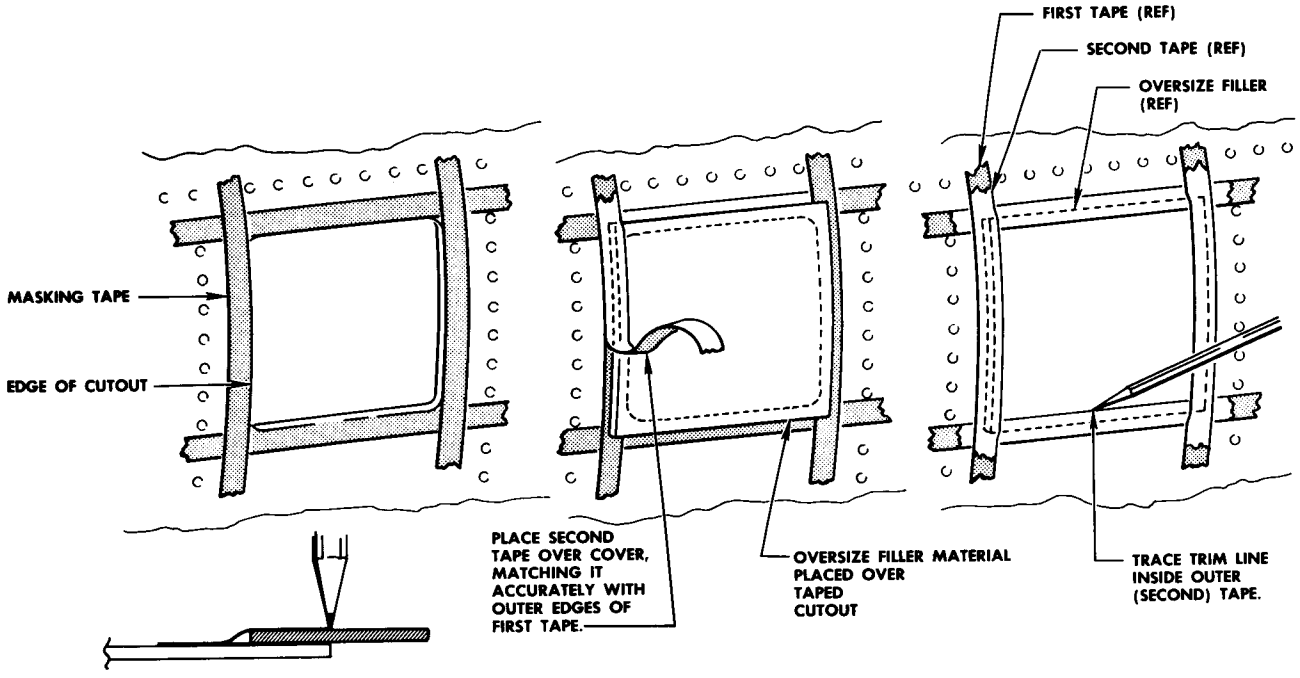


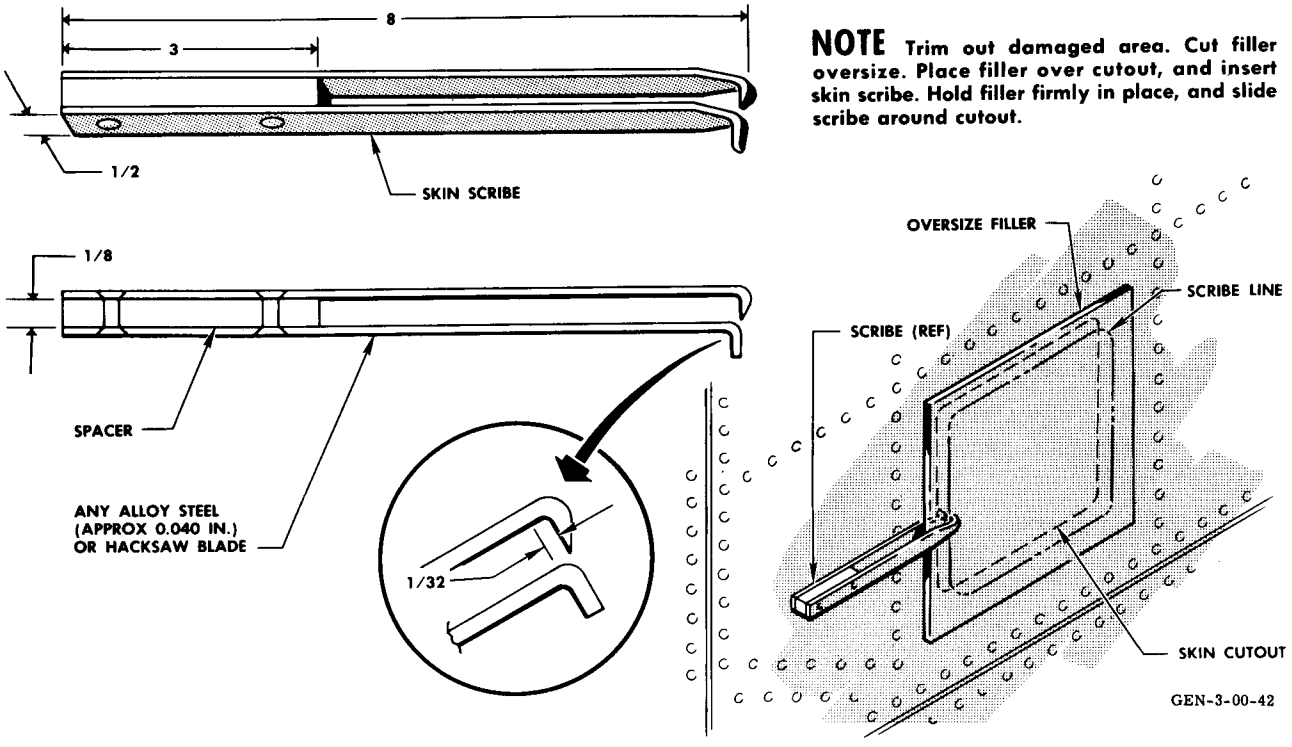
Figure 1-61. Locating Blind Holes

LOCATING TRIM LINES WITH MASKING TAPE

GENERAL



LOCATING TRIM LINES WITH SKIN SCRIBE

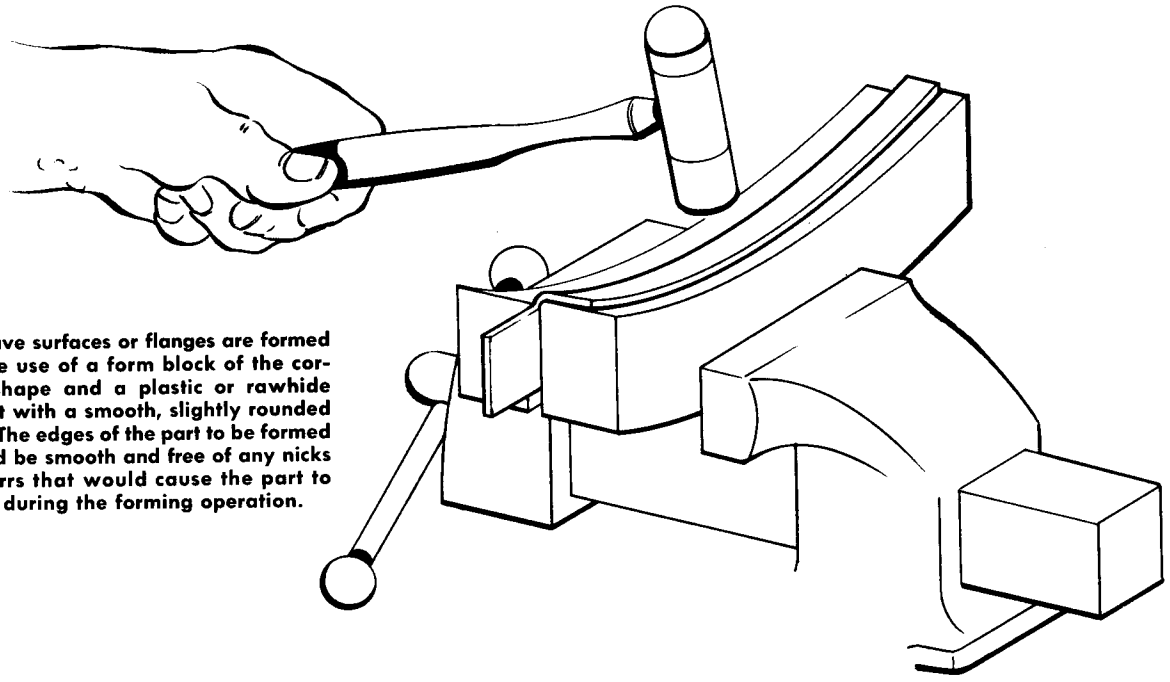


NOTE Trim out damaged area. Cut filler oversize. Place filler over cutout, and insert skin scribe. Hold filler firmly in place, and slide scribe around cutout.

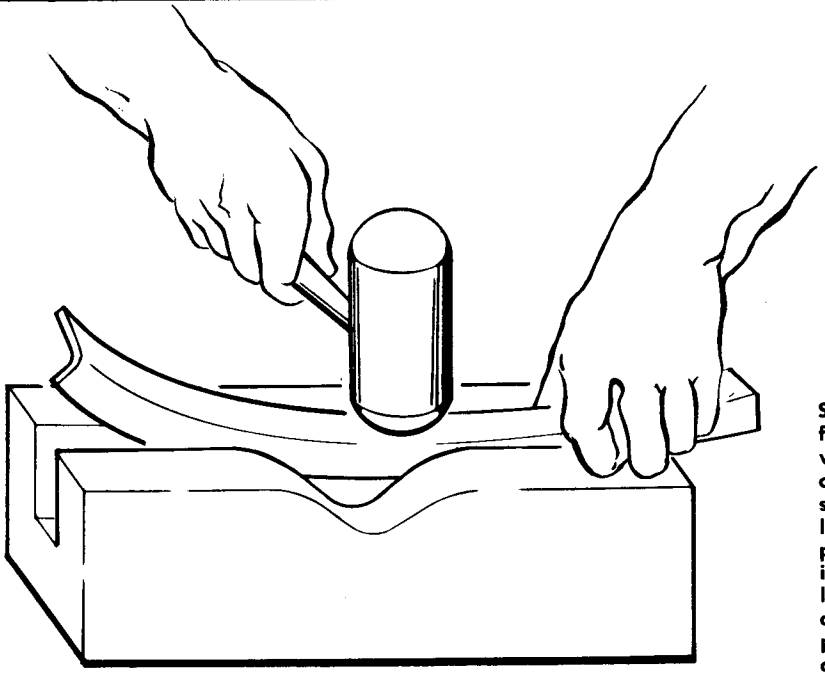
Figure 1-62. Locating Trim Lines

HAND-FORMING

GENERAL



Concave surfaces or flanges are formed by the use of a form block of the correct shape and a plastic or rawhide mallet with a smooth, slightly rounded face. The edges of the part to be formed should be smooth and free of any nicks or burrs that would cause the part to crack during the forming operation.



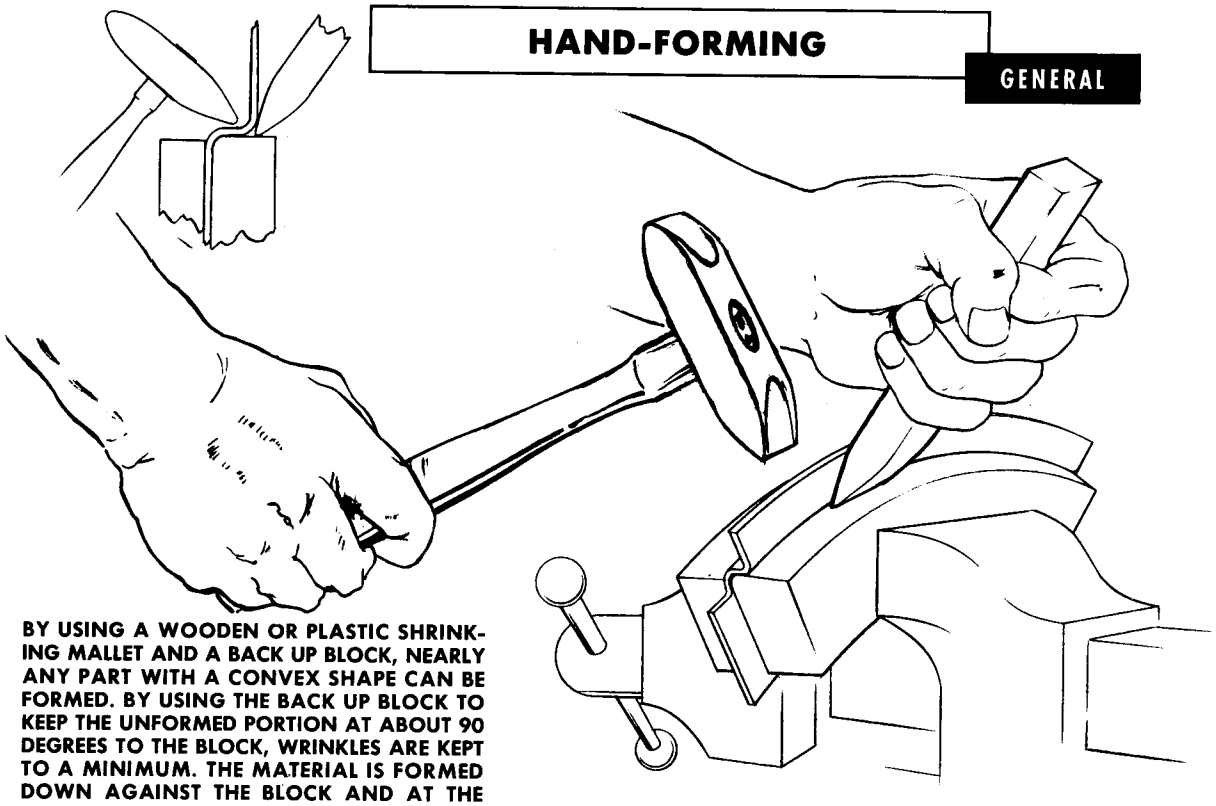
Sheet metal or extruded angles can be formed by hand with the use of a wooden block, slotted so that one leg of the angle will fit into it. The slot should be just wide enough so that the leg of the angle fits it snugly. This will prevent the angle from wrinkling as it is formed. To form the angle, tap it lightly with a plastic or rawhide mallet as it is moved along over the relieved portion of the block. Channels, "T's," and various other shapes can be formed in the same way by varying the size and number of slots cut into the block.

GEN-3-00-63

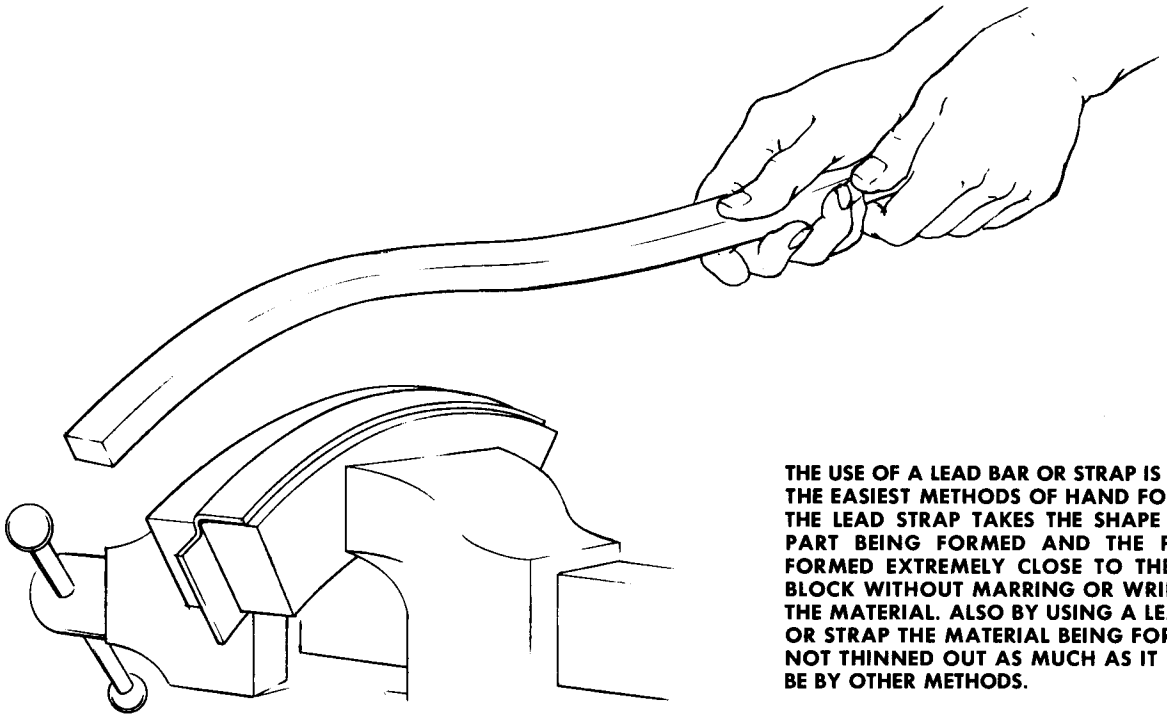
Figure 1-62A. Hand Forming (Sheet 1 of 2)

HAND-FORMING

GENERAL



BY USING A WOODEN OR PLASTIC SHRINKING MALLET AND A BACK UP BLOCK, NEARLY ANY PART WITH A CONVEX SHAPE CAN BE FORMED. BY USING THE BACK UP BLOCK TO KEEP THE UNFORMED PORTION AT ABOUT 90 DEGREES TO THE BLOCK, WRINKLES ARE KEPT TO A MINIMUM. THE MATERIAL IS FORMED DOWN AGAINST THE BLOCK AND AT THE SAME TIME AWAY FROM THE RADIUS OF THE FORM BLOCK.



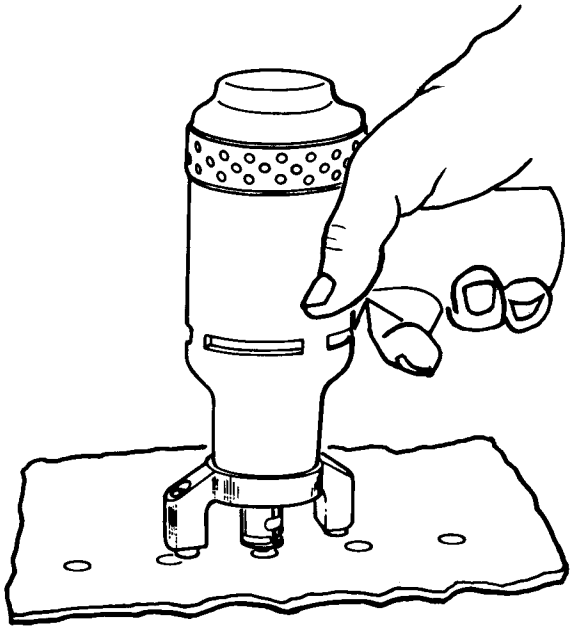
THE USE OF A LEAD BAR OR STRAP IS ONE OF THE EASIEST METHODS OF HAND FORMING. THE LEAD STRAP TAKES THE SHAPE OF THE PART BEING FORMED AND THE PART IS FORMED EXTREMELY CLOSE TO THE FORM BLOCK WITHOUT MARRING OR WRINKLING THE MATERIAL. ALSO BY USING A LEAD BAR OR STRAP THE MATERIAL BEING FORMED IS NOT THINNED OUT AS MUCH AS IT WOULD BE BY OTHER METHODS.

GEN-3-00-64

Figure 1-62A. Hand Forming (Sheet 2 of 2)

SHAVING FASTENER HEADS

GENERAL



NOTE The fastener heads may be milled with either the electric or pneumatic rivet mill to meet the flushness requirements below. The surface of the sheet must not be damaged, and the fastener head must not be reduced more than 5 percent of the head diameter.

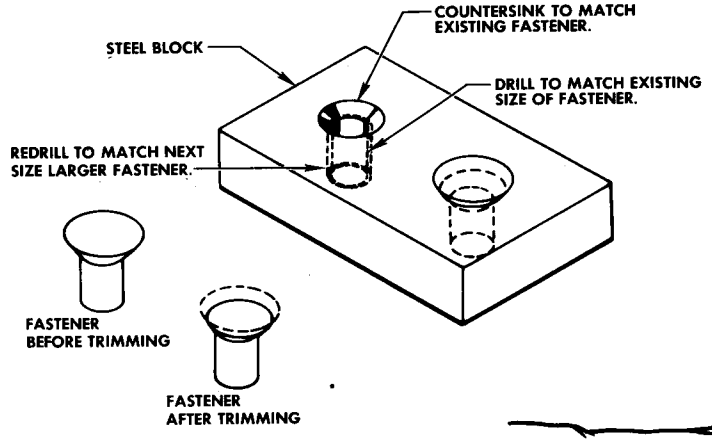
- Do not mill or shave the heads of Deutsch drive pin rivets or Jo-bolts.

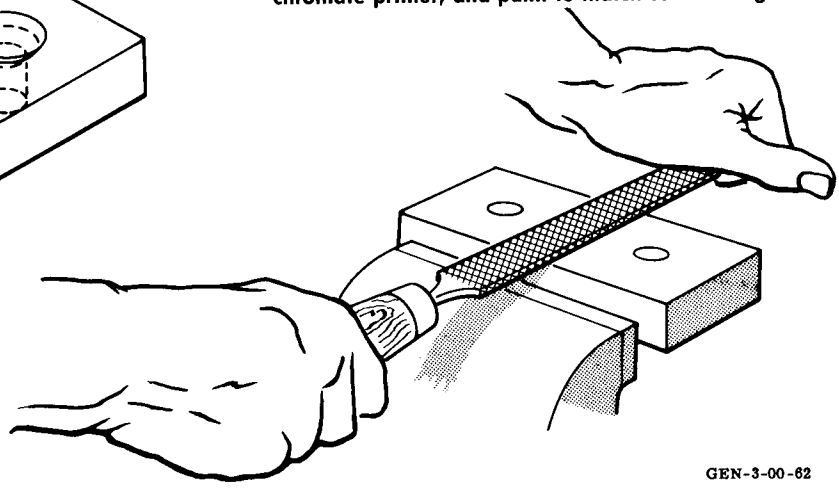
FLUSHNESS REQUIREMENTS		
	COUNTERSUNK HOLES	DIMPLED HOLES
BELOW SURFACE	0.002	0.002
ABOVE SURFACE	0.000	0.002

REDUCING FASTENER HEAD SIZE

In cases where a dimpled or countersunk hole has become enlarged, and it is not possible to redimple or countersink for the next size larger fastener, the following method may be used to trim the next size larger fastener head to match the existing dimples or countersinks.

1. Drill a hole in a steel block the same size as the existing fastener.
2. Countersink the drilled hole so that the existing size fastener head is flush with the surface of the steel block.
3. Redrill hole to next size larger fastener diameter.
4. Place the fastener to be used in the countersunk hole in the steel block, and file or grind off head flush with the surface of the block.
5. Install the fastener to be used in redrilled holes. After installation, coat head of fastener with two coats of zinc-chromate primer, and paint to match surrounding area.





NOTE This method is to be used only for the next size larger fastener.

GEN-3-00-62

Figure 1-62B. Trimming Fastener Heads

placed over the hole and held firmly in place while the scriber is drawn around it as illustrated in figure 1-62.

LOCATING SKIN TRIM LINES WITH MASKING TAPE.

Locating the trim lines for flush patch fillers, access covers, etc, when there is no access to the back side for marking, may be done with masking tape as shown in figure 1-62. This method is best when used on rectangular openings.

The cutout to be matched is filed or sanded smooth. The filler sheet is cut sightly oversize and formed to match mold line of repair area. Strips of masking tape are aligned along the edges of the opening to form a frame. The filler sheet is centered over the opening and taped down with tape of the same width as used underneath. Carefully align outer edges of outer or second tape with the first layer, and pull it tighty up over the filler. (Do not press tape into corners formed by edges of filler and base structure.) When thinner gages of filler material are used, the inner edge of the outer tape will closely match the edges of the cutout. When the filler material is thick, the outer tape will fall a little short of matching the edges of the cutout and, unless this is compensated for when marking, the filler will be too large for the opening. Unless otherwise specified, the skin gap is $\frac{1}{32}$ ($+\frac{1}{32}$, -0) inch.

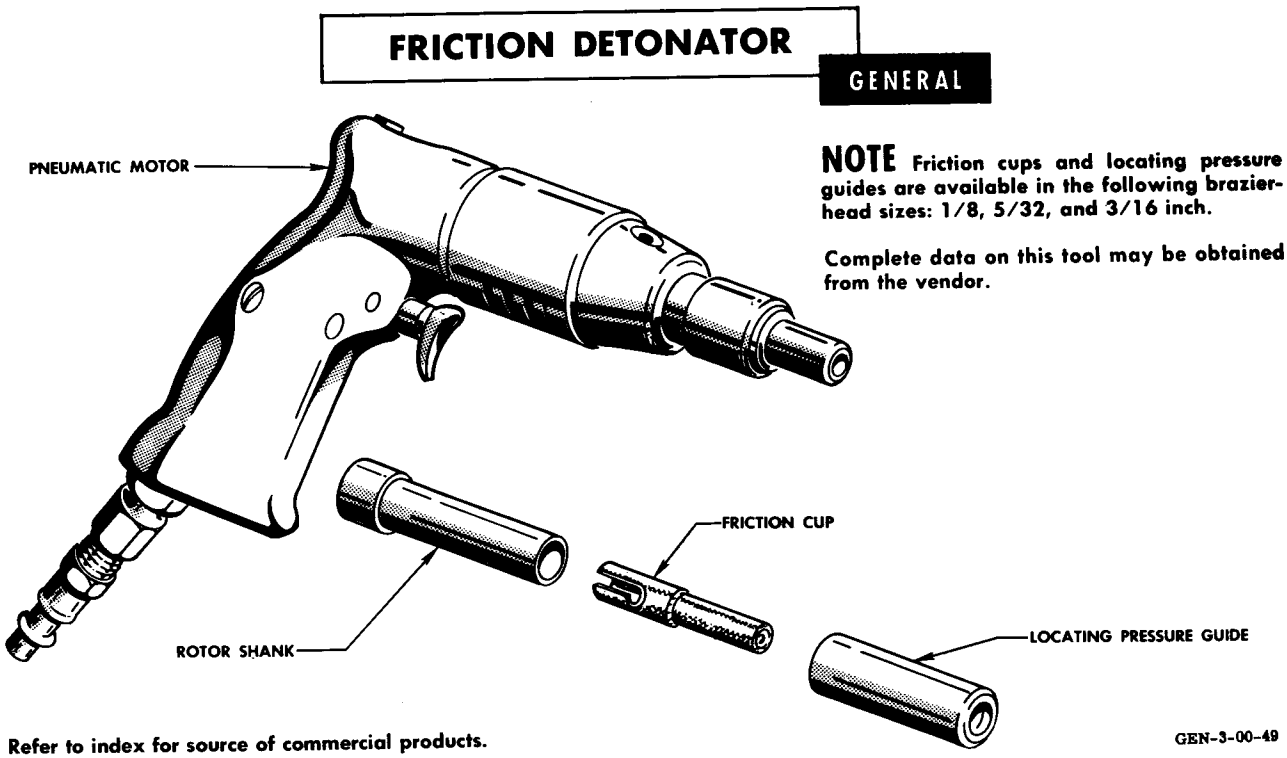


Figure 1-63. Explosive Rivet Detonator

HAND FORMING.

FLANGES AND ANGLES.

When making repairs, it is often necessary to make a part when replacement parts are not available. Several different methods of hand-forming concave and convex flanges and angles are shown in figure 1-62A.

TRIMMING FASTENER HEADS.

When an outer skin has been repaired or replaced, all new flush rivets should be inspected for flushness. If some of the rivet heads protrude excessively, they should be milled with either an electric or pneumatic rivet mill to meet the flushness requirements shown in figure 1-62B.

When the rivet hole sizes are not within the tolerances shown for standard rivets in figure 10-21, an oversized rivet should be used. However, if oversized rivets are not available, the next larger standard size must be used. Redimpling titanium for the next size larger rivet is *prohibited*. When using the next size larger rivet in titanium, the head must be trimmed to match the existing dimple.

Figure 1-62B shows the method of trimming the rivet head before installation.

When using aluminum rivets in aluminum skin, either method shown in figure 1-62B may be used.

SPECIAL TOOLS.

EXPLOSIVE RIVET FRICTION DETONATOR.

A friction detonating tool for installing explosive rivets, and which ensures proper time-temperature relationship and personnel safety, is commercially available. It consists of a high-speed pneumatic motor, a spring-loaded sleeve, and a specially treated micarta tip assembly.

The sleeve serves the dual purpose of centering the tool and preventing the rivet from turning. The tip assembly is rotated at high speed and creates the friction necessary to detonate the rivet. (See figure 1-63.) The friction detonator may be purchased from Sheridan Friction Products. Refer to index for sources of commercial products.

Detonating Tool Number SP-301

Tip Size	Type	Part Number
1/8	Braz Hd	JJ650-130 detail 101-1
5/32	Braz Hd	JJ650-130 detail 101-2
3/16	Braz Hd	JJ650-130 detail 101-3

DRILL EXTENSION.

The drill extension tool, shown in figure 1-64, with a complete assortment of chucks, makes an extension drill of any standard length drill bit. It is relatively

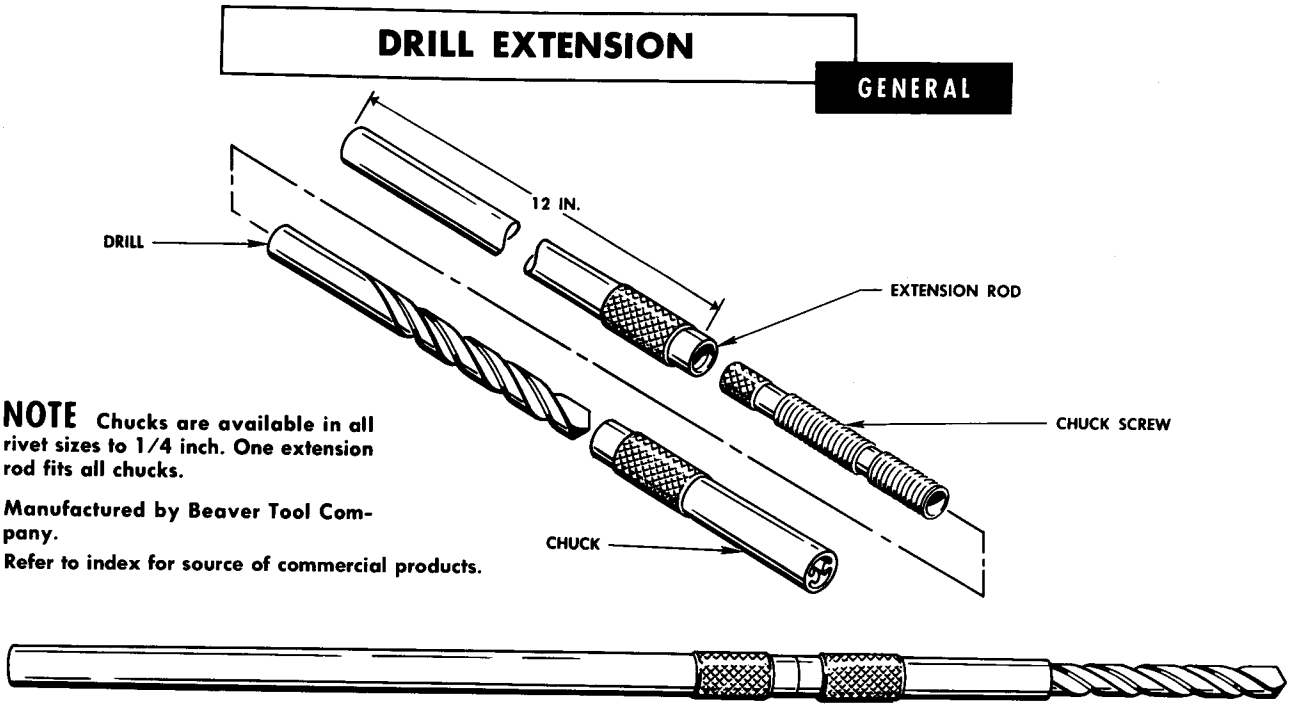


Figure 1-64. Drill Extension

safer to use than a regular extension drill because the extension is rigid. It will not whip or bend, as a slim extension drill tends to do when not properly controlled. It is limited, however, to working through holes of ¼-inch diameter or larger.

DISPOSABLE-CARTRIDGE SEALANT GUN.

The disposable-cartridge sealant gun (figure 1-65) provides a simple and convenient method for injecting sealants, adhesives, etc, into prepared holes, and for laying external fillets in air-tight or fluid-tight sections of the airplane.

All components of the gun which come in contact with the sealant are made of polyethylene plastic and are of a disposable nature. They may be cleaned and reused, however, by stripping cured compound from plastic parts, since the compound adheres very poorly to this plastic after it has cured.

IDENTIFICATION OF MATERIALS. BY CHEMICAL TESTING.

It is often necessary to distinguish 7075 from the other aluminum alloys, and magnesium alloys from those of aluminum.

IDENTIFICATION OF 7075 ALUMINUM ALLOY BY CHEMICAL TEST.

A chemical spot-test kit, containing the solutions and equipment shown in figure 1-70, will aid in this type of inspection. The formula for preparing the 7075 spot-test solution, with specification numbers of chemicals used, is as follows:

- 5 grams of cadmium sulphate (commercial)
- Milliliters of hydrochloric acid (Federal Specification O-A-86)
- 3 grams of sodium chloride (commercial)
- 100 milliliters of distilled water

Other materials and specification numbers are as follows:

- Chemical film solution (Specification MIL-C-5541)
- 20% solution chromic acid (Federal Specification O-C-303)
- Phosphoric acid cleaner (Specification MIL-C-5410, Type 1)
- Silver nitrate crystals (commercial)
- Methyl ethyl ketone (Federal Specification TT-M-261)
- Nitric acid (Federal Specification O-A-88)

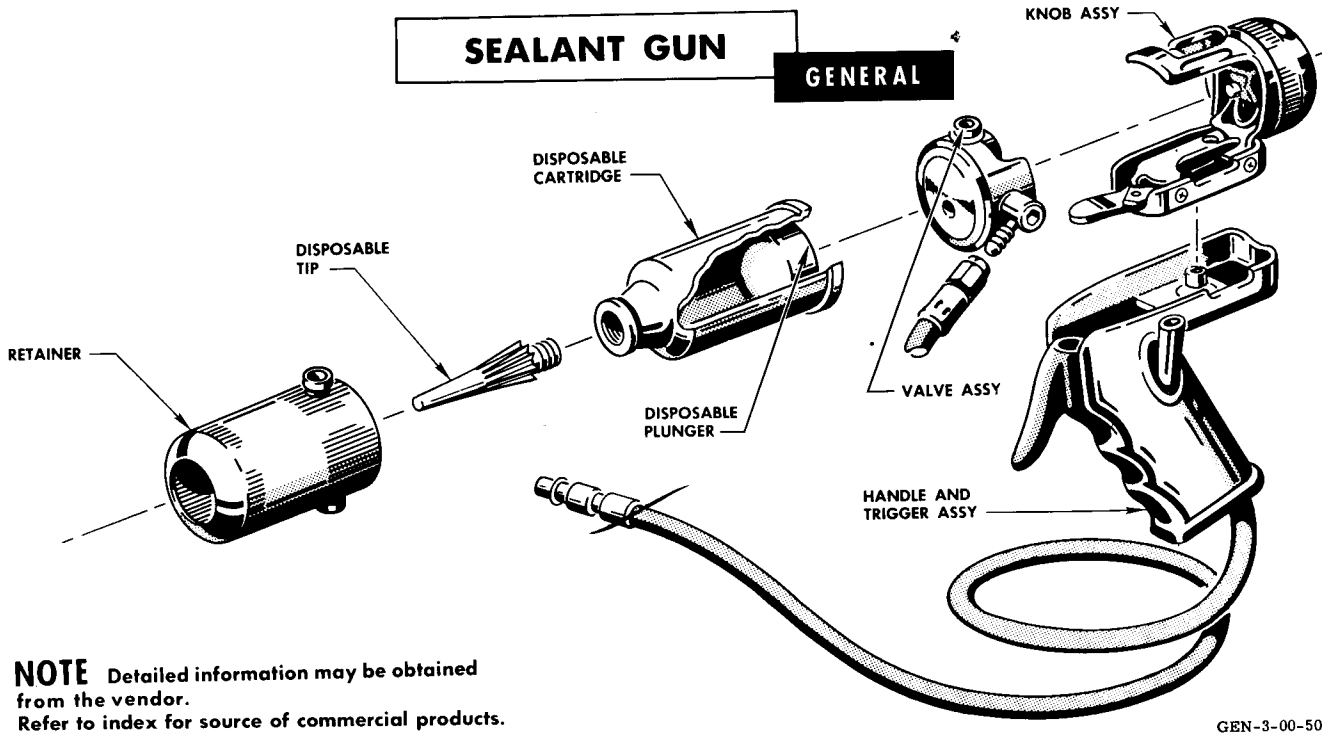


Figure 1-65. Disposable Cartridge Sealant Gun

Most of the solutions contained in this kit are both poisonous and corrosive. Reasonable precautions should be taken in handling them. If any solution is spilled on the skin, immediately wash the area well with cold, running water. If the area feels irritated (stings or feels warm) after washing, report to the dispensary for treatment.

Because 7075 is an aluminum alloy which contains zinc, it is quickly and accurately identified by a simple chemical test. Because there is one percent zinc in the cladding of 7075, this test will work on either bare or clad material.

Remove paint, if present, from an area about the size of a dime. Clean area thoroughly with solvent (bottle No. 3). Remove surface film by lightly sanding with No. 320 grit abrasive cloth. To avoid contamination, use a fresh piece of abrasive cloth for each test.

Place one drop of 7075 spot test solution (bottle No. 1) on the prepared surface. Allow to stand one minute; then rinse off the test solution with water and examine the spot. If the alloy is 7075, a dark gray deposit of cadmium will be seen. It is recommended that the sample pieces of 7075 and 2024 included in the kit be run with the test piece for comparison.

After the alloy is identified, it should be refinished as follows if material is to remain unpainted:

1. Remove any dark deposit with Bon Ami or very fine abrasive cloth.
2. Swab the area with phosphoric acid cleaner (bottle No. 5) and keep wet for 3 to 5 minutes.
3. Rinse thoroughly with water. Dry to avoid spotting.

After the alloy has been identified it should be refinished as follows if material is to be painted:

1. Clean area as outlined in the three steps above.
2. Swab cleaned area with chemical film solution (bottle No. 2) and keep wet for 3 to 5 minutes.
3. Rinse thoroughly with water and allow to dry. (Do not rub.)
4. Apply two coats of zinc chromate primer and any colored finish coats required.

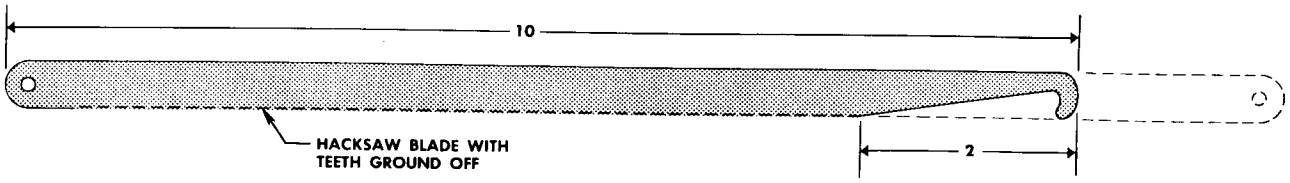
IDENTIFICATION OF MAGNESIUM ALLOYS BY CHEMICAL TEST.

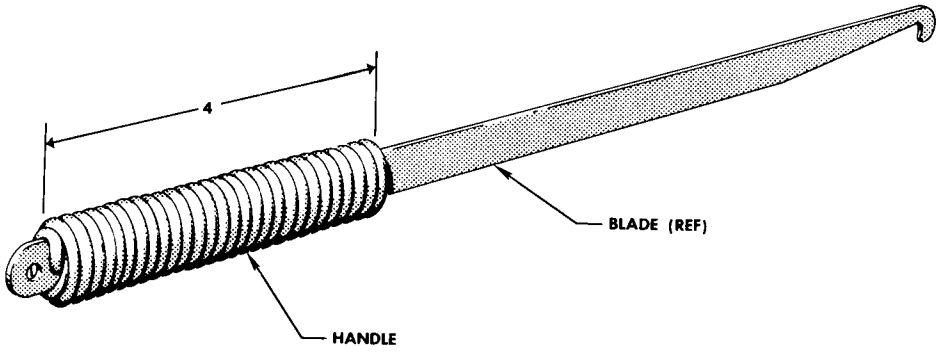
Magnesium alloy parts may be separated from those made of aluminum alloys by using the following spot-test procedure: Remove paint, if present, from an area about the size of a dime. Wipe area clean with solvent (bottle No. 3). If methyl ethyl ketone (MEK) is not available, lacquer thinner may be substituted.

Dissolve a few silver nitrate crystals (bottle No. 4) in a small amount of distilled water (bottle No. 7). Place

CHIP CHASER

GENERAL





NOTE Any thin steel may be used; however, a springy material will give more satisfactory service.

Handle may be made of wood, phenolic, scrap Plexiglas, etc. Rivet or tape to blade.

Use chip chaser to remove drill chips, etc, from between assemblies which cannot be separated after cutting operations.

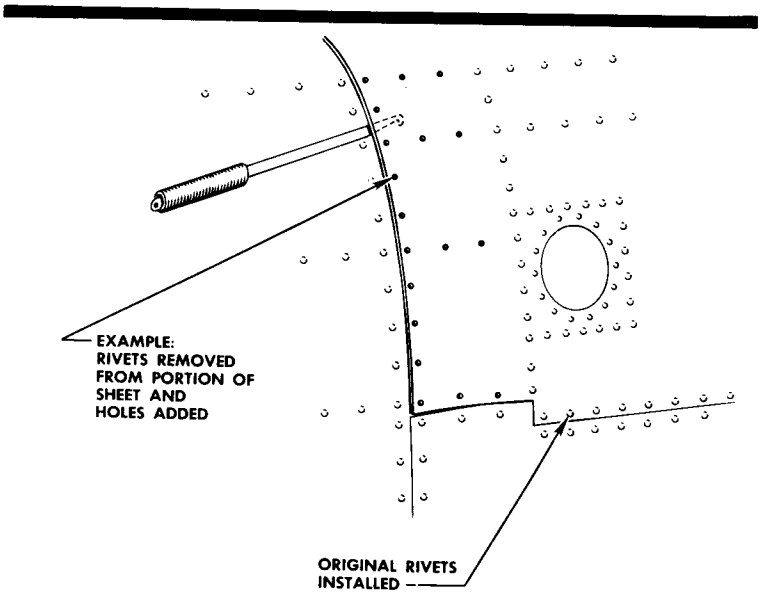


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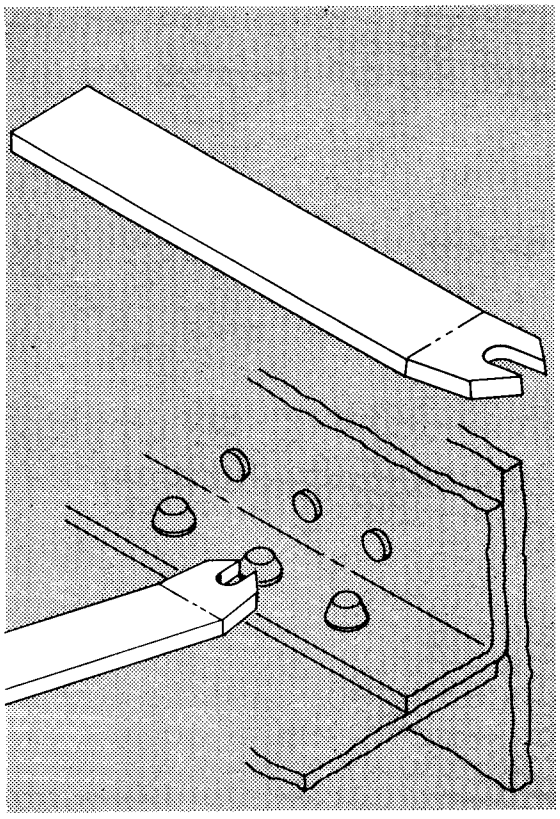
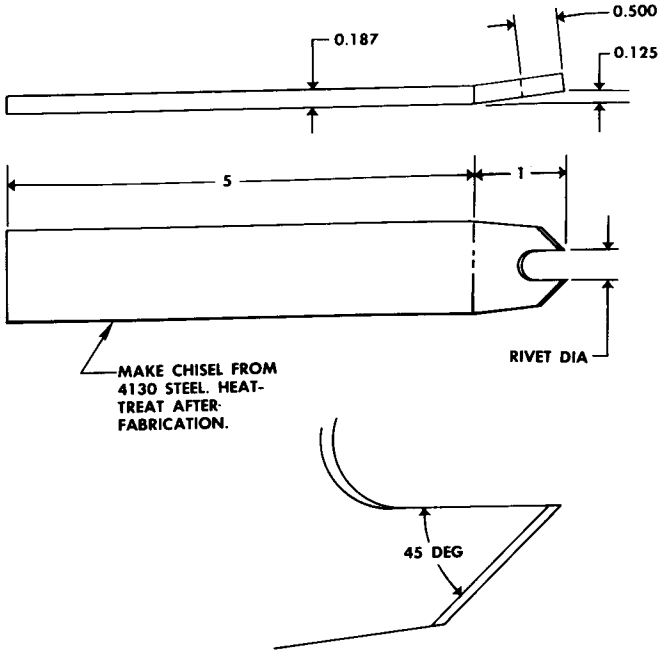
Figure 1-66. Chip Chaser

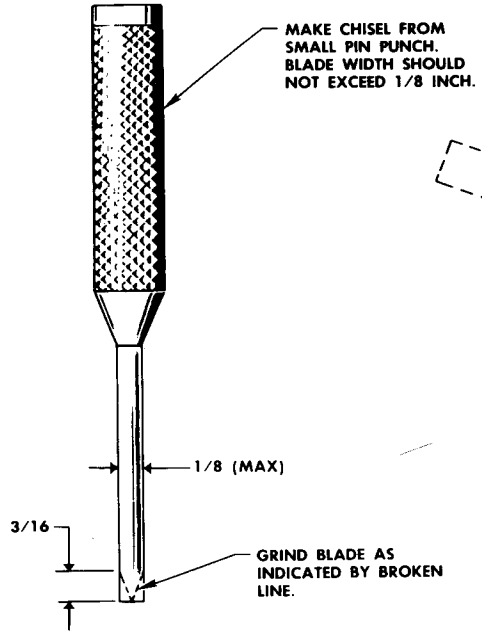
FORKED CHISEL

GENERAL





NARROW-BLADE CHISEL



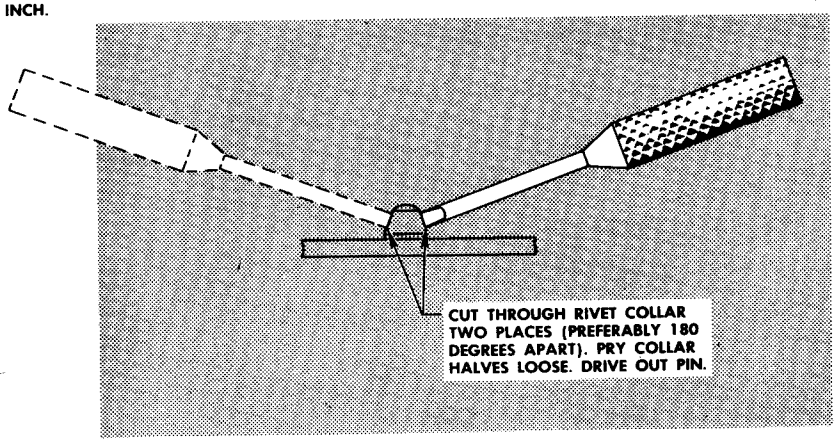


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Figure 1-67. Cutting Hi-shear Rivet Collars

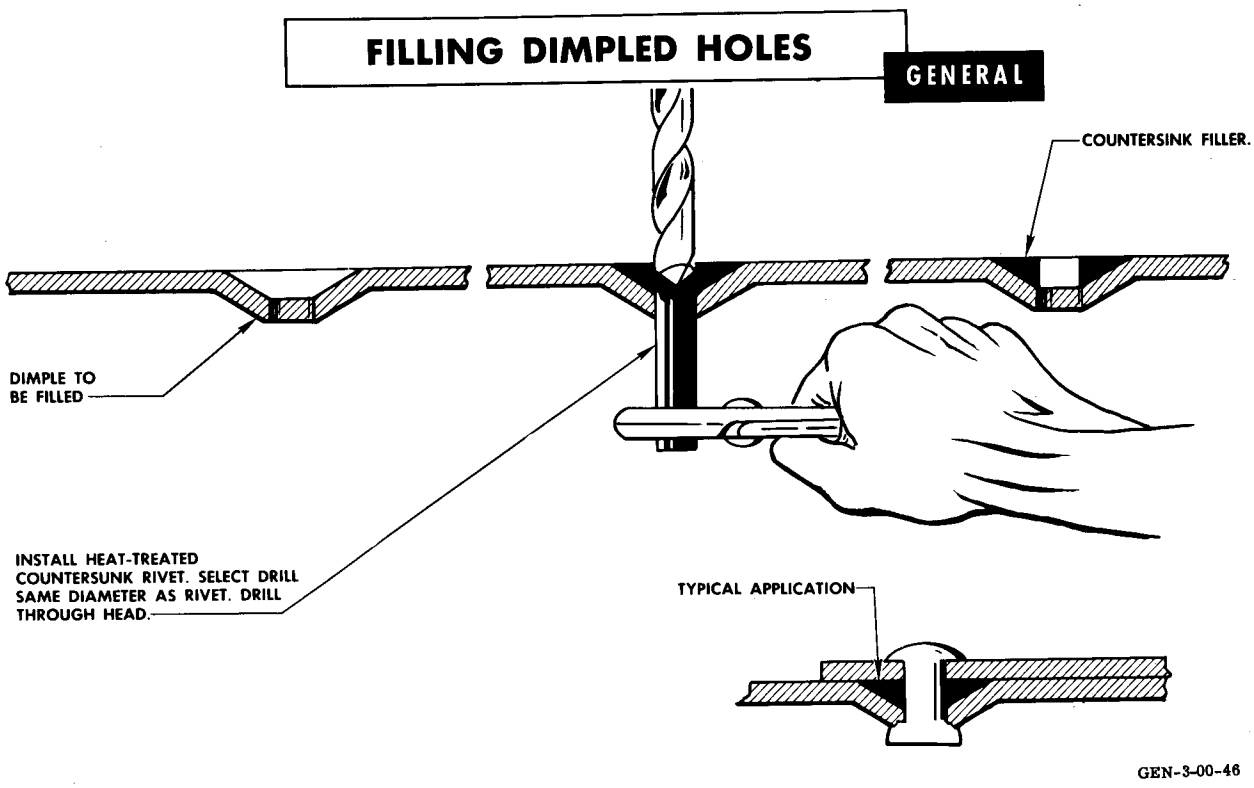


Figure 1-68. Filling Dimpled Holes

one or two drops of the silver nitrate solution on the test area. If the material is magnesium, the clear liquid will immediately turn black. If the material is an aluminum alloy there will be no reaction. It is recommended that the operator check the aluminum and magnesium samples included in the test kit to familiarize himself with these reactions.

Silver nitrate in solution cannot be stored for long periods. However, if tests are to be made frequently, a small bottle of solution may be made up and used until it starts to darken or become cloudy.

The area tested must be thoroughly washed with water and treated as follows: If the material is an aluminum alloy, refinish as directed under "Identification of 7075 Aluminum Alloy by Chemical Test."

If material is a magnesium alloy:

1. Wash test area thoroughly with water.

2. Remove black stain with Bon Ami or fine abrasive paper.

3. Swab with 20 percent solution of chromic acid (bottle No. 6). Allow to stand 2 minutes.

4. Rinse with water and dry.

5. Apply one coat of wash primer. (Specification MIL-C-15328 or Specification MIL-C-8514.)

6. Apply two coats of zinc-chromate primer (three coats on areas containing dissimilar metals).

7. Apply colored finish as required.

CHEMICAL TEST FOR DAMAGED CLADDING.

To determine if the cladding or alclad aluminum alloy has been penetrated by scratches or nicks, thoroughly clean the surface with methyl ethyl ketone (Federal Specification TT-M-261) and apply a 10-percent solution of sodium hydroxide (Specification MIL-S-11795) to the surface. If the damage has gone through the alclad coating, a black or dark brown discoloration will appear. Because of its corrosive action, the solution should not remain on the surface more than 2 minutes. After test is completed, the area must be thoroughly washed with water.

INSPECTION METHODS.

PENETRANT INSPECTION METHOD.

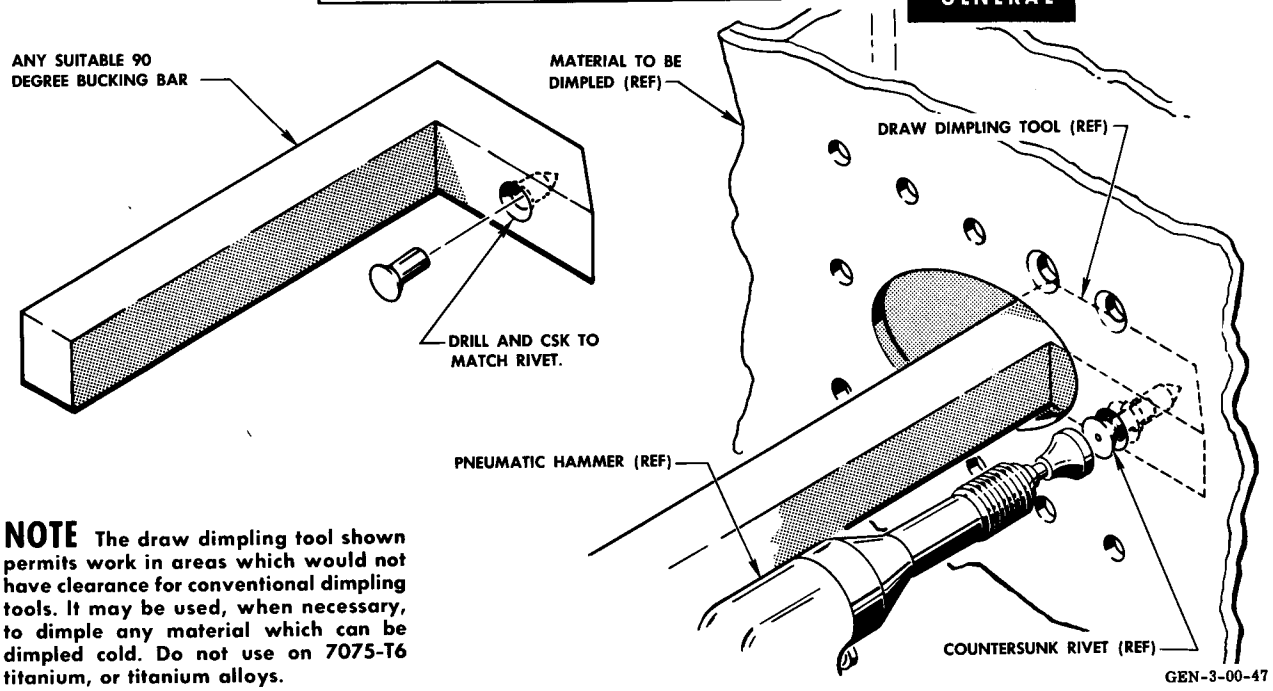
Penetrant inspection materials are of two types, fluorescent, which fluoresces or glows strongly when exposed to black light (near ultraviolet), and nonfluorescent, which contrasts strongly with its background when examined under white light. Fluorescent-penetrant inspection is preferred; however, there are times when only the dye-penetrant method can be used, such as:

1. When no electrical current is available.
2. When anodizing and chromate finishes affect sensitivity.
3. When a large area is required to be inspected.
4. When it may be difficult to rerun the parts.

The penetrant method of inspection may be used for detection of surface defects or indications on nonabsorbent, nonporous materials and parts which will not

DRAW DIMPLING TOOL

GENERAL



NOTE The draw dimpling tool shown permits work in areas which would not have clearance for conventional dimpling tools. It may be used, when necessary, to dimple any material which can be dimpled cold. Do not use on 7075-T6 titanium, or titanium alloys.

Figure 1-69. Draw Dimpling Tool

CHEMICAL SPOT-TEST KIT

GENERAL

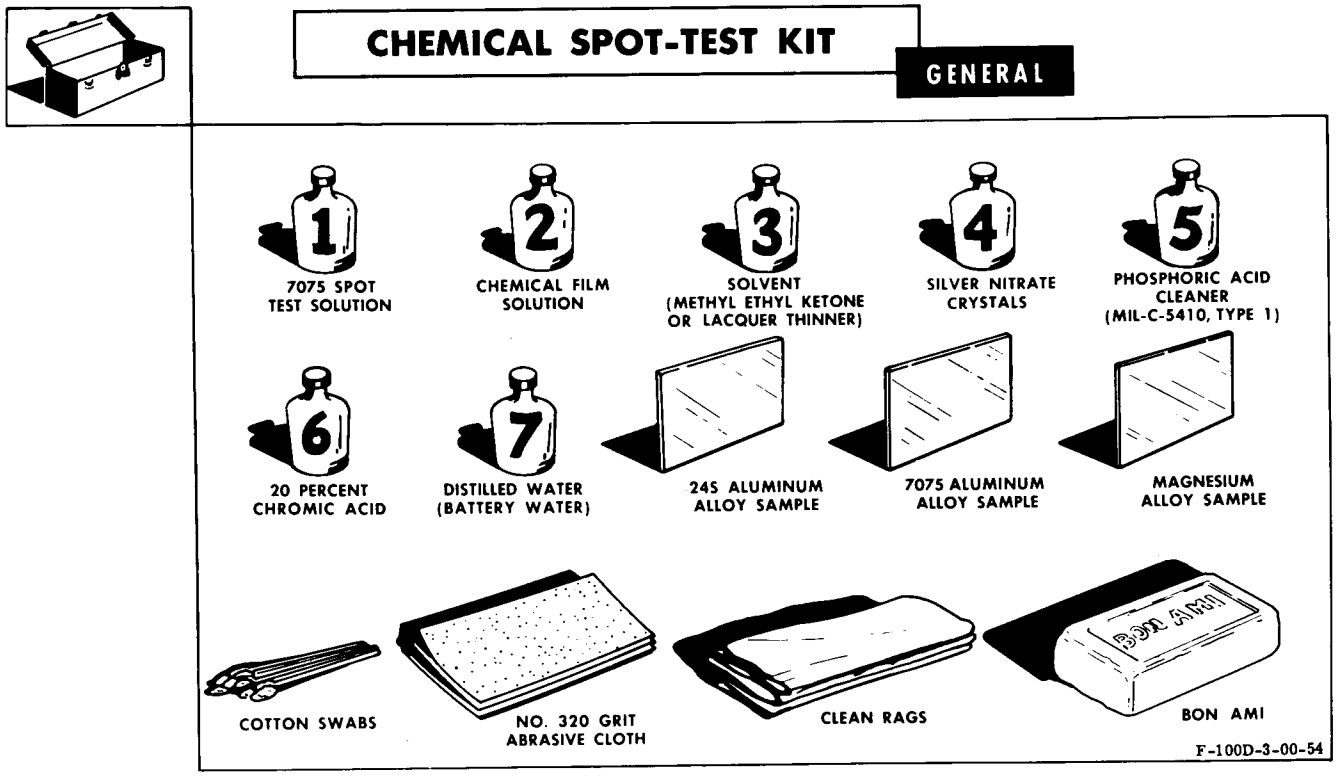
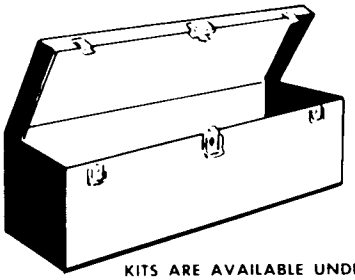


Figure 1-70. Chemical Spot-test Kit

NONFLUORESCENT PENETRANT INSPECTION FIELD KIT

GENERAL



KITS ARE AVAILABLE UNDER THE FOLLOWING TRADE NAMES:

DYE-CHEK

* Turco Products

SPOT-CHEK

* Magnaflux Corp

MET-L-CHEK

* Met-l-chek Co

* Refer to index for source of commercial products.

F-100D-3-00-55

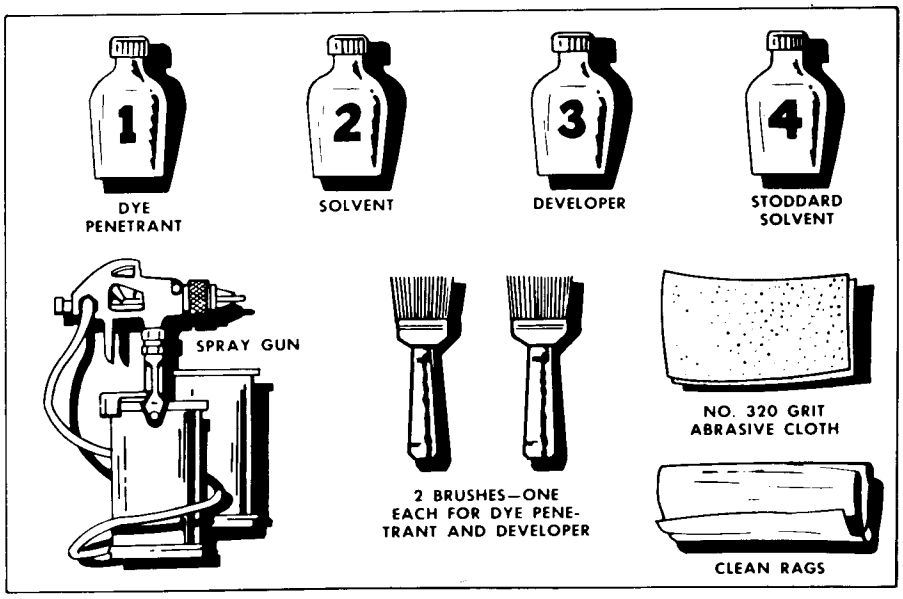


Figure 1-71. Nonfluorescent Penetrant Inspection Field Kit

FLUORESCENT AND NONFLUORESCENT PENETRANT ACTION

GENERAL

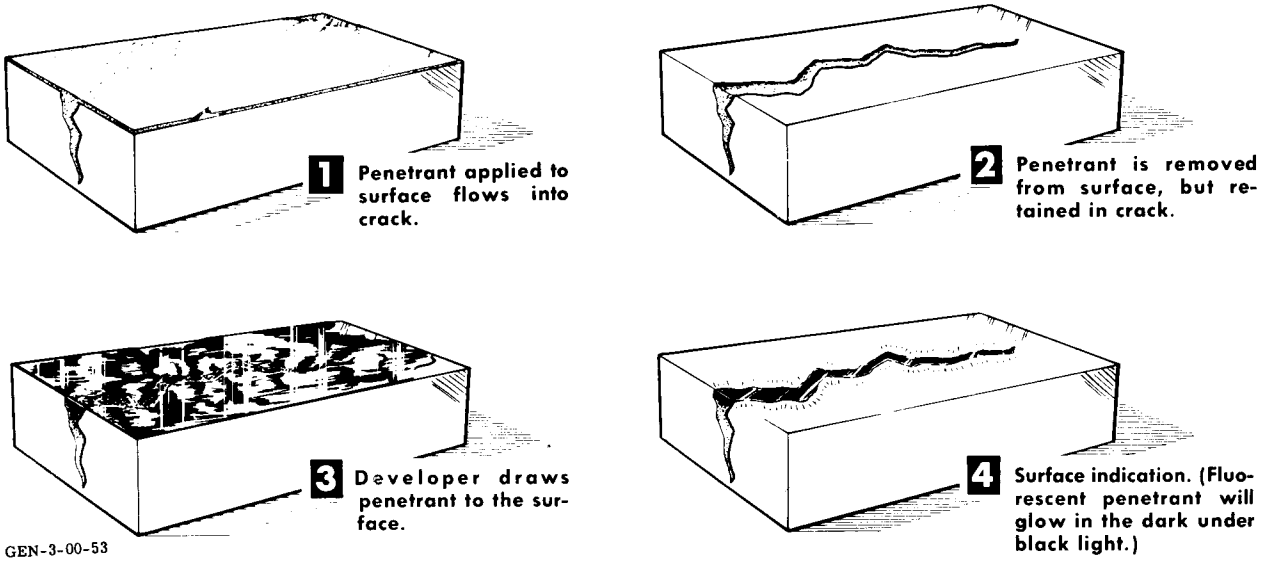


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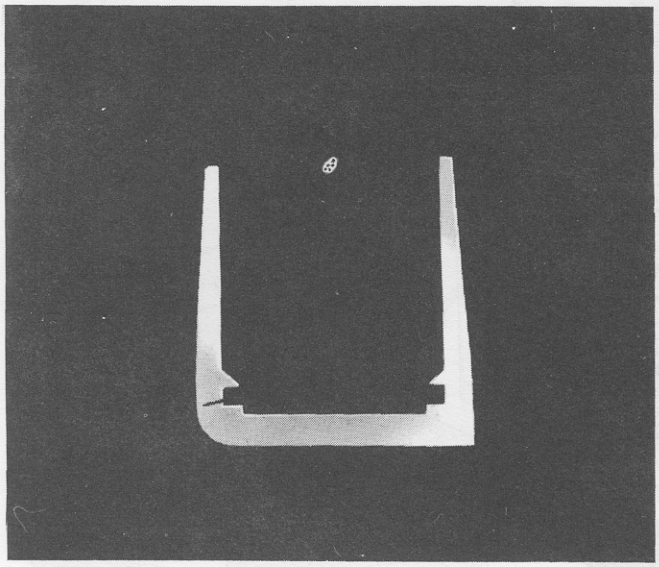
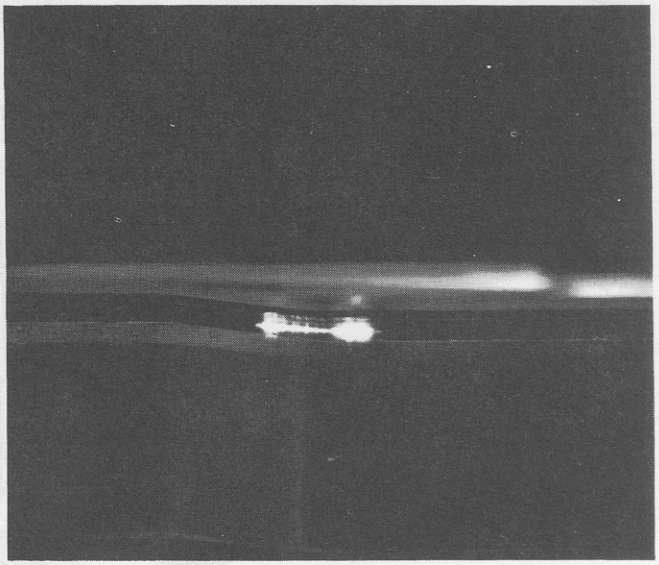
Figure 1-72. Fluorescent and Nonfluorescent Penetrant Action

FLUORESCENT PENETRANT INDICATION OF CRACK

GENERAL



CROSS SECTION VIEW OF CRACKED CHANNEL



FLUORESCENT PENETRANT INDICATION OF SAME CRACK AS SEEN IN A MIRROR

GEN-3-00-54

Figure 1-73. Fluorescent Penetrant Indication of Crack

be harmed by chemical or physical reaction with the inspection materials. Typical examples of items which may be inspected are machined steel parts, aluminum castings, glazed ceramic and glass items, and certain plastic parts.

Parts to be inspected must be free of paint and thoroughly cleaned, using approved paint removers and cleaning solvents. Chemicals will be handled and stored in accordance with applicable Technical Orders.

The surface to be inspected should be dry and free from dirt, scale, oil, or grease. Parts may be vapor-degreased or cleaned with solvent (bottle No. 2 or 4). (See figure 1-71.)

Apply dye penetrant (bottle No. 1) by brush or spray and allow to stand for about 2 minutes. Recover the same area with additional penetrant and allow to stand for another 3 minutes. For extremely small flaws such as grinding checks, apply penetrant three times and allow 5 minutes penetration time after each application.

Wipe off as much penetrant as possible with a clean, dry rag and follow with a rag impregnated with solvent (bottle No. 2). The surface should then be wiped dry with another clean rag. It is important to *thoroughly* remove all penetrant so as to avoid false indications. A cotton swab dipped in solvent (bottle No. 2) is helpful in drying drilled holes, threads, re-entrant angles, etc. Do not soak parts by immersing them in cleaner, as this might weaken indications. Do not dry parts with strong air blast.

The developer (bottle No. 3) must be kept thoroughly mixed while in use. Before applying it, shake the bottle vigorously. Apply a thin, even coat of developer and observe part closely as the developer dries. Blowing on the part gently with the mouth speeds the drying and is sometimes helpful in detecting small cracks.

PORTABLE X-RAY APPLICATIONS

GENERAL

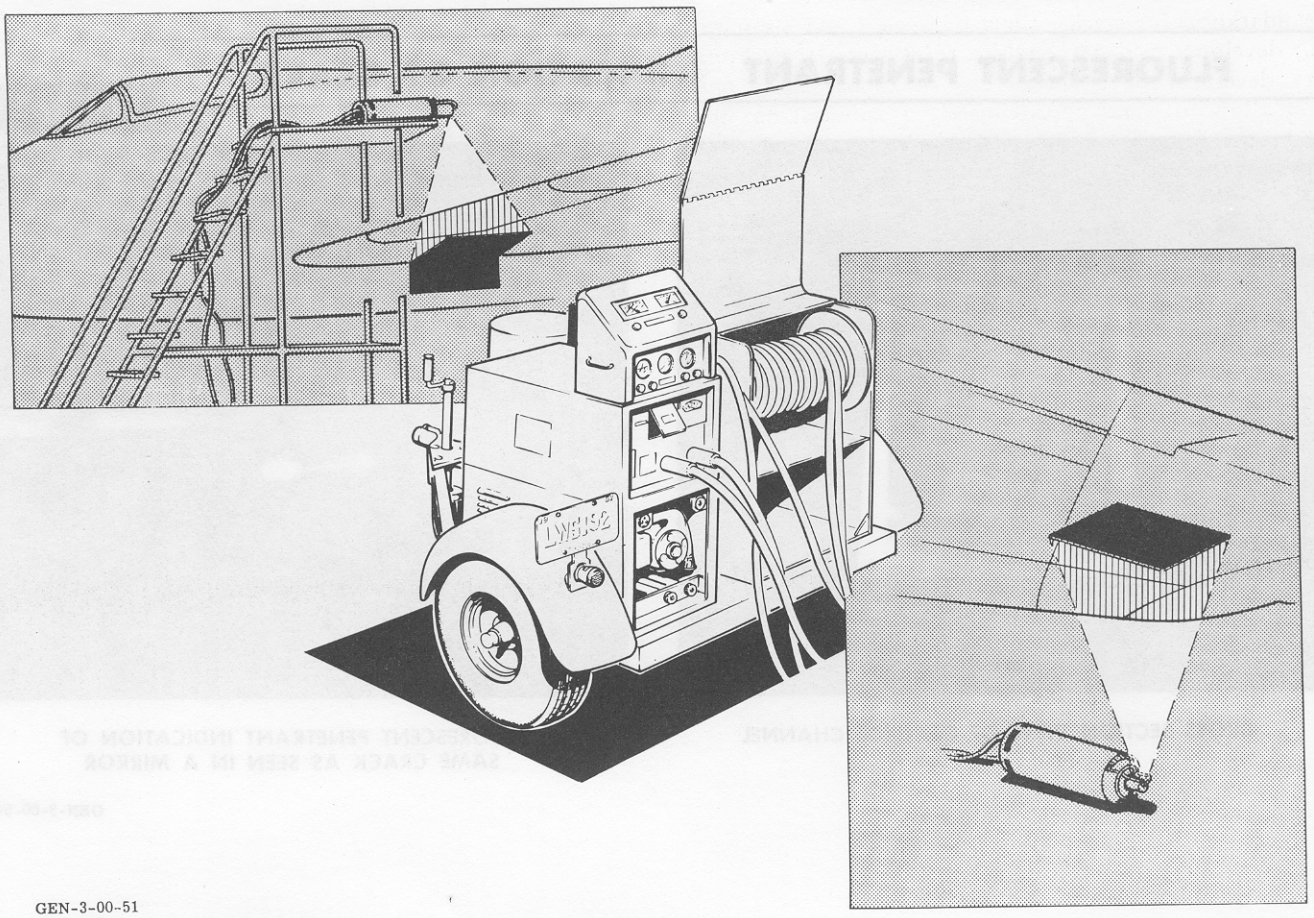


Figure 1-74. Portable X-ray Applications

Make only one application of developer, as repeated applications tend to wash away small indications.

Cracks or voids will exude dye (figures 1-72 and 1-73) and cause a bright red indication. A red line indicates a crack, cold shut or lap. Random red dots show pits or porosity. A series of red dots in a line indicates a very tight crack. The extent or volume of the discontinuity can be estimated by the width of the indication, or amount of spreading out. A diffused pink color shows that all of the penetrant was not thoroughly cleaned off before the developer was applied. If a very tight crack is suspected, allow several minutes for the indications to appear.

It is important to avoid contaminating the solvent and developer (bottles No. 2 and 3) with the dye penetrant. This dye is oil-soluble and can be removed from the hand, etc, with ordinary solvents. All four solutions are chemically noncorrosive and can be left in contact with metals as long as desired. Keep all solutions tightly capped when not in use, to minimize evaporation.

The procedure for fluorescent-penetrant inspection is as follows:

1. Clean inspection area in same manner as for dye-penetrant method.

2. Spray penetrant on area to be inspected so that surface is covered. Allow penetrant to settle into surface cracks for 10 minutes on aluminum and magnesium; 20 minutes on stainless steel; and 20 to 30 minutes on titanium and high-temperature alloys. Do not use penetrant in temperatures below 40°F. (70°F is normal usage temperature.)

3. When sufficient penetration time has been allowed, spray cleaner on a cloth and remove penetrant. Repeat, using clean areas of cloth until excess penetrant has been removed.

Note

View inspection area under black light, preferably in semidarkness, to be sure all excess penetrant has been removed.

4. After cleaning operation, apply either wet or dry developer. Before applying wet developer, shake can vigorously and then spray a thin, even coat onto inspection area. Before inspecting area under black light in semidarkness for cracks, allow between 2 and 3 minutes for developer to dry to a white, translucent layer. Apply dry developer by a shaker can or rubber spray bulb and allow to stand approximately 5 minutes. For surfaces that are rough or porous, dry developer gives superior results.

Use the following precautions while handling fluorescent-penetrant materials:

1. Use cleaner and developer in an adequately ventilated area and avoid prolonged contact with skin.

2. While using penetrant, wear neoprene gloves and keep penetrant off clothes.

3. Wash with soap and water to remove penetrant, cleaner, or developer from skin.

For further use, advantages, and disadvantages of penetrant inspection methods, refer to T.O. 33B1-2-1-2 and the manufacturer's field inspection kit, AZ-43, which consists of the following or equivalent:

- Zyglo cleaner ZC-7
- Zyglo penetrant ZL-22
- Zyglo dry developer ZP-4
- Zyglo developed ZP-9
- Black light—Magnaflux ZB-26

Refer to Appendix I for sources of commercial products.

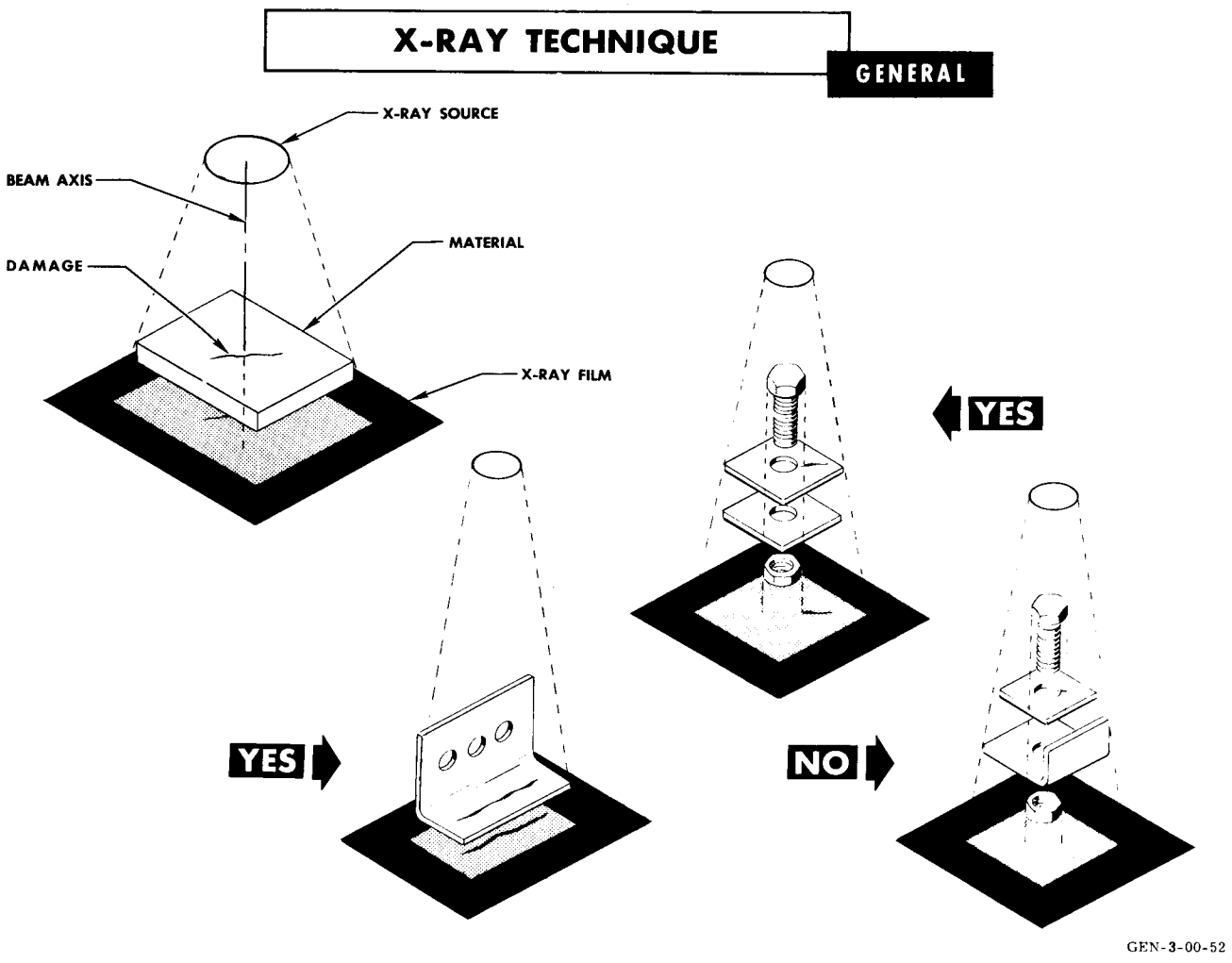


Figure 1-75. X-ray Technique

PORTABLE X-RAY INSPECTION METHOD.

X-rays are electromagnetic vibrations similar to light, but of a wave length short enough to pass, to a limited extent, through solid materials. X-rays also have the ability to expose photographic film while it is protected in a light-proof pack. Therefore, X-rays passing through material such as an aircraft part, are capable of producing a shadow picture on film. With proper exposure, slight changes in density of the material may be visible.

This method of inspection can be valuable in gaining information about the internal condition of materials. It does have limitations. Thorough training in methods and procedures in use of X-ray equipment is essential. To be able to understand what the X-ray reveals is also a skill which must be developed.

When inspecting airplanes which have been damaged by hard or unusual landings, or by exceeding established flight limits, X-ray can be a very valuable aid. It has also been widely used for periodic fatigue-checking of critical joints and fittings on airplanes which have accumulated a large number of flying hours.

X-rays will reveal cracks in hidden sheet metal parts such as ribs and spar webs. When exposures are taken with the X-ray beam axis parallel to the plane of mating surfaces, separation of those surfaces will be revealed. Enlarged bolt holes, and sheared or partially sheared fasteners may also be detected. Cracks in extrusions and heavy fittings can be seen when some portion of the surfaces of separation are parallel to the X-ray beam axis. Cracks cannot be seen when the X-ray beam is at an angle to the separated surfaces. (See figure 1-75.)

In X-ray inspection of assembled aircraft components, many factors must be considered. One of these factors will be how great and how confusing a mass of material must be penetrated along with the material in

question. When the unwanted material is too thick, or would produce confusing patterns on the film, it makes interpretation difficult, if not impossible. Also to be considered is the presence of sealing compound, which can partially conceal cracks and may give indications similar to cracks where there are none. When there is any doubt, the sealing compound should be removed and another exposure taken. Liquids are another factor to consider. X-rays are harmless to all aircraft materials and fluids including fuel. The transformer head is a sealed unit, and it may be used on a fueled airplane. However, when exposures are made through fuel or oil cells, the cells should be drained because even a small amount of liquid tends to cloud the indications.

When a request for X-ray services from another base is being made and personnel making the request are not familiar with portable X-ray procedure, the exact parts to be inspected should be stated in the request. Personnel familiar with the airplane and X-ray possibilities will then be able to determine whether or not the trip would be of practical value.

For further use, advantages, and disadvantages of X-ray inspection, refer to T.O. 33B3-1-1.

PORTABLE X-RAY INSPECTION METHOD SAFETY MEASURES.

Safety of personnel is of extreme importance. A portable unit cannot have all the built-in protection that stationary units have. Splash radiation from the object being X-rayed and from concrete floors must be considered. All safety information relative to the equipment in use should be obtained, and used in establishing procedures. The area which must remain clear, when X-ray equipment is being used, may be found by using a Geiger counter or scintillator. This should be done by the radiological defense officer, and the findings should be approved by the medical officer. Personnel regularly using X-ray equipment should wear radiation-exposure badges. These badges, which are miniature film packs, measure the amount of radiation the wearer has received. To avoid injury from overexposure, the film should be developed and examined at regular intervals. The radiological defense officer or medical officer will have information on how to get these badges and what to do with them.

MAGNETIC PARTICLE INSPECTION METHOD.

Magnetic particle inspection is a most dependable form of inspection; however, its use is limited by the accessibility of the part to be inspected and by the metallic composition of the part. The most common metals which *cannot* be inspected by this method are aluminum, magnesium, brass, copper, bronze, lead, titanium, all alloys of these metals, and some stainless steels.

Other limitations for magnetic particle inspection are plating or surface conditioners. Plating cannot be more than 0.004 inch thick. Surface conditioners, such as Dulite, Parkerizing, and lacquer coatings render magnetization impossible.

Magnetic particle inspection derives its value from the fact that a metallic part can be magnetized and, consequently, a crack or defect sets up its own polar system of north to south so that when the indicating powder is applied, a definite magnetic field will be indicated about the defect.

Parts to be inspected should be free from oil, grease, water, rust, and scale. Cleaning of parts is accomplished in the same manner as for other methods. Finally, the part should be checked for absolute dryness.

All removable parts should be magnetized and checked about their two major axes, since defect direction is not known, and the direction of the magnetic field will determine whether both major axes can be checked on certain parts.

The part is circularly magnetized by attaching leads to each end of the part to be inspected and by passing dc current directly through the part.

The part is longitudinally magnetized by wrapping coil three times around using 2/0 cable or five times around using 4/0 cable for maximum ampere turns and then activating the dc current. When the inspection part is tubular, insert cable through shaft and then activate dc current.

There are two methods of magnetic particle inspection: continuous and residual. Inspection by the continuous method is limited by the fact that some metals will not retain their magnetism after the current is shut off. In this case, the indicating powder should be applied as a light spray while current is flowing. Inspection by the residual method is the same as by the continuous method except the current can be turned off and the part removed and inspected.

There is no specific value of current that can be prescribed for a particular part in order to obtain the required magnetism. Only a range of current can be prescribed for a particular part configuration. In general, the range of current to be applied is 500 amperes for a half inch to 4000 amperes for 4 inches of greatest width or diameter, the increase being at a rate of 1000 amperes per inch.

Because of the high voltage and high current involved in operating the equipment, time limits are set on various current shots to avoid burning up the lines. Follow-are the recommended current time intervals:

- DC output duty cycle

Continuous	0 to 1250 amperes
One minute on, 5 minutes off	1250 to 2000 amperes
One minute on, 10 minutes off	2000 to 3000 amperes

- AC output duty cycle

Continuous	0 to 600 amperes
Two minutes on, 5 minutes off	600 to 1200 amperes
One minute on, 5 minutes off	1200 to 2000 amperes
One minute on, 10 minutes off	2000 to 3000 amperes

In general, parts that are magnetized must be demagnetized because of the following possible detrimental effects:

1. Affecting of nearby instruments.
2. Interference with later machining of part by attracting chips to part.
3. Interference with later magnetic particle inspection.
4. Interference with removal of magnetic particles during the cleaning process.

Demagnetization is accomplished by wrapping the coil around the part while the coil is plugged to the ac outlet, setting the current-regulating switch on a higher value of current than was used to magnetize, and slowly drawing the part from the coil or drawing the coil from the part. Drawing the coil from the part may be more convenient at times.

For further use, advantages, and disadvantages of magnetic particle inspection method, refer to T.O. 33B2-1-1 and 33B2-4-1. Equipment to be used is Model KCH-3D portable magnetic particle inspection equipment, Air Force Type III.

CHARTS AND TABLES.

BEND ALLOWANCE CHART.

See figure 1-76.

SETBACK CHART.

See figure 1-77.

MINIMUM BEND RADII TABLES.

See figure 1-80.

TORQUE TABLES.

See figure 1-79.

SHOP MATHEMATICS.

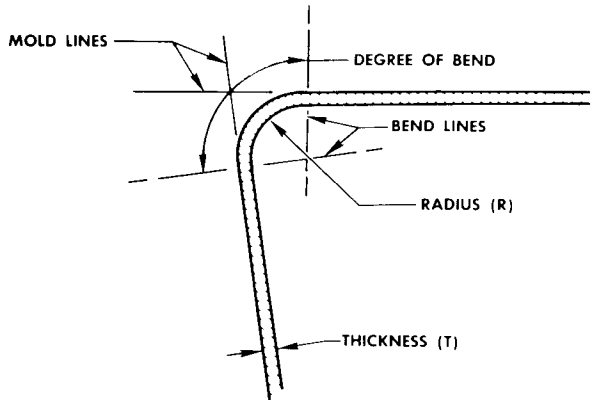
See figure 1-81.

BEND ALLOWANCE CHART

GENERAL

The empirical formula for finding the bend allowance of material required for a bend = (0.01743 × radius) × (0.0078 × thickness)

This equals the material required for one degree of bend.



1 Multiply the numbers in the table by the degrees of bend to determine the amount of material required for the bend.

Example:
95 deg bend, 1/8 in. radius, 0.040 in. material
0.00249 × 95 = 0.2365 in.

2 For bends larger than one inch radius, the amount of material required for the bend is calculated as the arc of a circle, the radius of which is taken to the center of the material.

Example:
90 deg bend, 1 in. radius, 0.064 in. material.
Radius to center of material = bend radius (1 in.) + 1/2 material thickness = 1 + 0.032 = 1.032 in.

$$\text{Circumference of arc} = \frac{2\ \pi\ R \times \text{degree of bend}}{360\ \text{(total degrees in circle)}}$$

BEND ALLOWANCE COEFFICIENTS FOR ALUMINUM ALLOYS, BASED ON ONE-DEGREE ANGLE

RADII	METAL THICKNESS											
	0.016	0.020	0.022	0.025	0.032	0.040	0.050	0.063	0.080	0.090	0.125	0.180
1/32	0.00067	0.00070	0.00072	0.00074	0.00079	0.00086	0.00094	0.00104	0.00117	0.00125	0.00154	0.00200
1/16	0.00121	0.00125	0.00126	0.00129	0.00135	0.00140	0.00149	0.00159	0.00171	0.00180	0.00209	0.00255
3/32	0.00176	0.00179	0.00180	0.00183	0.00188	0.00195	0.00203	0.00213	0.00226	0.00234	0.00263	0.00309
1/8	0.00230	0.00234	0.00235	0.00238	0.00243	0.00249	0.00258	0.00268	0.00281	0.00289	0.00317	0.00364
5/32	0.00285	0.00288	0.00290	0.00292	0.00297	0.00304	0.00312	0.00322	0.00335	0.00343	0.00372	0.00418
3/16	0.00339	0.00342	0.00344	0.00347	0.00352	0.00358	0.00367	0.00377	0.00390	0.00398	0.00426	0.00473
7/32	0.00394	0.00397	0.00398	0.00401	0.00406	0.00412	0.00421	0.00431	0.00444	0.00452	0.00481	0.00527
1/4	0.00448	0.00451	0.00454	0.00456	0.00461	0.00467	0.00476	0.00486	0.00499	0.00507	0.00535	0.00582
9/32	0.00503	0.00506	0.00507	0.00510	0.00515	0.00521	0.00530	0.00540	0.00553	0.00561	0.00590	0.00636
5/16	0.00557	0.00560	0.00562	0.00564	0.00570	0.00576	0.00584	0.00595	0.00608	0.00616	0.00644	0.00691
11/32	0.00612	0.00615	0.00616	0.00619	0.00624	0.00630	0.00639	0.00649	0.00662	0.00670	0.00699	0.00745
3/8	0.00666	0.00669	0.00671	0.00673	0.00679	0.00685	0.00693	0.00704	0.00717	0.00725	0.00753	0.00800
13/32	0.00721	0.00724	0.00725	0.00728	0.00733	0.00739	0.00748	0.00758	0.00771	0.00779	0.00808	0.00854
7/16	0.00775	0.00778	0.00780	0.00782	0.00787	0.00794	0.00802	0.00812	0.00826	0.00834	0.00862	0.00908
15/32	0.00829	0.00833	0.00834	0.00837	0.00842	0.00848	0.00857	0.00867	0.00880	0.00888	0.00917	0.00963
1/2	0.00884	0.00887	0.00889	0.00891	0.00896	0.00903	0.00911	0.00921	0.00935	0.00943	0.00971	0.01017
17/32	0.00938	0.00942	0.00943	0.00946	0.00951	0.00957	0.00966	0.00976	0.00989	0.00997	0.01025	0.01072
9/16	0.00993	0.00996	0.00998	0.01000	0.01005	0.01012	0.01020	0.01030	0.01043	0.01051	0.01080	0.01126
19/32	0.01047	0.01051	0.01052	0.01055	0.01058	0.01065	0.01073	0.01083	0.01098	0.01105	0.01133	0.01179
5/8	0.01102	0.01105	0.01107	0.01109	0.01114	0.01121	0.01129	0.01139	0.01152	0.01160	0.01189	0.01235
21/32	0.01156	0.01160	0.01161	0.01164	0.01170	0.01175	0.01183	0.01193	0.01207	0.01214	0.01245	0.01289
11/16	0.01211	0.01214	0.01216	0.01218	0.01223	0.01230	0.01238	0.01248	0.01261	0.01269	0.01298	0.01344
23/32	0.01265	0.01268	0.01269	0.01273	0.01276	0.01283	0.01291	0.01301	0.01316	0.01322	0.01351	0.01397
3/4	0.01320	0.01323	0.01324	0.01327	0.01332	0.01338	0.01347	0.01357	0.01370	0.01378	0.01407	0.01453
25/32	0.01374	0.01378	0.01379	0.01381	0.01386	0.01392	0.01401	0.01411	0.01425	0.01432	0.01461	0.01507
13/16	0.01429	0.01432	0.01433	0.01436	0.01441	0.01447	0.01456	0.01466	0.01479	0.01487	0.01516	0.01562
27/32	0.01483	0.01486	0.01487	0.01490	0.01494	0.01501	0.01509	0.01519	0.01534	0.01540	0.01569	0.01615
7/8	0.01538	0.01541	0.01542	0.01545	0.01550	0.01556	0.01565	0.01575	0.01588	0.01596	0.01625	0.01671
29/32	0.01592	0.01595	0.01596	0.01599	0.01604	0.01610	0.01619	0.01629	0.01643	0.01650	0.01679	0.01727
15/16	0.01646	0.01650	0.01651	0.01654	0.01659	0.01665	0.01674	0.01684	0.01697	0.01705	0.01734	0.01780
31/32	0.01701	0.01704	0.01705	0.01708	0.01712	0.01718	0.01727	0.01737	0.01752	0.01758	0.01787	0.01833
1	0.01755	0.01759	0.01760	0.01763	0.01768	0.01774	0.01783	0.01793	0.01806	0.01814	0.01843	0.01888

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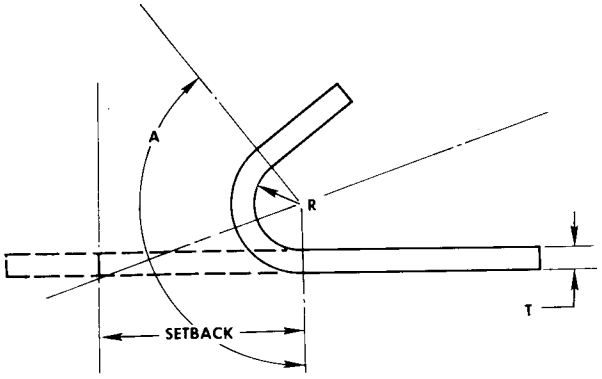
Figure 1-76. Bend Allowance Chart

SETBACK CHART

GENERAL



Example:
Angle of bend (A) = 140 deg
Material thickness (T) = 0.064 in.
Radius (R) = 0.125 in.
K—from table = 2.7475

SETBACK = K × (T + R) =
2.7475 × (0.064 + 0.125) =
2.7475 × 0.189 = 0.509
SETBACK = 0.509 IN.

ANGLE DEGREE	K	ANGLE DEGREE	K	ANGLE DEGREE	K	ANGLE DEGREE	K	ANGLE DEGREE	K	ANGLE DEGREE	K
1	0.00873	31	0.27732	61	0.58905	91	1.0176	121	1.7675	151	3.8667
2	0.01745	32	0.28674	62	0.60086	92	1.0355	122	1.8040	152	4.0108
3	0.02618	33	0.29621	63	0.61208	93	1.0538	123	1.8418	153	4.1653
4	0.03493	34	0.30573	64	0.62487	94	1.0724	124	1.8807	154	4.3315
5	0.04366	35	0.31530	65	0.63707	95	1.0913	125	1.9210	155	4.5107
6	0.05241	36	0.32492	66	0.64941	96	1.1106	126	1.9626	156	4.7046
7	0.06116	37	0.33459	67	0.66188	97	1.1303	127	2.0057	157	4.9151
8	0.06993	38	0.34433	68	0.67451	98	1.1504	128	2.0503	158	5.1455
9	0.07870	39	0.35412	69	0.68728	99	1.1708	129	2.0965	159	5.3995
10	0.08749	40	0.36397	70	0.70021	100	1.1917	130	2.1445	160	5.6713
11	0.09629	41	0.37388	71	0.71329	101	1.2131	131	2.1943	161	5.9758
12	0.10510	42	0.38386	72	0.72654	102	1.2349	132	2.2460	162	6.3137
13	0.11393	43	0.39391	73	0.73996	103	1.2572	133	2.2998	163	6.6911
14	0.12278	44	0.40403	74	0.75355	104	1.2799	134	2.3558	164	7.1154
15	0.13165	45	0.41421	75	0.76733	105	1.3032	135	2.4142	165	7.5957
16	0.14054	46	0.42447	76	0.78128	106	1.3270	136	2.4751	166	8.1443
17	0.14945	47	0.43481	77	0.79543	107	1.3514	137	2.5386	167	8.7769
18	0.15838	48	0.44523	78	0.80978	108	1.3764	138	2.6051	168	9.5144
19	0.16734	49	0.45573	79	0.82434	109	1.4019	139	2.6746	169	10.385
20	0.17633	50	0.46631	80	0.83910	110	1.4281	140	2.7475	170	11.430
21	0.18534	51	0.47697	81	0.85408	111	1.4550	141	2.8239	171	12.706
22	0.19438	52	0.48773	82	0.86929	112	1.4826	142	2.9042	172	14.301
23	0.20345	53	0.49858	83	0.88472	113	1.5108	143	2.9887	173	16.350
24	0.21256	54	0.50952	84	0.90040	114	1.5399	144	3.0777	174	19.081
25	0.22169	55	0.52057	85	0.91633	115	1.5697	145	3.1716	175	22.904
26	0.23087	56	0.53171	86	0.93251	116	1.6003	146	3.2708	176	26.636
27	0.24008	57	0.54295	87	0.94896	117	1.6318	147	3.3759	177	38.188
28	0.24933	58	0.55431	88	0.96569	118	1.6643	148	3.4874	178	57.290
29	0.25862	59	0.56577	89	0.98270	119	1.6977	149	3.6059	179	114.590
30	0.26795	60	0.57735	90	1.00000	120	1.7320	150	3.7320	180	INFINITE

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Figure 1-77. Set Back Chart

TAP DRILLS AND DECIMAL EQUIVALENTS

GENERAL

TAP DRILL SIZES

NC		NF	
Coarse Thread		Fine Thread	
SCREW SIZE	TAP DRILL	SCREW SIZE	TAP DRILL
7/16-14	U	7/16-20	25/64
1/2-13	27/64	1/2-20	29/64
9/16-12	31/64	9/16-18	33/64
5/8-11	17/32	5/8-18	37/64
11/16-11	19/32	11/16-11	5/8
3/4-10	21/32	3/4-16	11/16
13/16-10	23/32	7/8-14	13/16
7/8- 9	49/64	1 -14	15/16
15/16- 9	53/64	1-1/8-12	1-3/64
1 - 8	7/8	1-1/4-12	1-11/64
1-1/8- 7	63/64	1-1/2-12	1-27/64
1-1/4- 7	1-7/64		
1-1/2- 6	1-11/32		

MACHINE SCREW

SCREW SIZE	DRILLS FOR	
	TAP	CLEARANCE
2 -56	No. 50	No. 43
3 -48	No. 47	No. 40
4 -40	No. 43	No. 32
6 -32	No. 36	No. 28
8 -32	No. 29	No. 19
10 -24	No. 25	No. 10
10 -32	No. 21	No. 10
12 -24	No. 16	7/32
1/4 -20	No. 7	1/4
1/4 -28	No. 3	1/4
5/16 -18	F	5/16
5/16 -24	I	5/16
3/8 -16	5/16	3/8
3/8 -24	Q	3/8

PIPE

DIAMETER OF TAP	DRILLS FOR	
	TAP	CLEARANCE
1/8- 27	11/32	7/16
1/4- 18	7/16	7/16
3/8- 18	37/64	3/4
1/2- 14	23/32	7/8
3/4- 14	59/64	1-3/16
1 -11-1/2	1- 5/32	1-7/16
1-1/4-11-1/2	1- 1/2	1-3/4
1-1/2-11-1/2	1-47/64	2
2 -11-1/2	2- 7/32	2-1/2
2-1/2- 8	2- 5/8	3
3 - 8	3- 1/4	3-3/4
3-1/2- 8	3- 3/4	4-1/4
4 - 8	4- 1/4	4-3/4

DECIMAL EQUIVALENTS

NUMBER, LETTER, AND FRACTIONAL SIZE DRILLS

SIZE DRILL	DECIMAL	SIZE DRILL	DECIMAL	SIZE DRILL	DECIMAL
80	0.0135	1/8	0.1250	21/64	0.3281
79	0.0145	30	0.1285	Q	0.3320
1/64	0.0156	29	0.1360	R	0.3390
78	0.0160	28	0.1405	11/32	0.3437
77	0.0180	9/64	0.1406	S	0.3480
76	0.0200	27	0.1440	T	0.3580
75	0.0210	26	0.1470	23/64	0.3594
74	0.0225	25	0.1495	U	0.3680
73	0.0240	24	0.1520	3/8	0.3750
72	0.0250	23	0.1540	V	0.3770
71	0.0260	5/32	0.1562	W	0.3860
70	0.0280	22	0.1570	25/64	0.3906
69	0.0292	21	0.1590	X	0.3970
68	0.0310	20	0.1610	Y	0.4040
1/32	0.0313	19	0.1660	13/32	0.4062
67	0.0320	18	0.1695	Z	0.4130
66	0.0330	11/64	0.1719	27/64	0.4219
65	0.0350	17	0.1730	7/16	0.4375
64	0.0360	16	0.1770	29/64	0.4531
63	0.0370	15	0.1800	15/32	0.4687
62	0.0380	14	0.1820	31/64	0.4843
61	0.0390	13	0.1850	1/2	0.5000
60	0.0400	3/16	0.1875	33/64	0.5156
59	0.0410	12	0.1890	17/32	0.5312
58	0.0420	11	0.1910	35/64	0.5469
57	0.0430	10	0.1935	9/16	0.5625
56	0.0465	9	0.1960	37/64	0.5781
3/64	0.0469	8	0.1990	19/32	0.5937
55	0.0520	7	0.2010	39/64	0.6094
54	0.0550	13/64	0.2031	5/8	0.6250
53	0.0595	6	0.2040	41/64	0.6406
1/16	0.0625	5	0.2055	21/32	0.6562
52	0.0635	4	0.2090	43/64	0.6719
51	0.0670	3	0.2130	11/16	0.6875
50	0.0700	1/32	0.2187	45/64	0.7031
49	0.0730	2	0.2210	23/32	0.7187
48	0.0760	1	0.2280	47/64	0.7344
5/64	0.0781	A	0.2340	3/4	0.7500
47	0.0785	15/64	0.2344	49/64	0.7656
46	0.0810	B	0.2380	25/32	0.7812
45	0.0820	C	0.2420	51/64	0.7969
44	0.0860	D	0.2460	13/16	0.8125
43	0.0890	1/4	0.2500	53/64	0.8281
42	0.0935	F	0.2570	27/32	0.8437
3/32	0.0937	G	0.2610	55/64	0.8594
41	0.0960	17/64	0.2656	7/8	0.8750
40	0.0980	H	0.2660	57/64	0.8906
39	0.0995	I	0.2720	29/32	0.9062
38	0.1015	J	0.2770	59/64	0.9219
37	0.0140	K	0.2811	15/16	0.9375
36	0.0165	9/32	0.2812	61/64	0.9531
7/64	0.1093	L	0.2900	31/32	0.9687
35	0.1100	M	0.2950	63/64	0.9844
34	0.1110	19/64	0.2968	1	1.000
33	0.1130	N	0.3020		
32	0.1160	5/16	0.3125		
31	0.1200	O	0.3160		
		P	0.3230		

GEN-3-00-21

Figure 1-78. Tap Drill Sizes and Decimal Equivalents

TORQUE TABLE

GENERAL

The torque table is to be used as a guide in tightening nuts, bolts, and screws whenever specific torque values are not called out in maintenance procedures. Using the proper torque allows the structure to develop its designed strength and greatly reduces the possibility of failure due to fatigue. There are a few items you should be aware of when using this table:

1 To convert to foot-pounds, divide inch-pounds by 12.

2 Threads must be free from grease or oil. Lubrication changes the torque value and will result in overtorquing.

3 When castellated nuts are used, tighten them to the lower torque limit. Then continue tightening until cotter pin hole is aligned with slots in nut. Do not back off nut to align hole.

4 When it is necessary to tighten from the bolthead, use the high side of the torque range. If necessary, the maximum allowable tightening torque may be used.

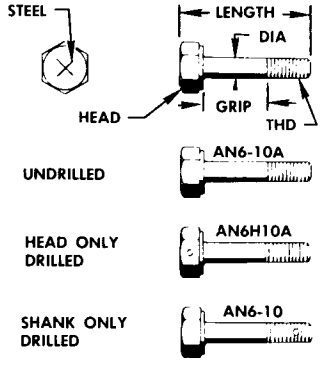
5 When corrosion-resistant steel bolts are used, they should be lubricated with an anti-seize compound. Corrosion-resistant steel bolts and nuts must be used together. Use shear nut torque values when tightening these bolts.

HEX NUTS

AN. NUMBER AND DESCRIPTION	STEEL CAD PLATED	STEEL CAD PLATED	CORR RESISTANT STEEL	STEEL CAD PLATED	CORR RESISTANT STEEL
AN363 REG HEIGHT 550°F MAX					
AN363C REG HEIGHT 800°F MAX					
AN364 THIN 250°F MAX					
AN365 REG HEIGHT 250°F MAX					
AN320 THIN 250°F MAX					
AN310 REG HEIGHT 250°F MAX					

BOLTS

BASIC NUMBER	DIAMETER	THREADS PER INCH
AN173	10*	32
174	1/4*	28
175	5/16	24
176	3/8	24
177	7/16	20
178	1/2	20
179	9/16	18
180	5/8	18
182	3/4	16
184	7/8	14
186	1	14



*BOLT SHANK SHOULD BE WITHOUT COTTER PIN HOLE WHEN USED WITH SELF-LOCKING NUTS.

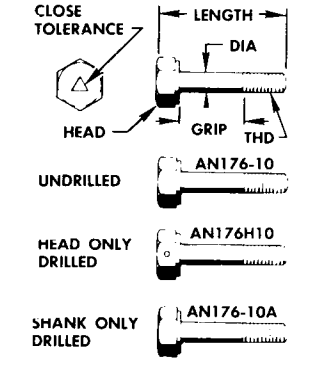
TORQUE VALUES IN INCH-POUNDS

STANDARD NUTS, BOLTS, AND SCREWS

WRENCH SIZE	BOLT, STUD, OR SCREW SIZE	TENSION-TYPE NUTS AN310 AND AN365	SHEAR-TYPE NUTS AN320 AND AN364
11/32	8-36	12 – 15	7 – 9
3/8	10-32	20 – 25	12 – 15
7/16	1/4-28	50 – 70	30 – 40
1/2	5/16-24	100 – 140	60 – 85
9/16	3/8-24	160 – 190	95 – 110
5/8	7/16-20	450 – 500	270 – 300
3/4	1/2-20	480 – 690	290 – 410
7/8	9/16-18	800 – 1000	480 – 60C
15/16	5/8-18	1100 – 1300	660 – 780
1-1/16	3/4-16	2300 – 2500	1300 – 1500
1-1/4	7/8-14	2500 – 3000	1500 – 1800
1-7/16	1-14	3700 – 5500	2200 – 3300

CLOSE-TOLERANCE BOLTS

BASIC NUMBER	DIAMETER	THREADS PER INCH
AN3	10*	32
4	1/4*	28
5	5/16	24
6	3/8	24
7	7/16	20
8	1/2	20
9	9/16	18
10	5/8	18
12	3/4	16
14	7/8	14
16	1	14



*BOLT SHANK SHOULD BE WITHOUT COTTER PIN HOLE WHEN USED WITH SELF-LOCKING NUTS.

GEN-3-00-22

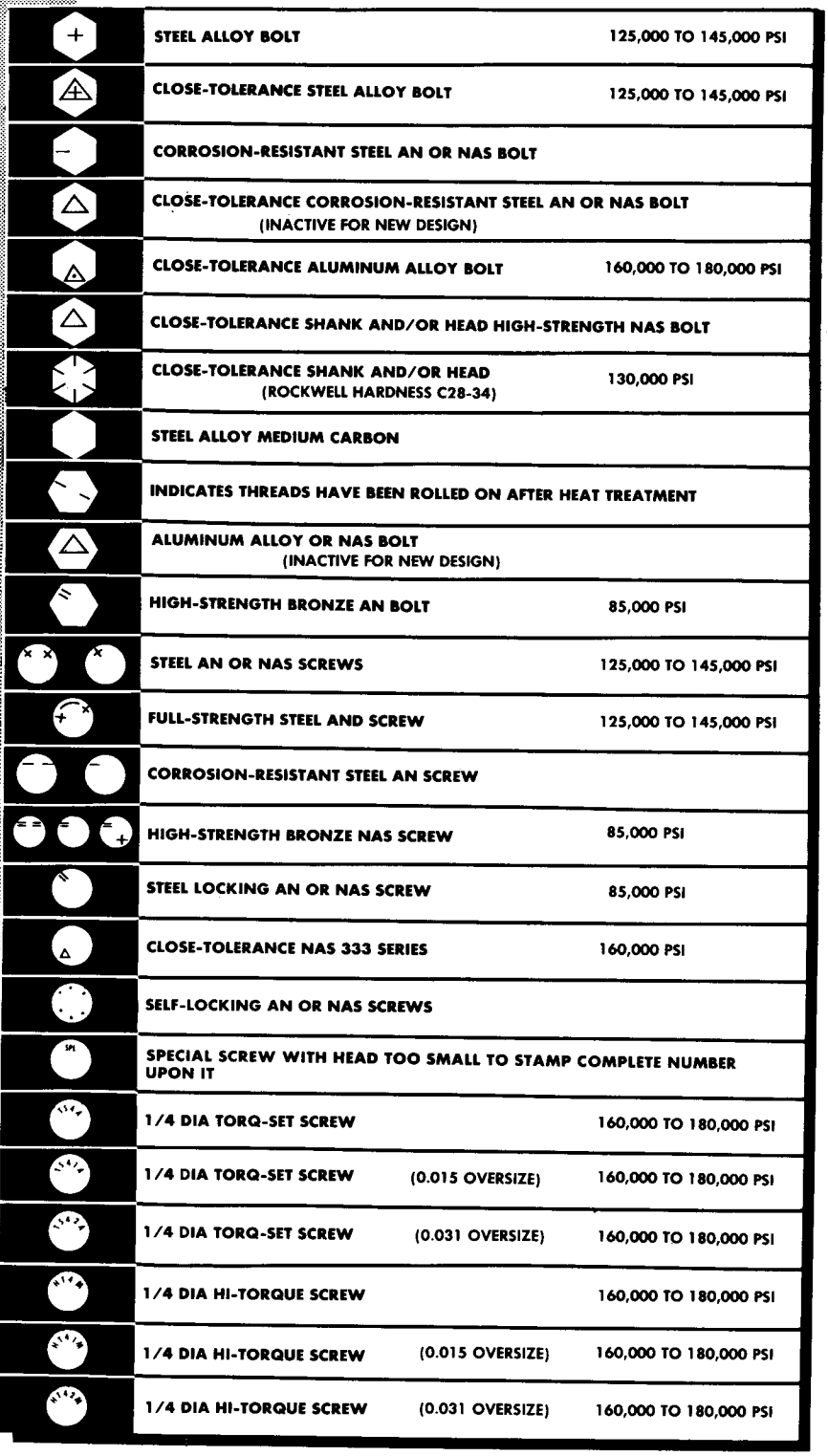
Figure 1-79. Torque Table

BOLT HEAD IDENTIFICATION

GENERAL

NOTE The head slots have been left off the screw and boltheads for clarity.

The figure "1" after the figure indicating the diameter of both Hi-Torque and Torq-Set bolts indicates that the screw is 0.015 inch oversize, and the figure "2" after the figure indicating the diameter indicates that the bolt is 0.031 inch oversize.



STEEL ALLOY BOLT		125,000 TO 145,000 PSI
CLOSE-TOLERANCE STEEL ALLOY BOLT		125,000 TO 145,000 PSI
CORROSION-RESISTANT STEEL AN OR NAS BOLT		
CLOSE-TOLERANCE CORROSION-RESISTANT STEEL AN OR NAS BOLT (INACTIVE FOR NEW DESIGN)		
CLOSE-TOLERANCE ALUMINUM ALLOY BOLT		160,000 TO 180,000 PSI
CLOSE-TOLERANCE SHANK AND/OR HEAD HIGH-STRENGTH NAS BOLT		
CLOSE-TOLERANCE SHANK AND/OR HEAD (ROCKWELL HARDNESS C28-34)		130,000 PSI
STEEL ALLOY MEDIUM CARBON		
INDICATES THREADS HAVE BEEN ROLLED ON AFTER HEAT TREATMENT		
ALUMINUM ALLOY OR NAS BOLT (INACTIVE FOR NEW DESIGN)		
HIGH-STRENGTH BRONZE AN BOLT		85,000 PSI
STEEL AN OR NAS SCREWS		125,000 TO 145,000 PSI
FULL-STRENGTH STEEL AND SCREW		125,000 TO 145,000 PSI
CORROSION-RESISTANT STEEL AN SCREW		
HIGH-STRENGTH BRONZE NAS SCREW		85,000 PSI
STEEL LOCKING AN OR NAS SCREW		85,000 PSI
CLOSE-TOLERANCE NAS 333 SERIES		160,000 PSI
SELF-LOCKING AN OR NAS SCREWS		
SPECIAL SCREW WITH HEAD TOO SMALL TO STAMP COMPLETE NUMBER UPON IT		
1/4 DIA TORQ-SET SCREW		160,000 TO 180,000 PSI
1/4 DIA TORQ-SET SCREW	(0.015 OVERSIZE)	160,000 TO 180,000 PSI
1/4 DIA TORQ-SET SCREW	(0.031 OVERSIZE)	160,000 TO 180,000 PSI
1/4 DIA HI-TORQUE SCREW		160,000 TO 180,000 PSI
1/4 DIA HI-TORQUE SCREW	(0.015 OVERSIZE)	160,000 TO 180,000 PSI
1/4 DIA HI-TORQUE SCREW	(0.031 OVERSIZE)	160,000 TO 180,000 PSI

GEN-3-00-92

Figure 1-79A. Bolthead Identification

MINIMUM BEND RADII TABLES

GENERAL

THICKNESS	ALUMINUM ALLOYS			
	2024-0	2024-T4	7075-0	7075-T6
0.020	1/32	1/16	1/16	1/8
0.025	1/16	1/16	1/16	1/8
0.032	1/16	3/32	1/16	1/8
0.040	1/16	3/32	1/16	3/16
0.050	1/16	1/8	3/32	1/4
0.063	3/32	5/32	3/32	5/16
0.071	1/8	7/32	5/32	3/8
0.080	1/8	1/4	3/16	7/16
0.090	1/8	9/32	1/4	1/2
0.100	5/32	11/32	5/16	9/16
0.125	3/16	7/16	11/32	3/4

THICKNESS	STEEL				TITANIUM‡	
	4130*		CORROSION-RESISTANT		PURE	ALLOY
	NORMALIZED	ANNEALED	ANNEALED	1/2 HARD	AMS—4900	AMS—4908
0.020	1/16	1/16	1/32	1/16	1/16	
0.025	1/16	1/16	1/32	1/16	3/32	3/32
0.032	1/8	1/16	1/32	1/16	1/8	1/8
0.036	1/8	3/32	1/16	1/8	1/8	1/8
0.040	1/8	3/32	1/16	1/8	1/8	1/8
0.050	1/8	3/32	1/16	1/8	5/32	5/32
0.063	1/8	3/32	1/16	1/8	3/16	3/16
0.080	5/32	3/32	3/32	3/16	9/32	9/32
0.090	3/16	3/32	3/32	3/16	5/16	5/16
0.112	1/4	1/8	1/8	1/4	13/32	13/32
0.125	1/4	1/8	1/8	1/4	7/16	7/16
0.190	3/8	3/16	3/16			
0.250	1/2	1/4	1/4			
OVER 0.250	2-1/2 T†	1-1/2 T†	2T†		3-1/2 T†	3-1/2 T†

*For normalized 4130 steel, bend radii equal to thickness may be obtained by heat forming. Material must be reheat-treated to normalized condition after forming.

†T = Material thickness

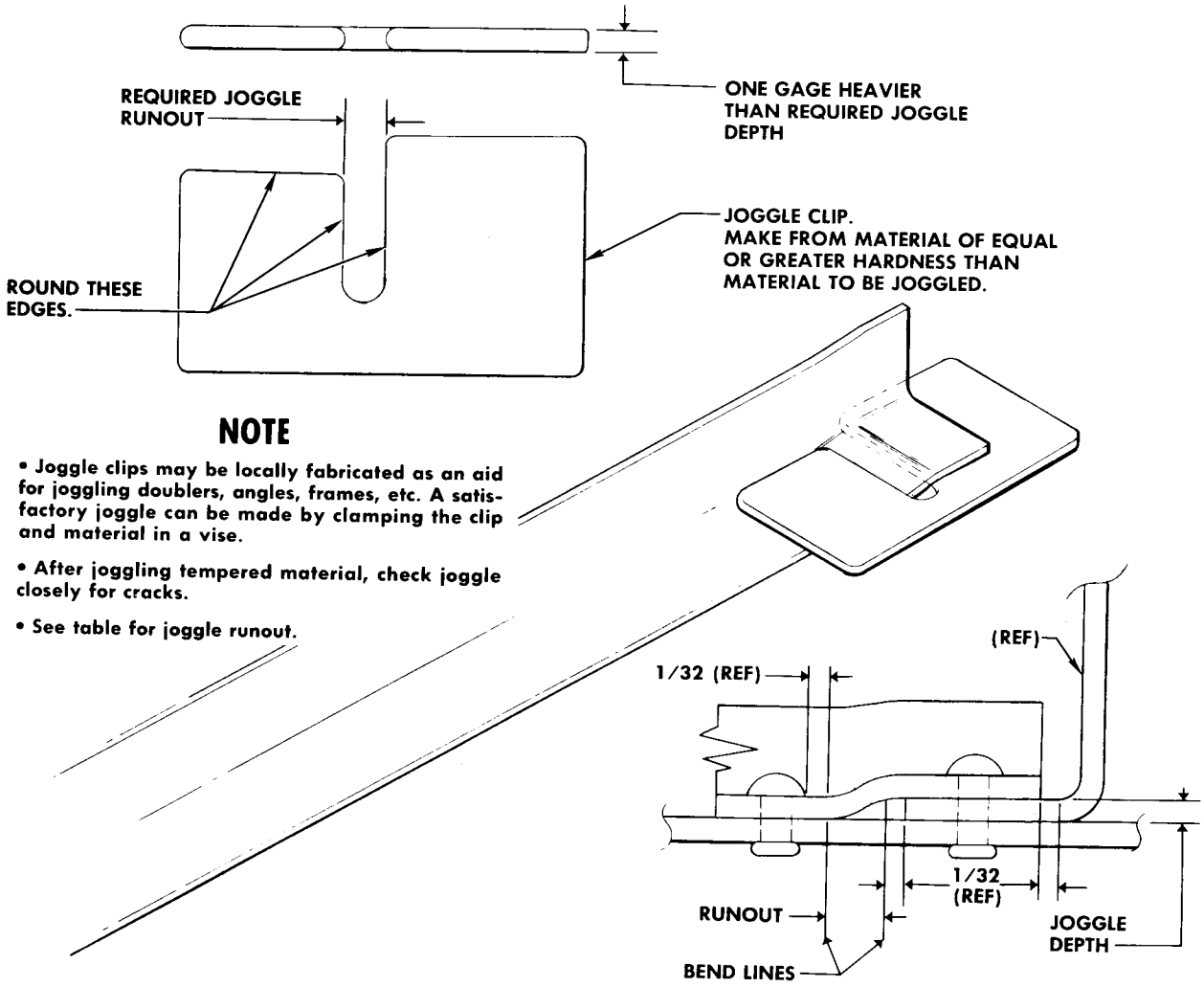
‡It is recommended that larger bend radii be used where possible. If forming difficulty is encountered, heat forming may be used. (Refer to index for titanium.)

GEN-3-00-23

Figure 1-80. Minimum Bend Radii

JOGGLES

GENERAL



NOTE

- Joggle clips may be locally fabricated as an aid for joggling doublers, angles, frames, etc. A satisfactory joggle can be made by clamping the clip and material in a vise.
- After joggling tempered material, check joggle closely for cracks.
- See table for joggle runout.

MATERIAL	RUNOUT	MATERIAL	RUNOUT
ALUMINUM		STEEL	
2024-O	3 X JOGGLE DEPTH	1010	4 X JOGGLE DEPTH
2024-T4	4 X JOGGLE DEPTH	1020	4 X JOGGLE DEPTH
5052-O	3 X JOGGLE DEPTH	4130 (Normalized)	4 X JOGGLE DEPTH
6061-O	3 X JOGGLE DEPTH		
6061-T4	4 X JOGGLE DEPTH		
6061-T6	4 X JOGGLE DEPTH	CRES	
7075-O	4 X JOGGLE DEPTH	301 (Annealed)	4 X JOGGLE DEPTH
7075-W	4 X JOGGLE DEPTH	302 (Annealed)	4 X JOGGLE DEPTH
7075-T6	8 X JOGGLE DEPTH	301 (1/2 Hard)	4 X JOGGLE DEPTH

GEN-3-00-93

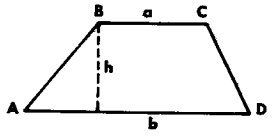
Figure 1-80A. Joggles

SHOP MATHEMATICS

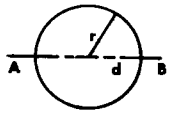
GENERAL

PLANE FIGURES

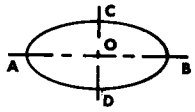
TRAPEZOID
(a parallel to b)
AREA $= \frac{(a+b)}{2} h$
(h is perpendicular to a and b)



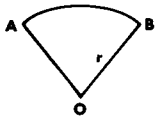
CIRCLE
CIRCUMFERENCE $= \pi d = 2\pi r$ $\left(\pi = \frac{\text{circumference}}{\text{diameter}}\right)$
AREA $= \pi r^2 = \frac{\pi}{4} d^2 = 0.7854 d^2$
$\pi = 3.1416$



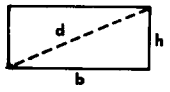
ELLIPSE
AREA $= \pi (OA) \times (OC)$
$= \frac{\pi}{4} (AB) \times (CD)$



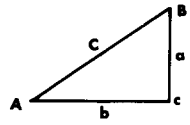
SECTOR
AREA $= \frac{r(\text{arc AB})}{2}$
$= \frac{\pi r^2 (\text{angle AOB})}{360 \text{ degrees}}$



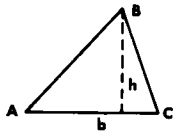
RECTANGLE
AREA $= bh$
DIAGONAL $d = \sqrt{b^2 + h^2}$



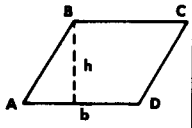
RIGHT TRIANGLE
AREA $= \frac{ba}{2}$
HYPOTENUSE $= c = \sqrt{a^2 + b^2}$
A + B + C = 180 degrees



ANY TRIANGLE
AREA $= \frac{bh}{2}$
(h is perpendicular to b)
A + B + C = 180 degrees

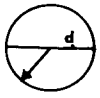


PARALLELOGRAM
(opposite sides parallel)
AREA = bh
(h is perpendicular to b)
A + B + C + D = 360 degrees

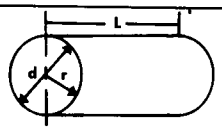


SOLID FIGURES

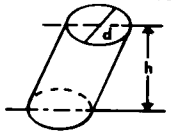
SPHERE
VOLUME $= \frac{4}{3} \pi r^3 = \pi \frac{d^3}{6}$
AREA $= 4\pi r^2 = \pi d^2$



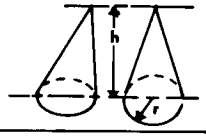
CYLINDER
VOLUME $= \pi r^2 L = 0.7854 d^2 L$
AREA OF CYLINDRICAL SURFACE $= \pi DL$



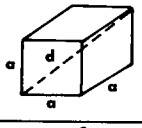
CYLINDER
VOLUME = (area base) h
AREA OF CYLINDRICAL SURFACE = (perimeter base) h



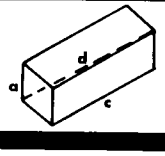
CONE
VOLUME (OF EITHER) $= \frac{1}{3} \pi r^2 h$
AREA (OF RIGHT CONE) $= \pi rs$



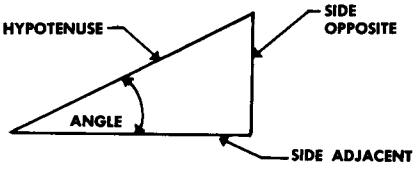
CUBE
VOLUME $= a^3$
DIAGONAL $= d = a\sqrt{3}$
TOTAL AREA $= 6a^2$



RECTANGULAR PRISM
VOLUME = (area of end) C
$= abc$
DIAGONAL $= d = \sqrt{a^2 + b^2 + c^2}$
TOTAL AREA $= 2(ab + bc + ca)$



FUNCTIONS OF ANGLES



SINE $= \frac{\text{Side Opposite}}{\text{Hypotenuse}}$
COSINE $= \frac{\text{Side Adjacent}}{\text{Hypotenuse}}$
TANGENT $= \frac{\text{Side Opposite}}{\text{Side Adjacent}}$
COTANGENT $= \frac{\text{Side Adjacent}}{\text{Side Opposite}}$
SECANT $= \frac{\text{Hypotenuse}}{\text{Side Adjacent}}$
COSECANT $= \frac{\text{Hypotenuse}}{\text{Side Opposite}}$

CONVERSION OF FRACTION TO DECIMAL

To convert fraction to decimal . . . .

Divide numerator (top) by denominator (bottom)

Example: ⅝

```
0.625
8) 5.000
   48
   20
   16
   40
   40
    0
```

⅝ = 0.625

GEN-3-00-24

Figure 1-81. Shop Mathematics

STOP DRILLING

GENERAL

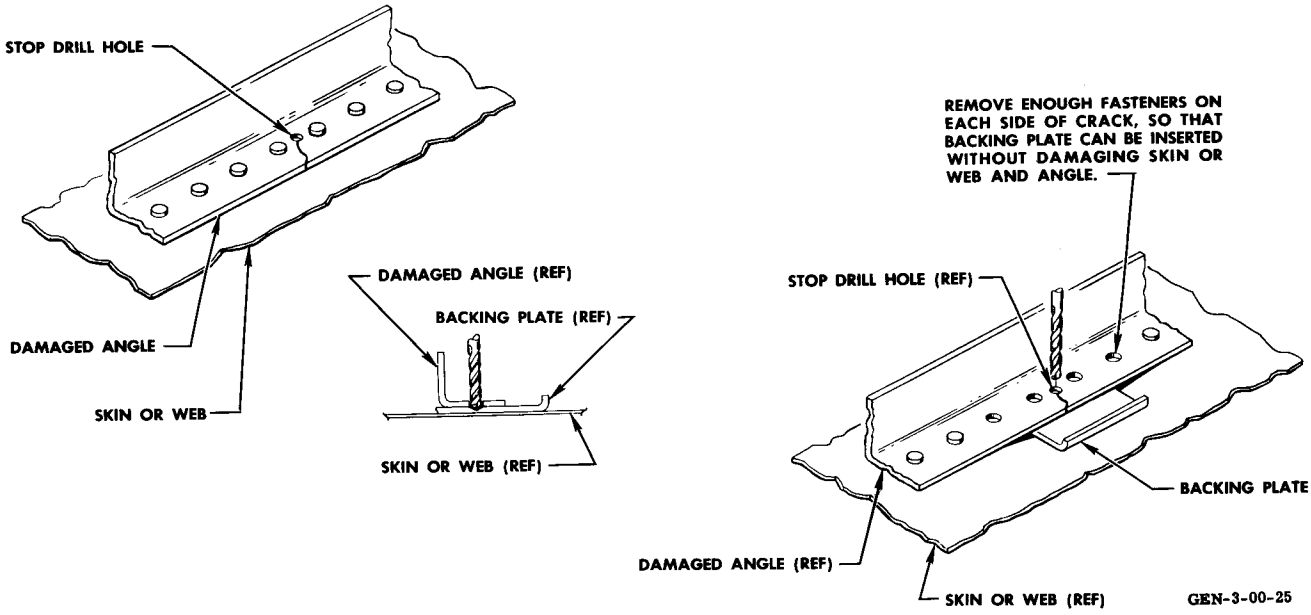


Figure 1-82. Stop-drilling

Section II

WING

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DESCRIPTION OF WING.

The wing used on the F-100D and F-100F Airplanes is a full cantilever, semimonocoque-type structure with a 45-degree sweep-back as measured at the 25-percent chord line. The wing has 400.2 square feet of surface area and the maximum skin-to-skin thickness at the root is 11.92 inches. The wing may be broken down into six parts: the leading edge and slat, main box section, trailing edge, flap aileron, and tip assemblies. The primary load-carrying structure is the box section, or the part of the wing between the front and rear spar. Full span slats, divided into five sections, extend along each outer panel forward of the front spar. These slats are attached to the leading edge and are considered as part of the leading edge. Partial span flaps and two segment ailerons are located along the trailing edge of each outer panel. The trailing edge is the part of the wing aft of the rear spar. The tip is considered to be the wing outboard of the sealed box section area. The box section is a sealed structure and is used as an integral fuel tank. Provisions are included for carrying a 275-gallon external fuel tank at wing station 108.357 and miscellaneous stores at wing stations 55, 106, and 155.

LEADING EDGE AND WING SLATS.

The leading edge structure contains the wing slats, slat tracks, and slat track support fittings along with related fairing structure. The leading edge assembly is attached by continuous hinges to the upper and lower front spar caps. The wing slats are constructed of 2024-T4 clad sheet with an extruded 2024-T42 trailing edge. The slat on each side of the wing is divided into five sections. Each section of the slat is supported by two tracks. These tracks are 4140 steel, heat-treated to 180,000-200,000 psi, with the exception of the track at wing station 300 on the left-hand wing, which is nonmagnetic beryllium-copper. The tracks are fixed to the slats and roll on rollers fixed in the wing leading edge. On early F-100D Airplanes, all slat sections on one side of the wing are flexibly linked together so that they will all move in unison without binding. On later airplanes, slat sections one, two, and three are flexibly linked together as a unit and sections four and five are linked together as a unit. Each of these units is able to

move independently of the other. The possibility of binding is eliminated by the use of side load rollers between slat sections three and four. The upper skins on the slats are 0.051-inch gage, and the lower skins are 0.040-inch gage material. The track support ribs are 0.064 gage and the air-load ribs are 0.040 gage. The track-to-rib attaching fittings are 2014-T6 aluminum alloy extrusions or die forgings.

BOX SECTION.

The center section structure which carries the wing bending loads across the fuselage, is a rectangular box. The upper and lower surfaces of this box consist of 7075-T6 machined plates fastened together with spacers to form a double skin structure for added strength. The center section spars are fabricated from 7075-T6 into spar caps and webs. The wing center section contains a bladder-type fuel cell and is fume-sealed for safety.

The outer panel main box structure consists primarily of upper and lower skins and front and rear spars. The spars are machined of 7075-T6 and 2014-T6 material. The inboard rear spar is integral with the trunnion fitting and distributes the landing gear loads into the wing. Intermediate, corrugated spars, along with widely spaced ribs, support the skins in the inboard wing section, while further outboard closely spaced ribs support the skin. The outer panel skins are taper-milled from 7075-T6 bare plate. Ridges, pads, ledges, islands, and integral stiffeners are milled into the skins for added support and strength. On airplanes not changed by T.O. 1F-100-1035, the skin thickness on both the upper and lower skin diminishes gradually along the span from a maximum of 1.400 inches at the root to a minimum of 0.070 inch at the tip. On airplanes changed by T.O. 1F-100-1035, a new heavier lower skin (2629-140144-1 and -2) is installed. The skin thickness for the new lower skin diminishes gradually along the span from a maximum of 1.587 inches at the root to a minimum of 0.070 inch at the tip. The new lower skin is machined with wider spar attachment lands and a "beefed-up" area at the root. The new skin is secured to the wing center-section box with Taper-Lok fasteners installed in specially reamed holes instead of the existing type bolts. The wing root rib provides for the attachment of the wing to the fuselage and is therefore considered a major structural member. This root-rib is forged from 2014-T6 material. Additional ribs are used in the box section to back up the aileron hinge fittings and other points of load concentration. The outer panel main box section is sealed to carry fuel in three compartments. Two cast aluminum ribs divide the compartments and a forging closes off the outboard end of the main box. The basic sealing of the main box section has been accomplished by the North American "integral attachment groove injection sealing method." (Refer to index.) Rubber-sealed doors provided for access to equipment also provide limited access for structural repair.

TRAILING EDGE.

The trailing edge is considered to be the structure aft of the rear spar. It is made up of ribs and skin along with a trailing edge extrusion of 6061-T6 aluminum extrusion. The ribs are fabricated mostly from 2024-T4 material. The skins are 7075-T6 material. The trailing edge houses the aileron and flap hinge fittings and includes the landing gear strut well.

FLAPS.

The flaps are attached to the wing by hinge fittings which are housed in the trailing edge structure of the wing. The wing flaps are all-metal structures with the primary load-carrying portion consisting of a one-cell box, between the flap front and rear spars. The upper and lower skins of the main box are machined from 2024-T4 aluminum alloy plate. The front spar is 2024-T4 aluminum alloy clad sheet with a formed lower cap angle and an extended 2024-T42 aluminum alloy upper cap angle. The rear spar is formed of 2024-T4 aluminum alloy clad sheet. All ribs are 2024-T4 clad sheet, except for two ribs in the leading edge at stations 84.920 and 93.920 which are corrosion-resistant steel sheet. The leading edge section, forward of the front spar, has an upper skin of corrosion-resistant steel sheet. The lower leading edge skin is 2024-T4 aluminum alloy clad sheet. The flap trailing edge is of aluminum alloy honeycomb sandwich construction. The inboard flap hinge is a 2014-T6 aluminum alloy die forging. The outboard hinge is 4340 steel die forging attached to a 7075-T6 aluminum alloy forging which also supports the landing gear side brace.

AILERONS.

There are two aileron segments on each wing. Each segment is supported by fittings extending aft of the wing rear spar to the aileron hinge point. The inboard aileron segment is supported at two hinge points; the outboard aileron segment is supported at three hinge points. The inboard aileron segment has partial static balance, and the outboard segment has complete static balance. The balance weights are located forward of the hinge line and are supported by the aileron leading edge structure. The aerodynamic seal between the ailerons and wing is achieved by utilizing a thin corrosion-resistant steel sheet attached to the wing trailing edge. This sheet overlaps and contacts the aileron leading edge to form the aerodynamic seal. The aileron skins are milled from 2024-T4 aluminum plate. The front spar cap and rib flanges are machined into the skins along with stiffener lands. The rib webs are 2024-T4 material and are riveted internally to flanges milled into the skins. Front and rear spar webs are 2024-T4 clad sheet channel sections. The trailing edge is of aluminum honeycomb sandwich construction. The hinge fittings on the ailerons are 2014-T6 extrusions and die forgings.

TIP.

The wing tip panel is the structure outboard of the sealed section of the wing. Conventional rib-stringer design is used in its construction. The ribs are 2024-T4 clad material and the stringers are 7075-T6 material. The milled skins on the tip are 7075-T6 material and the sheet skins are 2024-T4 material. The wing tip trailing edge is of honeycomb sandwich construction. The tips house the clearance lights and the left tip houses the magnetic compass. Because of the compass, the left wing tip fasteners are made of nonmagnetic material.

PRIMARY AND SECONDARY STRUCTURE.

Most of the structure in the wing is considered primary structure. A structure is considered primary if the failure of the structure would result in loss of the airplane, cause injury, death, or serious hazard to any crew member, or make the airplane unable to meet its basic design standards. Primary structure includes the box section, leading edge and slat (except the small leading edge fairing next to the fuselage), the trailing edge structure, most of the wing tip structure, and the ailerons (except the aileron trailing edge). The flap, aileron and tip trailing edge, wing tip fairing, and leading edge fairing next to the fuselage, are considered secondary structure. Repairs made to the structure in a primary area should make the structure as strong or stronger than the original.

ONE-TIME FLIGHT.

The one-time flight diagram (figure 2-1) indicates areas of the wing which may be damaged but will permit restricted ferry flight to a repair facility. These areas are mostly in secondary structure and no repair is required before making a ferry flight under restricted conditions. Ferry flight, with damage in indicated areas, is to be made at airspeeds not to exceed 300 knots and load conditions are not to exceed +3 G or −1 G.

EXTERNAL (NONFLUSH) REPAIRS.

Nonflush repairs may be used as temporary and one-time flight repairs on the wing, as outlined under "Temporary Repair" in Section I. A box section (sealed area) patch diagram is shown in figure 2-2. The fastener requirements and fastener layout patterns are shown in figure 2-10.

WING NEGLIGIBLE DAMAGE.

Damage consisting of holes, dents, cracks, and scratches within the limits shown on the illustrations for each wing component, is considered negligible. Negligible damage does not require structural repairs that will reinforce the structural component; however, in most cases some corrective measures must be taken. These corrective steps are to prevent the spreading of the damage and/or corrosion. (Refer to "Negligible Damage" in Section I.) Every crack must be evaluated by the structural repair officer, before repair work is started, to determine the cause of the damage. Every stop-drilled crack should be checked after each flight to determine if the crack is getting bigger. If a stop-drilled crack restarts, it must be considered major damage. Repair must be made using methods prescribed for repair of major damage. Stop-drilled cracks must be dye-penetrant checked to determine definitely that the crack does not extend beyond the drilled holes.

Dents within the limits shown are considered negligible. An attempt may be made to bring the larger dents back to the mold line dimensions if the area is accessible, and provided there are no sharp dents or wrinkles existing. If a dent has a radius less than 0.125 inch, it should be considered a "sharp-corner" dent. Any "sharp-corner" dent is to be considered major damage and must be repaired.

Scratches may occur either in a spanwise direction or chordwise direction. As a rule a spanwise scratch is not as critical as a chordwise scratch; therefore, a scratch 6 inches long in a spanwise direction might be negligible damage, while the same scratch in a chordwise direction might be major damage. Any scratch which exceeds the limits shown must be evaluated for the necessity of further repair.

Holes within the limits shown may be left without structural-type repair; however all holes must be cleaned out to remove sharp corners, and all holes in external surfaces must be plugged with a nonstructural-type repair.

WING SKIN CORROSION REMOVAL LIMITS.

See figure 2-24B.

LEADING EDGE AND SLAT NEGLIGIBLE DAMAGE.

See figure 2-21.

BOX SECTION NEGLIGIBLE DAMAGE.

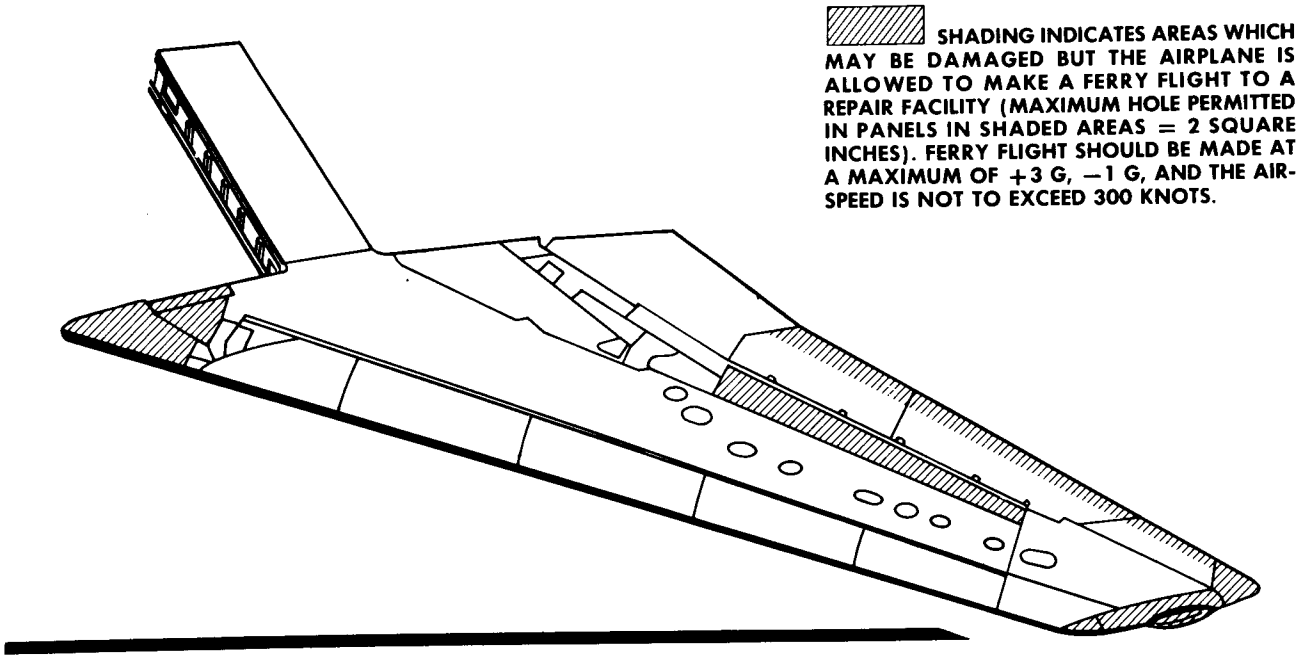
See figure 2-25.

Only very slight damage to the box section is considered negligible, because the box section is the primary load-carrying structure. No holes or cracks in the upper or lower skin on the fuel sealed area are considered negligible. Holes or cracks within the limits shown are allowed on the center section.

No scratches on the center section deeper than 0.003 inch may run over the edge of the skin at either outer edges, or internal holes, or at any row of rivets between stations 276 and 301. Any damage to the center section front or rear spar is considered major damage and must be repaired.

ONE-TIME FLIGHT

WING



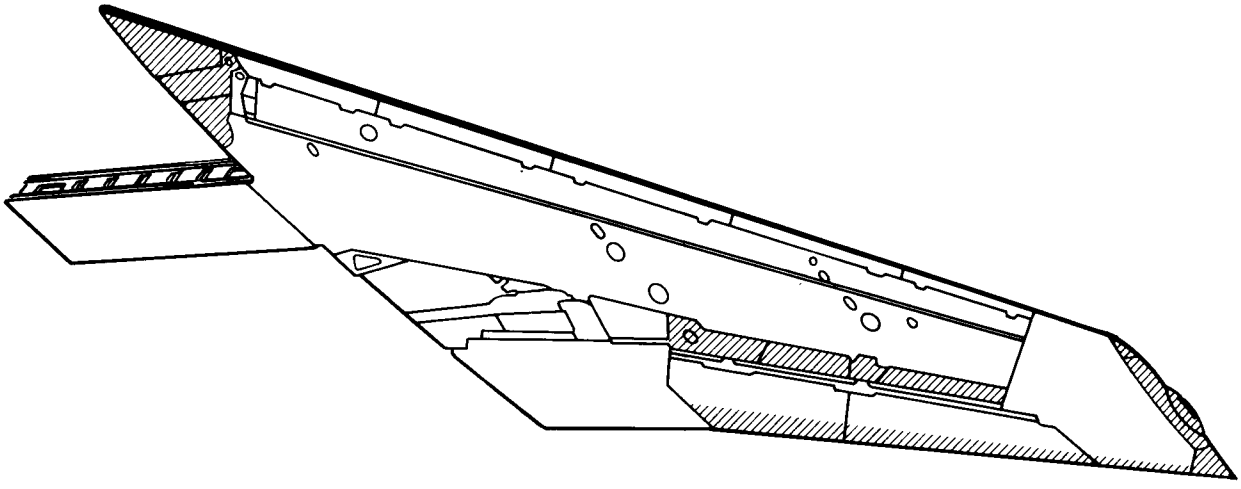


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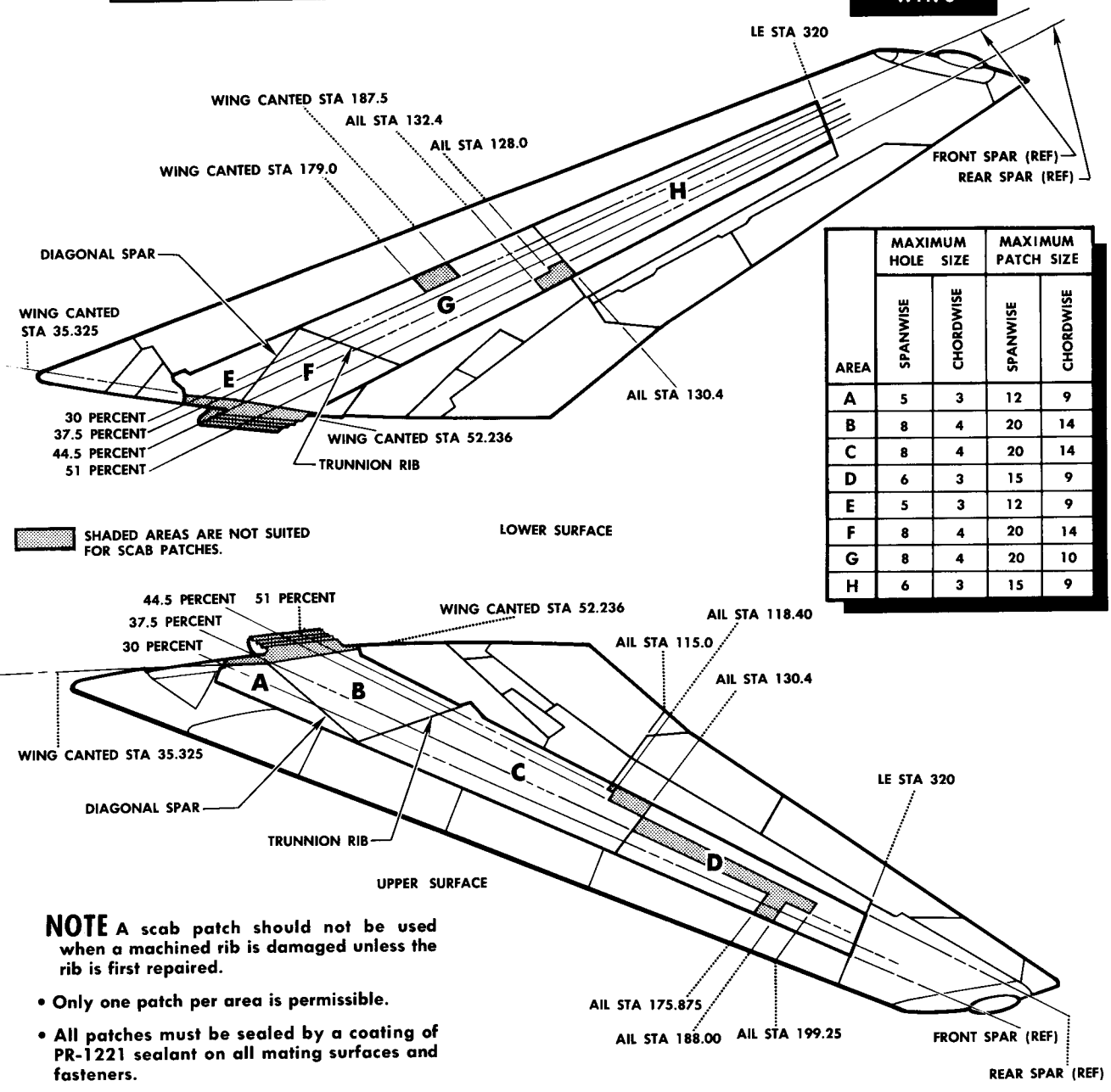
Figure 2-1. One-time Flight

EXTERNAL SKIN PATCH AREA TEMPORARY

WING



AREA	MAXIMUM HOLE SIZE		MAXIMUM PATCH SIZE	
	SPANWISE	CHORDWISE	SPANWISE	CHORDWISE
A	5	3	12	9
B	8	4	20	14
C	8	4	20	14
D	6	3	15	9
E	5	3	12	9
F	8	4	20	14
G	8	4	20	10
H	6	3	15	9

NOTE A scab patch should not be used when a machined rib is damaged unless the rib is first repaired.

- Only one patch per area is permissible.
- All patches must be sealed by a coating of PR-1221 sealant on all mating surfaces and fasteners.
- The patch thickness should be the same as the maximum skin thickness of the damaged skin. 7075-T6 plate is used except in areas "B" and "F." In these areas, 4130 steel heat-treated to 125,000 psi is used.
- All edges of the patch must be chamfered 30 degrees.
- The maximum thickness for a one-time flight patch is 0.375 inch. The maximum thickness for a temporary repair patch is 0.187 inch.
- Jo-bolts should be used for repair. (Refer to "Fastener Layout Instructions" for correct spacing.
- Refer to "Aerodynamic Smoothness" before using this repair on outer skins.

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Figure 2-2. External Skin Patch Areas

TRAILING EDGE NEGLIGIBLE DAMAGE.

See figure 2-45.

WING FLAP NEGLIGIBLE DAMAGE.

See figure 2-48.

AILERON NEGLIGIBLE DAMAGE.

See figure 2-53.

WING TIP NEGLIGIBLE DAMAGE.

See figure 2-59.

REPAIRING INTEGRAL FUEL TANK.

Personnel making repairs on the integral fuel tank on the wings should be familiar with the basic construction of the wing and the sealing method used. Successful wing repairs will depend on good workmanship. Bear in mind that every additional hole drilled into the tank area is a potential leak. Repairs that channel fuel from one section to another make it difficult, if not impossible, to detect the source of leakage. Scratches or nicks in any sealing surface may result in a poor seal. Great care has been taken in design and construction of the wing to avoid as many potential leaks as possible. When making a repair, some forethought should be given to the method of repair in order to retain this basic design. The following procedures and precautions are given to aid in making wing repairs.

- Most major repairs to the fuel section of the wing will require the removal of the upper surface. It is not ordinarily practical at present to attempt to replace a damaged skin with a new skin; however, it is possible to repair and replace the original skin. In some cases, the repair work may be done through the existing access holes in the spars and skins.

- Cleanness cannot be too strongly emphasized. Chips and foreign matter can cause leaks by holding sealing elements apart. Each bay of the wing must be inspected for loose raw stock and debris before close-out. Oil, grease, or dirt in the sealing grooves or on the surfaces that are to be sealed will prevent adhesion of sealant compounds. It is required that all surfaces to be sealed be wiped clean with solvent before the parts are fitted together.

- Close metal-to-metal fit is necessary for sealing purposes. The maximum gap between surfaces for injection sealing should not exceed 0.004 inch. Larger gaps will allow the sealing compound to squeeze out the side, causing a poor seal. Care must be taken when fitting repair members to contoured sections. These parts must fit together closely in order to ensure good sealing and strength properties.

- Before fasteners are installed in the fuel sections of the wing, they should be cleaned and degreased. Vapor-degreasing with trichlorethylene is recommended if available. Solvent-wiping is satisfactory. Close-tolerance fasteners and holes are required for sealing. A tolerance of +0.004 to −0.001 inch is used on all fastener holes, except those which are groove-sealed. The tolerance for groove-sealed holes is +0.002, −0.001 inch. Fasteners are chosen in the following descending order of preference: (1) groove-sealed bolts or screws (2) bolts with Stat-O-Seals (3) individually sealed screws (4) bolts or screws used with sealing dome nuts having PR-1221 sealant compound around and over the attachment lugs (5) Huck lock bolts installed with PR-1221 and (6) blind fasteners. If Stat-O-Seals are used, they should be in replaceable locations.

- Fuel has corrosive properties; therefore, it is necessary to take corrosion-preventive measures on the interior of the integral fuel tanks. All aluminum or aluminum alloy structures in the tank area are chemical film-treated and then given a protective covering of interior coating conforming to Specification MIL-S-4383. Any tank area in which bare metal has been exposed or new material has been installed should be chemically treated (Specification MIL-S-5002). Refer to index. It must be either sprayed or brush-coated with interior coating (Specification MIL-S-4383).

WING OUTER PANEL UPPER SKIN.

REMOVAL OF WING OUTER PANEL UPPER SKIN.

The outer wing panel box section is closed out and sealed to be used as an integral fuel tank. Access to most of the interior of this area is attained only by removal of the upper skin panel.

Before the upper skin can be removed, the following must be accomplished as directed in the maintenance handbooks:

- Remove wing outer panel from center section at the splice.
- Remove wing tip panel at leading edge station 320 splice.
- Remove aileron and aileron actuating cylinder.
- Remove wing flap and flap actuating cylinder.
- Remove leading edge and slat assemblies.
- Remove landing gear strut, actuator and doors.
- Remove all access doors on front spar, upper skin, and rear spar.

The upper skin may be removed without the use of a wing jig, provided care is taken to properly support the wing before starting removal. However, if a suitable wing jig is available it should by all means be used. The wing must be supported in such a way that it will not be subjected to any twisting or bending loads. It is suggested that supports with adjustable legs be used, so as to be able to level the wing regardless of floor irregularities. When the upper skin is removed, it is very important to keep the attaching screws in order so they can be reinstalled in the same holes they were removed from. The screw sizes vary in different locations along the wing and some screws may be oversize. If screws are not installed in the proper locations, nuts or nut plates may be damaged or a good seal may be impossible to obtain. As an aid in keeping the screws in order, the wing screw holes may be numbered in sequence and a board drilled and numbered to match the wing. (See figure 2-3.) As a screw is removed from the wing, place the screw in the corresponding numbered hole in the board. In this way, the screws will be kept in order, kept clean, and will be handy when needed to reinstall the skin. To remove the upper skin panel, proceed as follows: (Parts and procedures are identified in figure 2-3.)

1. Remove all fasteners through 223-14792 skin panel at lower inboard nose section, front spar, root rib, lower wing outer panel skin, fixed leading edge skin, and rib at leading edge canted station 89.168. Remove skin panel.

Note

Refer to index for recommended procedure for removing Deutsch rivets.

2. Remove all fasteners through 223-14017 skin panel at upper inboard nose section, front spar, root rib, upper wing outer panel skin, rib at leading edge canted station 89.168, and leading edge intercostal. Remove skin panel assembly.

3. Remove all fasteners through upper skin, from spar cap, rear spar cap, root rib, leading edge station 96, diagonal spar, wing canted station 107.013, wing canted station 116.73, all intermediate ribs, all intermediate spars, and tip rib at leading edge station 320.

4. Remove ten 192-12693-5 bolts and one 192-12693-3 bolt through upper skin and landing gear trunnion.

5. Remove all fasteners through trailing edge assembly and wing outer panel upper skin.

6. Remove four screws located inboard of root rib near front spar attachment of center section.

7. Make sure all fasteners are removed and then remove skin panel.

INSTALLATION OF WING OUTER PANEL UPPER SKIN.

After repairs are complete and before installing the upper skin panel, the wing fuel area must be cleaned and prepared for sealing. (See "Preparation of Wing for Fuel Tank Sealing.") With this preparation accomplished, proceed as follows: (Parts and procedures are identified in figure 2-4.)

1. Place skin panel on box section. Carefully guide skin panel under edge of trailing edge skin as panel is lowered into place.

2. Use drift pins of the proper size to aid in aligning the holes. Be careful not to elongate or damage holes, as this may result in leaks.

3. Start fastener screws in holes at scattered locations as the holes are aligned. Check numbers on skin panel and screw board to be sure the fasteners are put in the holes they were removed from. Do not tighten any fasteners until all possible holes are aligned and fasteners started. With the fasteners loose, it is sometimes possible to move the skin panel slightly to align other holes.

4. With fasteners started in all possible locations, tighten groove-sealed fasteners. Apply sealant to individually sealed fasteners and tighten. (Refer to "Sealing Individually Sealed Fasteners.") Check misaligned holes to see if tightening the fasteners aligned them. If so, install, seal (if required), and tighten fasteners.

5. Holes that will not align will have to be drilled out and an oversize screw or a substitute fastener installed. Great care must be used not to drill out the threads of the underlying nut plates if oversize screws are to be installed. Refer to figure 2-7 for permissible substitute fasteners.

6. After all fasteners have been installed and torqued to proper values, the fuel area must be sealed. (Refer to "Wing Fuel Tank Sealing.")

7. Install all access doors on front spar, upper skin, tip rib, and rear spar.

8. Check wing fuel tank for leaks and obstructions. Refer to "Wing Fuel Tank Leak and Obstruction Tests."

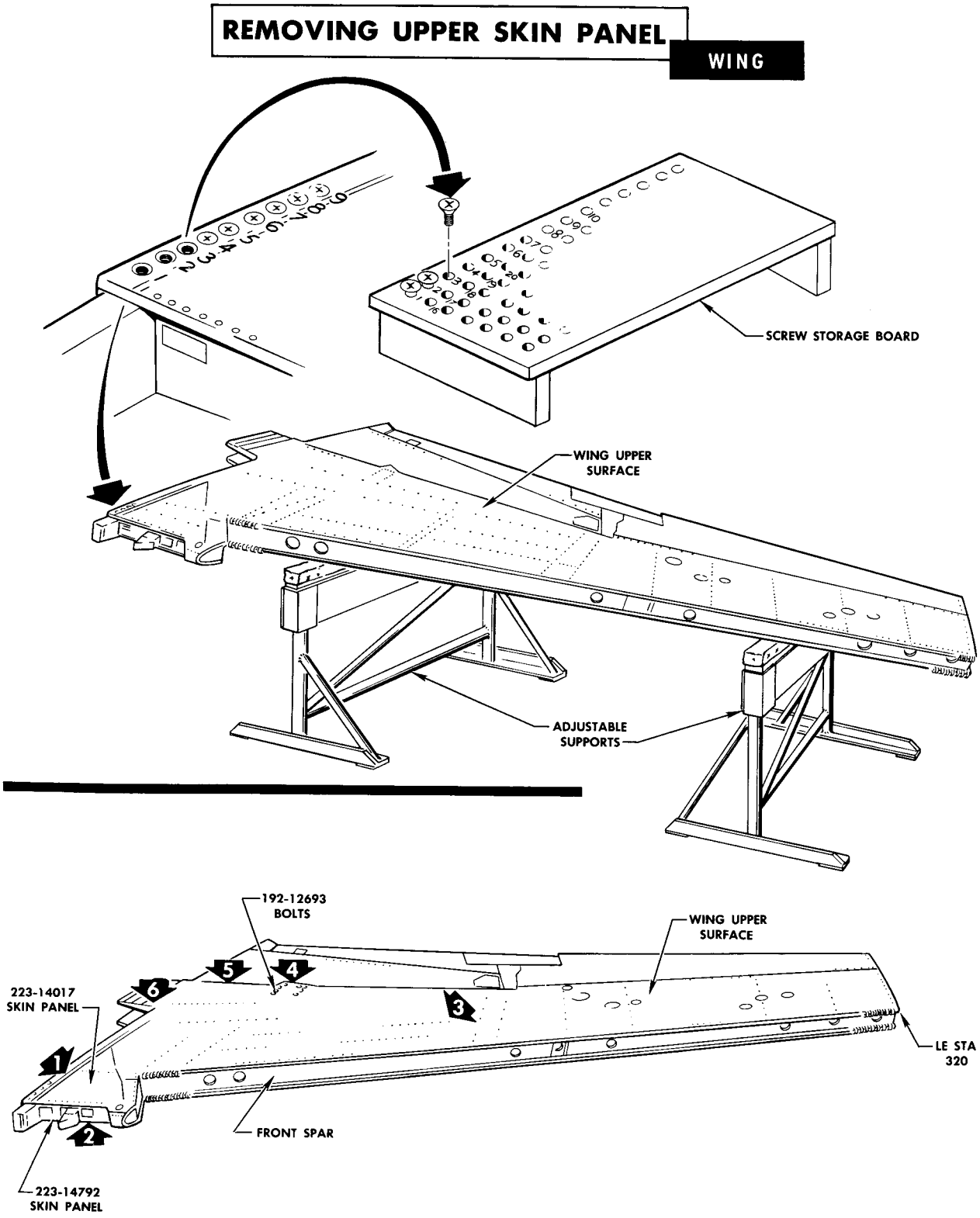
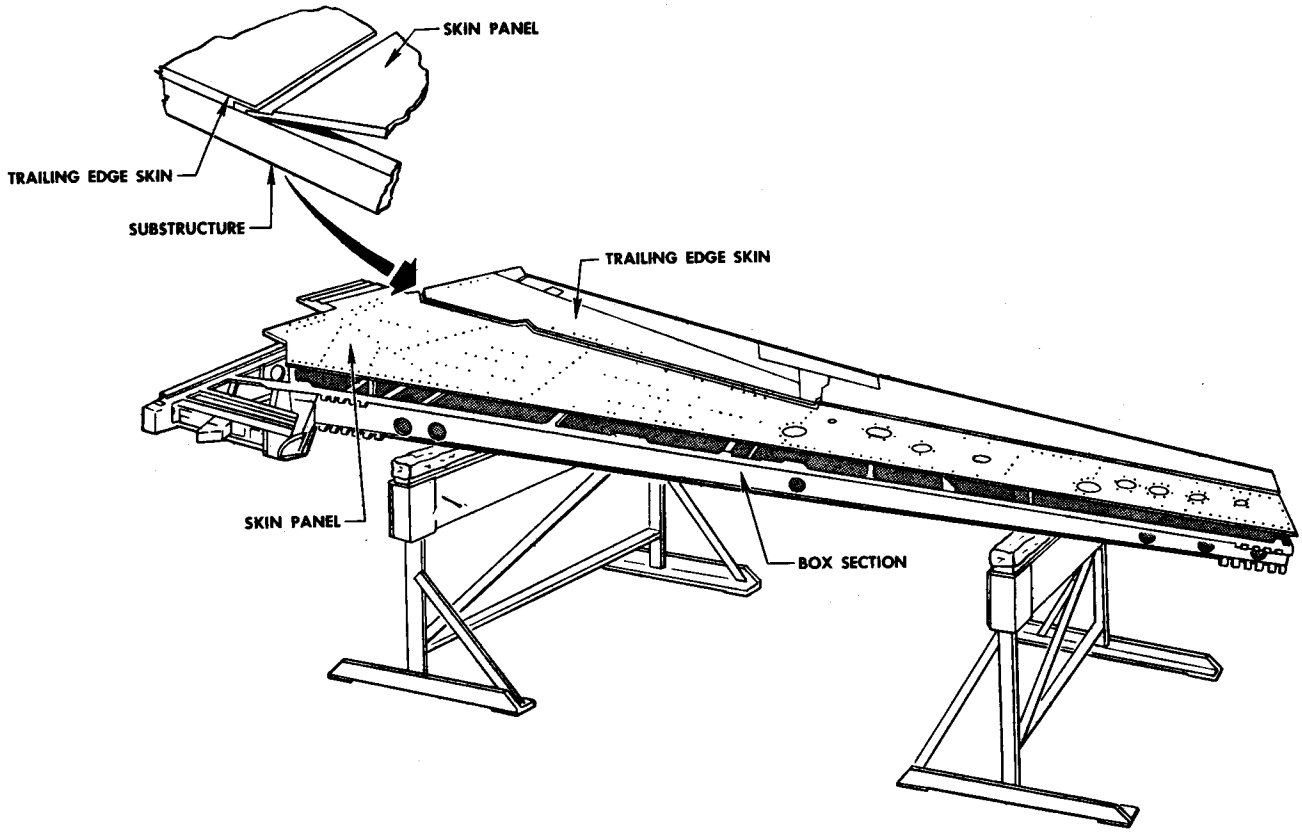


Figure 2-3. Removing Outer Wing Panel Upper Skin

INSTALLING UPPER SKIN PANEL

WING



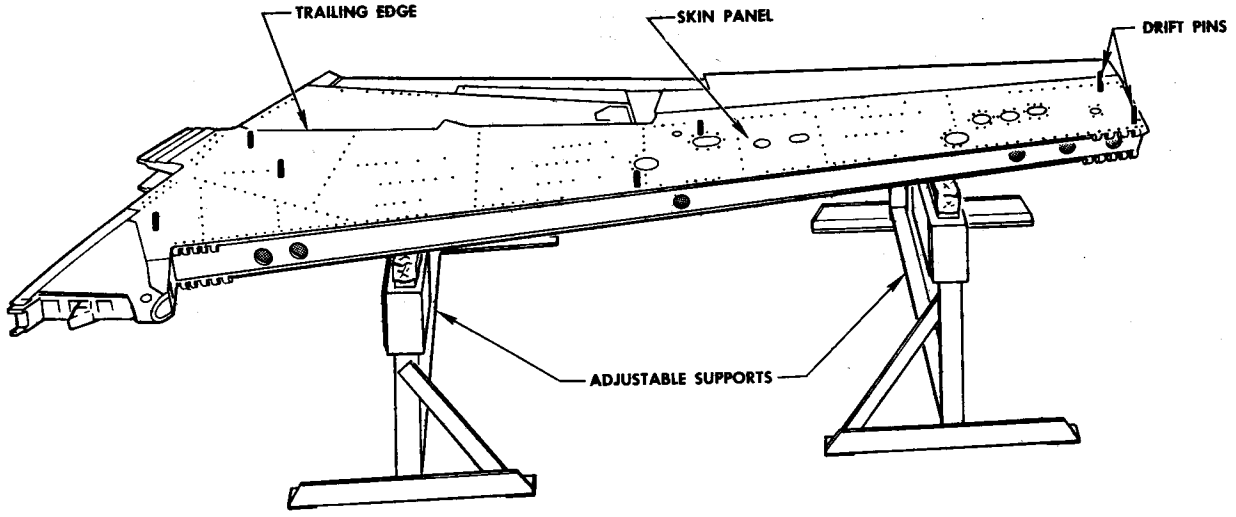


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Figure 2-4. Installing Outer Wing Panel Upper Skin

9. When the sealing and testing procedures are complete, the following must be accomplished as directed in the maintenance handbooks:

- Install landing gear strut, actuator, and doors.
- Install wing flaps and flap actuating cylinder.
- Install aileron and aileron actuating cylinder.
- Install wing tip panel at leading edge station 320 splice.
- Install wing outer panel to center section.
- Install leading edge and slat assemblies.

INTERIOR FINISHING OF INTEGRAL FUEL TANK.

In order to assure a fuel-tight seal and protect against corrosion, it is necessary to properly prepare the interior of the wing and the sealing surfaces before close-out. All bare interior parts of the integral fuel tank, except as noted, should receive a colored chemical film or anodic treatment and be coated with interior coating (MIL-S-4383).

- Standard clips and instrumentation and fuel system lines and components are not to be coated; however, castings and welded seams that form a barrier between wet and dry areas must be treated and coated with interior coating.
- Aluminum surfaces in contact with "Stat-O-Seal" type seals on fasteners passing from a wet to a dry area and all aluminum surfaces in contact with "O" rings or "Gask-O-Seals" should have a chemical or anodic film treatment but should not be coated with interior coating.

(Deleted)

- Gang channels on the bottom of the tank should be coated with interior coating after installation and torquing of fasteners and before final close-out. Areas to be electrically bonded may be stripped of interior coating and chemical or anodic film as required. Affected areas must be touched up with interior coating after bonding. All repair parts exposed to fuel must be chemical-film-treated or anodic-film-treated. Immediately after the chemical or anodic film treatment, two or more uniform coats of interior coating should be sprayed onto the exposed surfaces. All areas to be coated with interior coating must be free from oil or dirt in order to obtain adequate adhesion of the coating. If necessary, solvent-wipe the surface, before application of coating, with petroleum naphtha (Federal Specification TT-N-95) and then with methyl isobutyl ketone before spraying. After the protective coating has been applied it should be air-dried for a minimum of eight hours in a well ventilated area. If a film less susceptible to chip imbedding is desired, the coating may be air-dried for a minimum of 20 minutes and then baked at 275°F, for an additional 20 minutes. Baking at temperatures higher than 275°F is not permissible. Baking at temperatures less than 275°F for various periods of time is permissible, provided sufficient air-dry before baking to prevent blistering is provided. The final state of dry must be equivalent to the eight hours air-dry. (A 5-minute air-dry plus a 20 minute bake at 140-150°F produces a dried film equivalent to eight hours air-dry.)

All foreign material, chips, dirt, etc must be removed from fuel tank area before final close-out. Chips and dirt should be removed by vacuuming the area while pounding on the structure or skin to dislodge particles. It is suggested that narrow mouth attachments be used on the vacuum cleaner to facilitate the removal of chips, dirt, etc, from crevices and corners. It is very important that the fuel area be clean. With this thought in mind, wipe all interior surfaces of the tank lightly with tack rags to remove any chips, dirt, or lint not picked up by the vacuum cleaner. After cleaning the fuel area, check for areas where the protective coating may have been damaged. Any areas where bare metal has been exposed or where the coating has been damaged or removed must be touched up. Areas requiring touch-up must be wiped with a clean lint-free cloth dampened in cleaning solvent and immediately wiped with a clean dry lint-free cloth. After cleaning area, apply a chemical film touch-up (refer to index) and touch up coating by either a brush or spray application of interior coating. Counterbores within the tank but not in the sealing groove should be coated with zinc-chromate primer after cleaning with solvent. Chemical film touch-up is not required. It is suggested that a touch-up gun or self-spray can be used to spray the primer. Overspray should be held to a minimum but need not be removed, provided the surface beneath the overspray has been previously cleaned.

The interior coating of the fuel area should be examined after the drying to ensure proper adhesion, correct film thickness (0.7 to 1.5 mils) and uniform appearance. The coating should be free from pinholes, blisters, runs, alligatoring, or other imperfections. The adhesion of the coating should be tested by the "cross-hatch" method. This is done by making eleven parallel cuts about 1/32 inch apart, with a new razor blade, through the coating. A second set of eleven cuts about 1/32 inch apart are made at right angles and across the first set. A strip of Type I Grade A tape (Specification JAN-P-127) is then firmly pressed down over the cross-hatched area. When the tape is removed in one abrupt motion, one hundred percent of the squares must remain

attached to the metal surface. This test should be repeated at several different places on the coated surfaces. The cross-hatched area must be repaired after inspection by a brush or spray application of interior coating material. If the coating does not pass the test for adhesion, the area that is loose must be stripped of the coating, recleaned, and again coated with interior coating material.

Before final assembly of tank components for sealing, be sure all holes are deburred and clean all chips out of grooved areas. Carefully inspect fastener holes for evidence of oil or other contaminants by spot-checking at intervals with a clean white rag. If any oil or wax is present, as indicated by a discoloration on the rag, clean the holes carefully with cleaning solvent. Immediately before assembling, clean all grooved areas and mating surfaces by solvent-wiping with cleaning solvent. In each case, use a clean rag to apply the solvent and follow up with another clean, dry cloth to wipe the surface dry.

CAUTION

The ability of the groove-sealing compound to seal depends upon its adhesion. Oils and greases are adhesion breakers and must be completely removed from all sealing areas. In this group of contaminants are included hair oils, body oils, and protective hand creams.

After cleaning the wing box section of all foreign material, install wing skin to box section. (Refer to index.) All fasteners must be wiped clean with a solvent rag as they are installed. Torque all skin fasteners to recommended value. (Refer to figure 1-50.) To prevent foreign material from contaminating the sealant groove, all injection holes, or holes in which fasteners have not been installed, should be covered with masking tape until the sealing compound is to be injected.

WING SUBSTITUTE FASTENERS.

WING SCREW SUBSTITUTIONS.

Screws used in the integral wing fuel cell area on F-100D and F-100F Airplanes are manufactured by the American Screw Co under the name Torq-Set. These screws may be identified by the offset cross recess in the head (see figure 2-6) and the letters "TS" stamped on the head. Most of these screws have a standard diameter; however, in some random locations, screws having either a 0.015- or 0.030-inch oversize shank may have been used. These oversize screws may be identified only by part number. When it becomes necessary to replace one or more screws in the tank area, it is recommended that the replacement be the same as the original. If the proper Torq-Set screw is not available screws from two other manufacturers or regular NAS screws may be substituted.

Standard sizes of modified Reed and Prince or standard and oversize Hi-Torque screws may be used. These are substitute screws and should be replaced with the proper sized "Torq-Set" screws as soon as possible. The NAS series screw should be used only when none of the others are available. The screws used in some areas are made of nonmagnetic material. When it becomes necessary to replace screws in these areas, every effort should be made to obtain nonmagnetic replacements. Only in case of emergency or a one-time flight should magnetic screws be used in the nonmagnetic areas. If magnetic screws must be used, they should be replaced with the proper nonmagnetic screw as soon as possible. The screw substitution charts (figure 2-6) show the comparable screw part numbers of the different manufacturers. When making substitutions, use the same hole size and grip length as original. The charts also show the screw-manufacturer's number of the driver tool to be used with each type of screw. The last of the wing screw charts gives the AF stock number for the driver tool. If oversize screws are to be replaced, use a like oversize screw for replacement. Where oversize screws are available, the part number of the oversize screw is also given on the charts. To use the charts, find the size of the screw required along the top of the chart. Under the size, find the number of the driver tool required. Next, under the size, find the basic number of the screw—either standard or oversize. Find the grip length required on the left side of the chart and directly under the screw size, find the dash number of the required screw. Example—HT8M-8, Hi-Torque, ½-20, Magnetic 0.500 inch grip length, use drive tool No. HT3408M, Air Force stock number 9DHT-HT 3408M.

SEALED FASTENER SUBSTITUTION.

Sealed fasteners in the wing fuel tank area are installed with five types of nuts used in different areas.

- Nut plates held by retainers are used along the intermediate spars.
- Gang channel nut plates are used on portions of the front spar.
- Riveted nut plates are used in inaccessible locations on some ribs and partially on diagonal spar.
- Self-wrenching "bumper" nuts are used in accessible locations on some ribs and partially on the front, rear, and diagonal spars. The self-wrenching nut has a tail which "bumps" against the adjoining structure to prevent the nut from turning while being tightened.
- External wrenching nuts (EB), regular stop nuts (AN365), or corrosion-resistant steel stop nuts (AN363C) are used in accessible areas.

USAF STOCK TOOLS—TORQ-SET SCREWS

WING

TORQ-SET SCREW	TORQUE INCH-POUNDS (FOR TENSION-TYPE NUT, OR FULL THD LENGTH)	TORQ-SET SCREWDRIVER	SQUARE DRIVE ADAPTER	SOCKET WRENCH TORQUE INDICATING HANDLE
1/4-28	70-90	EX-170 1-4 (FSN 5120-511-0209)	SC-114 (FSN 5120-203-9618)	FSN 5120-221-7944*
5/16-24	140-203	EX-170 5-16 (FSN 5120-511-0211)	SC-314 (FSN 5120-203-9619)	FSN 5120-595-9073†
3/8-24	190-351	EX-170 3-8 (FSN 5120-511-0210)	SC-314 (FSN 5120-203-9619)	FSN 5120-595-9073†
7/16-20	500-756	EX-170 7-16 (FSN 5120-511-0212)	SC-314 (FSN 5120-203-9619)	FSN 5120-595-9073†
1/2-20	690-990	EX-170 1-2 (FSN 5120-511-0208)	HE-3320 (FSN 5120-203-9620	FSN 5120-595-9077§
9/16-18	1,000-1,440	EX-170 9-16 (FSN 5120-511-0207)	HE-3320 (FSN 5120-203-9620)	FSN 5120-595-9077¶

NOTE Use lower torque values for cotter pin or lockwire secured nuts. Obtain alignment of slot and hole by increasing torque only.

Use higher torque values for torquing from head end. A tolerance of 10 percent above the higher torque limit must not be exceeded.

* Alternate—Use 5120-204-0736 handle with 5120-243-1688 adapter (Part No. A6).

† Alternate—use 7900-428450 handle.

‡ Alternate—Use 7900-428450 handle. For maximum torque, use 5120-595-9077 handle with 5120-240-8702 adapter.

§ Alternate—Use 5120-595-9074 handle.

¶ Alternate—For minimum torque, use 5120-595-9073 handle with 5120-240-8703 adapter.

TORQUE HANDLES—TORQ-SET SCREW

TORQ-SET SCREW	TORQUE HANDLE FEDERAL STOCK NO.	DESCRIPTION
1/4-28	5120-221-7944	HANDLE—SOCKET WRENCH TORQUE INDICATING DUAL, MALE 1/4 IN. SQUARE DRIVE, 5 TO 150 IN. LB, TYPE I, SIZE 1, SPEC MIL-H-4034
5/16-24 3/8-24 7/16-20	5120-595-9073	HANDLE—SOCKET WRENCH TORQUE INDICATING DUAL, MALE 3/8 IN. SQUARE DRIVE, 100 TO 750 IN. LB, SPEC 50442. USE 7900-428450 (0 TO 600 IN. LB) UNTIL SUPPLY IS EXHAUSTED.
1/2-20 9/16-18	5120-595-9077	HANDLE—SOCKET WRENCH TORQUE INDICATING RATCHET, MALE 1/2 IN. SQUARE DRIVE, CAP. 0 TO 1600 INCH-POUNDS, SPEC 50343. WHEN SUPPLY IS EXHAUSTED, USE 5120-595-9074 (700 TO 1600 IN. LB).

F-100D-3-10-135A

Figure 2-5. Torq-Set Screw Torque Table

SCREW CHART

WING

	1 MODIFIED REED & PRINCE (Reed & Prince Mfg Co–Worcester, Mass.) 2 3 MAGNETIC STANDARD SIZE (160,000 PSI Min Ult Tens.)					
SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO.	914	915	916	917	918	919
BASIC NUMBER	RP334CRA	RP335CRA	RP336CRA	RP337CRA	RP338CRA	RP339CRA
GRIP LENGTH	DASH NUMBER					
0.312	– 6	– 6	– 7	– 7	–10	–10-5
0.375	– 6-5*	– 6-5	– 7-5	– 7-5	–10-5	–11
0.437	– 7*	– 7*	–10	–10	–11	–11-5
0.500	– 7-5	– 7-5*	–10-5	–10-5	–11-5	–12
0.562	–10*	–10*	–11*	–11*	–12	–12-5
0.625	–10-5*	–10-5*	–11-5*	–11-5	–12-5	–13
0.687	–11*	–11*	–12*	–12	–13	–13-5
0.750	–11-5	–11-5*	–12-5*	–12-5	–13-5	–14
0.812	–12	–12*	–13*	–13	–14	–14-5
0.875	–12-5*	–12-5*	–13-5*	–13-5	–14-5*	–15
0.937	–13*	–13*	–14*	–14	–15*	–15-5
1.000	–13-5	–13-5*	–14-5	–14-5	–15-5*	–16*
1.062	–14	–14*	–15	–15	–16*	–16-5*
1.125	–14-5	–14-5*	–15-5*	–15-5	–16-5*	–17*
1.187	–15	–15*	–16	–16	–17*	–17-5*
1.250	–15-5	–15-5	–16-5	–16-5	–17-5*	–20*
1.312	–16	–16	–17*	–17	–20*	–20-5*
1.375	–16-5	–16-5	–17-5*	–17-5	–20-5	–21*
1.437	–17	–17	–20*	–20	–21	–21-5*
1.500	–17-5	–17-5	–20-5*	–20-5	–21-5*	–22*
1.562	–20	–20	–21*	–21*	–22	–22-5*
1.625	–20-5	–20-5	–21-5*	–21-5*	–22-5	–23*
1.687	–21	–21	–22*	–22	–23	–23-5*
1.750	–21-5	–21-5	–22-5*	–22-5	–23-5	–24*
1.812	–22	–22	–23	–23	–24	–24-5
1.875	–22-5	–22-5	–23-5	–23-5	–24-5	–25
1.937	–23	–23	–24	–24	–25	–25-5
2.000	–23-5	–23-5	–24-5	–24-5	–25-5	–26

1 High-torque screws having differently slotted heads should not be intermixed on an airplane.

2 No oversize available. Use Hi-Torque series. (See chart No. 2.)

2 No nonmagnetic available. Use 7S27 or TS-7S27.

*Sizes marked thus * are called out on drawings.

F-100D-3-10-78

Figure 2-6. Wing Screw Charts (Sheet 1 of 8)

SCREW CHART

WING

	1 HI-TORQUE (Hi-Shear Rivet Tool Co, Los Angeles 45, Calif.) MAGNETIC (160,000 PSI Min Ult Tens.)					
SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO.	HT3304M	HT3305M	HT3306M	HT3407M	HT3408M	HT3409M
STD SIZE BASIC NO.	HT4M	HT5M	HT6M	HT7M	HT8M	HT9M
0.015 Oversize Basic No. 4	HT41M	HT51M	HT61M	HT71M	HT81M	HT91M
0.031 Oversize Basic No. 4	HT42M	HT52M	HT62M	HT72M	HT82M	HT92M
GRIP LENGTH	DASH NUMBER					
0.312	– 5	– 5	– 5	– 5	– 5	– 5
0.375	– 6*	– 6	– 6	– 6	– 6	– 6
0.437	– 7*	– 7*	– 7	– 7	– 7	– 7
0.500	– 8	– 8*	– 8	– 8	– 8	– 8
0.562	– 9*	– 9*	– 9*	– 9*	– 9	– 9
0.625	–10*	–10*	–10*	–10	–10	–10
0.687	–11*	–11*	–11*	–11	–11	–11
0.750	–12	–12*	–12*	–12	–12	–12
0.812	–13	–13*	–13*	–13	–13	–13
0.875	–14*	–14*	–14*	–14	–14*	–14
0.937	–15	–15*	–15*	–15	–15*	–15
1.000	–16*	–16*	–16	–16	–16*	–16*
1.062	–17	–17*	–17	–17	–17*	–17*
1.125	–18	–18*	–18*	–18	–18*	–18*
1.187	–19	–19*	–19	–19	–19*	–19*
1.250	–20	–20	–20	–20	–20*	–20*
1.312	–21	–21	–21*	–21	–21*	–21*
1.375	–22	–22	–22*	–22	–22	–22*
1.437	–23	–23	–23*	–23	–23	–23*
1.500	–24	–24	–24*	–24	–24*	–24*
1.562	–25	–25	–25*	–25*	–25	–25*
1.625	–26	–26	–26*	–26*	–26	–26*
1.687	–27	–27	–27*	–27	–27	–27*
1.750	–28	–28	–28*	–28	–28	–28*
1.812	–29	–29	–29	–29	–29	–29
1.875	–30	–30	–30	–30	–30	–30
1.937	–31	–31	–31	–31	–31	–31
2.000	–32	–32	–32	–32	–32	–32

1 High-torque screws having differently slotted heads should not be intermixed on an airplane.

4 Used occasionally in production repair. Not stock items.

* Sizes marked thus * are called out on drawings.

F-100D-3-10-79

Figure 2-6. Wing Screw Charts (Sheet 2 of 8)

SCREW CHART

WING

	1 TORQ-SET (American Screw Co, Willimantic, Conn.) MAGNETIC (160,000 PSI Min Ult Tens.)					
SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO. ADAPTER NO.	EX-170-1/4 SC-114	EX-170-5/16 SC-314	EX-170-3/8 SC-314	EX-170-7/16 SC-314	EX-170-1/2 HE-3320	EX-170-9/16 HE-3320
STD SIZE BASIC NO.	TS4A	TS5A	TS6A	TS7A	TS8A	TS9A
0.015 Oversize Basic No. 4	TS41A	TS51A	TS61A	TS71A	TS81A	TS91A
0.031 Oversize Basic No. 4	TS42A	TS52A	TS62A	TS72A	TS82A	TS92A
GRIP LENGTH	DASH NUMBER					
0.312	− 5	− 5	− 5	− 5	− 5	− 5
0.375	− 6*	− 6	− 6	− 6	− 6	− 6
0.437	− 7*	− 7*	− 7	− 7	− 7	− 7
0.500	− 8	− 8*	− 8	− 8	− 8	− 8
0.562	− 9*	− 9*	− 9*	− 9*	− 9	− 9
0.625	−10*	−10*	−10*	−10	−10	−10
0.687	−11*	−11*	−11*	−11	−11	−11
0.750	−12	−12*	−12*	−12	−12	−12
0.812	−13	−13*	−13*	−13	−13	−13
0.875	−14*	−14*	−14*	−14	−14*	−14
0.937	−15*	−15*	−15*	−15	−15*	−15
1.000	−16	−16*	−16*	−16	−16*	−16*
1.062	−17	−17*	−17	−17	−17*	−17*
1.125	−18	−18*	−18*	−18	−18*	−18*
1.187	−19	−19*	−19	−19	−19*	−19*
1.250	−20	−20	−20	−20	−20*	−20*
1.312	−21	−21	−21*	−21	−21*	−21*
1.375	−22	−22	−22*	−22	−22	−22*
1.437	−23	−23	−23*	−23	−23	−23*
1.500	−24	−24	−24*	−24	−24*	−24*
1.562	−25	−25	−25*	−25*	−25	−25*
1.625	−26	−26	−26*	−26*	−26	−26*
1.687	−27	−27	−27*	−27	−27	−27*
1.750	−28	−28	−28*	−28	−28	−28*
1.812	−29	−29	−29	−29	−29	−29
1.875	−30	−30	−30	−30	−30	−30
1.937	−31	−31	−31	−31	−31	−31
2.000	−32	−32	−32	−32	−32	−32

1 High-torque screws having differently slotted heads should not be intermixed on an airplane.

4 Used occasionally in production repair. Not stock items.

* Sizes marked thus * are called out on drawings.

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Figure 2-6. Wing Screw Charts (Sheet 3 of 8)

SCREW CHART

WING

1 TORQ-SET (American Screw Co, Willimantic, Conn.) NONMAGNETIC STANDARD SIZE (150,000 PSI Min Ult Tens.)						
SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO. ADAPTER NO.	EX-170-1/4 SC-114	EX-170-5/16 SC-314	EX-170-3/8 SC-314	EX-170-7/16 SC-314	EX-170-1/2 HE-3320	EX-170-9/16 HE-3320
BASIC NUMBER	TS-7S27 -416	TS-7S27 -516	TS-7S27 -616	TS-7S27 -716	TS-7S27 -816	TS-7S27 -916
GRIP LENGTH	DASH NUMBER					
0.312	-13			-15	-17	
0.344		-14	-15			-19
0.375	-14*			-16	-18	
0.406		-15	-16			-20
0.437	-15			-17	-19	
0.469		-16	-17			-21
0.500	-16			-18	-20	
0.531		-17	-18			-22
0.562	-17*			-19	-21	
0.594		-18	-19*			-23
0.625	-18*			-20	-22	
0.656		-19	-20*			-24
0.687	-19			-21	-23	
0.719		-20	-21			-25
0.750	-20			-22	-24	
0.781		-21	-22			-26
0.812	-21			-23	-25	
0.844		-22	-23			-27
0.875	-22			-24	-26	
0.906		-23	-24			-28
0.937	-23			-25	-27	
0.969		-24	-25			-29
1.000	-24			-26	-28	
1.031		-25	-26			-30
1.062	-25			-27	-29	
1.094		-26	-27			31
1.125	-26			-28	-30	
1.156		-27	-28			-32
1.187	-27			-29	-31	
1.219		-28	-29			-33
1.250	-28			-30	-32	
1.281		-29	-30			-34
1.312	-29			-31	-33	
1.344		-30	-31			-35
1.375	-30			-32	-34	
1.406		-31	-32			-36
1.437	-31			-33	-35	
1.469		-32	-33			-37
1.500	-32			-34	-36	
1.531		-33	-34			-38
1.562	-33			-35	-37	
1.594		-34	-35			-39
1.625	-34			-36	-38	
1.656		-35	-36			-40
1.687	-35			-37	-39	
1.719		-36	-37			-41
1.750	-36			-38	-40	
1.781		-37	-38			-42
1.812	-37			-39	-41	
1.844		-38	-39			-43
1.875	-38			-40	-42	
1.906		-39	-40			-44
1.937	-39			-41	-43	
1.969		-40	-41			45
2.000	-40			-42	-44	

1 High-torque screws having differently slotted heads should not be intermixed on an airplane

* Sizes marked thus * are called out on drawings.

F-100D-3-10-81

Figure 2-6. Wing Screw Charts (Sheet 4 of 8)

SCREW CHART

WING

NORTH AMERICAN AVIATION, INC. (Std)
NONMAGNETIC STANDARD SIZE (125,000 PSI Min Ult Tens.)

SIZE	10-32	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO.	NO. 2 Phillips-head	NO. 3 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head
BASIC NUMBER	7S21 -10	7S21 -416	7S21 -516	7S21 -616	7S21 -716	7S21 -816	7S21 -916
GRIP LENGTH	DASH NUMBER						
0.312	-12	-13	-14	-15	-15	-17	-19
0.344							
0.375	-13	-14*			-16	-18	
0.406			-15	-16			-20
0.437	-14*	-15			-17	-19	
0.469			-16	-17			-21
0.500	-15	-16			-18	-20	
0.531			-17	-18			-22
0.562	-16*	-17*			-19	-21	
0.594			-18	-19*			-23
0.625	-17*	-18*			-20	-22	
0.656			-19	-20*			-24
0.687	-18	-19			-21	-23	
0.719			-20	-21			-25
0.750	-19	-20			-22	-24	
0.781			-21	-22			-26
0.812	-20	-21			-23	-25	
0.844			-22	-23			-27
0.875	-21	-22			-24	-26	
0.906			-23	-24			-28
0.937	-22	-23			-25	-27	
0.969			-24	-25			-29
1.000	-23	-24			-26	-28	
1.031			-25	-26			-30
1.062	-24	-25			-27	-29	
1.094			-26	-27			-31
1.125	-25	-26			-28	-30	
1.156			-27	-28			-32
1.187	-26	-27			-29	-31	
1.219			-28	-29			-33
1.250	-27	-28			-30	-32	
1.281			-29	-30			-34
1.312	-28	-29			-31	-33	
1.344			-30	-31			-35
1.375	-29	-30			-32	-34	
1.406			-31	-32			-36
1.437	-30	-31			-33	-35	
1.469			-32	-33			-37
1.500	-31	-32			-34	-36	
1.531			-33	-34			-38
1.562	-32	-33			-35	-37	
1.594			-34	-35			-39
1.625	-33	-34			-36	-38	
1.656			-35	-36			-40
1.687	-34	-35			-37	-39	
1.719			-36	-37			-41
1.750	-35	-36			-38	-40	
1.781			-37	-38			-42
1.812	-36	-37			-39	-41	
1.844			-38	-39			-43
1.875	-37	-38			-40	-42	
1.906			-39	-40			-44
1.937	-38	-39			-41	-43	
1.969			-40	-41			-45
2.000	-39	-40			-42	-44	

* Sizes marked thus * are called out on drawings.

F-100D-3-10-46

Figure 2-6. Wing Screw Charts (Sheet 5 of 8)

SCREW CHART

WING

NORTH AMERICAN AVIATION, INC. (Std)
NONMAGNETIC STANDARD SIZE (150,000 PSI Min Ult Tens.)

SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO.	NO. 3 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head
BASIC NUMBER	7S27 -416	7S27 -516	7S27 -616	7S27 -716	7S27 -816	7S27 -916
GRIP LENGTH	DASH NUMBER					
0.312	-13	-14	-15	-15	-17	-19
0.344				-16	-18	
0.375	-14*					
0.406	-15	-15	-16	-17	-19	-20
0.437		-16	-17			-21
0.469	-16			-18	-20	
0.500						
0.531	-17*	-17	-18	-19	-21	-22
0.562		-18*	-19*			-23
0.594	-18*			-20	-22	
0.625						
0.656	-19	-19	-20*	-21	-23	-24
0.687		-20	-21			-25
0.719	-20			-22	-24	
0.750						
0.781	-21	-21	-22	-23	-25	-26
0.812		-22	-23			-27
0.844	-22			-24	-26	
0.875						
0.906	-23	-23	-24	-25	-27	-28
0.937		-24	-25			-29
0.969	-24			-26	-28	
1.000						
1.031	-25	-25	-26	-27	-29	-30
1.062		-26	-27			-31
1.094	-26			-28	-30	
1.125						
1.156	-27	-27	-28	-29	-31	-32
1.187		-28	-29			-33
1.219	-28			-30	-32	
1.250						
1.281	-29	-29	-30	-31	-33	-34
1.312		-30	-31			-35
1.344	-30			-32	-34	
1.375						
1.406	-31	-31	-32	-33	-35	-36
1.437		-32	-33			-37
1.469	-32			-34	-36	
1.500						
1.531	-33	-33	-34	-35	-37	-38
1.562		-34	-35			-39
1.594	-34			-36	-38	
1.625						
1.656	-35	-35	-36	-37	-39	-40
1.687		-36	-37			-41
1.719	-36			-38	-40	
1.750						
1.781	-37	-37	-38	-39	-41	-42
1.812		-38	-39			-43
1.844	-38			-40	-42	
1.875						
1.906	-39	-39	-40	-41	-43	-44
1.937		-40	-41			-45
1.969	-40			-42	-44	
2.000						

* Sizes marked thus * are called out on drawings.

F-100D-3-10-82

Figure 2-6. Wing Screw Charts (Sheet 6 of 8)

SCREW CHART

WING

NAS SERIES MAGNETIC STANDARD SIZE (160,000 PSI Min Ult Tens.)						
SIZE	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20	9/16-18
DRIVER TOOL NO.	NO. 3 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head	NO. 4 Phillips-head
BASIC NUMBER	NAS 334 CPA	NAS 335 CPA	NAS 336 CPA	NAS 337 CPA	NAS 338 CPA	NAS 339 CPA
GRIP LENGTH	DASH NUMBER					
0.312	– 6	– 6	– 7	– 7	–10	–10-5
0.375	– 6-5	– 6-5	– 6-5	– 7-5	–10-5	–11
0.437	– 7	– 7	–10	–10	–11	–11-5
0.500	– 7-5	– 7-5	–10-5	–10-5	–11-5	–12
0.562	–10	–10	–11	–11	–12	–12-5
0.625	–10-5	–10-5	–11-5	–11-5	–12-5	–13
0.687	–11	–11	–12	–12	–13	–13-5
0.750	–11-5	–11-5	–12-5	–12-5	–13-5	–14
0.812	–12	–12	–13	–13	–14	–14-5
0.875	–12-5	–12-5	–13.5	–13-5	–14-5	–15
0.937	–13	–13	–14	–14	–15	–15-5
1.000	–13-5	–13-5	–14-5	–14-5	–15-5	–16
1.062	–14	–14	–15	–15	–16	–16-5
1.125	–14-5	–14-5	–15-5	–15-5	–16-5	–17
1.187	–15	–15	–16	–16	–17	–17-5
1.250	–15-5	–15-5	–16-5	–16-5	–17-5	–20
1.312	–16	–16	–17	–17	–20	–20-5
1.375	–16-5	–16-5	–17-5	–17-5	–20-5	–21
1.437	–17	–17	–20	–20	–21	–21-5
1.500	–17-5	–17-5	–20.5	–20-5	–21-5	–22
1.562	–20	–20	–21	–21	–22	–22-5
1.625	–20-5	–20-5	–21-5	–21-5	–22-5	–23
1.687	–21	–21	–22	–22	–23	–23-5
1.750	–21-5	–21-5	–22-5	–22-5	–23-5	–24
1.812	–22	–22	–23	–23	–24	–24-5
1.875	–22-5	–22-5	–23-5	–23-5	–24-5	–25
1.937	–23	–23	–24	–24	–25	–25-5
2.000	–23-5	–23-5	–24-5	–24-5	–25-5	–26

Example of NAS Part No.: NAS 334 CPA22-5

=¼–28 steel bolt, 1.875 grip, plated shank, Phillips recess, without cotter pin hole. (Screw must be cadmium plated.)

F-100D-3-10-120

Figure 2-6. Wing Screw Charts (Sheet 7 of 8)

SCREW TYPE	DRIVER NO.	FEDERAL STOCK NUMBER
REED & PRINCE	915	5120-596-4068
	916	5120-596-4067
	917	5120-596-4049
	918	5120-596-4050
	919	5120-596-4051
HI-TORQUE	HT 3304M	5120-315-4004
	HT 3305M	5120-315-4005
	HT 3306M	5120-315-4006
	HT 3407M	5120-315-4007
	HT 3408M	5120-315-4008
	HT 3409M	5120-335
TORQ-SET	EX-170 1/4	5120-511-0209
	EX-170 5/16	5120-511-0211
	EX-170 3/8	5120-511-0210
	EX-170 7/16	5120-511-0212
	EX-170 1/2	5120-511-0208
	EX-170 9/16	5120-511-0207
(ADAPTERS)	SC-114	5120-203-9618
	SC-314	5120-203-9619
	HE-3320	5120-203-9620

F-100D-3-10-119A

Figure 2-6. Wing Screw Charts (Sheet 8 of 8)

When a nut plate for a groove-sealed screw or an individually-sealed screw is damaged, a repair may be made as follows (repairs are listed in order of preference):

- Replace fastener with same type screw and nut plates if possible.
- Replace fastener with same type screw, using a self-wrenching nut, an "EB" nut or an AN365 or AN363C stop nut in accessible areas.
- Replace fastener with a blind fastener, subject to limitations. For the location, size and type of screws, type of nut or nut plate, and thickness of material to determine fastener grip length, refer to figure 2-7 or 2-8.

BLIND FASTENER LIMITATION.

Occasionally, because of manufacturing tolerances, it has been necessary to use 0.015-inch or 0.030-inch oversize screws in place of the originally required nominal-diameter screw. For this reason, it is necessary to measure the hole before making any replacement of a sealed fastener. Some of the blind fasteners used as substitutions require hole sizes so close to the nominal fastener standard size that it is not possible to use them in the 0.015-inch oversize holes. The following will aid in determining the correct size blind fastener to use.

- The ¼-inch diameter Jo-bolt requires a "G" (0.261-inch) diameter hole and thus may be used to replace either a standard ¼-inch diameter screw or an 0.015-inch oversize screw.
- The ¼-inch diameter Deutsch rivet requires an "F" (0.257-inch) diameter hole and cannot be used to replace a ¼-inch diameter 0.015-inch oversize screw.
- 5/16-inch diameter Jo-bolts and Deutsch rivets require a nominal size hole and may be used to replace standard 5/16-inch diameter srcews. If a 5/16-inch 0.015-inch oversize screw is to be replaced, the next size larger (⅜-inch diameter) Jo-bolt or Deutsch rivet should be used.
- The ⅜-inch diameter Deutsch rivet requires a "W" (0.386-inch) diameter hole and thus may be used to replace either a standard ⅜-inch diameter screw or a ⅜-inch diameter 0.015-inch oversize screw.
- A ⅜-inch diameter Jo-bolt cannot be used to replace a ⅜-inch diameter 0.015-inch oversize screw.

For fasteners that may be substituted and the type and quantity of substitute fasteners permitted, see figure 2-7 or 2-8.

INSTALLATION OF BLIND FASTENERS.

In areas where blind fasteners may be used, the installation may be made in either of two ways, depending upon conditions. The first and definitely the preferred way to install a blind fastener, is with the blind fastener grip corresponding to the thickness of the structure. This means if a nut plate is attached inside, it must be partially or completely removed. There are several ways to eliminate nut plate interference. Riveted nut plates may be dislodged into the tank by driving with a punch the size of the hole. Nut plates in a gang channel may be removed in the same way. Nut plates on the intermediate spars float in the channel spar caps and are positioned by a retainer strip which engages grooves inside the channel. Some early airplanes have nut plates which consist of a regular nut in a swayed retainer used along the intermediate spar. These nuts can be driven out of the retainer and permitted to fall into the tank interior. This is done by pushing the nut plate to one side as far as possible (use up all available float) and by contacting one edge of the nut with a flat-ended punch. Several sharp blows will usually dislodge the nut, leaving the retainer and retainer strip in place in the spar channel. Neither retainer nor retaining strip will interfere with a blind fastener. Later airplanes have one-piece nut plates in the intermediate spar channels. These nut plates cannot be completely removed; however, it is possible to force them inward so as to be beyond interference with the blind fastener. To force the nut plate inward, drill through the nut plate to remove all threads. Use the correct size drill for the blind fastener to be installed. Force the nut plate inward away from the spar cap. This may be done by using a 3/16-inch diameter punch with a 30-degree bevel ground on the end. Push the nut plate to

one side as far as possible to use up all available float. Insert punch in the hole and make contact with the nut plate. Wrap a piece of masking tape around the punch ¼ inch from the skin surface, for use as a depth gage. The nut plate must be forced inward a minimum of ¼ inch. Several sharp blows with a hammer will cause the retaining strip to bend inward enough to keep the nut plate from interfering with the blind fastener.

Note

Deforming the retaining strip may affect adjacent nut plates. While it probably will not let the nut plates drop loose if adjacent screws are later removed, it may misalign the nut plates so that difficult threading or cross-threading results. **For this reason, it is always desirable to back screws only part way out, leaving a few threads engaged, when resealing any individually sealed screws.**

The hole in the nut plate should be kept in alignment with the fastener hole in the skin, since the end of the Jo-bolt will extend a short distance into the enlarged hole in the nut plate before being driven. On the upper surface of the wing, gravity will cause the nut plate to fall away from the spar cap, making it easy to install the blind fastener. On the lower surfaces of the wing, however, gravity causes the nut plate to rest on the spar cap. In this position, the nut plate interferes with the proper installation of a Jo-bolt. To overcome this problem, it is necessary to hold the nut plate away from the spar cap while the fastener is driven. After forcing the nut plate inward ¼ inch as described, press a small bead of sealant compound PR-703 firmly on the end of the Jo-bolt. The bead should be the same diameter as the Jo-bolt and should be 5/16 to 3/8 inch in length. When the Jo-bolt is inserted in the hole, the sealant will lift the nut plate and hold it away from the Jo-bolt. The Jo-bolt may then be driven in a normal manner.

The second and least preferred method of installing a Jo-bolt is through both the structure and an existing nut plate. This method is acceptable if a better method cannot be used. There are several disadvantages to this method of installation, one being that the nut portion of the nut plate may be cracked, or the nut plate may have a plastic lock ring against which the upset end of the Jo-bolt will rest; also, it is hard to determine the correct grip length to use, because the nut plates of different manufacturers vary in height. If it is necessary to use this type of installation, it will be necessary to drill out the nut plate threads to permit the Jo-bolt to pass through. Chips should be kept out of the tank as much as possible.

Regardless of the type of blind fastener used, the following steps are necessary:

- The hole must be reamed to proper size for Jo-bolt or Deutsch rivet. (Refer to index for installation data.)
- The screw recess in the wing must be solvent-wiped in order to assure a good seal.
- Prepare and seal Jo-bolt itself with PR-1221 sealant. (See figure 2-9.) The exterior of the Jo-bolt should be free of all grease or hand oil when installed, to assure a good seal.
- Prepare and seal Deutsch rivets by dipping in PR-1221 sealant, and drive rivet while the sealant is still wet.
- On an individually sealed hole, install fastener part way into hole and extrude a bead of sealant compound PR-703 around annular groove. Press blind fastener into place, allowing excess sealant to extrude from beneath the head, and then complete installation of fastener.
- When fastener heads are more than 0.015 inch below the surface of the skin, they should be filled with aerodynamic filler. (Refer to index.)
- If possible, remove any loose nuts, etc, that drop into the tank.

If a screw in an inaccessible area cannot be properly torqued and cannot be removed due to the turning of the nut or nut plate, refer to figure 2-11 for removal procedure.

WING FUEL TANK SEALING.

The wing fuel tanks on the F-100D and F-100F Airplanes are sealed against fuel leakage by the North American "integral attachment groove injection sealing method." Grooves 0.050 inch deep and 0.188 inch wide are machined into one structural member at mating surfaces. These grooves run along the centerlines of rows of closely spaced screws or bolts, which hold the parts together. A spotface in the groove at each fastener provides a continuous sealant path which permits sealing the fastener while continuing the mating surface seal. A suitable nonhardening sealing compound is injected into the grooves which interconnect on the various parts of the fuel-carrying part of the wing. The sealing compound is injected through special holes spaced a few inches apart along the grooves. After injection of the sealing compound, these holes are closed with self-tapping screws. Injection is done with a pressure gun operated either by air pressure or by hand. (See figure 2-12.) This method of sealing the wing fuel tank makes it possible to inject the sealing compound from the exterior of the wing. It also enables the periodic repair of the sealed area to be done by reinjecting the grooves with more nonhardening sealing compound, if required. Where mating surface sealing is not required (e.g. intermediate spars, etc), screws are individually sealed during installation. A special counterbore under the head of each screw provides a space for sealing compound which is injected under the screwhead during installation. (See figure

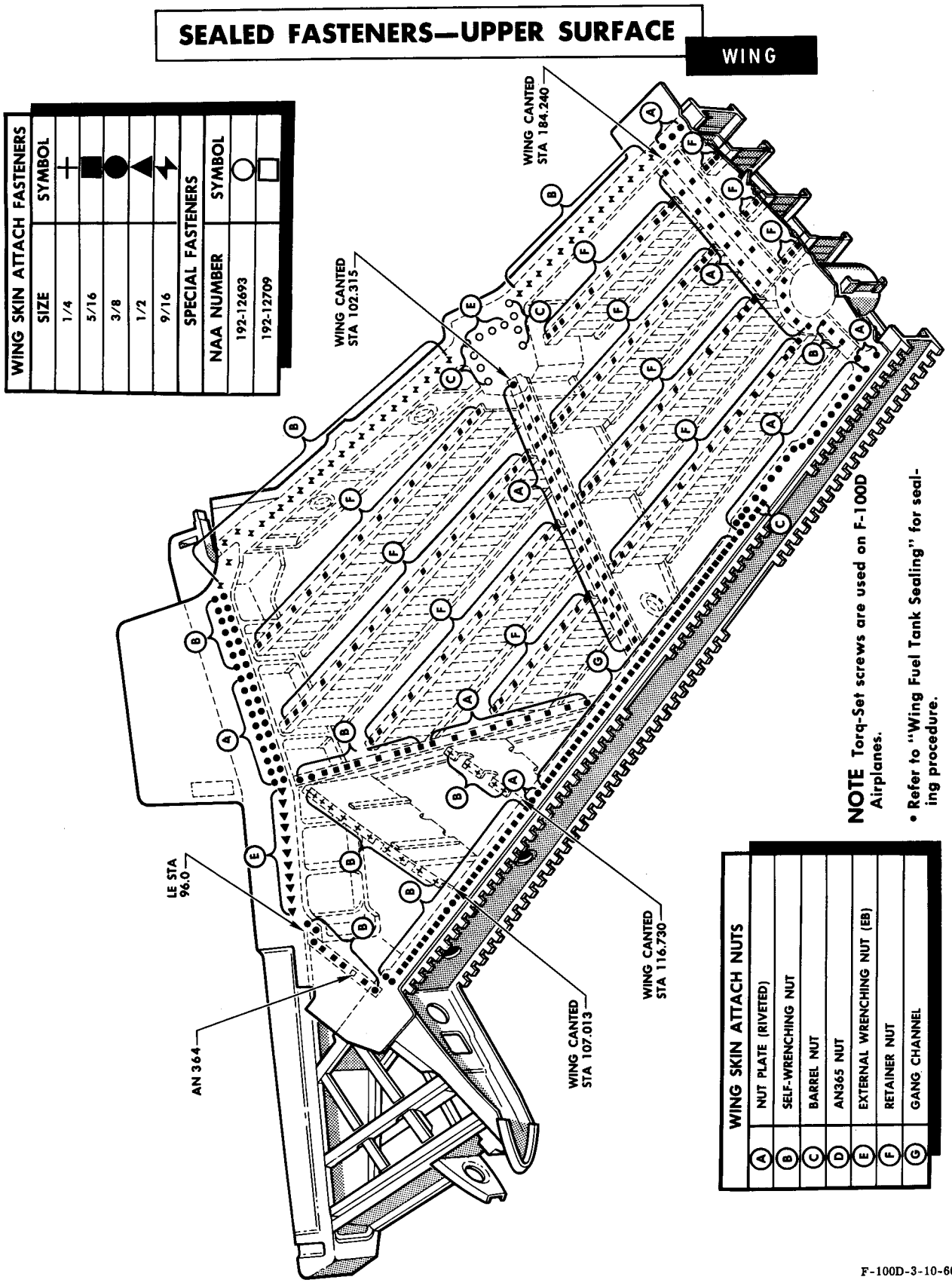


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 1 of 6)

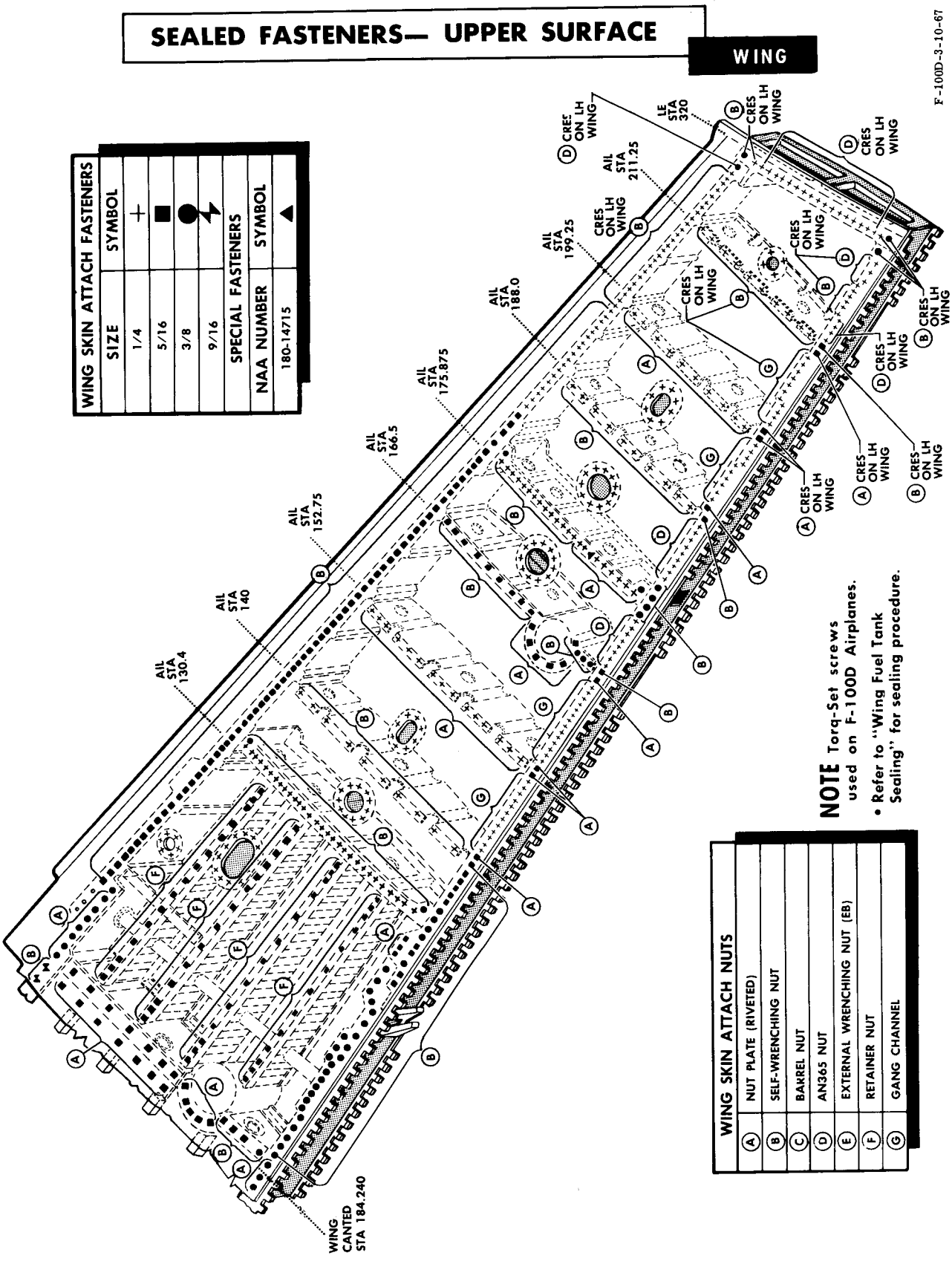


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 2 of 6)

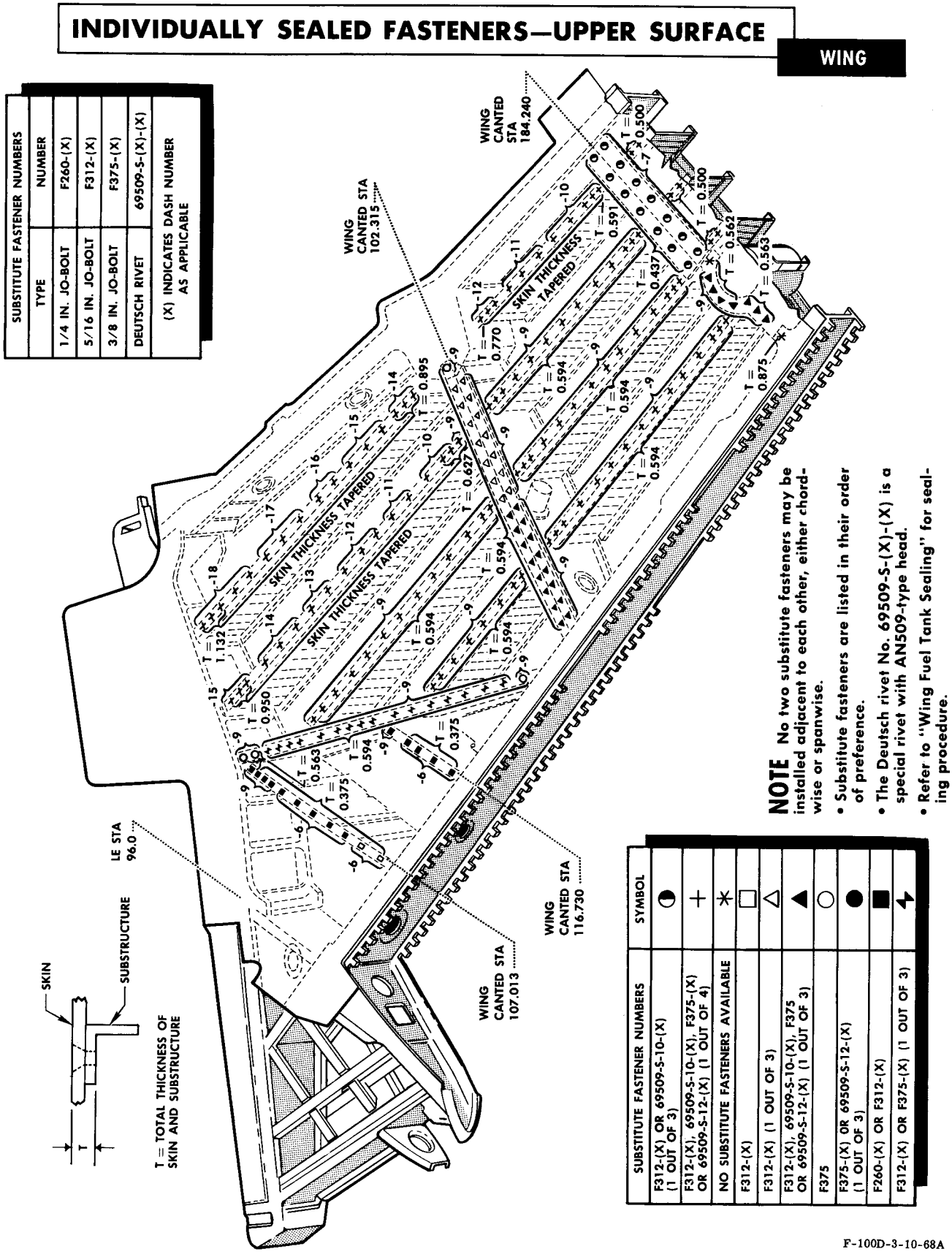


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 3 of 6)

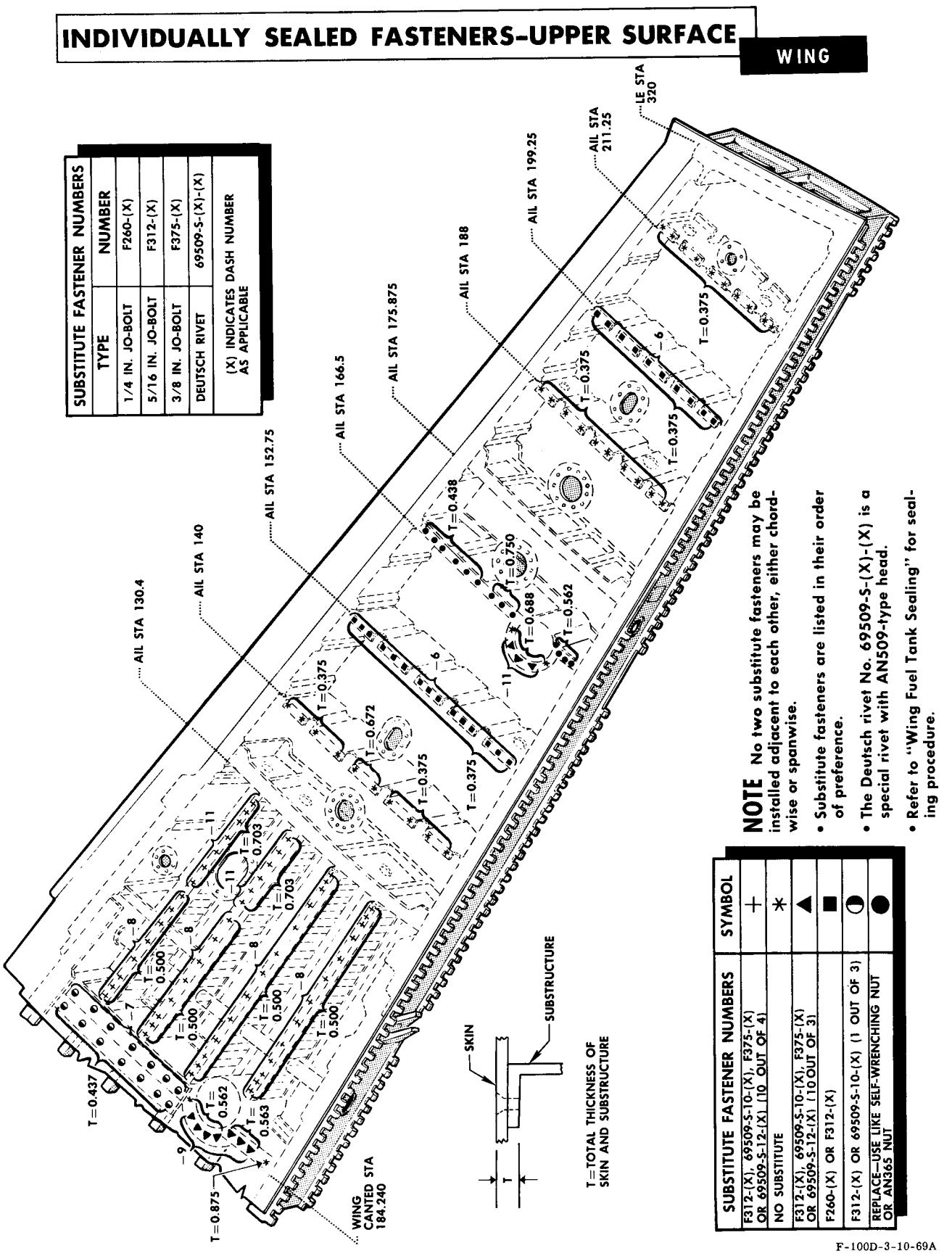


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 4 of 6)

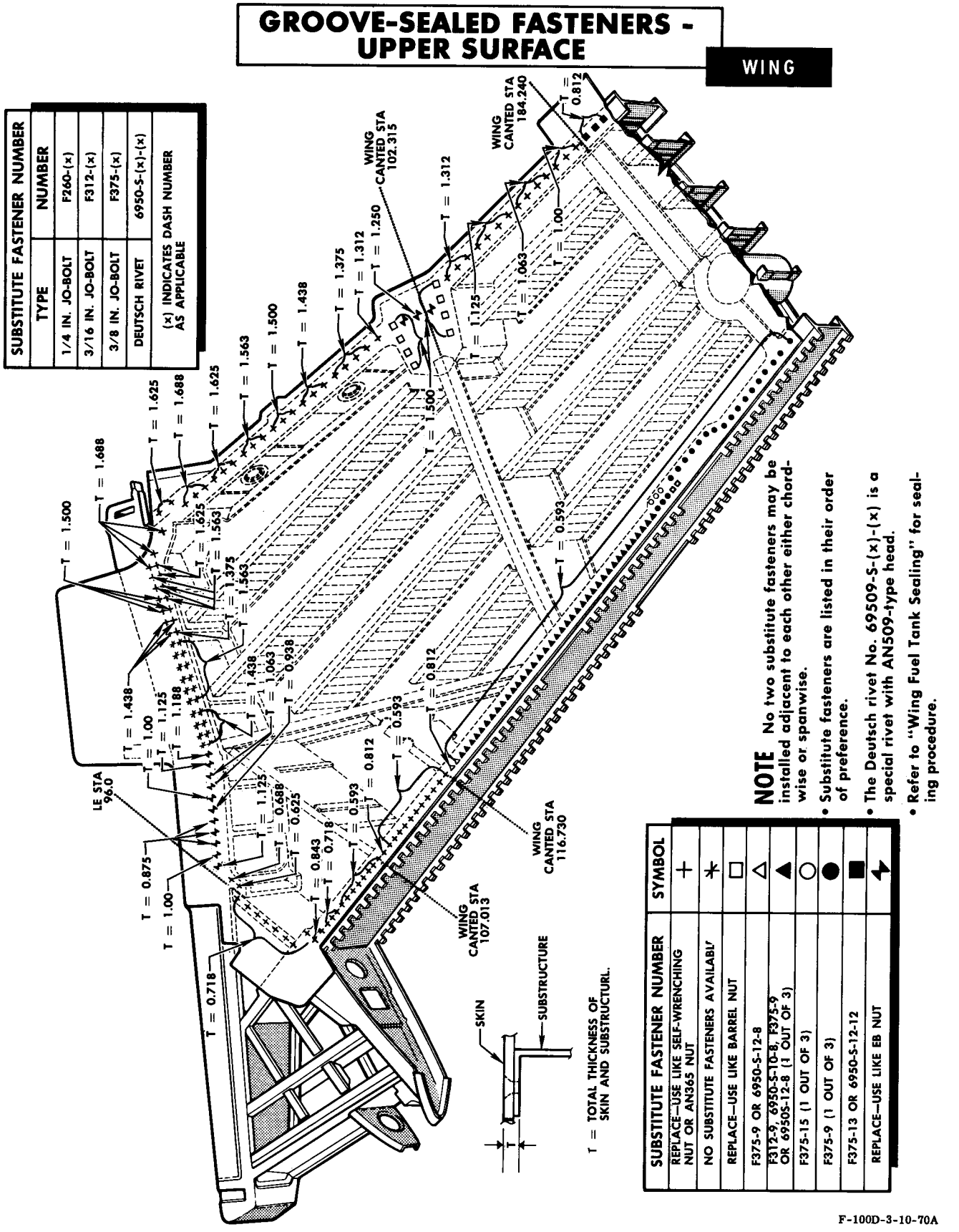


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 5 of 6)

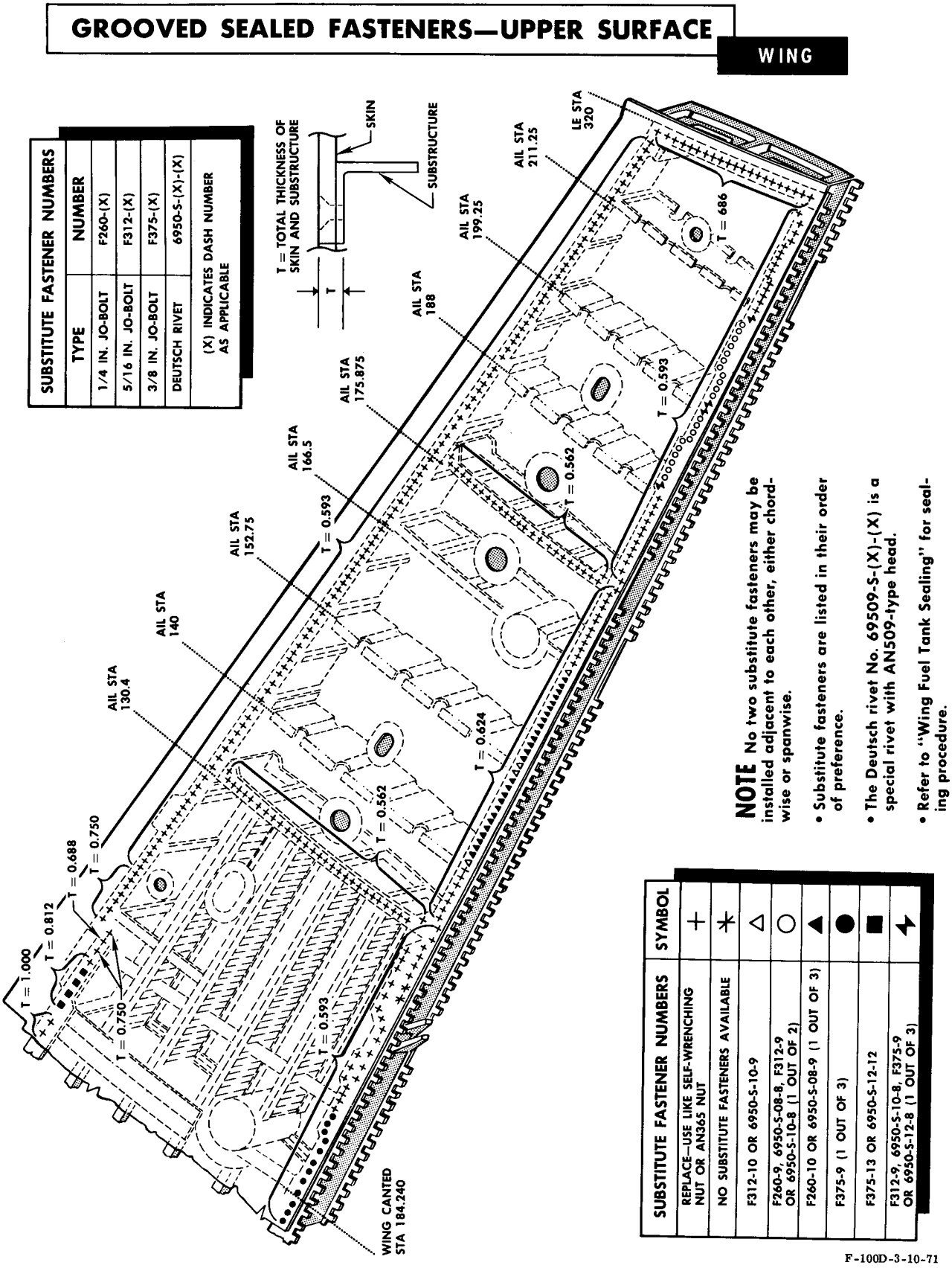


Figure 2-7. Sealed Fasteners—Upper Surface (Sheet 6 of 6)

SEALED FASTENERS - LOWER SURFACE

WING

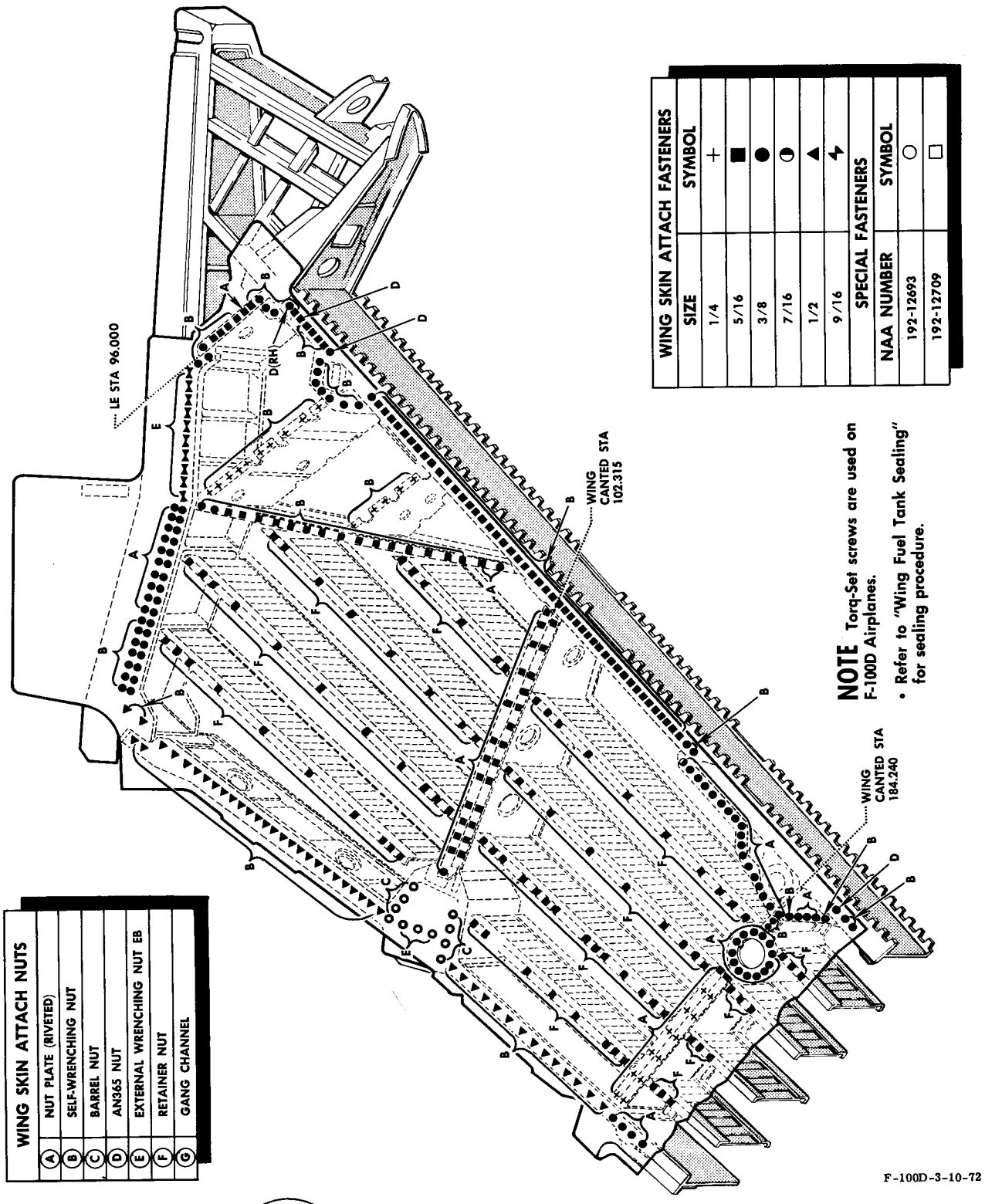


Figure 2-8. Sealed Fasteners—Lower Surface (Sheet 1 of 6)

See Rev-Op. Suppl. # 13 dtd. 11/17/67

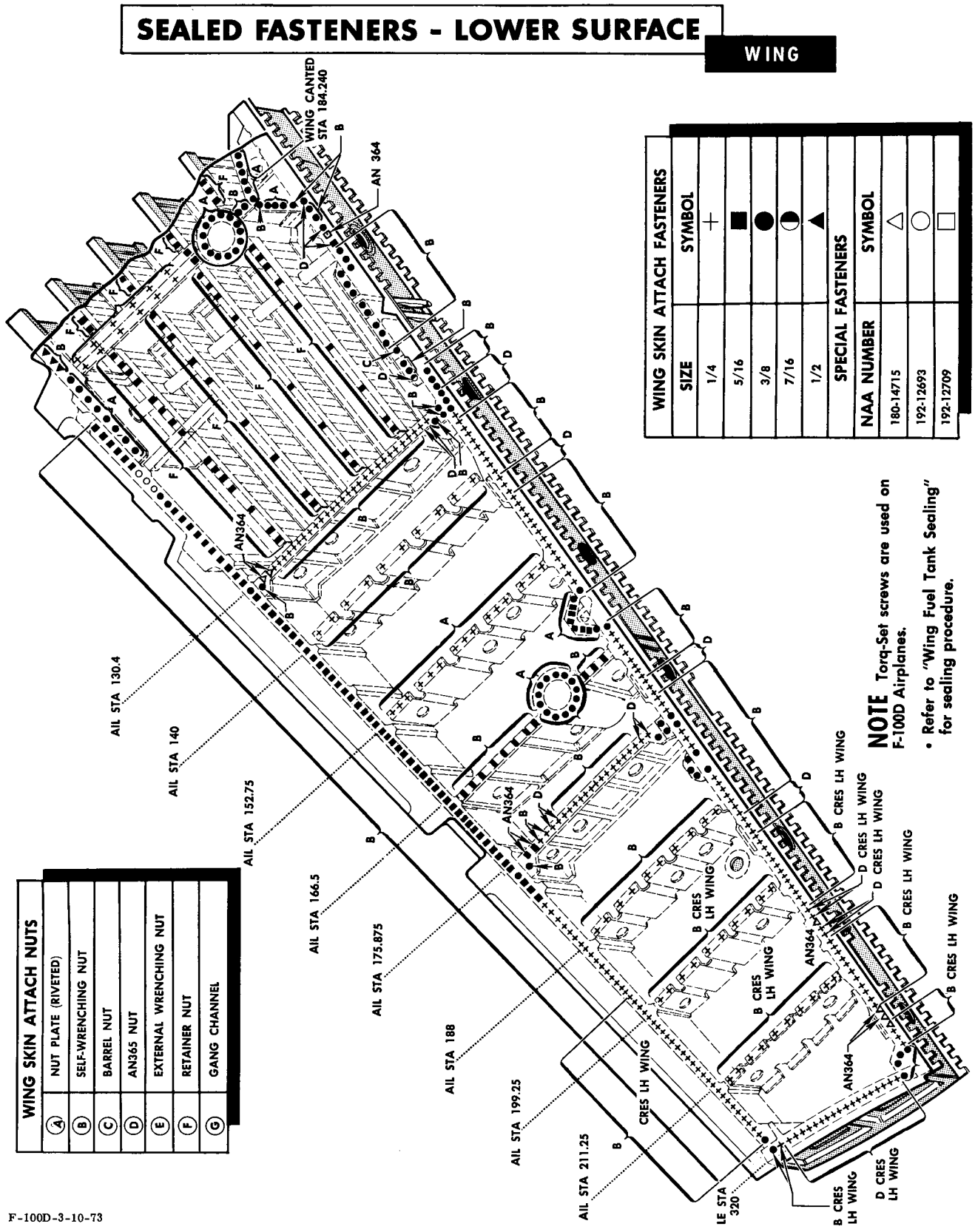


Figure 2-8. Sealed Fasteners—Lower Surface (Sheet 2 of 6)

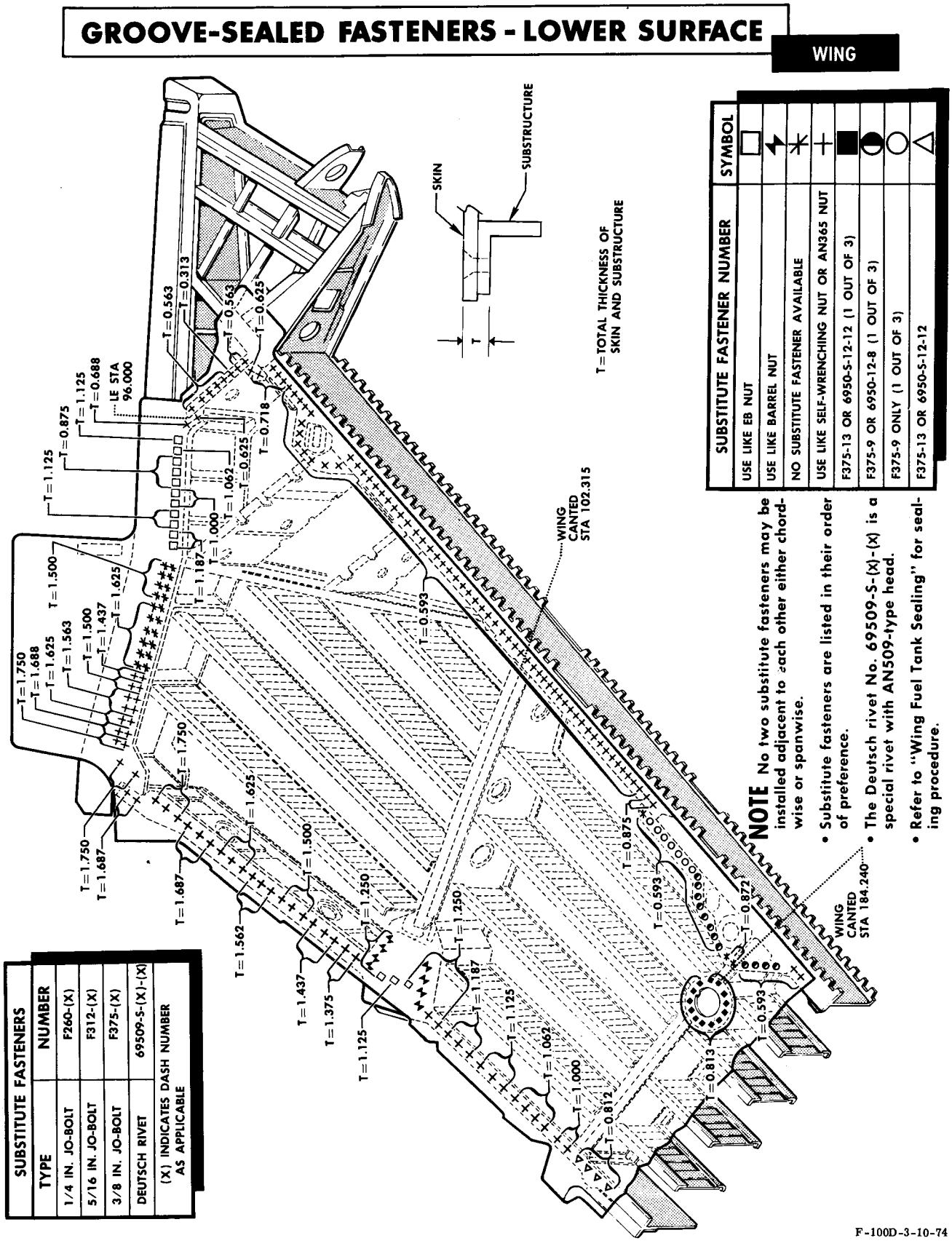


Figure 2-8. Sealed Fasteners—Lower Surface (Sheet 3 of 6)

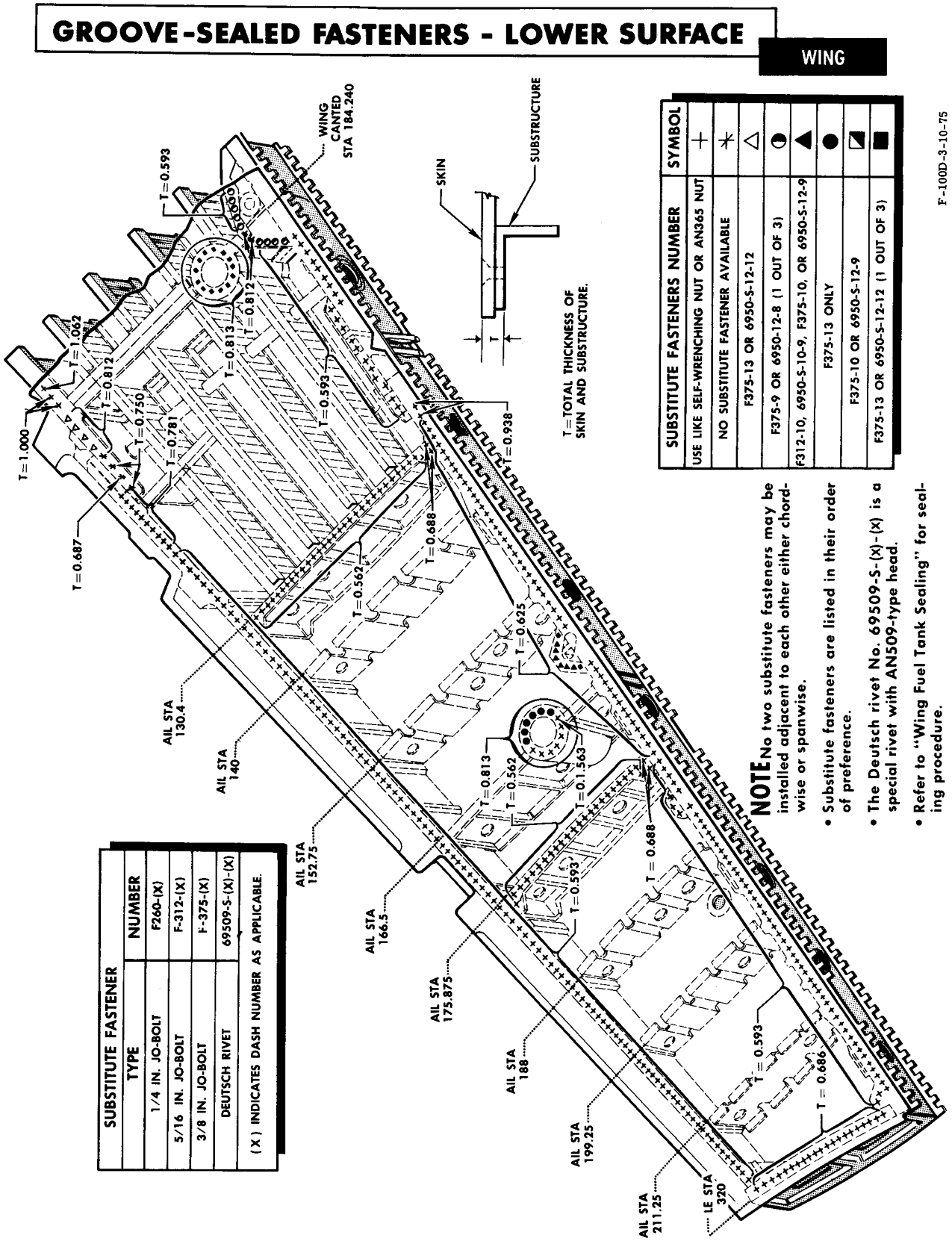


Figure 2-8. Sealed Fasteners—Lower Surface (Sheet 4 of 6)

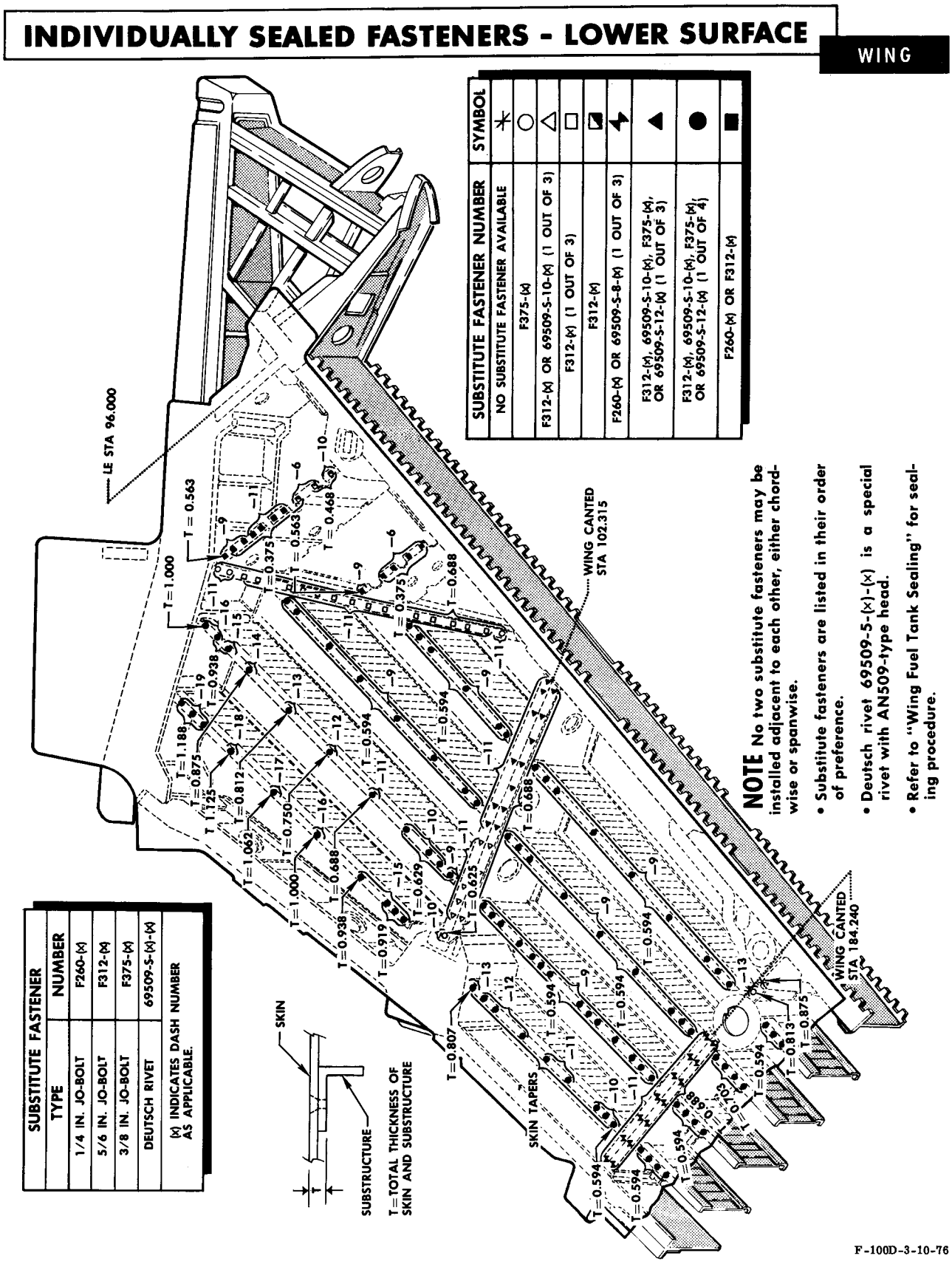
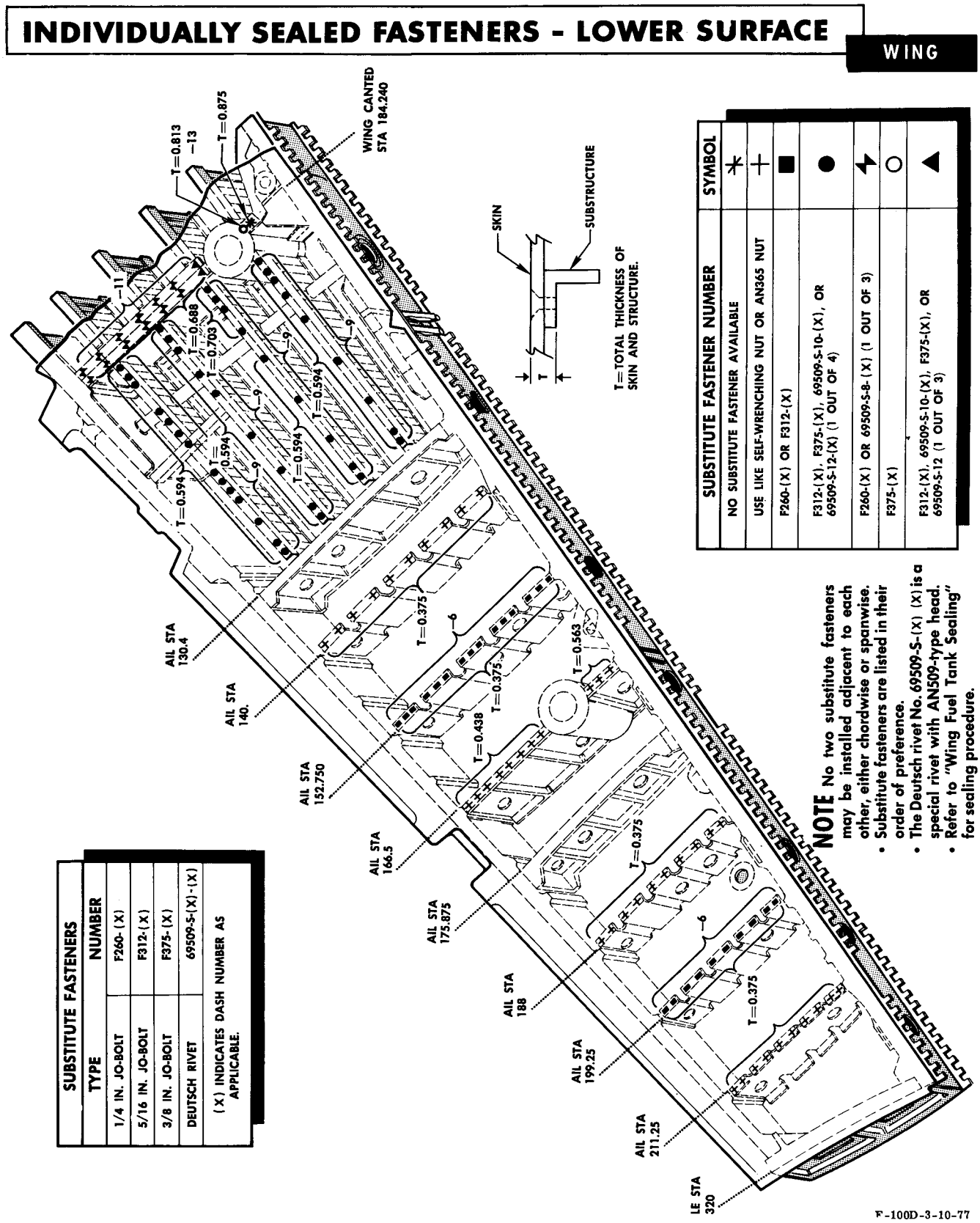


Figure 2-8. Sealed Fasteners—Lower Surface (Sheet 5 of 6)


INDIVIDUALLY SEALED FASTENERS - LOWER SURFACE
WING
WING CANTED STA 184.240
T=0.813 -13
T=0.875
-11
T=0.688
T=0.703
T=0.594
T=0.594
T=0.594

JO-BOLT IN FUEL SEALED AREA

WING

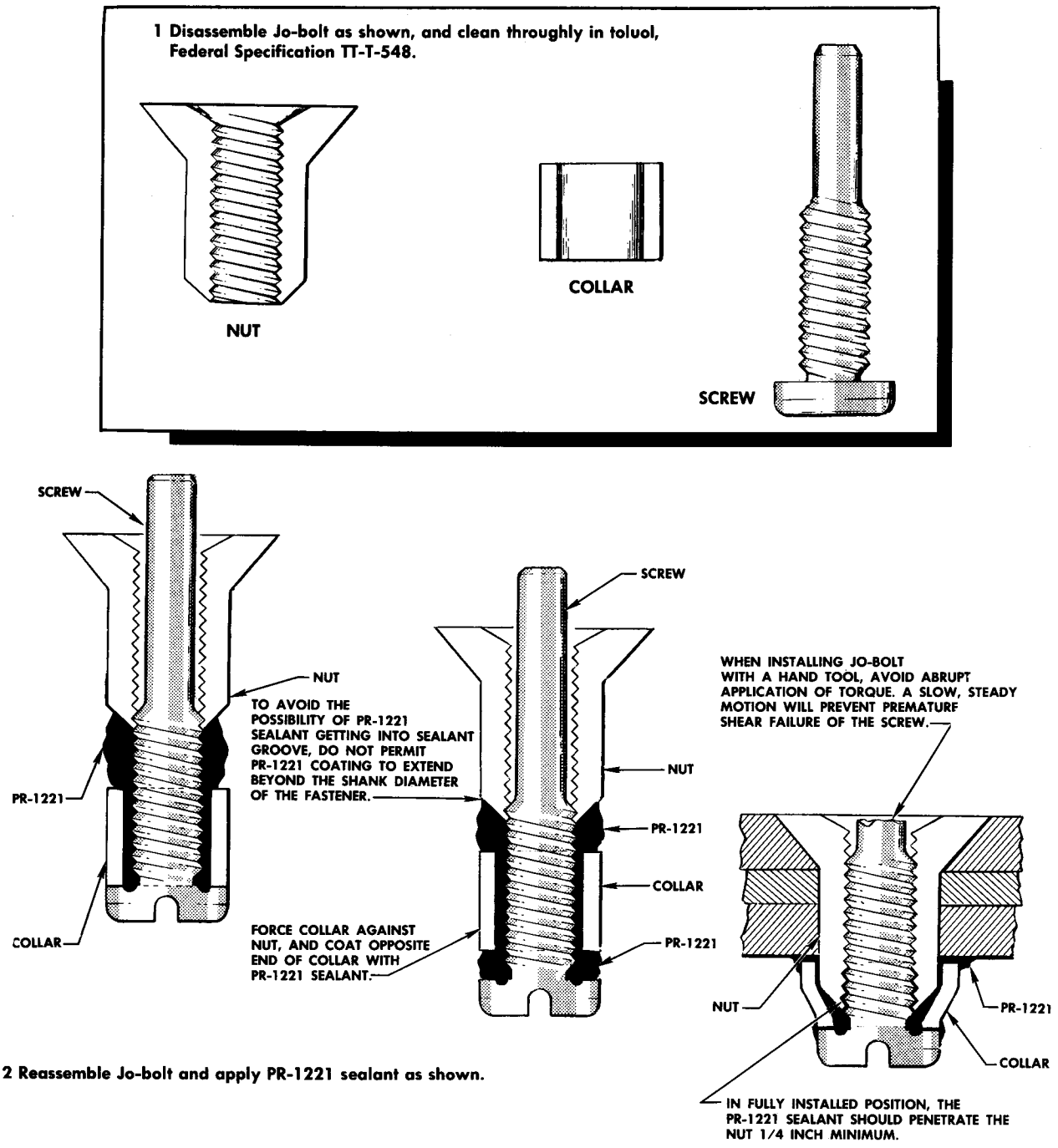


Figure 2-9. Jo-bolt Preparation and Installation in Fuel Sealed Area

FASTENER LAYOUT INSTRUCTIONS

WING

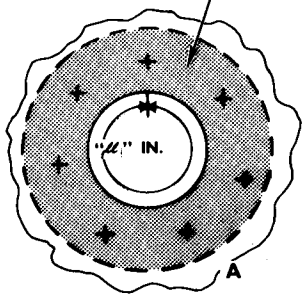
PROCEDURE

To determine the number of fasteners required on each side, measure damage "L." Multiply "L" by number of fasteners required for one inch of damage. (See chart.) This is the number of fasteners required for each side of "L" dimension (shaded areas). Substituting "ℓ" for "L" dimension, use same procedure to find number of fasteners required in unshaded area. For a repair on a skin edge, as in "B," the fasteners required for "ℓ" are installed on one side only (unshaded area).

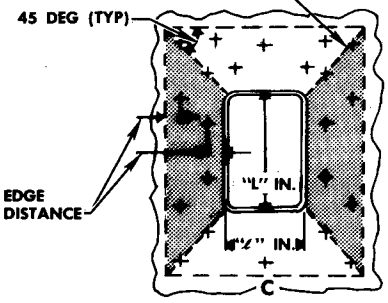
EXAMPLE

If "ℓ" is 3 inches and the number of fasteners per one-inch break is 2, then $3 \times 2 = 6$ fasteners are required per side. If "ℓ" is 1-1/2 inches and the number of fasteners required per one-inch break is 2, then $1\text{-}1/2 \times 2 = 3$ fasteners are required per side. The required number of fasteners for the repair should be placed on both sides of the measured distances except on a repair such as "B," where fasteners are on only one side of "ℓ" dimension. Use fastener spacing as shown on repair illustration, or use chart spacing as required.

FOR A CIRCULAR REPAIR AS IN "C," USE "$\mathcal{U}$" DIMENSION MULTIPLIED BY FASTENERS REQUIRED FOR ONE INCH OF DAMAGE. THE CIRCUMFERENCE OF CIRCLE "$\mathcal{U}$" IS FOUND BY MULTIPLYING 3.4 TIMES THE DIAMETER OF THE DAMAGED AREA. SPACE FASTENERS EQUALLY AROUND DAMAGED AREA.



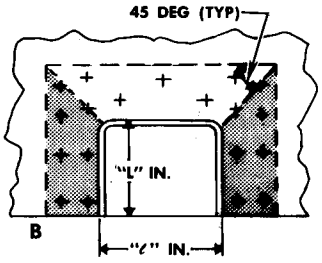
A FASTENER ON CORNER LINE IS COUNTED AS 1/2 FASTENER FOR EACH ADJACENT AREA WHEN DETERMINING FASTENER PATTERN.



NOTE The following fasteners may be used on repairs in accordance with the chart.

7S28-3 to 7S28-7
NAS333 to NAS337
HT3M to HT7M
F200 to F375 (Jo-bolt)

- Use ESNA2500 series dome nuts. Attach nut plates to doubler only, or use nuts with Stat-O-Seal washers.



DETERMINATION OF SIZE & NO. OF SCREWS

SKIN THICKNESS	FASTENERS PER ONE IN. BREAK					EDGE DISTANCE OF FASTENERS					MIN FASTENER SPACING					MAX FASTENER SPACING				
	3/16	1/4	5/16	3/8	7/16	3/16	1/4	5/16	3/8	7/16	3/16	1/4	5/16	3/8	7/16	3/16	1/4	5/16	3/8	7/16
0.064-0.079	1					9/16					15/16					1-13/32				
0.080-0.101		1					13/16					1-5/16					1-25/32			
0.102-0.124			1					1					1-5/8					2-1/4		
0.125-0.155				1					1-3/16					1-15/16					2-1/2	
0.156-0.174				1	1				1-3/16	1-5/16				1-15/16	2				↑	2-1/2
0.175-0.236				1-1/2	1				7/8	1-5/16				1-3/8	2					↓
0.237-0.261				1-1/2	1-1/2				7/8	7/8				1-3/8	1-7/16					
0.262-0.300				2	1-1/2				5/8	7/8				1	1 7/16					
0.301-0.374				2-1/2	2				9/16	3/4				7/8	1-1/8					
0.375-0.407				3	2				1/2	3/4				13/16	1-1/8					
0.408-0.450				3	2-1/2				↑	5/8				13/16	15/16					
0.451-0.500				3-1/2	2-1/2				↓	5/8				3/4	15/16					
0.501-0.523				3-1/2	3				1/2	9/16				↕	7/8					
0.524-0.600				4	3				7/16	↕				3/4	7/8					
0.601-0.610				4-1/2	3				3/8	9/16				5/8	7/8					
0.611-0.672				4-1/2	3-1/2				↑	1/2				↕	13/16					
0.673-0.710				5	↕				↓	1/2				↓	↕				↓	↓
0.748				5	3-1/2				3/8	1/2				5/8	13/16				2-1/2	2-1/2

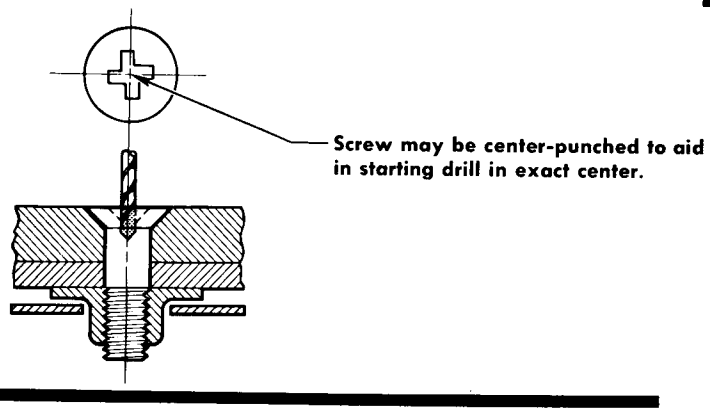
F-100D-3-10-133B

Figure 2-10. Fastener Layout Instructions

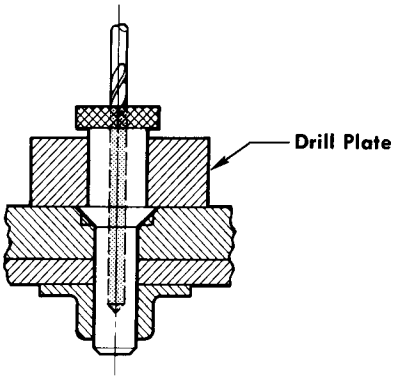
REMOVAL OF SCREW IN DAMAGED NUT PLATE

WING

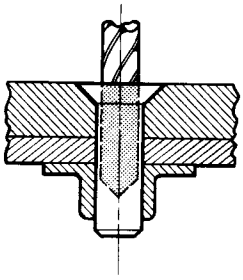
1 Start No. 30 hole at center of screw.



2 Use drill plate to ensure center drilling of screw. Drill to depth shown.

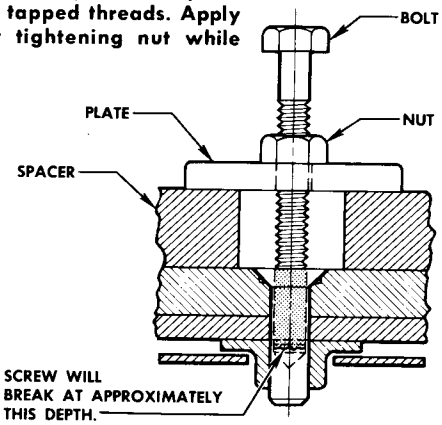


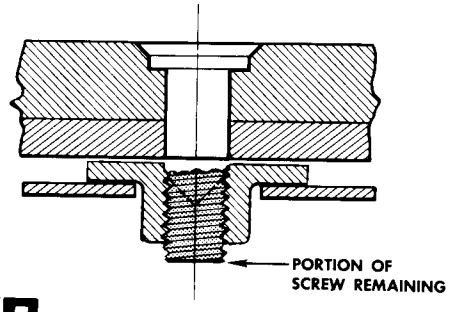
3 Drill hole to "A" diameter. Tap hole to depth as shown, using "B" size tap.



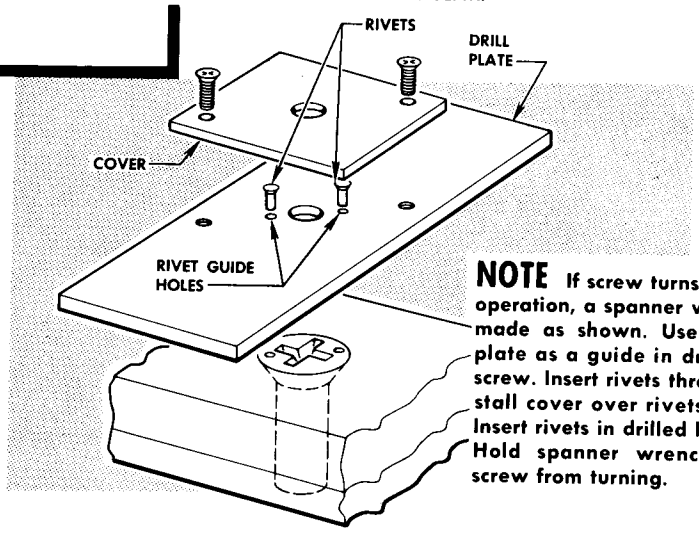
FASTENER SIZE	DRILL "A"	TAP "B"
1/4 IN.	No. 21 (0.159 IN.)	10–32
5/16 IN.	No. 3 (0.213 IN.)	1/4–28
3/8 IN.	No. 1 (0.272 IN.)	5/16–24

4 Install bolt through plate and spacer to bottom of tapped threads. Apply leverage by tightening nut while holding bolt.





5 After main portion of screw is removed, the remaining part may be drilled out or knocked out with the nut plate. Every effort should be made to remove any parts falling into fuel cell areas.



NOTE If screw turns during drilling operation, a spanner wrench may be made as shown. Use holes in drill plate as a guide in drilling holes in screw. Insert rivets through plate. Install cover over rivets with screws. Insert rivets in drilled holes in screw. Hold spanner wrench to prevent screw from turning.

F-100D-3-10-53

Figure 2-11. Removal of Screw in Damaged Nut Plate

2-14.) A second method used to seal access doors and some screws involves the use of rubber-type seals. "Stat-O-Seals," which are "O" rings bonded inside metal washers, are used to seal access door attaching screws and a few other screws. The access doors are sealed by especially shaped "O" rings bonded directly into the doors. These seals cannot be replaced. If door seals are damaged, the door must be replaced. Rivets are not used in the fuel area where they may become loose and cause leaks.

MATERIALS FOR FUEL TANK SEALING.

MATERIAL	SOURCE
Sealing compound (for injection)	PR-703 Products Research Co*
Sealing compound (for repairs)	PR-1221 (MIL-S-7502) Products Research Co*
Interior coating	Spec MIL-C-27725
Cleaning solvent	NT-S-1 Shell Oil Co, Inc*
Solvent—dry-cleaning	Federal Spec P-D-680
Toluol	Federal Spec TT-T-548

***Refer to index for source of commercial products.**

PREPARATION OF MATERIAL FOR FUEL TANK SEALING.

The sealing compound PR-703 is a nonhardening putty-like material which remains plastic over a wide range of temperatures. Injection can, with some difficulty, be accomplished at temperatures as low as 42°F; however, the most desirable temperature for injection is above 77°F and injection should be accomplished above this temperature if at all possible. The material is premixed and no further preparation is required before it is used. Sealing compound PR-1221 (Specification MIL-S-7502) is a fuel-resistant, curing-type material (brown in color) that is used to seal most repairs. This sealing compound is received in two parts: accelerator and base compound. The sealing compound is prepared for use by thoroughly blending one part, by weight, of accelerator with each ten parts of base compound. The compound may be considered properly blended when the accelerator is uniformly dispersed throughout the mixture (no further evidence of streaks or discoloration). After installation and cure of a sealed repair, the exposed PR-1221 surface should be coated with interior coating conforming to Specification MIL-C-27725.

A fuel-resistant interior coating (Specification MIL-C-27725) shall be sprayed or brushed on the interior surfaces of integral tanks to prevent corrosion. This coating shall be prepared and applied in accordance with procedures outlined in Technical Manual for Preparation, Inspection, and Repair of Aircraft Fuel, Oil, and Water-alcohol Cells and Integral Tanks, T.O. 1-1-3.

WARNING

Odorless vapors that are given off during application and curing of this material are extremely hazardous. Observe warning outlined in T.O. 1-1-3.

EQUIPMENT FOR FUEL TANK SEALING.

Two guns are available for the injection of the sealing compound (PR-703) into the sealant grooves. Both guns are manufactured by the Grover Smith Mfg Corp at San Gabriel, California. The Grover sealant injection gun No. 223 is air-operated and the Grover sealant injection gun No. 225 is hand-operated. The sealing compound is packed into the sealant chamber where pressure is applied to force the sealing compound into the pressure injection chamber. Pressure to force the sealing compound into the injection chamber is applied by an air piston on gun No. 223 and by spring pressure on gun No. 225. The spring on gun No. 225 is compressed by turning the "T" handle at the end of the sealant chamber. As sealing compound is used, the "T" handle must be screw in to maintain pressure against the spring and, in turn, the sealing compound. On gun No. 223 (air-operated), air pressure is directed against the high-pressure injection piston when the trigger is actuated. This piston then forces the sealing compound

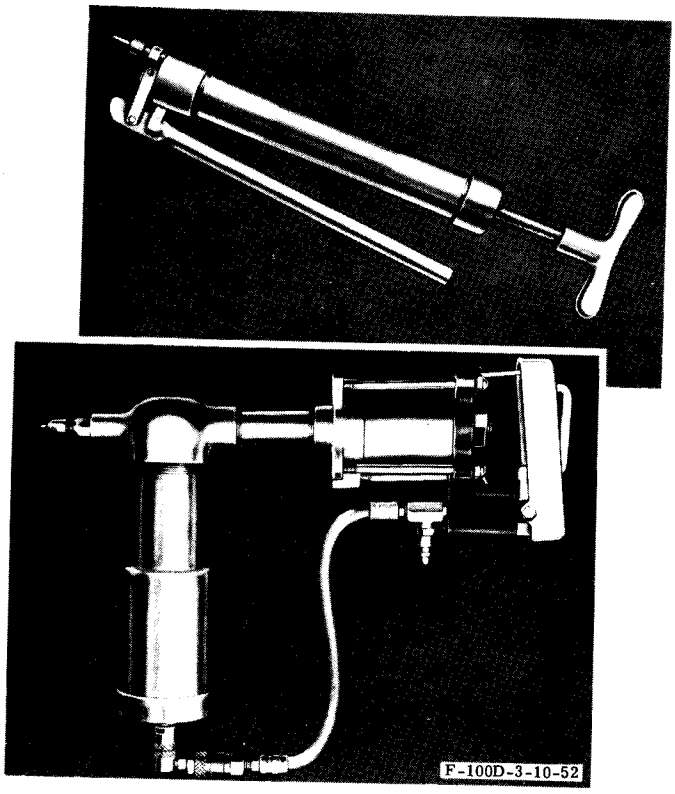


Figure 2-12. Sealant Injection Guns

through the nozzle into the sealant grooves. On gun No. 225 (hand-operated) the lever handle applies pressure to an injection piston which forces the sealing compound through the nozzle into the sealant grooves. To load sealant gun No. 223 (air-operated), proceed as follows: (Parts referred to are identified in figure 2-13.)

1. Remove loading chamber from gun barrel and push loading piston to its fully bottomed position.

2. Roll sealing compound (PR-703) into a cylindrical shape about 1¾ to 2 inches in diameter.

Note

Hands should be clean and oil-free before handling sealing compound.

3. Insert plug of sealing compound into loading chamber of gun. Tamp sealing compound in with a wooden stick until all air bubbles are forced out.

4. Reassemble loading chamber to gun barrel.

To load sealant gun No. 225 (hand-operated), proceed as follows: (Parts referred to are identified in figure 2-13.)

1. Back off "T" handle until spring tension is released and handle is fully extended.

2. Unscrew pressure head from loading chamber.

3. Roll sealing compound (PR-703) into a cylindrical shape about one inch in diameter.

4. Insert plug of sealing compound into barrel of gun. Tamp sealing compound in with a wooden stick until all air bubbles are forced out.

5. Reassemble pressure head to loading chamber.

6. Screw "T" handle in until spring tension is felt. As the sealing compound is forced out of the gun by operation, the "T" handle must be screwed in further to maintain spring tension.

SEALING PROCEDURE.

SEALING TANK GROOVES.

Sealing the tank grooves should be done with care to avoid future leaks in the wing area. After loading sealant gun (refer to index), proceed as follows:

1. Place the nozzle tip of the gun into the countersink of an injection hole. Hold the gun firmly in position and depress the trigger until the sealing compound is injected about half the distance to the adjacent injection holes. (See figure 2-14.) This will be accomplished when the plunger travel indicator has traveled about half of the total stroke on the Grover gun.

2. Move the gun tip to one of the adjoining injection holes. Hold gun firmly in place and depress the trigger until the sealing compound flows out of the hole previously injected for a minimum of ⅛ inch above the skin surface.

Note

A single burst of the gun will usually suffice for smaller fasteners. Several bursts may be required to seal around the larger fasteners or if the temperature is below 70°F. To decrease the time required for injection, it is permissible to heat the guns to a maximum of 110°F.

3. Repeat step 2. Continue injecting around the tank until all of the grooved areas have been injected.

4. Install PK79X2-6-4 injection hole screws in sealed area at any convenient time during the injection sequence.

5. After final installation of injection-hole screws, clean all excess groove-sealing compound off wing by scraping away most of it with a wood or plastic blade. The remainder of the sealing compound may be removed with toluol or cleaning solvent. Toluol should not be used on any surface coated with corrosion-resistant finish.

SEALING INDIVIDUALLY SEALED FASTENERS.

Individually sealed fasteners should be sealed as follows:

1. Remove all traces of oil or grease from the sealing

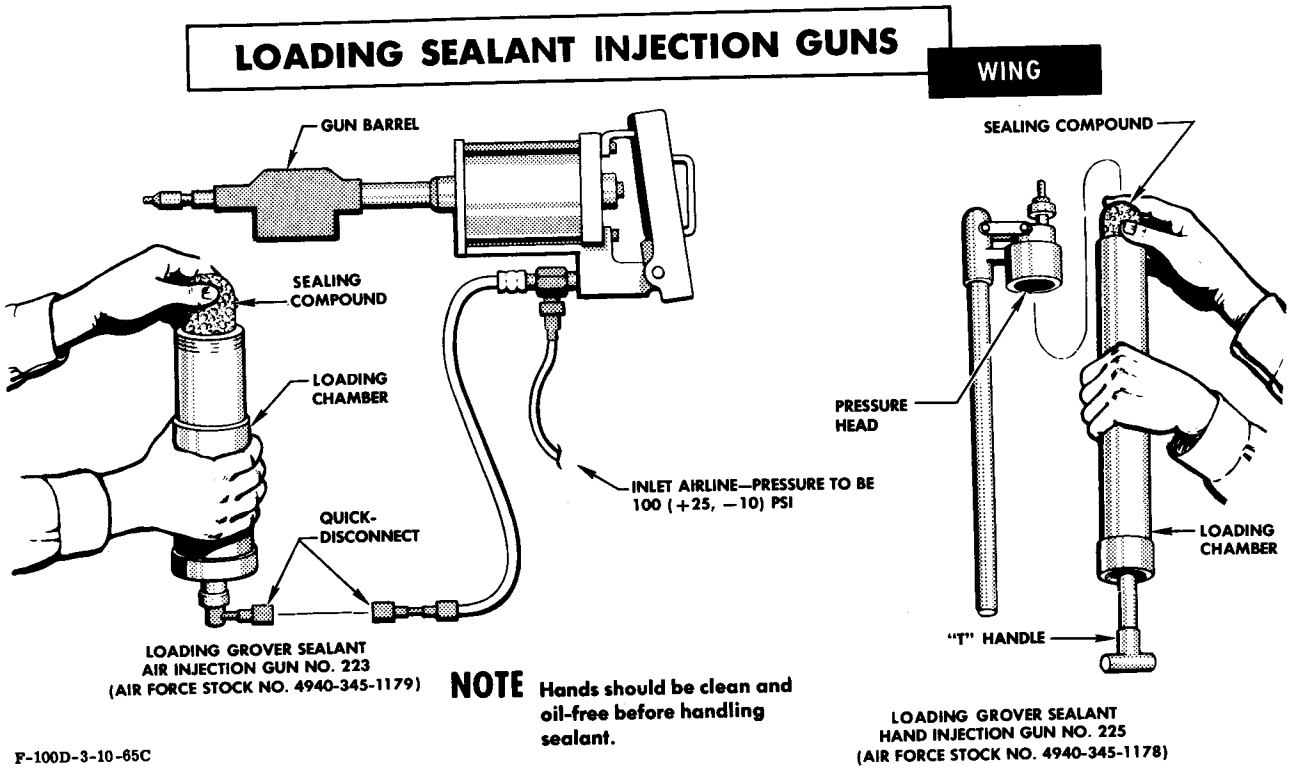


Figure 2-13. Loading Sealant Injection Guns

areas on the screws and from the screw holes by solvent-wiping.

2. Place a bead of groove-sealing compound around the shank of the screw below the tapered part of the screwhead. (See figure 2-14.) Do not coat the screw threads. The bead may be applied with the screw partly inserted in the hole or before it is inserted. Use an excess of sealing compound and tamp the ends of the bead together. Be sure the tool used to tamp the bead is oil-free. If the fingers are used to tamp the bead, they must be clean and free of grease and oil. After the screw is driven, the sealing compound must squeeze out completely around the screwhead. If squeeze-out does not occur, remove screw, clean and apply more sealing compound.

3. Clean the excess sealing compound off the wing surface by scraping with a wood or plastic blade. Remove remaining sealing compound with toluol or cleaning solvent. Toluol should not be used on any surface coated with corrosion-resistant finish.

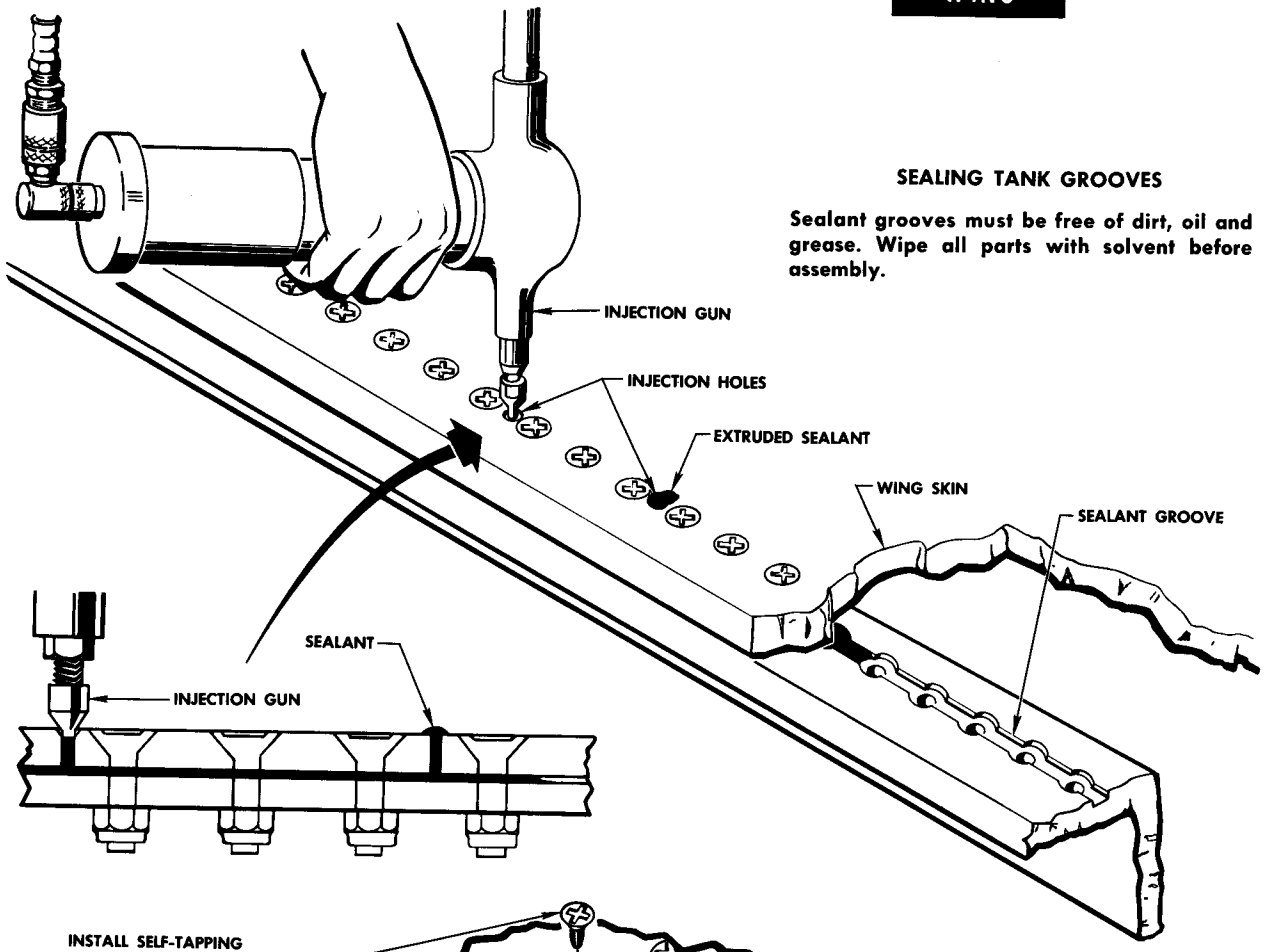
Fasteners sealed with "O" rings must be installed with new "O" rings. Roll or slide the unused "O" ring over the threaded portion of the screw and continue to move the "O" ring until it is against the screwhead. *Do not use petrolatum or lubricant of any type.* Use care not to cut or damage the "O" ring seals. Cut or damaged seals must be replaced with unused "O" ring seals. Install screw and torque to required value.

REWORK OF INDIVIDUALLY SEALED SCREWS.

Individually sealed screws which leak after having been installed and torqued according to proper procedures must be reworked. Inspection of the screw might show the threads bottoming in the nut or the screwhead not seating properly. Replacing the long screw with one of proper length should correct the first condition. Improper seating may be caused by an elongated countersink or a crooked hole. (See figure 2-15.) This condition should be corrected by reworking the hole. Rework should be done with a one-piece tool which cuts both the 100-degree countersink and the counterbore at the same time. If this tool is not available, separate countersink and counterbore is acceptable. Open tools (without depth stop) with pilots which will closely fit the existing holes must be used. If separate tools are used, the countersinking operation must be performed first, followed by the counterbore. Only the minimum amount of material necessary to seat the screwhead should be removed. The cut must be made with great care, backing out the tool frequently to observe results. A screwhead which is set deeper than 0.020 inch below the surface should be checked by the structural repair officer. (See figure 2-16.) On a properly seated screw, temporarily tightened to about 10 inch-pounds, it should not be possible to insert the edge of a 0.002-inch feeler gage anywhere under the head. Screws which are recessed deeper than 0.015-inch below the surface should be filled with aerodynamic filler. (Refer to index.)

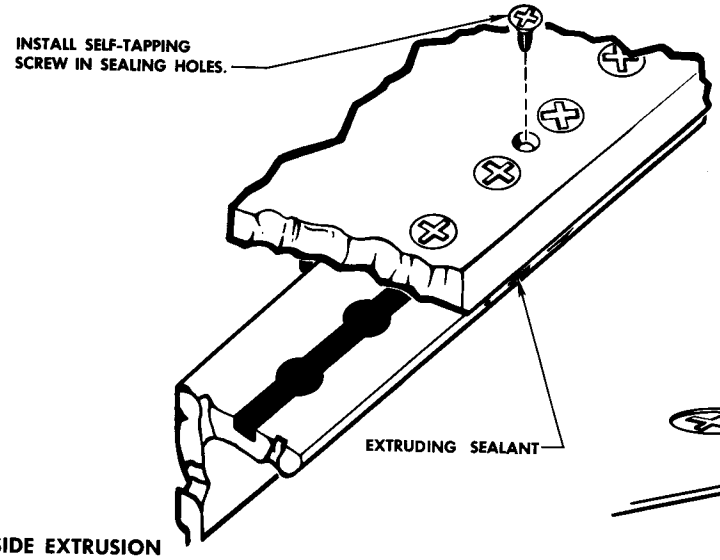
SEALING INTEGRAL FUEL TANKS

WING



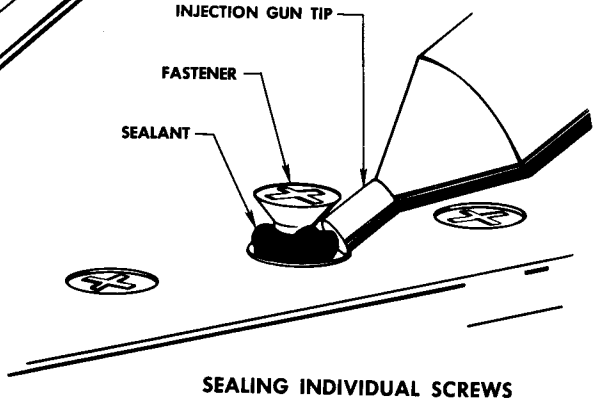
SEALING TANK GROOVES

Sealant grooves must be free of dirt, oil and grease. Wipe all parts with solvent before assembly.



SIDE EXTRUSION

Side extrusion is not permissible. Loose bolts, poor metal-to-metal fit, or the attempt to force sealant through too long a portion or groove will cause side extrusion.



SEALING INDIVIDUAL SCREWS

Screws must be free from dirt and oil. Wipe with solvent before installation. Do not roll sealant in fingers to apply to screws. After applying bead completely around screw, tamp ends of bead together.

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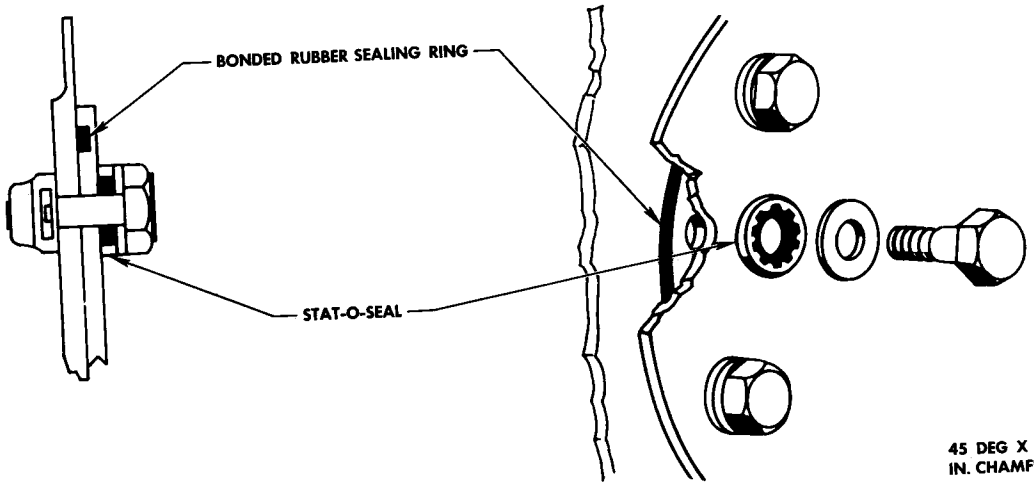
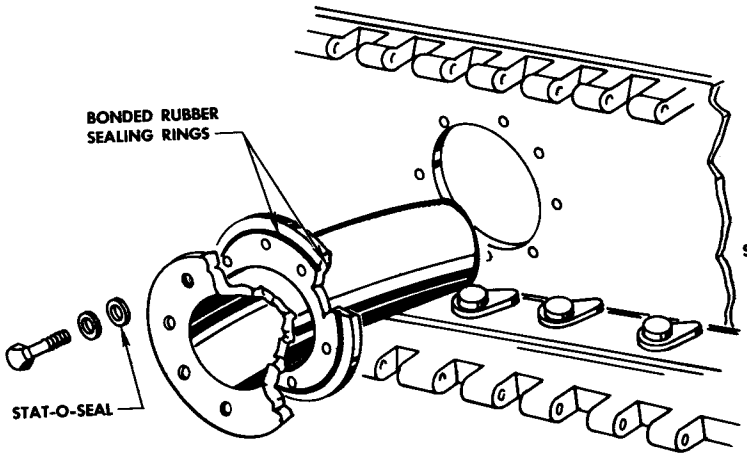
Figure 2-14. Sealing Integral Fuel Tanks (Sheet 1 of 2)

SEALING INTEGRAL FUEL TANKS

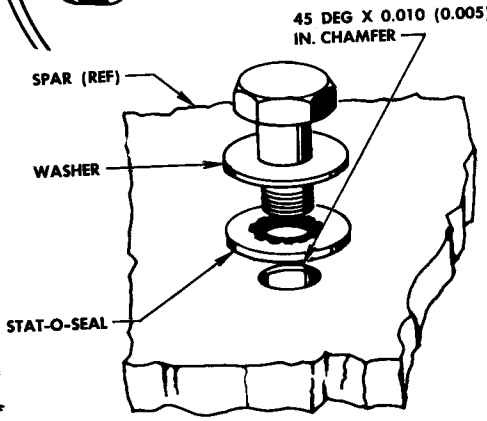
WING





BONDED RUBBER SEALING RING INSTALLATION

Surface must be clean and free of nicks and scratches. Replace ring after several removals, or if seals show any evidence of wear or damage.



STAT-O-SEAL INSTALLATION

Fasteners should not be turned on rubber-type seals. A 1/16-inch washer should always be used between a bolthead and Stat-O-Seals. Special washers are provided for countersunk screws. Fasteners should be torqued from nut side whenever possible.

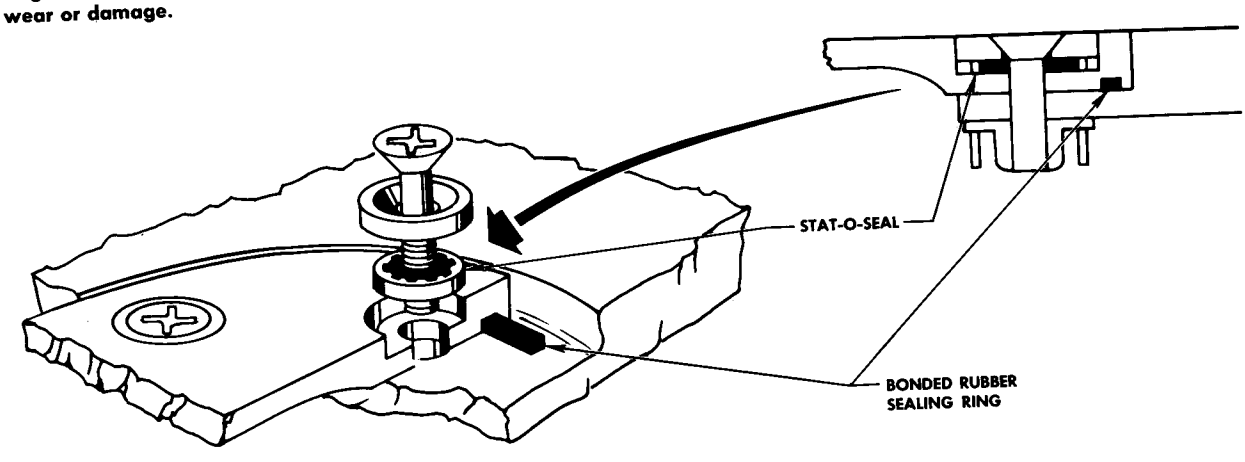


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Figure 2-14. Sealing Integral Fuel Tanks (Sheet 2 of 2)

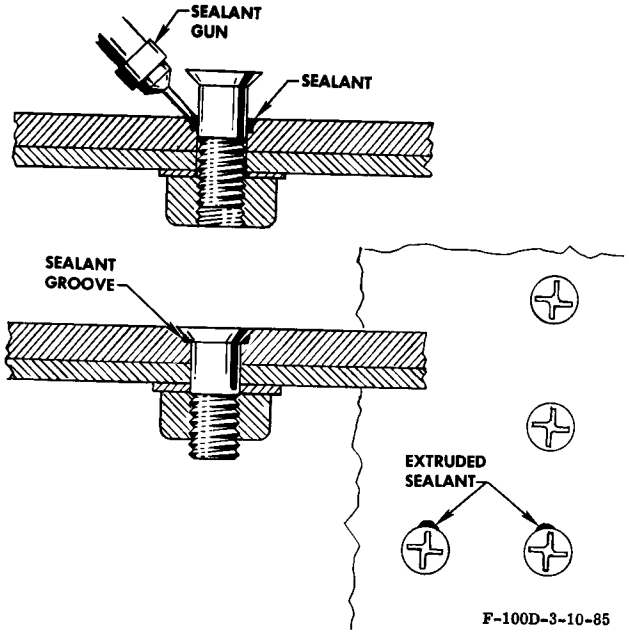


Figure 2-15. Extruded Sealant Indicating Poor Seal

REWORK OF INDIVIDUALLY SEALED SCREWS—F-100F-20 AIRPLANES.

All individually sealed screws on the wing of F-100F-20 Airplanes are sealed with "O" rings. The "O" rings may be replaced in the following manner:

1. Partially back out affected screw. Do not completely disengage screw from nut. This is especially true where bumper nuts are used, since the nut may drop into the tank.

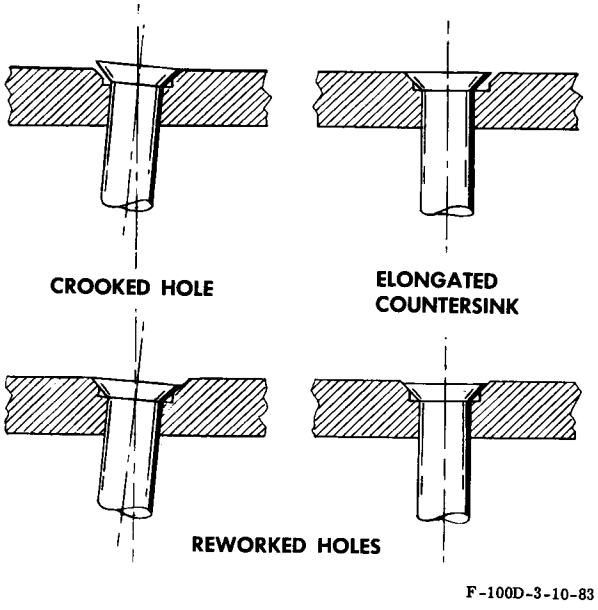


Figure 2-16. Reworking Individually Sealed Holes

2. Remove old or leaking "O" ring by cutting it with a knife or by stretching it over screw head.

3. Make sure countersink and counterbore area in skin is thoroughly clean and absolutely free of all chips and foreign matter.

4. Install a new "O" ring seal by stretching it very carefully over screw head and positioning it on screw shank adjacent to screw head.

5. Tighten screw and torque to required torque value.

CAUTION

- Do not use petrolatum or lubricant of any type.
- Use caution not to cut or damage "O" ring seal. Damaged "O" ring seals must be replaced with unused "O" ring seals.

WING FUEL TANK LEAK OBSTRUCTION TESTS.

CAUTION

Because of the highly detrimental effect of mercury on metals used in airplane systems, instruments (manometers, thermometers, etc) containing mercury are prohibited from use in testing fuel systems.

The following procedure can be used to detect leaks in the sealed area of the wing when the wing is removed from the airplane. For leak-checks with the wing installed on the airplane, refer to Maintenance Handbook "F-100D Fuel and Utility Systems," T.O. 1F-100D-2-2. The leak check is best accomplished by securely closing all openings using fittings such as shown in figure 2-18, and applying air pressure inside the tank. Attach the air supply source through a suitable regulator and filter to fitting No. 4 (see figure 2-18), located on single-point refueling line. Proceed as follows:

1. Apply air pressure at single-point refueling line of 20 psi maximum.

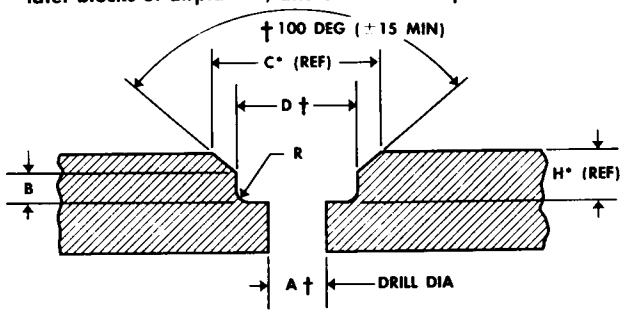
2. Brush castile soap solution (Specification MIL-S-4282) on all sealing surfaces, individually sealed fasteners, access doors and all fittings. Watch for any leaks as indicated by bubbles in soap solution.

3. Reseal leaking areas until all leaks are stopped. Leaks in groove-sealed areas can be stopped by removing the injection hole screws in leaking area and reinjecting sealing compound until leak stops. Individually sealed fasteners may be backed partially out, cleaned and resealed as necessary to stop leaks. Access doors that leak may be retorqued to stop leaks. If retorquing does not

DIMENSIONS FOR SEALED FASTENERS

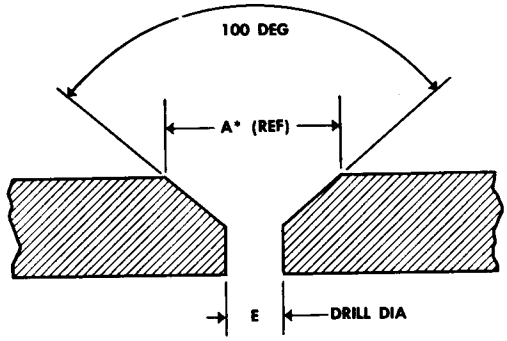
WING

INDIVIDUALLY SEALED SCREW RECESS DIMENSIONS
F-100D-1 Airplanes AF54-2126 thru -2132, and F-100D-5 and later blocks of airplanes , and all F-100F airplanes.



SCREW SIZE	A † DRILL DIA +0.004, −0.001	D † ±0.005	R ±0.005	B ±0.002	C* REF ONLY	H* REF ONLY
1/4	0.250	0.415	0.031	0.090	0.499	0.125
5/16	0.312	0.500	0.062	0.097	0.626	0.149
3/8	0.375	0.562	0.062	0.097	0.752	0.176

GROOVE SEALED SCREW RECESS DIMENSIONS.

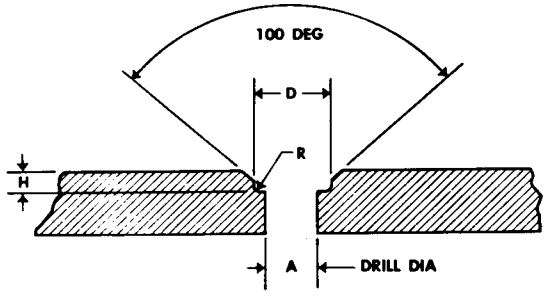


SCREW SIZE	DRILL DIA +0.002, −0.001	E* REF ONLY
1/4	0.250	0.499
5/16	0.312	0.626
3/8	0.375	0.752
7/16	0.4375	0.879
1/2	0.500	1.005
9/16	0.5625	1.132

*These dimensions are for reference only. Screw to be flush within 0.005 in. Above and 0.002 in. below surface.

†These dimensions are to be concentric within 0.003 inch of total indicator reading.

INDIVIDUALLY SEALED SCREW RECESS DIMENSIONS—
F-100D-1 Airplanes AF54-2121 thru -2125



SCREW SIZE	A DRILL DIA +0.004, −0.001	D ±0.010	R ±0.005	H ±0.010
1/4	0.250	0.415	0.031	0.100
5/16	0.312	0.500	0.062	0.120
3/8	0.375	0.625	0.062	0.140

F-100D-3-10-51B

Figure 2-17. Dimensions for Sealed Fastener Holes

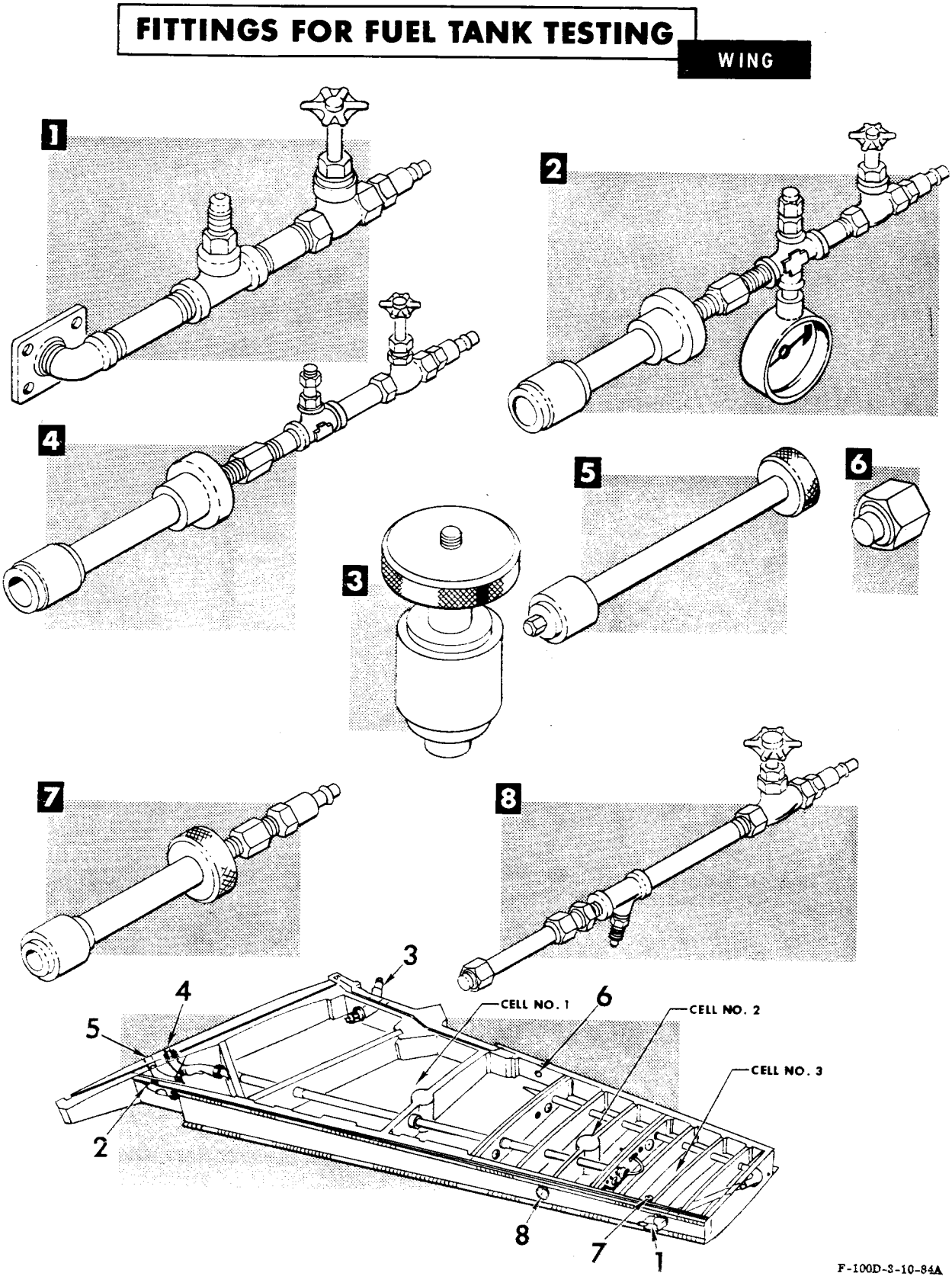


Figure 2-18. Fittings for Wing Fuel Tank Testing

stop leak, remove access door and check for cracks, tears, or damage to seal. Damaged access doors must be replaced.

4. After all leaks are stopped, repressurize the tank to 20 psi and let the tank stand for 5 minutes. If the pressure does not drop 2 psi, continue test procedure. If pressure does drop more than 2 psi, repeat steps 2 and 3 until rate of pressure drop is less than 2 psi per 5 minutes.

Use the following procedure to ensure that fuel lines are unobstructed, that the check valves seal properly, and that cell bulkheads are free of leaks. (Refer to figure 2-18.)

1. Connect air pressure of 10 psi maximum to fitting No. 2. Check that air flows from fuel transfer line at fitting No. 1.

2. Remove pressure from fitting and connect it to the vent line at fitting No. 7. Check that air flows from the drains of all three cells. Airflow from these points indicates that the check valve in the vent line and the check valves in the bulkheads are unobstructed. Remove pressure source from fitting.

3. Connect air pressure of 10 psi maximum at single-point line fitting (No. 4). Open drain valve of cell No. 3. Air should blow from the open drain valve, indicating that the single-point line has no obstructions. Remove pressure line from fitting.

4. Remove cover plate of cell No. 2 nearest the bulkhead between cells No. 1 and No. 2 and insert a suitable plug in the forward flapper valve hole. (This flapper valve has a small hole drilled in the body that will allow air to pass from cell No. 1 to cell No. 2 if not plugged.)

5. Connect air pressure line and air manometer to fitting No. 2. Apply air pressure to cell No. 1 until the pressure in the cell reaches 10 psi as read on the manometer. Close off the air pressure source and observe the manometer for a period of five minutes. Ensure that the pressure in cell No. 1 does not drop more than five inches of water. This test is to determine that the bulkheads are properly sealed, that check valves close and seal properly, and that "O" ring seals on lines and fuel-tight electrical connectors seal properly. If air pressure leakage rate is more than 5 inches of water in 5 minutes, listen carefully at the opening in cell No. 2. Any leaks will usually make a hissing sound. Put hand into the cell and feel along the bulkhead. A draft can be felt many times. Small leaks can sometimes be found by applying soap solution along the suspected areas. Correct the leaking condition by resealing as necessary. When leakage rate is acceptable, remove plug from flapper valve hole and reinstall cover plate on cell No. 2.

6. Apply air pressure to cell No. 2 through fitting No. 8, with manometer also attached to fitting. Repeat step 5 procedure on cell No. 2. Air should not leak into cell No. 3. Leakage rate for cell No. 2 is 7 inches of water in 5 minutes. When intercell leakage rate is acceptable, proceed with tests.

7. With air pressure source and manometer still connected at fitting No. 8, connect another air source and manometer to fitting No. 4, apply air pressure of 10 psi to fitting No. 8, and air pressure of 25 psi to fitting No. 4. Allow the pressures to stabilize; then observe the 25 psi manometer for a period of 5 minutes. Ensure that during the 5 minute period the pressure in the refueling line does not decrease by more than four inches of water as shown by manometer. This check ensures that the pressure valve seals properly when a pressure of 25 psi is applied to the refueling line and a pressure of 10 psi is applied to cell No. 2.

8. Disconnect air pressure sources and manometers from fittings. Remove fittings from wing and cover all openings to prevent entry of foreign material.

WING REPAIRS.

WING REPAIR INDEX.

See figure 2-19.

INSTALLING OVERSIZE TAPER-LOK FASTENERS—AIRPLANES CHANGED BY T.O. 1F-100-1035.

In the event that oversize Taper-Lok fasteners must be installed in a predrilled Taper-Lok hole, the following procedure must be used:

1. Determine from the following table, the part number of oversize Taper-Lok fastener and reamer to be used.

Note

Use smallest diameter oversize Taper-Lok fastener available that will fall within required tolerances.

OVERSIZE TAPER-LOK FASTENERS AND REAMERS

EXISTING TAPER-LOK	OVERSIZE TAPER-LOK NO.	OVERSIZE REAMER NO.
TL 400-7-33	1st oversize—TL 410-7-33	TLD 2060BR-4-7
	2nd oversize—TL 430-7-33	TLD 2060BR-5-7
	3rd oversize—TL 440-7-33	TLD 2060BR-6-7

EXISTING TAPER-LOK	OVERSIZE TAPER-LOK NO.	OVERSIZE REAMER NO.
TL 400-8-34	1st oversize—TL 410-8-34	TLD 2060BR-4-8
	2nd oversize—TL 430-8-34	TLD 2060BR-5-8
	3rd oversize—TL 440-8-34	TLD 2060BR-6-8
TL 400-8-38	1st oversize—TL 410-8-38	TLD 2060BR-4-8
	2nd oversize—TL 430-8-38	TLD 2060BR-5-8
	3rd oversize—TL 440-8-38	TLD 2060BR-6-8

2. Install selected oversize reamer in a long-travel microstop countersink (Briles Mfg No. TLA 4900-38M or equivalent).

3. Place microstop countersink in a drill motor with a spindle speed of approximately 250 rpm (Rockwell Corp No. 41D or equivalent).

4. Set stop on countersink for an extremely light cut. Insert reamer in fastener hole and make first cut. After each cut, remove all chips and shavings from hole, insert oversize Taper-Lok in hole, tap lightly with a wooden or plastic mallet to seat fastener, and check for fastener head protrusion as indicated in step 5. Continue reaming and checking until hole is within the tolerances given in step 5.

5. Check fastener head protrusion for the proper distance between lower or inner face of fastener head and skin surface. The correct protrusion for a 7/16-inch Taper-Lok fastener is from 0.130 (minimum) to 0.288 (maximum) inch. The proper protrusion for a 1/2-inch Taper-Lok fastener is from 0.144 (minimum) to 0.317 (maximum) inches. A go no-go head protrusion gage may be used if available (Briles Mfg No. TLG 5130-1).

6. When correct fastener head protrusion has been achieved, remove fastener and lightly coat entire shank of fastener with Prussian blue dye, Federal Stock No. 8010-247-8706. Insert fastener in hole and tap lightly with a wooden or plastic mallet; then remove fastener. Check hole to determine that a dye transfer has occurred over at least 80 percent of hole surface. If an 80 percent dye transfer is not achieved, remove all dye from hole and take another very light cut with reamer. Check that fastener head protrusion is still acceptable and repeat dye check.

7. If fastener hole has acceptable bearing, remove all traces of Prussian blue dye from fastener hole, using a swab dipped in toluene (Federal Specification TT-T-548). The fastener hole should be clean and bright with no traces of burrs, scratches, or rifling, and with a 0.040 (+0.010, −0.000) inch radius on the surface adjacent to the fastener head.

8. Apply a coating of zinc-chromate primer (Specification MIL-P-8585) to surface of skin around Taper-Lok fastener hole. Do not allow primer to enter fastener hole.

9. Install a new Taper-Lok fastener in hole. Do not use Taper-Lok fastener used for dye check if another fastener of same size and type is available. The Prussian blue dye damages the cetyl-alcohol finish applied to the fastener by the manufacturer. If another Taper-Lok fastener is not available, remove all traces of Prussian blue dye with toluene and inspect fastener for damage. Dip cleaned fastener in a prepared solution of one pound of hexadecanol dissolved in one gallon of JP-4 fuel and allow to drip dry; then install fastener in prepared fastener hole.

10. Install new Washernuts on Taper-Lok fasteners. Use TLN 1010-7 on 7/16-inch Taper-Lok fasteners and TLN 1010-8 on 1/2-inch Taper-Lok fasteners. A TLN 1500-7 nut with an MS20002-7 washer may be substituted for the TLN 1010-7 Washernut, and a TLN 1500-8 nut and a MS2002-8 washer may be substituted for the TLN 1010-8 Washernut.

11. Torque 7/16-inch nuts to 550-590 inch-pounds. Torque 1/2-inch nuts to 770-830 inch-pounds.

Note

A click-type or snap-type torque wrench is advised because of the inaccessibility of the installation.

12. Check fastener thread protrusion through torqued nut. Minimum acceptable thread protrusion is one full thread; maximum thread protrusion is two full threads. If thread protrusion is below minimum, the next longer grip fastener must be installed. If the thread protrusion is above maximum limits, a maximum of three steel washers may be installed under the nut to bring the thread protrusion in tolerance.

13. Check that Taper-Lok head is seated against skin. If head is not seated, check that highest permissible torque is applied. Do not overtorque.

WING FRONT SPAR UPPER CAP AND SKIN REPAIR.

It is necessary to remove the upper skin panel of the wing to gain access to make this repair. (See figure 2-29.) Since this repair is in the sealed area, it must be made with care in order to assure a good fuel-tight seal. Proper fit of the repair parts is very important as large gaps will lead to leaks under the structural loads. As repair parts are assembled, apply a thin coat of PR-1221 sealing compound between all parts and apply a fillet of the same material around all repair edges in wet area. After curing, apply fuel-resistant coating over exposed PR-1221 sealant. Bolts used in the repair should be installed with the nuts forward and Stat-O-Seal washers under all nuts. The rubber of the Stat-O-Seal must not touch the bolt threads after installation. Bolt retainers should be used where needed to permit later replacement of Stat-O-Seals from outside. The minimum fastener edge distance is two times the fastener diameter except as noted on the illustration. The AN426AD4 rivets on each side of the sealant groove, in the gaps of the repair parts, are installed to aid in making a fuel-tight seal. The rivets on the spar should be flush in the upper surface while the rivets in the skin should be flush on both surfaces.

SKIN SPLICE—SLATS.

Cut slat assembly skins for splice shown in figure 2-23B as follows:

Part No. 192-17101-1 and -2—Cut upper and/or lower skin through center of rib bay between station 101.901 and 108.277.

Part No. 192-17201-1 and -2, 192-17202-1 and -2—Cut upper and/or lower skin through center of any rib bay except slat track attaching bays. Part No. 192-17203-1 and -2—Cut upper skin through center of any rib bay except slat track attaching bays. Cut lower skin through center of any rib bay between station 173.101 and 199.761.

Part No. 192-17204-1, -2, -101, -102, -201, and -202—Cut upper skin through center of any rib bay between station 224.385 and 251.049. Cut lower skin through center of any rib bay except slat track bays.

Part No. 192-17205-1 and -2—Cut upper and/or lower skin through center of any rib bay between station 274.651 and 298.295.

SEALING SURFACE AT ACCESS DOOR REPAIR.

This repair is to be used on either recessed or flat doors on the spar only. (See figure 2-33.) Damage not closer than one fastener diameter to a bolt hole may have a depth of 20 percent of the thickness at the door seal area, but must not exceed 0.100 inch. Damage closer than one fastener diameter to a bolt hole may have a depth of 10 percent of the thickness at the door seal area, but must not exceed 0.050 inch. After cleaning up the damaged area and removing all sharp edges, fabricate the doubler. The doubler is made from 2024-T4 or 7075-T6 0.064-inch sheet. The doubler outside diameter should be as near the door diameter as possible, but must clear the radius of the recess in the spar. The inside diameter of the doubler should be the same size

as the access hole. Bolt holes are drilled in the doubler to match the holes in the access door. The surface finish on the doubler must be the same as the finish on the door seal area. This is required to assure the rubber seal on the access door seating properly. Upon assembly of repair, seal doubler by applying PR-1221 sealant between doubler and door sealing area of spar. Apply fillet of PR-1221 sealant around edges of doubler after installation. All exposed PR-1221 must be overcoated with fuel-resistant coating. (Refer to "Materials for Fuel Tank Sealing.")

WING SKIN AT INTERMEDIATE SPAR REPAIR.

To gain access for this repair, it is necessary to cut out damaged skin large enough to insert repair doubler on both sides of spar. (See figure 2-39.) Trim out damaged spar cap and spar web as required. In order to assure a fuel-tight seal, care must be taken to be sure the parts fit together with no excessive gaps that may be hard to seal. The spar web is repaired by a repair web of 7075-T6 clad one gage heavier than the existing web and formed to fit over the existing web. Use a filler of the same gage and material as existing web, as required. The spar cap filler is fabricated from 0.125 inch 7075-O clad which is heat-treated to -T6 condition after forming. Nut plates as shown are fastened to the shim and the shim in turn is fastened to the spar cap filler. These nut plates replace the original nut plates removed with damaged spar cap. The repair angles are fabricated from 0.125 inch 4130 steel sheet. These angles should be heat-formed and heat-treated to 165,000 psi minimum after forming. Use tapered shim made of 7075-T6 as required to fit repair angle to spar cap. Refer to corrosion prevention procedures in Section I for treatment where steel and aluminum parts are in contact. The dome nuts installed on the repair doubler must be sealed before the doubler is installed. Coat under, around, and over dome nuts with PR-1221 sealing compound. After curing, exposed PR-1221 must be overcoated with fuel-resistant interior coating. (Refer to "Materials for Fuel Tank Sealing.") All mating surfaces of the repair must be covered with a thin coating of PR-1221 sealing compound before assembly. Fasteners used along the spar should be sealed by countersinking and counterboring the holes to match existing holes and sealing with PR-703 sealing compound. When using Deutsch rivets, dip pin in PR-1221 sealing compound before driving and drive while still wet. PR-1221 should also be applied under the heads of the rivets.

WING SKIN AT DOOR LAND REPAIR.

This repair may be used on damaged access doors in the wing skins. (See figure 2-40.) To make this repair, it is necessary to remove the upper skin panel. This provides access to the inner sides of both skin panels. It is necessary to mill the skin to a constant thickness between ribs in the doubler area in order to make a good, sound, leak-free repair. When the repair doubler extends over a rib land, alter the rib as required, using standard rib repair. Do not cut away existing rib lands. The repair doubler is machined, to fit damaged area, from 7075-T6 aluminum alloy plate. Stiffeners are machined on the doubler and add strength to the repair where the skin stringers have been removed. The filler is made from 7075-T6 material the same thickness as the skin after milling. A hole is cut out of the filler the size of the access door used on the access being repaired. If the existing access door is not damaged it may be used; otherwise, the existing door and fasteners must be duplicated. The repair doubler must be milled in the door area to accommodate any difference in thickness between door and skin. The repair must be sealed during assembly by applying PR-1221 sealing compound between faying surfaces and applying a bead of PR-1221 around all interior edges. After the PR-1221 is cured, it should be coated with fuel-resistant coating. (Refer to "Materials for Fuel Tank Sealing.")

MACHINED SKIN TRAILING EDGE REPAIR.

When the aft edge of the wing lower machined skin is damaged between stations 68 and 80, one of the repair straps shown in figure 2-40A may be used for a repair. The damage may extend as far forward as the rear spar plane. If the damage extends farther forward than the rear spar plane, the structural repair officer should be consulted for a specific repair. The trimmed-out area may vary in length if it does not exceed the limits shown on the illustrations and the required number of fasteners are used. The trimmed-out area must end midway between the existing fasteners. When the damaged area has been trimmed and burred, the edges of the cutout must be dye-penetrant checked to be absolutely certain that all of the damage has been removed. The repair strap is machined from 7075-T6 aluminum bar stock. The machined repair angle used in conjunction with the repair strap for the longest damage length is also machined from 7075-T6 aluminum bar stock or extrusion.

WEAR, REAM, AND BUSH LIMITS FOR WING TRAILING EDGE FITTINGS.

See figure 2-46B.

FLAP SKIN AND RIB REPAIR.

To make this repair, it is necessary to gain access to the inside of the flap by removing the front spar. (See figure 2-50.) The rib web is 2024-T4 clad sheet, 0.063-inch gage. At some stations the ribs have full webs, extending from the front to the rear spar. If web is damaged, it should be replaced. The milled legs on the skin must be removed in the damaged area to make room for the doubler. These legs should be cut flush within the inner surface of the skin. The cleaned-out damaged area need not be circular as shown; however, the filler

must be cut to match the cleaned-out area. The doubler is made of 2024-T4 clad sheet. It must be at least 0.010 inch heavier than the damaged skin. The rib repair angles are to be machined from 2024-T4 or 7075-T6 bar stock. The legs of the repair angle should be made 0.015 inch heavier than the original rib leg. The repair angle must have a radius along the back to clear the radius on the rib leg and must be milled to go over the repair doubler. Fillers, made of 2024-T4 clad sheet the same gage as the rib leg thickness should be used in the area where the rib web is attached. This filler fills the space between the rib web and the repair angle. The flap skin, repair doubler, and skin filler should be dimpled to enable the use of flush rivets. The repair angle is to be countersunk to match. In areas where the skin thickness is stepped, a filler of the proper thickness is added under the doubler.

TRAILING EDGE HONEYCOMB REPAIR.

See figures 2-57, 2-60, 2-61, 2-62, and 2-63.

Repair of the honeycomb trailing edge must not be any wider than 8 inches at the trailing edge. The maximum number of repairs for any trailing edge is two. If more repairs are needed, the trailing edge should be replaced. The finished repair should be flush (±0.015 inch) with mold line. When trimming out damaged area and honeycomb core to make room for splice doubler, it is not necessary to cut the honeycomb core straight or have a smooth cut. Any irregularities will be filled with the epoxy adhesive. The splice doublers are made of balsa wood and covered with 0.016-inch, 6061-T6 aluminum alloy sheet formed to fit inside the existing skin. If necessary, 2024-T4 clad sheet may be substituted for 6061-T6 aluminum alloy sheet. The balsa wood filler is cut to match the splice doublers with the metal covering in place. The repair skin is 0.016 inch 6061-T6 or 2024-T4 aluminum sheet formed to match the existing trailing edge. As the repair parts are assembled, apply plenty of adhesive to bond all parts together and to fill any gaps or other irregularities on repair. This repair may be cold-cured if necessary. Refer to index for general instruction for repairing honeycomb sandwich panels. After repairs are complete and the adhesive is cured, the surfaces should be painted with aluminized lacquer.

HOLDING FIXTURE DIMENSIONS.

Figures 2-67, 2-68, 2-69, and 2-70 show the necessary dimensions for the construction of fixtures to hold the wing components while repairs are being made. These dimensions and attach points are not intended for use in manufacturing jigs for the building of a complete component. They are intended only for manufacturing holding fixtures to maintain the proper attach points, and to ensure that the part being repaired will fit when reinstalled.

REPAIR OF SWAY-BRACE HELI-COIL INSERTS – TYPE I AND TYPE III PYLONS.

Heli-Coil inserts when loose or damaged should be replaced in accordance with the procedures in "General—Installation of Heli-Coil Inserts," T.O. 44H1-1-117. On Type I pylons, Heli-Coil inserts with internal thread length of less than 9/16 inch (nine thread turns), or inserts on Type III pylons with an internal thread length of less than 5/8 inch (10 thread turns) should be replaced.

REPAIR OF STRIPPED THREADS.

Stripped threads in the sway-brace casting may be repaired using the following procedure:

1. Drill the stripped hole to 15/16 inch diameter. Maintain the blueprint tolerances for location and alignment.

2. Thread the 15/16-inch hole with a 1-14NF3 tap.

3. Fabricate a threaded bushing and install in casting. See figure 2-71. A minimum driving torque of 300 inch-pounds must be obtained before and after full engagement of the threads.

4. Lock bushing in place by drilling 3/32-inch diameter, 5/16-inch long hole in bushing and casting threads, and install NAS561-3-5 Rollpin.

Note

Locate the hole in the heavy side of casting.

5. Spot-face, as necessary, upper and lower end of bushing to original casting dimensions. Chamfer end of threaded hole.

6. Apply zinc-chromate primer to spot-faced areas.

REPAIR OF FLANGED BUSHING.

Pressed-in flanged bushings are serviceable if the bushing has not been unseated. If the bushing has been unseated, the following procedure may be used for repair:

1. Remove bushing and two pins that secured flange.

2. Drill hole to 1 3/64 inch diameter. Maintain blueprint tolerances for location and alignment.

3. Thread hole with 1 1/8-12NF3 tap.

4. Chamfer counterbored end of hole 45 degrees by 1 3/16 inch diameter.

5. Mark position of existing pin holes.

6. Fabricate flanged threaded bushing and install it in casting. (See figure 2-71.) Maintain minimum driving torque of 300 inch-pounds before seating flange.

7. Lock bushing in place by drilling two 1/8-inch diameter, 1/2-inch long holes in flange and installing two NAS561-4-8 Rollpins. Locate pins 180 degrees apart and clear of original pin holes and threads.

8. Spot-face flange end and anti-flange end of bushing to blueprint dimensions.

9. Apply zinc-chromate primer to spot-faced ends of bushing.

PERMANENT REPAIR OF WING OUTER PANEL SLAT INTERCONNECT FITTINGS.

When wear tolerances of a wing slat interconnector fitting are exceeded permanent repair may be accomplished as follows:

1. Enlarge the worn oversize hole (normally 5/8 inch inside diameter) to 0.6870 inch minimum. Whenever it is necessary to enlarge the hole above 0.6870 inch, remove only the minimum amount of material required to "clean up" the wear and only up to maximum of 0.7036 inch inside diameter. Replace all fittings exceeding the above 0.7036 tolerance.

2. Make a bushing to fit enlarged hole as follows:

a. Use 3/4-inch-diameter 4130 steel bar.

b. Make bushing OD 0.0002 to 0.0004 inch larger than ID of enlarged hole.

c. Make bushing ID 0.625 (+0.005, −0.001) inch.

d. Make bushing length 0.750 ±1/32) inch.

e. Cadmium-plate bushing per Federal Specification QQ-P-416.

3. Apply zinc-chromate primer; then press bushing into fitting.

REWORK OF AILERON BELL CRANK SUPPORT.

See figure 2-72.

WING SKIN TONGUE AREA REPAIR.

Damage or elongation of the wing-to-center-box-section boltholes in the upper skin may occur when bolts are inserted through these holes to hold the wing in the shipping fixture. These holes may be repaired by reaming holes oversize and inserting a bushing. To insert these bushings, proceed as follows:

1. Ream out damaged holes to 1/8-inch oversize maximum (7/16-inch holes to 9/16-inch oversize, 1/2-inch holes to 5/8-inch oversize). Paint inside of oversize holes with Alodine (Specification MIL-C-5541).

NOTE Do not ream more than three holes in a single row or ream holes over the maximum of 1/8-inch oversize, without first contacting SMAMA structural engineering for repair instructions.

2. Manufacture repair bushing for damaged holes. (See figure 2-41B.)

3. Paint outside of bushing with a liberal coat of zinc-chromate primer (Specification MIL-P-8585 or equivalent), and, while wet, press bushing into oversize hole.

4. Ream bushing, if necessary, to fit applicable bolt. (Refer to applicable Illustrated Parts Breakdown for bolt sizes.)

UPPER WING SKIN MLG TRUNNION BOLT COUNTERBORES.

When upper wing skin MLG trunnion bolt counterbores are discovered cracked – consult Sacramento ALC drawing 7335823 for counterbore crack classification, disposition and rework instructions.

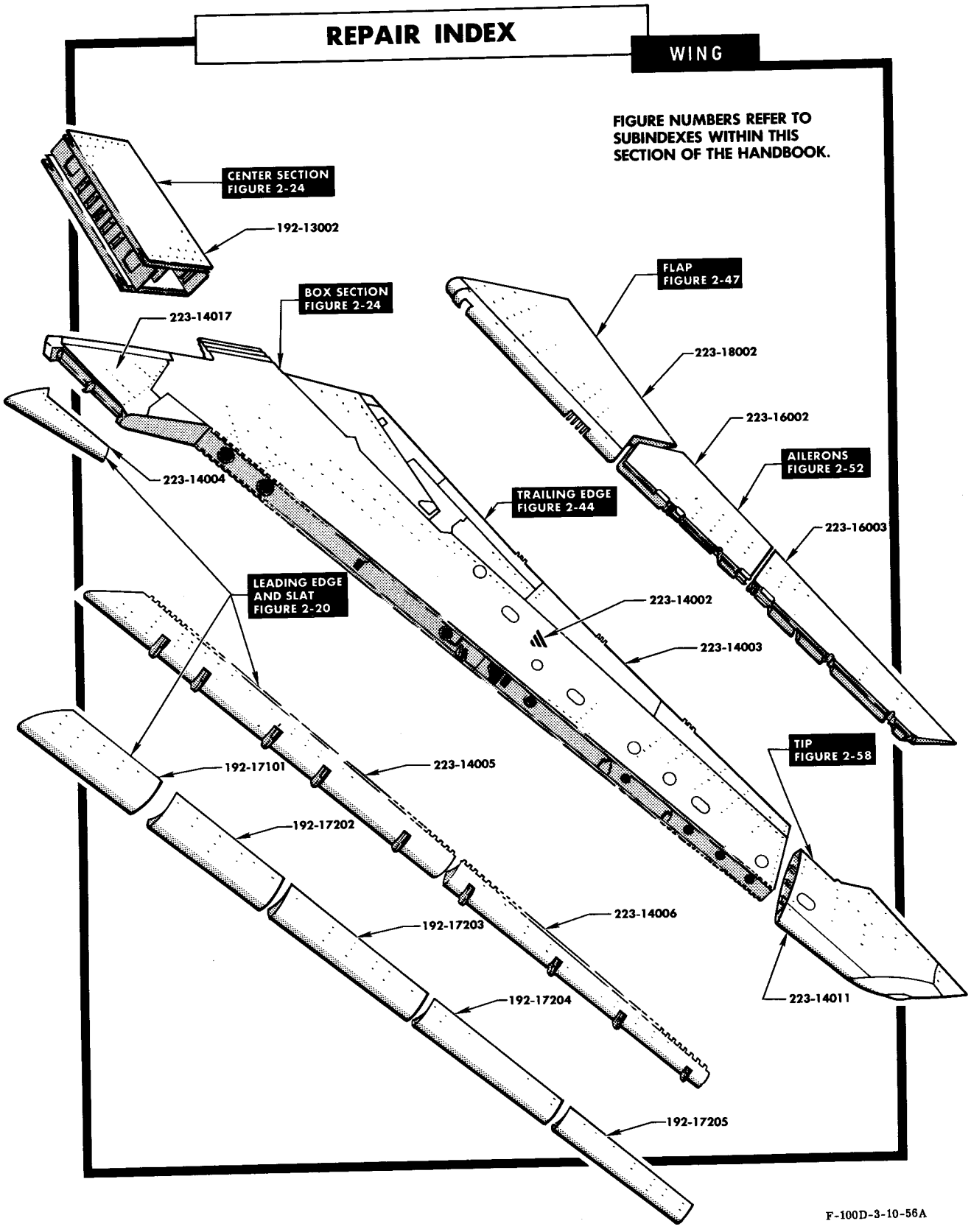


Figure 2-19. Wing Repair Index

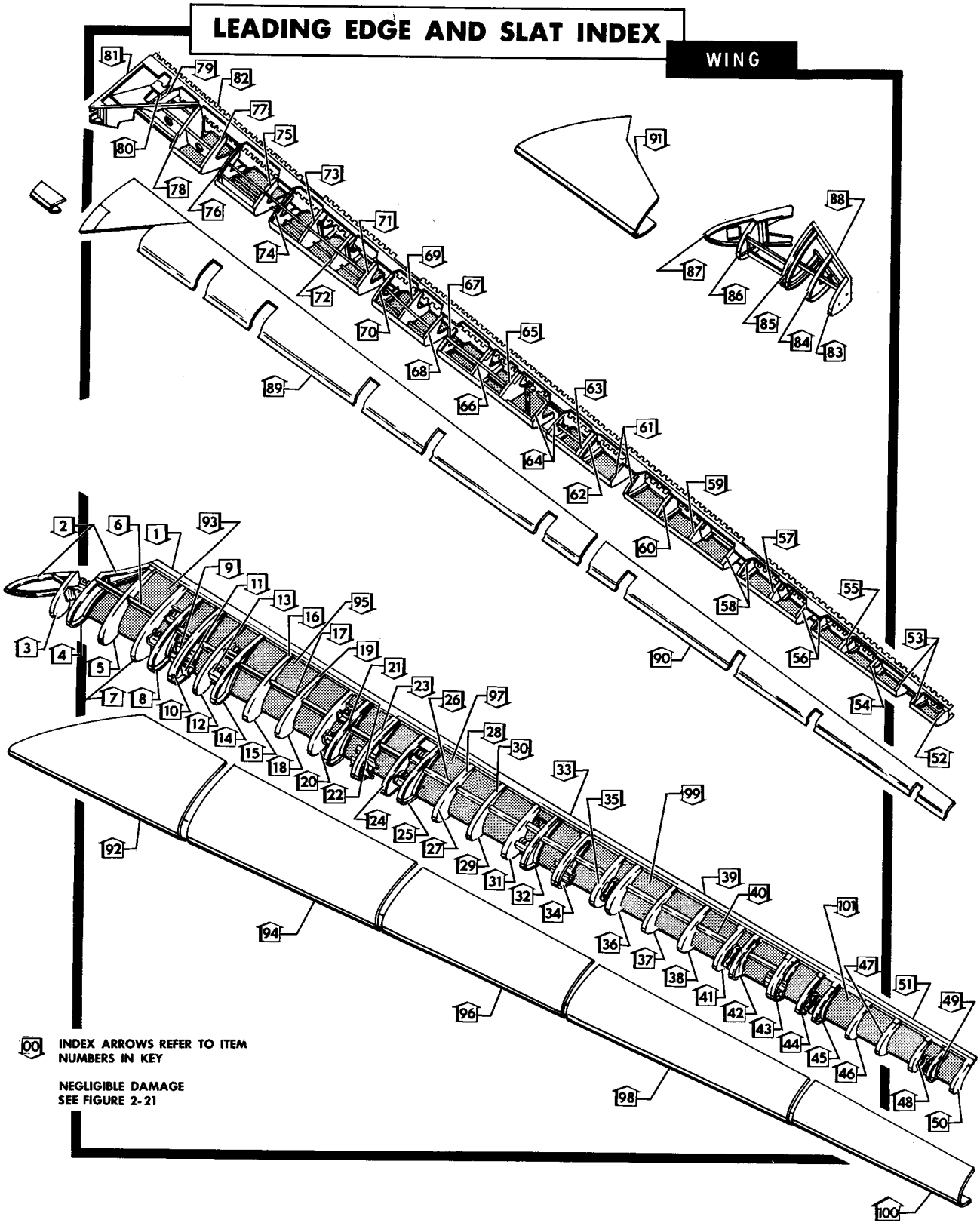


Figure 2-20. Leading Edge and Slat Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-17216-13	TE	Extr	2024-T42	
2	192-17114	Rib Assy	0.040	2024-T4 Clad	2-22
3	192-17103	Rib Assy	0.064	2024-T4 Clad	2-22
4	192-17104	Rib Assy	0.064	2024-T4 Clad	2-22
5	192-17013	Rib	0.040	2024-T4 Clad	2-22
6	192-17106	Spar	0.064	2024-T4 Clad	
7	192-17214	Rib	0.064	2024-T4 Clad	2-22
8	192-17215	Rib	0.064	2024-T4 Clad	2-22
9	192-17213-19	"Z"	0.040	2024-T4 Clad	
10	192-17016	Rib	0.040	2024-T4 Clad	2-22
	192-17017	Rib	0.040	2024-T4 Clad	2-22
11	192-17203-15	"Z"	0.040	2024-T4 Clad	
12	192-17218-3	Rib	0.064	2024-T4 Clad	2-22
13	192-17213-17	"Z"	0.040	2024-T4 Clad	
14	192-17219-3	Rib	0.064	2024-T4 Clad	2-22
15	192-17020-3	Rib	0.040	2024-T4 Clad	2-22
16	192-17020-7	Rib	0.040	2024-T4 Clad	2-22
17	192-17207	Spar			
18	192-17020-5	Rib	0.040	2024-T4 Clad	2-22
19	192-17020-9	Rib	0.040	2024-T4 Clad	2-22
20	192-17218-5	Rib	0.064	2024-T4 Clad	2-22
21	192-17219-5	Rib	0.064	2024-T4 Clad	2-22
22	192-17216-5	TE	Extr	2024-T42	
23	192-17021	Rib	0.040	2024-T4 Clad	2-22
	192-17022	Rib	0.040	2024-T4 Clad	2-22
24	192-17223-3	Rib	0.064	2024-T4 Clad	2-22
25	192-17224-3	Rib	0.064	2024-T4 Clad	2-22
26	192-17208	Spar	0.064	2024-T4 Clad	
27	192-17025	Rib	0.040	2024-T4 Clad	2-22
28	192-17025-7	Rib	0.040	2024-T4 Clad	2-22
29	192-17025-5	Rib	0.040	2024-T4 Clad	2-22
30	192-17025-9	Rib	0.040	2024-T4 Clad	2-22
31	192-17223-5	Rib	0.064	2024-T4 Clad	2-22
32	192-17224-5	Rib	0.064	2024-T4 Clad	2-22
33	192-17216-7	TE	Extr	2024-T42	
34	192-17226	Rib			2-22
	192-17227	Rib			
35	192-17228-3	Rib	0.064	2024-T4 Clad	
36	192-17229	Rib	0.064	2024-T4 Clad	
37	192-17030-14	Rib	0.064	2024-T4 Clad	
38	192-17030-11	Rib	0.040	2024-T4 Clad	
39	192-17216-9	TE	Extr	2024-T42	
40	192-17209	Spar	0.064	2024-T4 Clad	
41	192-17228-5	Rib	0.064	2024-T4 Clad	
42	192-17229	Rib	0.064	2024-T4 Clad	
43	192-17031	Rib	0.040	2024-T4 Clad	
43	192-17032	Rib	0.040	2024-T4 Clad	
44	192-17233	Rib	0.064	2024-T4 Clad	
45	192-17234	Rib	0.064	2024-T4 Clad	
46	192-17035-3	Rib	0.040	2024-T4 Clad	
47	192-17035-5	Rib	0.040	2024-T4 Clad	
48	192-17233-5	Rib	0.064	2024-T4 Clad	
49	192-17234-5	Rib	0.064	2024-T4 Clad	
50	192-17036	Rib Assy	0.020	CRES MIL-S-5059	
51	192-17216-11	TE	Extr	2024-T42	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 2-20

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
52	180-14212	Rib	0.040	2024-T4 Clad	2-23
53	192-14227	Rib	0.064	2024-T4 Clad	2-23
54	192-12203	Rib Assy	0.040	2024-T4 Clad	2-23
55	192-12203	Rib Assy	0.040	2024-T4 Clad	2-23
56	192-14227	Rib	0.064	2024-T4 Clad	2-23
57	192-12203	Rib Assy	0.040	2024-T4 Clad	2-23
58	192-14227	Rib	0.064	2024-T4 Clad	2-23
59	192-12211	Rib Assy	0.064	2024-T4 Clad	2-23
60	223-14212	Rib Assy	0.051	2024-T4 Clad	2-23
61	192-14227	Rib	0.064	2024-T4 Clad	2-23
62	223-14210	Rib	0.051	2024-T4 Clad	2-23
63	223-142-07	Rib	0.051	2024-T4 Clad	2-23
64	192-12239	Rib	0.051	2024-T4 Clad	2-23
65	223-14323	Rib	0.051	2024-T4 Clad	2-23
66	223-14233	Rib	0.040	2024-T4 Clad	2-23
67	223-14215	Rib	0.050	2024-T4 Clad	2-23
68	223-14218	Rib	0.051	2024-T4 Clad	2-23
69	223-14231	Rib	0.040	2024-T4 Clad	2-23
70	223-14214	Rib	0.051	2024-T4 Clad	2-23
71	223-14213	Rib	0.051	2024-T4 Clad	2-23
72	223-14234	Rib	0.063	2024-T4 Clad	2-23
73	223-14235	Rib	0.051	2024-T4 Clad	2-23
74	223-14228	Rib	0.051	2024-T4 Clad	2-23
75	223-14225	Rib	0.051	2024-T4 Clad	2-23
76	223-14223	Rib	0.051	2024-T4 Clad	2-23
77	223-14222	Rib	0.051	2024-T4 Clad	2-23
78	223-14237	Rib	0.051	2024-T4 Clad	2-23
79	223-14221	Rib	0.051	2024-T4 Clad	2-23
80	223-14239	Rib	0.063	2024-T4 Clad	2-23
81	223-14208	Rib Assy	0.063	2024-T4 Clad	2-23
82	223-14169	Joint	Extr	7075-T6	
83	223-14217	Rib	0.040	2024-T4 Clad	2-23
84	223-14216	Rib	0.063	2024-T4 Clad	2-23
85	223-14202	Rib Assy	0.063	7075-T6 Clad	2-23
86	192-12215	Rib	0.040	2024-T4 Clad	2-23
87	192-12201	Rib Assy	Extr	2024-T42	2-23
88	223-14547	Angle	Extr	2024-T42	
89	223-14005-3	Skin	0.040	2024-T4 Clad	**2-23A, 2-23C**
90	223-14006-3	Skin	0.040	2024-T4 Clad	**2-23A, 2-23C**
91	223-14004-3	Skin	0.063	2024-T4 Clad	
92	223-17201-3	Skin	0.051	2024-T4 Clad	2-23B
93	223-17201-5	Skin	0.040	2024-T4 Clad	2-23B
94	223-17202-3	Skin	0.051	2024-T4 Clad	2-23B
95	223-17202-5	Skin	0.040	2024-T4 Clad	2-23B
96	223-17203-3	Skin	0.051	2024-T4 Clad	2-23B
97	223-17203-5	Skin	0.040	2024-T4 Clad	2-23B
98	223-17204-3	Skin	0.051	2024-T4 Clad	2-23B
99	223-17204-5	Skin	0.040	2024-T4 Clad	2-23B
100	223-17205-3	Skin	0.040	2024-T4 Clad	2-23B
101	223-17205-5	Skin	0.040	2024-T4 Clad	2-23B

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

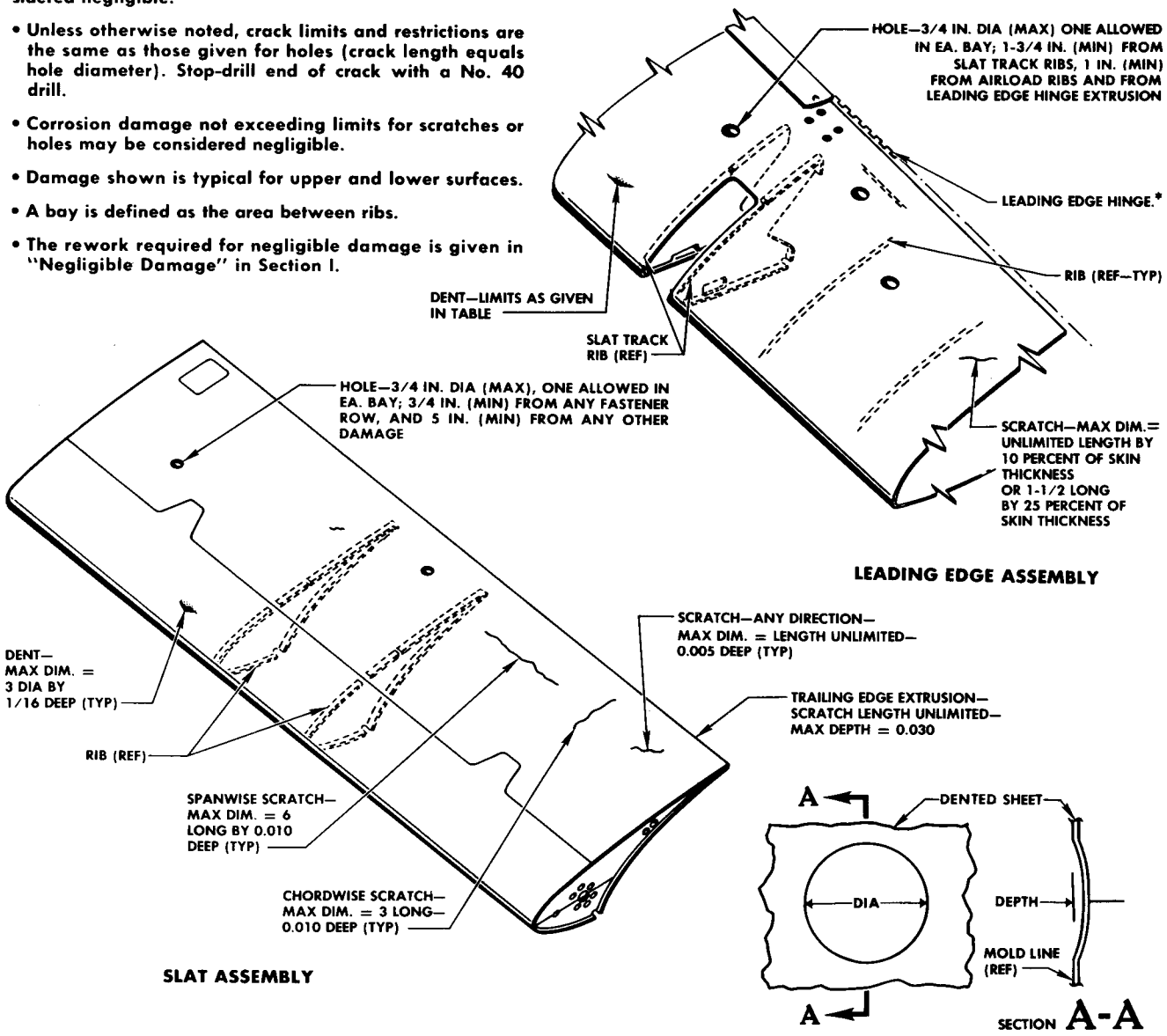
Key to Figure 2-20 (Continued)

LEADING EDGE AND SLAT NEGLIGIBLE DAMAGE

WING

NOTE Damage within the limits shown may be considered negligible.

- Unless otherwise noted, crack limits and restrictions are the same as those given for holes (crack length equals hole diameter). Stop-drill end of crack with a No. 40 drill.
- Corrosion damage not exceeding limits for scratches or holes may be considered negligible.
- Damage shown is typical for upper and lower surfaces.
- A bay is defined as the area between ribs.
- The rework required for negligible damage is given in "Negligible Damage" in Section I.



* A single broken lug may be left unrepaired unless it is either within 2 inches of any slat track support fitting or within one inch of front spar stations 116.95, 163.95, or 245.9.

Two broken lugs may be left unrepaired in any 5-inch area, provided they are separated by at least one good lug and are not located within 2 inches of any slat track support or within one inch of front spar stations 116.95, 163.95, or 245.9.

NEGLIGIBLE DENTS

MIN DIMENSION	1 IN.	2 TO 3 IN.	4 IN. AND OVER
MAX DEPTH	1/32	1/16	3/32

NOTE See figure 2-23D.

F-100D-3-10-122C

Figure 2-21. Leading Edge and Slat Negligible Damage

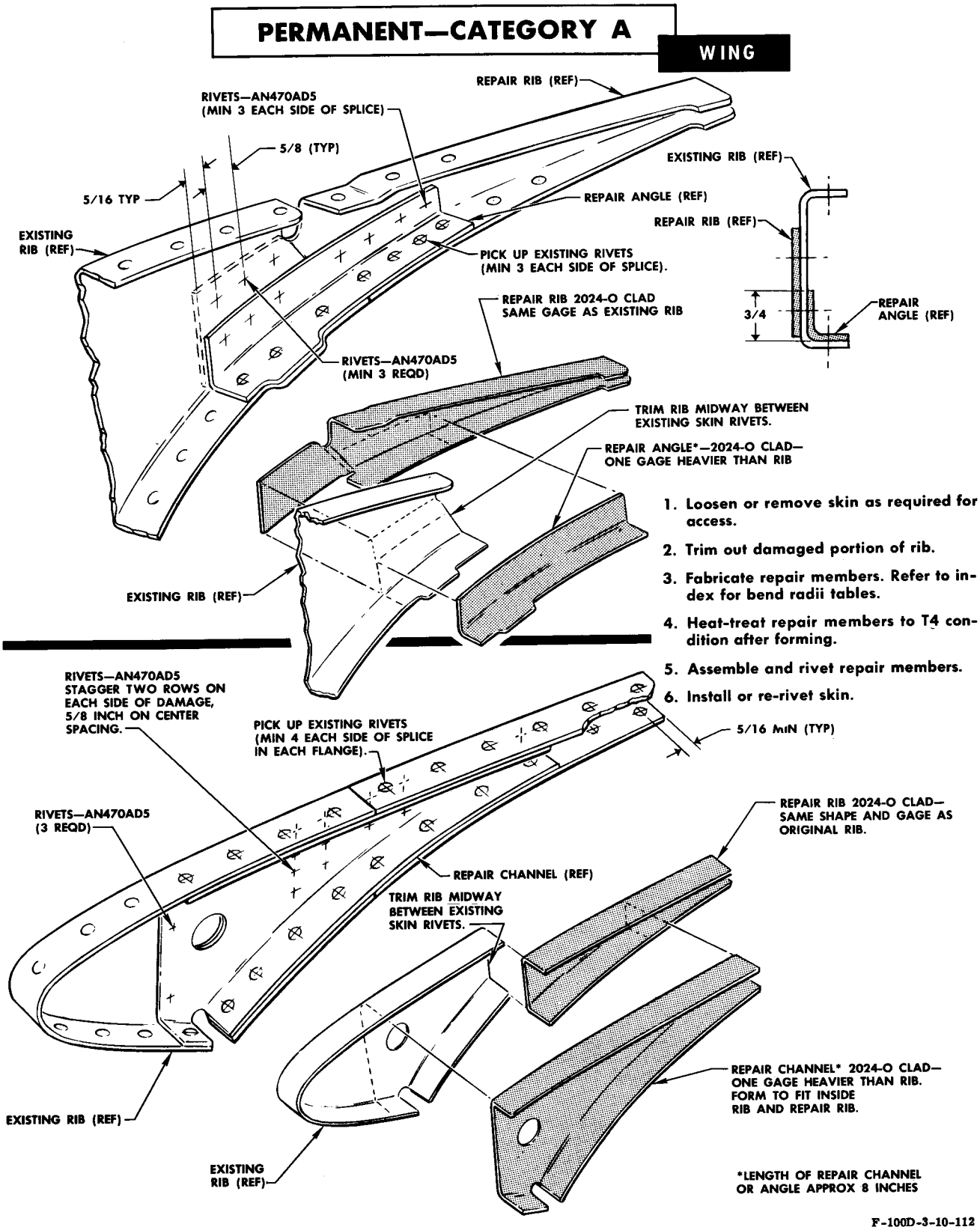
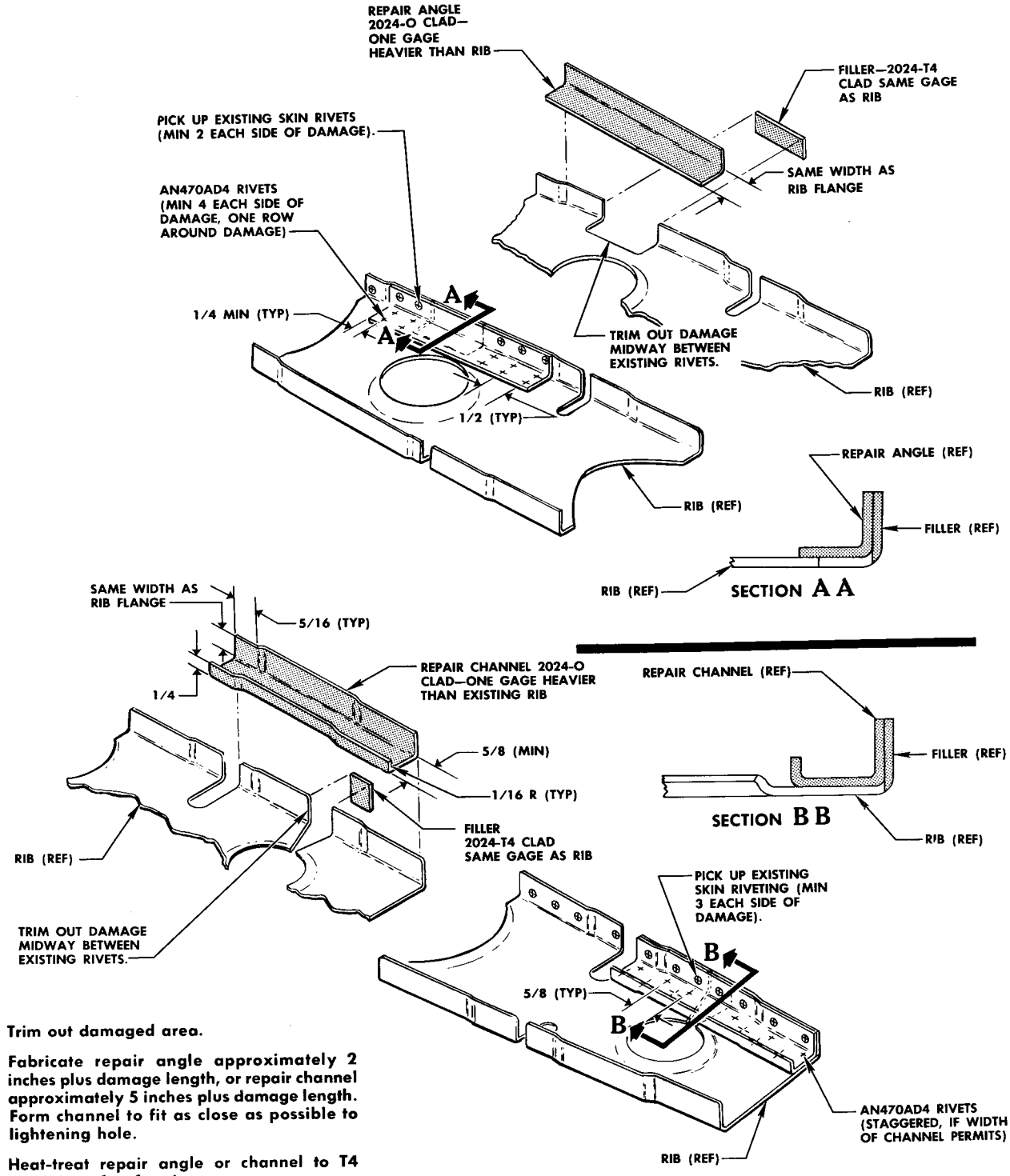


Figure 2-22. Slat Rib

PERMANENT—CATEGORY A

WING

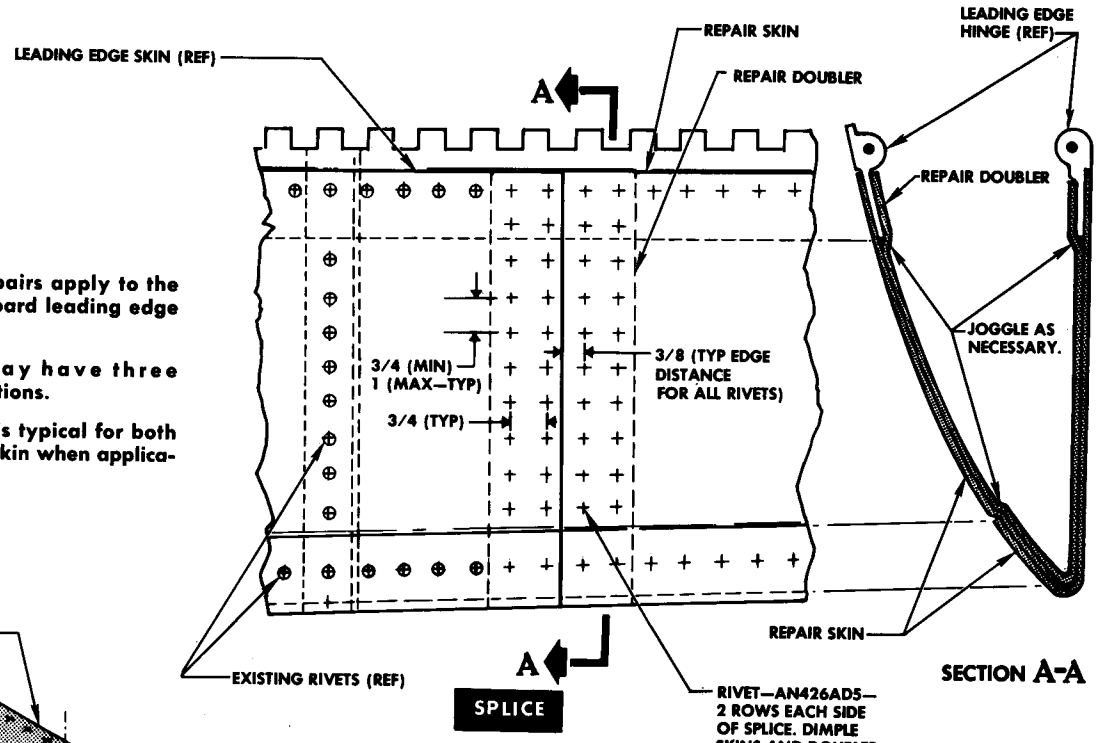


1. Trim out damaged area.
2. Fabricate repair angle approximately 2 inches plus damage length, or repair channel approximately 5 inches plus damage length. Form channel to fit as close as possible to lightening hole.
3. Heat-treat repair angle or channel to T4 condition after forming.
4. Install repair parts on rib.

Figure 2-23. Rib Flanges

PERMANENT CATEGORY A

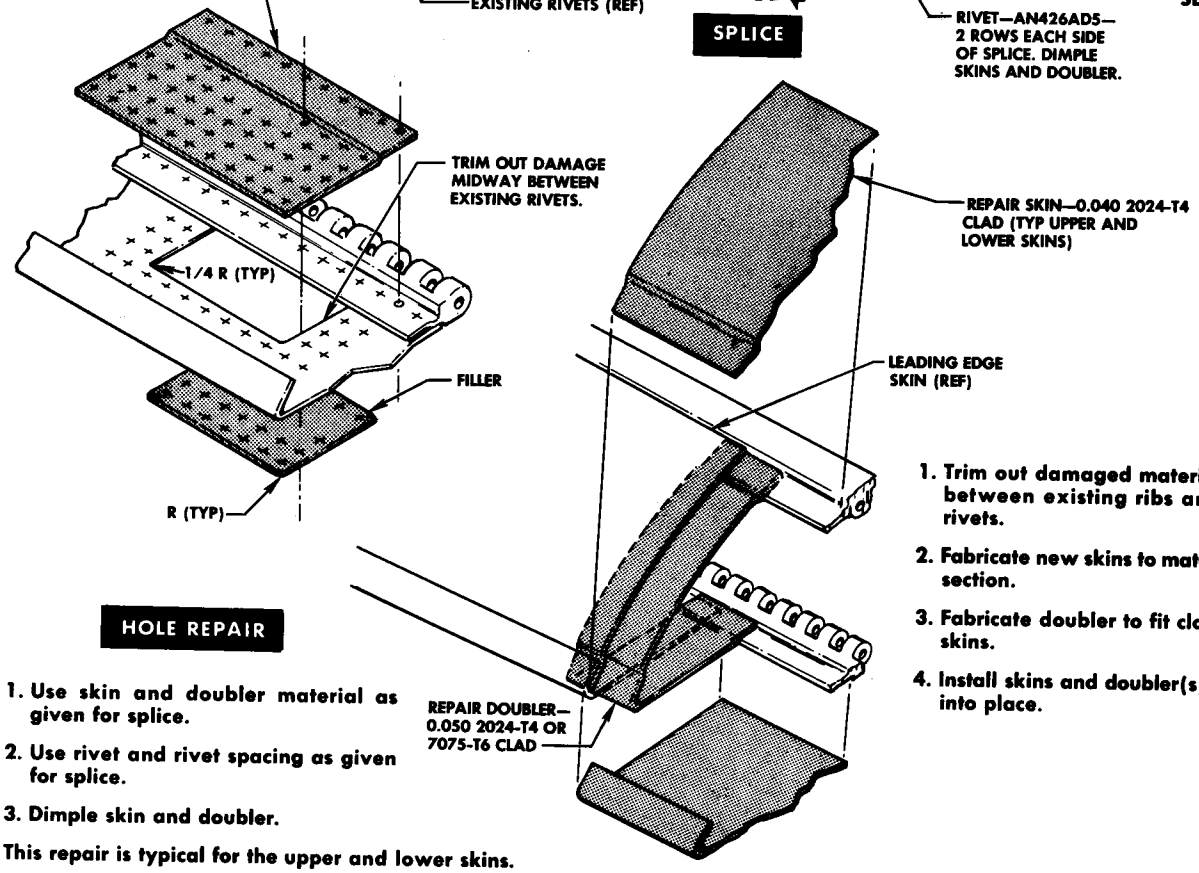
WING



NOTE These repairs apply to the inboard and outboard leading edge sections.

Each section may have three spliced-in skin sections.

The splice shown is typical for both ends of the repair skin when applicable.



HOLE REPAIR

1. Use skin and doubler material as given for splice.
2. Use rivet and rivet spacing as given for splice.
3. Dimple skin and doubler.

This repair is typical for the upper and lower skins.

1. Trim out damaged material midway between existing ribs and existing rivets.
2. Fabricate new skins to match removed section.
3. Fabricate doubler to fit closely inside skins.
4. Install skins and doubler(s), and rivet into place.

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Figure 2-23A. Leading Edge Skins

PERMANENT CATEGORY A

WING

NOTE

Refer to text for skin cut locations.

- Increase width of doubler at door cutouts in lower skins to provide for three rows of rivets at spacing given.
- Trim doubler to match door cutout in skin.
- All blind close-out fasteners must match those removed.
- Prime contact surfaces of skin and repair doubler before installation.

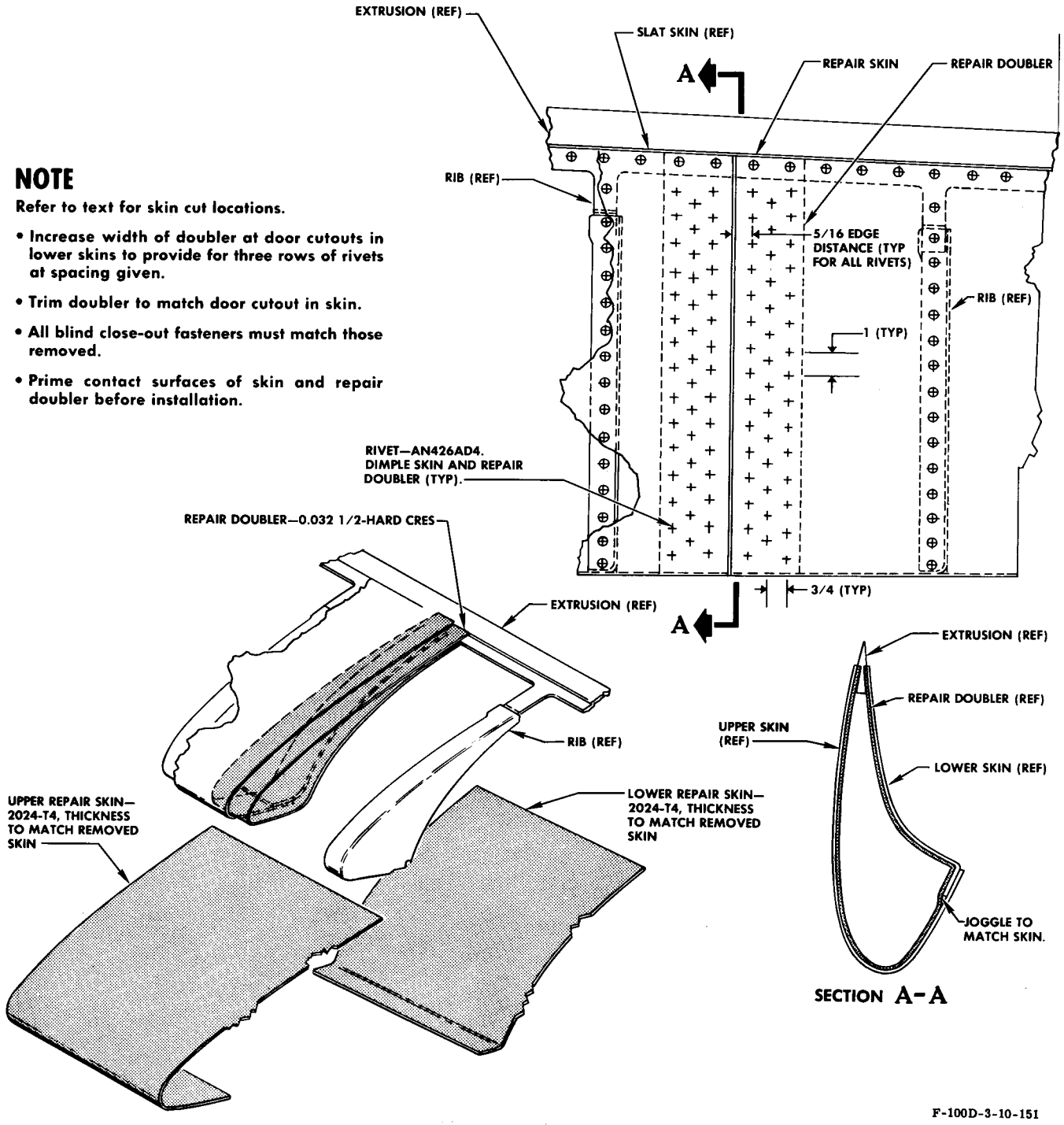
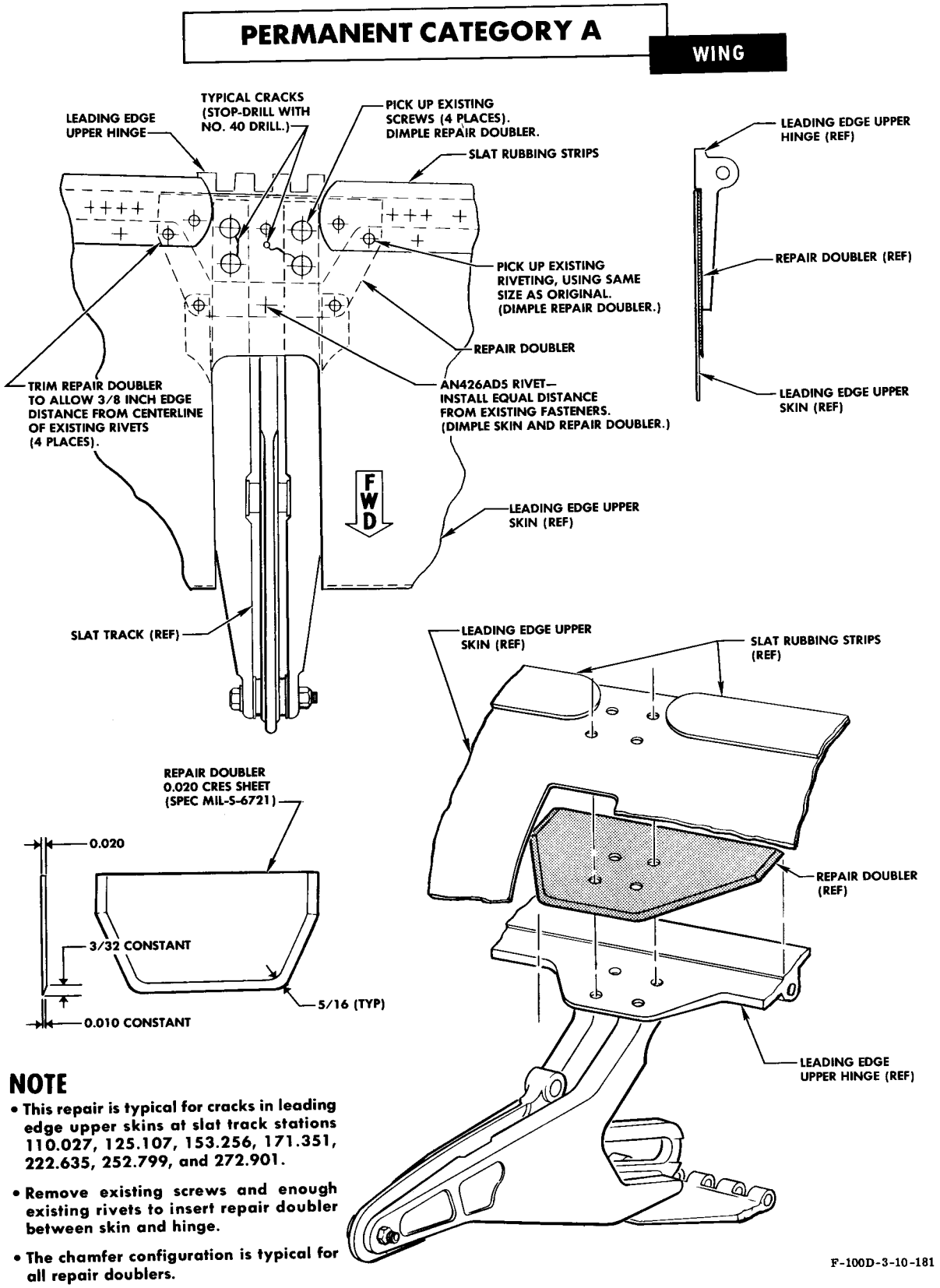


Figure 2-23B. Skin Splice—Slat



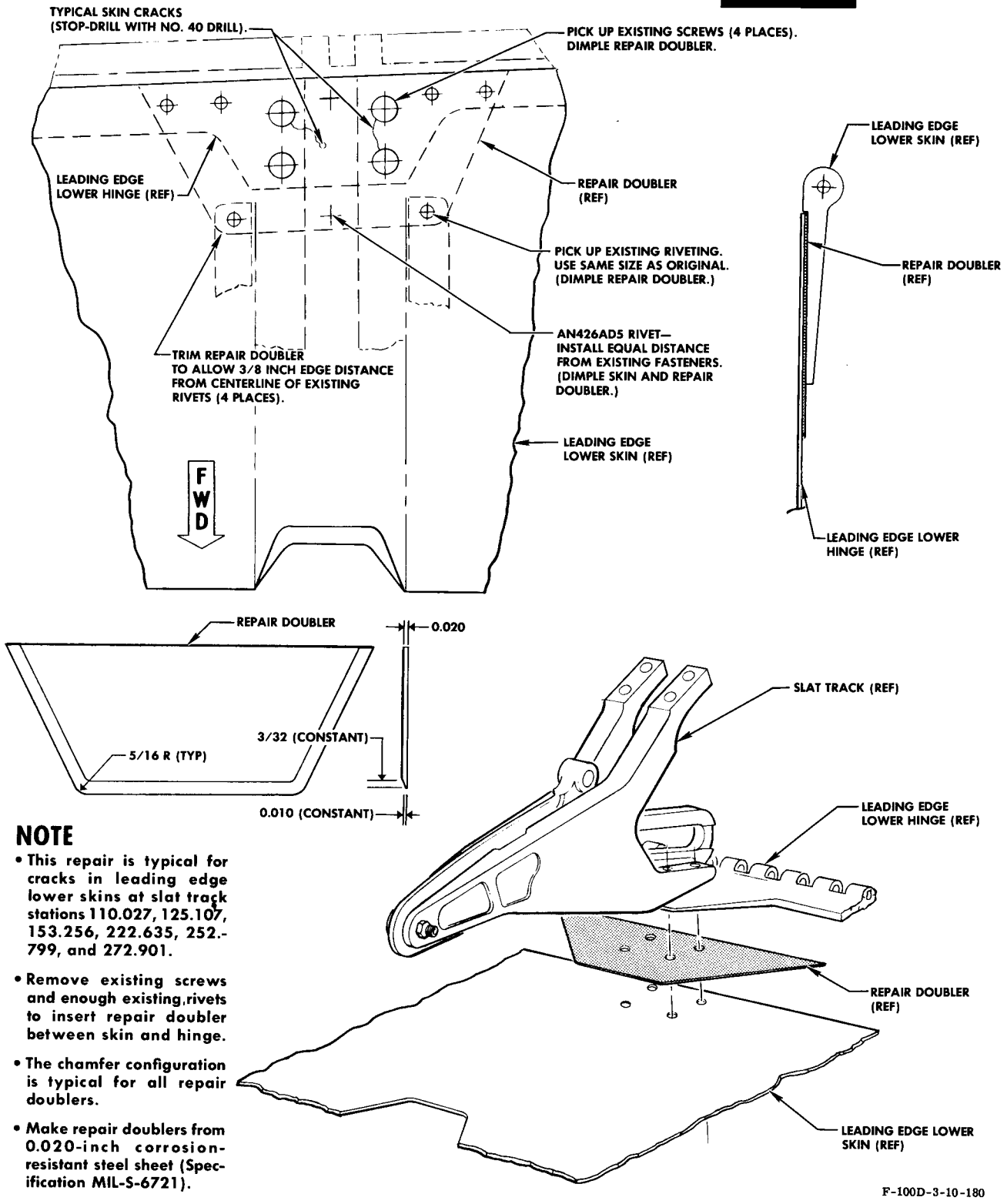
NOTE

- This repair is typical for cracks in leading edge upper skins at slat track stations 110.027, 125.107, 153.256, 171.351, 222.635, 252.799, and 272.901.
- Remove existing screws and enough existing rivets to insert repair doubler between skin and hinge.
- The chamfer configuration is typical for all repair doublers.

Figure 2-23C. Cracks in Leading Edge Skins at Slat Tracks (Sheet 1 of 5)

PERMANENT CATEGORY A

WING



NOTE

- This repair is typical for cracks in leading edge lower skins at slat track stations 110.027, 125.107, 153.256, 222.635, 252.799, and 272.901.
- Remove existing screws and enough existing rivets to insert repair doubler between skin and hinge.
- The chamfer configuration is typical for all repair doublers.
- Make repair doublers from 0.020-inch corrosion-resistant steel sheet (Specification MIL-S-6721).

Figure 2-23C. Cracks in Leading Edge Skins at Slat Tracks (Sheet 2 of 5)

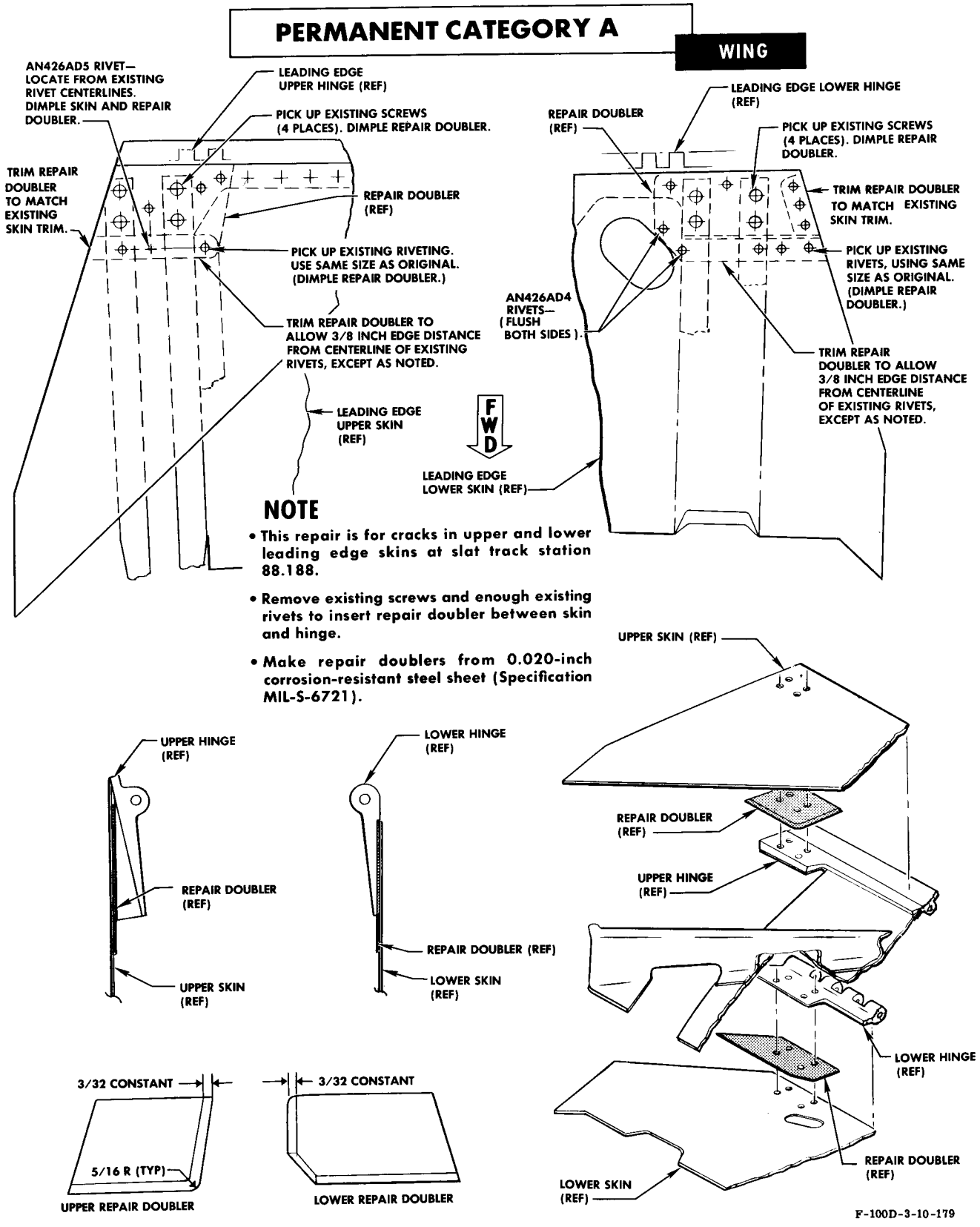
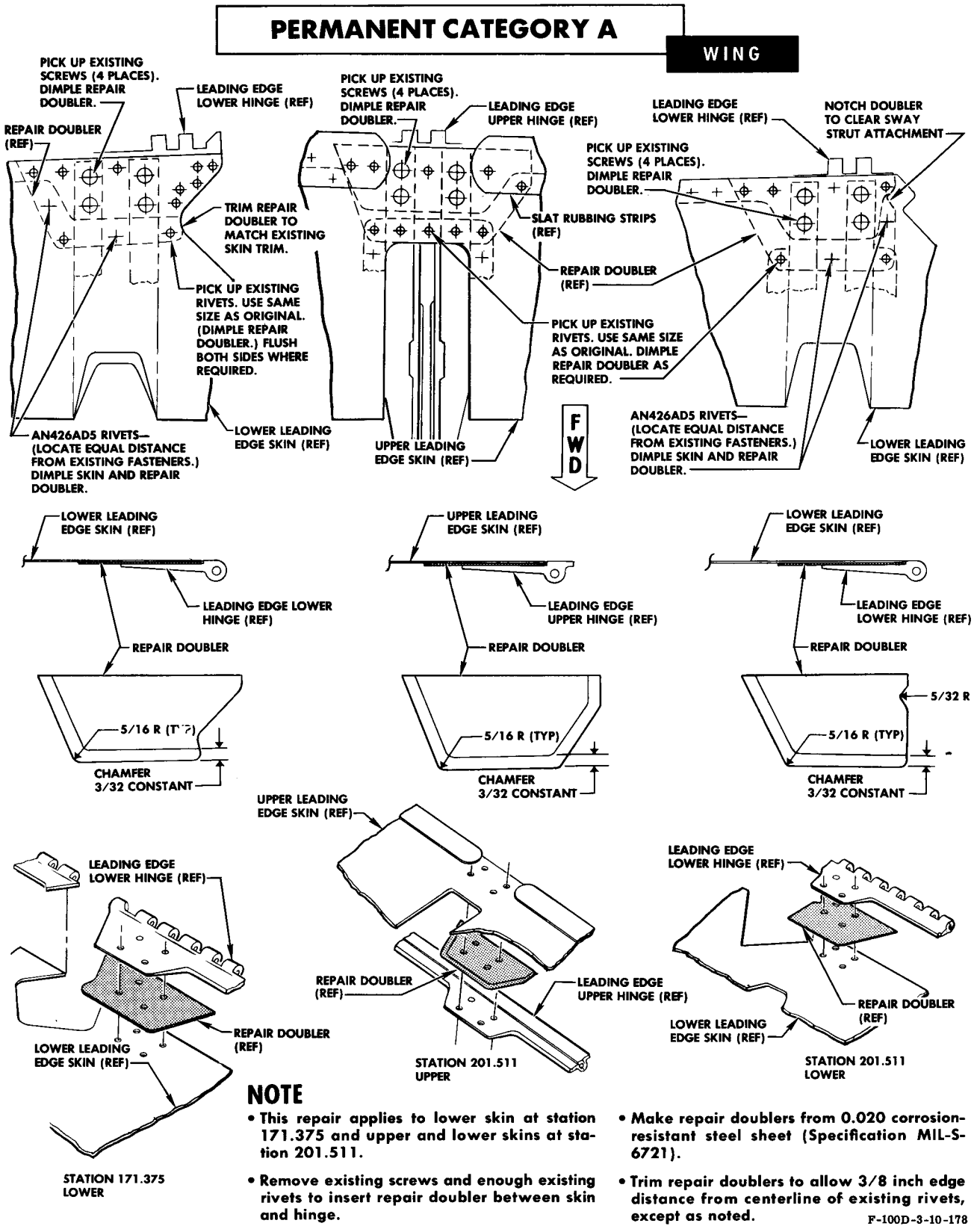


Figure 2-23C. Cracks in Leading Edge Skins at Slat Tracks (Sheet 3 of 5)



NOTE

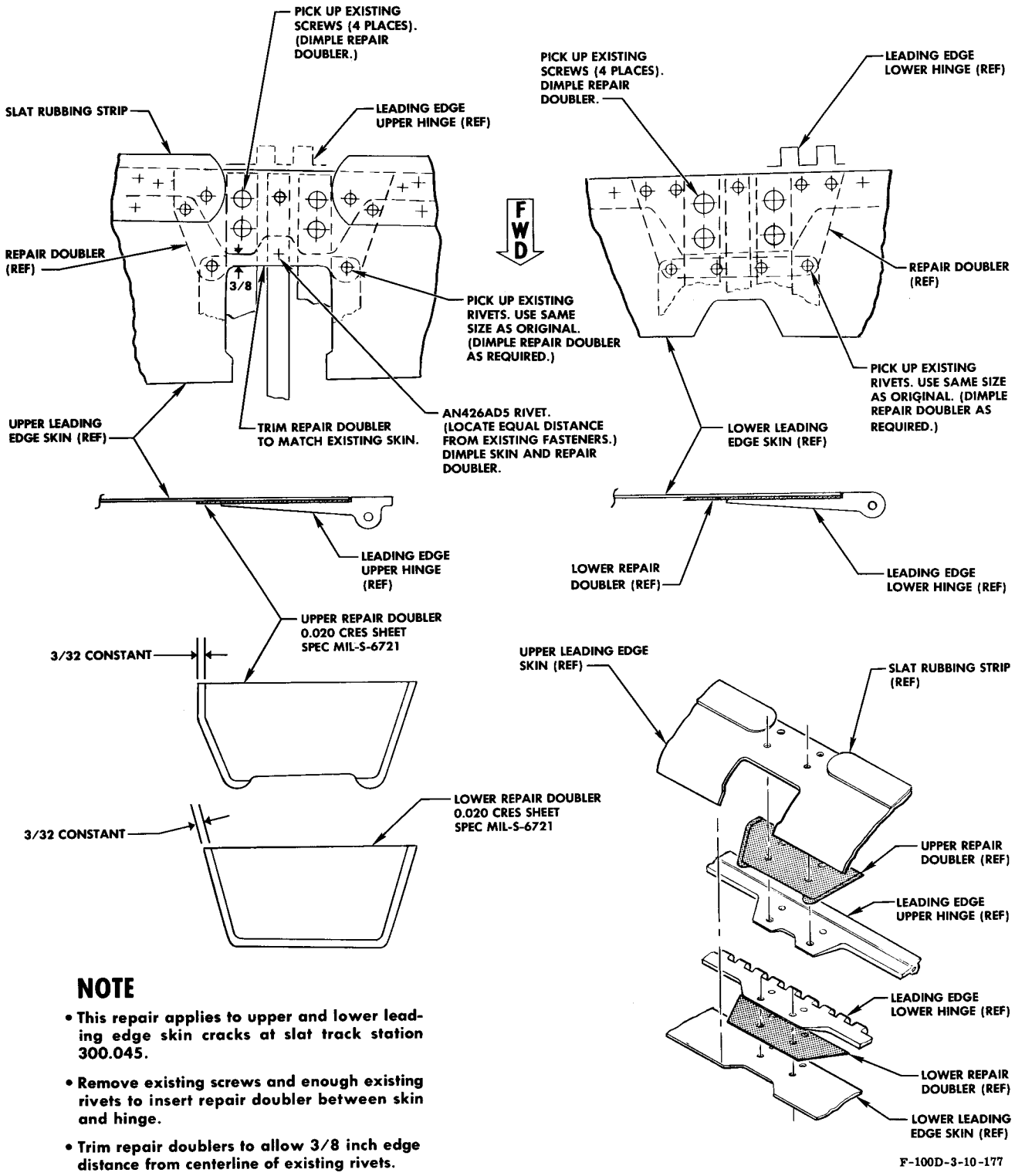
- This repair applies to lower skin at station 171.375 and upper and lower skins at station 201.511.
- Remove existing screws and enough existing rivets to insert repair doubler between skin and hinge.
- Make repair doublers from 0.020 corrosion-resistant steel sheet (Specification MIL-S-6721).
- Trim repair doublers to allow 3/8 inch edge distance from centerline of existing rivets, except as noted.

F-100D-3-10-178

Figure 2-23C. Cracks in Leading Edge Skins at Slat Tracks (Sheet 4 of 5)

PERMANENT CATEGORY A

WING



NOTE

- This repair applies to upper and lower leading edge skin cracks at slat track station 300.045.
- Remove existing screws and enough existing rivets to insert repair doubler between skin and hinge.
- Trim repair doublers to allow 3/8 inch edge distance from centerline of existing rivets.

Figure 2-23C. Cracks in Leading Edge Skins at Slat Tracks (Sheet 5 of 5)

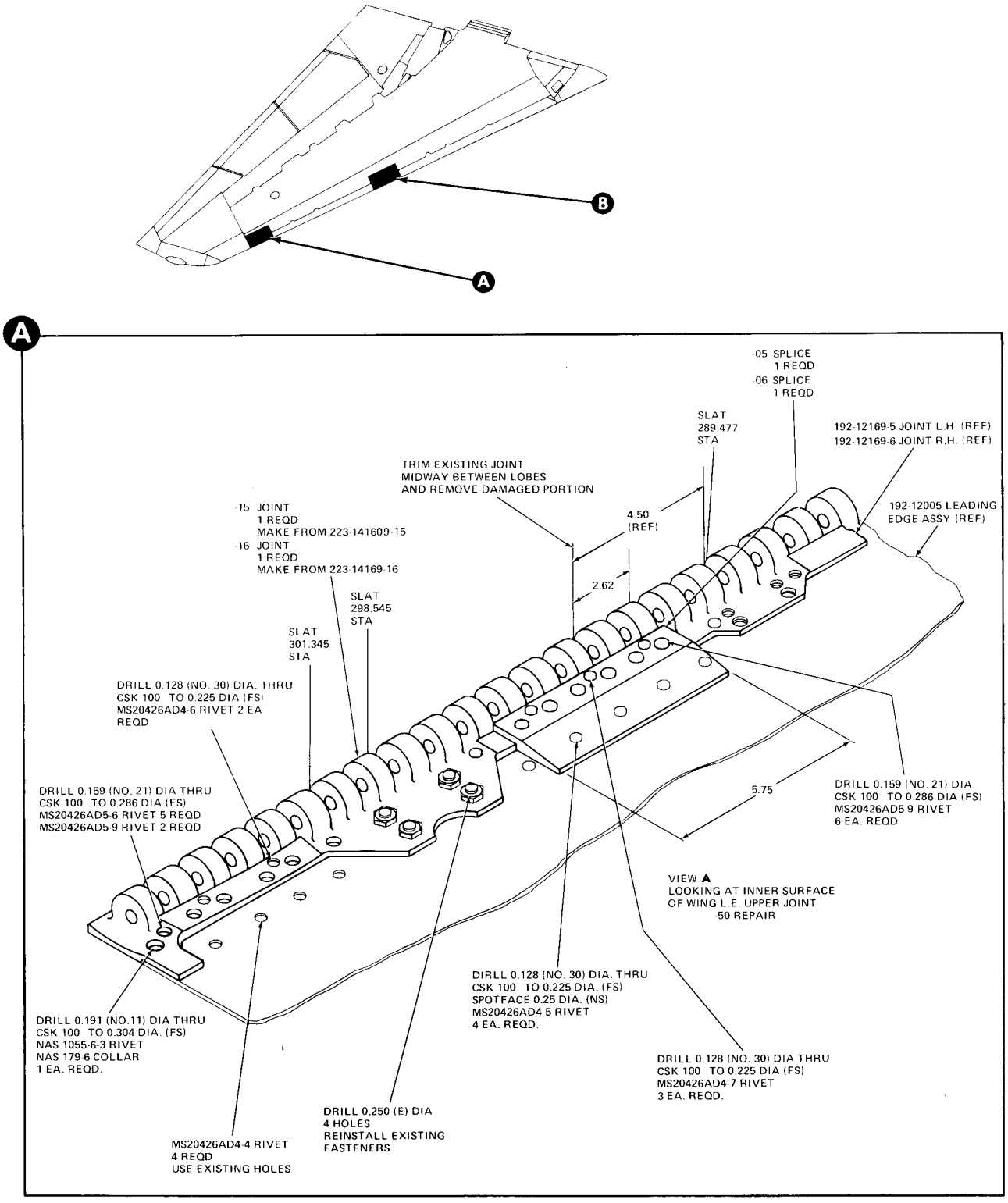
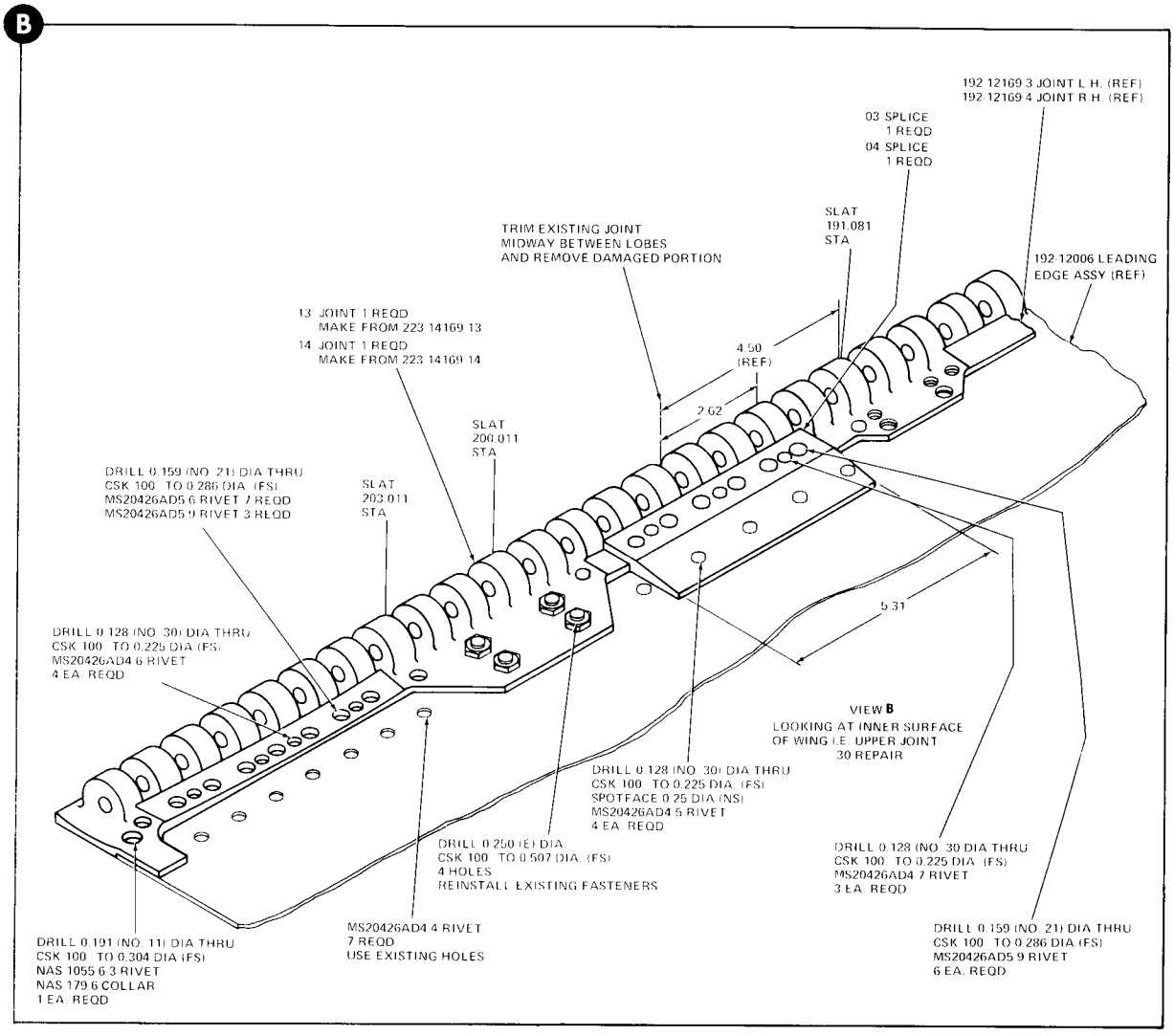


Figure 2-23D. Wing Repair Leading Edge Upper Joint (Sheet 1 of 2)



QTY REQD						LIST OF MATERIALS				
60	50	40	30		SYM	NOMENCLATURE	CODE IDENT	IDENTIFYING NO	SPECIFICATION	MATERIAL
						WING REPAIR	98749	7334889		
						WING REPAIR		30		
						WING REPAIR		40		
						WING REPAIR		50		
						WING REPAIR		60		
			1			SPLICE		03	QQ-A-200/11 7075-T6	0.25 THK 1.50 x 6.00
		1				SPLICE		04	"	"
	1					SPLICE		05	"	"
1						SPLICE		06	QQ-A-200/11 7075-T6	0.25 THK 1.50 x 6.00
			1			JOINT		13	MAKE FROM 223-14169-13	
		1				JOINT		14	MAKE FROM 223-14169-14	
	1					JOINT		15	MAKE FROM 223-14169-15	
1						JOINT	98749	16	MAKE FROM 223-14169-16	

Figure 2-23D. Wing Repair Leading Edge Upper Joint (Sheet 2 of 2)

SLAT TRACK IN-SERVICE WEAR TOLERANCE AND REPAIR LIMITS

Slat tracks worn less than 0.0075 inch on the bearing surface may be considered serviceable.

Worn areas may be chrome plated to a maximum of 0.015 inch plating thickness, in accordance with established procedures. (Refer to Section XI ELECTROPLATING, this manual and T.O. 42C-1-7.)

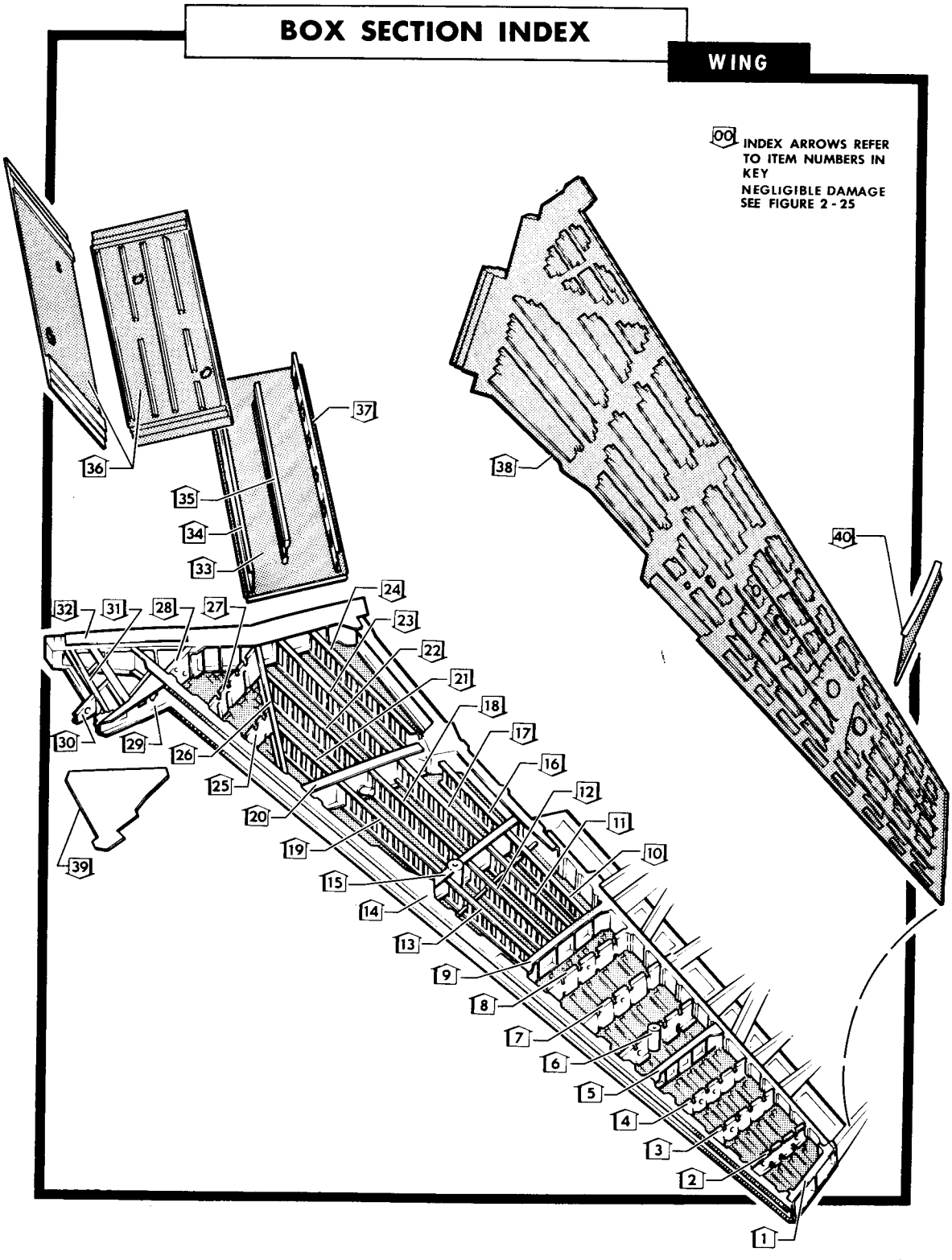


Figure 2-24. Box Section Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-14060	Rib		7075-T6 Die Forging	
2	223-14059	Rib Assy	0.040	7075-T6 Clad	2-33
3	223-14058	Rib Assy	0.040	7075-T6 Clad	2-33
4	223-14057	Rib Assy	0.040	7075-T6 Clad	2-33
5	223-14056	Rib		356-T6 Sand Casting	
6	223-14055	Support		7075-T6 Die Forging	
7	223-14051	Rib Assy	0.040	7075-T6 Clad	2-33
8	223-14054	Rib Assy	0.040	7075-T6 Clad	2-33
9	223-14053	Rib Assy		356-T6 Sand Casting	
10	223-14115	Spar	0.125	7075-T6 Clad	2-28
11	223-14111	Spar	0.063	7075-T6 Clad	2-28
12	223-14107	Spar Assy	0.032	7075-T6 Clad	2-28
13	223-14104	Spar Assy	0.032	7075-T6 Clad	2-28
14	223-14100	Front Spar Assy			2-29, 2-30, 2-31, 2-32, 2-33
15	223-14052	Spar Assy	0.032	7075-T6 Clad	
16	192-12114	Spar Assy	0.032	7075-T6 Clad	2-28
17	223-14110	Spar Assy	0.032	7075-T6 Clad	2-28
18	223-14106	Spar Assy	0.032	7075-T6 Clad	2-28
19	223-14103	Spar Assy	0.032	7075-T6 Clad	2-28
20	223-14050	Rib Assy	0.125	7075-T6 Clad	
21	192-12102	Spar Assy	0.032	7075-T6 Clad	2-28
22	192-12105	Spar Assy	0.032	7075-T6 Clad	2-28
23	192-12107	Spar Assy	0.040	7075-T6 Clad	2-28
24	192-12113	Spar Assy	0.040	7075-T6 Clad	2-28
25	223-14067	Rib Assy	0.063	2024-T4	2-23
26	223-14118	Spar Assy	0.102	7075-T6 Clad	2-28
27	223-14066	Rib Assy	0.100	7075-T6 Clad	2-23
28	223-14065	Rib Assy		7075-T6 Bar Plate	2-23
29	223-14240	Rib		2024-T6 Die Forging	
30	223-14224	Rib	0.064	7075-T4 Clad	2-23
31	223-14117	Intercostal		7075-T6 Die Forging	
32	192-12371	Rib Assy			2-26
33	223-13010	Skin—Lwr Assy	¾	7075-T6 Plate	
33A‡	2629-130008-11	Skin—Lwr Assy	¾	7075-T6 Plate	
34	223-13003	Spar—Front Assy			
35	223-13005	Spar—Ctr Assy			2-28
36	223-13007	Skin—Up.	¾	7075-T6 Plate	
37	223-13004	Spar—Rear Assy			
38	223-14007	Skin—Upper	Milled	7075-T6 Plate	2-34, 2-34A, 2-35,
	223-14008	Skin—Lower	Milled	7075-T6 Plate	2-36, 2-37, 2-38,
	2629-140144-1, -2§	Skin—Lower	Milled	7075-T6 Plate	2-39, 2-40, 2-41, 2-41A, 2-41B
39	223-14152	Skin	Milled	7075-T6 Plate	
40	223-14866	Fence—Wing	Extr	2014-T62	2-43A

***Drawing numbers are for reference only and are not to be used for ordering parts.**

†For additional repairs, refer to Section X.

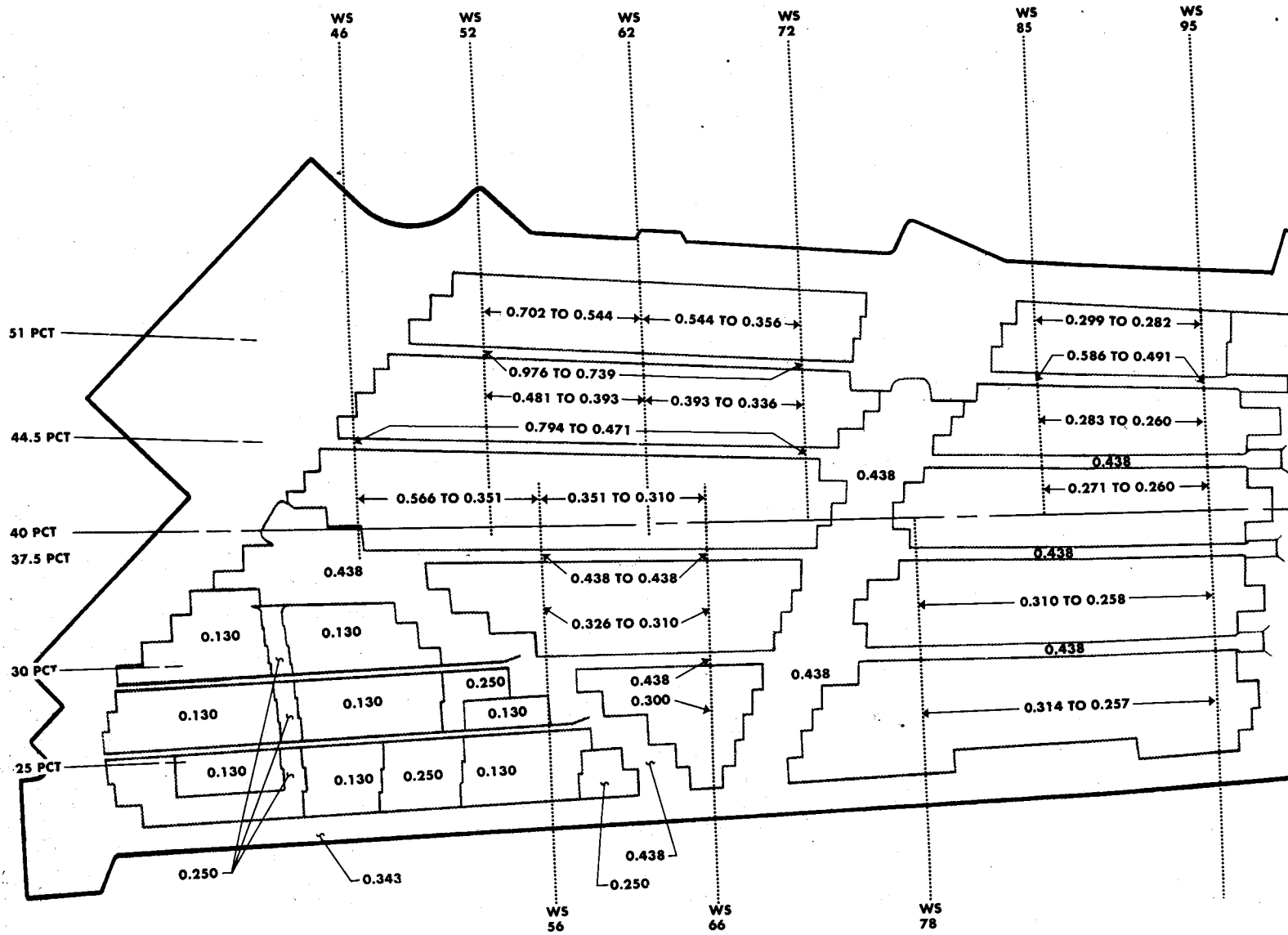
‡Airplanes changed by T.O. 1F-100-1028.

§Airplanes changed by T.O. 1F-100-1035.

Key to Figure 2-24

WING SKIN THICKNESS DIAGRAM— WING

F-100D WING SKIN—UPPER
(VIEW LOOKING IN)



NOTE For clarity, the thicknesses of only the larger areas are shown. To obtain the thickness of any specific area not shown, refer to NAA drawing No. 223-14140.

- All stations shown are taken from the 40 pct line.
- Taper dimensions are given from wing station to wing station to establish the taper. The taper does not end at the wing stations but continues on to the rib land. The tapers are uniform except where local variances may interfere with the taper at points along the length of the taper.

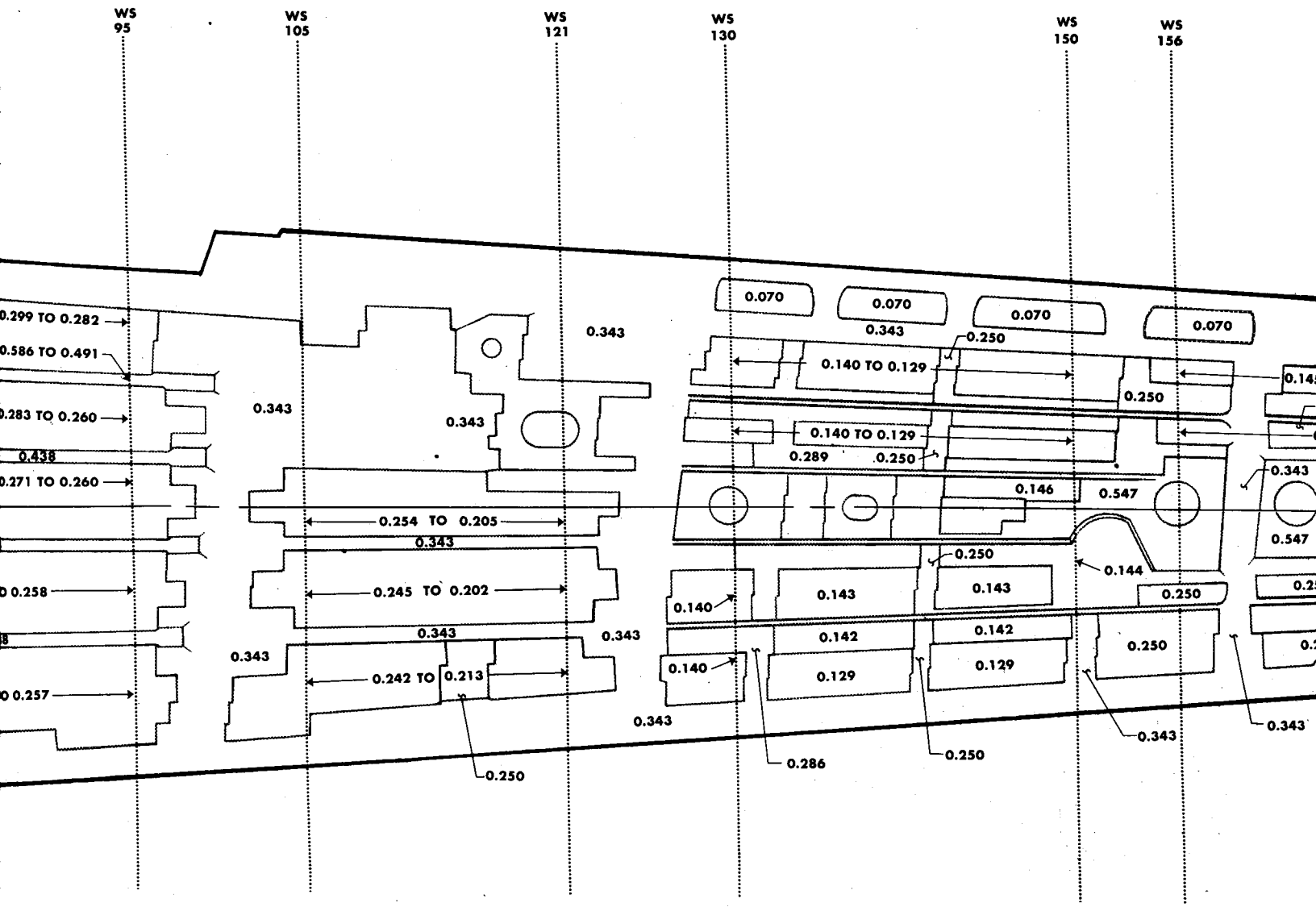


Figure 2-24A. Upper Wing

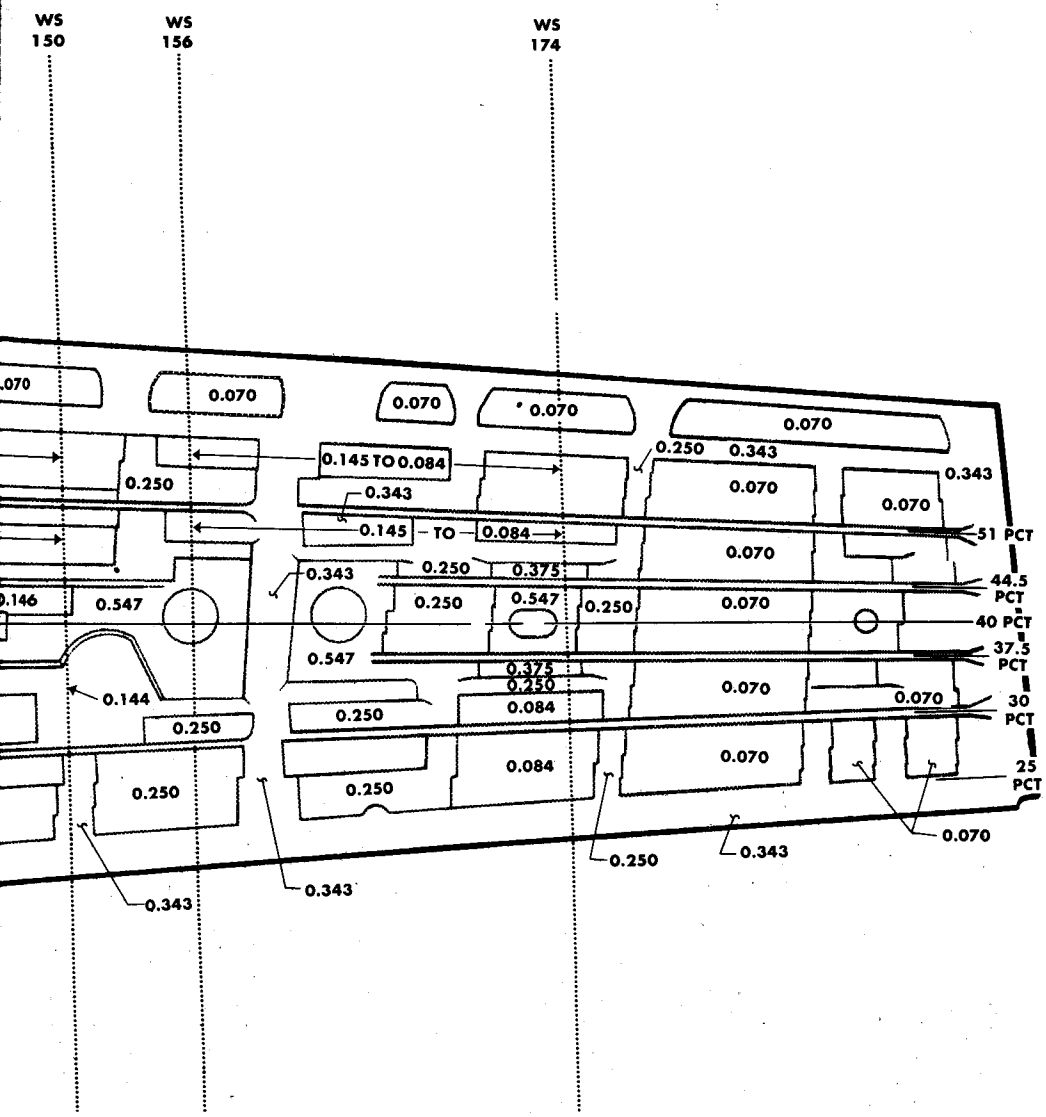
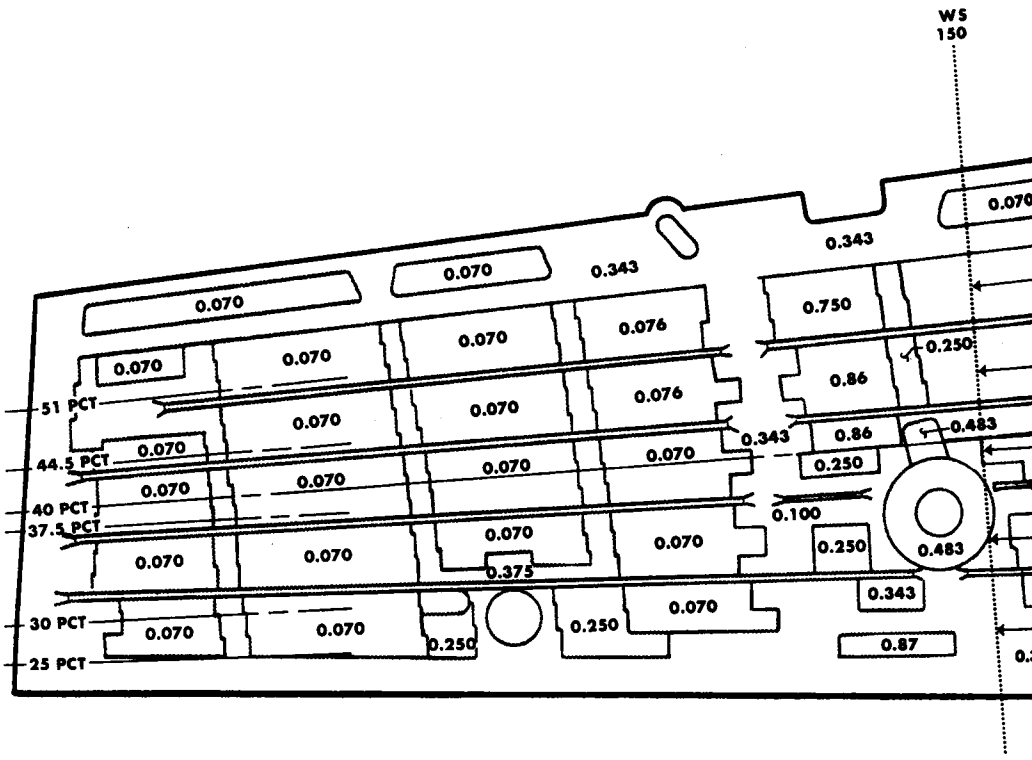


Figure 2-24A. Upper Wing Skin Thickness Diagram

WING SKIN THICKNESS DIAGRAM

F-100D WING SKIN—LOWER
(VIEW LOOKING IN)



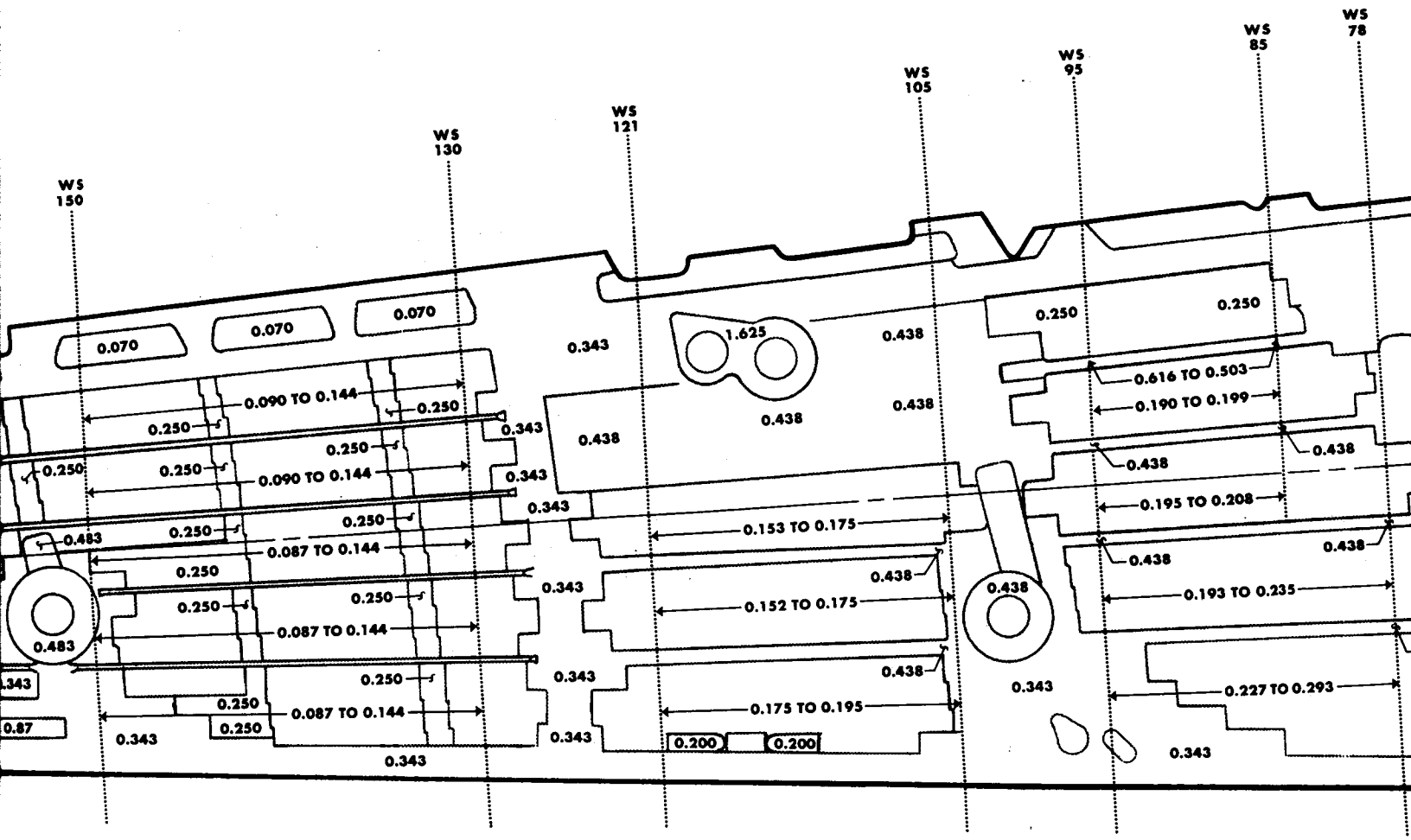
NOTE For clarity, the thicknesses of only the larger areas are shown. To obtain the thickness of any specific area not shown, refer to NAA drawing No. 223-14142.

- All stations shown are taken from the 40 pct line.
- Taper dimensions are given from wing station to wing station, to establish the taper. The taper does not end at the wing stations, but continues on to the rib land. The tapers are uniform, except where local variances may interfere with the taper at points along the length of the taper.

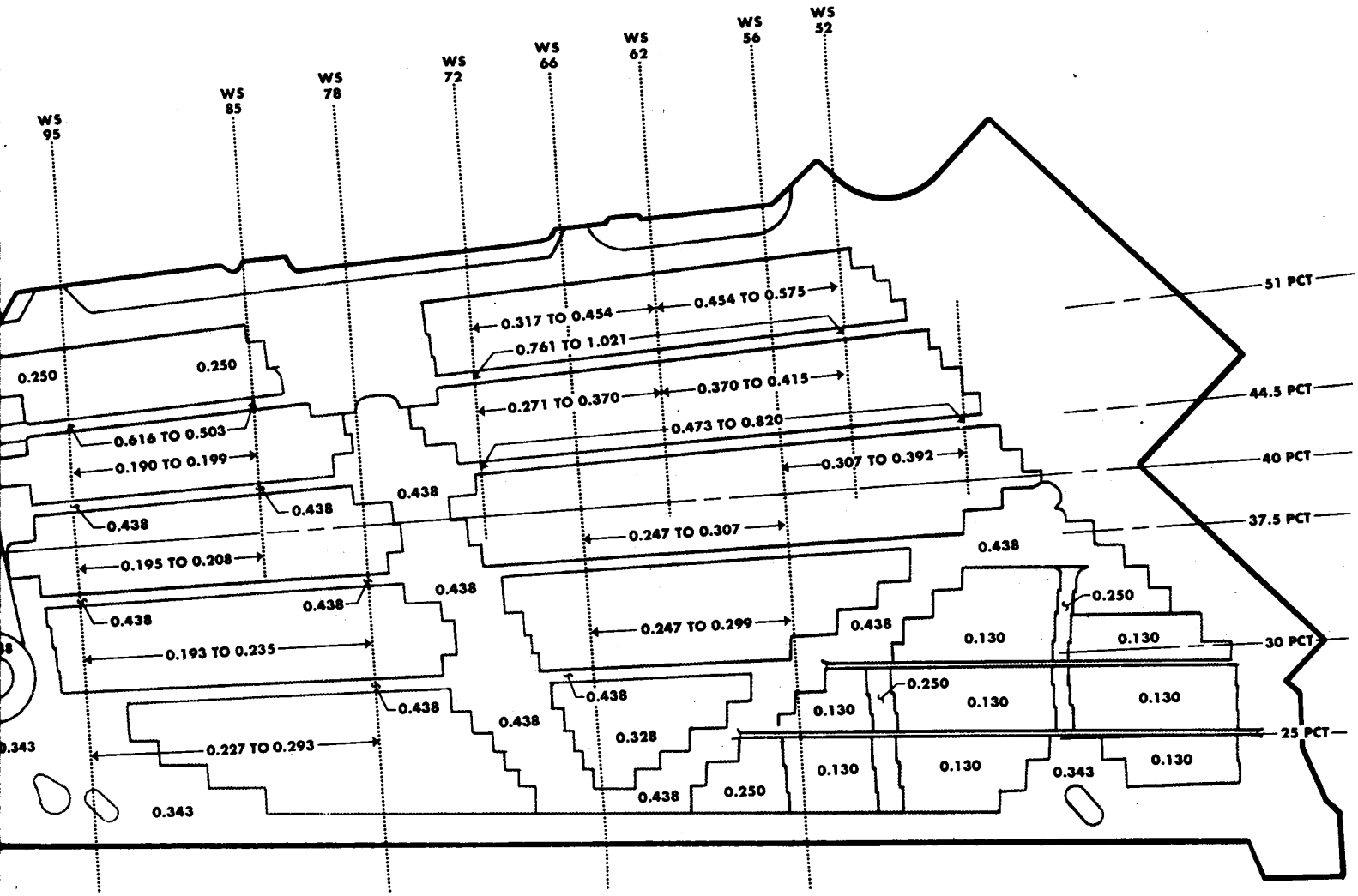
Figure 2-24A-1. Lower Wing Skin Thickness Diagram—Airplanes Not Changed

GRAM

WING

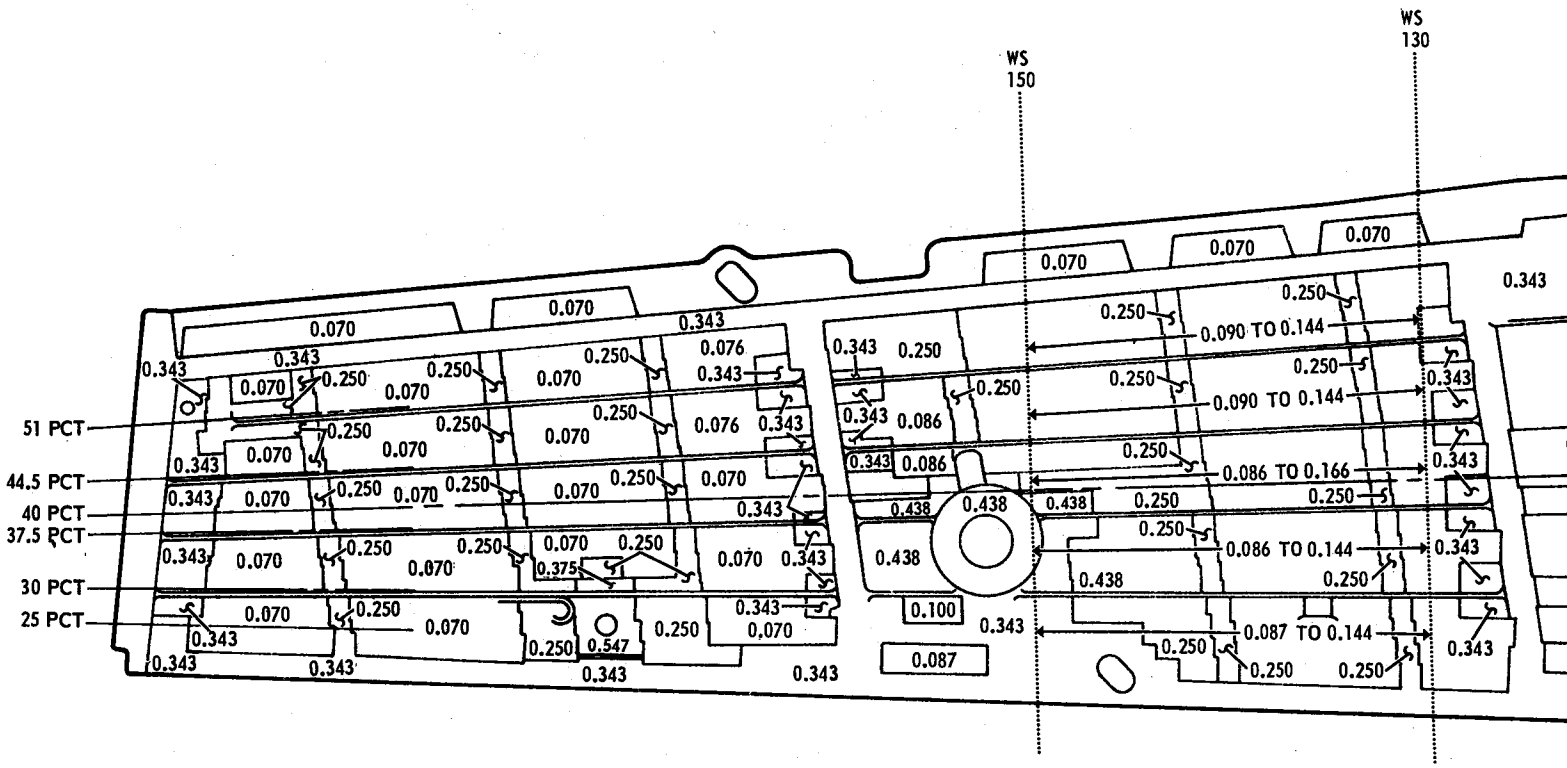


F-100D-3-10-182


WS 95
WS 85
WS 78
WS 72
WS 66
WS 62
WS 56
WS 52
51 PCT
44.5 PCT
40 PCT
37.5 PCT
30 PCT
25 PCT
0.317 TO 0.454
0.454 TO 0.575
0.761 TO 1.021
0.271 TO 0.370
0.370 TO 0.415
0.473 TO 0.820
0.307 TO 0.392
0.247 TO 0.307
0.247 TO 0.299
0.616 TO 0.503
0.190 TO 0.199
0.195 TO 0.208
0.193 TO 0.235
0.227 TO 0.293
0.250
0.438
0.130
0.328
0.343

WING SKIN THICKNESS DIAGRAM

F-100D WING SKIN - LOWER
(VIEW LOOKING IN)



NOTE For clarity, the thicknesses of only the larger areas are shown. To obtain the thickness of any specific area not shown, refer to NR drawing No. 2629-140144.

- All stations shown are taken from the 40 percent line.
- Taper dimensions are given from wing station to wing station, to establish the taper. The taper does not end at the wing station, but continues on to the rib land. The tapers are uniform, except where local variances may interfere with the taper at points along the length of the taper.

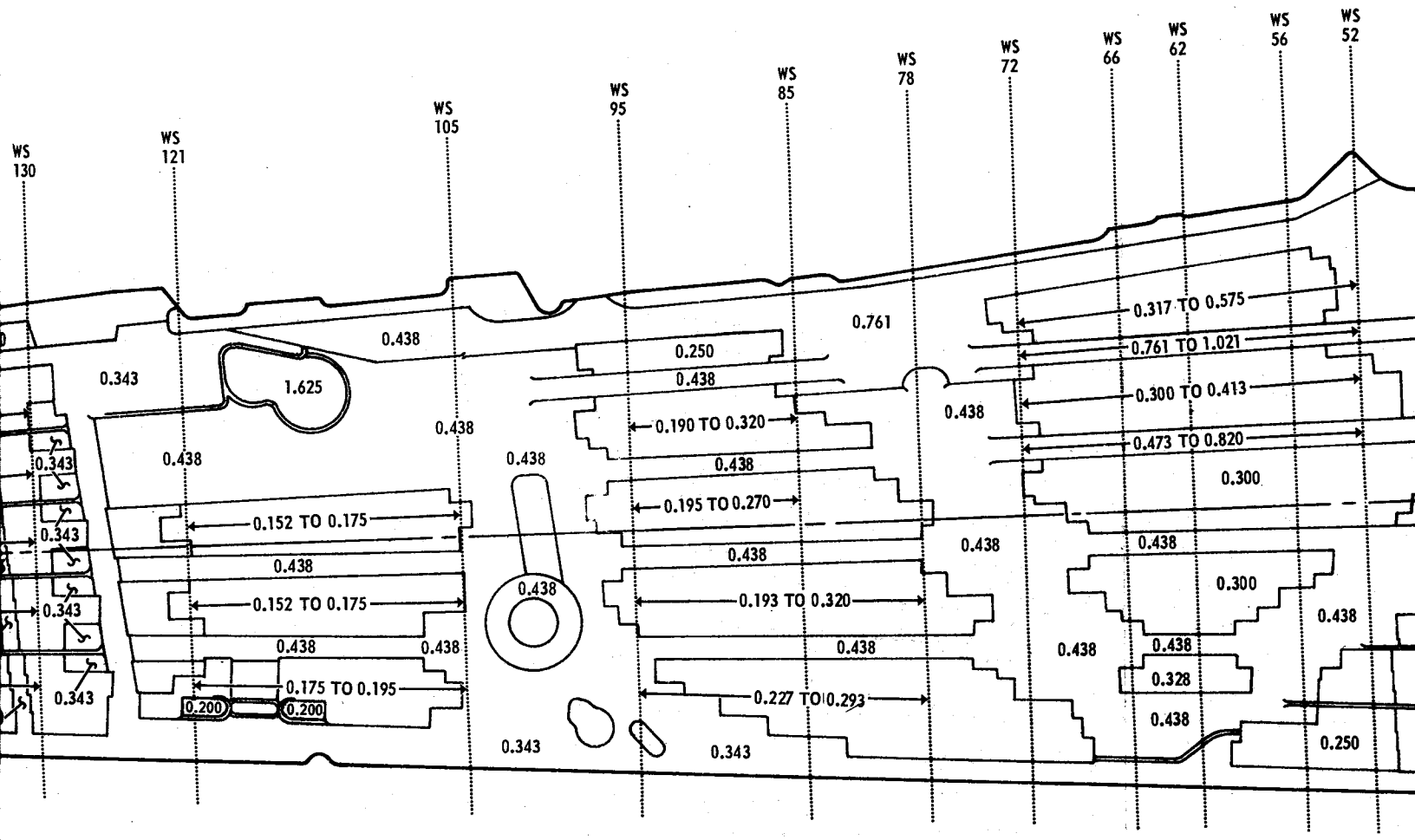
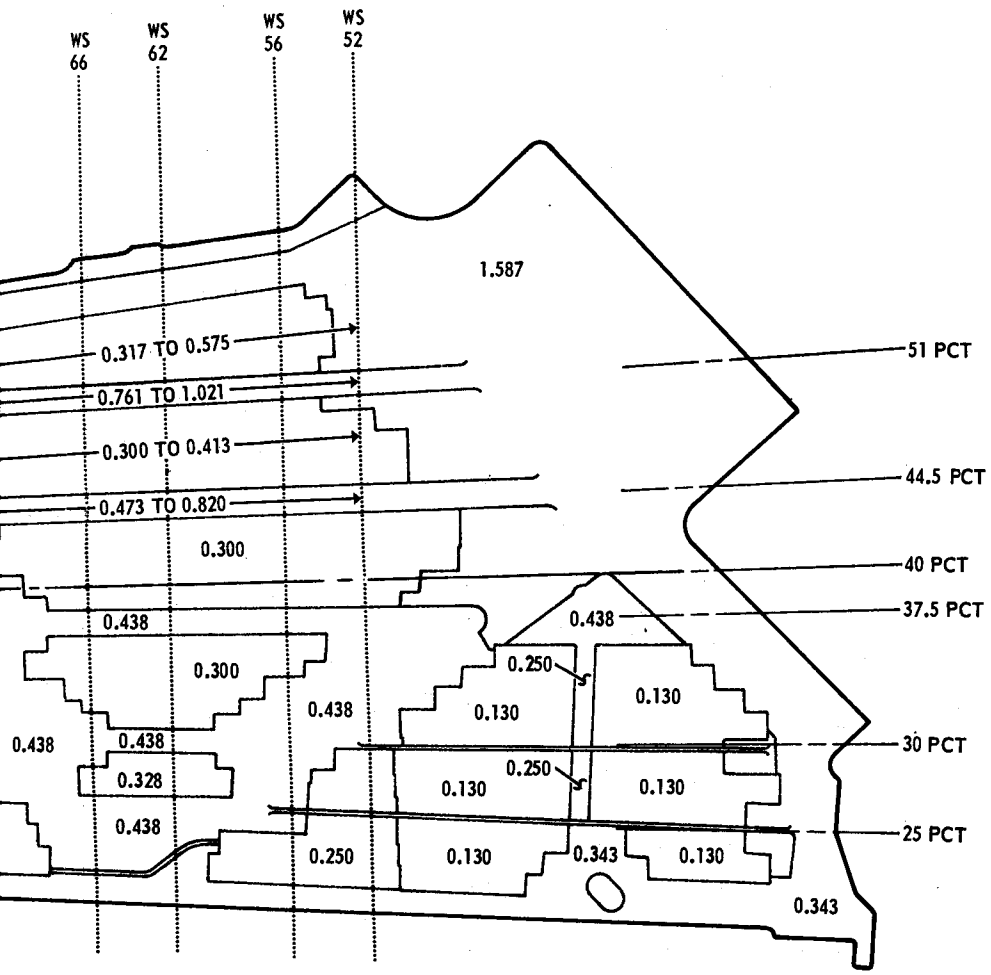


Figure 2-24A-2. Lower Wing Skin Thickness Diagram—A



100D-3-10-196

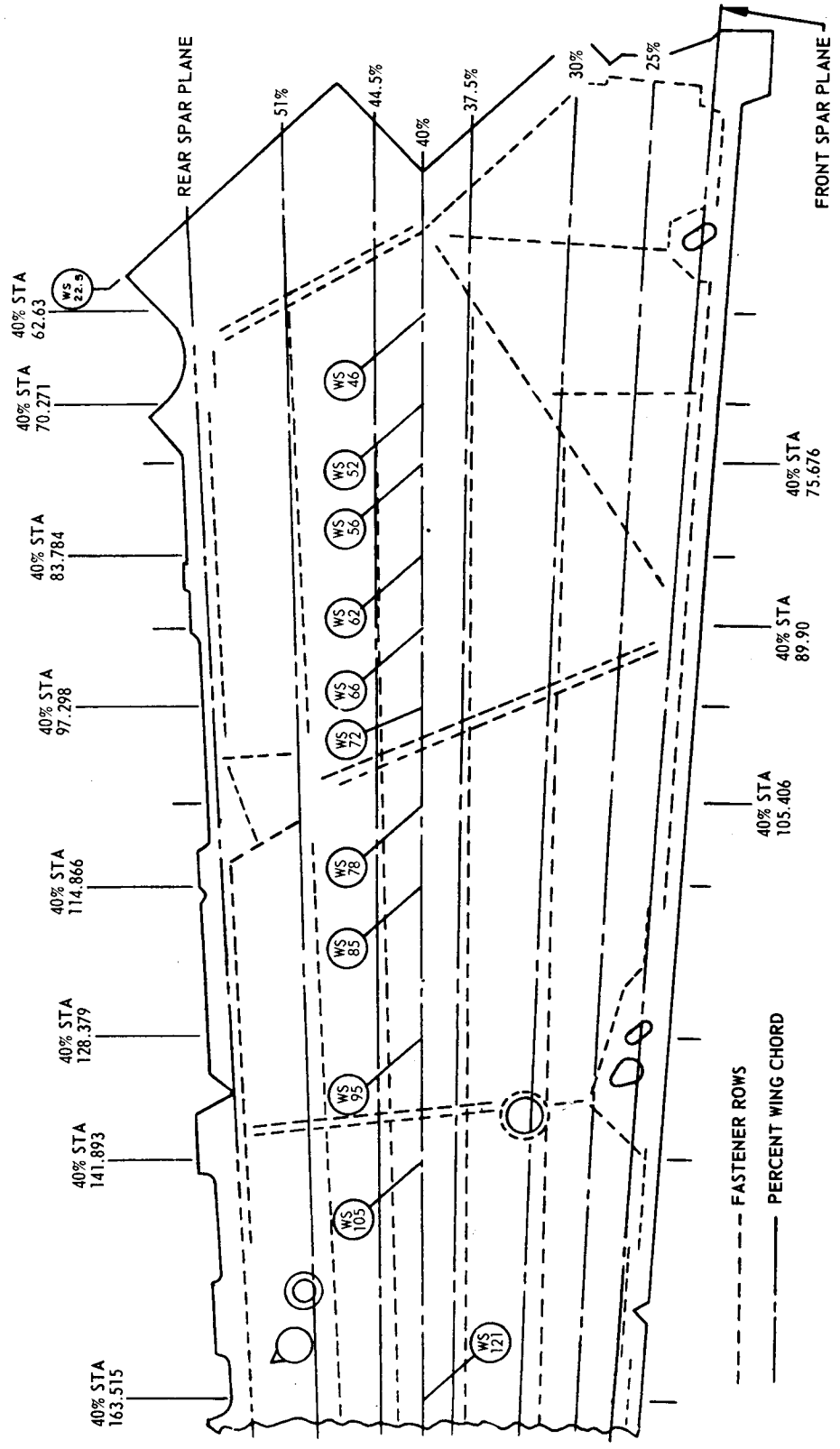
er Wing Skin Thickness Diagram—Airplanes Changed by T.O. 1F-100-1035

Changed 22 November 1968

WING SKIN CORROSION REMOVAL LIMITS

WING

WING SKIN—LOWER



NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit $= \frac{0.120}{3} = 0.040$ inch.

F-100D-3-10-188

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 1 of 7)

WING FILLETS CHAFING LOWER WING, SKIN, F.S. 267 AND W.S. 38.0 (T.O. 1F-100-1035, NEW SKINS ONLY).

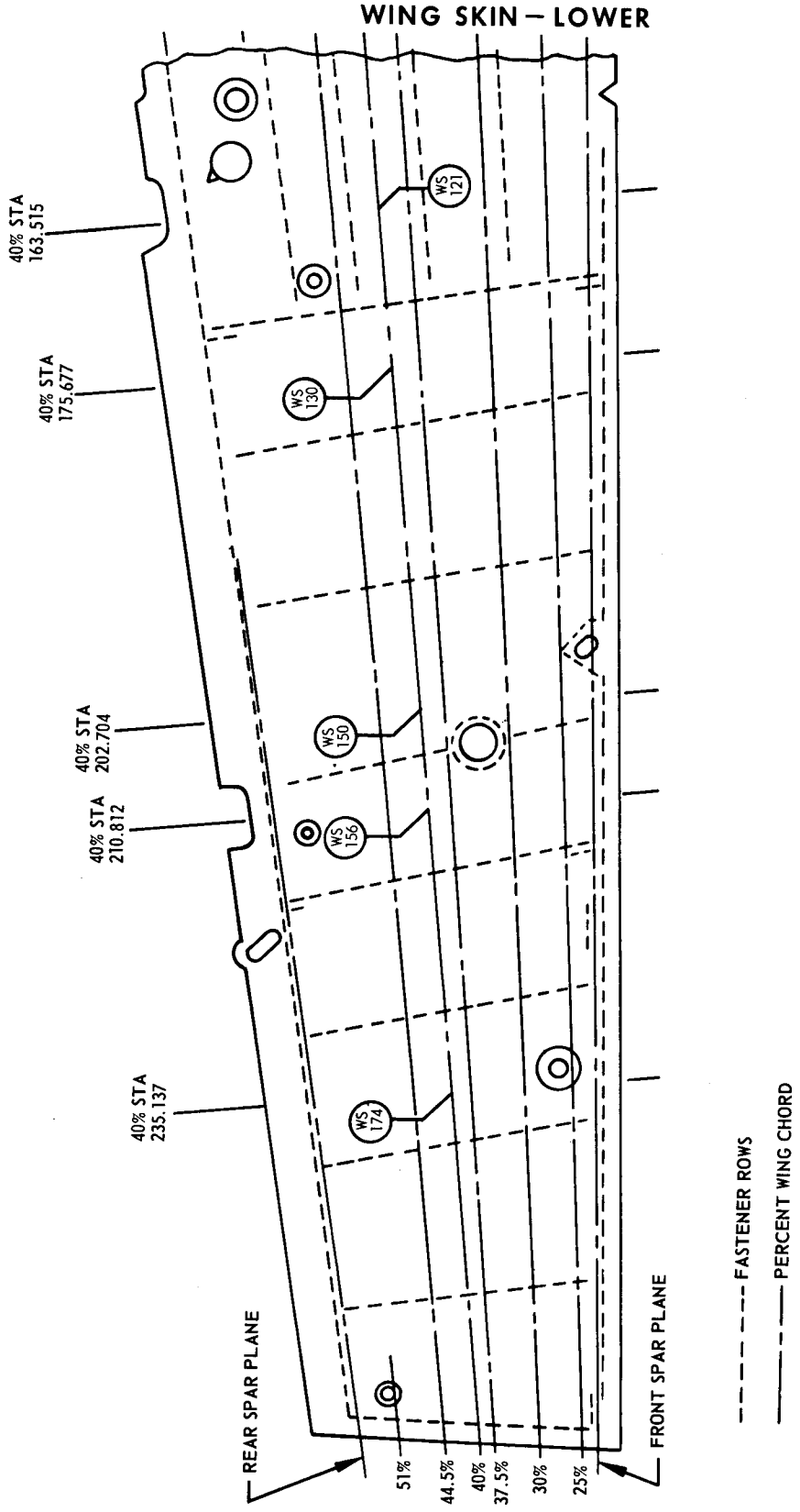
1. If the fuselage to lower wing skin fillet, 2629-310008-1 and -2 aft edge of the forward tab, shows evidence of chafing the lower wing skin, it can be corrected as follows:

a. The tab may be reworked by springing the tab down from the wing skin and twisting until it lays flat.

b. The chafing gouge should be ground and burnished and filled with aero-dynamic skin filler, PR-341.

WING SKIN CORROSION REMOVAL LIMITS

WING



NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit = $\frac{0.120}{3}$ = 0.040 inch.

F-100D-3-10-189

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 2 of 7)

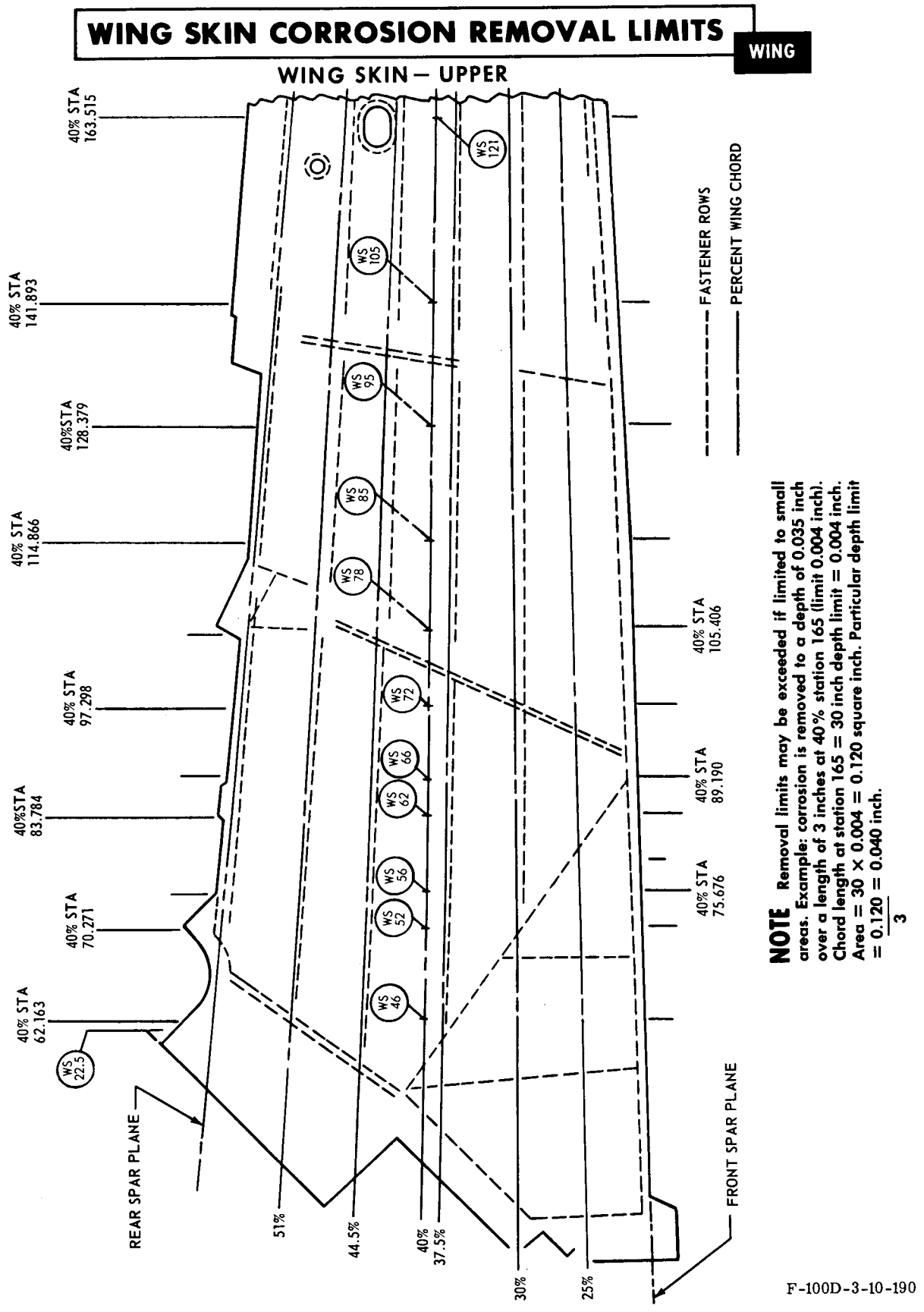
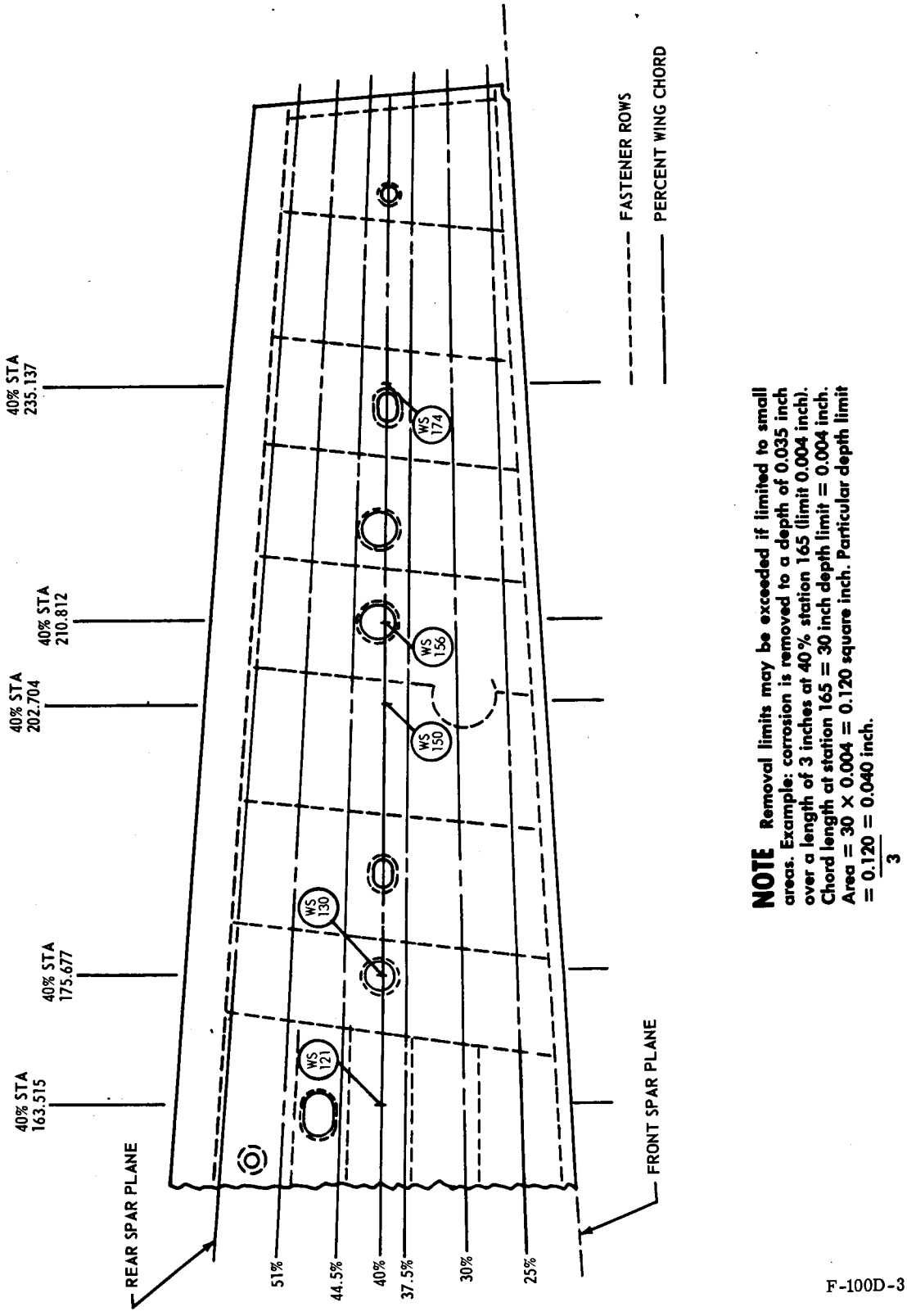


Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 3 of 7)

WING SKIN CORROSION REMOVAL LIMITS

WING



NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit = $\frac{0.120}{3}$ = 0.040 inch.

F-100D-3-10-191

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 4 of 7)

WING SKIN CORROSION REMOVAL LIMITS

WING

LOCATION BETWEEN 40% STA'S	AND	WING SKIN LOWER MAX SKIN REMOVAL INCHES	WING SKIN UPPER MAX SKIN REMOVAL INCHES
40.4	62.6	0.051	0.051
62.6	78.2	↕	0.035
78.2	100.1	↕	0.026
100.1	110.0	0.051	↕
110.0	121.4	0.049	0.026
121.4	133.4	0.008	0.024
133.4	145.7	0.015	0.023
145.7	157.9	0.031	0.026
157.9	170.1	0.027	0.004
170.1	182.0	0.022	↕
182.0	194.0	0.018	0.004
194.0	206.1	0.010	0.005
206.1	217.2	0.005	0.015
217.2	228.1	——	0.013
228.1	239.2	0.010	0.026
239.2	251.2	0.033	0.034
251.2	263.5	0.033	0.033

NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit = $\frac{0.120}{3}$ = 0.040 inch.

F-100D-3-10-192

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 5 of 7)

WING SKIN CORROSION REMOVAL LIMITS

WING

LOCATION BETWEEN 40% STA'S	AND	WING SKIN LOWER		WING SKIN UPPER	
		AT FRONT SPAR	AT REAR SPAR	AT FRONT SPAR	AT REAR SPAR
		MAX SKIN REMOVAL AROUND FASTENERS	MAX SKIN REMOVAL AROUND INCHES	MAX SKIN REMOVAL AROUND FASTENERS	MAX SKIN REMOVAL AROUND INCHES
41.5	62.6	0.051	—	0.051	—
62.6	67.1		—		—
67.1	78.2		0.051		0.051
78.2	100.1				
100.1	104.0				
104.0	114.1		0.040		0.040
114.1	121.4		0.051		0.051
121.4	130.9				
130.9	133.4			0.035	
133.4	145.7			0.051	
145.7	157.9				
157.9	170.1				
170.1	182.0				
182.0	194.0				
194.0	206.1				
206.1	217.2				
217.2	228.1				
228.1	239.2				
239.2	251.2				
251.2	263.5				
263.5	265.5		—		—

NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit = $\frac{0.120}{3}$ = 0.040 inch.

F-100D-3-10-193

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 6 of 7)

WING SKIN CORROSION REMOVAL LIMITS

WING

LOCATION ALONG	WING SKIN LOWER	WING SKIN UPPER
	MAX SKIN REMOVAL AROUND FASTENERS INCHES	MAX SKIN REMOVAL AROUND FASTENERS INCHES
30% SPAR	0.051	0.051
37.5% SPAR		0.051 INBD CWS 0.015 OUTBD 102.315
44.5% SPAR		0.026 INBD CWS 0.015 OUTBD 102.315
51% SPAR		0.035 INBD CWS 0.010 OUTBD 102.315
DIAGONAL SPAR		0.026
ROOT RIB		0.051
LE STA 96.0		
WC STA 107.013		
WC STA 116.730		0.051
WC STA 102.315	0.051	0.026
WC STA 184.240	0.015	0.023
AIL STA 130.4	0.022	0.004
AIL STA 140.0	0.018	0.004
AIL STA 152.75	0.010	0.005
AIL STA 166.5	0.005	0.015
AIL STA 175.875	—	0.013
AIL STA 188.0	0.010	0.026
AIL STA 199.25	0.033	0.034
AIL STA 211.25		
LE STA 320.0	0.033	0.034

NOTE Removal limits may be exceeded if limited to small areas. Example: corrosion is removed to a depth of 0.035 inch over a length of 3 inches at 40% station 165 (limit 0.004 inch). Chord length at station 165 = 30 inch depth limit = 0.004 inch. Area = 30 × 0.004 = 0.120 square inch. Particular depth limit = $\frac{0.120}{3}$ = 0.040 inch.

F-100D-3-10-194

Figure 2-24B. Wing Skin Corrosion Removal Limits (Sheet 7 of 7)

BOX SECTION—NEGLIGIBLE DAMAGE

WING

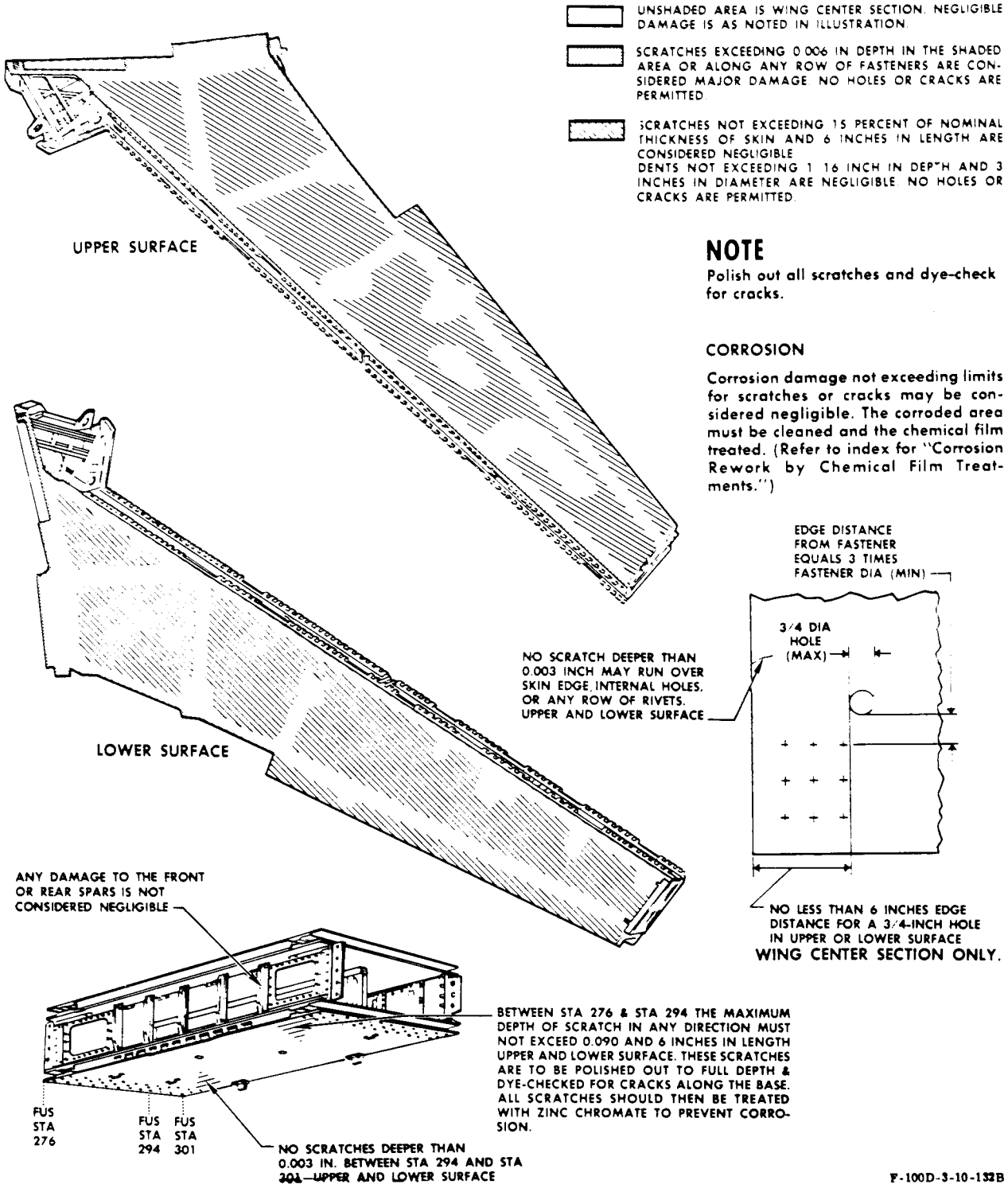
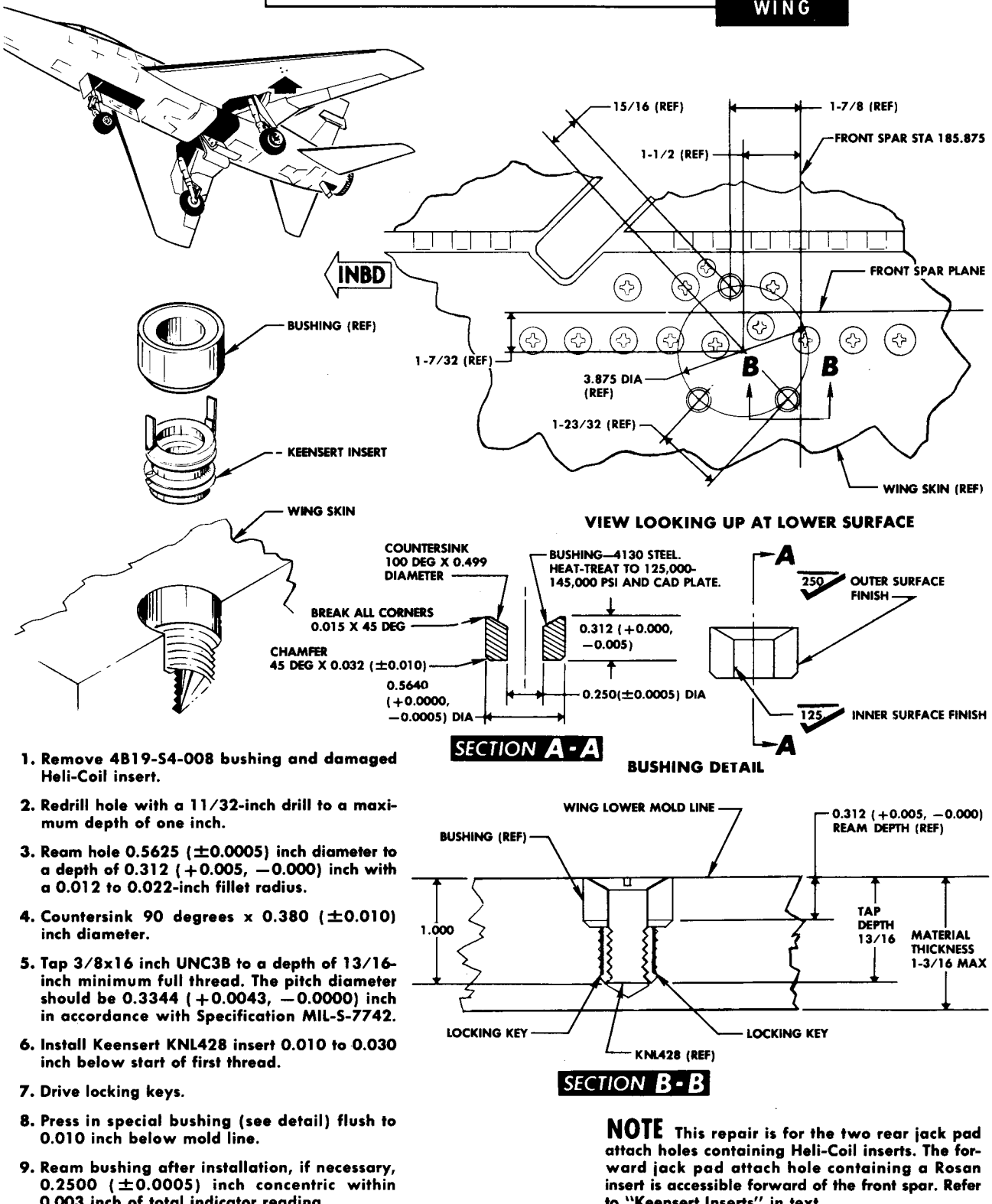


Figure 2-25. Box Section Negligible Damage

PERMANENT CATEGORY A

WING



1. Remove 4B19-S4-008 bushing and damaged Heli-Coil insert.
2. Redrill hole with a 11/32-inch drill to a maximum depth of one inch.
3. Ream hole 0.5625 (±0.0005) inch diameter to a depth of 0.312 (+0.005, −0.000) inch with a 0.012 to 0.022-inch fillet radius.
4. Countersink 90 degrees x 0.380 (±0.010) inch diameter.
5. Tap 3/8x16 inch UNC3B to a depth of 13/16-inch minimum full thread. The pitch diameter should be 0.3344 (+0.0043, −0.0000) inch in accordance with Specification MIL-S-7742.
6. Install Keensert KNL428 insert 0.010 to 0.030 inch below start of first thread.
7. Drive locking keys.
8. Press in special bushing (see detail) flush to 0.010 inch below mold line.
9. Ream bushing after installation, if necessary, 0.2500 (±0.0005) inch concentric within 0.003 inch of total indicator reading.

NOTE This repair is for the two rear jack pad attach holes containing Heli-Coil inserts. The forward jack pad attach hole containing a Rosan insert is accessible forward of the front spar. Refer to "Keensert Inserts" in text.

F-100D-3-10-163

Figure 2-25A. Heli-Coil Repair—Wing Jack Pad Attachment

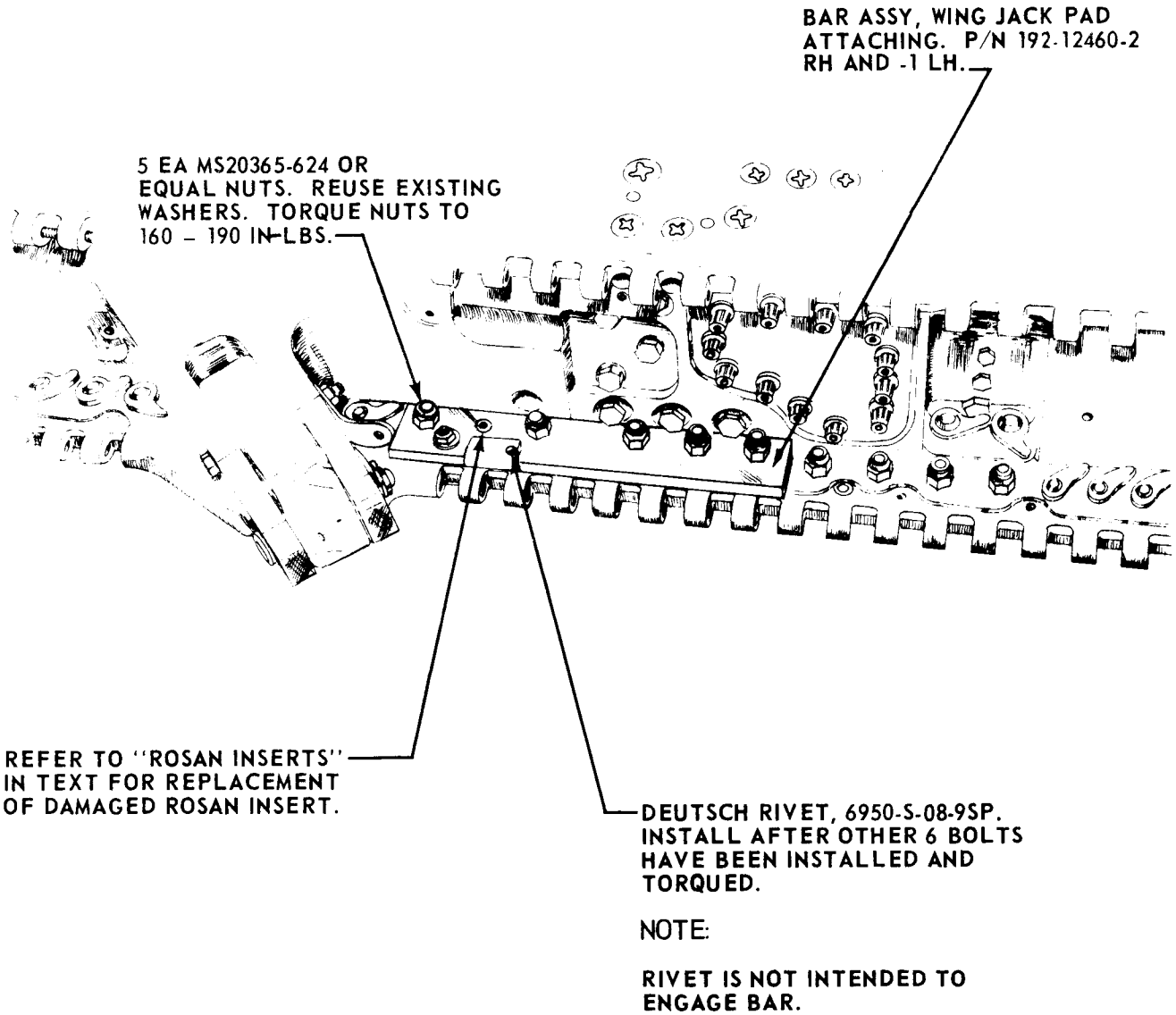
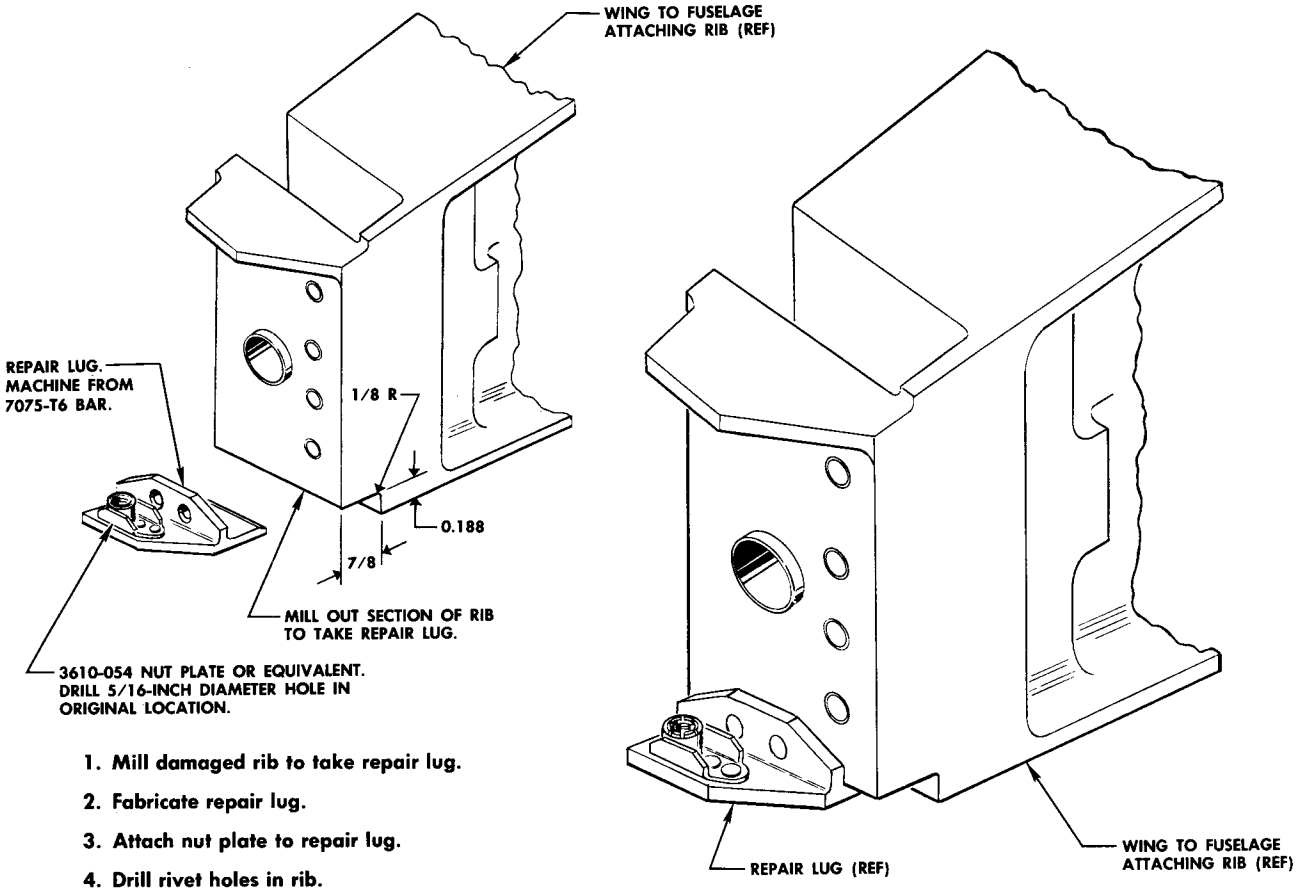


Figure 2-25B. Rework and Installation of Wing Jack Pad Attaching Bar

PERMANENT—CATEGORY A

WING



1. Mill damaged rib to take repair lug.
2. Fabricate repair lug.
3. Attach nut plate to repair lug.
4. Drill rivet holes in rib.
5. Install repair lug with rivets.

NOTE This repair is for upper and lower lugs.

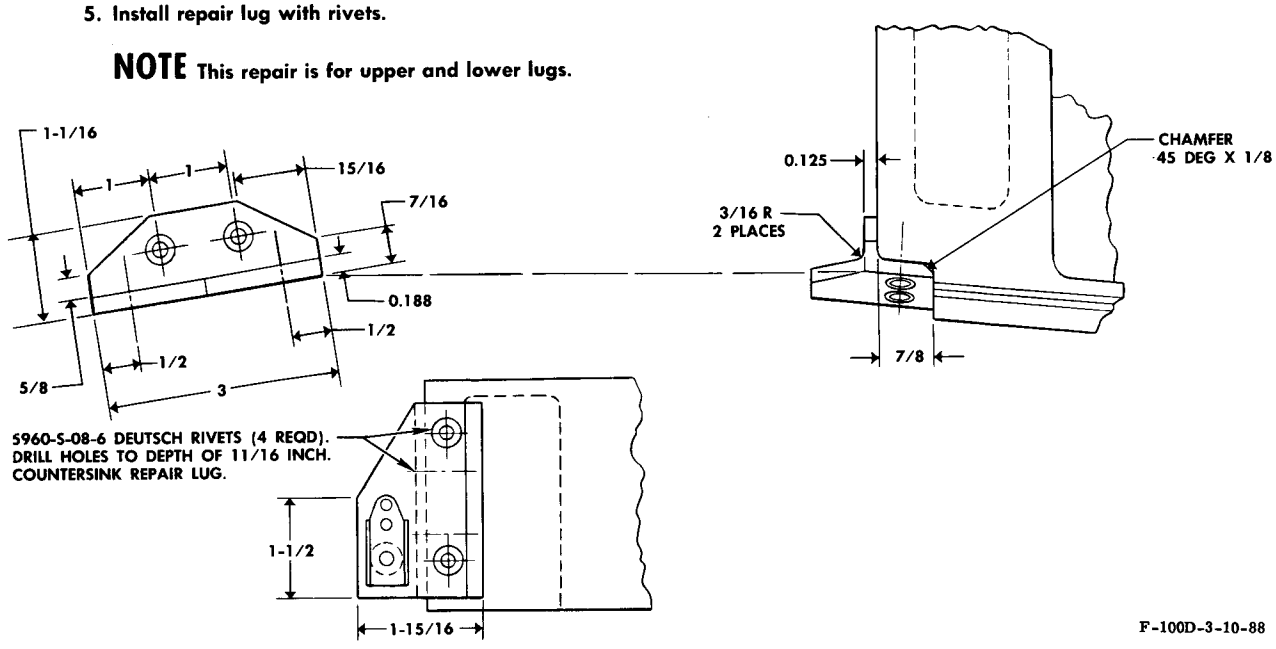
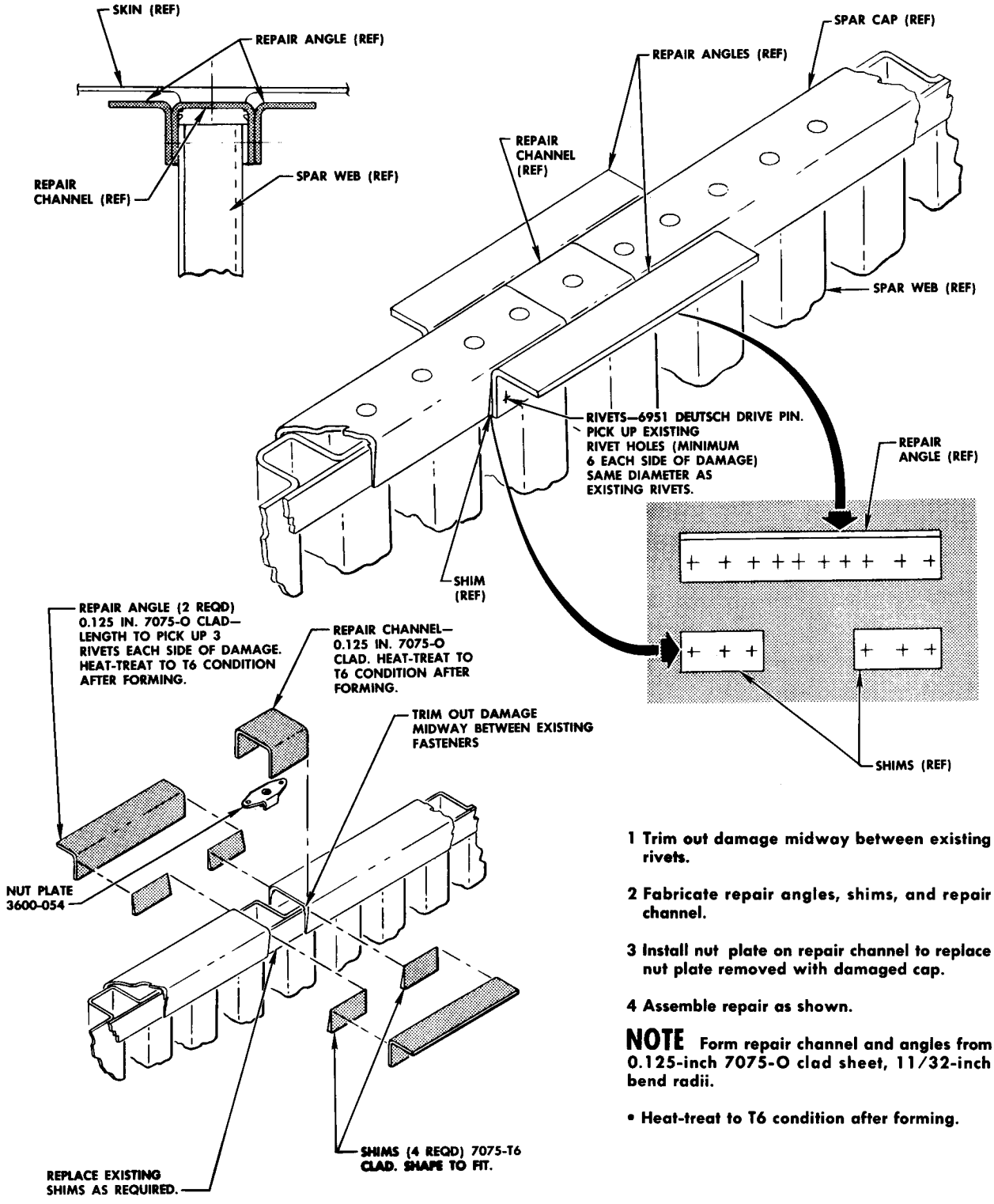


Figure 2-26. Wing-to-Fuselage Attaching Rib Lug

PERMANENT—CATEGORY B

WING



1 Trim out damage midway between existing rivets.

2 Fabricate repair angles, shims, and repair channel.

3 Install nut plate on repair channel to replace nut plate removed with damaged cap.

4 Assemble repair as shown.

NOTE Form repair channel and angles from 0.125-inch 7075-O clad sheet, 11/32-inch bend radii.

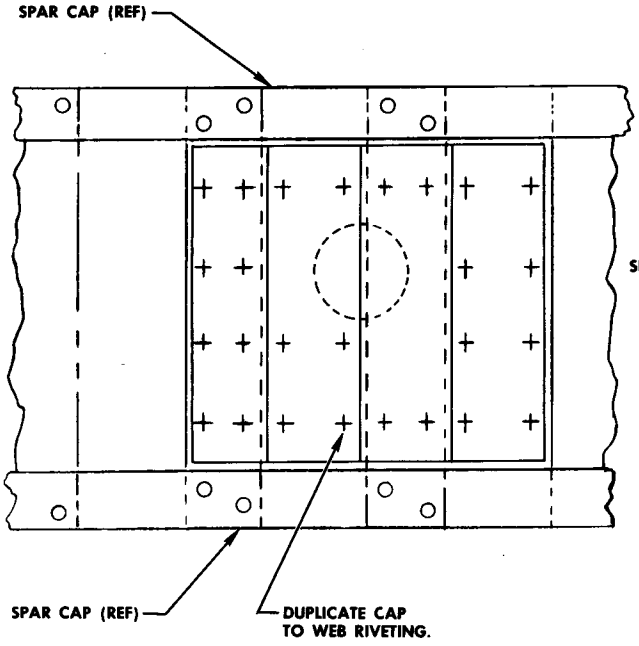
• Heat-treat to T6 condition after forming.

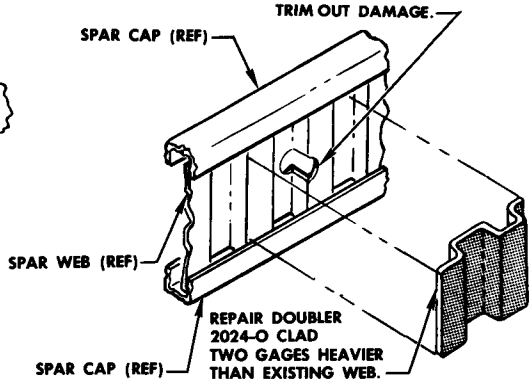
F-100D-3-10-89A

Figure 2-27. Wing Corrugated Spar Cap

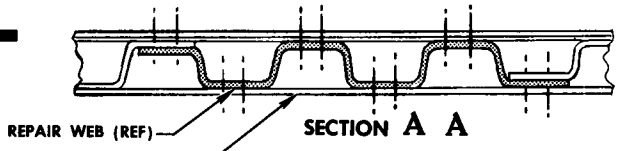
PERMANENT—CATEGORY B

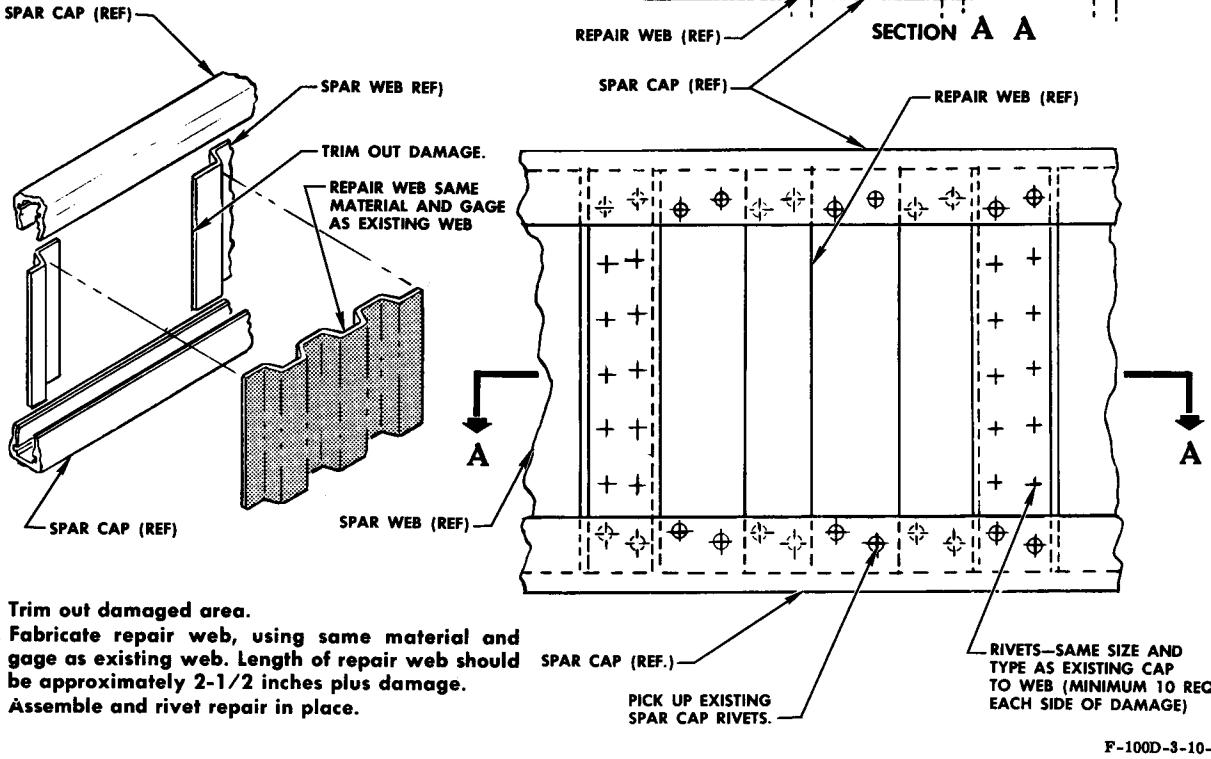
WING





1. Trim out damaged area.
2. Fabricate repair doubler, using 2024-O clad material. Make repair doubler large enough to extend a minimum of one inch above and below damage and to extend over a minimum of one undamaged spanwise web panel each side of damage.
3. Heat-treat repair doubler to T4 condition after forming.
4. Assemble and rivet repair in place.





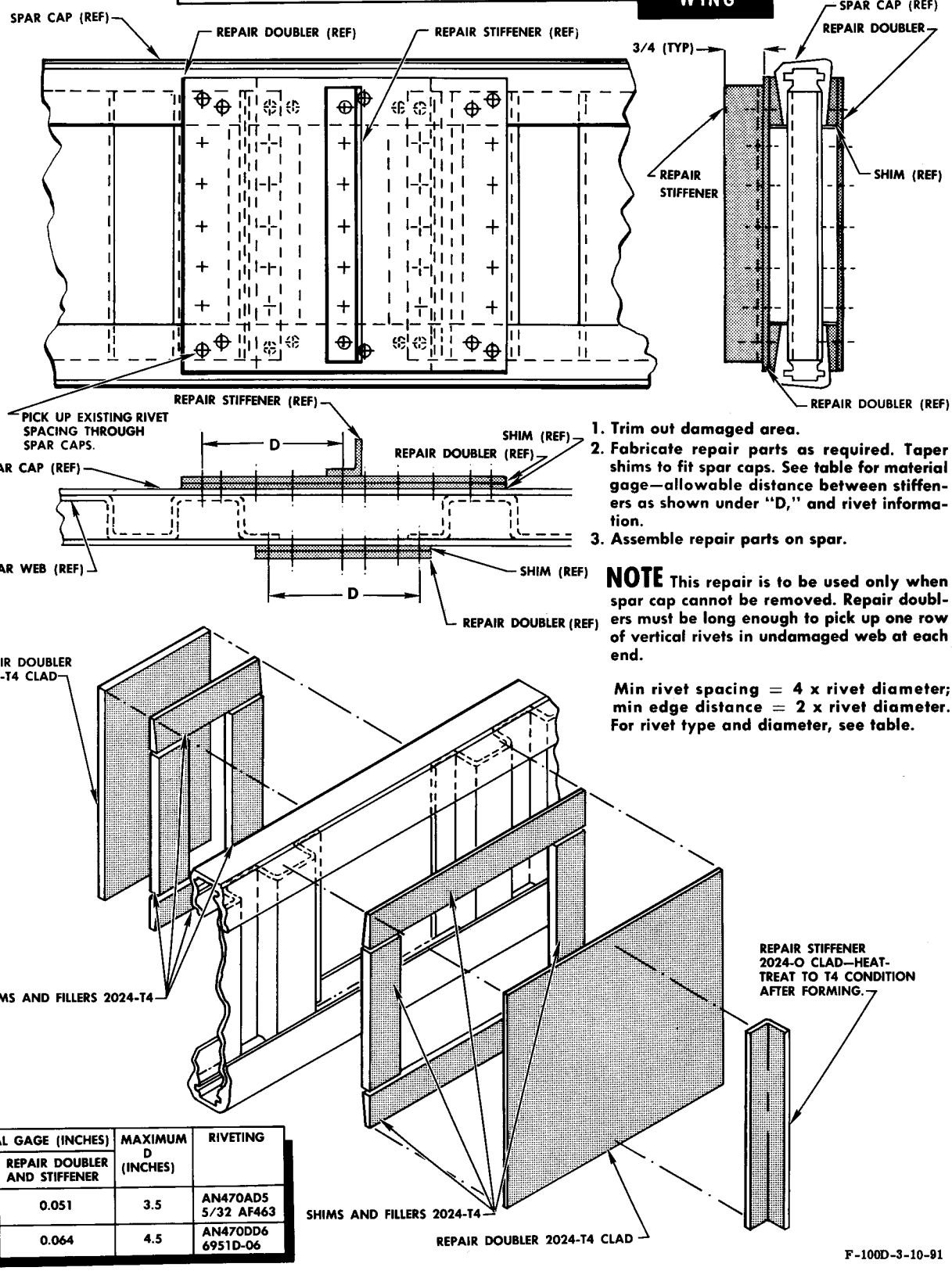
1. Trim out damaged area.
2. Fabricate repair web, using same material and gage as existing web. Length of repair web should be approximately 2-1/2 inches plus damage.
3. Assemble and rivet repair in place.

F-100D-3-10-90

Figure 2-28. Wing Corrugated Spar Web (Sheet 1 of 2)

PERMANENT—CATEGORY B

WING



1. Trim out damaged area.
2. Fabricate repair parts as required. Taper shims to fit spar caps. See table for material gage—allowable distance between stiffeners as shown under "D," and rivet information.
3. Assemble repair parts on spar.

NOTE This repair is to be used only when spar cap cannot be removed. Repair doublers must be long enough to pick up one row of vertical rivets in undamaged web at each end.

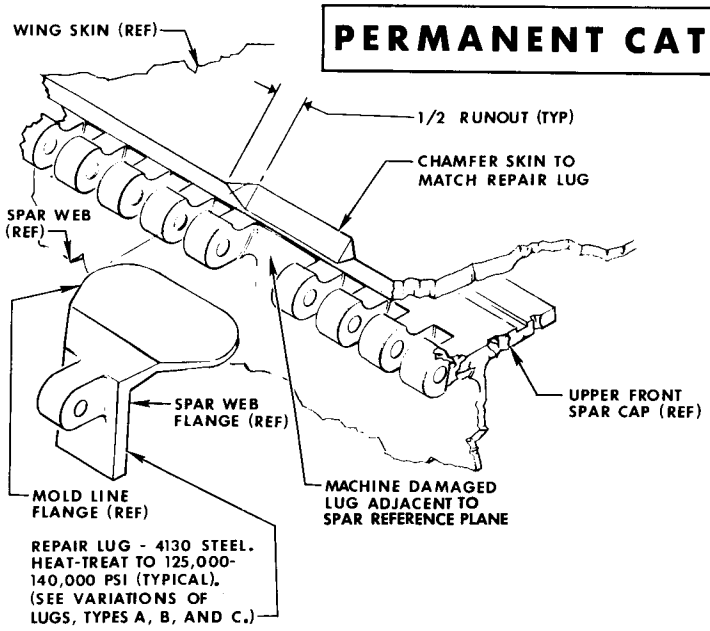
Min rivet spacing = 4 x rivet diameter; min edge distance = 2 x rivet diameter. For rivet type and diameter, see table.

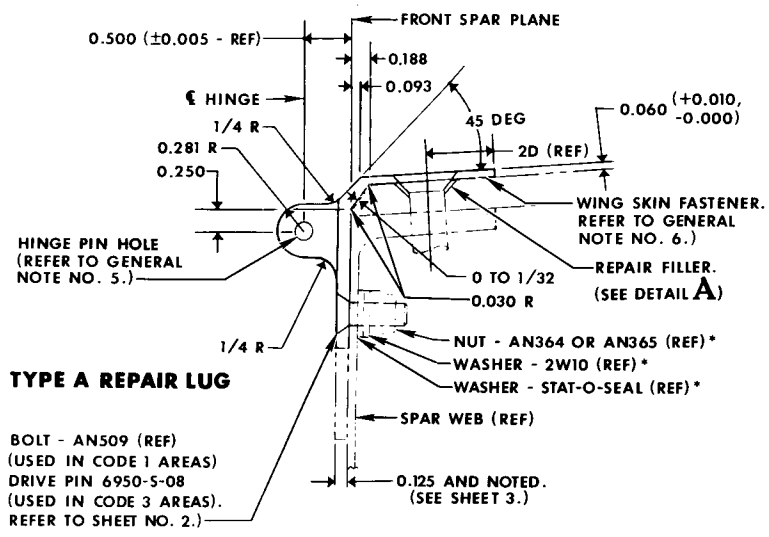
MATERIAL GAGE (INCHES)		MAXIMUM D (INCHES)	RIVETING
WEB	REPAIR DOUBLER AND STIFFENER		
0.032	0.051	3.5	AN470AD5 5/32 AF463
0.040	0.064	4.5	AN470DD6 6951D-06

Figure 2-28. Wing Corrugated Spar Web (Sheet 2 of 2)

PERMANENT CATEGORY B

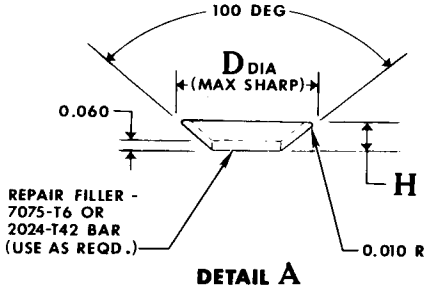
WING





TYPE A REPAIR LUG

*CODE 1 AREAS ONLY



DETAIL A

FOR USE WITH	D	H
1/4 BOLT	0.504 0.501	0.106 (+0.010, -0.000)
5/16 BOLT	0.633 0.628	0.133 (+0.010, -0.000)
3/8 BOLT	0.760 0.755	0.160 (+0.010, -0.000)

GENERAL NOTES

1. Three types of repair lugs are shown.

   a. Type A—to be used in accessible areas (where washers and nuts can be installed).

   b. Type B—to be used in blind areas (where blind fasteners must be used).

   c. Type C—special application repairs for three restricted areas. (Refer to sheet No. 5.)

2. Areas where each repair may be used are shown in sheets No. 6 and 7. Also shown in sheets No. 6 and 7 are (1) the exact (and only) areas on the front spar web where fasteners may be installed, and (2) the fastener type applicable to each of these areas. Minor modification of lines etc. mounted on the spar may be required.

3. The following applies to Type A and B repair lugs.

   (a) Both types have typical mold line flange configurations which depend on the location of existing skin fasteners with respect to the hinge lugs. (Refer to sheet No. 2.)

   (b) Spar web flange configurations for Type A and B repair lugs are shown in sheets 3 and 4. The views shown represent the *minimum* and *maximum* configurations and a few of the intermediate possibilities. The flange may be varied within the limits shown to match permissible fastener locations on the spar web. Drive pin blind fasteners, 6950-S, may be used in Type A and B repairs when the spar web area is coded for their use. (Refer to sheets 6 and 7.)

4. All fasteners which project into the spar web are 1/4-inch diameter.

5. The hinge pin hole sizes as required are 0.191, 0.3125, and 0.3160 inch. Tolerances for each size are +0.005, −0.000 inch. The hinge pin holes in the repair lugs must be aligned with the existing lug holes. Line drilling or reaming on assembly is recommended.

6. The replacement wing skin fastener must be selected for proper thread engagement. Install repair filler (detail A) flush to 0.010 inch below surface of wing skin.

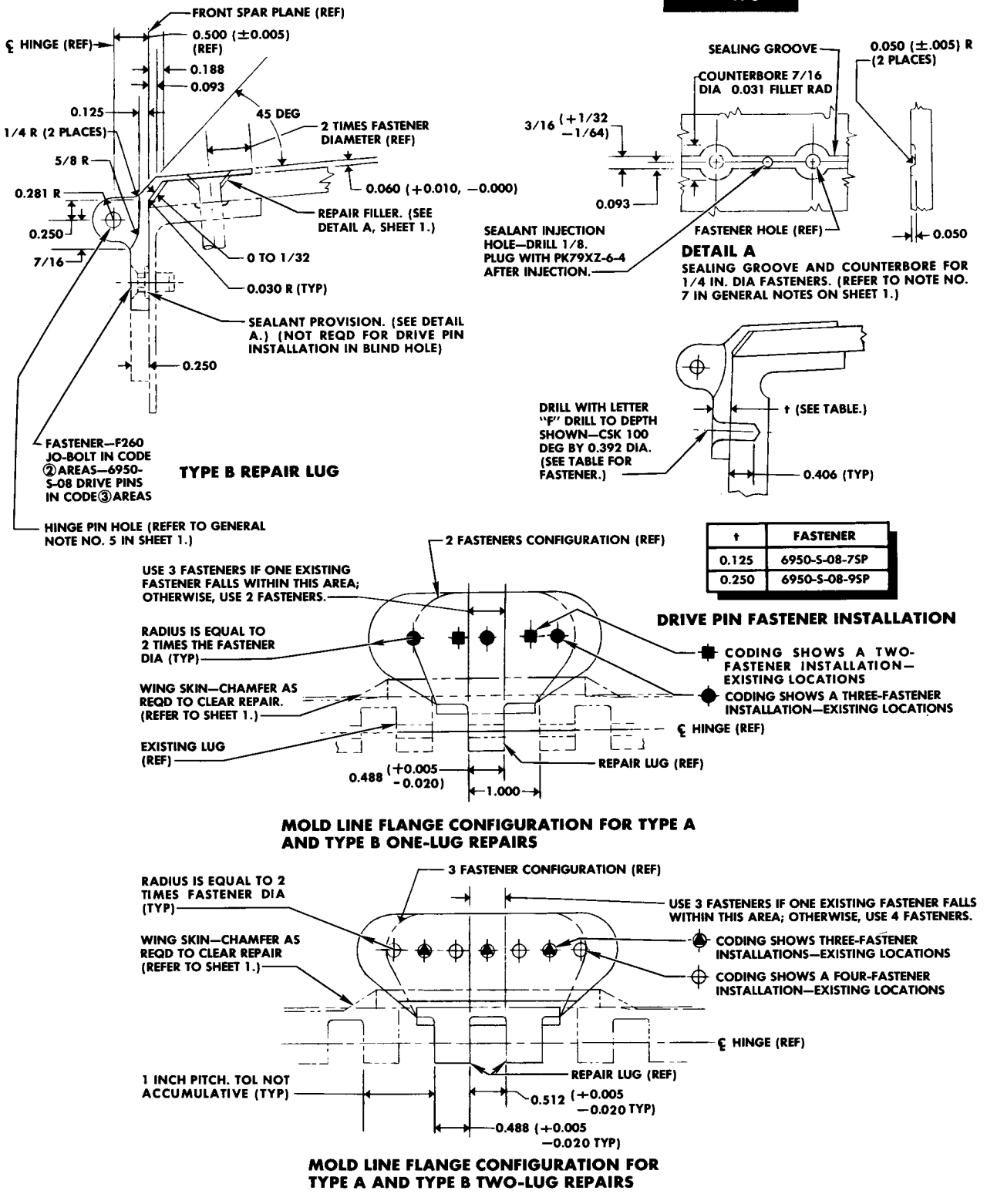
7. For sealing instructions, refer to "Wing Fuel Tank Sealing."

F-100C-3-10-55

Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 1 of 7)

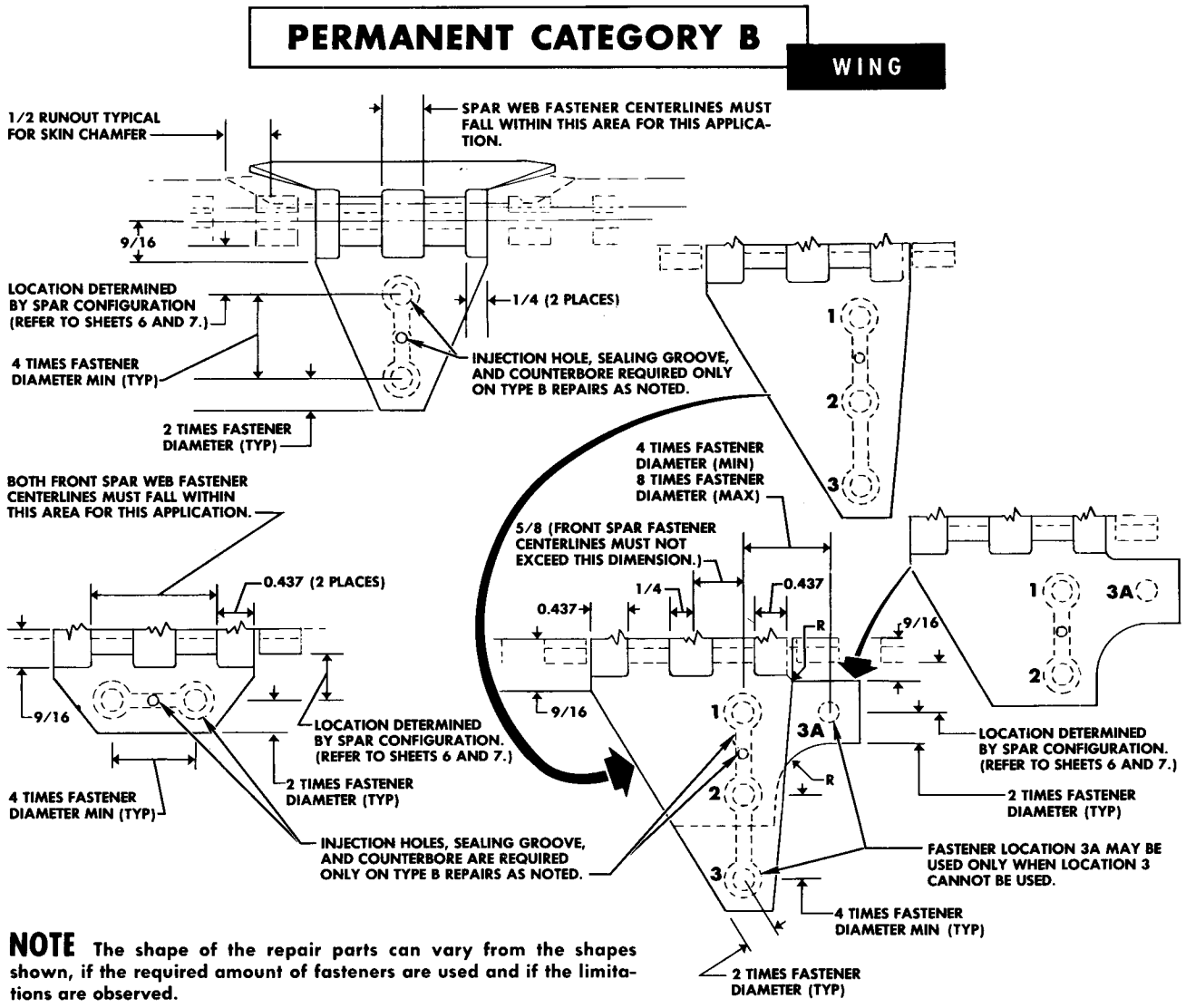
PERMANENT CATEGORY B

WING



†	FASTENER
0.125	6950-S-08-7SP
0.250	6950-S-08-9SP

Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 2 of 7)



NOTE The shape of the repair parts can vary from the shapes shown, if the required amount of fasteners are used and if the limitations are observed.

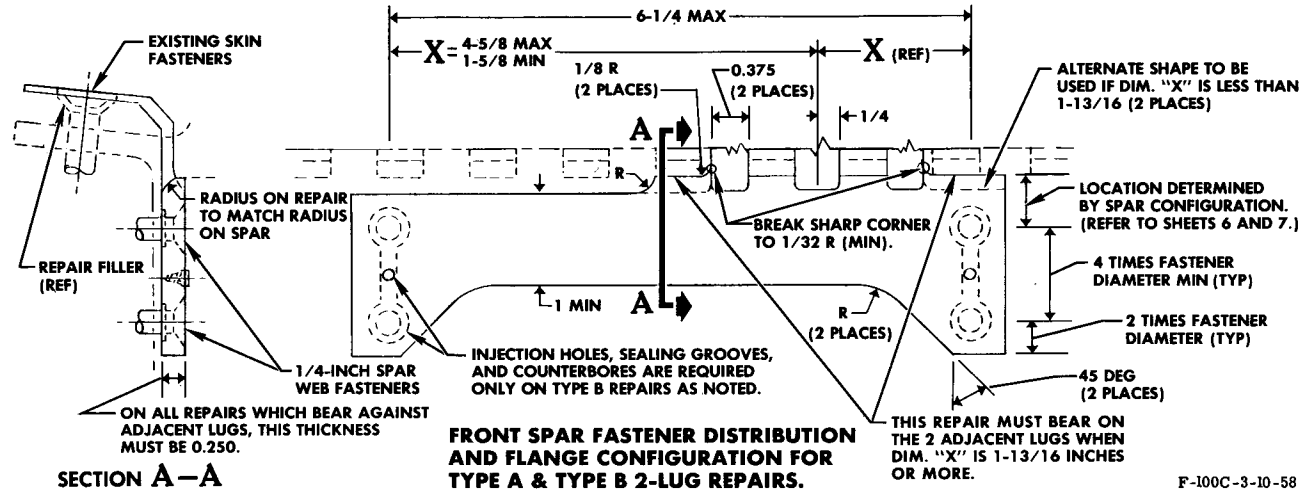


Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 3 of 7)

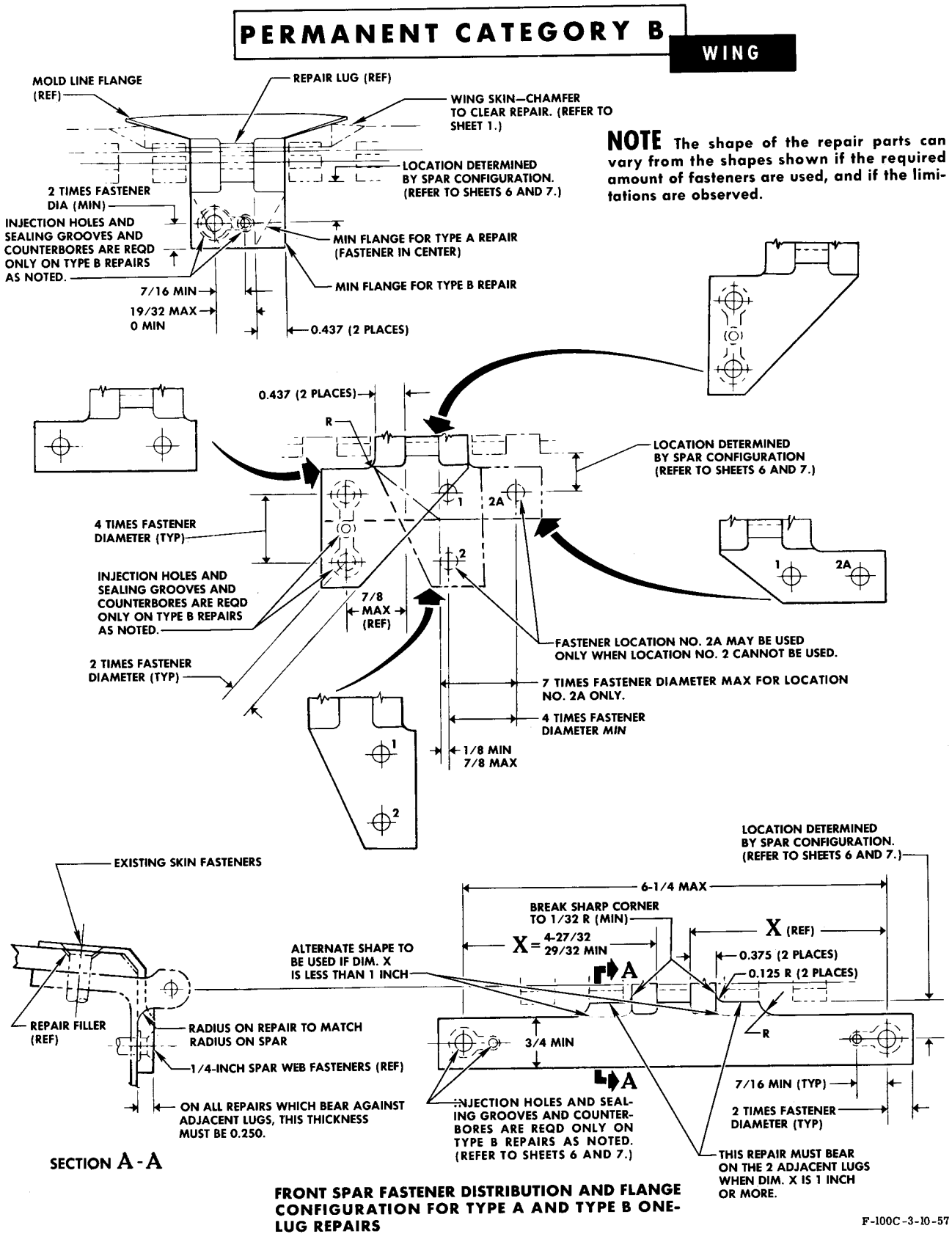


Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 4 of 7)

PERMANENT CATEGORY B

WING

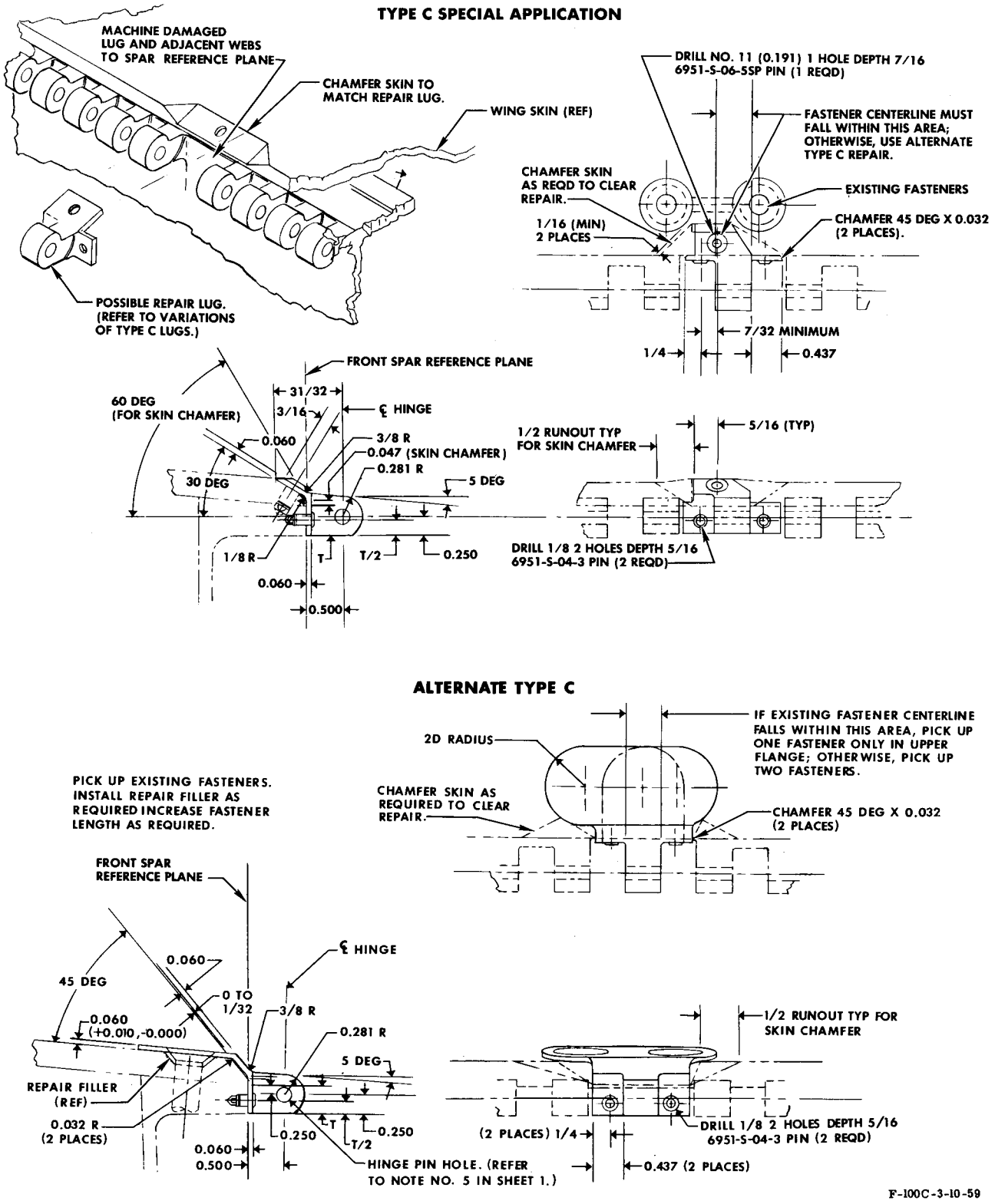


Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 5 of 7)

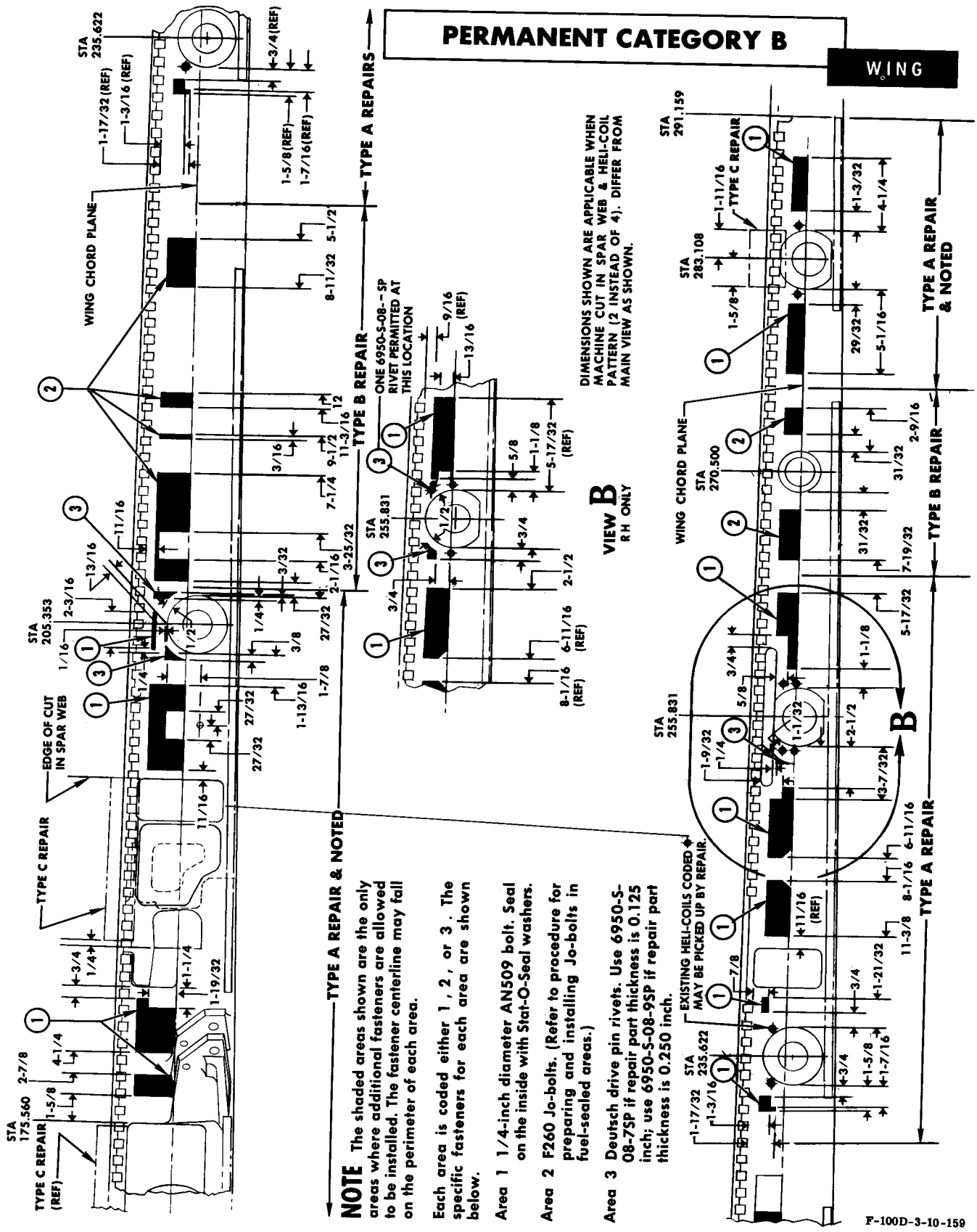


Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 6 of 7)

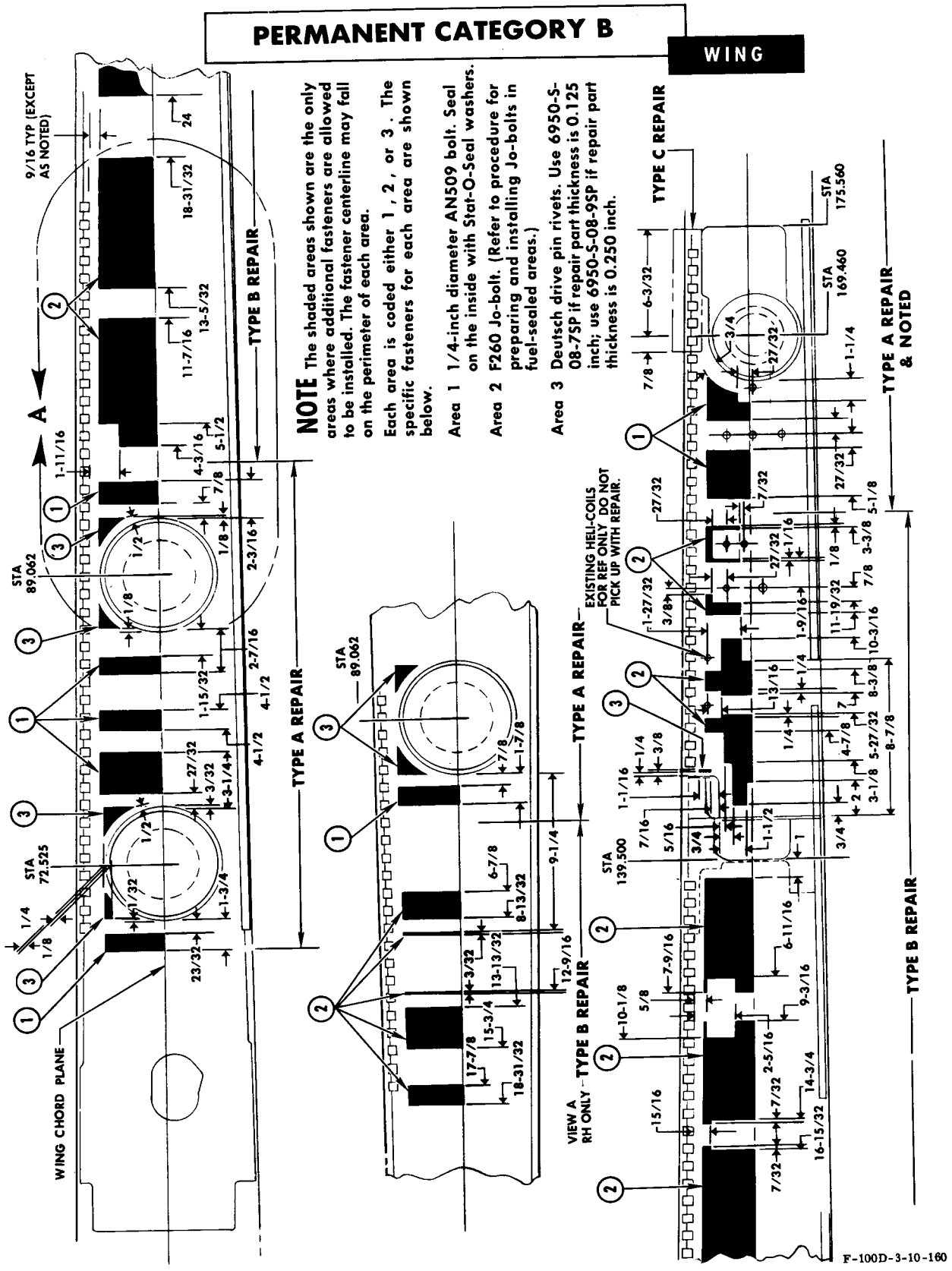
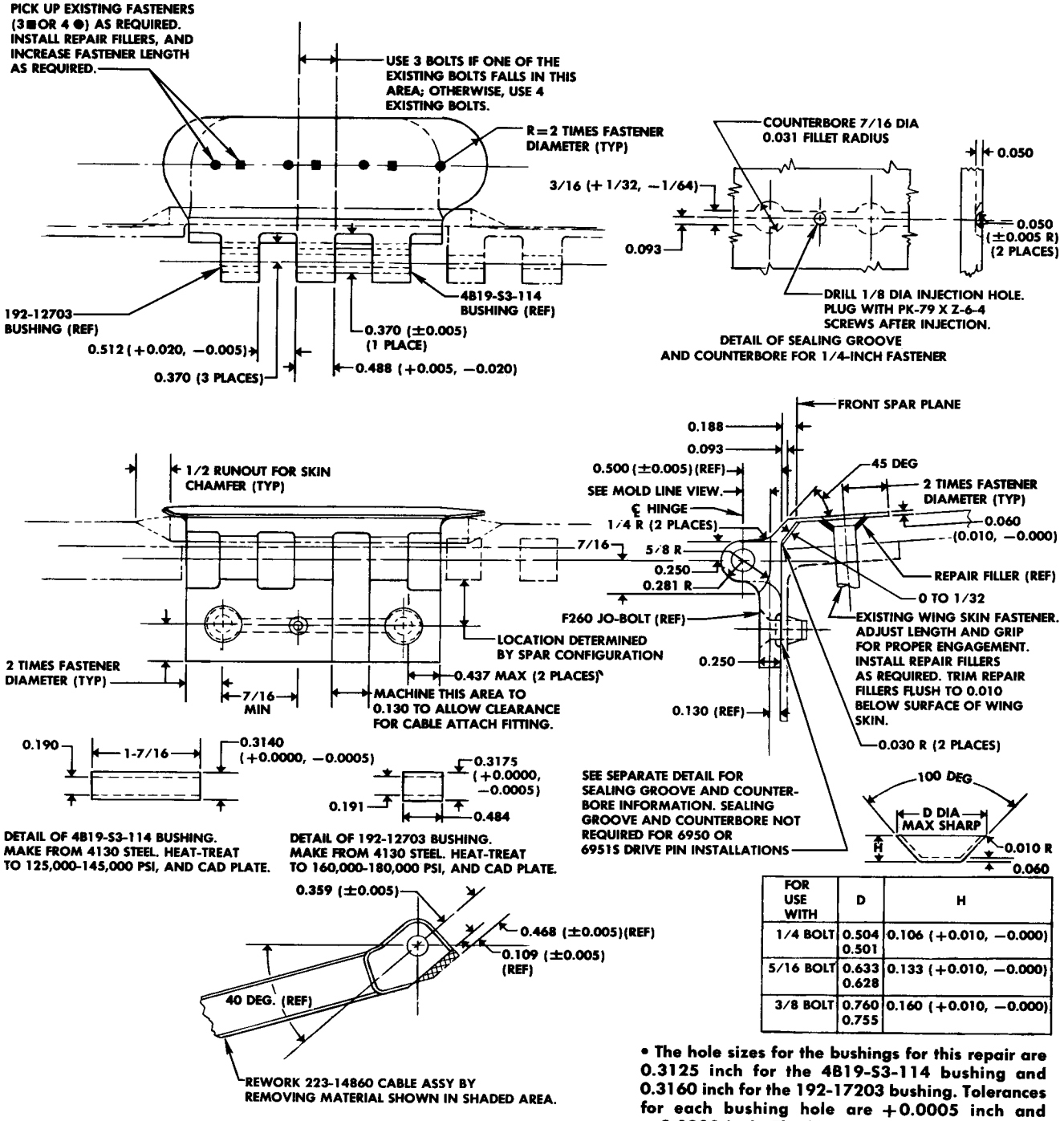


Figure 2-28A. Hinge Repair—Front Spar—Upper (Sheet 7 of 7)

PERMANENT CATEGORY B

WING



FOR USE WITH	D	H
1/4 BOLT	0.504 0.501	0.106 (+0.010, −0.000)
5/16 BOLT	0.633 0.628	0.133 (+0.010, −0.000)
3/8 BOLT	0.760 0.755	0.160 (+0.010, −0.000)

NOTE This repair was specifically designed for use at the drop cable attach point at station 116.718.

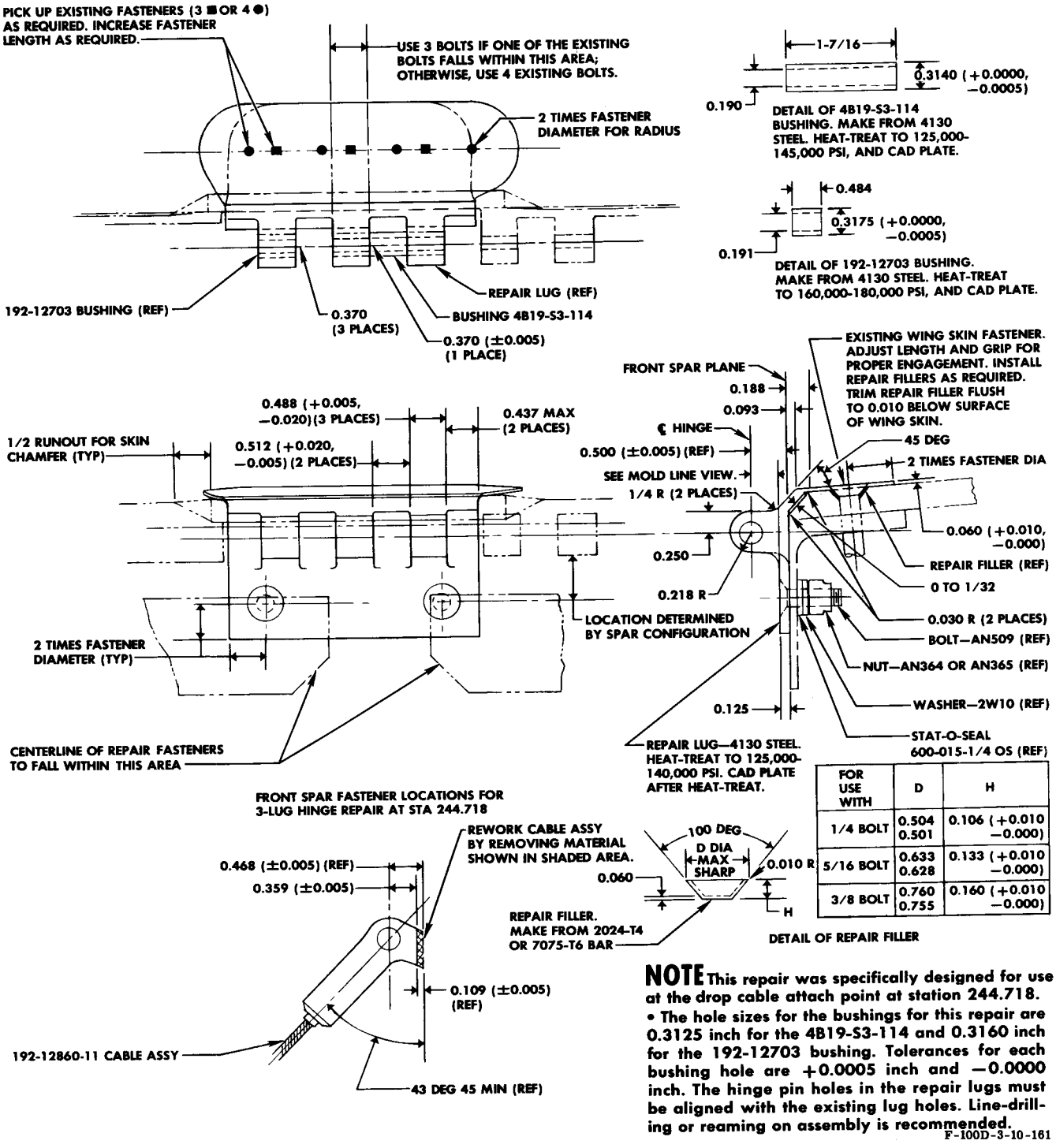
• The hole sizes for the bushings for this repair are 0.3125 inch for the 4B19-S3-114 bushing and 0.3160 inch for the 192-17203 bushing. Tolerances for each bushing hole are +0.0005 inch and −0.0000 inch. The hinge pin holes in the repair lugs must be aligned with the existing lug holes. Line-drilling or reaming on assembly is recommended.

F-100D-3-10-162

Figure 2-28B. Specific Three-lug Hinge Repair—Front Spar—Upper—Station 116.718

PERMANENT CATEGORY B

WING



FOR USE WITH	D	H
1/4 BOLT	0.504 0.501	0.106 (+0.010 −0.000)
5/16 BOLT	0.633 0.628	0.133 (+0.010 −0.000)
3/8 BOLT	0.760 0.755	0.160 (+0.010 −0.000)

NOTE This repair was specifically designed for use at the drop cable attach point at station 244.718.

- The hole sizes for the bushings for this repair are 0.3125 inch for the 4B19-S3-114 and 0.3160 inch for the 192-12703 bushing. Tolerances for each bushing hole are +0.0005 inch and −0.0000 inch. The hinge pin holes in the repair lugs must be aligned with the existing lug holes. Line-drilling or reaming on assembly is recommended.

Figure 2-28C. Specific Three-lug Hinge Repair—Front Spar—Upper—Station 244.718

PERMANENT—CATEGORY B

WING

1. Clean up damaged area.
2. Fabricate repair parts.
3. Apply PR1221 sealant to mating surfaces on all parts. Do not allow PR1221 to get in the sealant groove.
4. Install spar repair parts on spar.
5. Install skin repair parts on skin.
6. After curing of PR1221 sealant is complete, cover all exposed surfaces with fuel-resistant coating.
7. Inject PR703 sealant into sealant groove. Refer to "wing fuel tank sealing."

NOTE For additional information, refer to text.

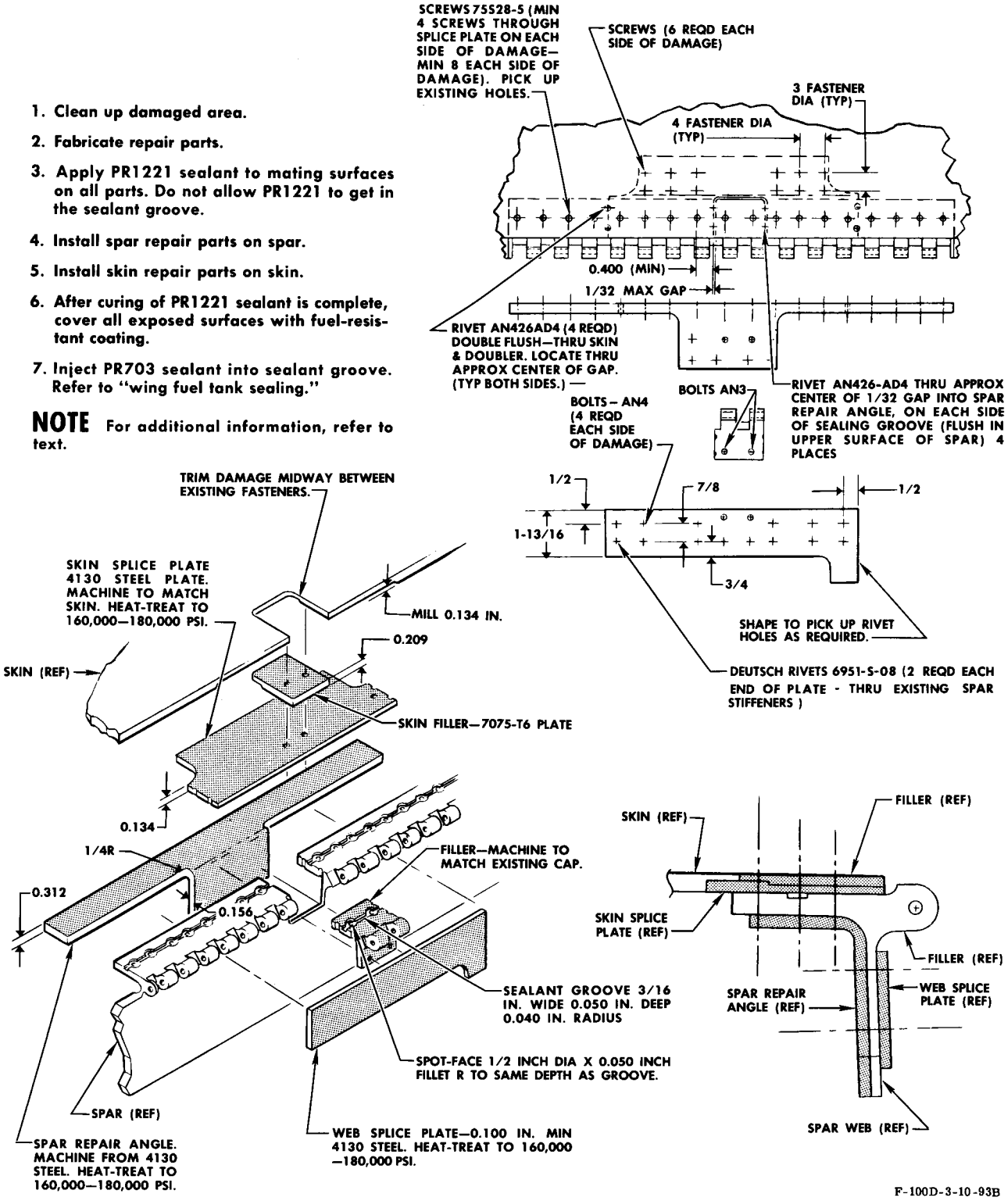
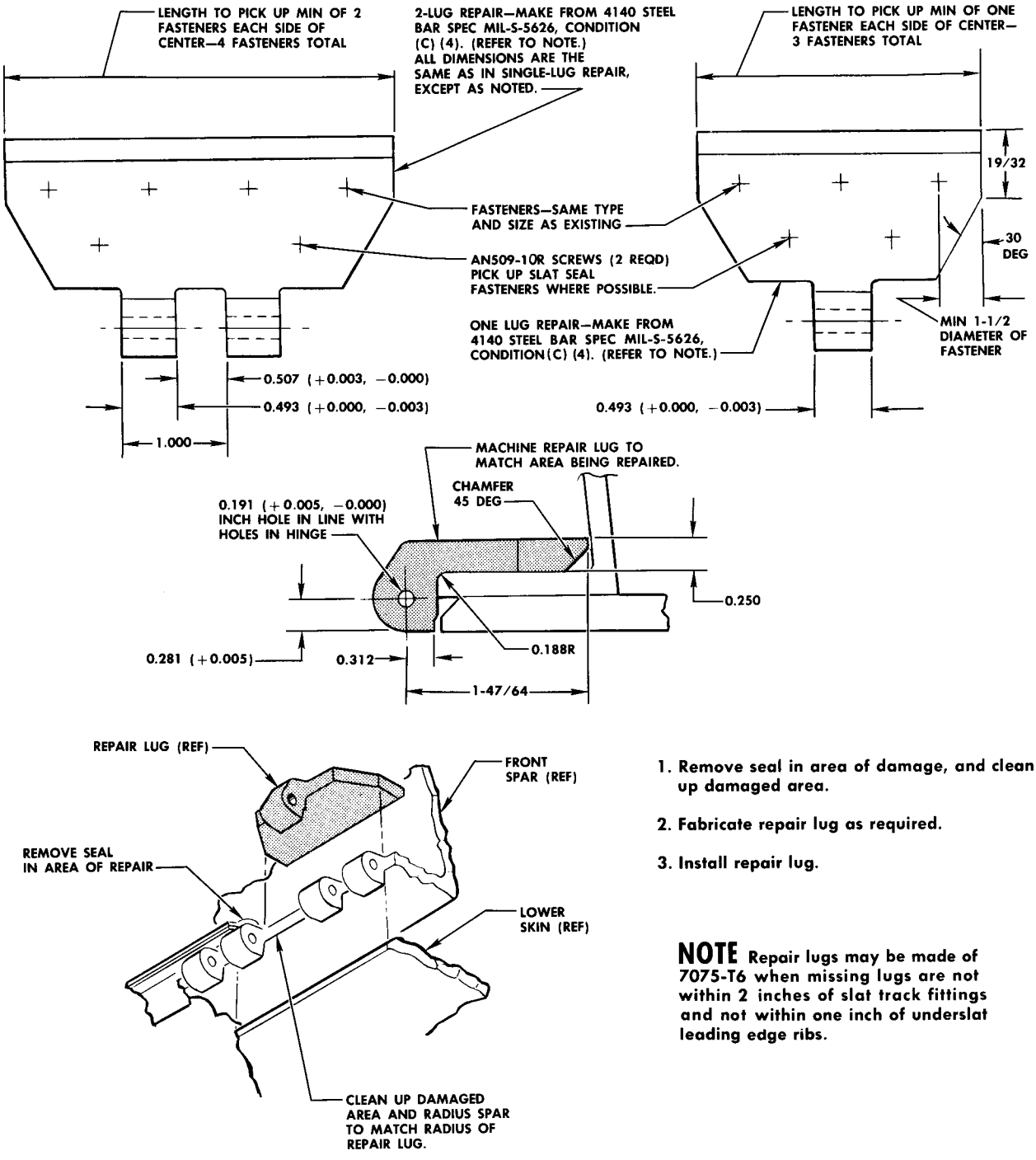


Figure 2-29. Wing Front Spar Upper Cap and Skin

PERMANENT—CATEGORY A

WING



1. Remove seal in area of damage, and clean up damaged area.
2. Fabricate repair lug as required.
3. Install repair lug.

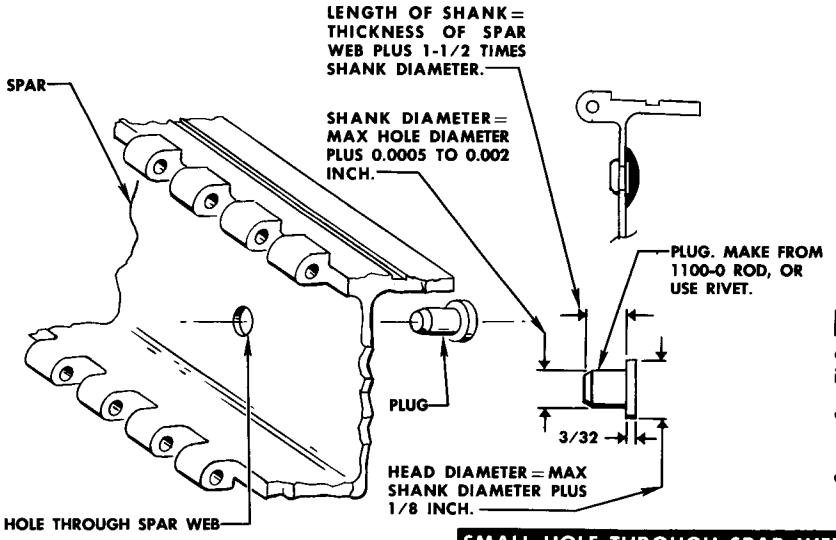
NOTE Repair lugs may be made of 7075-T6 when missing lugs are not within 2 inches of slat track fittings and not within one inch of underslat leading edge ribs.

F-100D-3-10-92

Figure 2-30. Lower Leading Edge Hinge

PERMANENT CATEGORY B

WING



1 Clean up hole to smallest diameter possible. (The maximum diameter of the cleaned hole must not exceed 1/2 inch.)

2 Fabricate plug as shown.

3 Install plug with manufactured head on fuel side of spar.

4 Apply coat of PR-1221 sealant over head of plug. After curing, overcoat PR-1221 sealant with fuel-resistant coating.

NOTE The minimum distance between the edge of the cleaned hole and any near fastener is equal to the diameter of the hole.

- No more than one repair of this type is permitted in any one spar panel.
- A spar panel is a section of the spar web between the stiffeners.

SMALL HOLE THROUGH SPAR WEB

1 Clean up hole to proper size.

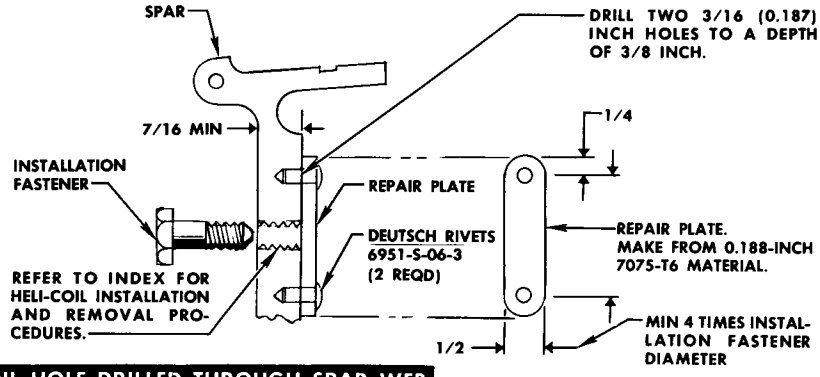
2 Fabricate repair plate as shown.

3 Install repair plate with PR-1221 sealant between repair plate and spar. Add fillet of PR-1221 sealant around edges of repair plate after installation.

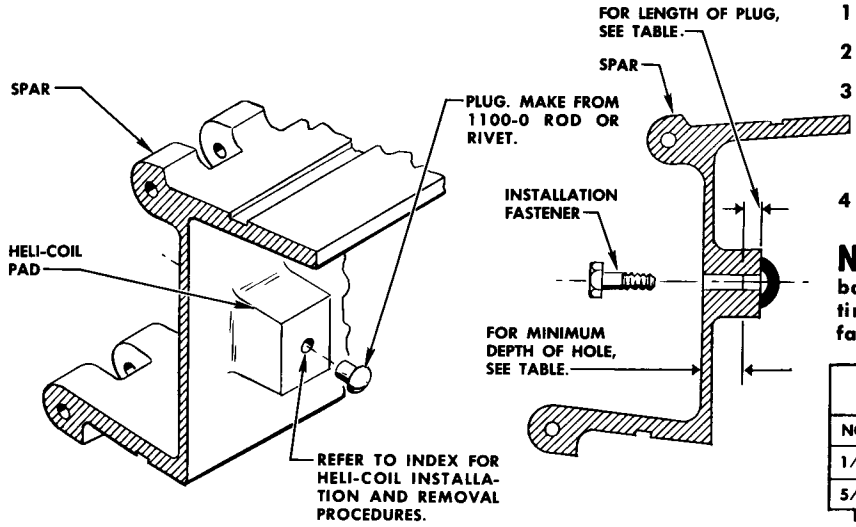
4 After curing, overcoat exposed PR-1221 sealant with fuel-resistant coating.

NOTE The spar web must be thick enough for the installation of buried Deutsch rivets.

- The minimum distance between the Deutsch rivets in the repair plate and any near spar web fastener should be two times the diameter of the installation fastener.



HELI-COIL HOLE DRILLED THROUGH SPAR WEB



1 Clean up hole to proper size.

2 Fabricate plug to fit as shown.

3 Coat plug with PR-1221 sealant, and install from fuel side. After installation, cover head of plug and area around head with PR-1221 sealant.

4 After curing, overcoat PR-1221 sealant with fuel-resistant coating.

NOTE The minimum depth of the hole is based on the Heli-Coil insert length of 1-1/2 times the major diameter of the installation fastener.

HELI-COIL INSERT	MINIMUM DEPTH	LENGTH OF PLUG (+0, −1/32 IN.)
NO. 10 SCREW	1/2 IN.	5/32 IN.
1/4 IN. SCREW	5/8 IN.	3/16 IN.
5/16 IN. SCREW	7/8 IN.	5/16 IN.

HELI-COIL HOLE DRILLED THROUGH SPAR PAD

F-100C-3-10-54

Figure 2-31. Hole Through Spar Web

PERMANENT CATEGORY B

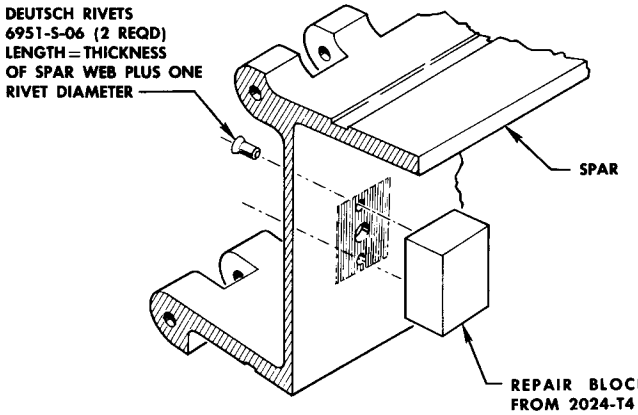
WING

1 Remove damaged Heli-Coil pad flush with spar web.

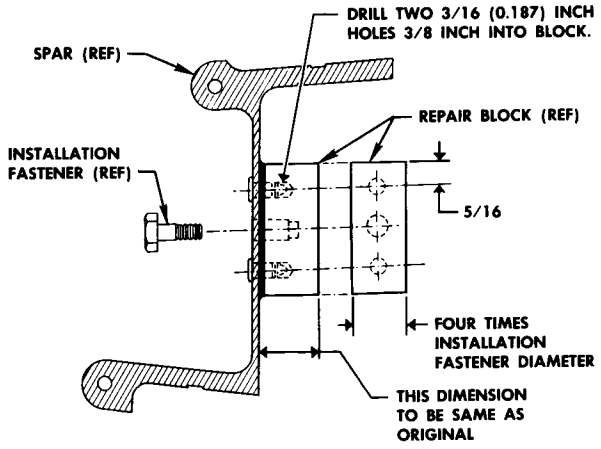
2 Fabricate repair block as shown.

3 Apply PR-1221 sealant between repair block and spar web, and rivet block in place.

4 Apply a fillet of PR-1221 sealant around edges of block. After curing, all exposed PR-1221 sealant must be overcoated with fuel-resistant coating. (Refer to index.)



DAMAGED HELI-COIL PAD



NOTE This repair is used when the Heli-Coil hole cannot be successfully plugged.

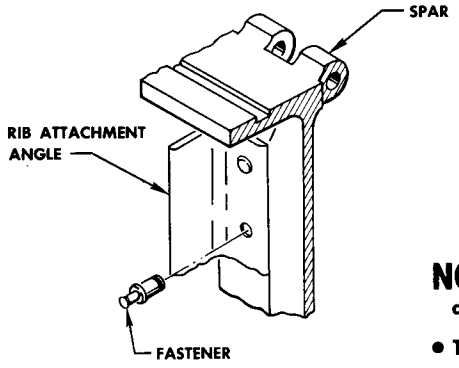
- The minimum distance between the Deutsch rivets and any near spar web fastener should be two times the installation fastener diameter.
- Only one repair of this type is allowed in any one panel.

---

1 Ream hole to correct size for fastener one size larger than original fastener.

2 Install oversize fastener.

NOTE When a blind fastener of the size and type required is not available, an AN. bolt or equivalent may be substituted.

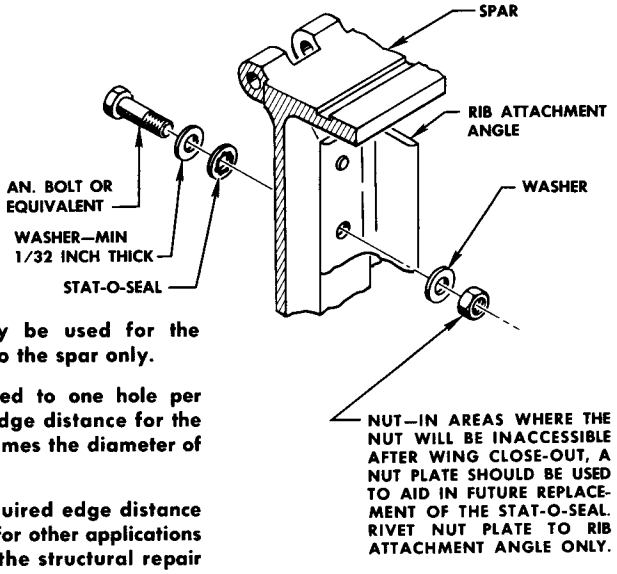


ELONGATED OR ENLARGED HOLE

1 Ream hole to correct size for bolt one size larger than original fastener.

2 Chamfer edge of hole 45 degrees X 0.010 (±0.005) inch.

3 Install bolt as shown.



NOTE This repair may be used for the attachment of the ribs to the spar only.

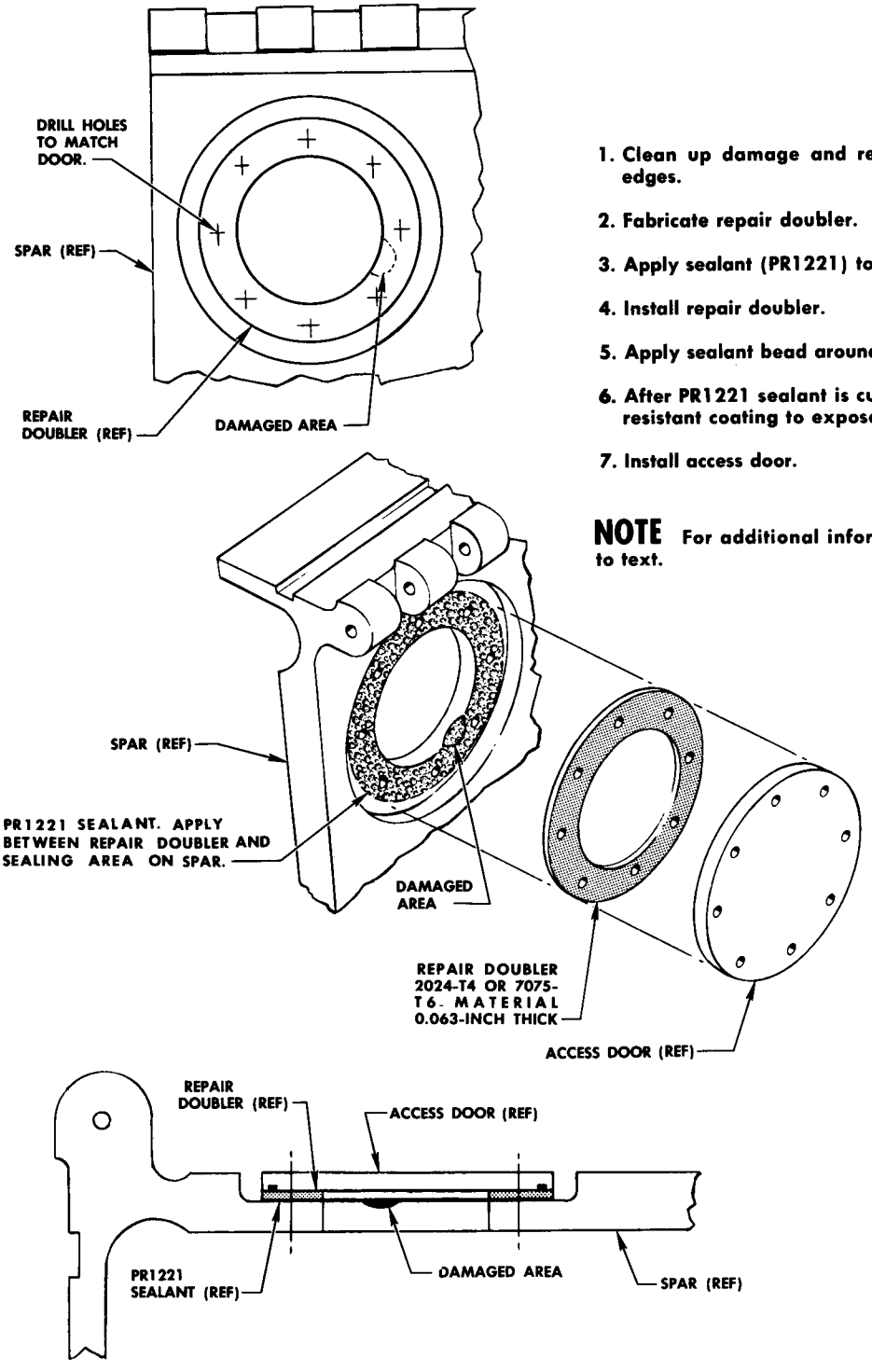
- This repair is not limited to one hole per angle. The minimum edge distance for the attach angle is 1-1/2 times the diameter of the fastener used.
- In areas where the required edge distance cannot be obtained, or for other applications of this type of repair, the structural repair officer must be consulted.

F-100C-3-10-53

Figure 2-32. Heli-Coil Hole Through Spar Web

PERMANENT—CATEGORY A

WING



1. Clean up damage and remove all sharp edges.
2. Fabricate repair doubler.
3. Apply sealant (PR1221) to surfaces.
4. Install repair doubler.
5. Apply sealant bead around doubler edge.
6. After PR1221 sealant is cured, apply fuel-resistant coating to exposed surface.
7. Install access door.

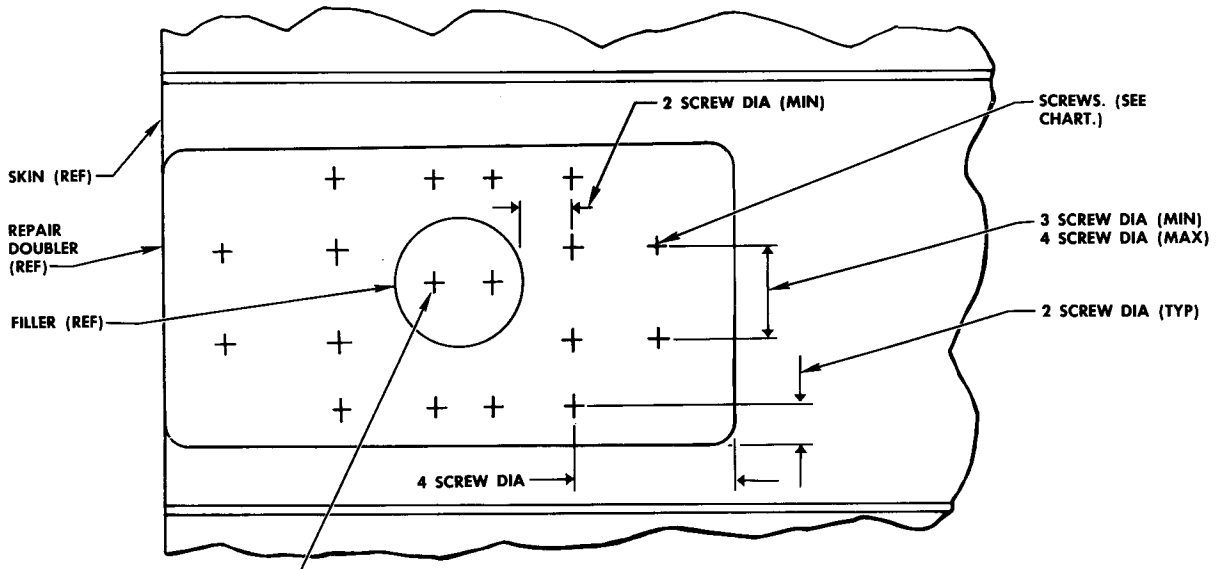
NOTE For additional information, refer to text.

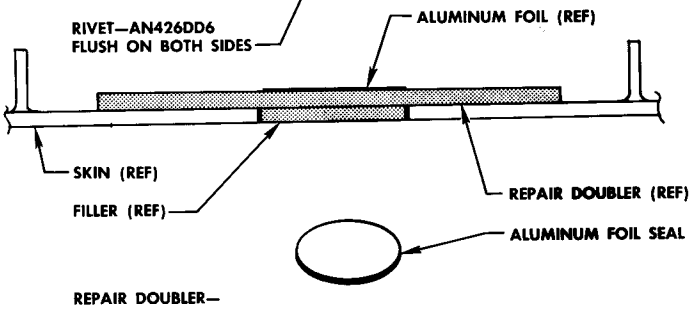
F-100D-3-10-96A

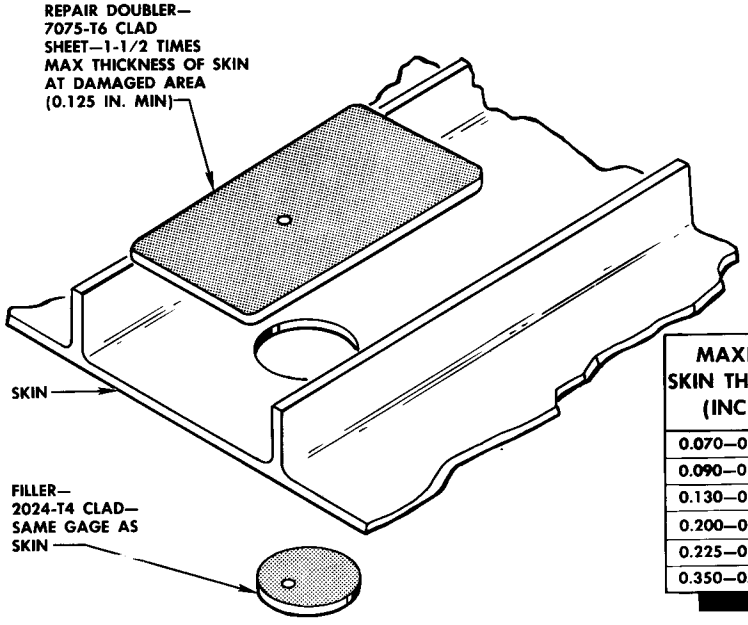
Figure 2-33. Sealing Surface at Access Door on Spar

PERMANENT CATEGORY B

WING







1 Trim out damaged area. (Maximum dimension = 1-1/2 inches.)

2 Fabricate repair doubler, filler, and foil seal.

3 Apply a thin coat of PR-1221 sealing compound to all mating surfaces, and assemble repair as shown. (Use Stat-O-Seal washers in accessible areas or ESNA dome nuts overcoated with PR-1221 sealing compound.)

4 Apply a bead of PR-1221 sealing compound around edges of repair doubler.

5 Place aluminum foil seal in place over filler rivets. Bond with PR-1221 sealing compound.

6 Apply overcoating of fuel-resistant coating on all exposed PR-1221 sealing compound after curing.

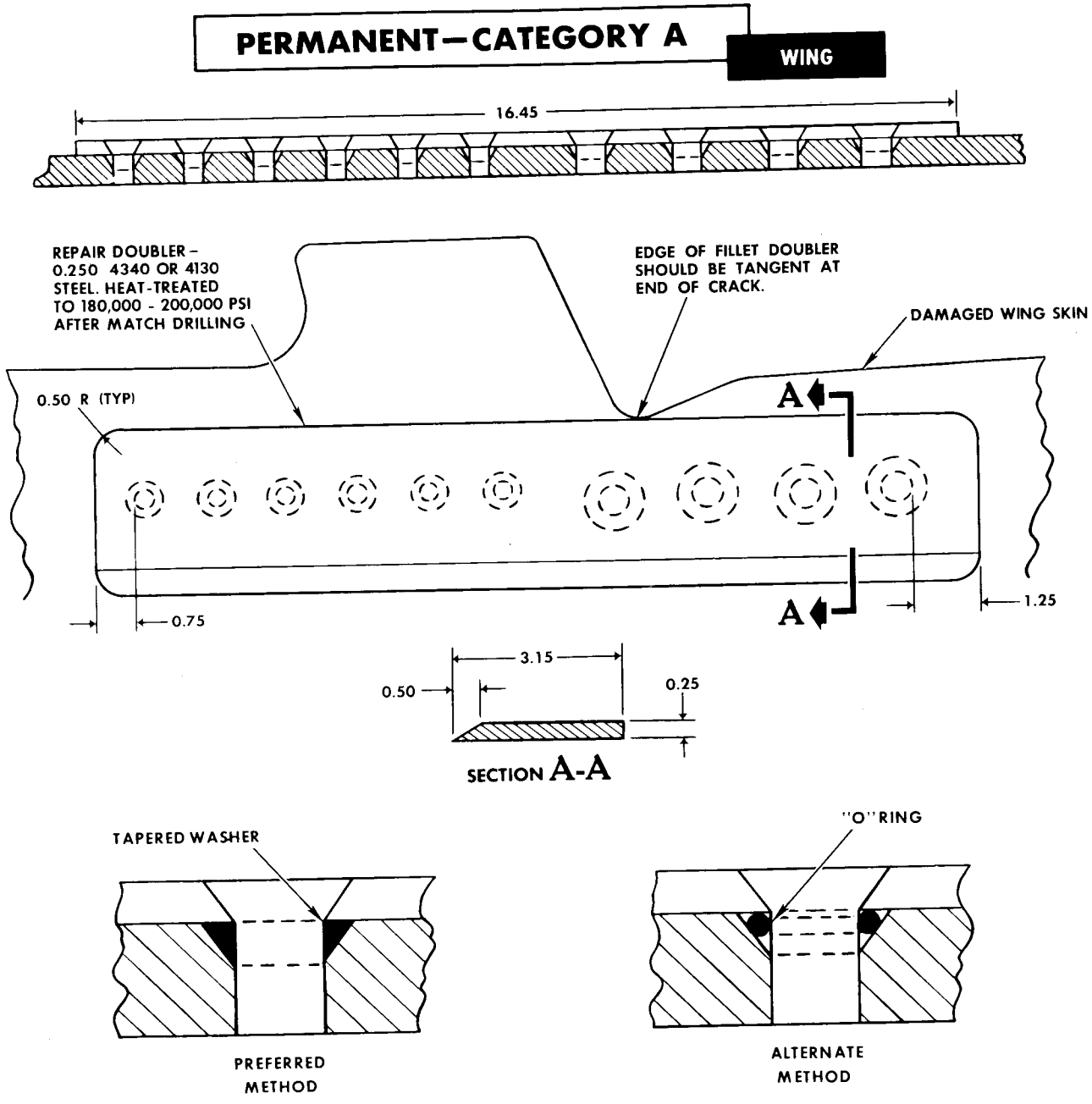
* HT3M, HT4M, and HT5M screws may be substituted for NAS333, NAS334, and NAS335 screws, respectively.

† If 7S28 screws are not available, the next size NAS or HT-M screw may be substituted.

MAXIMUM SKIN THICKNESS (INCHES)	SCREW SIZE	NUMBER OF FULL CHORDWISE ROWS	DOUBLER WIDTH PLUS DAMAGE
0.070—0.090	NAS333*	1	1-1/2 INCHES
0.090—0.130	7S28-3†	1	1-1/2 INCHES
0.130—0.200	NAS334*	1	2 INCHES
0.200—0.225	7S28-4†	1	2 INCHES
0.225—0.350	NAS335*	1	2-1/2 INCHES
0.350—0.500	7S28-5†	2	2-1/2 INCHES

F-100D-3-10-97A

Figure 2-34. Wing Skins—0.070 to 0.500-inch Thick



For Doubler Manufacture

1. Break all edges 0.010 inch.
2. Machine all surfaces.
3. Match drill holes in doubler (shown) before heat-treating and cadmium-plating.

For Assembly

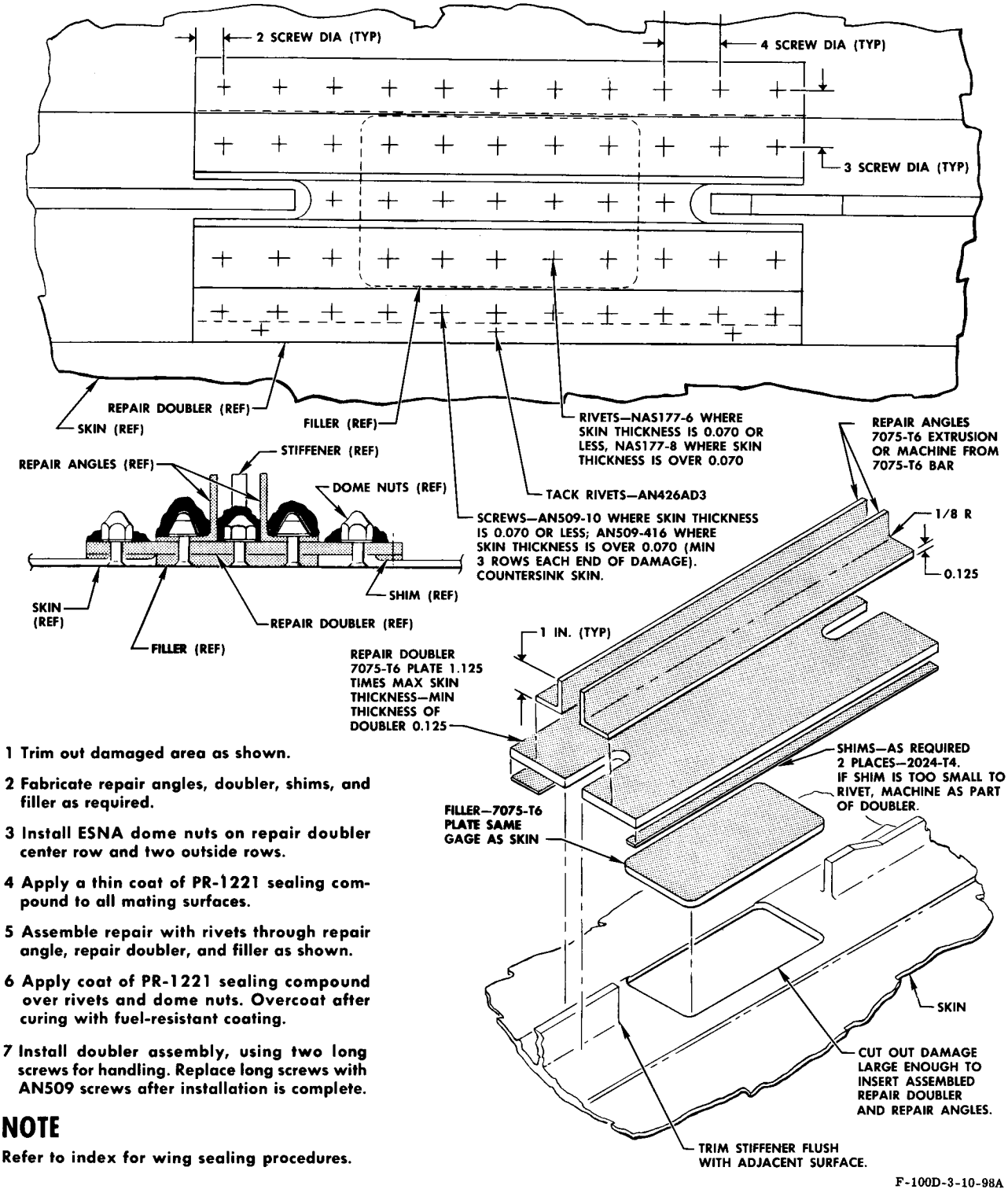
1. Remove paint from area of skin under doubler.
2. Place countersink washers NAS33C and NAS339 (preferred) or "O" rings MS29513-12 and MS28775-113 in holes in skin.
3. Fasten doubler to skin with adhesive (Specification MIL-S-8802). Reseal wing skin by rolling bolt threads in standard wing sealant before installation, and apply sealant to wing nuts during installation.
4. Use bolts NAS339-20 and NAS336-20 (preferred) or NAS586-18 and NAS589-20 or longer.

100D-3-10-195

Figure 2-34A. External Doubler—Lower Wing Skin at Station 184.240

PERMANENT—CATEGORY A

WING



1 Trim out damaged area as shown.

2 Fabricate repair angles, doubler, shims, and filler as required.

3 Install ESNA dome nuts on repair doubler center row and two outside rows.

4 Apply a thin coat of PR-1221 sealing compound to all mating surfaces.

5 Assemble repair with rivets through repair angle, repair doubler, and filler as shown.

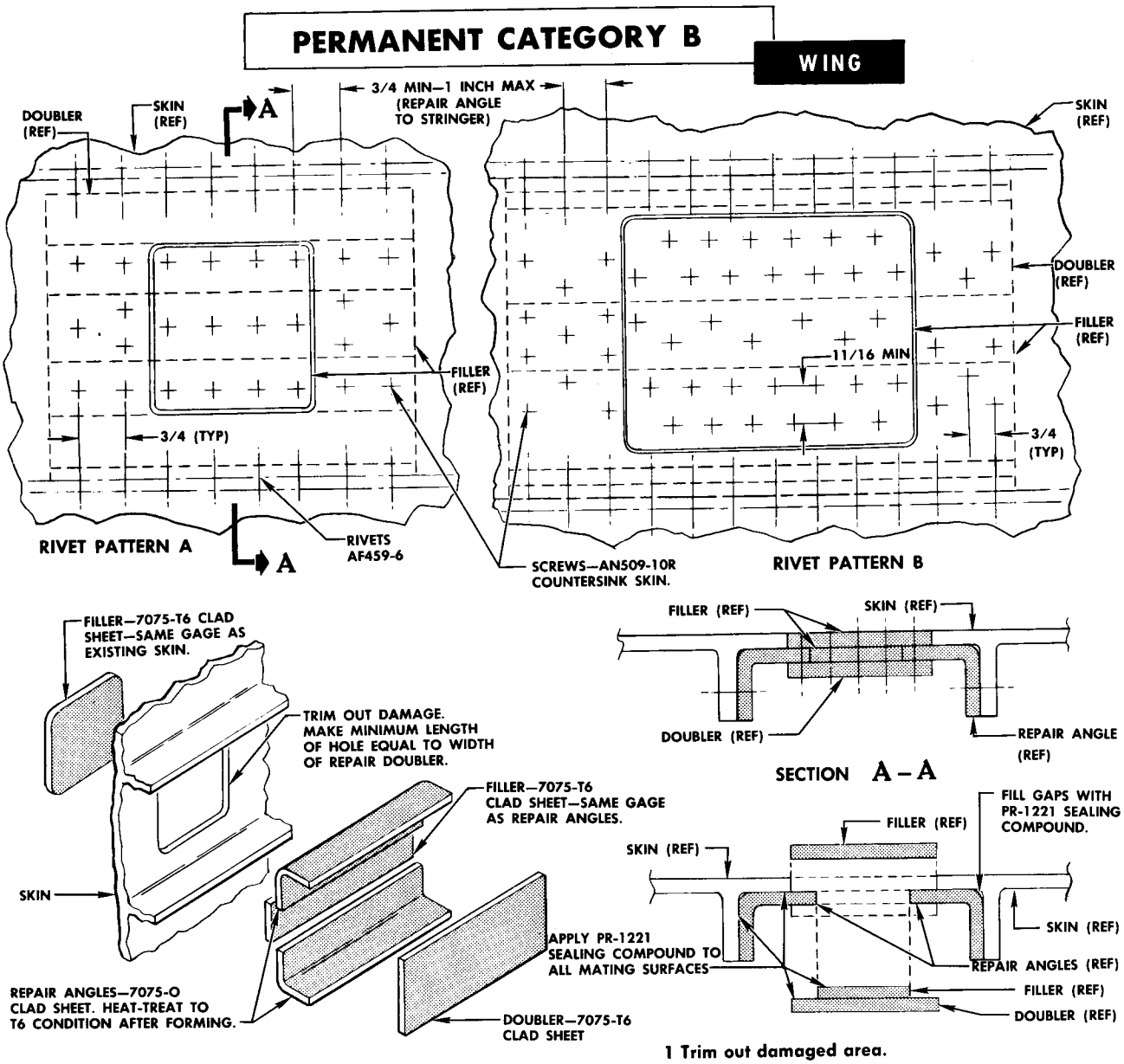
6 Apply coat of PR-1221 sealing compound over rivets and dome nuts. Overcoat after curing with fuel-resistant coating.

7 Install doubler assembly, using two long screws for handling. Replace long screws with AN509 screws after installation is complete.

NOTE

Refer to index for wing sealing procedures.

Figure 2-35. Wing Skin and Stringer



EXISTING SKIN GAGE	REPAIR DOUBLER GAGE	REPAIR ANGLE GAGE	REPAIR ANGLE BEND RADII	RIVET PATTERN
IN.	IN.	IN.	IN.	———
0.070	0.081	0.091	1/4	A
0.081	0.091	0.102	5/16	B
0.091	0.102	0.125	11/32	B
0.102	0.125	0.125	11/32	B
0.125	0.156	0.156	15/32	B
0.156	0.188	0.188	9/16	B

1 Trim out damaged area.

2 Fabricate repair angles, fillers, and doubler.

3 Install ESNA A2500-02 dome nuts on doubler. Apply PR-1221 sealing compound around nutshells and rivets.

4 Apply PR-1221 sealing compound to all mating surfaces and install repair angles. Fill all gaps with PR-1221 sealing compound.

5 Apply PR-1221 sealing compound to mating surfaces and install doubler and fillers as shown.

Length of repair for rivet pattern A = about 3 inches plus damage.

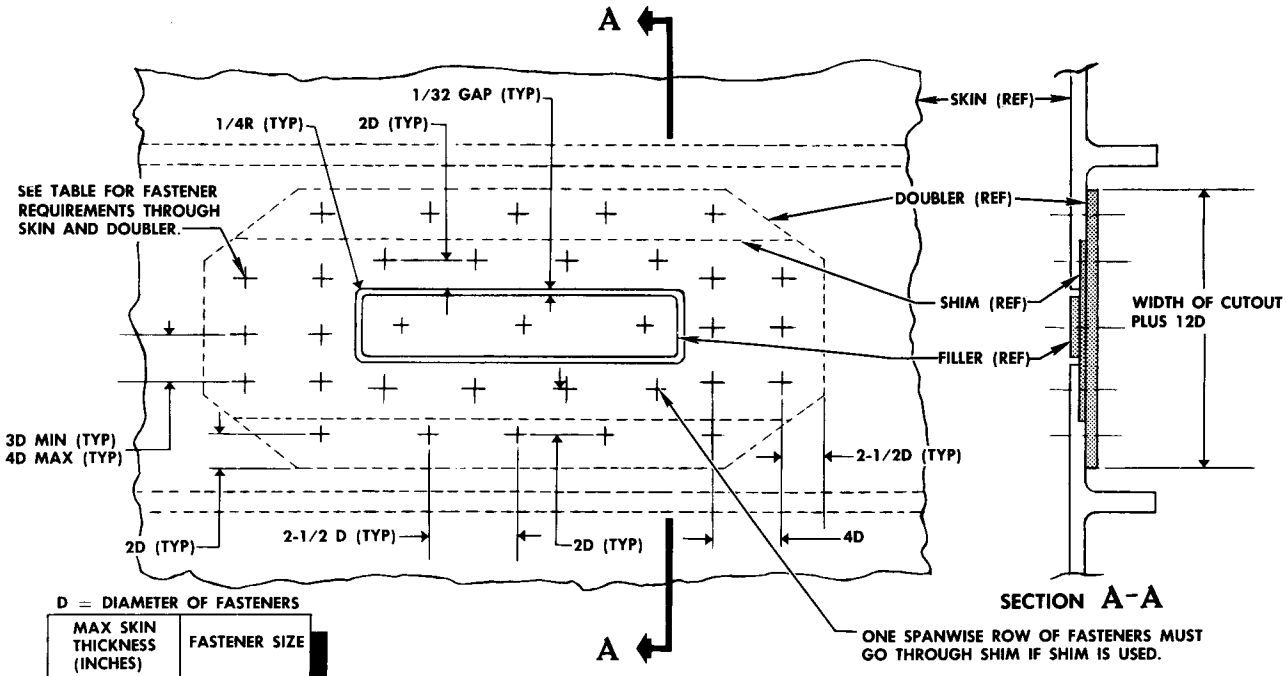
- Length of repair for rivet pattern B = about 4-1/2 inches plus damage.
- Minimum rivet edge distance = 3/8 inch.

F-100D-3-10-99A

Figure 2-36. Wing Skins—Maximum Thickness 0.156 Inch

PERMANENT—CATEGORY A

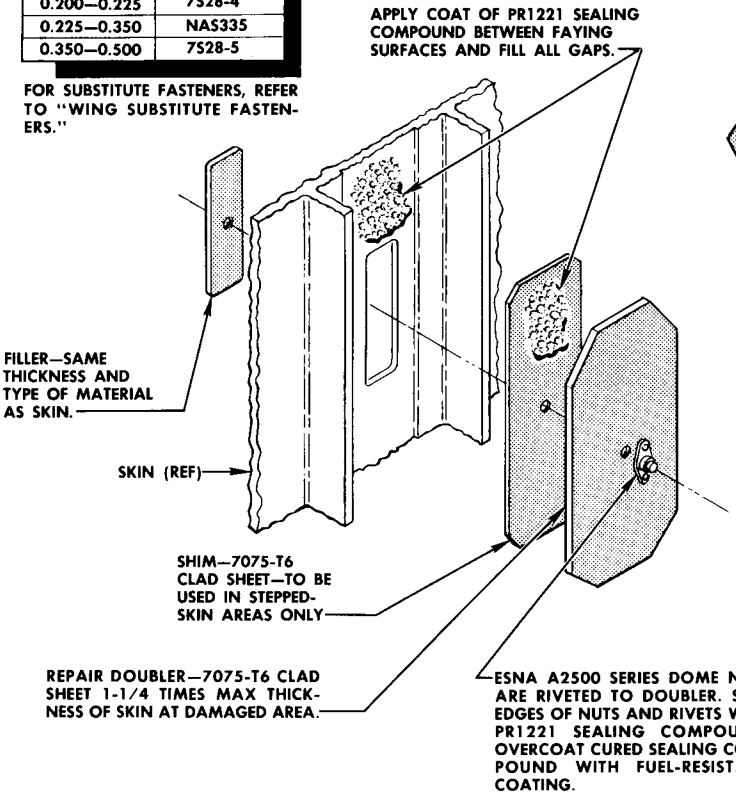
WING

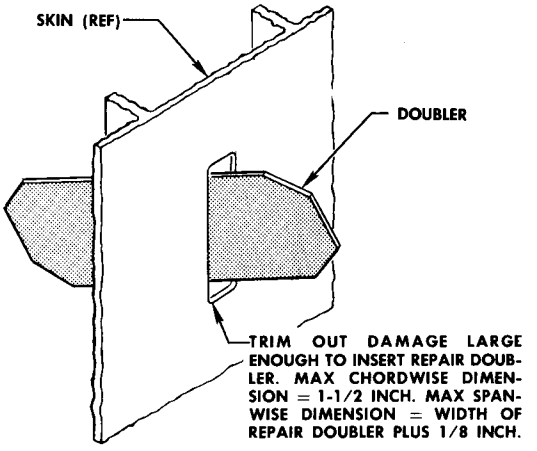


D = DIAMETER OF FASTENERS

MAX SKIN THICKNESS (INCHES)	FASTENER SIZE
0.070—0.090	NAS333
0.090—0.130	7S28-3
0.130—0.200	NAS334
0.200—0.225	7S28-4
0.225—0.350	NAS335
0.350—0.500	7S28-5

FOR SUBSTITUTE FASTENERS, REFER TO "WING SUBSTITUTE FASTENERS."





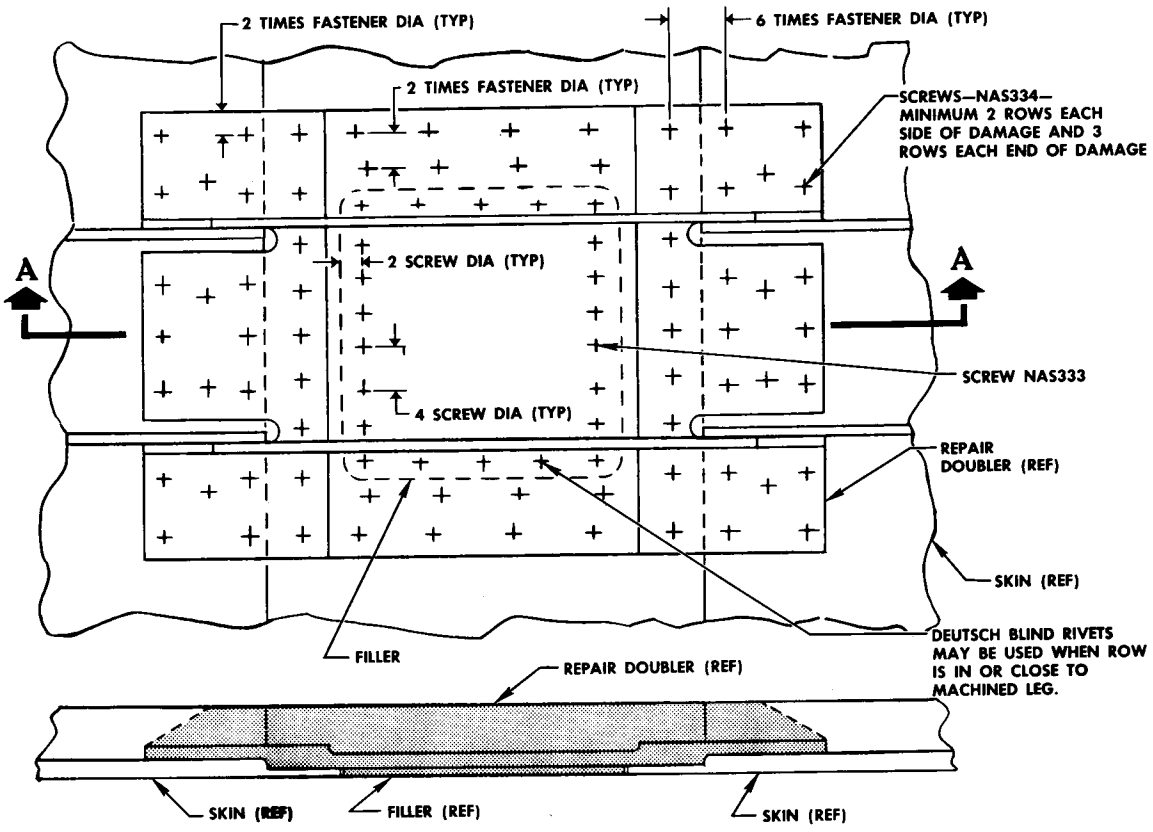
1. Trim out damaged area.
2. Fabricate repair parts.
3. Lay out and drill fastener holes.
4. Install dome nuts on doubler.
5. Seal dome nuts and apply sealing compound between shim and doubler and between shim and skin.
6. Insert shim and doubler through hole, and fasten in place.
7. Install filler in hole.

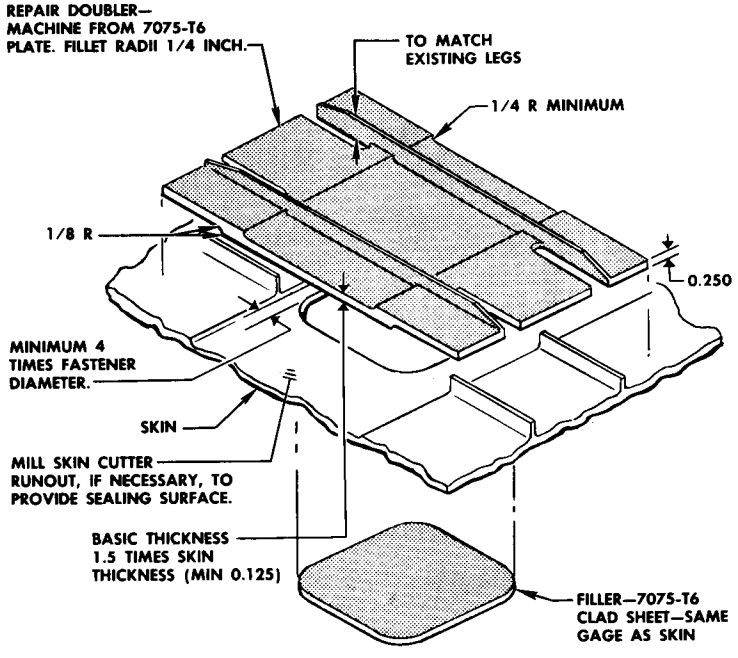
F-100D-3-10-100

Figure 2-37. Wing Skin Between Stringers

PERMANENT CATEGORY B

WING





1. Trim out damaged area.
2. Mill skin cutter runout to provide sealing surface if necessary.
3. Fabricate repair doubler and filler.
4. Install ESNA-A2500 series dome nuts on doubler if repair is in nonaccessible area. Use nuts and Stat-O-Seal washers in accessible areas.
5. Coat all mating surfaces with a thin coat of PR-1221 sealing compound before assembly.
6. Assemble repair as shown.
7. Run a fillet of PR-1221 sealing compound around all doubler edges and around and over dome nuts if used.
8. Overcoat all exposed PR-1221 sealing compound with fuel-resistant coating after curing.

NOTE Refer to index for wing sealing procedure.

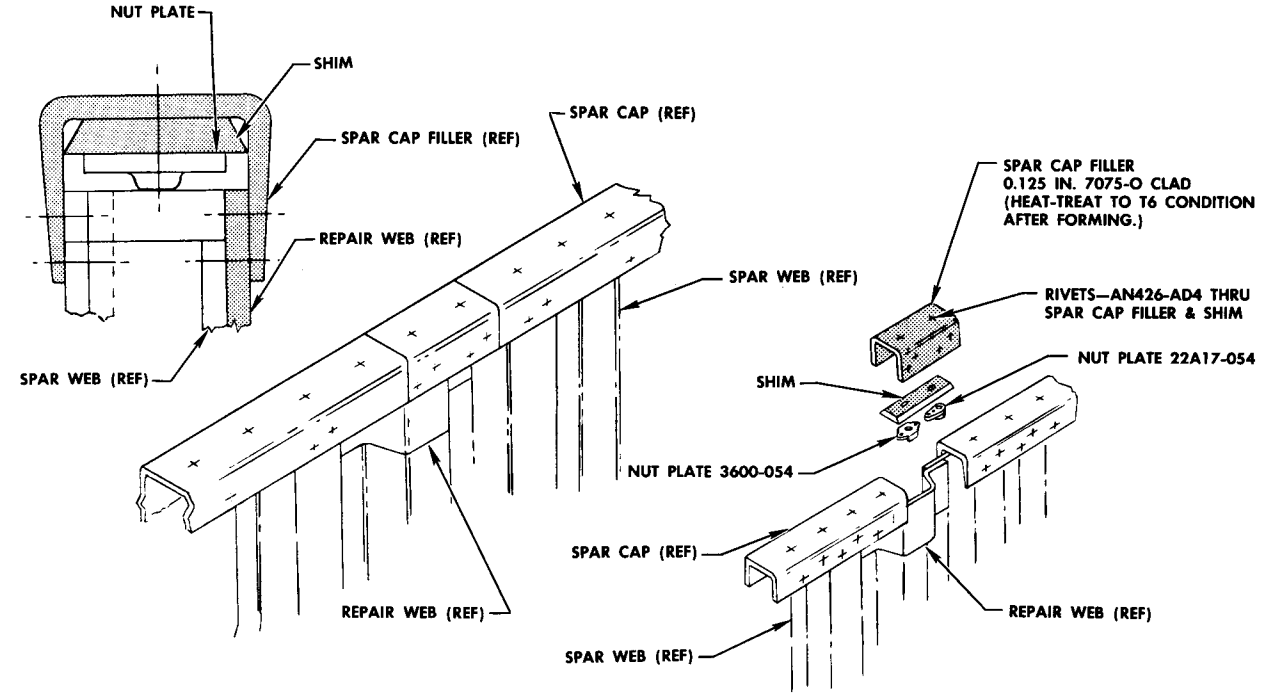
- When repair doubler extends over a rib land, alter rib as required, using standard rib repair.

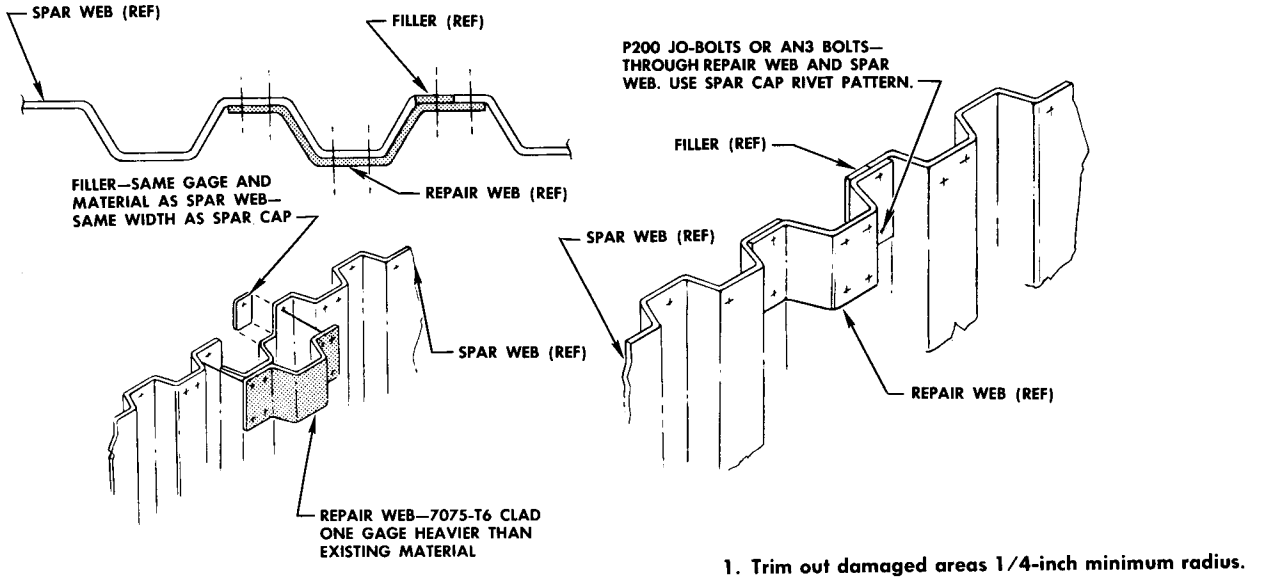
F-100D-3-10-101A

Figure 2-38. Wing Skin Between Ribs

PERMANENT—CATEGORY B

WING

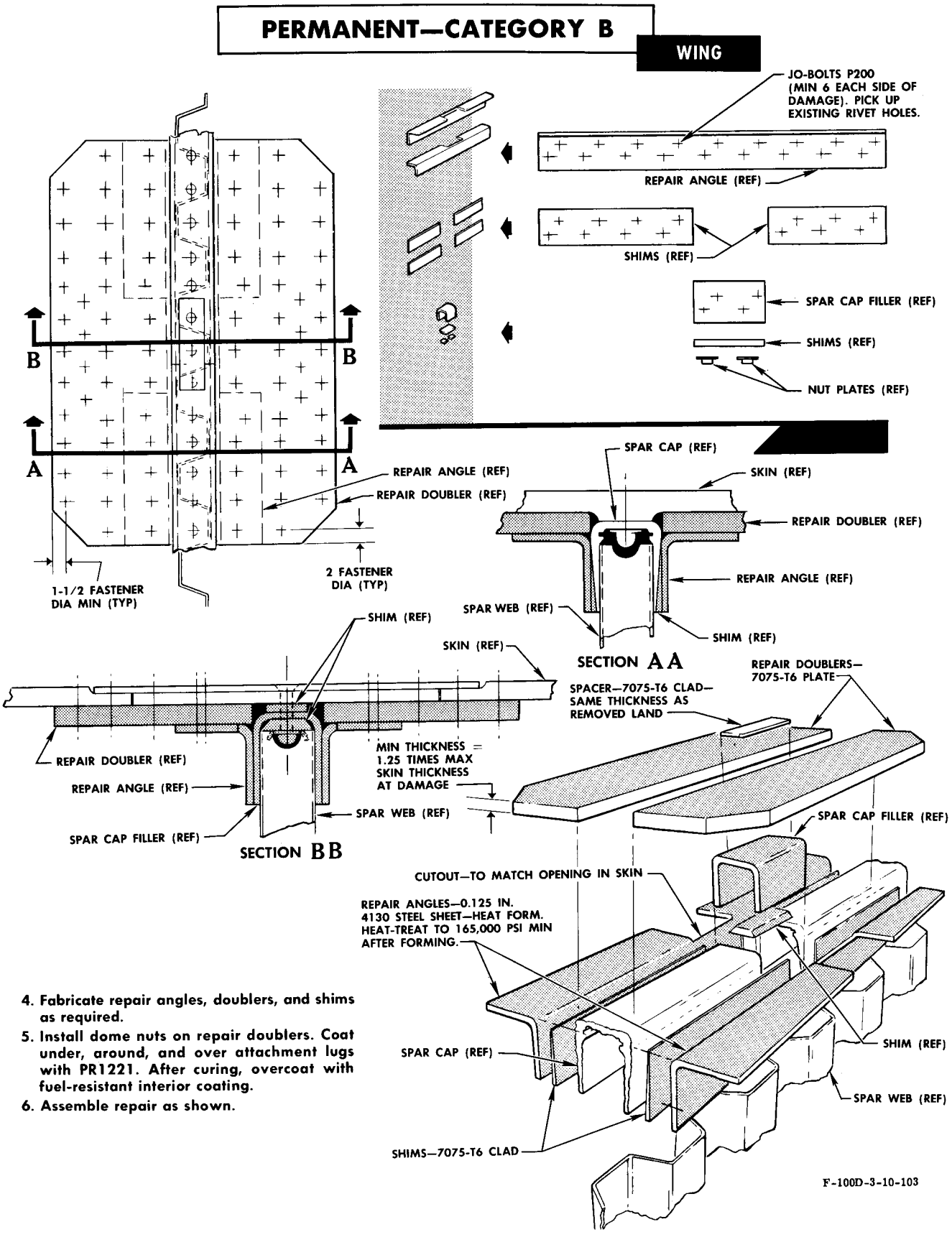




1. Trim out damaged areas 1/4-inch minimum radius.
2. Fabricate spar web and cap repair parts as required.
3. Assemble repair as shown.

F-100D-3-10-104

Figure 2-39. Wing Skin at Intermediate Spar (Sheet 1 of 3)

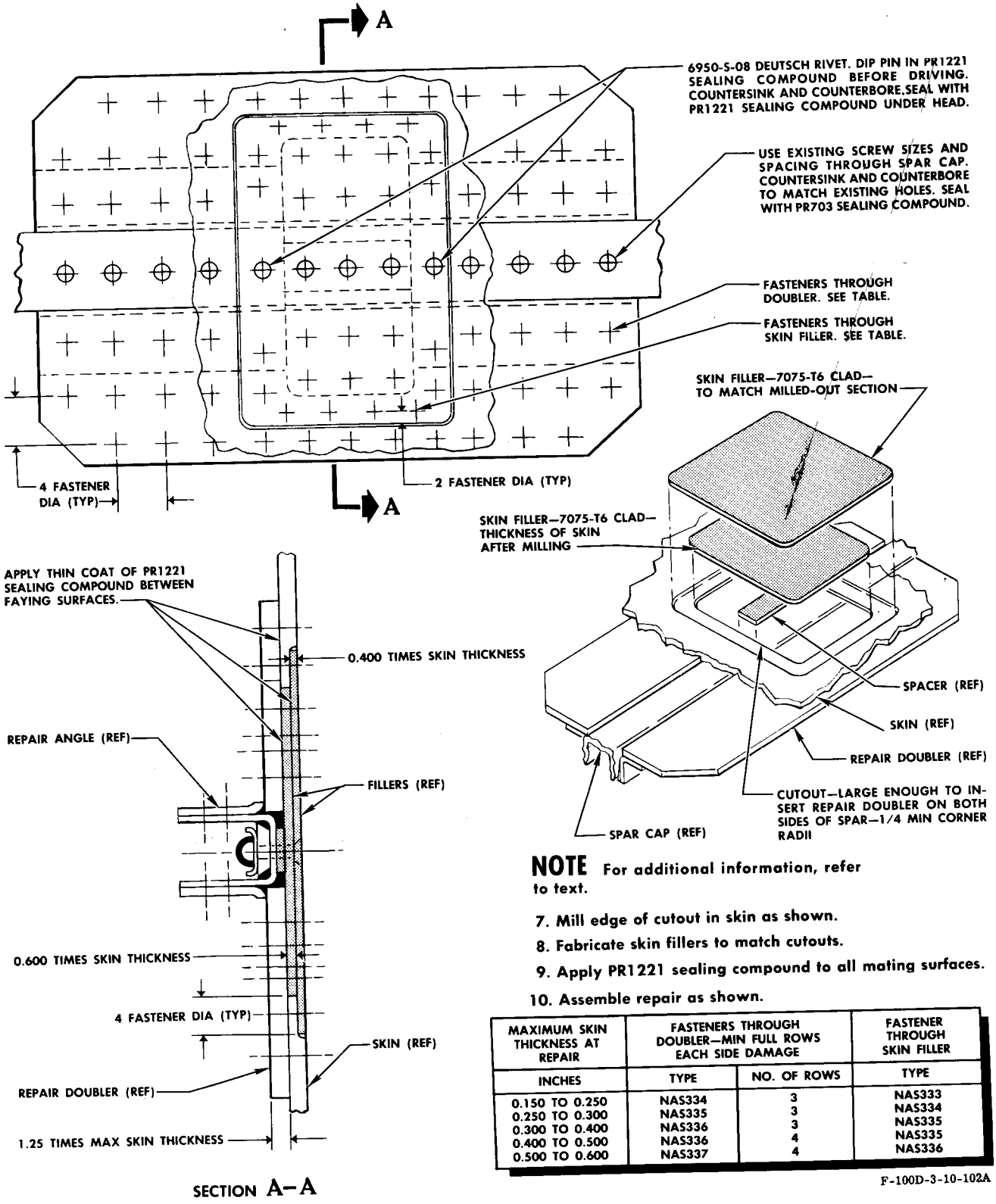


4. Fabricate repair angles, doublers, and shims as required.
5. Install dome nuts on repair doublers. Coat under, around, and over attachment lugs with PR1221. After curing, overcoat with fuel-resistant interior coating.
6. Assemble repair as shown.

Figure 2-39. Wing Skin at Intermediate Spar (Sheet 2 of 3)

PERMANENT—CATEGORY B

WING



NOTE For additional information, refer to text.

7. Mill edge of cutout in skin as shown.
8. Fabricate skin fillers to match cutouts.
9. Apply PR1221 sealing compound to all mating surfaces.
10. Assemble repair as shown.

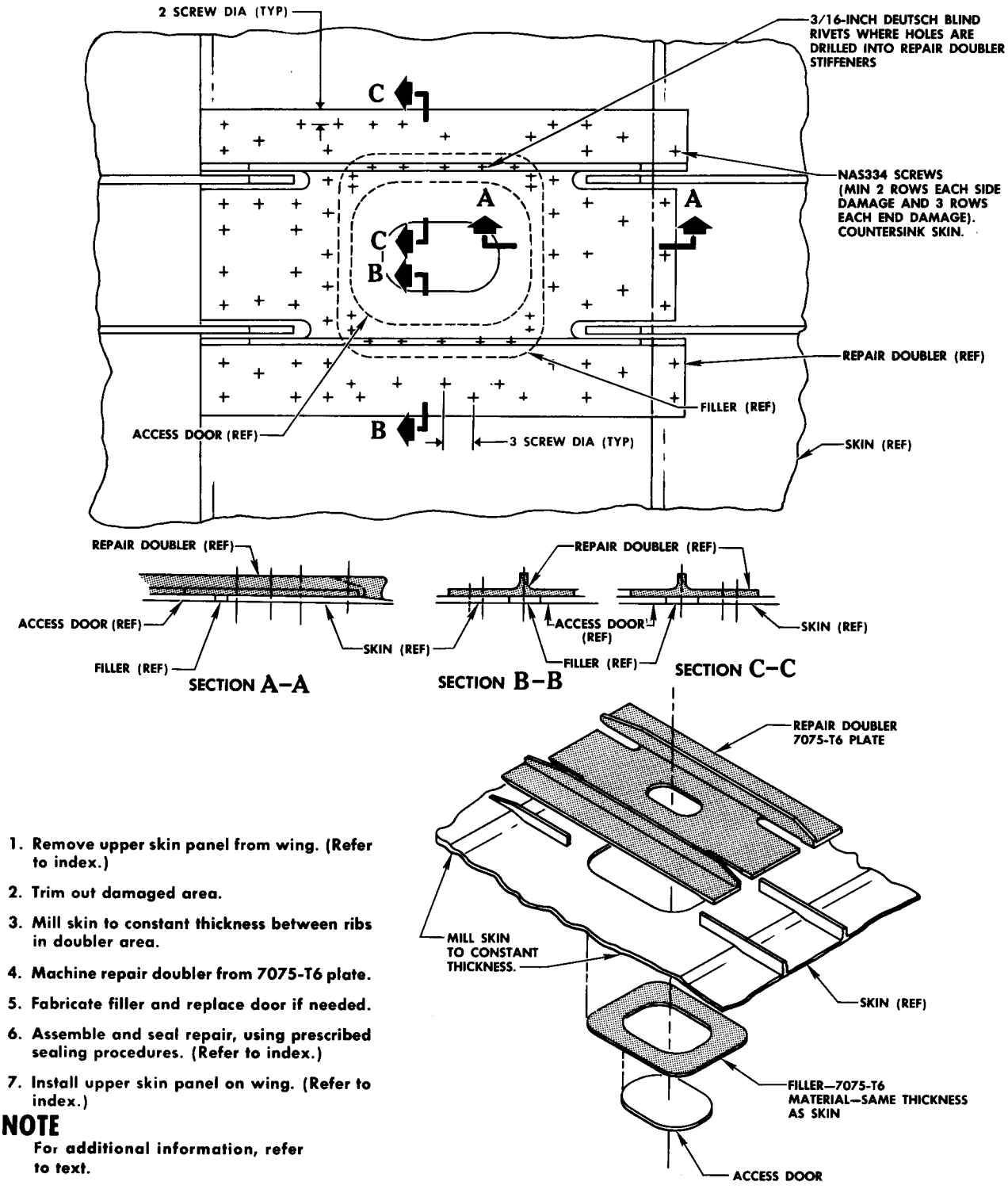
MAXIMUM SKIN THICKNESS AT REPAIR	FASTENERS THROUGH DOUBLER—MIN FULL ROWS EACH SIDE DAMAGE		FASTENER THROUGH SKIN FILLER
INCHES	TYPE	NO. OF ROWS	TYPE
0.150 TO 0.250	NAS334	3	NAS333
0.250 TO 0.300	NAS335	3	NAS334
0.300 TO 0.400	NAS336	3	NAS335
0.400 TO 0.500	NAS336	4	NAS335
0.500 TO 0.600	NAS337	4	NAS336

F-100D-3-10-102A

Figure 2-39. Wing Skin at Intermediate Spars (Sheet 3 of 3)

PERMANENT—CATEGORY B

WING



1. Remove upper skin panel from wing. (Refer to index.)
2. Trim out damaged area.
3. Mill skin to constant thickness between ribs in doubler area.
4. Machine repair doubler from 7075-T6 plate.
5. Fabricate filler and replace door if needed.
6. Assemble and seal repair, using prescribed sealing procedures. (Refer to index.)
7. Install upper skin panel on wing. (Refer to index.)

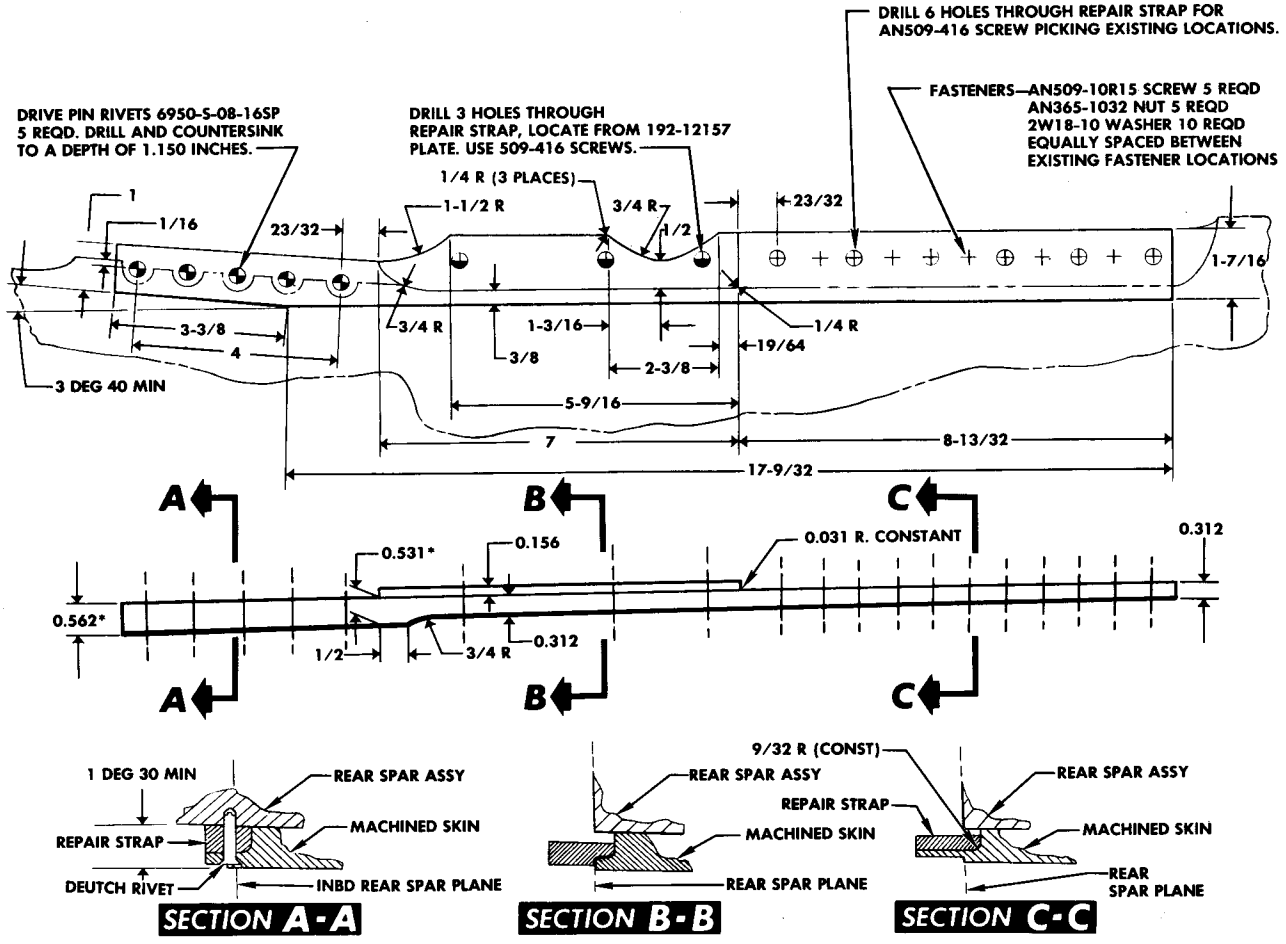
NOTE

For additional information, refer to text.

Figure 2-40. Wing Skin at Door Land

PERMANENT CATEGORY B

WING

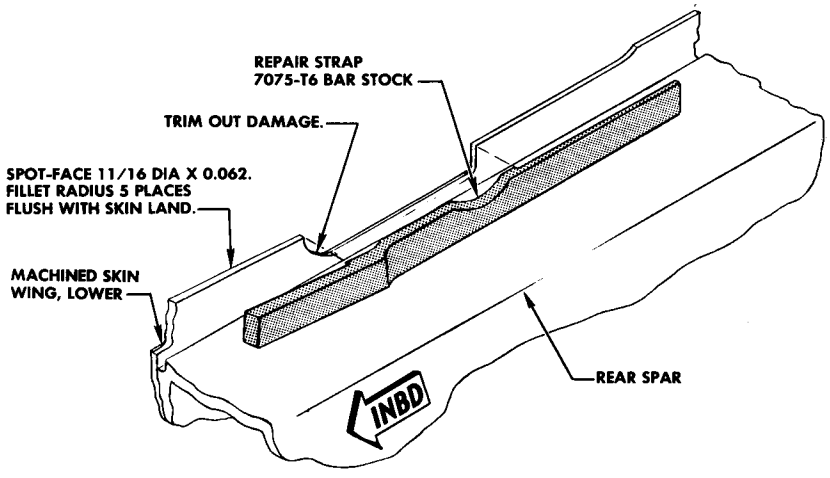


1. Trim out damaged skin midway between existing fastener locations.
2. Damage length may vary from dimensions shown, but must not exceed 7 inches in length. For more extensive damage, refer to repair straps shown on sheet 2.
3. Machine repair strap from 7075-T6 bar stock to the dimensions shown.
4. Fasten repair strap in place using the size and type of fasteners shown and the spacing called out.

*These dimensions taken on aft face of repair bar.

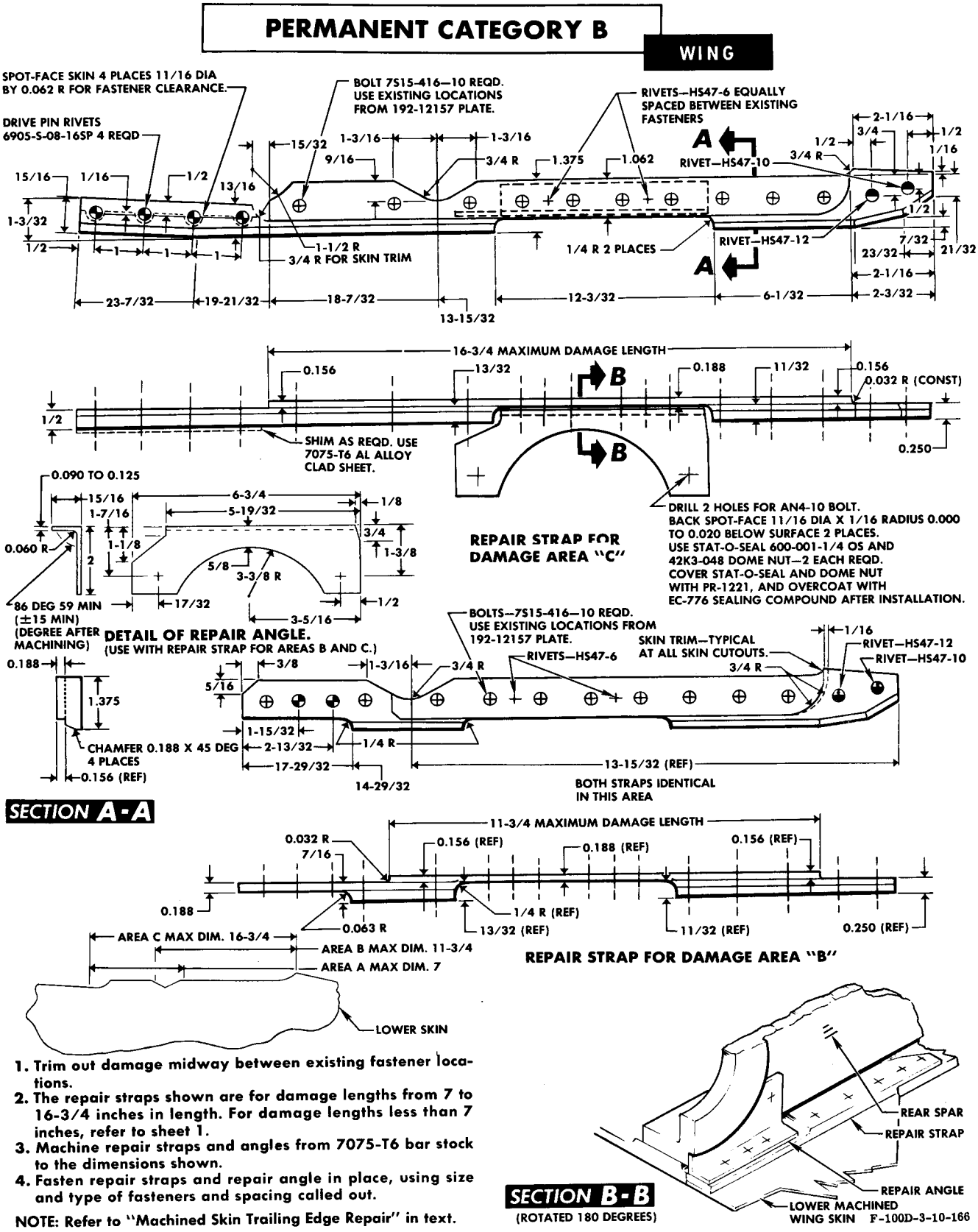
NOTE

Refer to "machined skin trailing edge repair" in text.



F-100D-3-10-164

Figure 2-40A. Strap—Lower Skin and Rear Spar (Sheet 1 of 2)



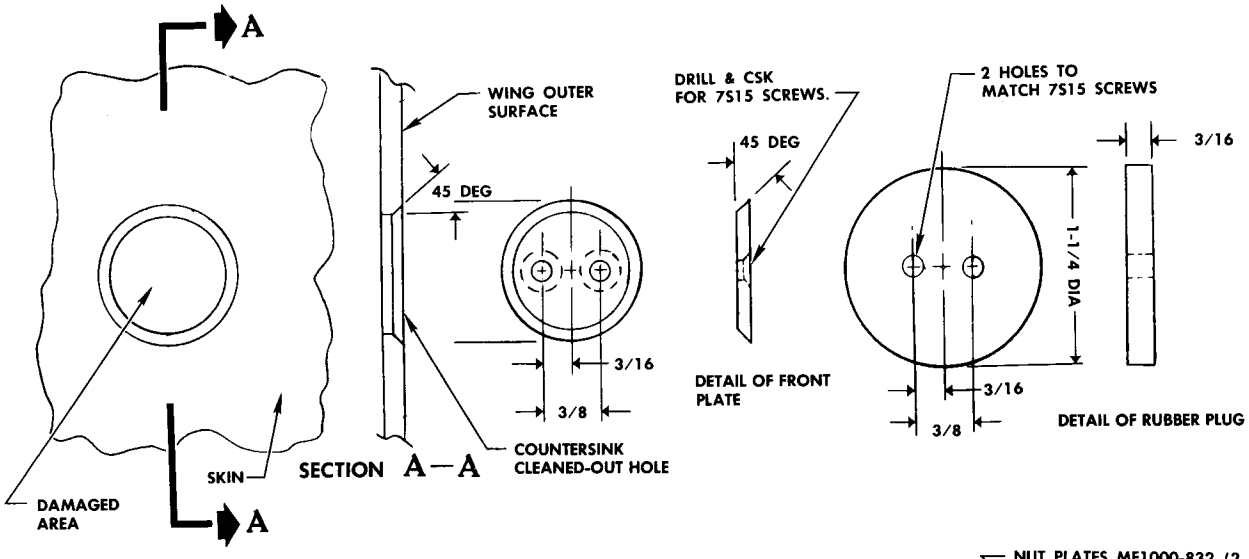
1. Trim out damage midway between existing fastener locations.
2. The repair straps shown are for damage lengths from 7 to 16-3/4 inches in length. For damage lengths less than 7 inches, refer to sheet 1.
3. Machine repair straps and angles from 7075-T6 bar stock to the dimensions shown.
4. Fasten repair straps and repair angle in place, using size and type of fasteners and spacing called out.

NOTE: Refer to "Machined Skin Trailing Edge Repair" in text.

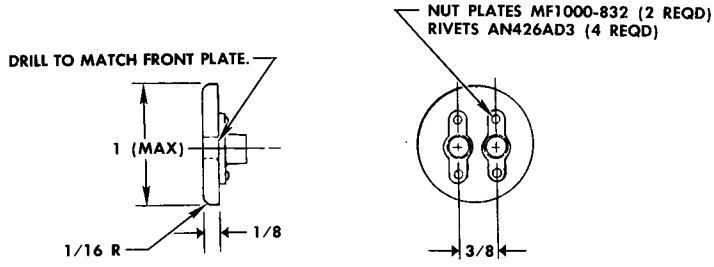
Figure 2-40A. Strap—Lower Skin and Rear Spar (Sheet 2 of 2)

ONE-TIME FLIGHT REPAIR

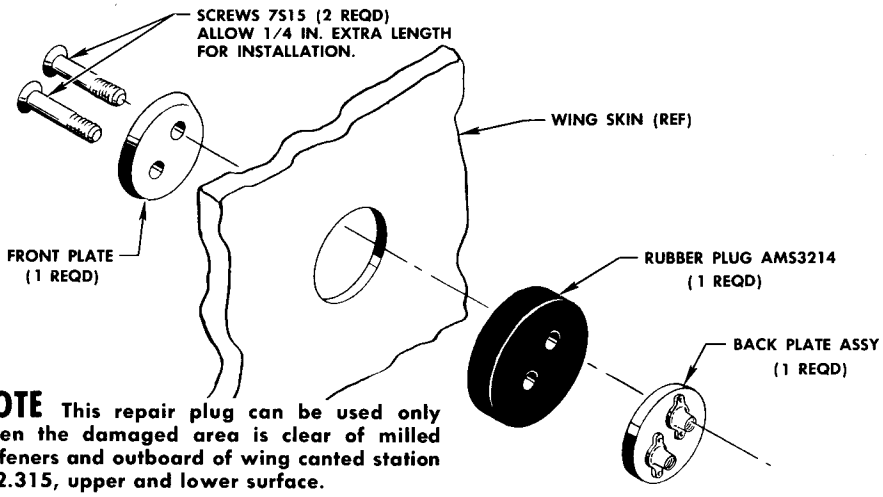
WING

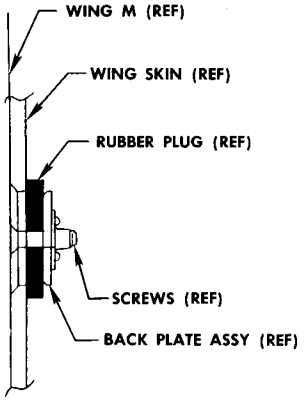


1. Trim out damaged area to a one-inch maximum diameter circular hole.
2. Countersink hole to one-half skin thickness. If skin thickness is 1/8 inch or less, countersink full depth of skin.
3. Make front plate to match countersunk hole. Drill two holes as shown.
4. Make rubber plug from AM53214 stock, as shown.
5. Make backup plate, and assemble with two nut plates as shown.
6. Seal repair plug parts with PR-1221 sealing compound and install as shown. Tighten screws snugly. If the screws are overtorqued, the rubber plug will tear, and leakage will start.



DETAIL OF BACKUP PLATE





INSTALLED PLUG ASSY

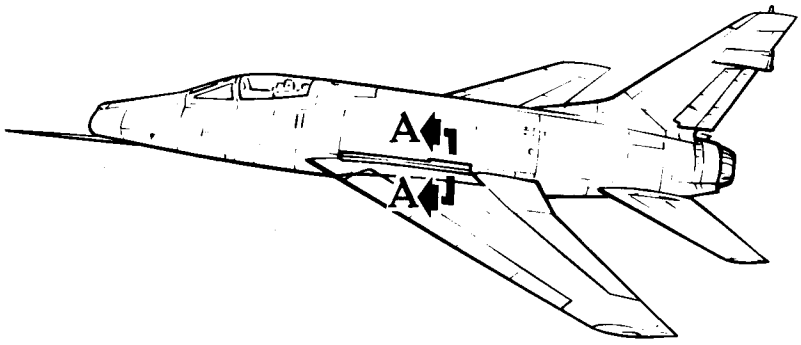
NOTE This repair plug can be used only when the damaged area is clear of milled stiffeners and outboard of wing canted station 102.315, upper and lower surface.

F-100D-3-10-134A

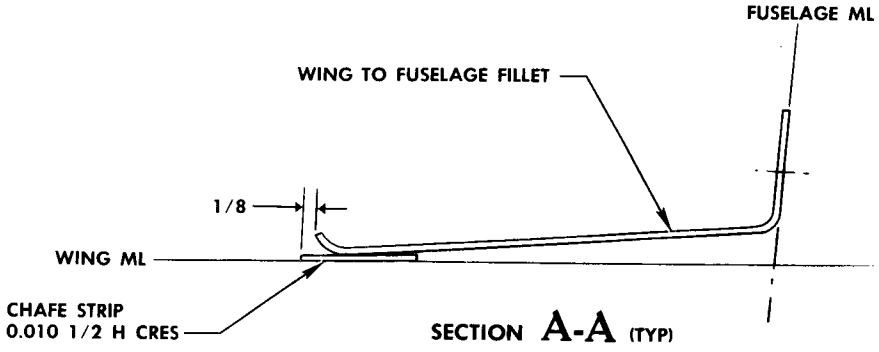
Figure 2-41. Integral Fuel Tank Skin Plug

PERMANENT—CATEGORY A

WING



NOTE Repair is applicable to the upper and lower wing surfaces.



1 Clean corroded area with solvent. Refer to index for solvent cleaning.

2 Remove corrosion by means of a hand scraper or abrasive wheel, using care not to remove excessive metal. After removing corrosion, blend and smooth area to a fine surface finish. Refer to index for removal of corrosion from bare aluminum alloys. Check area to determine depth of metal removed.

NOTE Removal of more than 0.032 inch of upper or lower surface metal will require a major skin repair in addition to this repair procedure.

3 Treat worked area with Alodine 1200 (MIL-C-5541). Refer to index for protective treatment for aluminum alloys.

4 Make chafe strips of 0.010 x 1 x 2 one-half hard corrosion-resistant steel sheet.

5 Locate three chafe strips equally spaced under each fillet and 1/8 inch outboard of the fillet edge as indicated.

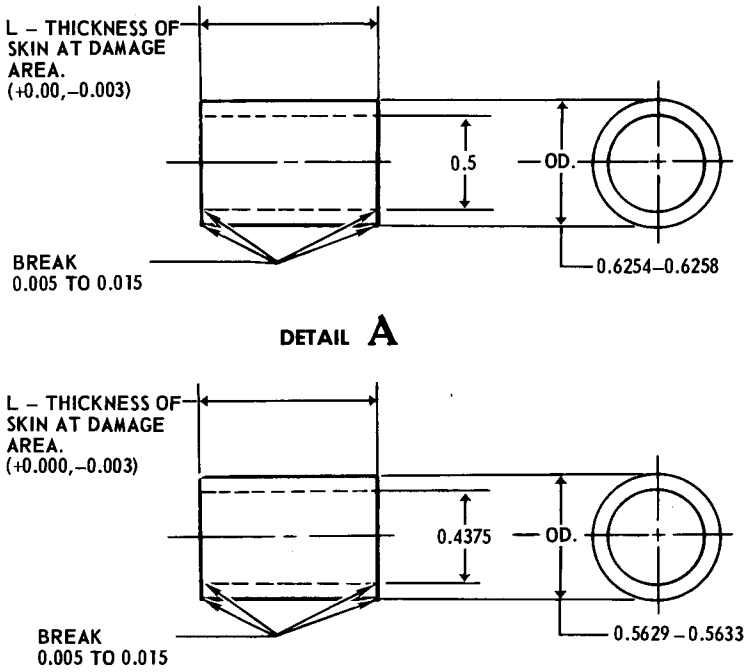
6 Attach chafe strips to skin with Epon VIII adhesive or equivalent. Surfaces must be clean and dry before the adhesive is applied.

F-100D-3-10-185

Figure 2-41A. Wing Skin Corrosion Damage Repair

PERMANENT—CATEGORY A

WING



MATERIAL: STAINLESS STEEL BAR ROD OR TUBE STOCK IN THE 300 SERIES, 1/4 OR 1/2 HARD (NO HEAT TREAT REQUIRED).

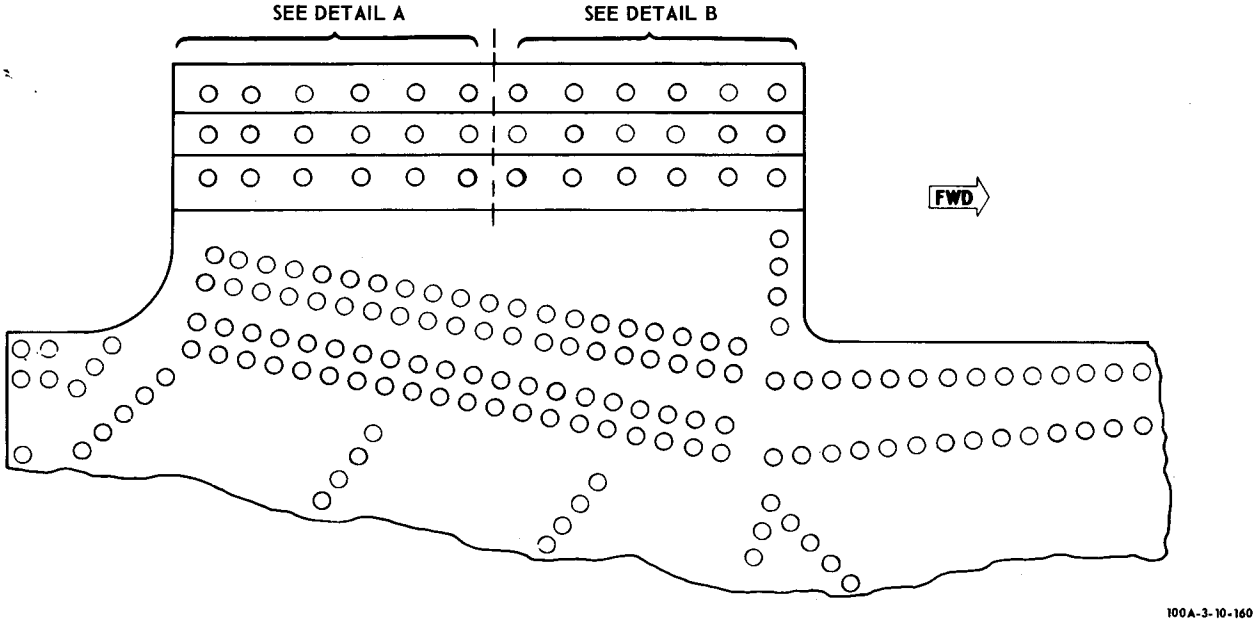
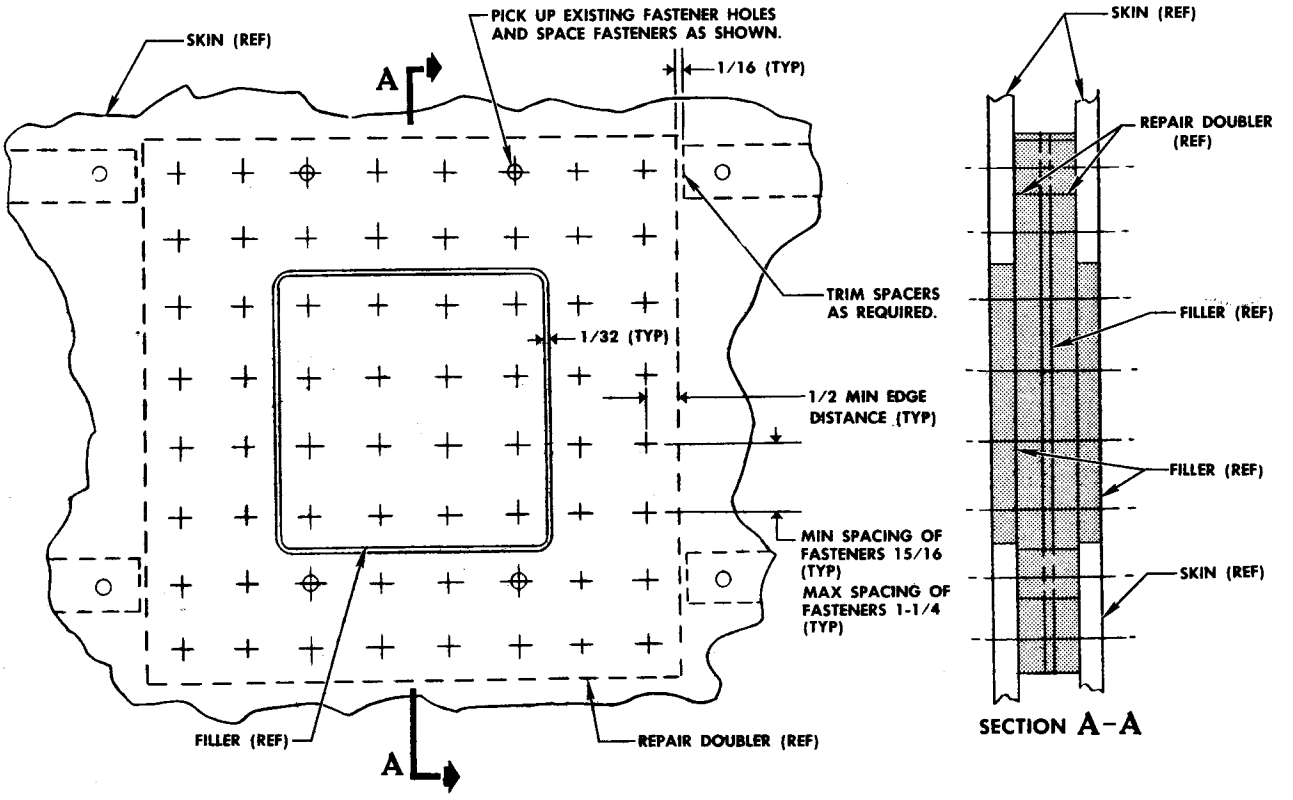
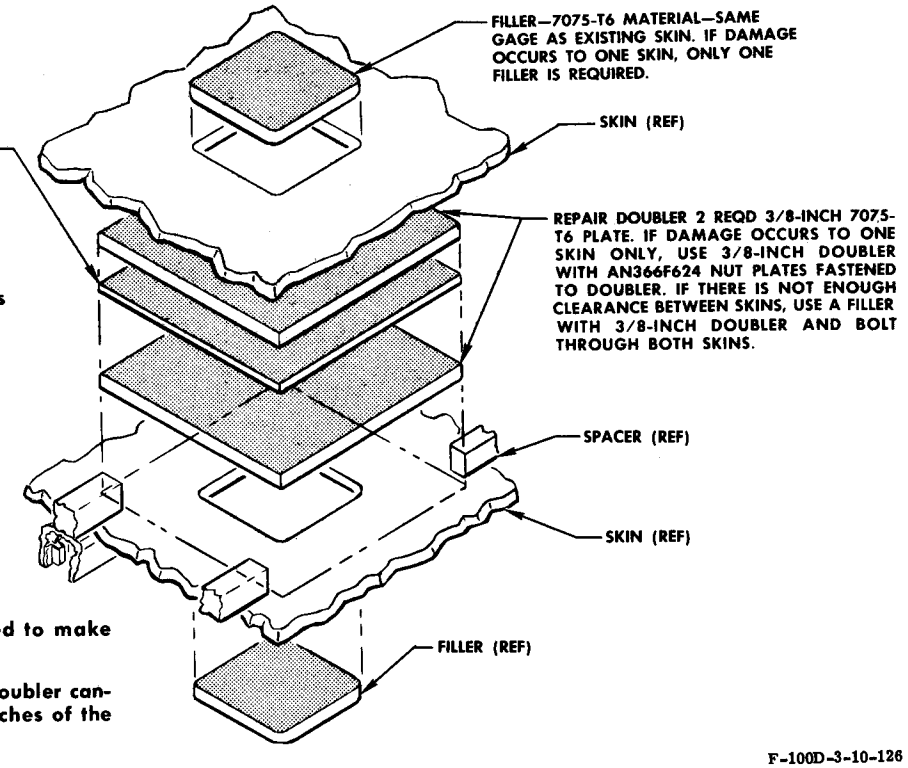


Figure 2-41B. Wing Skin Tongue Area Repair Bushing

PERMANENT-CATEGORY A

WING





1. Trim out damaged area.
2. Cut out spacers as required.
3. Fabricate doublers and fillers as required.
4. Assemble repair and install fasteners. Use three MS20004 bolts per inch of damage. (Refer to "Fastener Layout Instructions.")

NOTE The skin may be removed to make the repair, if necessary.

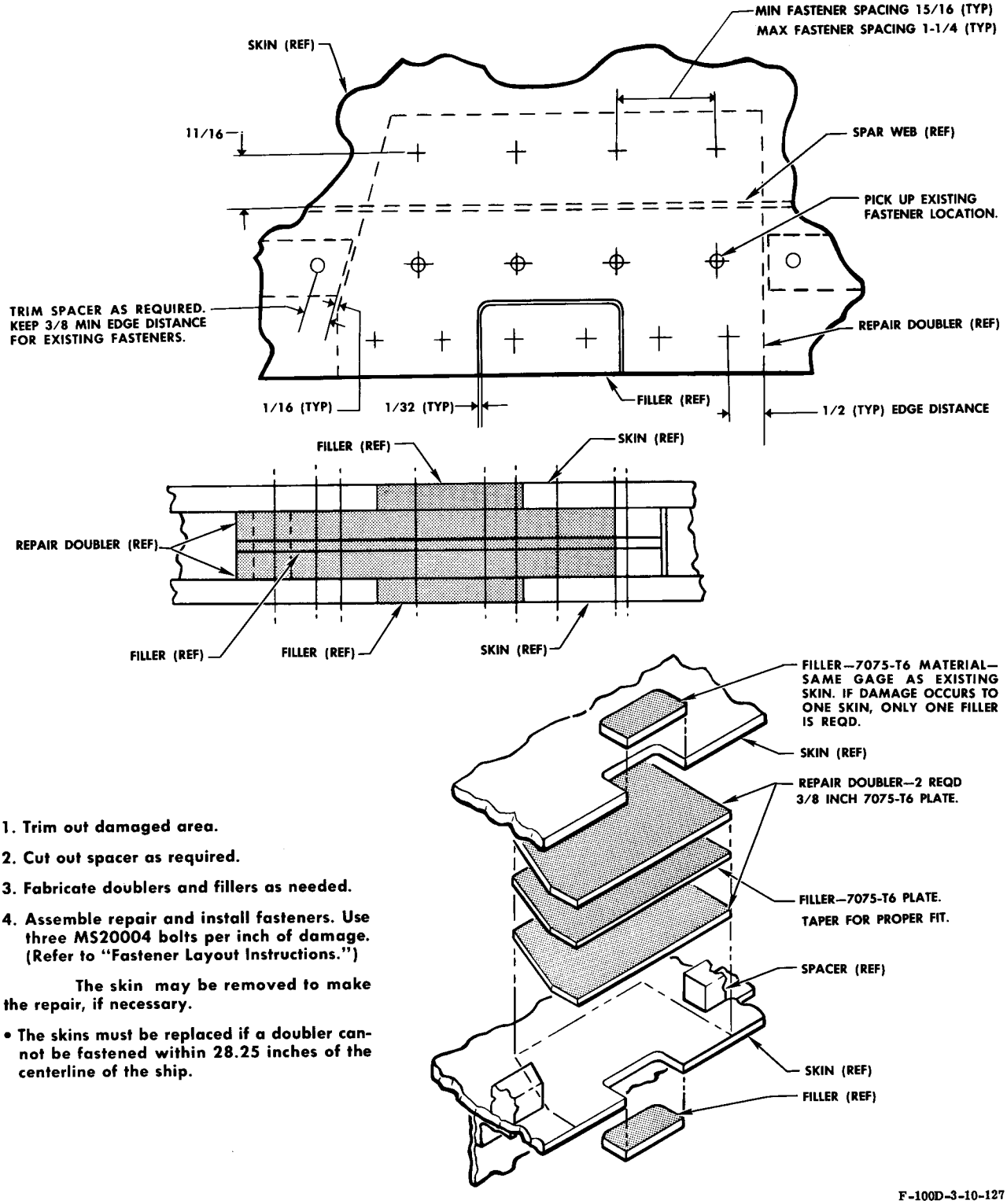
- The skins must be replaced if a doubler cannot be fastened within 28.25 inches of the centerline of the ship.

F-100D-3-10-126

Figure 2-42. Center Section Skin—Inner and Outer

PERMANENT - CATEGORY A

WING



1. Trim out damaged area.
2. Cut out spacer as required.
3. Fabricate doublers and fillers as needed.
4. Assemble repair and install fasteners. Use three MS20004 bolts per inch of damage. (Refer to "Fastener Layout Instructions.")

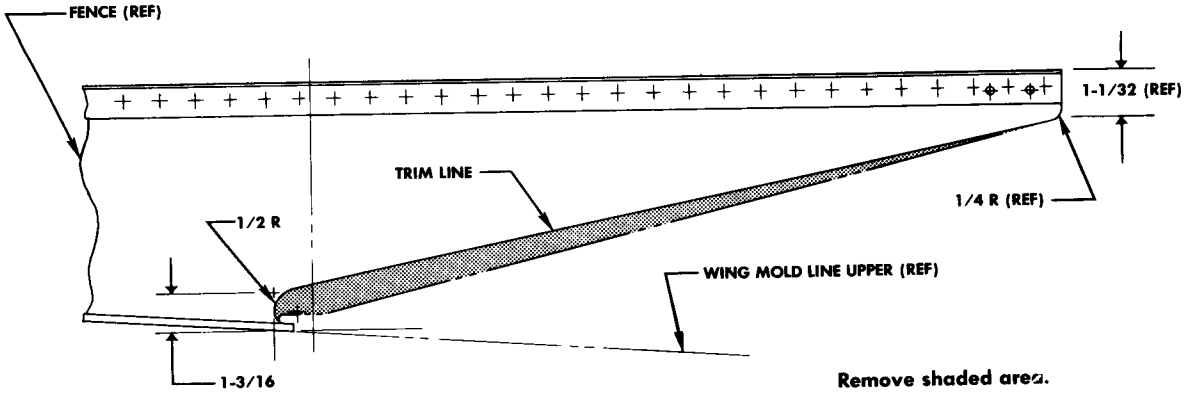
The skin may be removed to make the repair, if necessary.

- The skins must be replaced if a doubler cannot be fastened within 28.25 inches of the centerline of the ship.

Figure 2-43. Center Section Skin—Edge

PERMANENT—CATEGORY A

WING

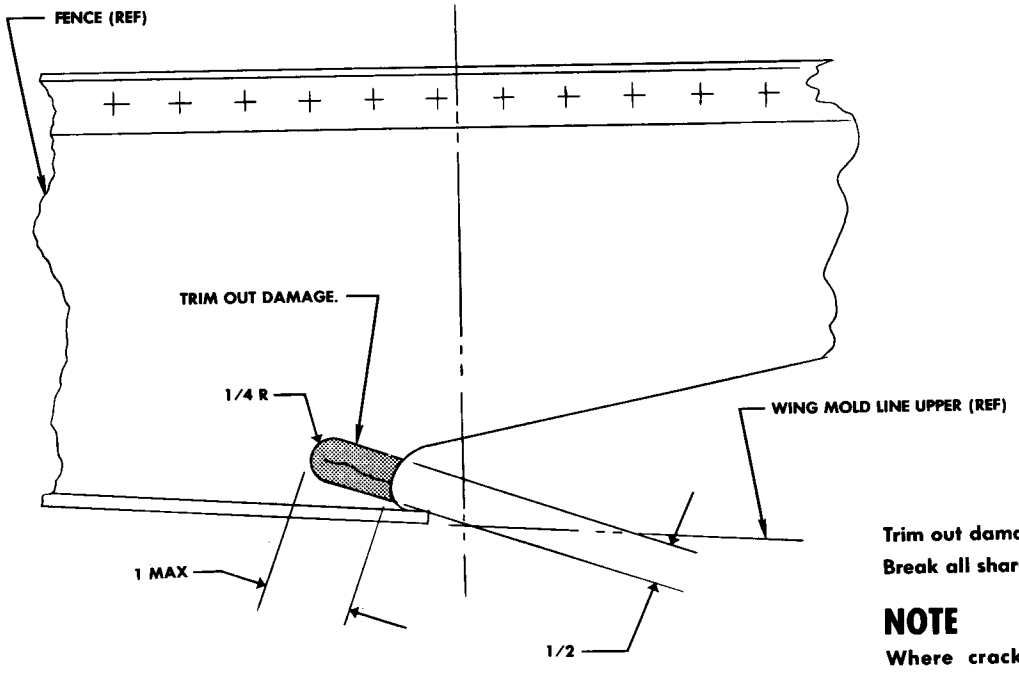


Remove shaded area.

Break all sharp edges.

NOTE

This rework will aid in preventing small cracks resulting from aero—dynamic forces.



Trim out damaged area (shaded).

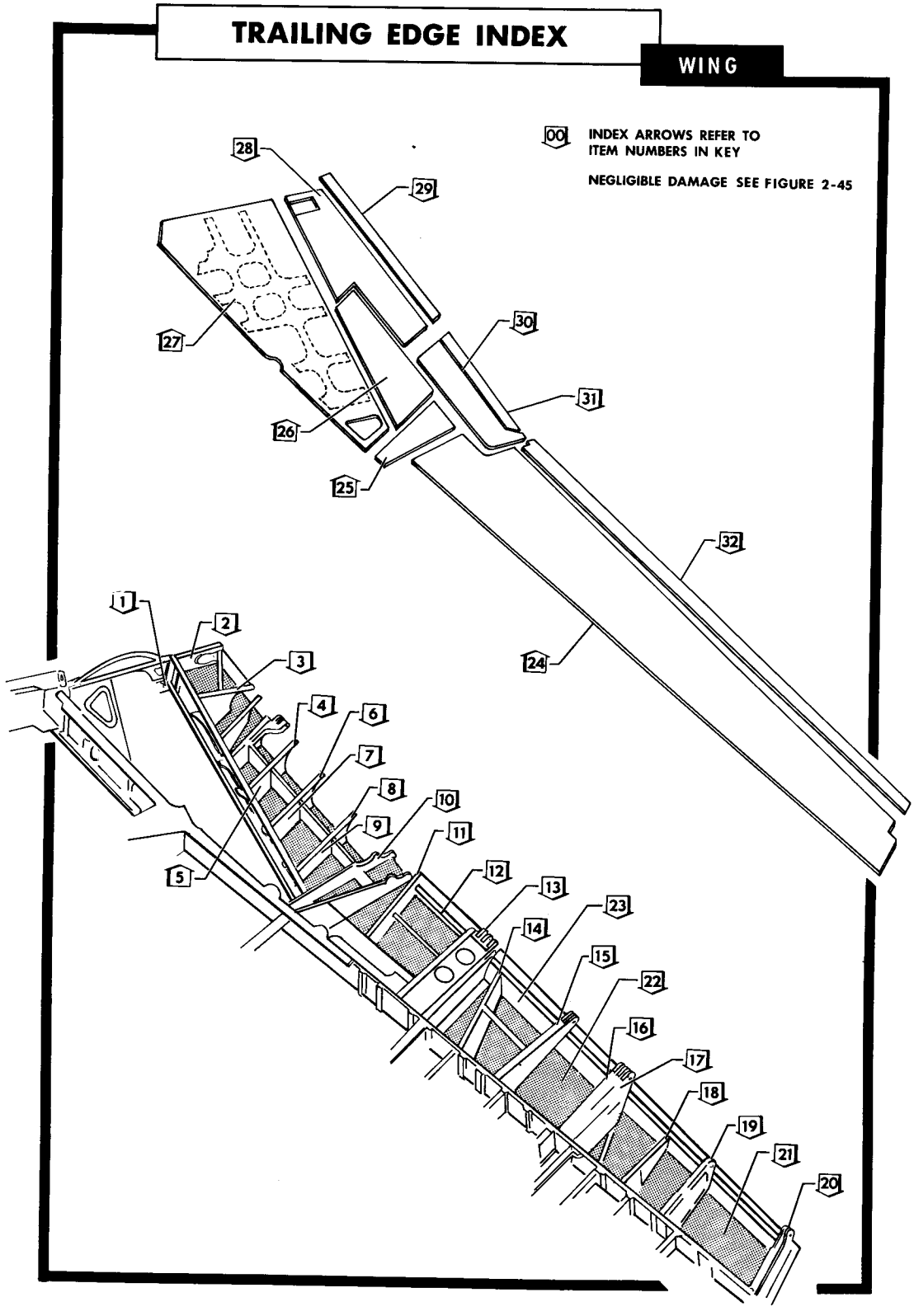
Break all sharp edges.

NOTE

Where crack exceeds one inch in length, replace fence.

F-100D-3-10-1

Figure 2-43A. Wing Fence Repair



F-100D-3-10-F

Figure 2-44. Trailing Edge Repair Index

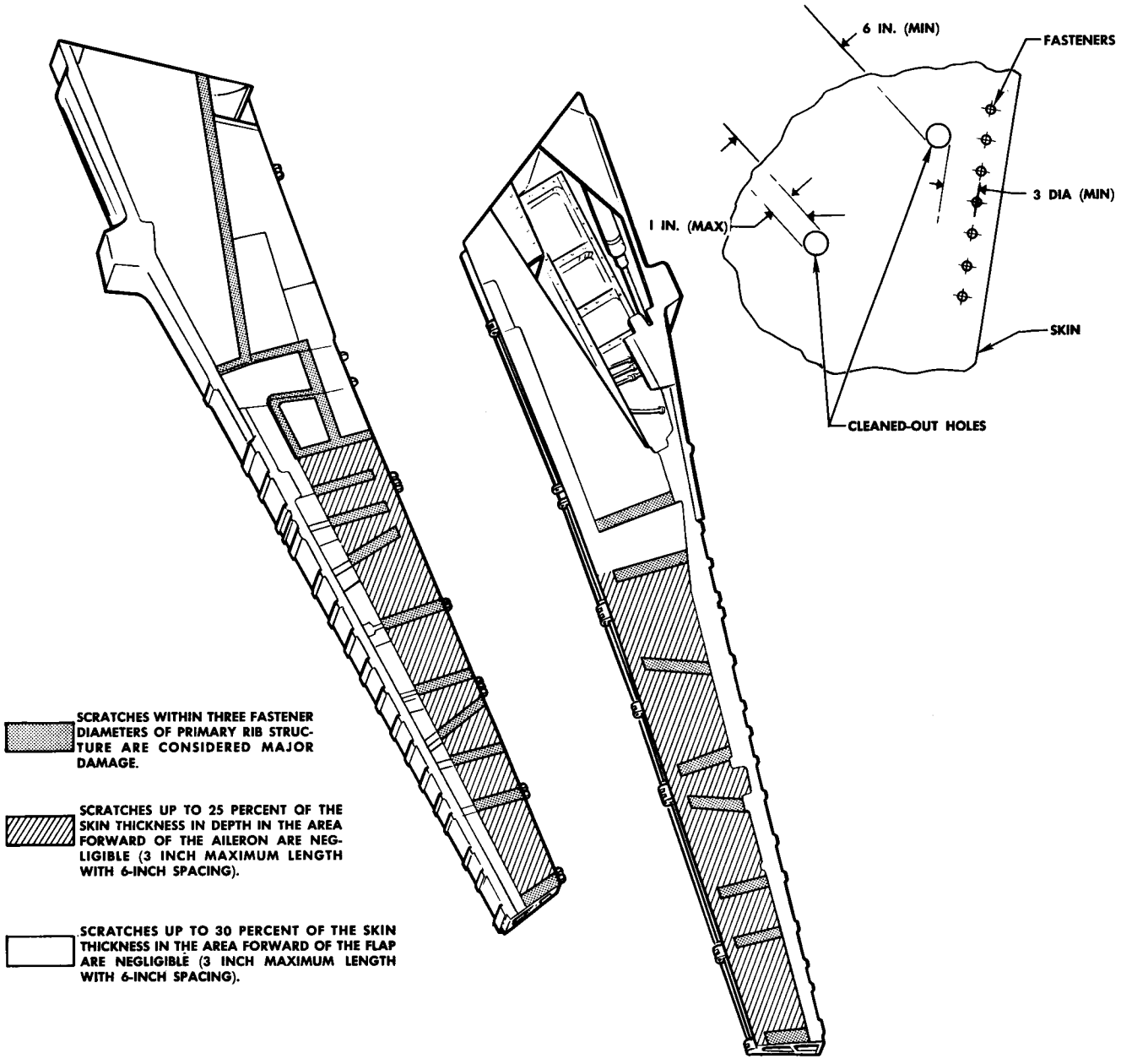
ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	180-14362	Doubler Assy	0.040	2024-T4 Clad	2-46
2	223-14268	Rib Assy	0.063	Cres	
3	223-14269	Rib Assy	0.040	Cres	
4	223-14272	Rib	0.063	2024-T4 Clad	
5	223-14266	Rib	0.063	2024-T4 Clad	
6	223-14273	Rib	0.063	2024-T4 Clad	
7	223-14267	Rib	0.050	2024-T4 Clad	
8	223-14264-11	Rib	0.040	Cres	
9	223-14264-3	Rib	0.064	2024-T4 Clad	
10	223-14359	Hinge Assy		4140 Stl Die Forging	
11	223-14552	Angle Assy	Extr	2024-T42	
12	223-14091	Spar Assy	0.040	2024-T4 Clad	
13	223-14317	Fitting		2014-F Die Forging	2-46B
14	223-14247	Rib Assy	0.051	2024-T4 Clad	2-23
15	180-14359	Fitting		2014-T6 Die Forging	2-46B
16	223-14308	Fitting			
17	223-14309	Fitting			
18	223-14250	Rib Assy	0.051	2024-T4 Clad	2-23
19	223-14251	Fitting		2014-T6 Die Forging	2-46B
20	223-14242	Rib		2014-T6 Die Forging	2-46B
21	223-14003-5	Skin	0.040	7075-T6 Clad	
22	223-14003-7	Skin	0.040	7075-T6 Clad	
23	223-14854	Trailing Edge	Extr	6061-T6	
24	223-14003-3	Skin	0.040	7075-T6 Clad	
25	223-14151	Skin	0.188	7075-T6 Plate	
26	223-14009	Skin	0.063	7075-T6 Clad	
27	223-14146	Skin	0.250	7075-T6 Plate	2-46, 2-46A
28	223-14009	Skin	0.063	7075-T6 Clad	
29	223-14869	Trailing Edge	Extr	6061-T6 Extr	
30	223-14793	Cover	0.063	2024-T4 Clad Sh	
31	223-14869	Trailing Edge	Extr	6061-T6 Extr	
32	223-14853	Trailing Edge	Extr	6061-T6 Extr	

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 2-44

TRAILING EDGE NEGLIGIBLE DAMAGE

WING



NOTE Cleaned-out holes of one inch maximum diameter with a minimum of three times the fastener diameter from any fastener and a minimum of six inches between the edges of the holes are considered negligible.
All surface holes should be plugged. For minor hole repairs, refer to index.

CORROSION

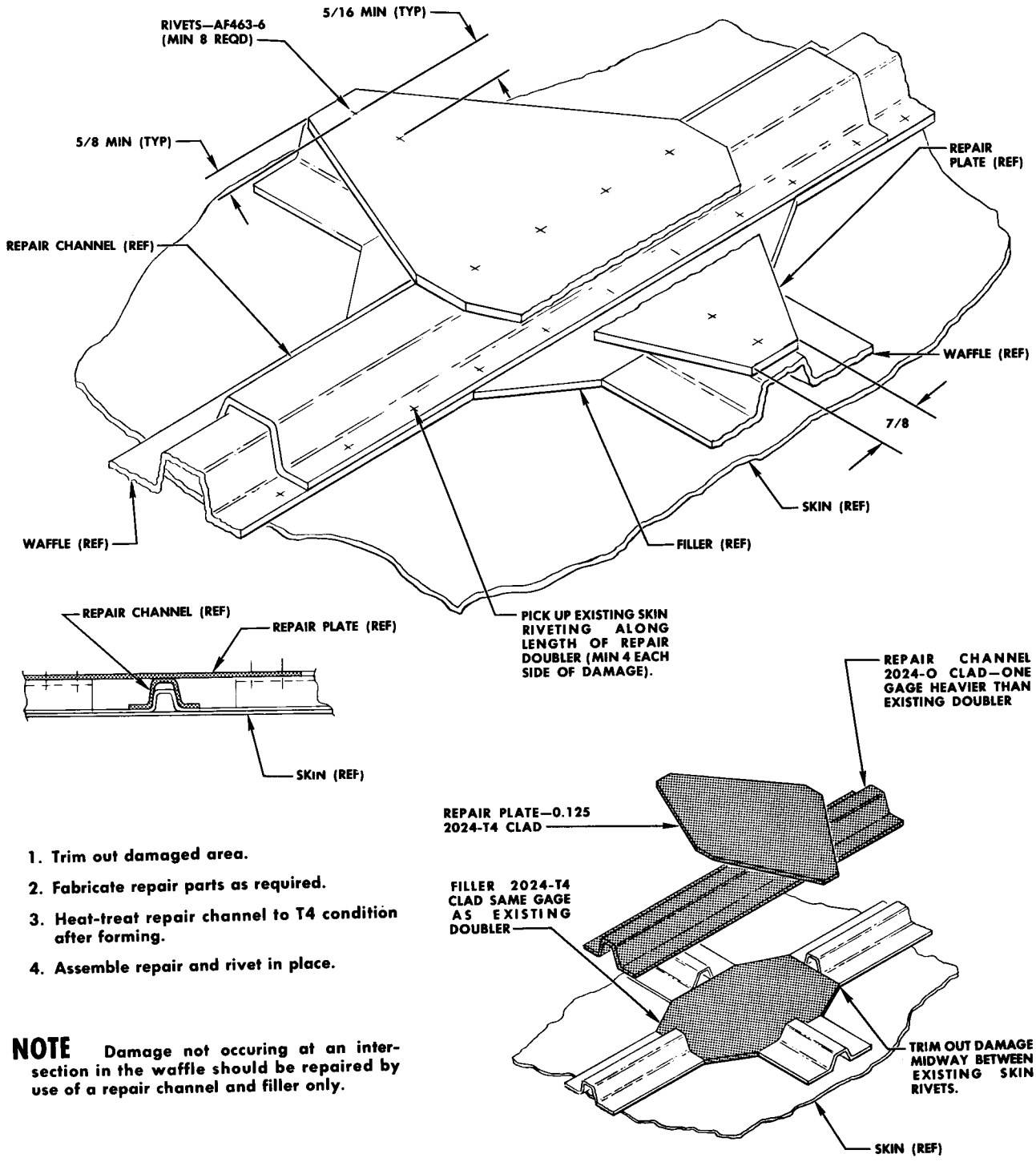
Corrosion damage not exceeding limits for cracks or scratches may be considered negligible. The corroded area must be cleaned and the chemical film treated. (Refer to index for "Corrosion Rework by Chemical Film Treatments.")

F-100D-3-10-[illegible]1B

Figure 2-45. Trailing Edge Negligible Damage

PERMANENT—CATEGORY A

WING



1. Trim out damaged area.
2. Fabricate repair parts as required.
3. Heat-treat repair channel to T4 condition after forming.
4. Assemble repair and rivet in place.

NOTE Damage not occuring at an intersection in the waffle should be repaired by use of a repair channel and filler only.

Figure 2-46. Wheel Well Waffle

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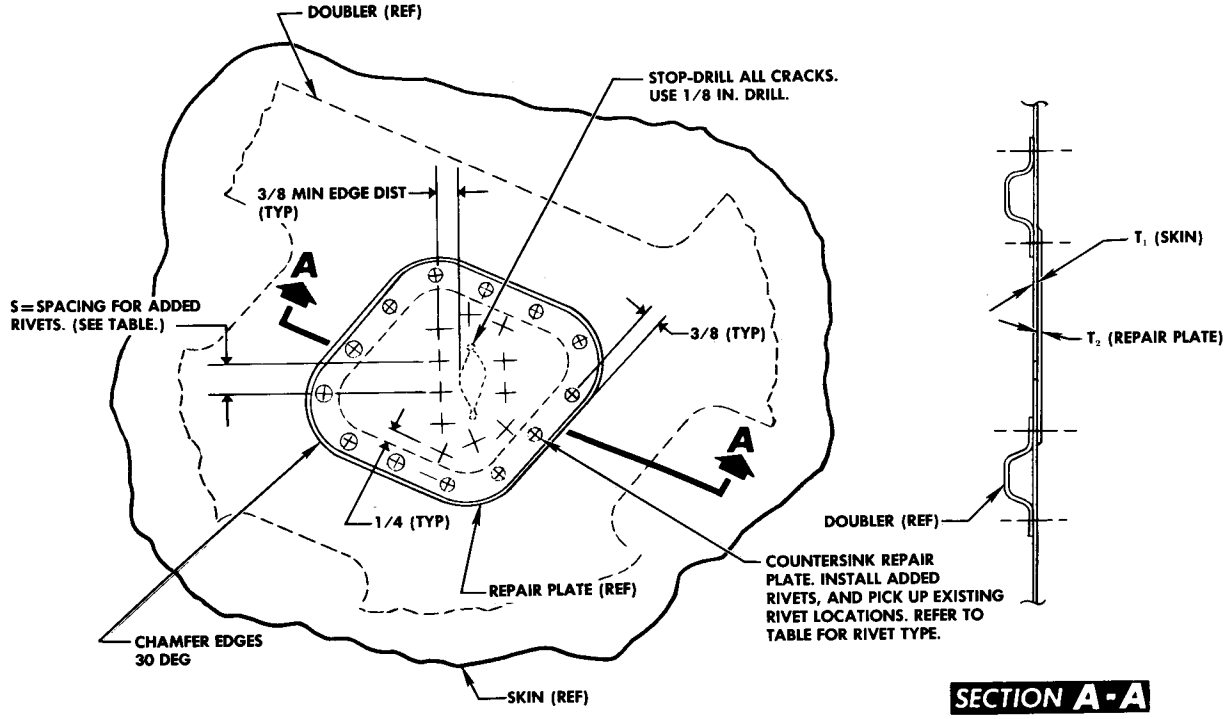
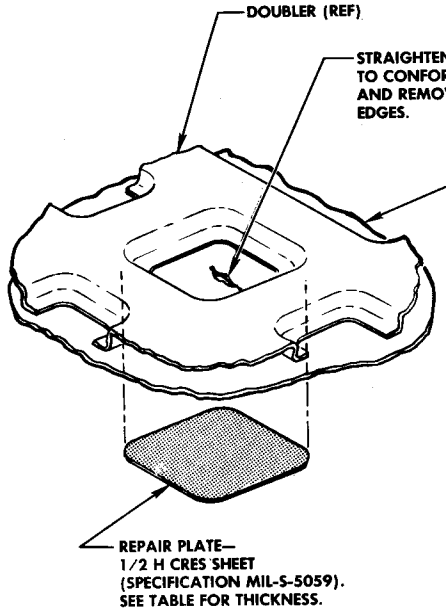
TEMPORARY—CATEGORY A

WING





REPAIR TABLE			
T_1	T_2	S	RIVET
0.062	0.050	1/2 TO 7/8	AN426AD5
0.125	0.080	5/8 TO 1	AN426DD6
0.250	0.125	5/8 TO 1	HS47-6-

1. Straighten damaged area, and remove rough edges.
2. Stop-drill cracks.
3. Fabricate repair plate.
4. Countersink and install fasteners in repair plate, using repair table for required spacing and rivet type.

NOTE

Damage not exceeding 3 inches maximum length may be repaired in this category.

- Where doubler does not completely surround damage, trim repair plate to pick up existing rivets approximately 1-1/2 inches from damage.
- Add rivets as necessary, using repair table for rivet type and spacing. Maintain 3/8-inch minimum edge distance.

F-100D-3-10-173

Figure 2-46A. Wing Skin Repair

PERMANENT—CATEGORY A

WING

PERMISSIBLE WEAR ALLOWANCES

ITEM NO.	PART DESCRIPTION	SHAFT DESCRIPTION	MFD MAX HOLE ID	MFD MIN SHAFT OD	MFD MAX ALLOWANCE	WEAR MAX TOLERANCE
1	223-14242 RIB	EWB18-4H20 BOLT	0.254	0.2485	0.0055	0.0016
2	223-14251 FITTING	EWB18-4H19 BOLT	0.254	0.2485	0.0055	0.0016
3	180-14312 FITTING	180-16035 BOLT	0.5630	0.5616	0.0014	0.0005
4	180-14359 FITTING	EWB18-4H2A BOLT	0.254	0.2485	0.0055	0.0016
5	223-14316 FITTING	223-16035 BOLT	0.5630	0.5616	0.0014	0.0005
6	180-14311 FITTING	192-523178 BOLT	0.8755	0.8736	0.0019	0.0005
7	180-14310 FITTING	192-523178 BOLT	0.8755	0.8736	0.0019	0.0005
8	223-14317 FITTING	223-52318 PIN	0.8755	0.8736	0.0019	0.0005
9	223-14319 FITTING	AN6-20A BOLT	0.379	0.371	0.008	0.002

PERMISSIBLE REAMING AND BUSHING LIMITS

ITEM NO.	PART DESCRIPTION	MAX ALLOW REAM DIA	BUSHING MATERIAL	BUSHING FIN. MIN ID	BUSHING FIN. MAX ID
*1	223-14242	0.3130	MIL-S-7720 STEEL	0.250	0.254
*2	223-14251	0.3130	MIL-S-7720 STEEL	0.250	0.254
†3	180-14312	0.6255	MIL-S-7720 STEEL	0.5625	0.5630
*4	180-14359	0.3130	MIL-S-7720 STEEL	0.250	0.251
†5	223-14316	0.6255	MIL-S-7720 STEEL	0.5625	0.5630
†6	180-14311	0.9380	MIL-S-7720 STEEL	0.8750	0.8755
†7	180-14310	0.9380	MIL-S-7720 STEEL	0.8750	0.8755
8	223-14317		SEE SHEET 2.		
‡9	223-14319	0.4380	MIL-S-7720 STEEL	0.375	0.379

*Interference fit 0.0003-0.0006
†Interference fit 0.0005-0.0010
‡Interference fit 0.0004-0.0008

NOTE Treat reamed part of hole with one coat of zinc-chromate primer, MIL-P-8585.

- Bushings to be installed in annealed condition while zinc-chromate primer is still wet.

VIEW A (SEE SHEET 2.)

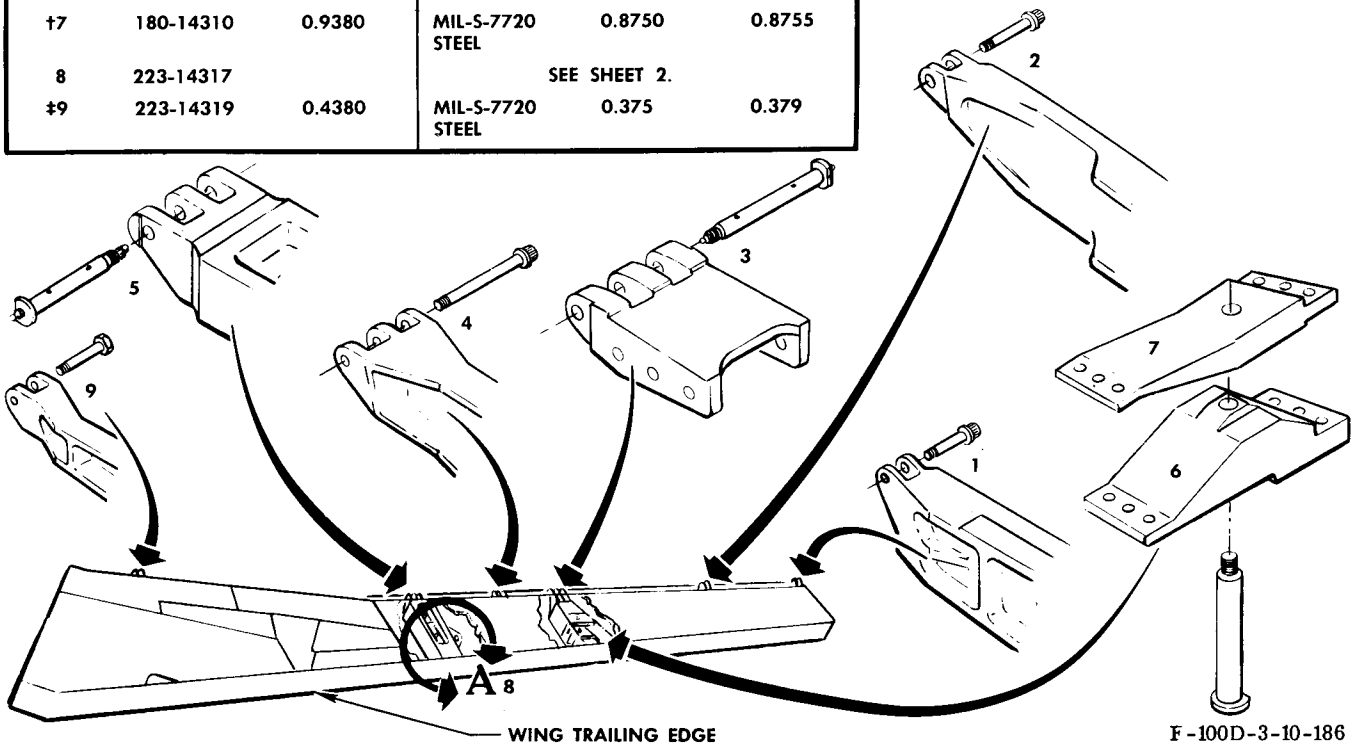
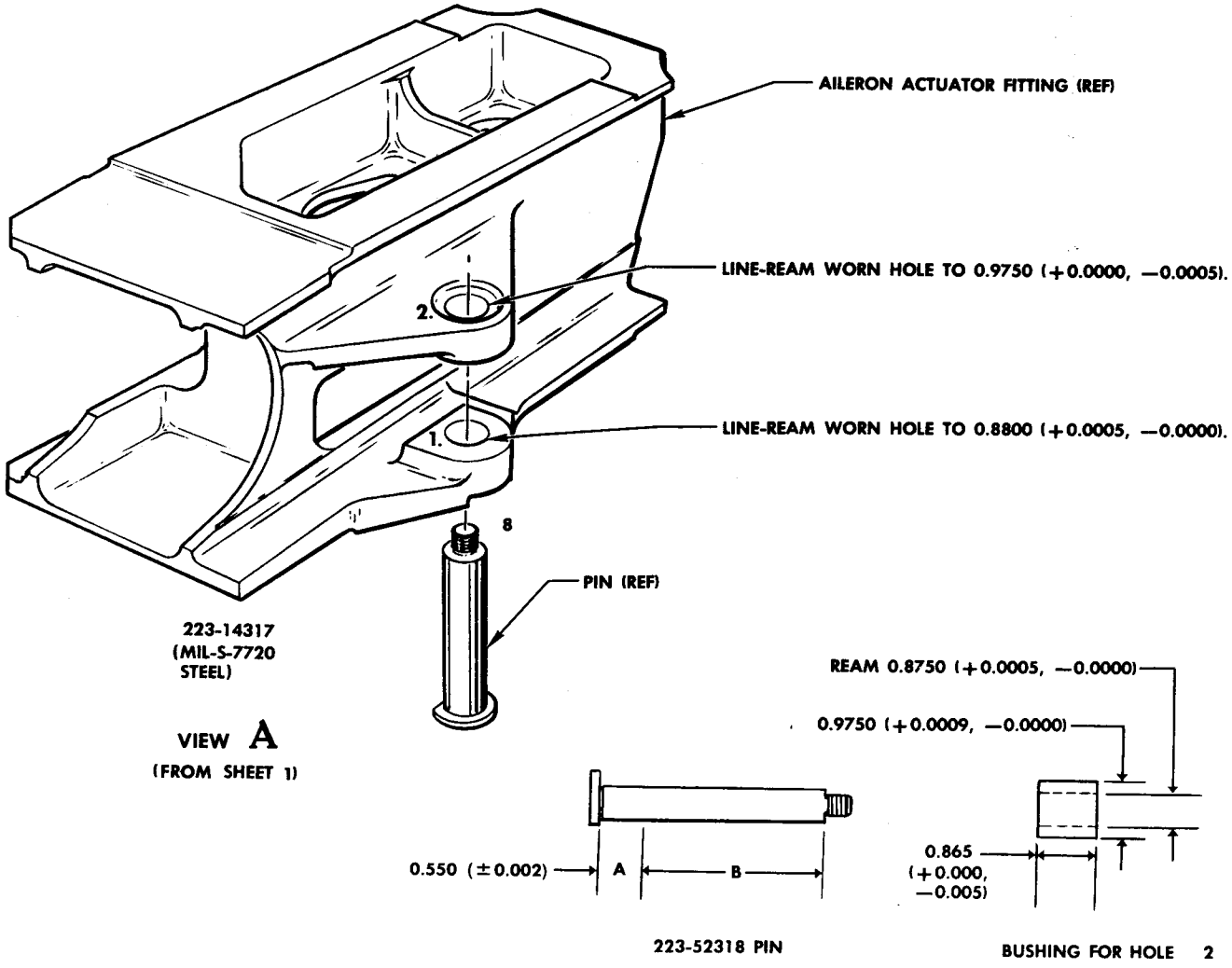


Figure 2-46B. Wear, Ream, and Bush Limits for Wing Trailing Edge Fittings (Sheet 1 of 2)

PERMANENT—CATEGORY A

WING



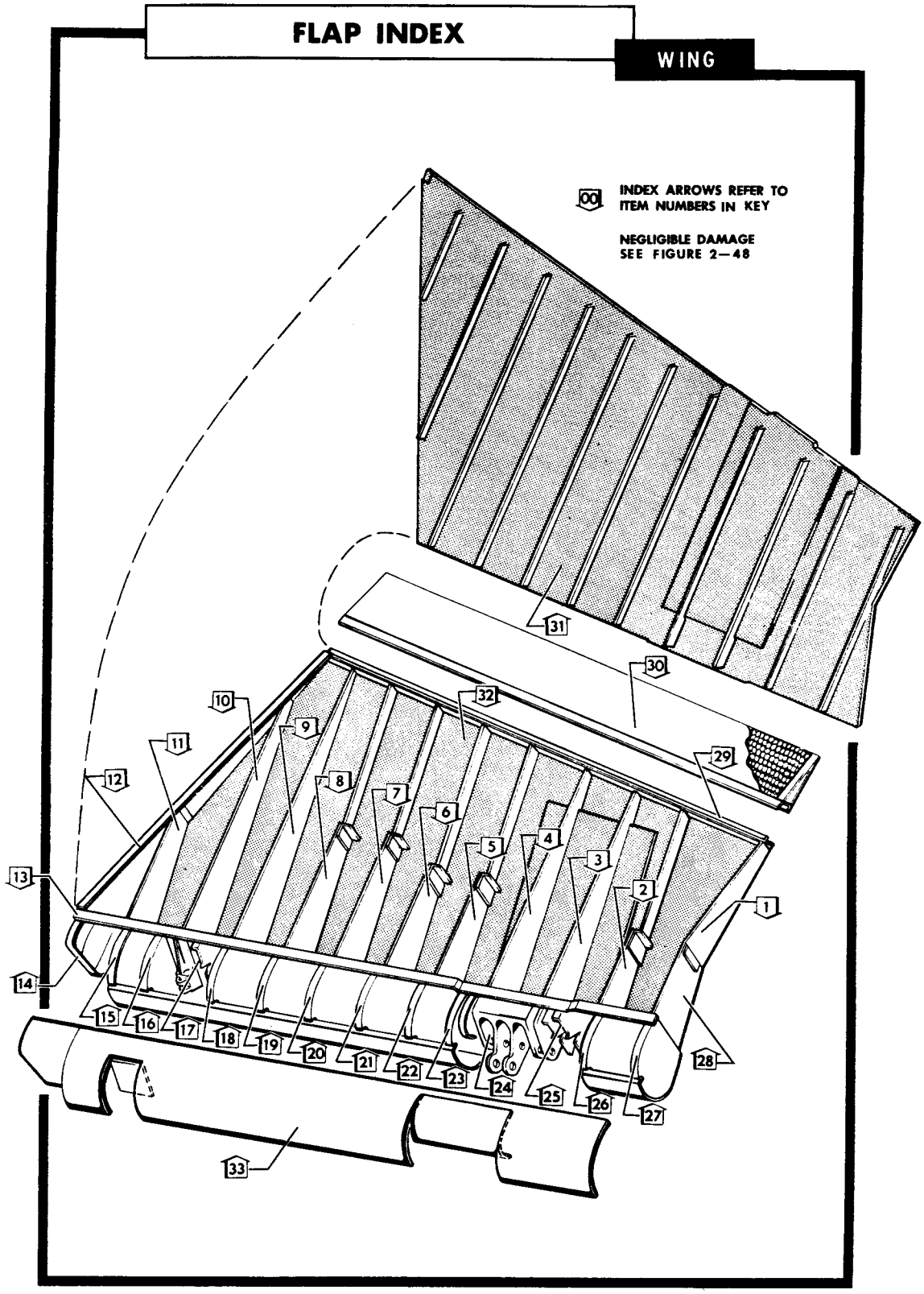
NOTE Line-ream worn hole or holes to diameters shown.

- Treat reamed holes with one coat zinc-chromate primer, MIL-P-8585.
- Strip chrome plate from pin shank and rechrome plate per MIL-P-6871, to allow for grinding.
- Grind Area A of pin to 0.8790 (+0.0000, —0.0005).
- Grind Area B of pin to 0.8741 (+0.0000, —0.0005).
- Fabricate bushing of steel, MIL-S-7720, to the dimensions shown.
- Install annealed bushing in hole 2, maintaining an 0.0005 to 0.0014 interference fit.

CONDITION AND TREATMENT FOR HOLE AND PIN					
			TREATMENT		
COND	HOLE 1	HOLE 2	HOLE 1	HOLE 2	SHAFT
1	IN LIMIT	OUT OF LIMIT	NONE	REAM AND BUSH	NONE
2	OUT OF LIMIT	OUT OF LIMIT	REAM	REAM AND BUSH	REPLATE AND GRIND
3	OUT OF LIMIT	IN LIMIT	REAM	NONE	REPLATE AND GRIND

100D-3-10-187A

Figure 2-46B. Wear, Ream, and Bush Limits for Wing Trailing Edge Fittings (Sheet 2 of 2)



F-100D-3-10-62

Figure 2-47. Wing Flap Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-18041	Rib	0.040	2024-T4 Clad	
2	223-18058	Web	0.064	2024-T4 Clad	
3	223-18056	Web	0.064	2024-T4 Clad	
4	223-18055	Web	0.064	2024-T4 Clad	
5	223-18054	Web	0.064	2024-T4 Clad	
6	223-18054	Web	0.064	2024-T4 Clad	
7	223-18053	Web	0.064	2024-T4 Clad	
8	223-18053	Web	0.064	2024-T4 Clad	
9	223-18052	Web	0.064	2024-T4 Clad	
10	223-18051	Web	0.064	2024-T4 Clad	
11	223-18051	Web	0.064	2024-T4 Clad	
12	223-18040	Rib	0.040	2024-T4 Clad	2-22
13	223-18003	Spar Assy	0.064	2024-T4 Clad	2-48B
14	223-18025	Rib	0.040	2024-T4 Clad	
15	223-18026	Rib Assy	0.051	2024-T4 Clad	
16	223-18019	Fitting		2024-T6 Die Forging	
17	223-18028	Rib	0.040	2024-T4 Clad	
18	223-18029	Rib	0.040	2024-T4 Clad	
19	223-18030	Rib	0.040	2024-T4 Clad	
20	223-18031	Rib	0.040	2024-T4 Clad	
21	223-18032	Rib	0.040	2024-T4 Clad	
22	223-18033	Rib	0.040	2024-T4 Clad	
23	223-18034	Rib	0.040	2024-T4 Clad	
24	223-18045	Fitting		4340 Stl Die Forging	
25	223-18035	Rib	0.040	2024-T4 Clad	
26	223-18036	Rib	0.040	2024-T4 Clad	
27	223-18037	Rib	0.040	2024-T4 Clad	
28	223-18057	Web	0.064	2024-T4 Clad	
29	223-18005	Spar	0.051	2024-T4 Clad	
30	223-18006-1	TE		Honeycomb Const	
31	223-18010-3	Skin	0.040	2024-T4 Clad	2-49, 2-50, 2-51, 2-51A
32	223-18010-5	Skin	0.040	2024-T4 Clad	2-24, 2-50, 2-51, 2-51A, 2-73
33	223-18002	Skin	0.032	2024-T4 Clad	

***Drawing numbers are for reference only and are not to be used for ordering parts.**

†For additional repairs, refer to Section X.

Key to Figure 2-47

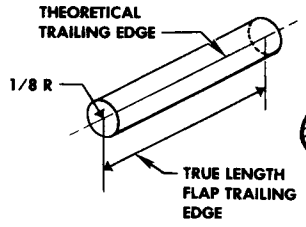
FLAP NEGLIGIBLE DAMAGE

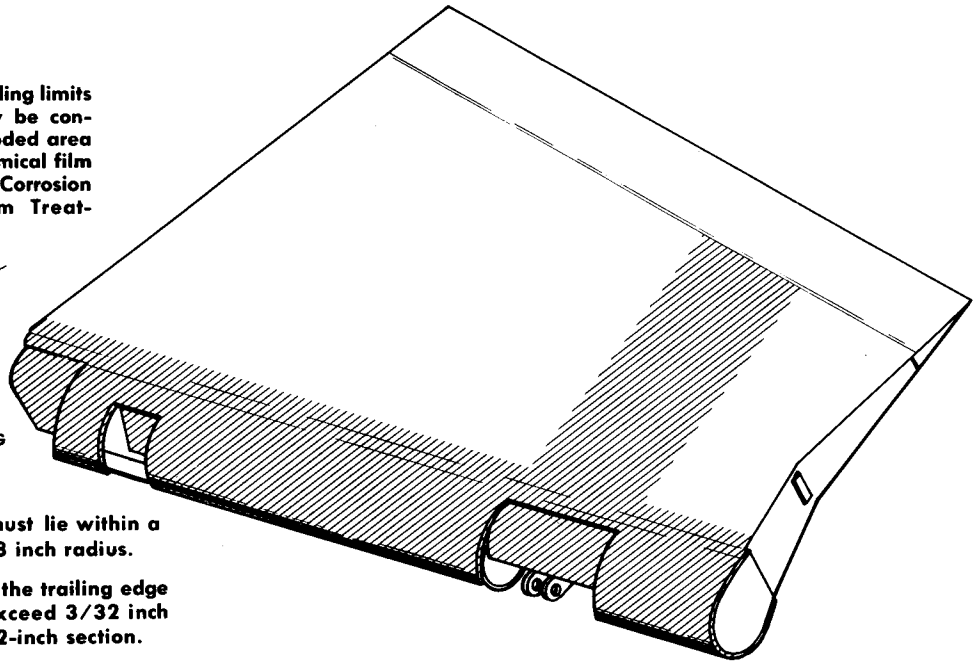
WING

SKIN DAMAGE IN THIS AREA IS CONSIDERED MAJOR DAMAGE.

CORROSION

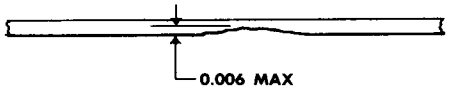
Corrosion damage not exceeding limits for scratches or cracks may be considered negligible. The corroded area must be cleaned and the chemical film treated. (Refer to index for "Corrosion Rework by Chemical Film Treatments.")



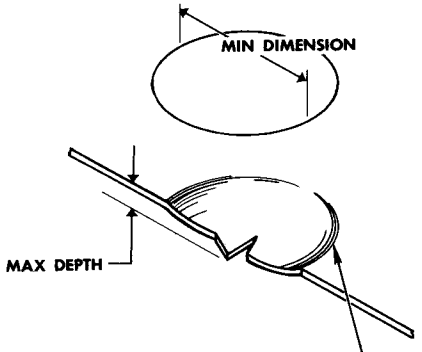


NOTE The trailing edge must lie within a theoretical cylinder with 1/8 inch radius.

- The maximum deviation of the trailing edge in any direction must not exceed 3/32 inch from a straight line in any 12-inch section.

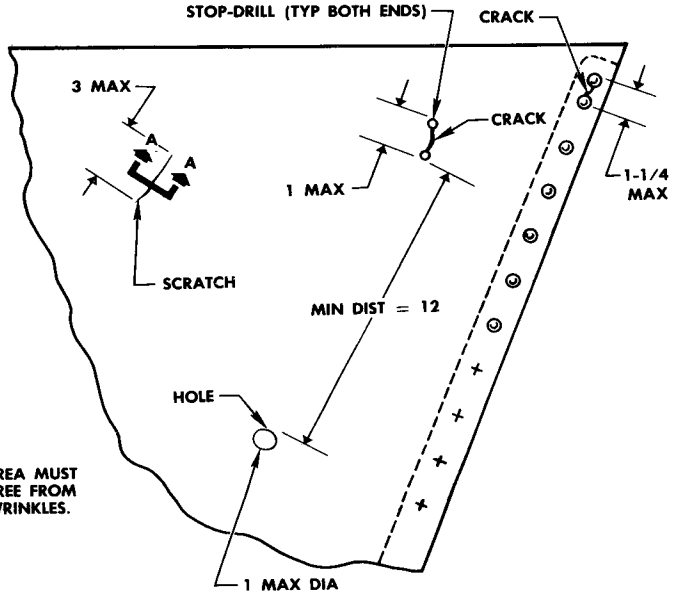


SECTION A-A



MIN DIMENSION	1 IN.	2 IN.	4 IN. AND OVER
MAX DEPTH	1/32	1/16	3/32

NOTE Dents which are not more than the dimensions given in the table are considered negligible.



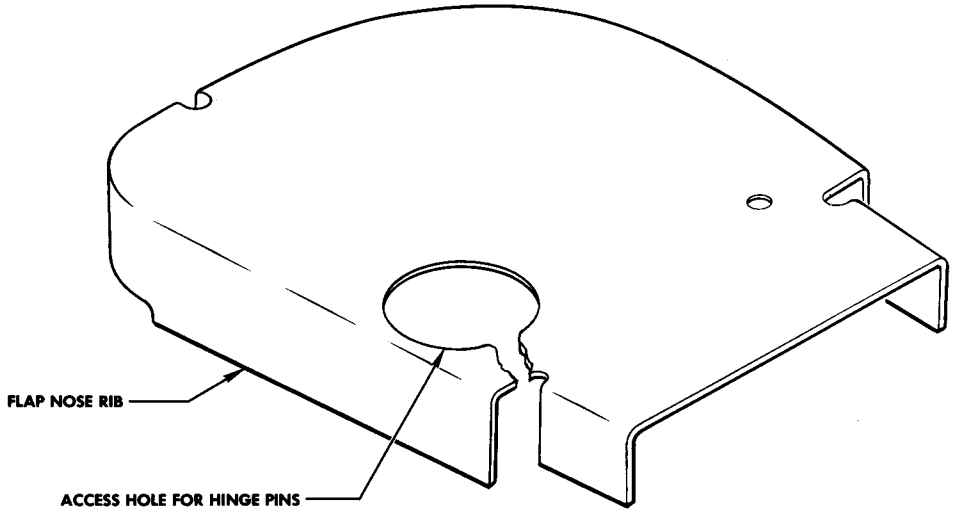
NOTE All surface holes should be plugged. For minor hole repairs, refer to index.

F-100D-3-10-130C

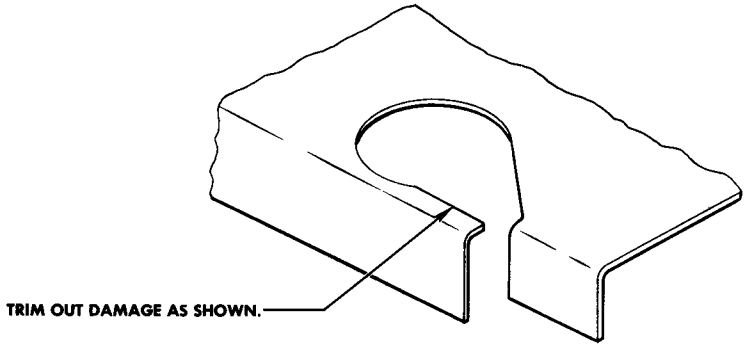
Figure 2-48. Flap Negligible Damage

PERMANENT CATEGORY A

WING

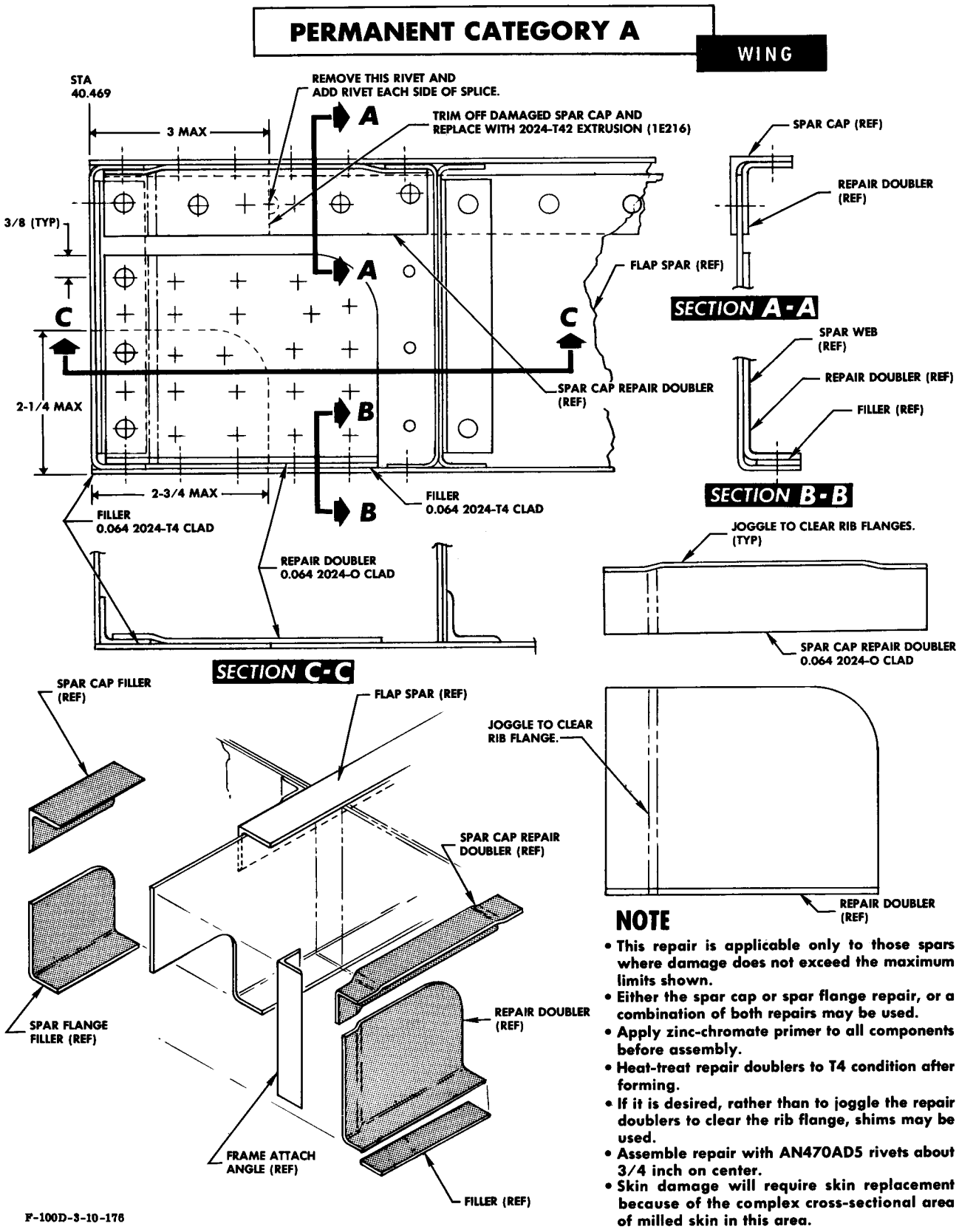


The narrow portion of the flap rib between the access hole for the outer hinge pin and the relief cutout for the rear spar is very easily damaged when the flap is removed or installed. If this type of damage occurs and damage does not extend into or between attachment holes of rib flange, repair rib by trimming out damaged area as shown. If rib is damaged above limits shown, replace rib with a serviceable part.



F-100D-3-10-165

Figure 2-48A. Flap Rib—Hinge Access Hole



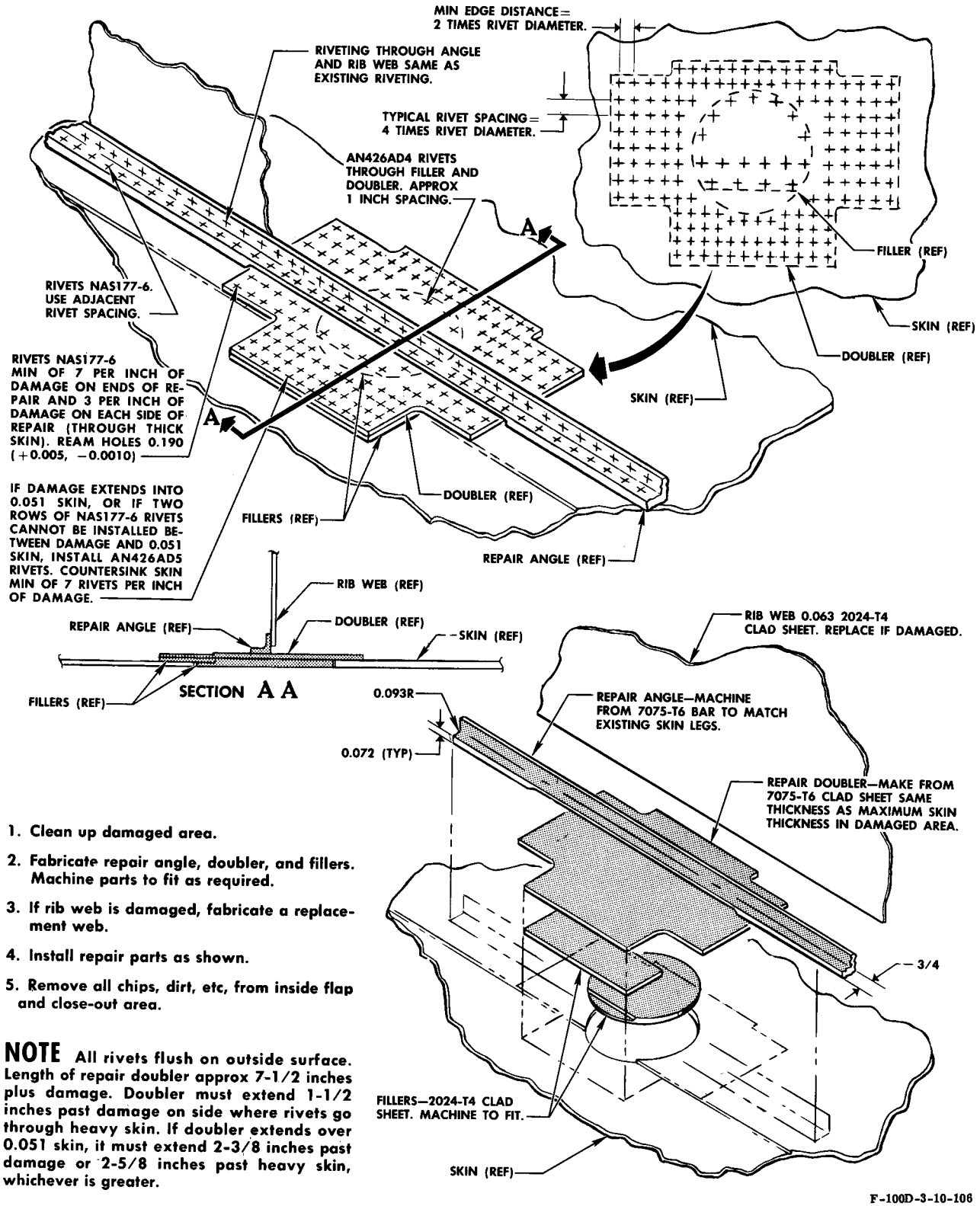
NOTE

- This repair is applicable only to those spars where damage does not exceed the maximum limits shown.
- Either the spar cap or spar flange repair, or a combination of both repairs may be used.
- Apply zinc-chromate primer to all components before assembly.
- Heat-treat repair doublers to T4 condition after forming.
- If it is desired, rather than to joggle the repair doublers to clear the rib flange, shims may be used.
- Assemble repair with AN470AD5 rivets about 3/4 inch on center.
- Skin damage will require skin replacement because of the complex cross-sectional area of milled skin in this area.

Figure 2-48B. Flap Spar Repair

PERMANENT—CATEGORY A

WING



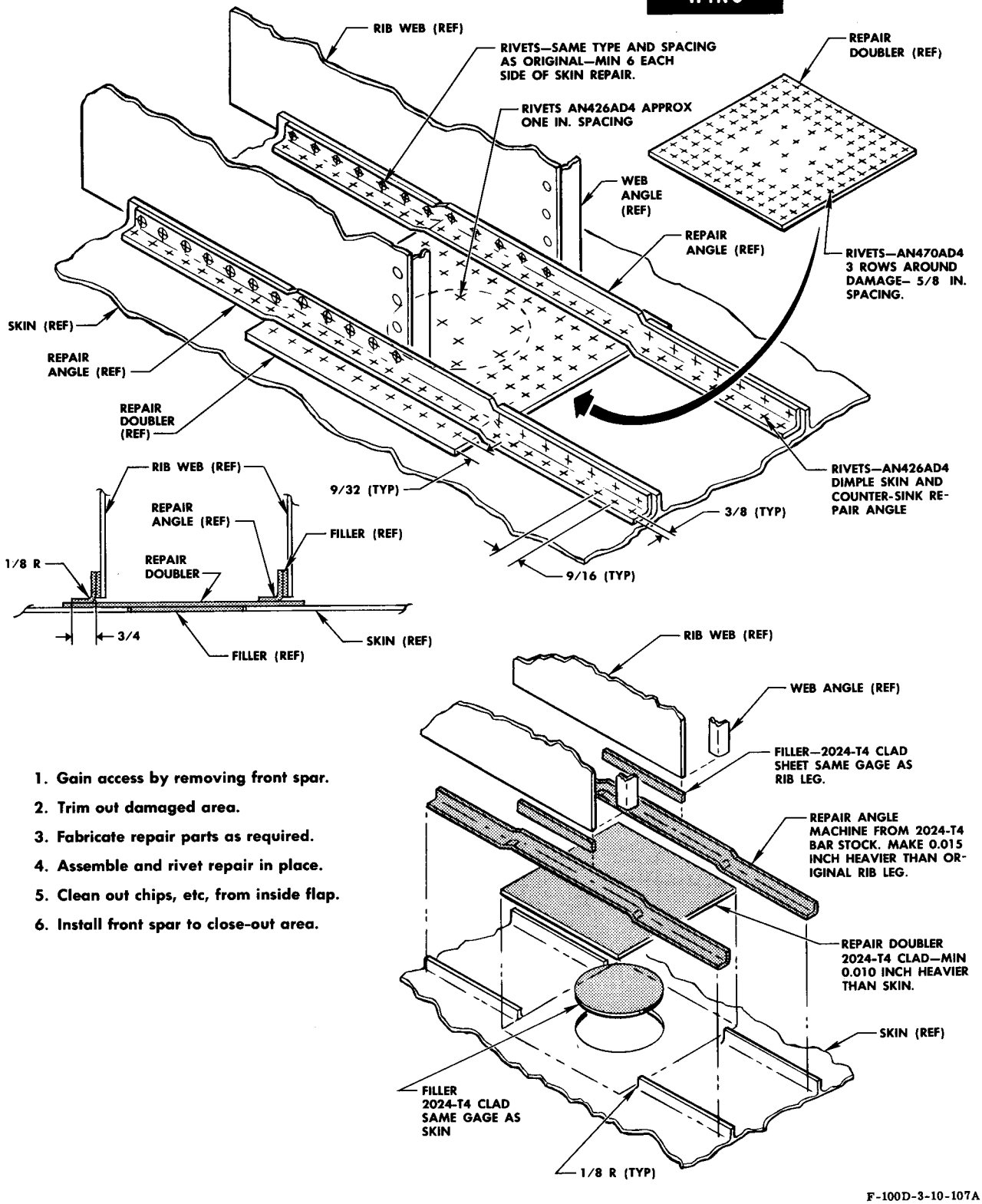
1. Clean up damaged area.
2. Fabricate repair angle, doubler, and fillers. Machine parts to fit as required.
3. If rib web is damaged, fabricate a replacement web.
4. Install repair parts as shown.
5. Remove all chips, dirt, etc, from inside flap and close-out area.

NOTE All rivets flush on outside surface. Length of repair doubler approx 7-1/2 inches plus damage. Doubler must extend 1-1/2 inches past damage on side where rivets go through heavy skin. If doubler extends over 0.051 skin, it must extend 2-3/8 inches past damage or 2-5/8 inches past heavy skin, whichever is greater.

Figure 2-49. Flap Skin and Actuator Ribs—Station 86.43 and 92.295

PERMANENT—CATEGORY B

WING

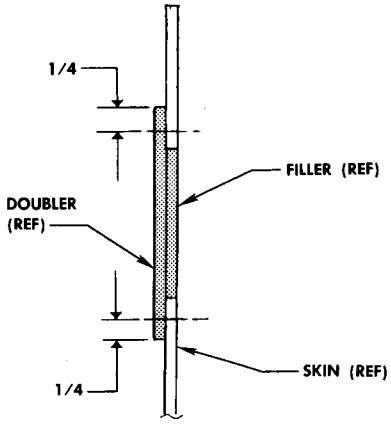


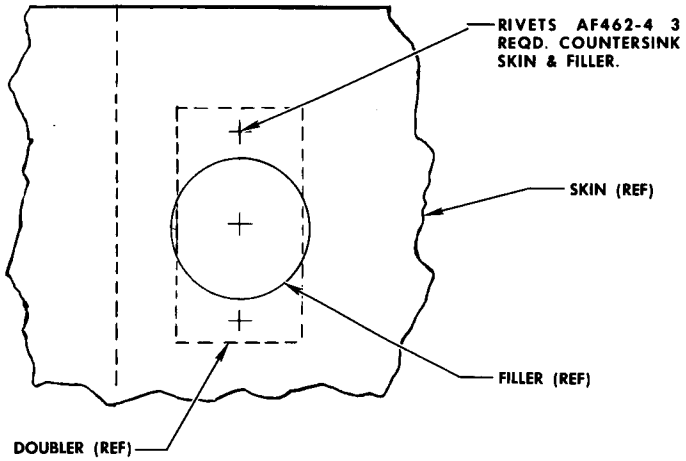
1. Gain access by removing front spar.
2. Trim out damaged area.
3. Fabricate repair parts as required.
4. Assemble and rivet repair in place.
5. Clean out chips, etc, from inside flap.
6. Install front spar to close-out area.

Figure 2-50. Flap Skin and Ribs

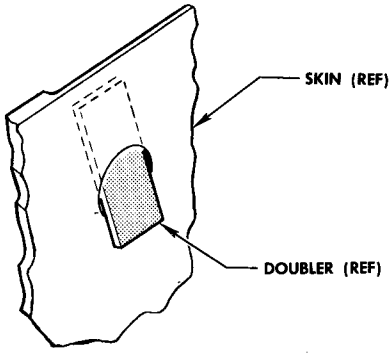
PERMANENT—CATEGORY A

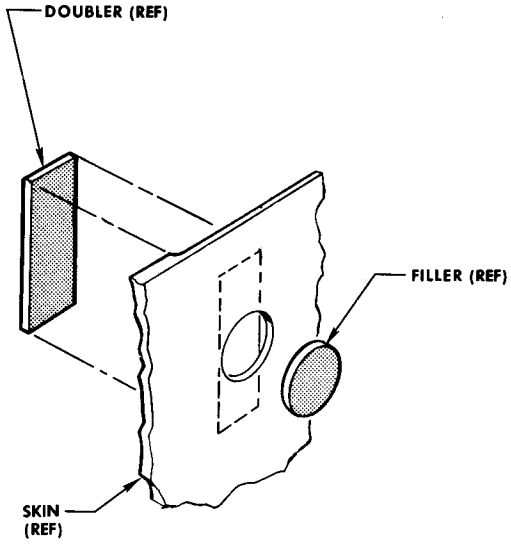
WING





1. Trim out damaged area.
2. Make doubler, using 2024-T4 clad material, 0.010-inch heavier gage than damaged skin. The length should be equal to the hole diameter plus one inch; the width should be equal to the hole diameter minus 1/32 inch. Make filler the same gage as skin, using 2024-T4 clad material.
3. Rivet doubler in place using center hole for positioning. Hold while riveting.
4. Rivet filler to doubler.





NOTE This repair applicable only when cleaned-out hole does not touch ribs, stiffeners, or heavy skin lands, and does not exceed 1.0 inch diameter.

Min distance between repairs 12 inches.
Max allowable repairs three per panel between ribs or stiffeners.

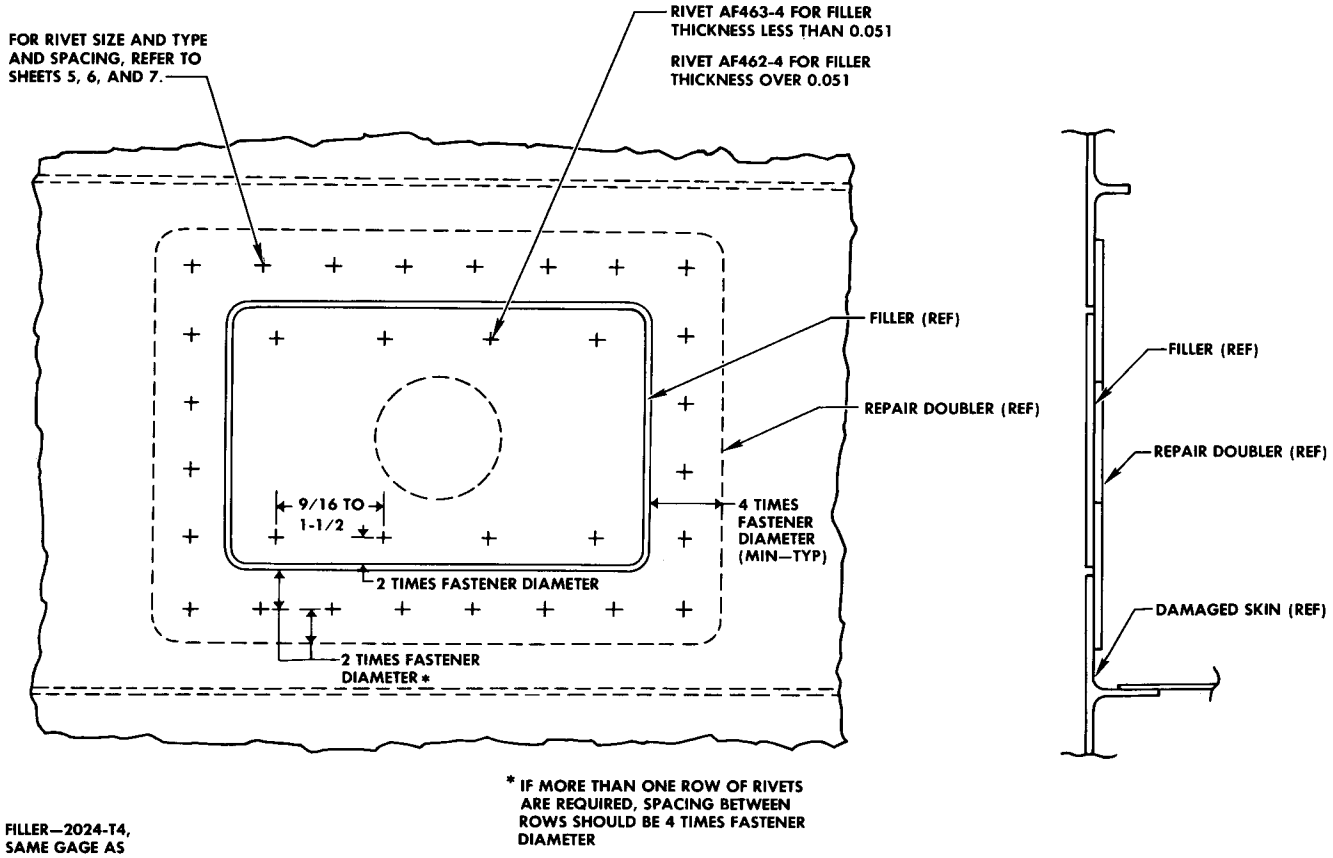
F-100D-3-10-110A

Figure 2-51. Flap and Aileron Skins—Blind Repair

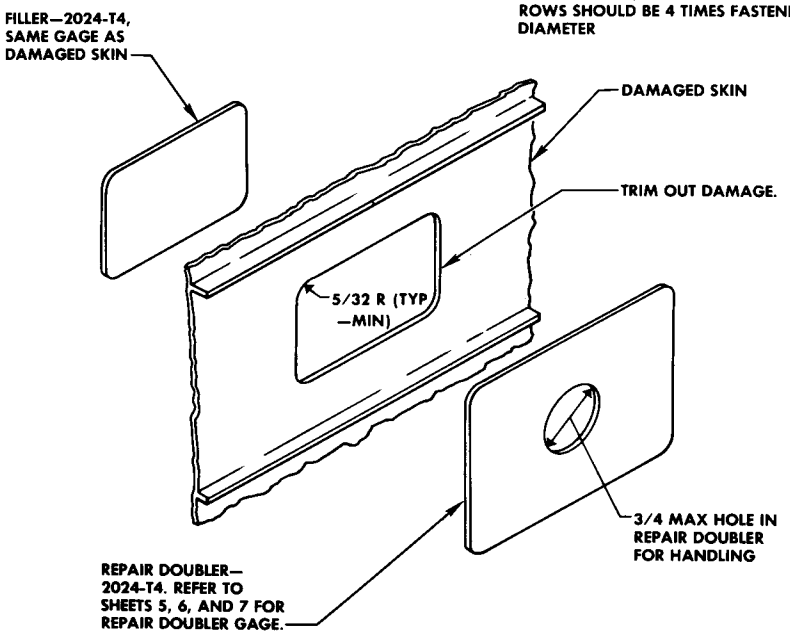
PERMANENT CATEGORY A

WING

AILERON AND FLAP



* IF MORE THAN ONE ROW OF RIVETS ARE REQUIRED, SPACING BETWEEN ROWS SHOULD BE 4 TIMES FASTENER DIAMETER



Do not use this repair if damage occurs in skin thickness greater than 0.072 inch.

Before repairing aileron, determine if it can be balanced with additional weight of repair parts. (Refer to "Flight Control Surface Static Balance" in Section I.)

1. Trim out damage.
2. Smooth all edges and corners.
3. Make repair doubler and filler. Refer to sheets 5, 6, and 7 for repair doubler gage. The rivet pattern determines the size and shape.
4. Fasten all repair parts in place, using size and type of rivets and spacing called out in sheets 5, 6, and 7.
5. Rebalance aileron.

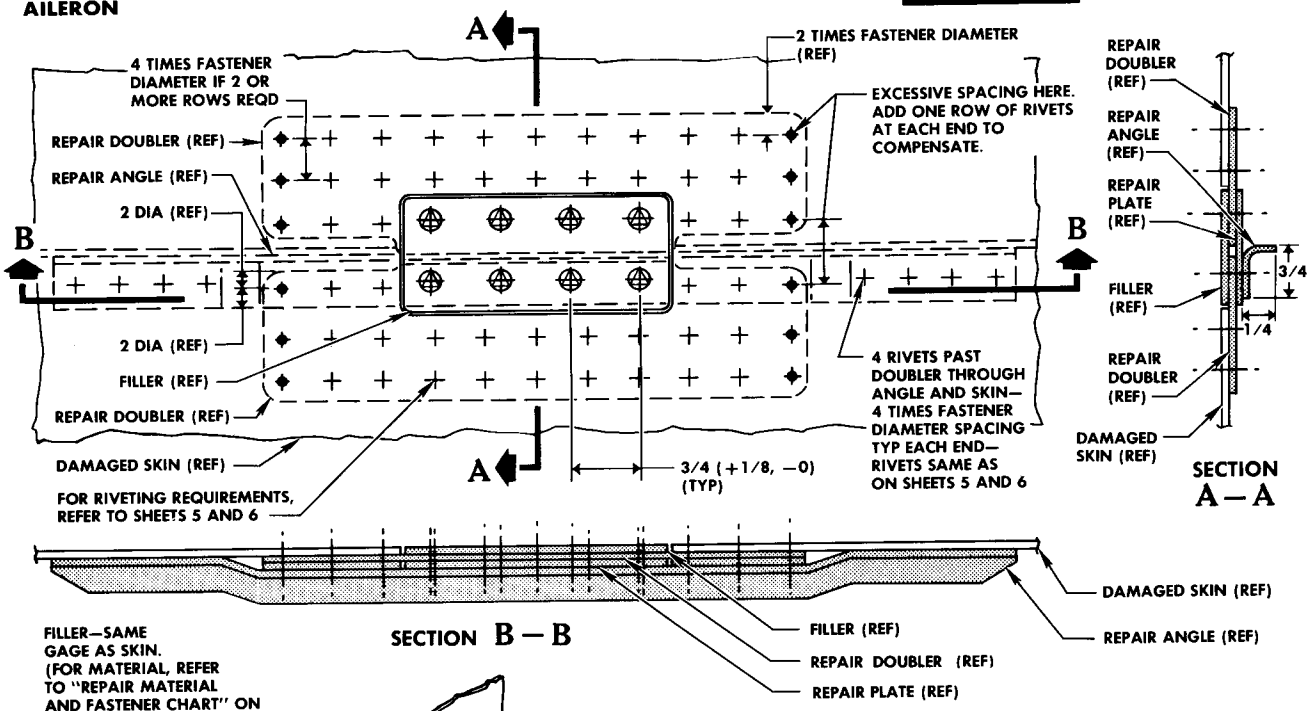
F-100D-3-10-144

Figure 2-51A. Milled Skins — Flap and Aileron — Blind (Sheet 1 of 7)

PERMANENT—CATEGORY A

WING

AILERON



Before repairing an aileron, determine if it can be balanced with the additional weight of the repair parts. (Refer to "Flight Control Surface Static Balance" in Section I.)

1. Trim out damage.
2. Smooth all edges and corners.
3. Make repair doublers, repair plate, repair angle, shims, and filler. Refer to "Repair Material and Fastener Chart" on this page for necessary information. The rivet pattern determines the doubler size. The rivet edge distance equals two times the fastener diameter.
4. Fasten repair parts in place. Refer to sheets 5 and 6 for number of rivets required. (The shims may be tack-riveted to the repair angle for ease of fabrication.)
5. Rebalance aileron.

REPAIR MATERIAL AND FASTENER CHART

	OUTBOARD AILERON PANELS 2, 3, 7, 8, 9, AND 10			ALL INBOARD AILERON PANELS AND OUTBOARD PANELS 1, 4, 5, 6, AND 11 THRU 22		
	MATERIAL	GAGE (IN.)	FASTENERS	MATERIAL	GAGE (IN.)	FASTENERS
REPAIR DOUBLERS	301 CRES 1/2 H	0.050	REFER TO SHEETS 5 AND 6	2024-T4	0.071	REFER TO SHEETS 5 AND 6
REPAIR PLATE	301 CRES 1/2 H	0.032	CR563-6 ⊕	2024-T4	0.040	AF463-6 ⊕
REPAIR ANGLE	2024-T4	0.050	REFER TO SHEETS 5 AND 6	2024-T4	0.050	REFER TO SHEETS 5 AND 6
FILLER	301 CRES 1/2 H	0.032 MIN OR SKIN T, WHICHEVER IS GREATER	CR563-6 ⊕	2024-T4	0.040	AF463-6 ⊕

NOTE If damage extends into two panels, select material thickness and riveting requirements from the panel requiring the most rivets per inch.

- Do not countersink skin if material thickness is less than 0.050 inch.

F-100D-3-10-145

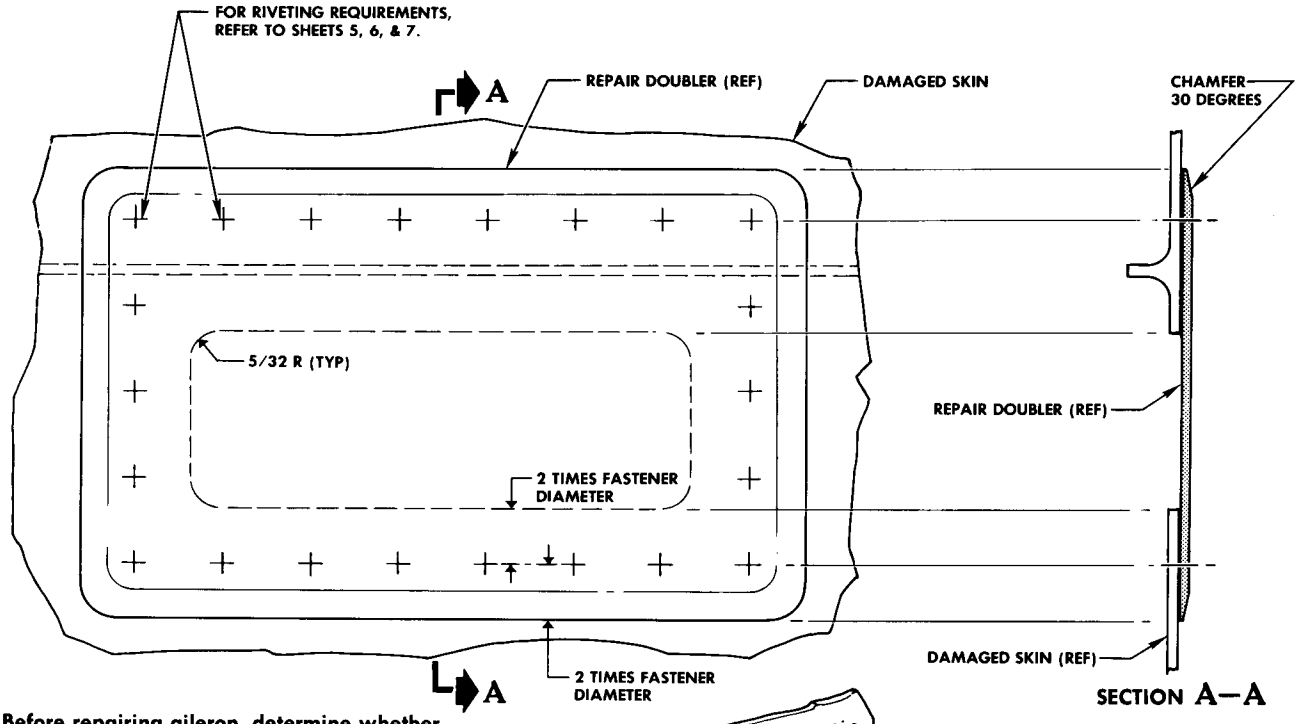
Figure 2-51A. Milled Skins — Flap and Aileron — Blind (Sheet 2 of 7)

TEMPORARY—CATEGORY A

WING

AILERON AND FLAP

NOTE Do not use this repair if damage occurs in skin thickness greater than 0.072 inch.

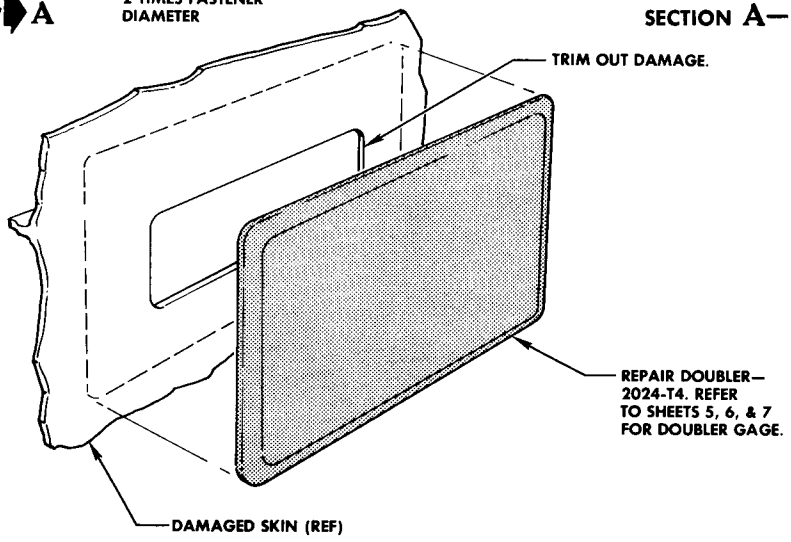


- Before repairing aileron, determine whether it can be balanced with additional weight of repair parts. (Refer to "Flight Control Surface Static Balance" in Section I.)

1. Trim out damage.
2. Make repair doubler, and chamfer 30 degrees. The rivet pattern determines the size and shape.
3. Fasten repair doubler in place, using size and type of fastener and spacing shown on sheets 5, 6, and 7.
4. Rebalance aileron.

NOTE If damage occurs in two panels, select doubler gage and riveting requirements from panel requiring most rivets per inch.

- If more than one row of fasteners is required, the spacing between the rows should be four times the rivet diameter.



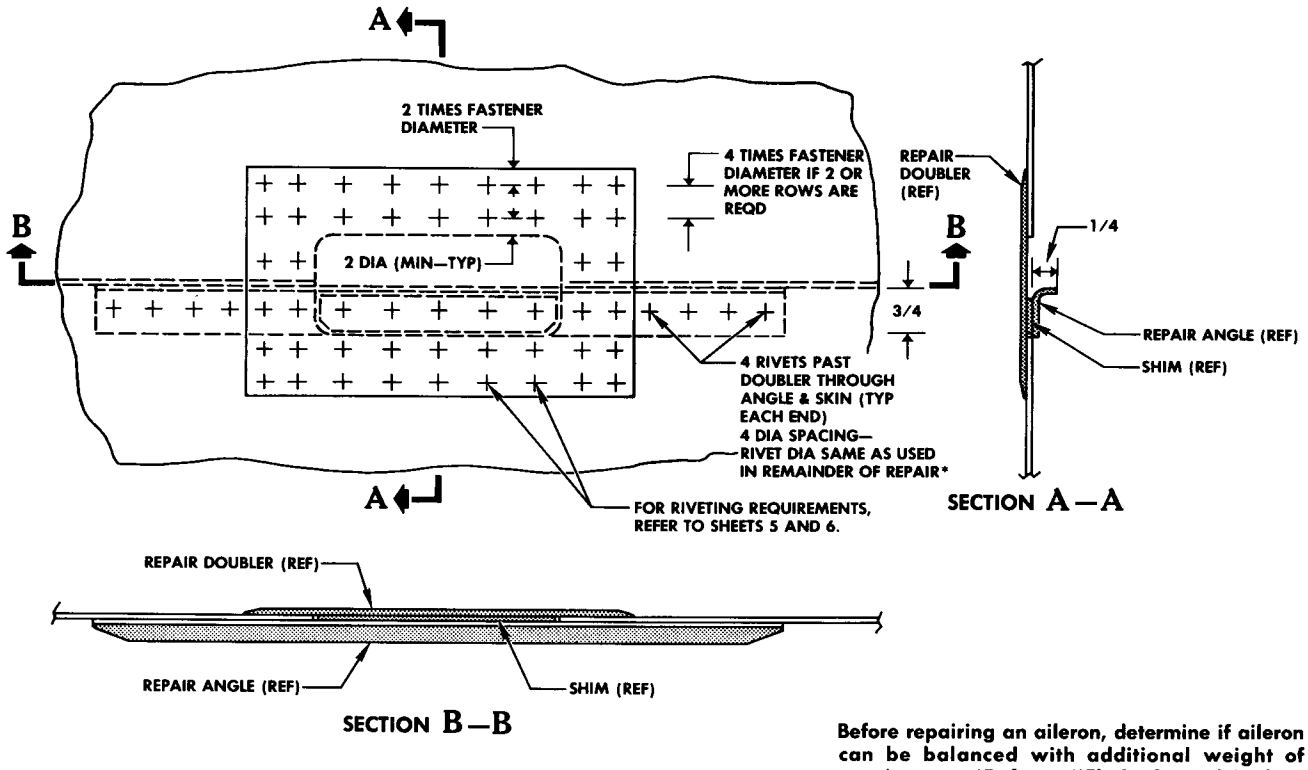
F-100D-3-10-146

Figure 2-51A. Milled Skins — Flap and Aileron — Blind (Sheet 3 of 7)

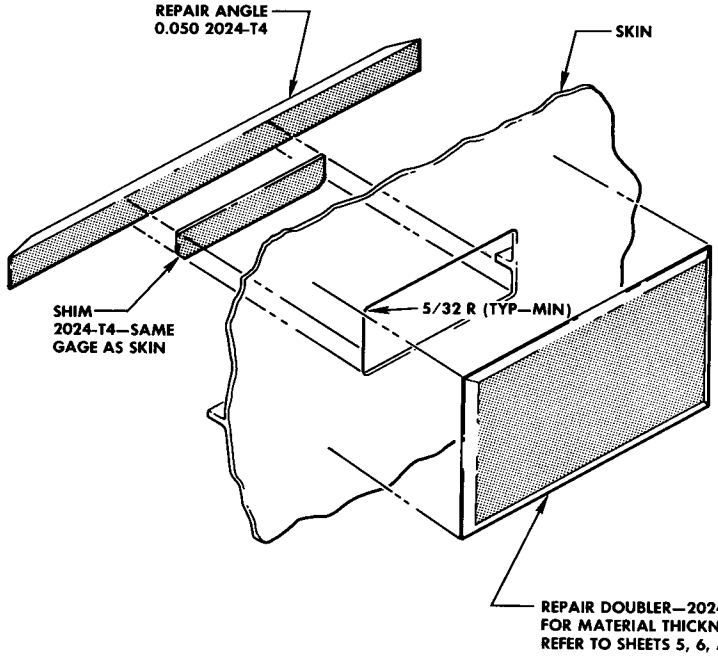
TEMPORARY—CATEGORY A

WING

AILERONS



SECTION B—B



Before repairing an aileron, determine if aileron can be balanced with additional weight of repair parts. (Refer to "Flight Control Surface Static Balance" in Section I.)

1 Trim out damage.

2 Smooth all edges and corners.

3 Make repair doubler, shim, and repair angle. Use repair tables in sheets 5, 6, and 7 to determine material thickness for repair doubler. The rivet pattern determines the size and shape.

4 Fasten repair parts in place. Refer to sheets 5 and 6 for number of rivets required.

5 Rebalance aileron.

NOTE If damage extends into two panels, select doubler thickness and riveting requirements from panel requiring most rivets per inch.

* Do not countersink skin if material thickness is less than 0.050 inch.

F-100D-3-10-147

Figure 2-51A. Milled Skins — Flaps and Aileron — Blind (Sheet 4 of 7)

PERMANENT AND TEMPORARY— CATEGORY A

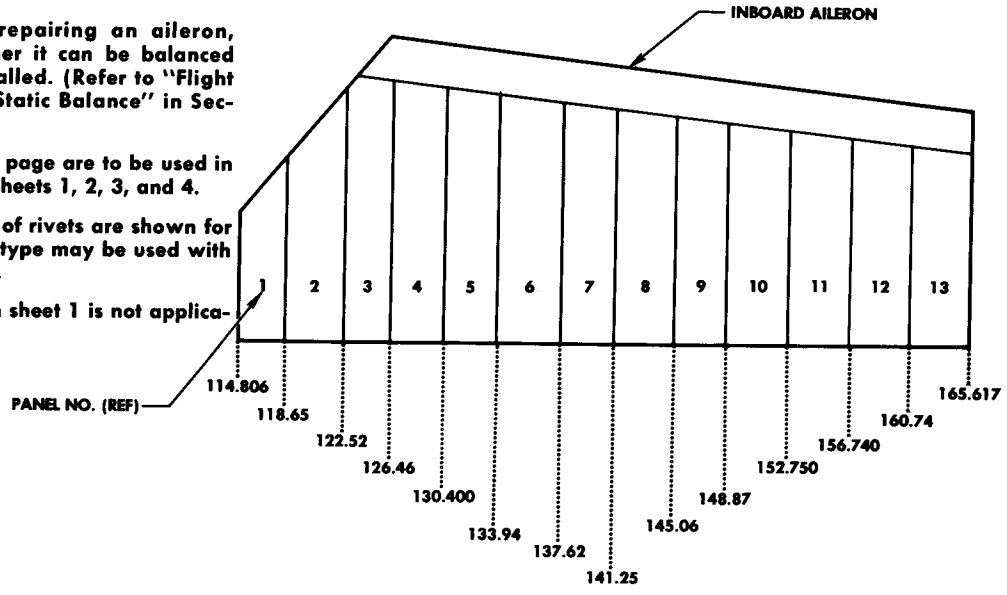
WING

AILERON

NOTE Before repairing an aileron, determine whether it can be balanced with repairs installed. (Refer to "Flight Control Surface Static Balance" in Section I.)

- The charts on this page are to be used in conjunction with sheets 1, 2, 3, and 4.
- Where two types of rivets are shown for one panel, either type may be used with the noted spacing.

*Repair shown on sheet 1 is not applicable to this panel.



AIL STA (APPROX UP. & LWR STIFFENER LOCATION)

EXTERNAL DOUBLER				
PANEL NO.	DOUBLER (†) (IN.)	RIVET REQUIREMENTS		
		TYPE	ROWS	SPACING (IN.)
1	0.040	AF463-4	1	1
2	0.040		1	1
3	0.040		1	1
4	0.040		1	1/2
*5	0.050	AF463-4	2	5/8
6	0.050	CR179-4 AF463-5	1	15/32 17/32
7	0.050	CR179-4 AF463-5	1	1/2 9/16
8	0.050	AF463-4	1	1/2
9	0.050	CR179-4	1	1/2
10	0.040	CR179-4	1	7/16
11	0.040	CR179-4	1	1/2
12	0.040	AF463-4	1	7/8
13	0.040	AF463-4	1	1

INTERNAL DOUBLER				
PANEL NO.	DOUBLER (†) (IN.)	RIVET REQUIREMENTS		
		TYPE	ROWS	SPACING (IN.)
1	0.050	AF462-4	1	1
2	0.050		1	1
3	0.050		1	1
4	0.050		1	1/2
5	0.050		2	19/32
6	0.050	AF462-4	2	11/16
7	0.063	CR178-4 AF462-5	1 1	1/2 9/16
8	0.063	AF462-4	2	29/32
9	0.050	AF462-4	2	23/32
10	0.050	CR178-4	1	7/16
11	0.050	CR178-4	1	1/2
12	0.050	AF462-4	1	7/8
13	0.050	AF462-4	1	1

F-100D-3-10-148

Figure 2-51A. Milled Skins — Flaps and Aileron — Blind (Sheet 5 of 7)

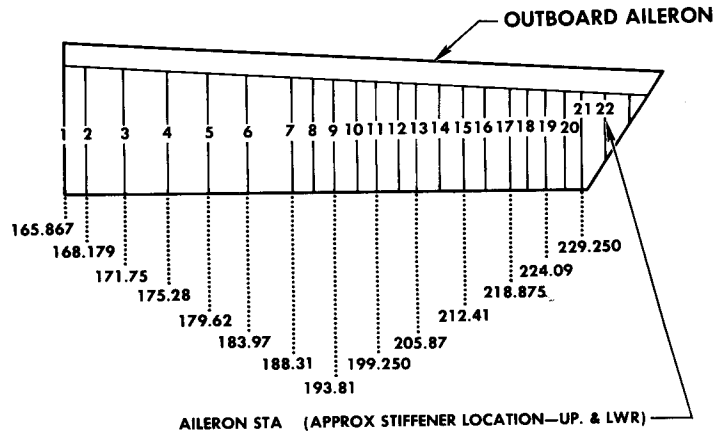
PERMANENT AND TEMPORARY— CATEGORY A

WING

AILERON

NOTE The charts on this page are to be used in conjunction with sheets 1, 2, 3, and 4.

Where two types of rivets are shown for one panel, either type may be used with the noted spacing.



INTERNAL DOUBLER

PANEL NO.	DOUBLER THICKNESS	RIVET REQUIREMENTS TYPE	ROWS	SPACING (IN.)
1	INTERNAL REPAIRS NOT APPLICABLE IN THIS PANEL			
*2	0.063	CR179-4 AF463-5	2 2	21/32 5/8
3	0.063	AF463-4	2	19/32
4	0.071	CR178-4 AF463-5	2 2	1/2 3/4
†5	0.071	AF462-4 AF463-4	3 2	17/32 1/2
†6	0.063	AF462-4 AF463-4	3 2	9/16 17/32
*7	0.050	AF463-4	2	9/16
*8	↕	↕	↕	↕
*9				
*10				
*11				
*12				
*13				
*14	0.050			9/16
*15	0.040			1/2
*16	↕			↕
*17				
*18			2	1/2
*19			2	13/16
*20			2	13/16
21			1	11/16
22	0.040	AF463-4	1	1

*Repairs shown on sheet 1 are not applicable to these panels.

EXTERNAL DOUBLER

PANEL NO.	DOUBLER THICKNESS	RIVETS REQUIREMENTS TYPE	ROWS	SPACING (IN.)
1	0.063	AF462-4	1	1
2	0.063	CR178-4 AF462-5	2 2	21/32 5/8
3	0.063	AF462-4	2	1/2
4	0.071	↕	2	1/2
†5	0.071		2	1/2
†6	0.063		2	7/16
7	↕		2	17/32
8			2	17/32
9	↕		2	9/16
10	0.063		2	9/16
11	0.050		2	1/2
12	↕		↕	↕
13				
14				
15				
16				
17			↕	↕
18			2	1/2
19			2	13/16
20			2	13/16
21	↕	↕	1	11/16
22	0.050	AF462-4	1	1

†Repairs shown on sheets 2 and 4 are not applicable to these panels.

F-100D-3-10-149

Figure 2-51A. Milled Skins — Flaps and Aileron — Blind (Sheet 6 of 7)

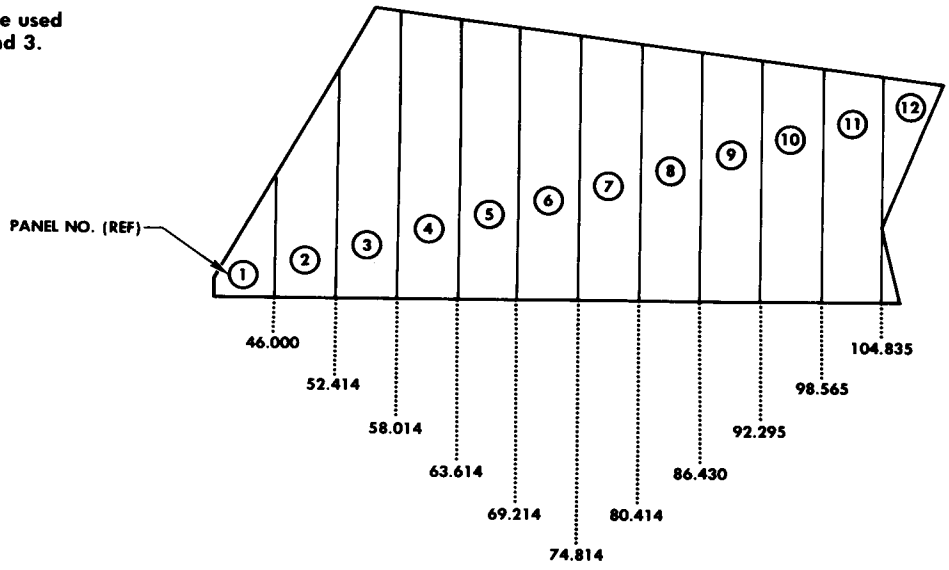
PERMANENT AND TEMPORARY— CATEGORY A

WING

FLAPS

NOTE

The charts on this page are to be used in conjunction with sheets 1 and 3.



FLAP STATION (STIFFENER LOCATIONS APPROX)

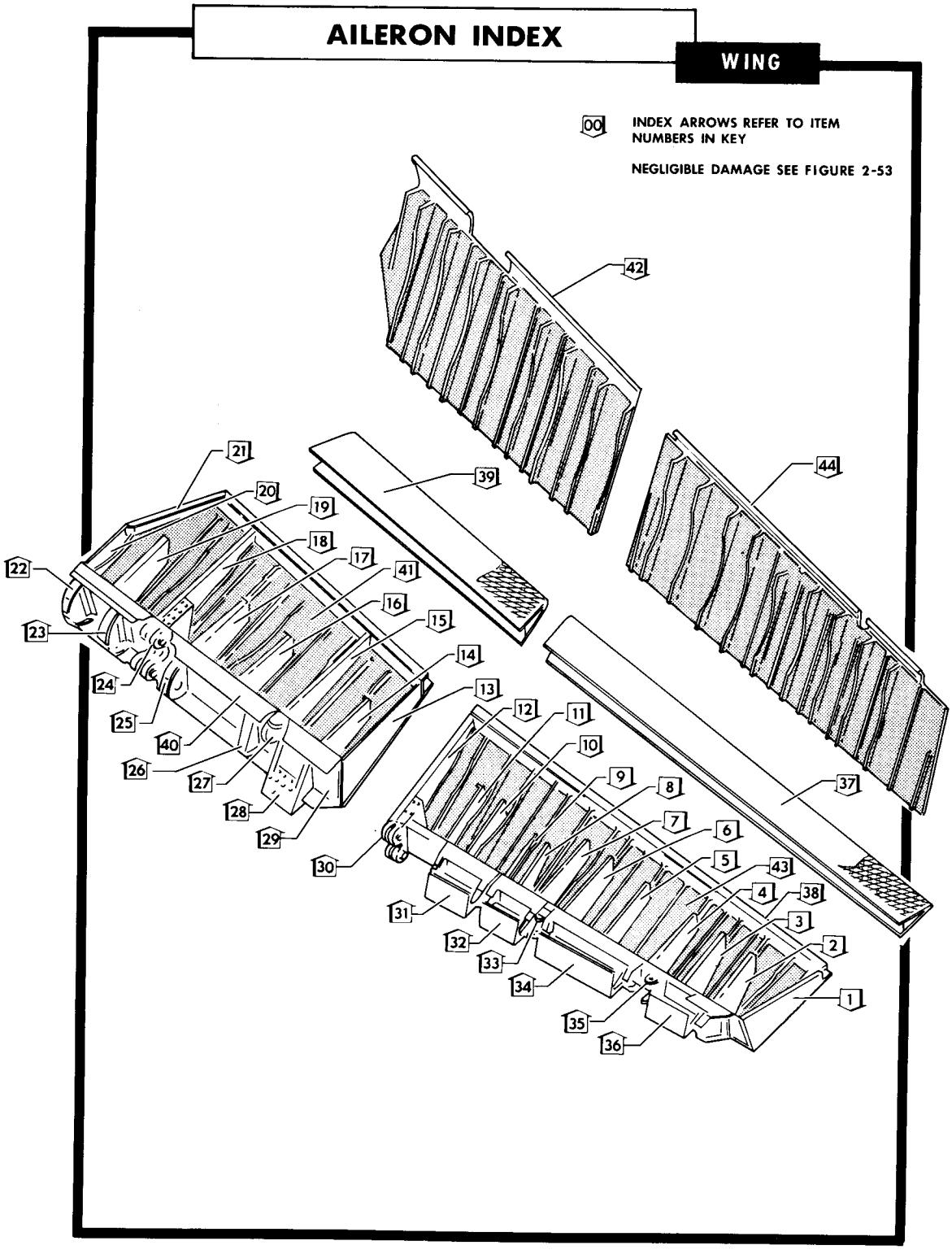
INTERNAL DOUBLER				
PANEL NO.	DOUBLER THICKNESS (t) (IN.)	RIVET REQUIREMENTS		
		TYPE	ROWS	SPACING (IN.)
1	0.050	AF463-4	1	1
2	0.050	AF463-4	1	1
*3	0.050	AF463-4	1	1
*4	0.050	AF463-4	1	23/32
5	0.050	AF463-4	1	17/32
6	0.050	AF463-4	1	9/16
7	0.050	AF463-4	1	23/32
8	0.063	AF463-4	1	21/32
9	0.063	AF462-4	1	1
10	0.063	AF463-4	1	27/32
11	0.050	AF463-4	1	1
12	0.050	AF462-4	1	1

EXTERNAL DOUBLER				
PANEL NO.	DOUBLER THICKNESS (t) (IN.)	RIVET REQUIREMENTS		
		TYPE	ROWS	SPACING (IN.)
1	0.050	AF462-4	1	1
2	0.050	AF462-4	1	1
3	0.050	AF462-4	1	1
4	0.050	AF463-4	1	23/32
5	0.050	AF463-4	1	17/32
6	0.050	AF463-4	1	9/16
7	0.050	AF462-4	1	19/32
8	0.063	AF462-4	1	9/16
9	0.063	AF462-4	1	1
10	0.063	AF463-4	1	21/32
11	0.050	AF462-4	1	1
12	0.050	AF462-4	1	1

* DO NOT ATTEMPT A BLIND REPAIR WITH AN INTERNAL DOUBLER ON PANELS NO. 3 AND 4 OF THE UPPER SKIN.

F-100D-3-10-150

Figure 2-51A. Milled Skins—Flaps and Aileron — Blind (Sheet 7 of 7)



F-100D-3-10-61A

Figure 2-52. Aileron Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-16040	Rib	0.032	2024-T4 Clad	2-54
2	192-16621	Web	0.040	2024-T4 Clad	
3	192-16621	Web	0.040	2024-T4 Clad	
4	192-16623	Web	0.040	2024-T4 Clad	
5	192-16623	Web	0.040	2024-T4 Clad	
6	192-16623	Web	0.040	2024-T4 Clad	
7	192-16622	Web	0.040	2024-T4 Clad	
8	192-16380	Web	0.040	2024-T4 Clad	
9	192-16380	Web	0.040	2024-T4 Clad	
10	192-16380	Web	0.040	2024-T4 Clad	
11	192-16380	Web	0.040	2024-T4 Clad	
12	223-16016	Rib	0.025	2024-T4 Clad	2-54
13	223-16023	Web	0.040	2024-T4 Clad	2-54
14	223-16019	Web	0.040	2024-T4 Clad	
15	223-16022	Web	0.050	2024-T4 Clad	2-54
16	223-16019	Web	0.040	2024-T4 Clad	
17	223-16019-5	Web	0.040	2024-T4 Clad	
18	223-16021	Web	0.063	2024-T4 Clad	2-54
19	223-16019-3	Web	0.040	2024-T4 Clad	
20	223-16020	Web	0.032	2024-T4 Clad	
21	223-16018	Rib	0.032	2024-T4 Clad	
22	223-16030	Rib	0.032	6061-T6 Clad	2-22
23	223-16031	Rib	0.032	6061-T6 Clad	2-22
24	223-16028	Fitting Assy		2014-T6 Die Forging	2-22
25	223-16032	Rib	0.032	6061-T6 Clad	2-22
26	223-16038	Rib	0.032	6061-T6 Clad	2-22
27	223-16029	Fitting		2014-T6 Extr	
28	223-16034	Rib	0.050	2024-T4 Clad	
29	223-16033	Rib	0.050	2024-T4 Clad	
30	192-16360	Fitting		2024-T4 Die Forging	
31	223-16036	Weight		Lead Alloy Cstg NAVY-L-16	
32	223-16057	Weight		Lead Alloy Cstg NAVY-L-16	
33	223-16026	Fitting	Extr	2014-T6	
34	192-16648	Weight		Leaded Naval Brass Bar	
35	223-16027	Fitting	Extr	2014-T6 Extr	
36	192-16531	Weight		Lead Alloy Cstg NAVY-L-16	
37	223-16015	TE Assy		Honeycomb	2-57, 2-60
38	223-16013	Spar	0.051	2024-T3 Clad	
39	223-16025	TE Assy		Honeycomb	2-57, 2-60
40	223-16012	Spar	0.040	2024-T3 Clad	
41	223-16004-3	Skin	Milled	2024-T4 Plate	2-51, 2-55, 2-56, 2-51A, 2-73
42	223-16004-1	Skin	Milled	2024-T4 Plate	2-51, 2-55, 2-51A
43	223-16005-3	Skin	Milled	2024-T4 Plate	2-51, 2-55, 2-56, 2-51A, 2-73
44	223-16005-1	Skin	Milled	2024-T4 Plate	2-51, 2-55, 2-51A

***Drawing number are for reference only and are not to be used for ordering parts.**

†For additional repairs, refer to Section X.

Key to Figure 2-52

AILERON—NEGLIGIBLE DAMAGE

WING

CORROSION

Corrosion damage not exceeding limits for cracks or scratches may be considered negligible. The corroded area must be cleaned and the chemical film treated. (Refer to index for "Corrosion Rework by Chemical Film Treatments.")

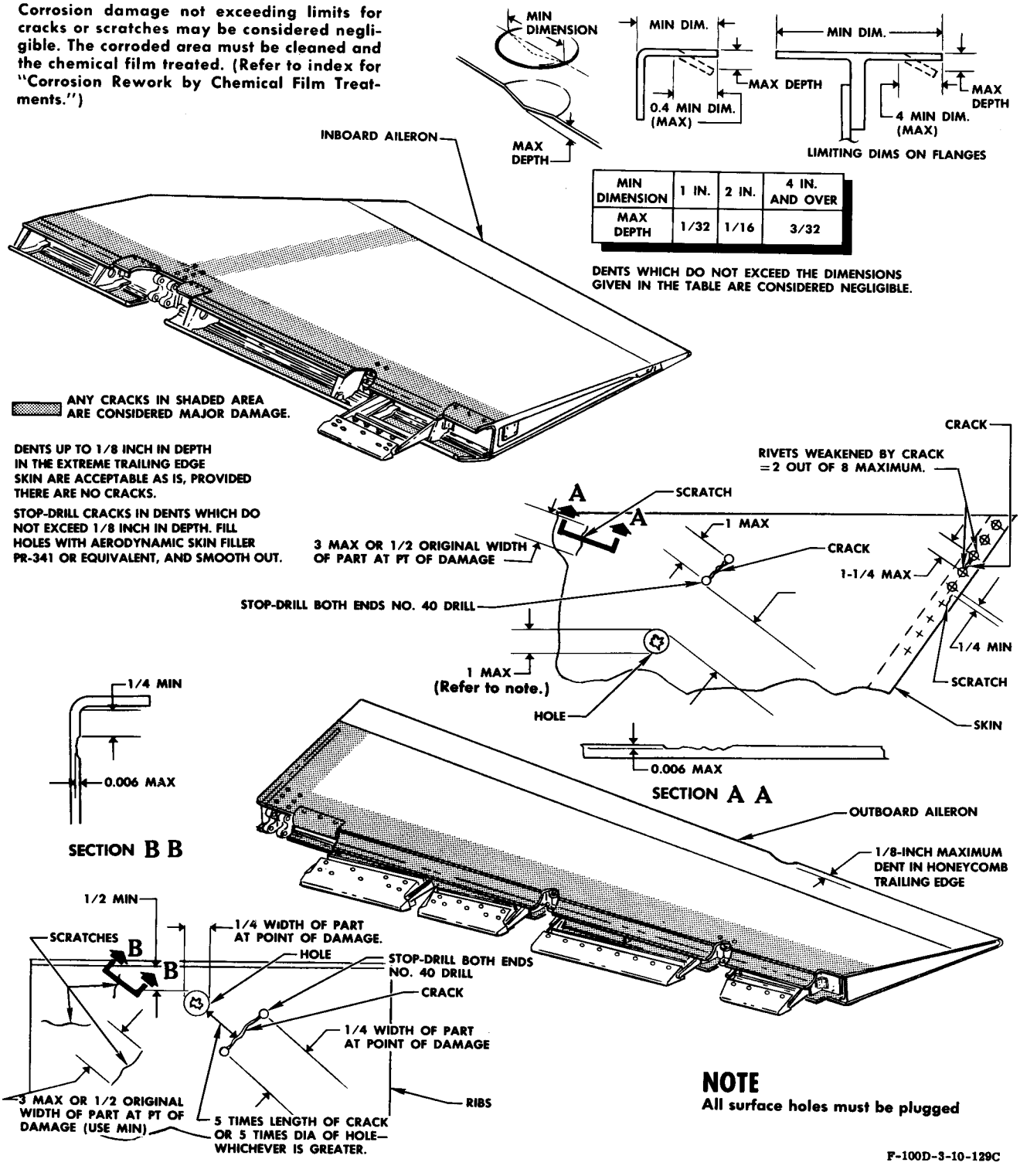
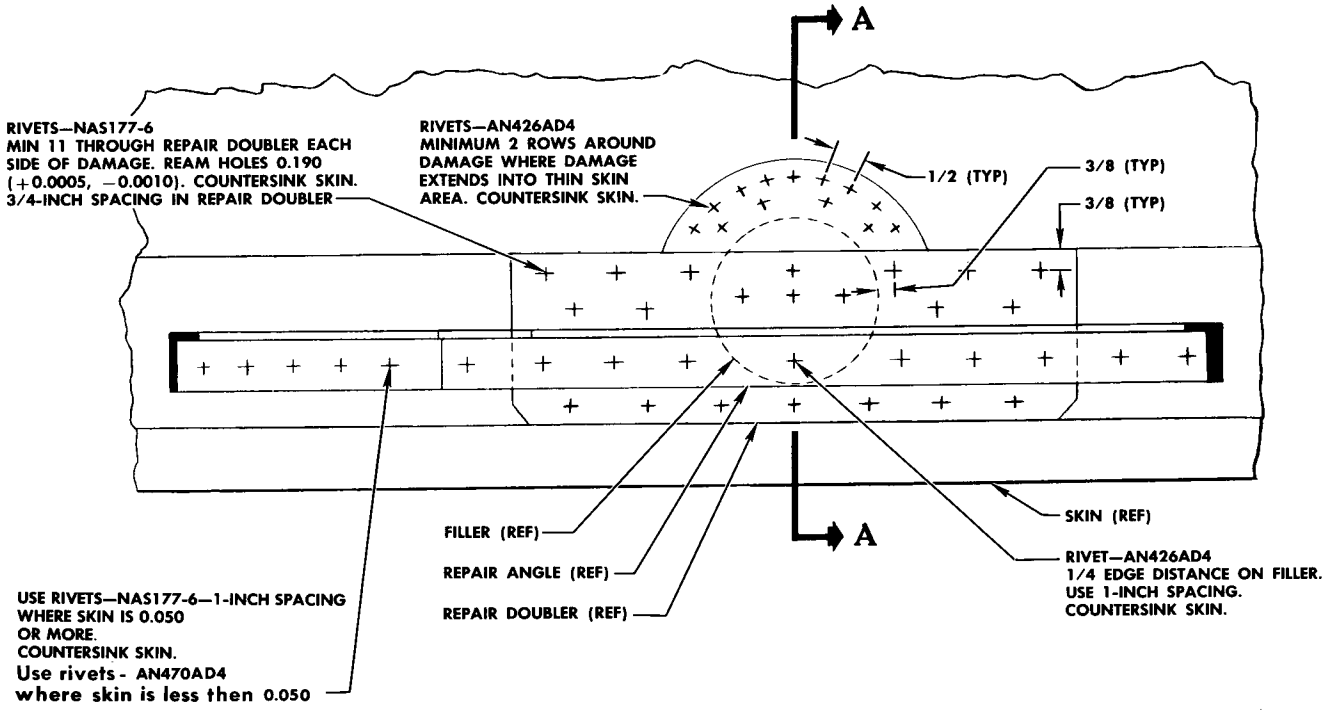
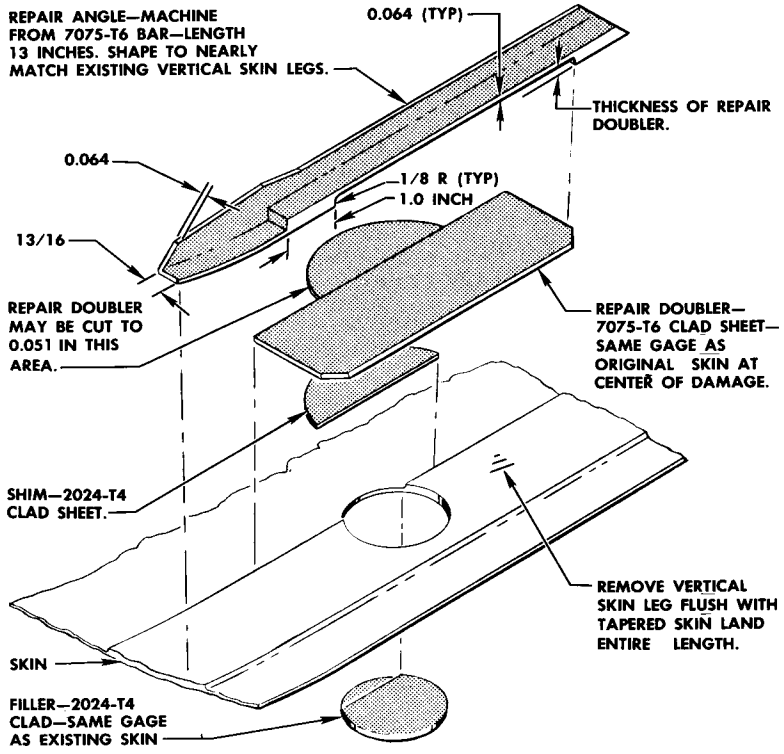


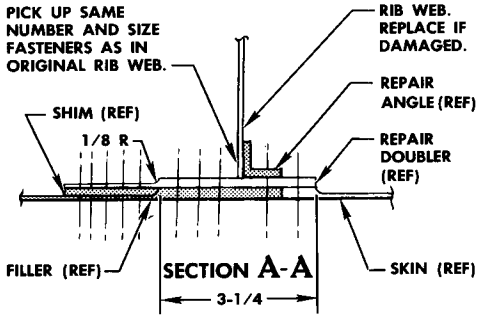
Figure 2-53. Aileron Negligible Damage

PERMANENT CATEGORY B

WING







1 Trim out damaged area.

2 Remove vertical skin leg flush with tapered skin land.

3 Fabricate repair angle, doubler, filler, and shim, if required.

4 Assemble repair as shown and install fasteners.

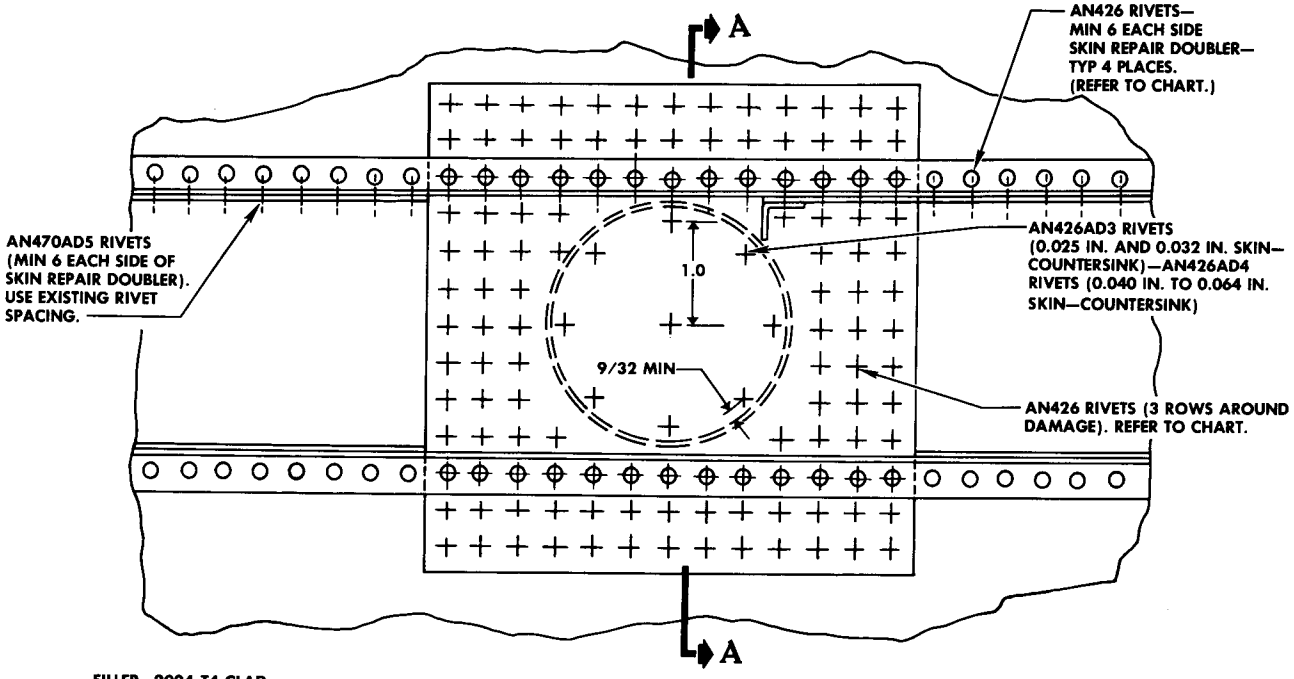
NOTE The length of the repair doubler is about 5-1/2 inches plus damage. Where the damage extends into the thin skin or so close to it that it is impossible to add one row of NAS177-6 rivets between the damage and the edge of the thick skin, the repair doubler must extend 1-1/2 inches past the damage or past the edge of the heavy land, whichever is greater.

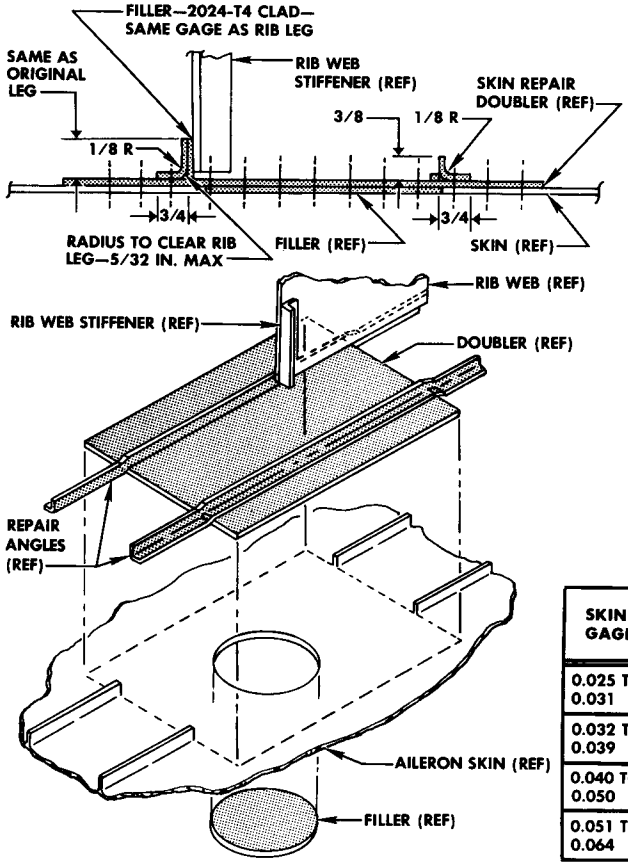
F-100D-3-10-108B

Figure 2-54. Outboard Aileron Skin and Actuator Rib—Station 168.179

PERMANENT—CATEGORY A

WING





1. Trim out damaged area, removing stiffeners as shown.
2. Make repair angles using 0.051-inch 2024-T4 clad material.
3. Make doubler, using 2024-T4 clad material at least 0.010 inch heavier than existing aileron skin. The length of each side should be about 4 inches plus the damage.
4. Make filler to match damaged area cutout, using 2024-T4 clad material the same gage as skin.
5. Assemble and rivet, using rivets as indicated.

NOTE The damaged area cutout does not need to be circular.

SKIN GAGE	RIVET SIZE	FLUSHING OPERATIONS FOR AN426 RIVETS THROUGH AILERON SKIN	RIVET SPACING	EDGE DISTANCE	DISTANCE BETWEEN ROWS
0.025 TO 0.031	AN426 AD3	DIMPLE SKIN AND REPAIR DOUBLER. COUNTERSINK REPAIR ANGLES.	9/16	9/32	9/16
0.032 TO 0.039	AN426 AD3	DIMPLE SKIN AND REPAIR DOUBLER. COUNTERSINK REPAIR ANGLES.	9/16	9/32	9/16
0.040 TO 0.050	AN426 AD4	COUNTERSINK SKIN.	9/16	9/32	9/16
0.051 TO 0.064	AN426 AD4	DIMPLE SKIN AND REPAIR DOUBLER. COUNTERSINK REPAIR ANGLES.	9/16	9/32	9/16

F-100D-3-10-109A

Figure 2-55. Aileron Skin at Ribs or Stiffeners

PERMANENT—CATEGORY A

WING

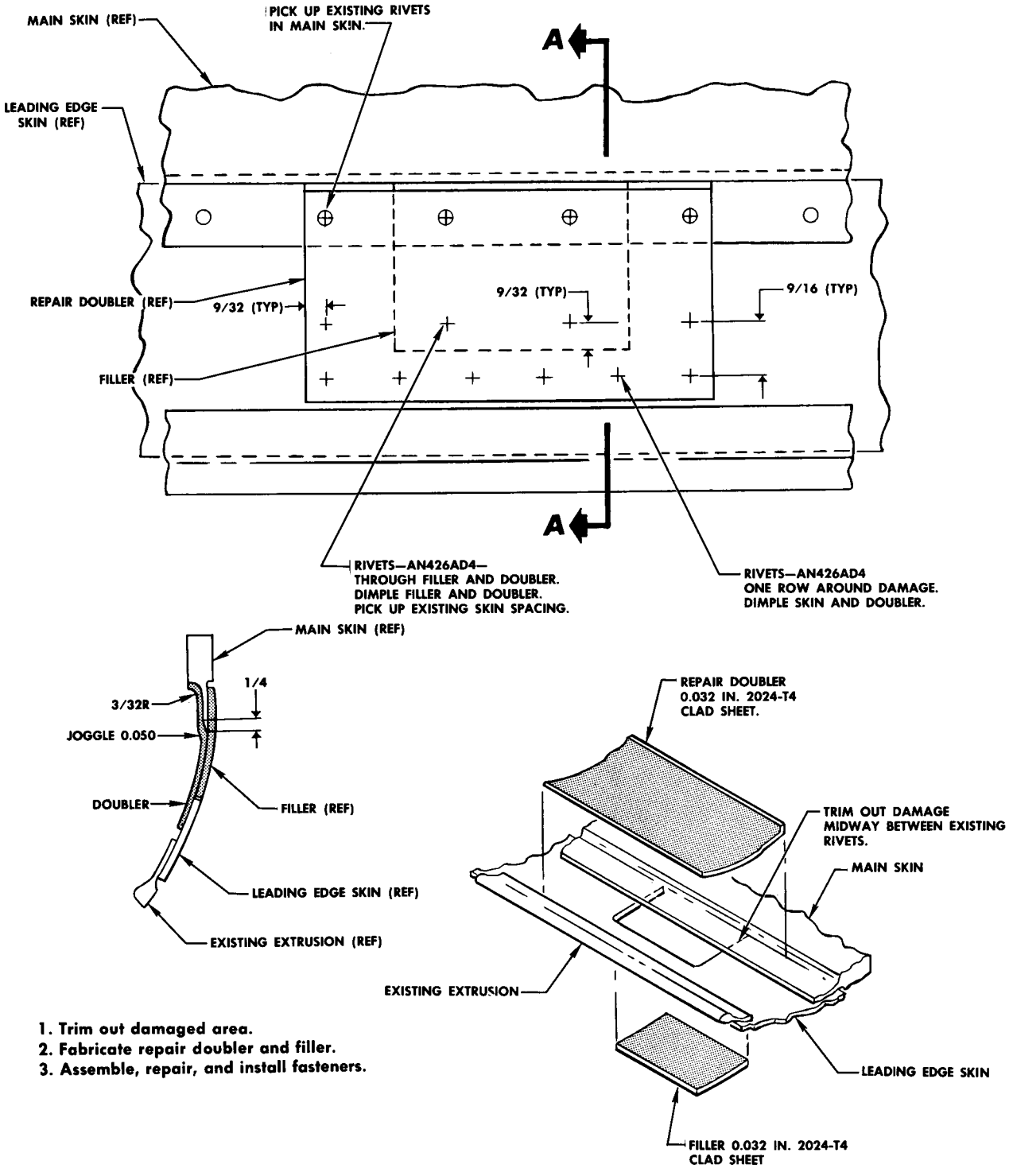
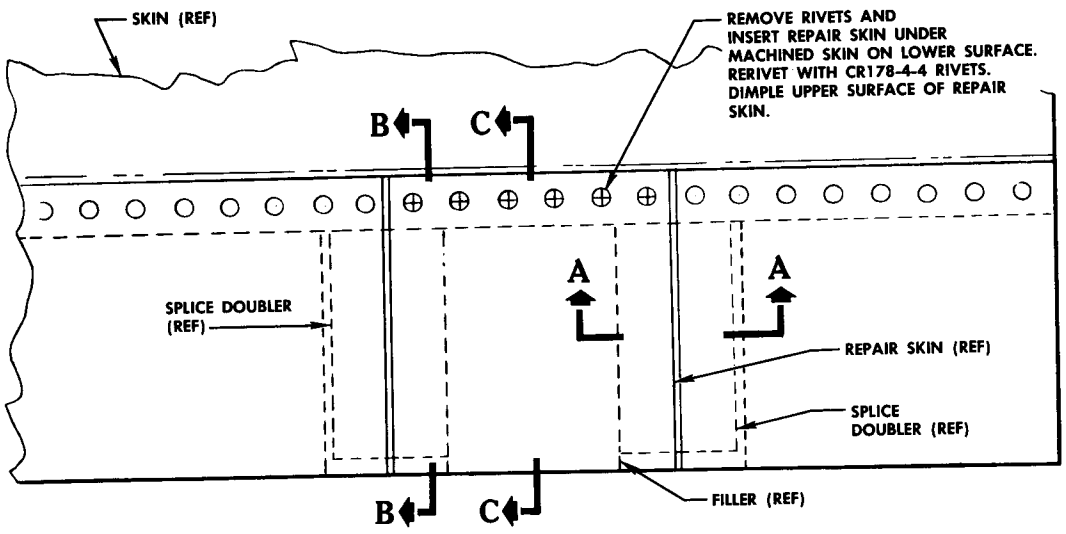


Figure 2-56. Aileron Lower Leading Edge

PERMANENT—CATEGORY A

WING



1 Trim out damaged area to aft spar face, and cut out honeycomb core.

2 Fabricate splice doubler, filler, and repair skin. Use balsa wood (Specification MIL-S-7998).

3 Apply plenty of adhesive, and assemble repair parts as shown. Refer to index for general instructions for repairing honeycomb sandwich panels.

4 Rivet repair skin in place.

5 Clean off excess adhesive.

NOTE For additional information, Refer to text.

- 2024-T4 clad sheet may be substituted for 6061-T6.
- The maximum number of repairs for any trailing edge is two.

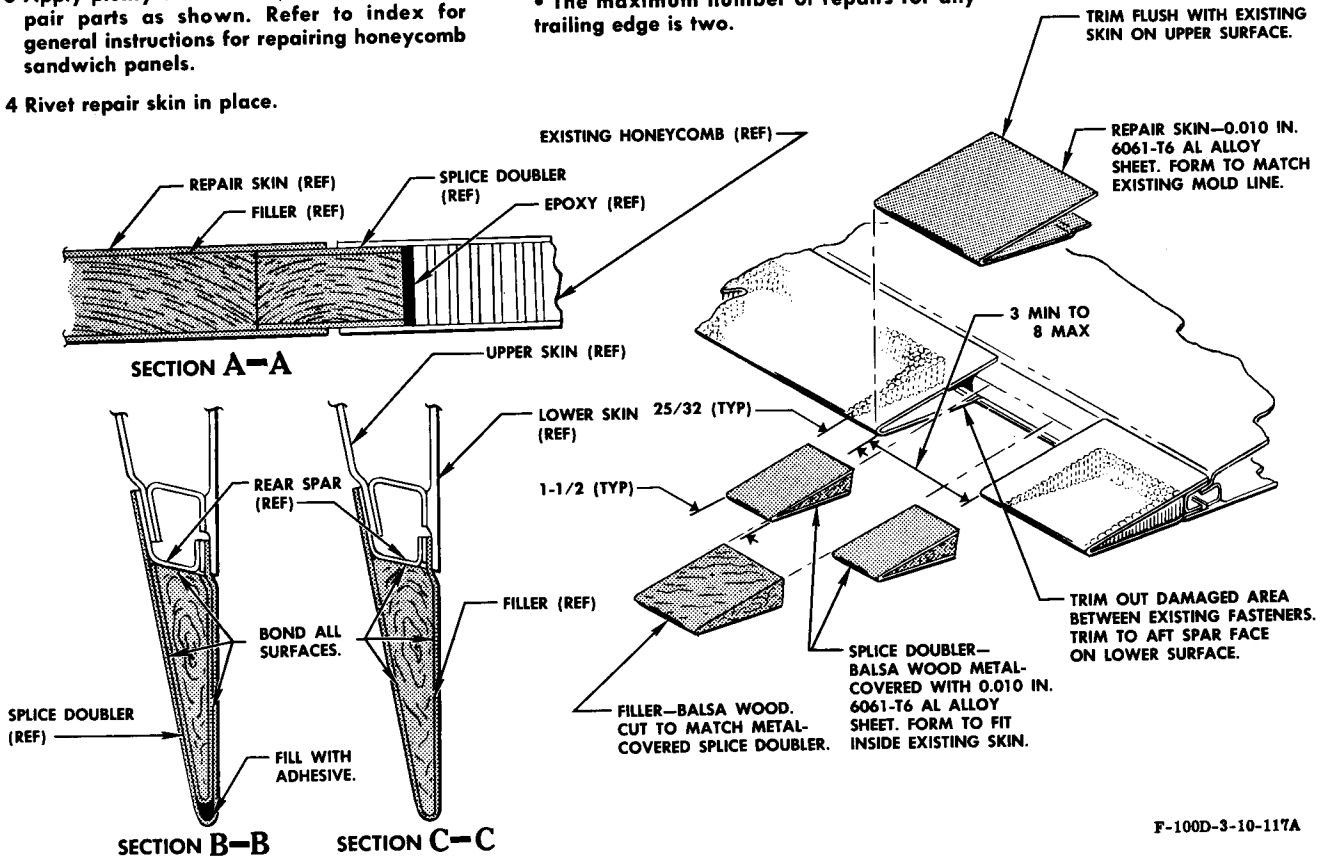
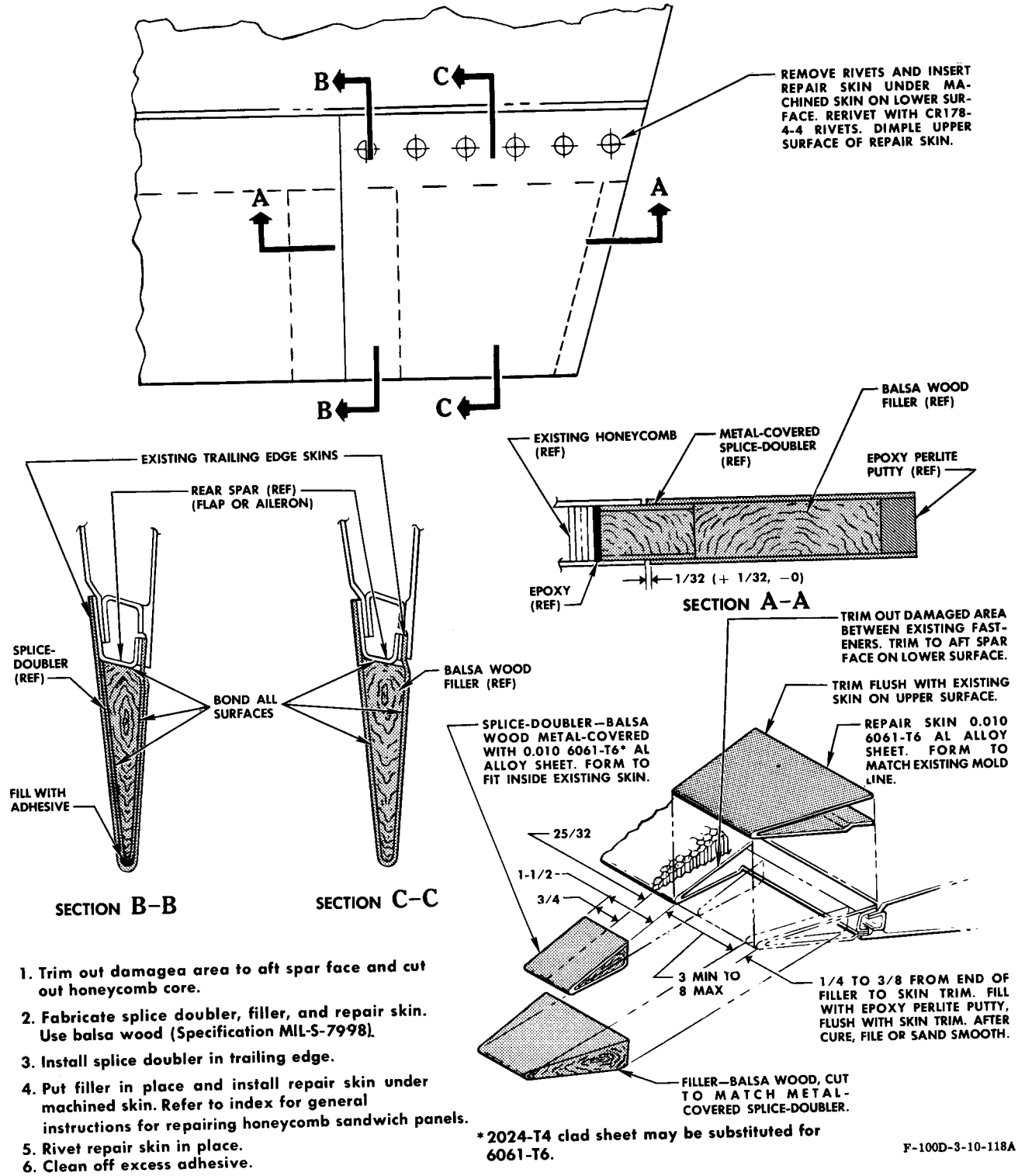


Figure 2-57. Aileron and Flap Trailing Edge—Honeycomb (Sheet 1 of 2)

PERMANENT—CATEGORY A **WING**



1. Trim out damagea area to aft spar face and cut out honeycomb core.
2. Fabricate splice doubler, filler, and repair skin. Use balsa wood (Specification MIL-S-7998).
3. Install splice doubler in trailing edge.
4. Put filler in place and install repair skin under machined skin. Refer to index for general instructions for repairing honeycomb sandwich panels.
5. Rivet repair skin in place.
6. Clean off excess adhesive.

*2024-T4 clad sheet may be substituted for 6061-T6.

Figure 2-57. Aileron and Flap Trailing Edge—Honeycomb (Sheet 2 of 2)

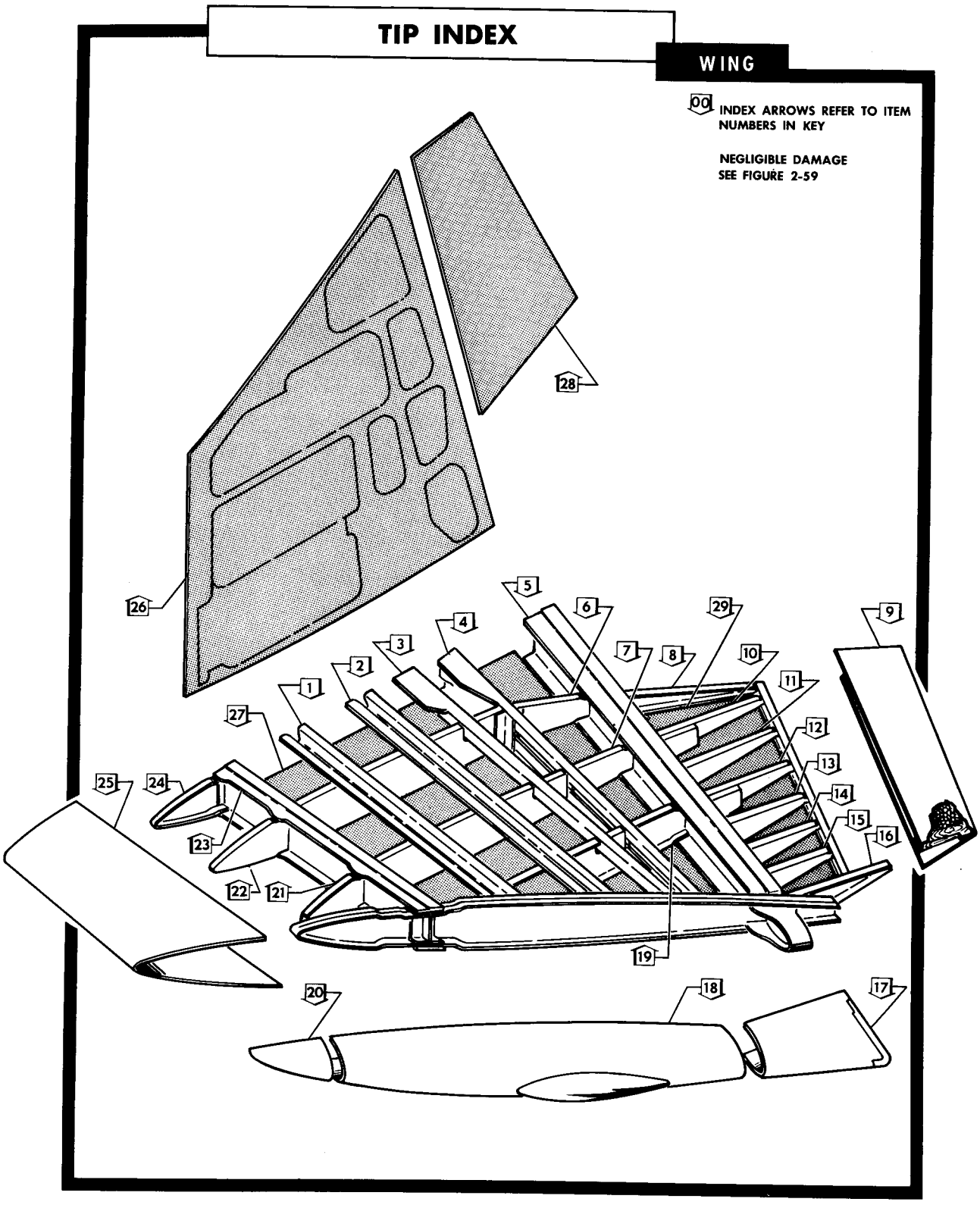


Figure 2-58. Wing Tip Repair Index

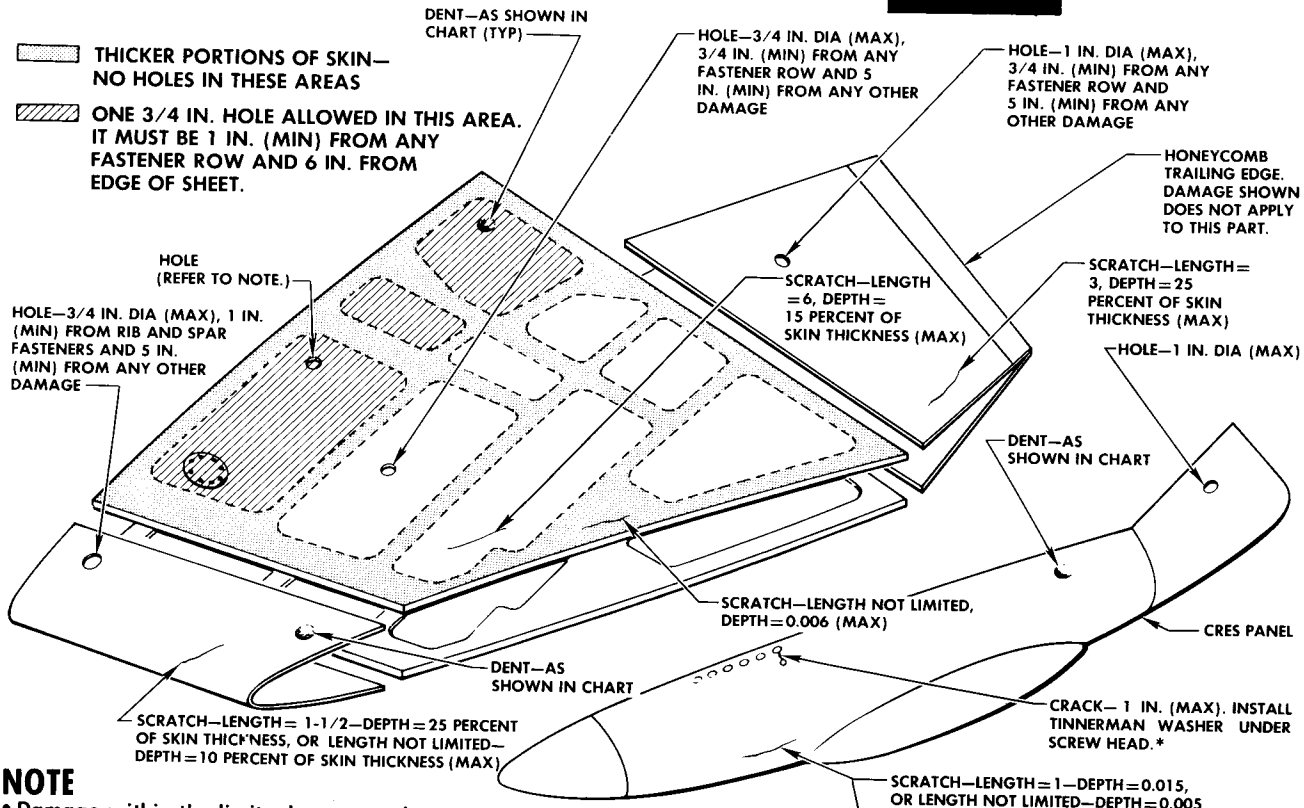
ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-14147-5	Stringer	Extr	7075-T6	2-59B
2	223-14147-7	Stringer	Extr	7075-T6	2-59B
3	223-14147-9	Stringer	Extr	7075-T6	2-59B
4	223-14119	Spar Assy	Extr	7075-T6	
5	223-14121	Spar Assy	Extr	7075-T6	
6	192-14257	Rib	0.040	2024-T4 Clad	
7	192-14257	Rib	0.040	2024-T4 Clad	
8	223-14254	Rib	0.051	2024-T4 Clad	2-22
9	223-14861	TE Assy		Honeycomb Const	2-60, 2-61, 2-62, 2-63
10	223-14255	Rib	0.040	2024-T4 Clad	2-22
11	192-14257-11	Rib	0.032	2024-T4 Clad	2-22
12	192-14257-13	Rib	0.040	2024-T4 Clad	2-22
13	192-14257-15	Rib	0.032	2024-T4 Clad	2-22
14	192-14257-17	Rib	0.032	2024-T4 Clad	2-22
15	192-14257-19	Rib	0.032	2024-T4 Clad	2-22
16	192-12258	Rib	0.032	2024-T4 Clad	
17	192-12013	Tip Assy Aft	0.051	5052-H34 Clad	
18	192-12012	Tip Assy Aft	0.020	CRES MIL-S-5059 Cond A	2-59A
19	192-12276	Rib Assy	0.040	2024-T4 Clad	2-22
20	192-12016	Front Tip Panel Assy			
	192-12016-3	Skin	0.051	5052-H34 Clad	
	192-12016-7	Rib	0.032	2024-T4 Clad	2-22
21	192-12243	Rib Assy	0.051	2024-T4 Clad	2-22
22	192-12277	Rib Assy	0.040	2024-T4 Clad	2-22
23	192-12120	Spar Assy	Extr	7075-T4	
24	180-14208	Rib Assy	0.051	2014-T6 Clad	
25	223-140113	Skin	0.051	2024-T4 Clad	
26	223-14147-21	Skin	Milled	7075-T6 Sheet	
27	223-14148-1	Skin	Milled	7075-T6 Sheet	
28	223-14015-5	Skin	0.032	2024-T4 Clad	
29	223-14015-3	Skin	0.032	2024-T4 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 2-58

TIP NEGLIGIBLE DAMAGE

WING



NOTE

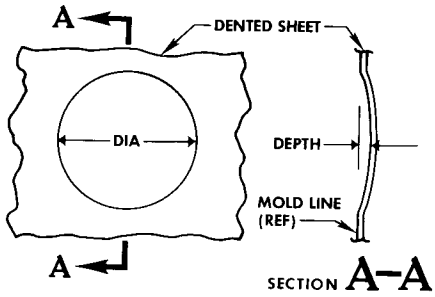
- Damage within the limits shown may be considered negligible.
- Unless otherwise noted, crack limits and restrictions are the same as those given for holes (crack length equals hole diameter). Stop-drill end of crack with a No. 40 drill.
- Outer wing panel cracked leading edge skins should be repaired or left "as is," depending on the degree of damage as noted in the following categories:

  1. Crack damage requiring minor repair or no repair—a maximum of four interconnected cracked holes (3 inches maximum length), or a maximum combination of four cracked holes separated by a good hole (combined length, 3 inches maximum). Cracks not terminating in a rivet hole should be stop-drilled, using a No. 40 (0.098-inch diameter) drill.

  2. Damage requiring skin replacement or structural repairs—a crack or combination of cracks exceeding the acceptable limits given in 1.

- Corrosion damage not exceeding limits for scratches or holes may be considered negligible.
- The rework required for negligible damage is given in "Negligible Damage" in Section I.)
- Damage shown is typical for upper and lower surfaces.

*This panel is nonstructural. No limit is set for the number of cracks a panel may have; however, damage should not be allowed to progress to the point where it would not be economical to repair the panel.



NEGLIGIBLE DENTS

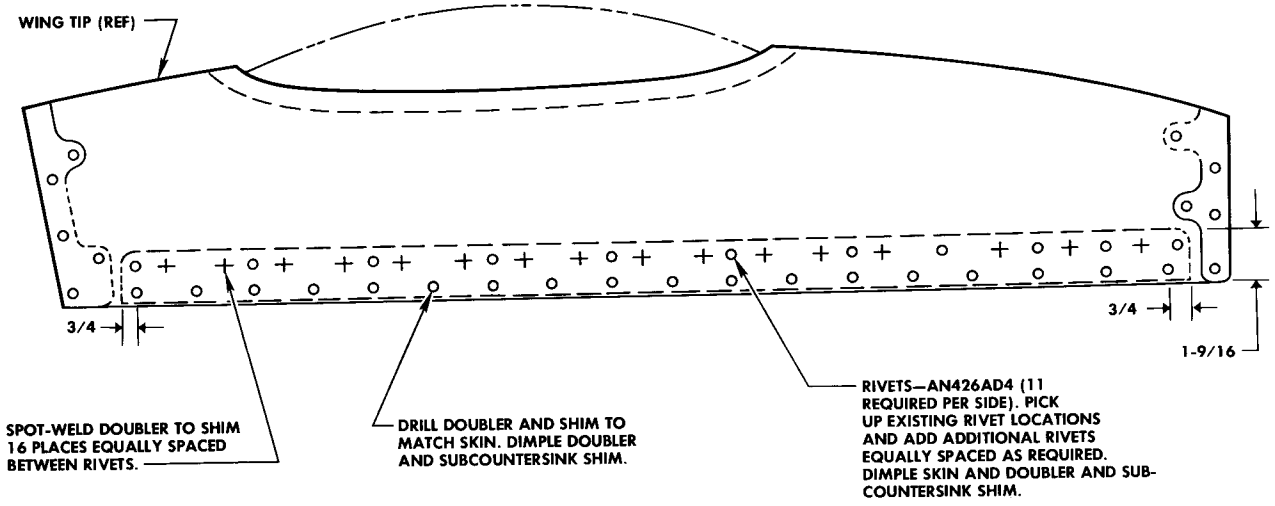
MIN DIA	1 IN.	2 TO 3 IN.	4 IN. AND OVER
MAX DEPTH	1/32	1/16	3/32

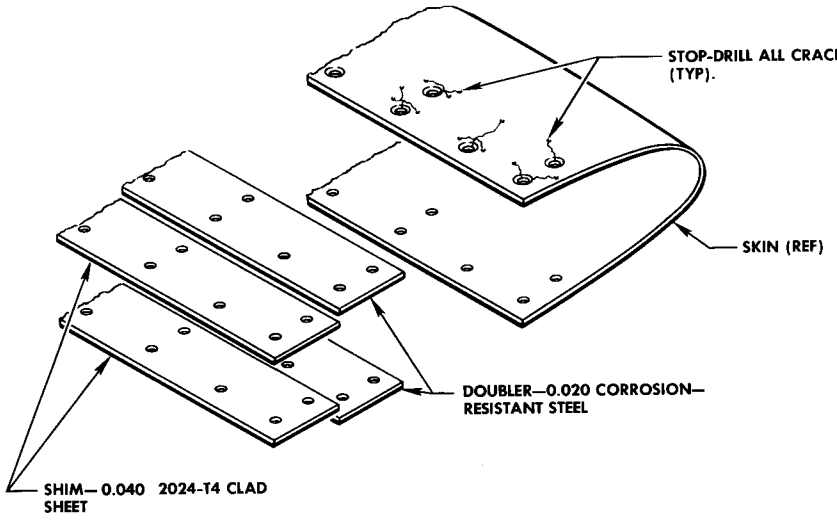
F-100D-3-10-142A

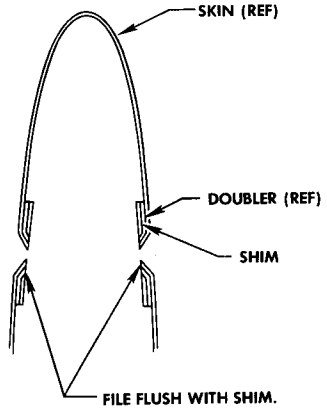
Figure 2-59. Wing Tip Negligible Damage

PERMANENT CATEGORY A

WING







1. Remove existing shim.
2. Stop-drill all cracks.
3. Fabricate and install new doubler and shim.

NOTE When repair is made on one surface, upper or lower, it should be duplicated on the opposite surface even though no cracks are apparent.

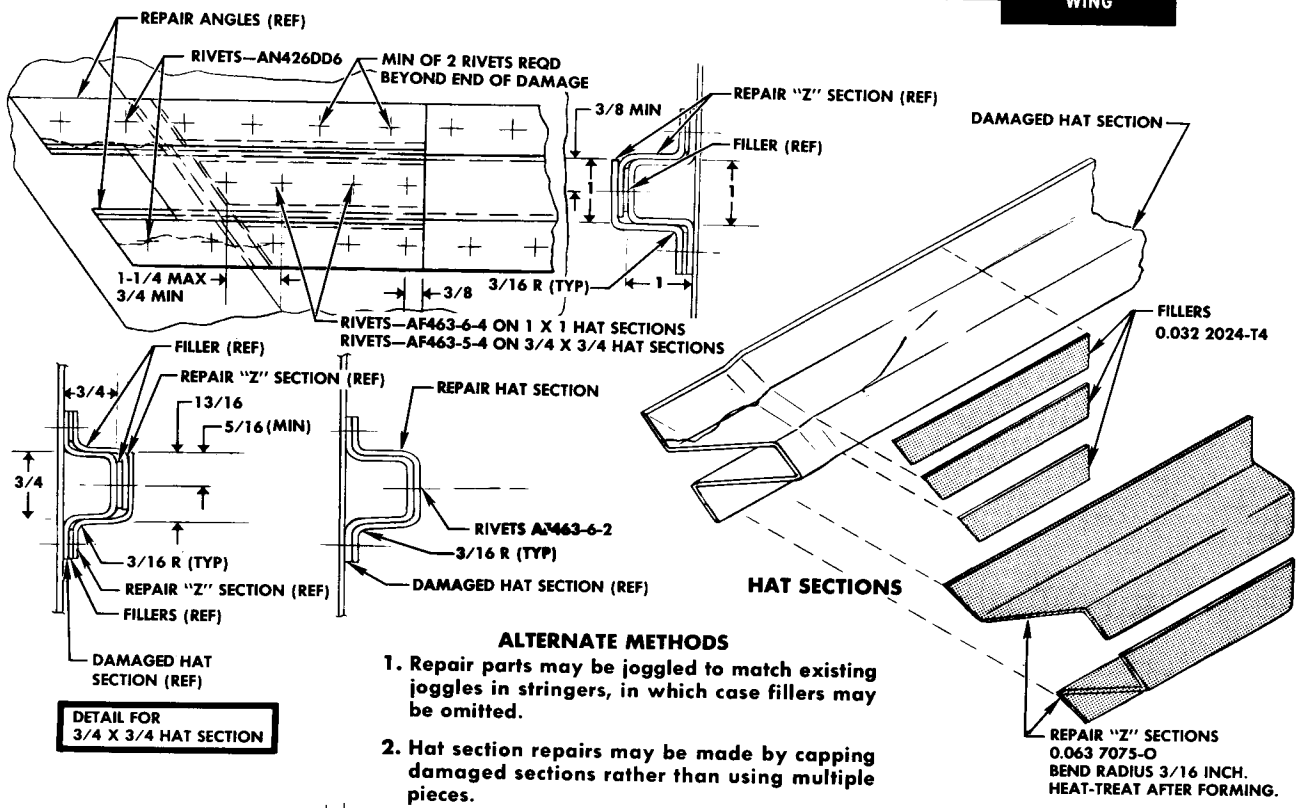
- 7075-T6 clad sheet may be substituted for 2024-T4.

F-100D-3-10-141

Figure 2-59A. Wing Tip Attachment Points

PERMANENT CATEGORY A

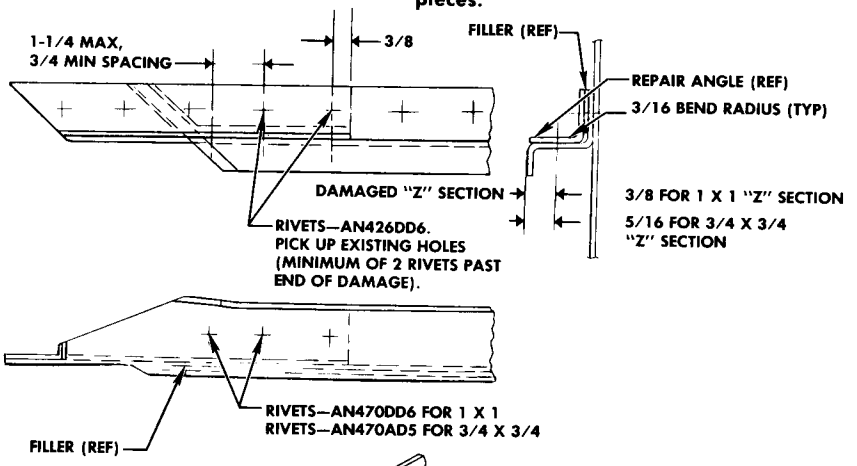
WING



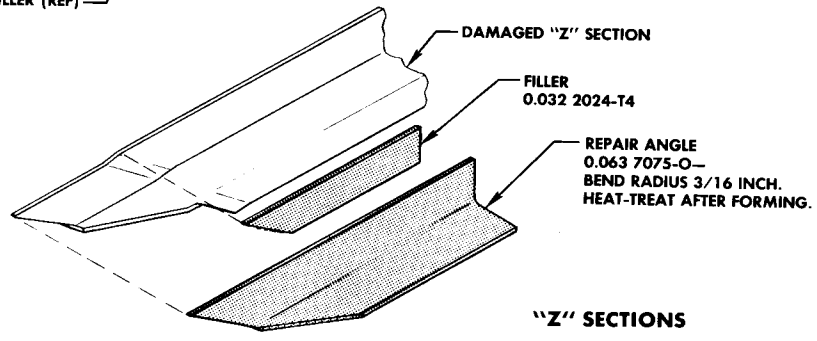
DETAIL FOR 3/4 X 3/4 HAT SECTION

ALTERNATE METHODS

1. Repair parts may be joggled to match existing joggles in stringers, in which case fillers may be omitted.
2. Hat section repairs may be made by capping damaged sections rather than using multiple pieces.



1. Stop-drill all cracks with a 1/8-inch diameter drill. Do not allow drill to touch the skin.
2. Make repair angles or "Z" sections and fillers as required. Heat-treat and heat-age the repair angles or "Z" sections to T6 condition after forming.
3. Trim the ends of the repair angles or "Z" sections to match the end of the damaged part.
4. Fasten repair parts in place, using size and type of fastener and spacing called out.



NOTE

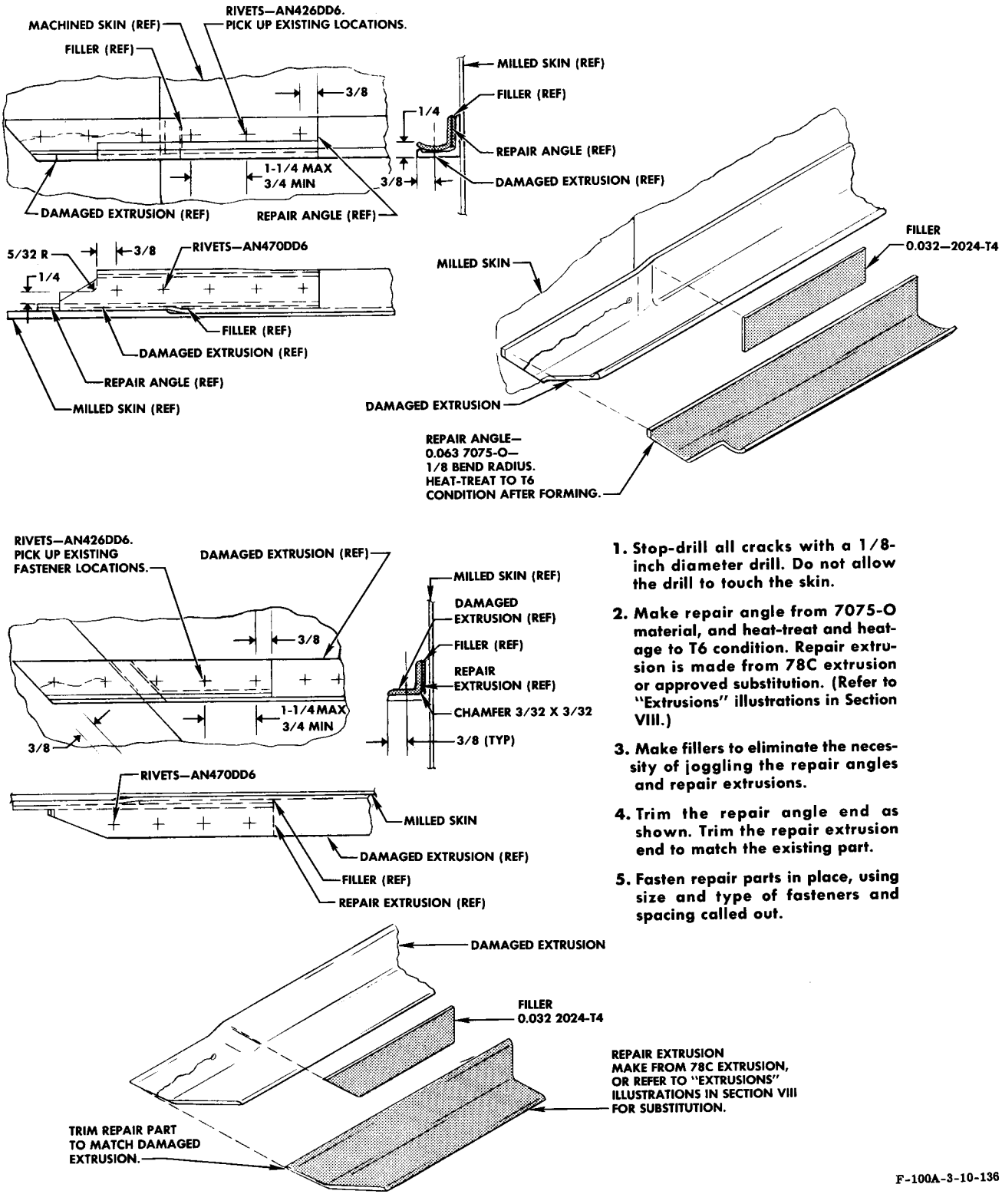
On damaged hat sections, use repair "Z" sections on both flanges even though only one flange is damaged.

F-100A-3-10-135

Figure 2-59B. Wing Tip Stringer Repair (Sheet 1 of 2)

PERMANENT CATEGORY A

WING



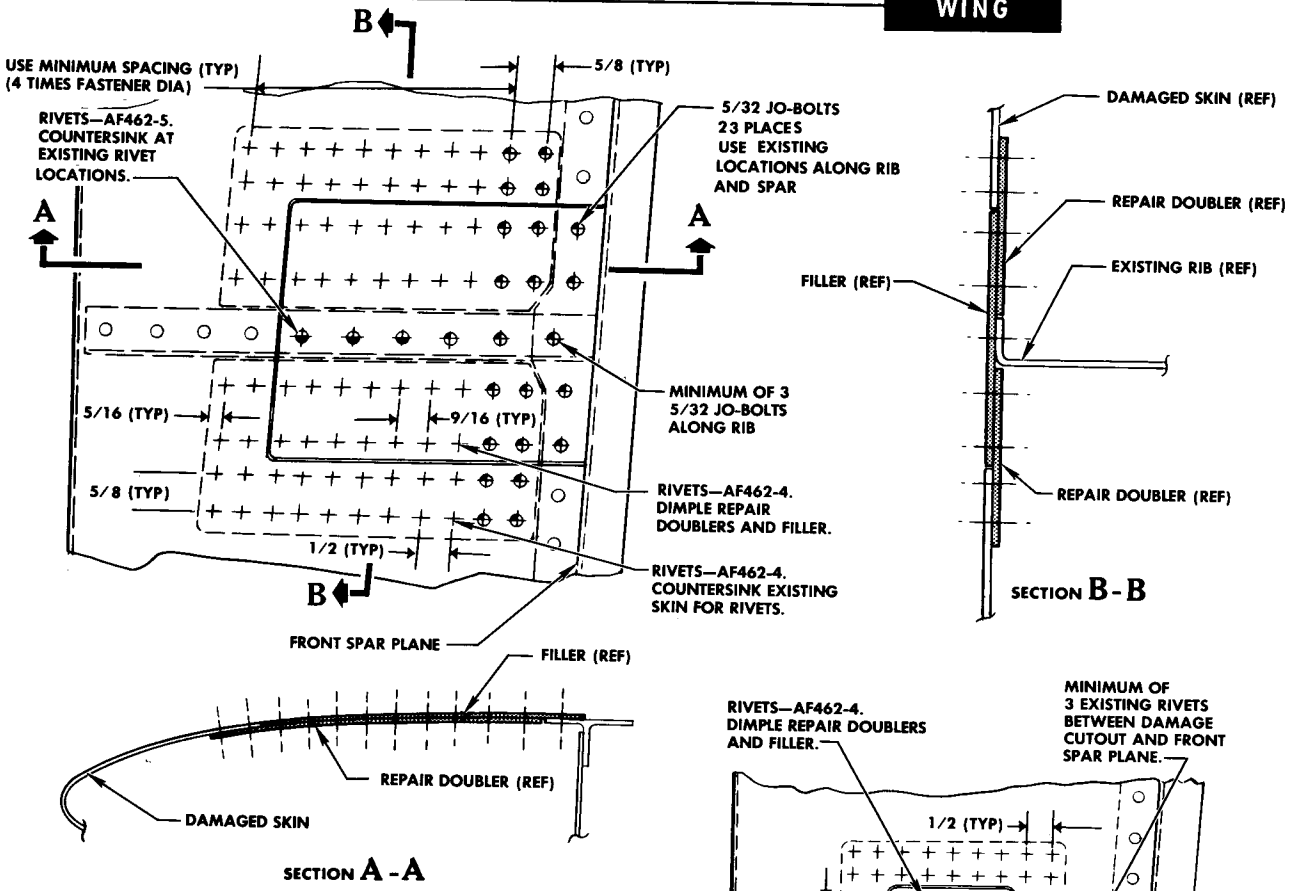
1. Stop-drill all cracks with a 1/8-inch diameter drill. Do not allow the drill to touch the skin.
2. Make repair angle from 7075-O material, and heat-treat and heat-age to T6 condition. Repair extrusion is made from 78C extrusion or approved substitution. (Refer to "Extrusions" illustrations in Section VIII.)
3. Make fillers to eliminate the necessity of joggling the repair angles and repair extrusions.
4. Trim the repair angle end as shown. Trim the repair extrusion end to match the existing part.
5. Fasten repair parts in place, using size and type of fasteners and spacing called out.

F-100A-3-10-136

Figure 2-59B. Wing Tip Stringer Repair (Sheet 2 of 2)

PERMANENT—CATEGORY A

WING



1 Trim out damage midway between existing fasteners.

2 Smooth all edges and corners after trimming.

3 Make repair doublers and fillers. Contour them to match existing mold line.

4 Place repair doublers in position on outside of damaged skin. Drill holes in corners of repair doublers, enabling attachment to damaged skins.

5 Insert repair doublers in place on inside of damaged skin, and complete drilling fastener pattern.

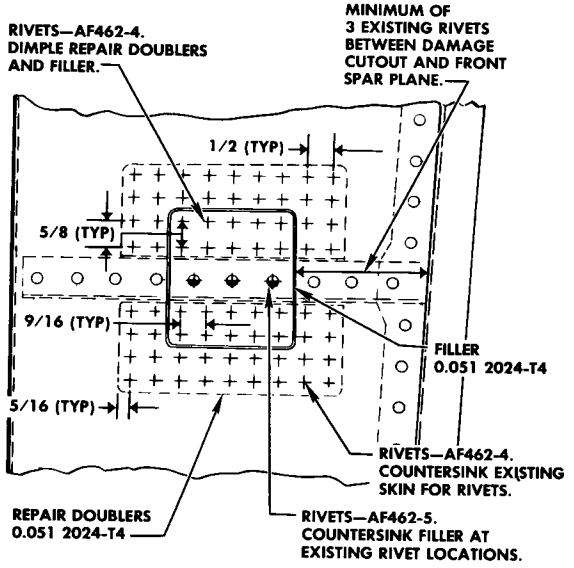
6 Remove repair doublers, and dimple both repair doublers and damaged skin.

7 Reinstall repair doublers and fasten them in place, using size and type of fasteners called out.

8 Place filler in position, pick up existing fastener locations required, and complete drilling fastener pattern.

9 Countersink filler, and fasten it in place with required size and type of fasteners.

NOTE Minimum fastener spacing is four times fastener diameter. Minimum edge distance is two times fastener diameter.



ALTERNATE REPAIR

A minimum of three existing rivets between damage cutout and front spar plane is necessary before this repair can be used.

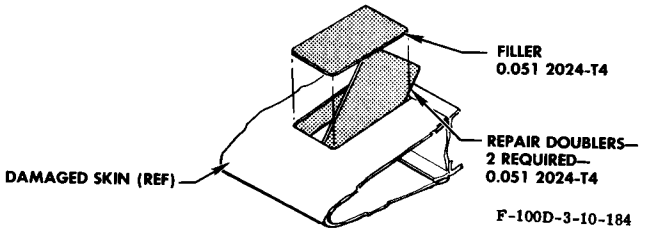
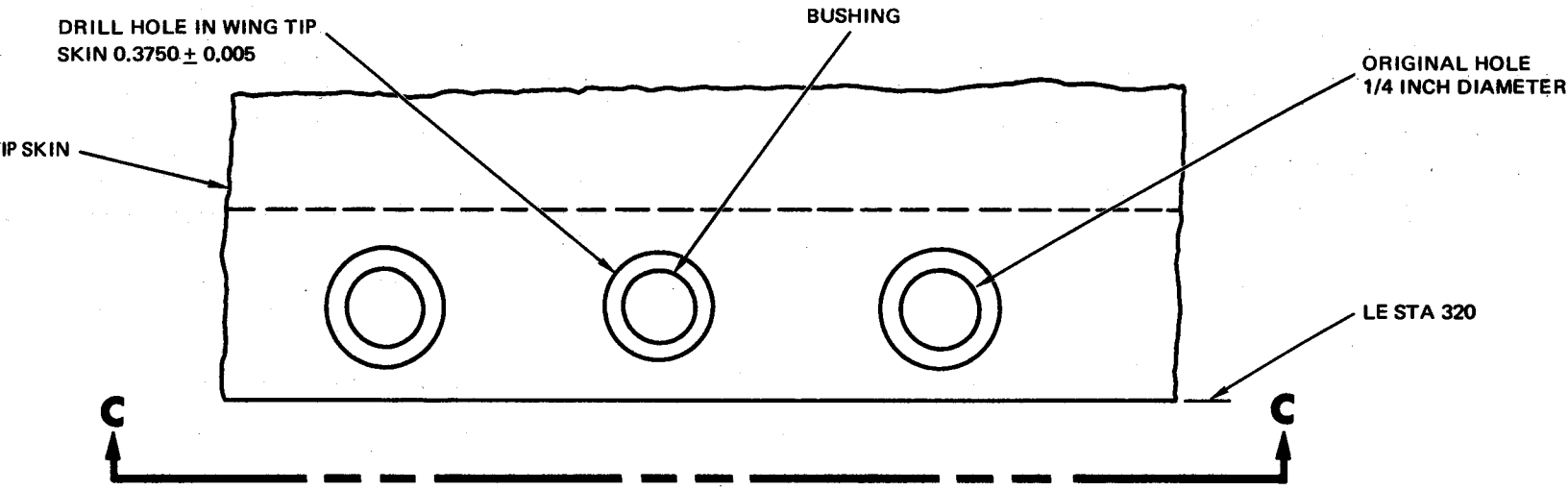
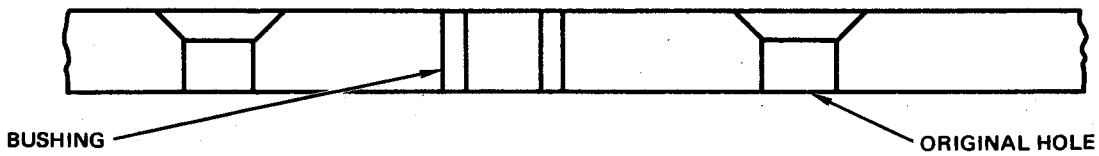


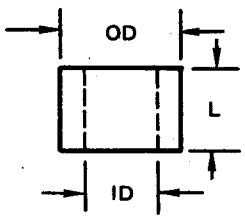
Figure 2-59C. Wing Tip Leading Edge Skin Repair



VIEW LOOKING DOWN AT A SEGMENT OF THE WING TIP SKIN ALONG LE STA 320

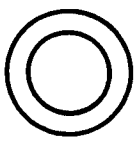


SECTION C-C



L	=	0.1780	+0.0000 −0.0050
OD	=	0.3765	+0.0000 −0.0005
ID	=	0.2500	+0.0040 −0.0000

MATERIAL – 7075 T6



BUSHING

Figure 2-59D. Wing Tip Panel Corrosion Damage Repair - LE STA 320, F-100D & F. (Sheet 1 of 2)

WING TIP CORROSION REMOVAL LIMITS AT LE STA 320

HOLE LOCATION ALONG LE STA 320	LOWER WING TIP SKIN			UPPER WING TIP SKIN		
	MAX SKIN REMOVAL AROUND ATTACH FASTENERS, INCHES			MAX SKIN REMOVAL AROUND ATTACH FASTENERS, INCHES		
	DIA = 1/4	DIA = 5/16	DIA = 3/8	DIA = 1/4	DIA = 5/16	DIA = 3/8
1	0.108	0.106	0.096	0.098	0.100	0.092
2						
3						
4						
5						
6	0.108	0.106	0.096			
7	0.068	0.065	0.054			
8						
9				0.098	0.100	0.092
10		0.065	0.054	0.056	0.055	0.043
11		0.071	0.061			
12						
13				0.056	0.055	0.043
14				0.044	0.045	0.034
15				0.044	0.045	0.034
16				0.068	0.071	0.061
17						
18	0.068	0.071	0.061			
19	0.108	0.106	0.106	0.068	0.071	0.061
20				0.098	0.100	0.092
21	0.108	0.106	0.106			
22				0.098	0.100	0.092

NOTE REMOVAL LIMITS MAY BE EXCEEDED BY 0.005 IF LIMITED TO NOT MORE THAN TWO HOLES IN ANY GROUP OF THREE.

- NORMAL FASTENER HOLE DIAMETER AT THE WTP/WOP ATTACH FLANGE ALONG LE STA 320 IS 1/4 INCH. FASTENER HOLES MAY BE ENLARGED TO A MAXIMUM DIAMETER OF 3/8 INCH TO REMOVE CORROSION. INSTALL A BUSHING ON HOLES ENLARGED TO 3/8 INCH DIAMETER AND REDRILL TO 1/4 INCH DIAMETER. MAKE BUSHING OUT OF 7075-T6 ALUMINUM ALLOY. APPLY STRONTIUM CHROMATE PRIMER MIL-P-23377 TO BUSHINGS AND INSTALL WHEN WET.

REPAIR PROCEDURE:

1. REMOVE CORROSION FROM AFFECTED AREAS. REFER TO APPLICABLE –3 PROCEDURE.
2. DETERMINE DEPTH OF METAL REMOVED. REFER TO APPLICABLE –3 PROCEDURE FOR GRIND OUT LIMITS. EXCEEDANCE OF GRIND OUT LIMITS WILL REQUIRE INSTALLATION OF AN EXTERNAL DOUBLER. FABRICATE DOUBLER AS REQUIRED IAW FABRICATION INSTRUCTIONS.
3. APPLY AERODYNAMIC FILLER (SEE APPENDIX 1 of –3) TO BUILD UP NOMINAL THICKNESS AND COMPENSATE FOR THICKNESS LOSS DURING GRINDING OUT OF CORRODED AREAS.
4. APPLY 2 COATS OF ZINC CHROMATE PRIMER MIL–P–8585 TO INTERIOR SURFACE OF ALL FASTENER HOLES AND INTERIOR SURFACE OF WTP SKIN AND STRINGERS.

(1) DO STEPS 5 THRU 7 IF DOUBLER IS REQUIRED.

5. APPLY STRONTIUM CHROMATE PRIMER MIL–P–23377 TO CONTACT SURFACE BETWEEN DOUBLER AND WTP SKIN.
6. FILL IN COUNTERSINKS ON WTP SKIN AT LE STA 320 WITH TAPERED ALUMINUM WASHERS.
7. ATTACH DOUBLER TO WTP SKIN. WET DIP FASTENERS IN MIL–P–8585. TIGHTEN TO PROPER TORQUE. WTP IS READY FOR ATTACHING TO WOP.
8. APPLY 2 COATS OF MIL–P–8585 TO WOP MATING SURFACE AND INSIDE FASTENER HOLES. INSTALL WTP. WET DIP FASTENERS IN MIL–P–8585. TIGHTEN FASTENERS TO PROPER TORQUE.
9. APPLY SEALANT MIL–S–8802D, TYPE B–4, IN THE VOID BETWEEN THE WOP AND WTP SKINS.
10. CHECK SPANWISE ALIGNMENT OF WTP WITH WOP. ALIGNMENT MAY BE CHECKED BY PLACING A STRAIGHT EDGE ACROSS THE WOP AND WTP. IF A DOUBLER IS INSTALLED, A NOTCH MUST BE CUT OUT OF THE STRAIGHT EDGE SO AS TO ALLOW IT TO REST FLUSH AGAINST THE TWO SURFACES.

DOUBLER FABRICATION:

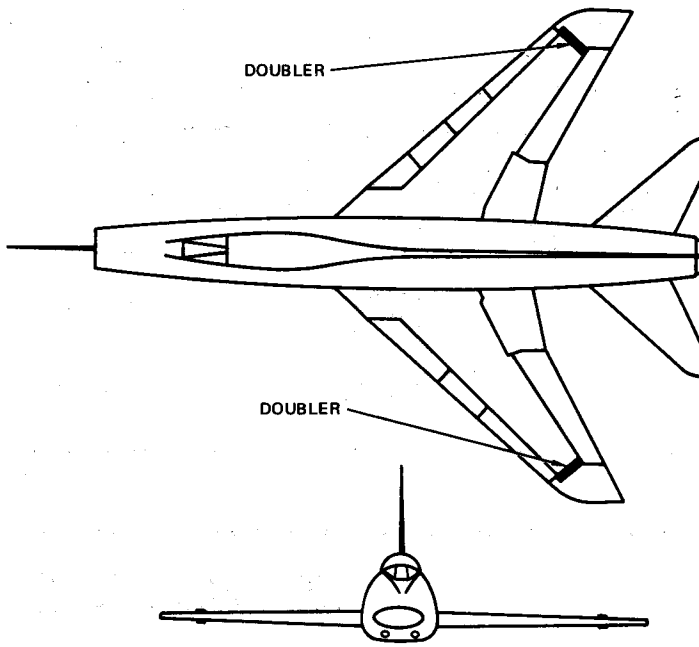
1. DETERMINE CHORDWISE LENGTH OF DOUBLER. THE LOCATION AND NUMBER OF HOLES EXCEEDING GRIND OUT LIMITS DETERMINE LENGTH OF DOUBLER. MAKE DOUBLER LONG ENOUGH TO PICK UP TWO ADDITIONAL HOLES BEYOND DAMAGED FASTENER HOLES. MINIMUM CHORDWISE LENGTH IS THE DISTANCE BETWEEN TWO STRINGERS. MINIMUM FASTENER EDGE DISTANCE IS TWO TIMES THE FASTENER DIAMETER. SPANWISE LENGTH OF DOUBLER EXTENDS OVER 4 FASTENERS OUTBOARD OF LE STA 320, SEE SECTION B–B. IF A DOUBLER IS ALSO REQUIRED ON THE SAME SURFACE FOR THE WOP, THE REQUIRED DOUBLER IS A COMBINATION OF THAT FOR THE WOP AND WTP. MAKE THE COMBINATION DOUBLER A SINGLE CONTINUOUS SPANWISE DOUBLER. SEE THE WOP CORROSION DAMAGE REPAIR PROCEDURE FOR DIMENSIONS OF WOP DOUBLER.
2. FABRICATE DOUBLER FROM 7075T6 SHEET 1/8–INCH THICK.
3. CHAMFER EDGES 45°.
4. MACHINE ALL SURFACES AND EDGES 125
5. ROUND OFF SHARP EDGES.
6. TRANSFER APPLICABLE HOLE PATTERN ON WTP TO DOUBLER. HOLES MUST BE DRILLED AND COUNTERSUNK ON A DRILL PRESS TO INSURE PROPER SEATING OF SCREWS.
7. APPLY ZINC CHROMATE PRIMER MIL–P–8585 TO MATING SURFACE OF DOUBLER AND INSIDE HOLE SURFACES.

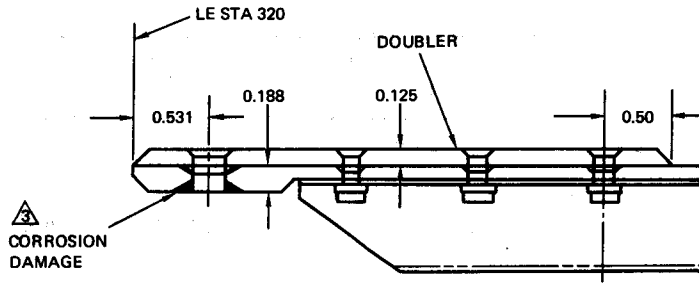
NOTE

(1) IF A DOUBLER IS ALSO REQUIRED ON THE SAME SURFACE FOR THE WOP, THE COMBINATION DOUBLER MUST BE INSTALLED FIRST ON THE WOP. REFER TO WOP CORROSION DAMAGE REPAIR PROCEDURE FOR INSTALLATION OF DOUBLER.

(2) IF A COMBINATION DOUBLER IS REQUIRED, USE NUTPLATES INSTEAD OF SELF–WRENCHING NUTS TO ATTACH DOUBLER TO WTP. ATTACH NUTPLATES TO INTERIOR SURFACE OF WTP.

(3) THIS REPAIR PROCEDURE APPLIES ONLY TO CORROSION DAMAGE AT THE WTP/WOP ATTACH FLANGE AT LE STA 320.





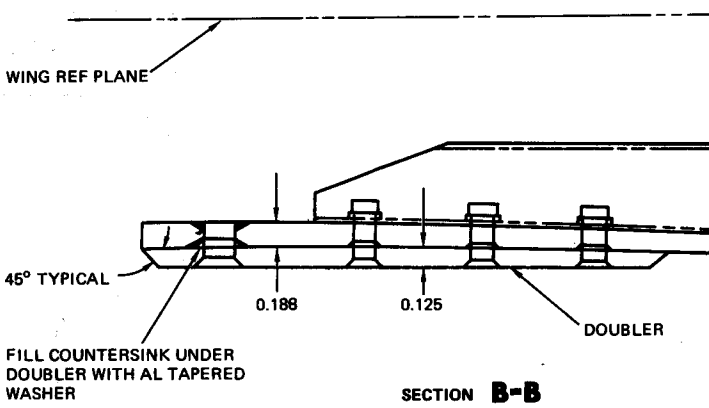
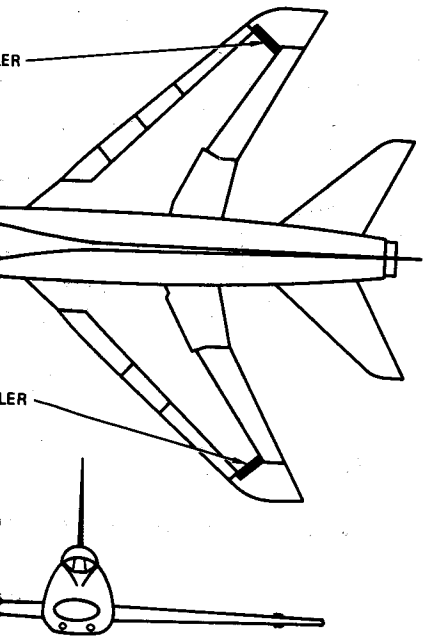
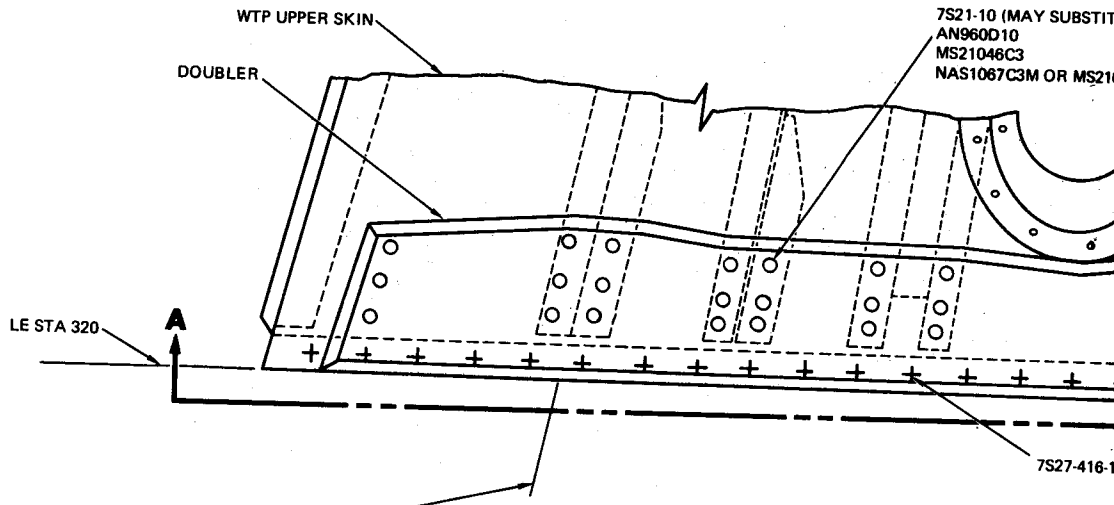


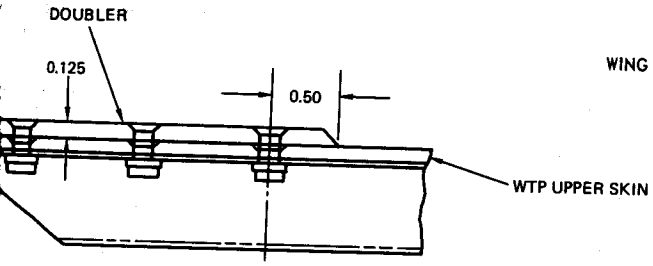
Figure 2-59D. Wing Tip Panel Corrosion Damage Repair - LE STA 320, F-100D & F. (Sheet 2 of 2)

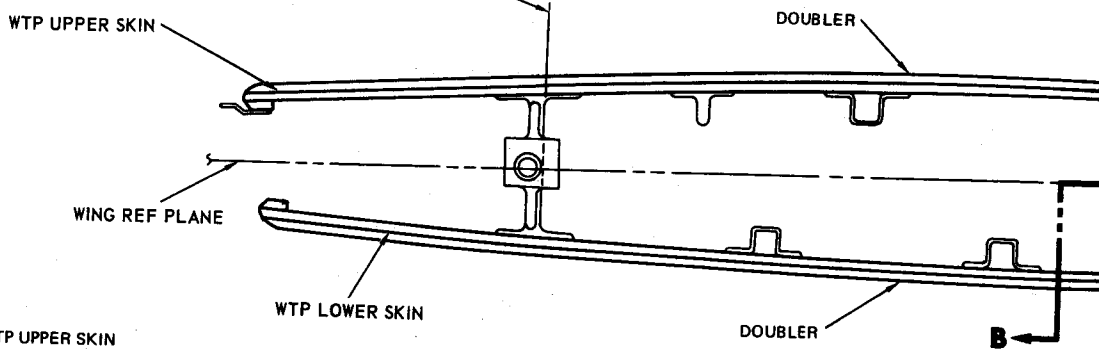




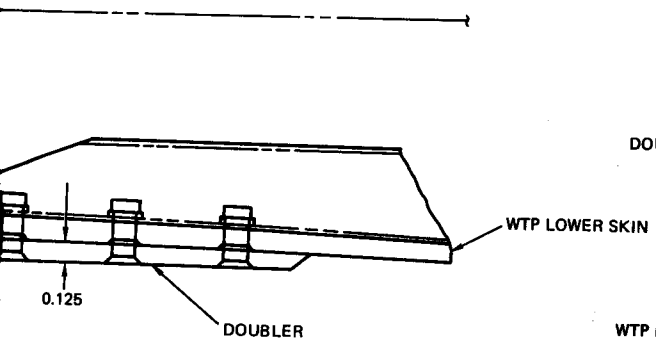
WTP FWD INTERM SPAR PLANE

VIEW LOOKING DOWN ON LEFT WING TIP PANEL

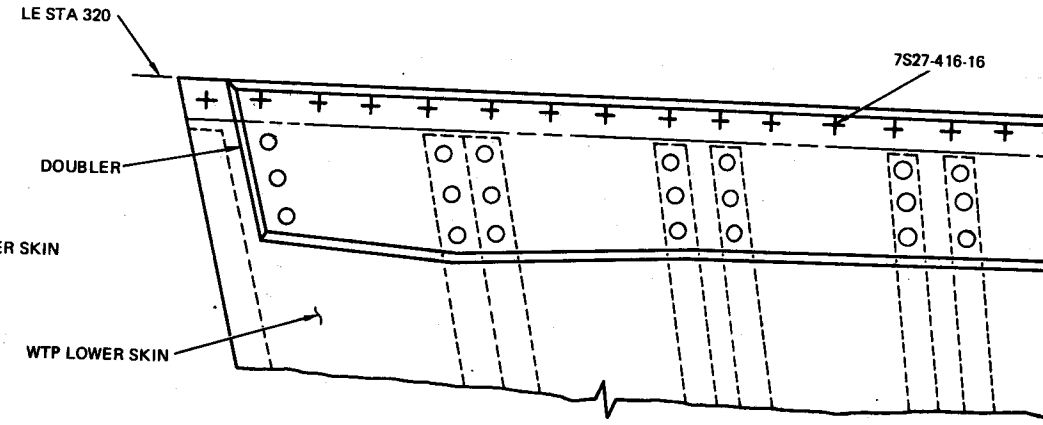




SECTION **A-A**

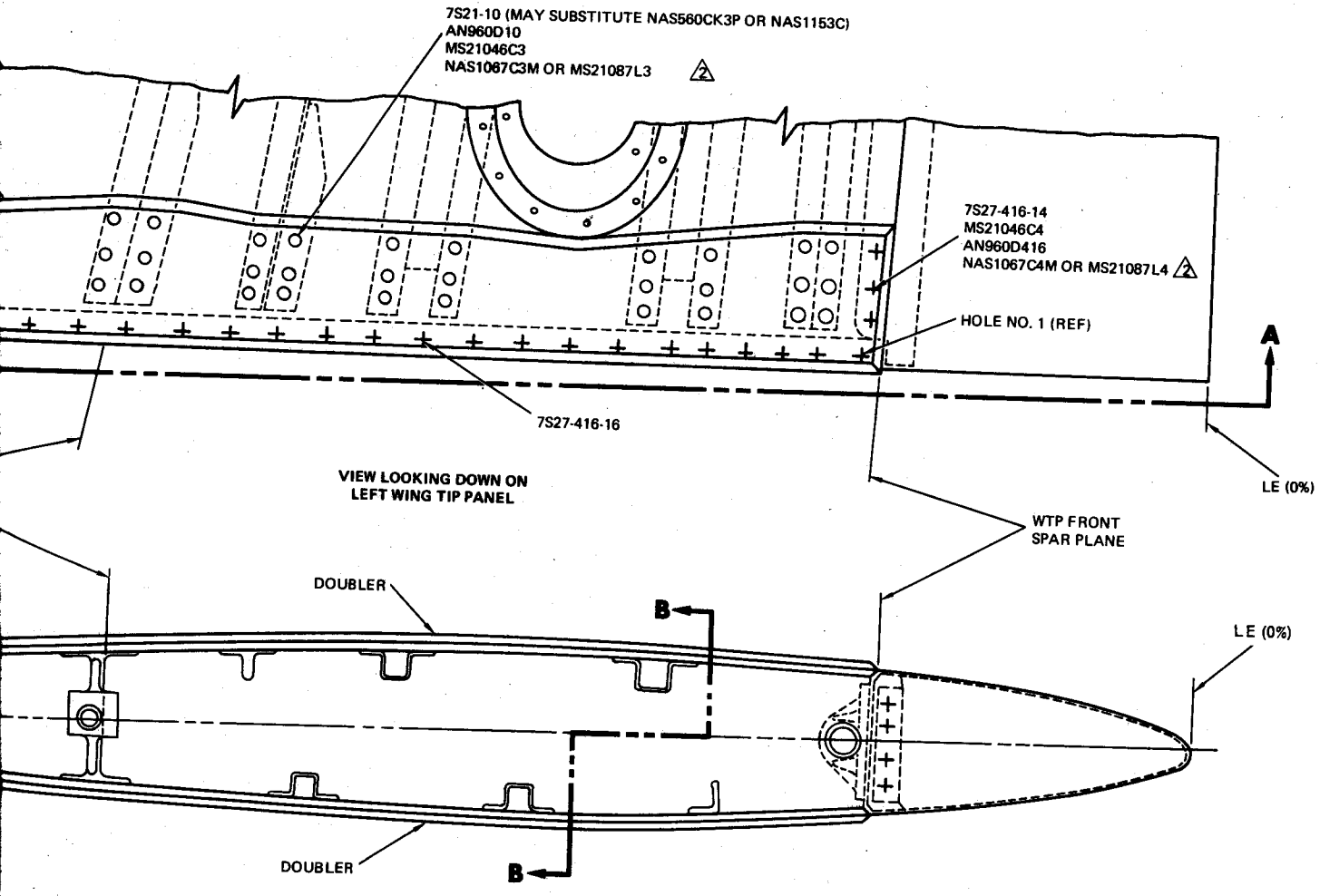


SECTION **B-B**



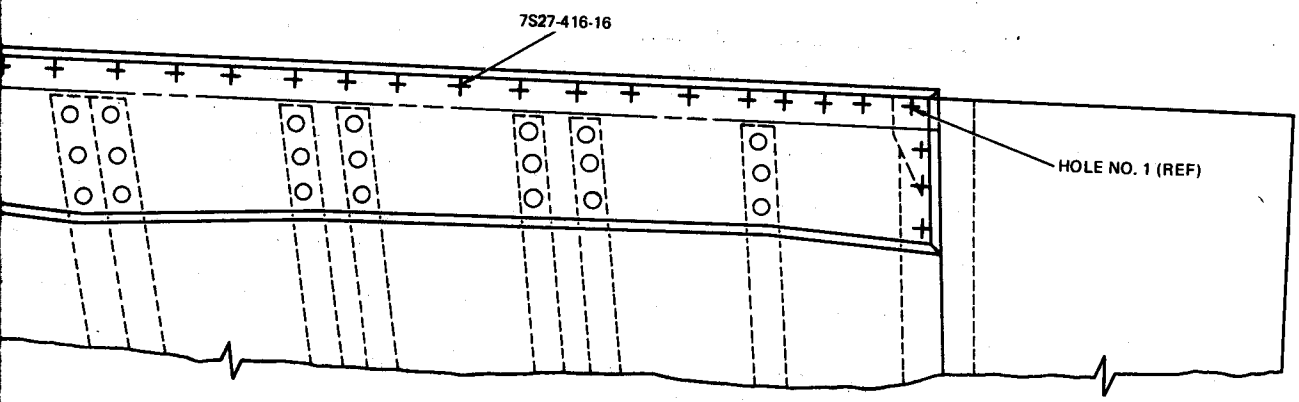
CAUTION IF DOUBLER EXTENDS TO TRAILING EDGE OF WTP INSURE THAT AILERON TRAVEL IS NOT RESTRICTED BY REPAIR FASTENERS

VIEW LOOKING UP AT LEFT WING TIP PANEL LOWER SURFACE



VIEW LOOKING DOWN ON
LEFT WING TIP PANEL

SECTION **A-A**

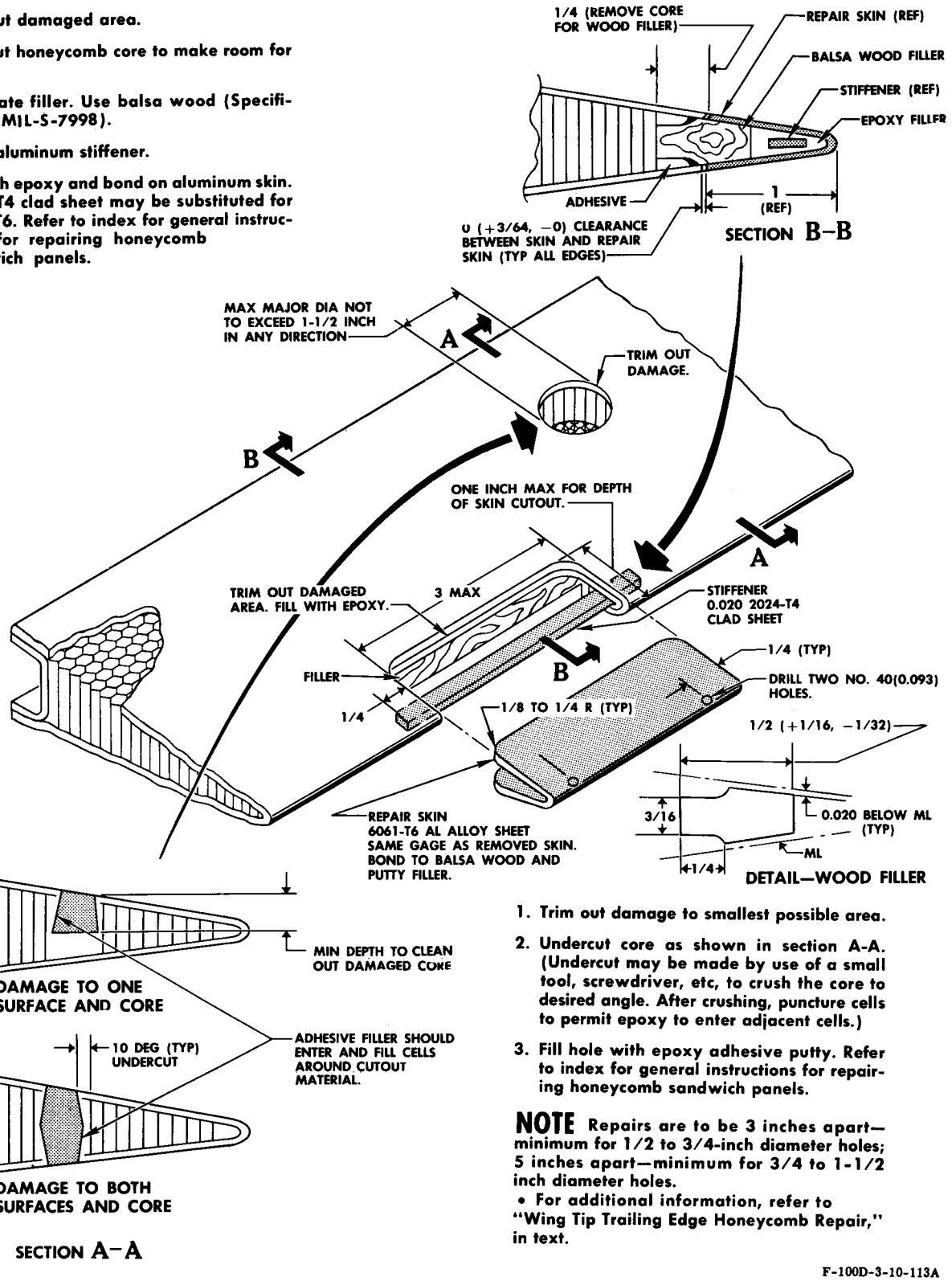


VIEW LOOKING UP AT LEFT
WING TIP PANEL LOWER SURFACE

PERMANENT—CATEGORY A

WING

1. Trim out damaged area.
2. Trim out honeycomb core to make room for filler.
3. Fabricate filler. Use balsa wood (Specification MIL-S-7998).
4. Insert aluminum stiffener.
5. Fill with epoxy and bond on aluminum skin. 2024-T4 clad sheet may be substituted for 6061-T6. Refer to index for general instructions for repairing honeycomb sandwich panels.



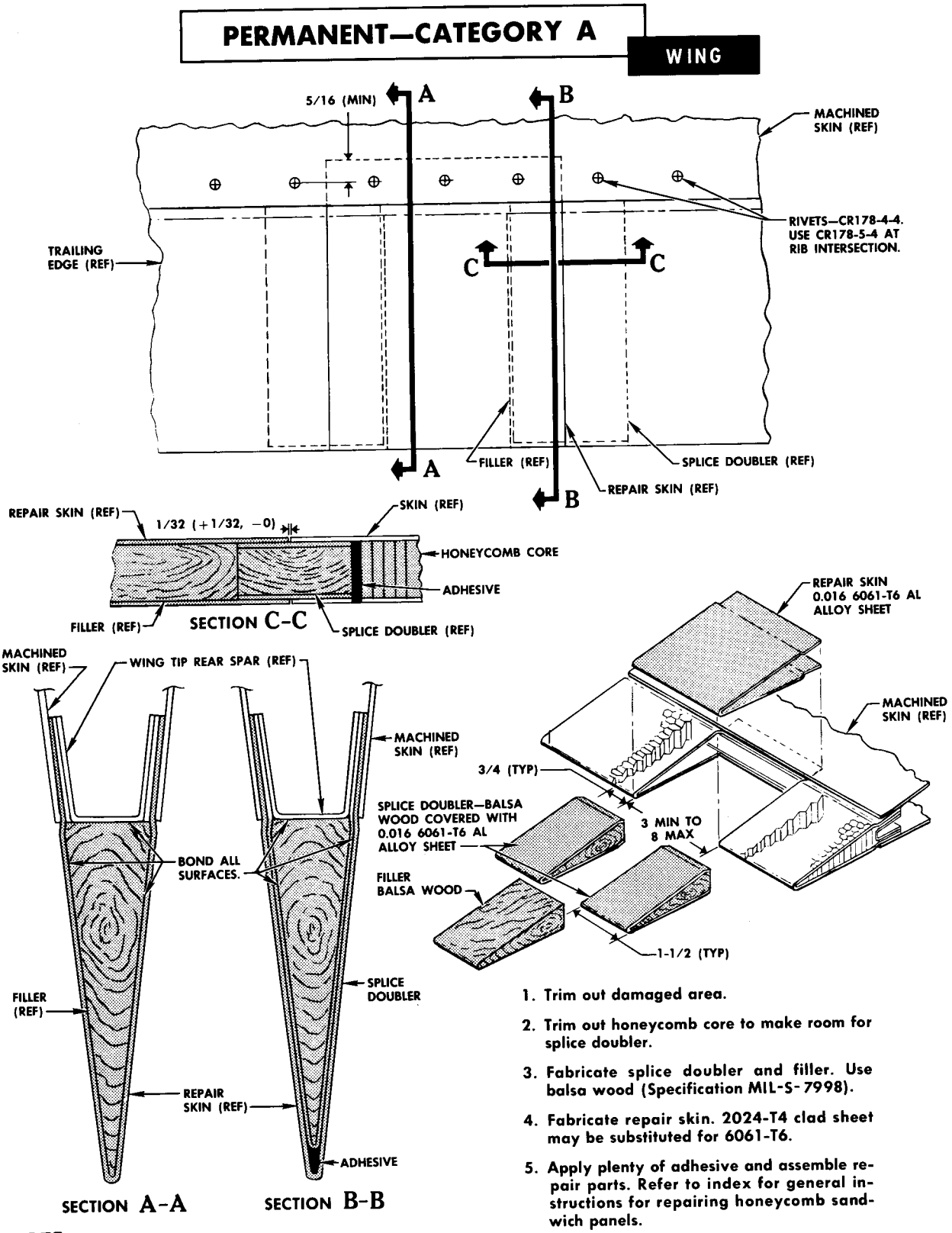
1. Trim out damage to smallest possible area.
2. Undercut core as shown in section A-A. (Undercut may be made by use of a small tool, screwdriver, etc, to crush the core to desired angle. After crushing, puncture cells to permit epoxy to enter adjacent cells.)
3. Fill hole with epoxy adhesive putty. Refer to index for general instructions for repairing honeycomb sandwich panels.

NOTE Repairs are to be 3 inches apart—minimum for 1/2 to 3/4-inch diameter holes; 5 inches apart—minimum for 3/4 to 1-1/2 inch diameter holes.

- For additional information, refer to "Wing Tip Trailing Edge Honeycomb Repair," in text.

F-100D-3-10-113A

Figure 2-60. Nonstructural Repair of Honeycomb Section

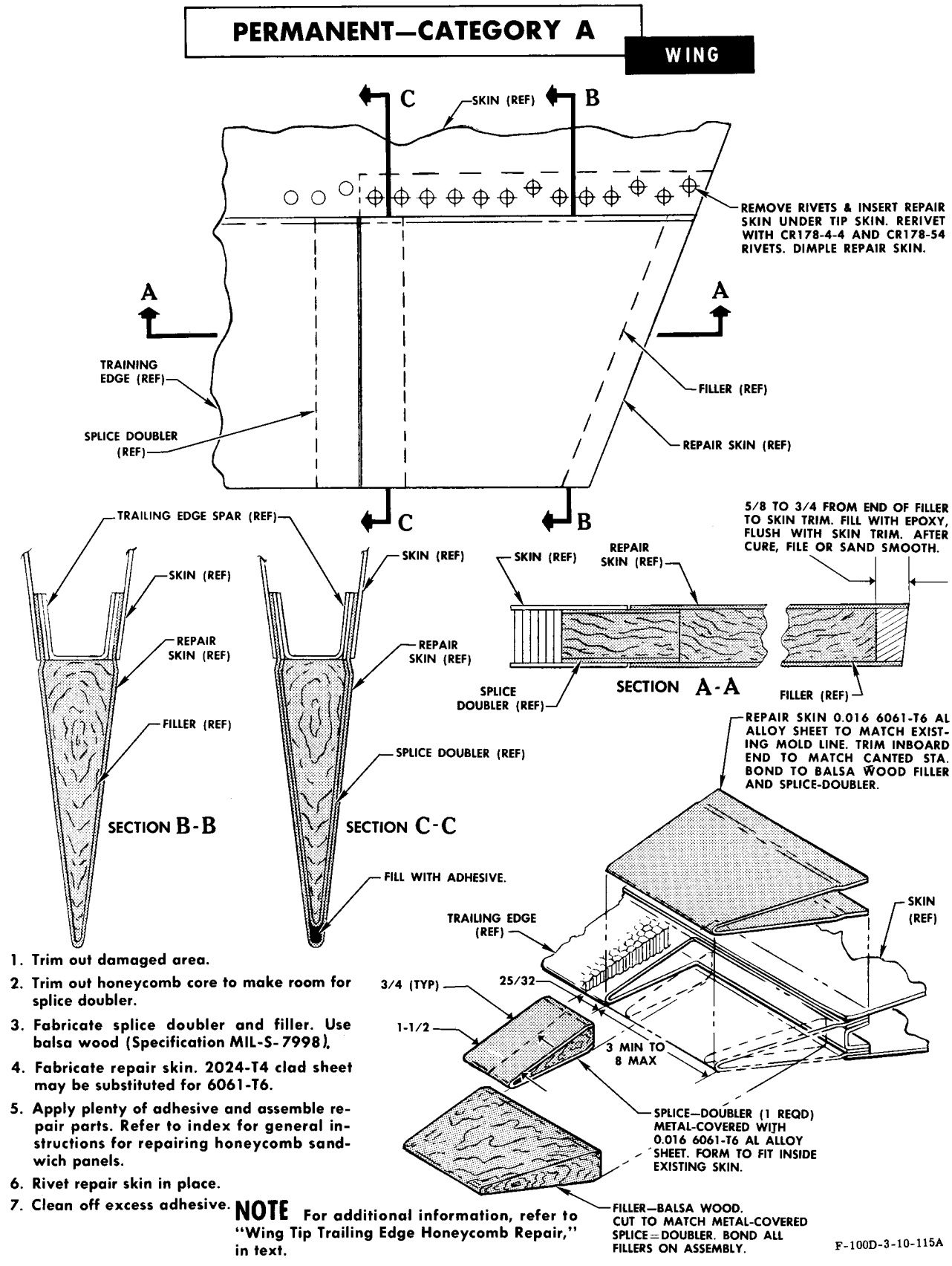


NOTE For additional information, refer to "Wing Tip Trailing Edge Honeycomb Repair," in text.

1. Trim out damaged area.
2. Trim out honeycomb core to make room for splice doubler.
3. Fabricate splice doubler and filler. Use balsa wood (Specification MIL-S-7998).
4. Fabricate repair skin. 2024-T4 clad sheet may be substituted for 6061-T6.
5. Apply plenty of adhesive and assemble repair parts. Refer to index for general instructions for repairing honeycomb sandwich panels.
6. Rivet repair skin in place.
7. Clean off excess adhesive.

F-100D-3-10-114A

Figure 2-61. Wing Tip Trailing Edge—Honeycomb—Center Repair



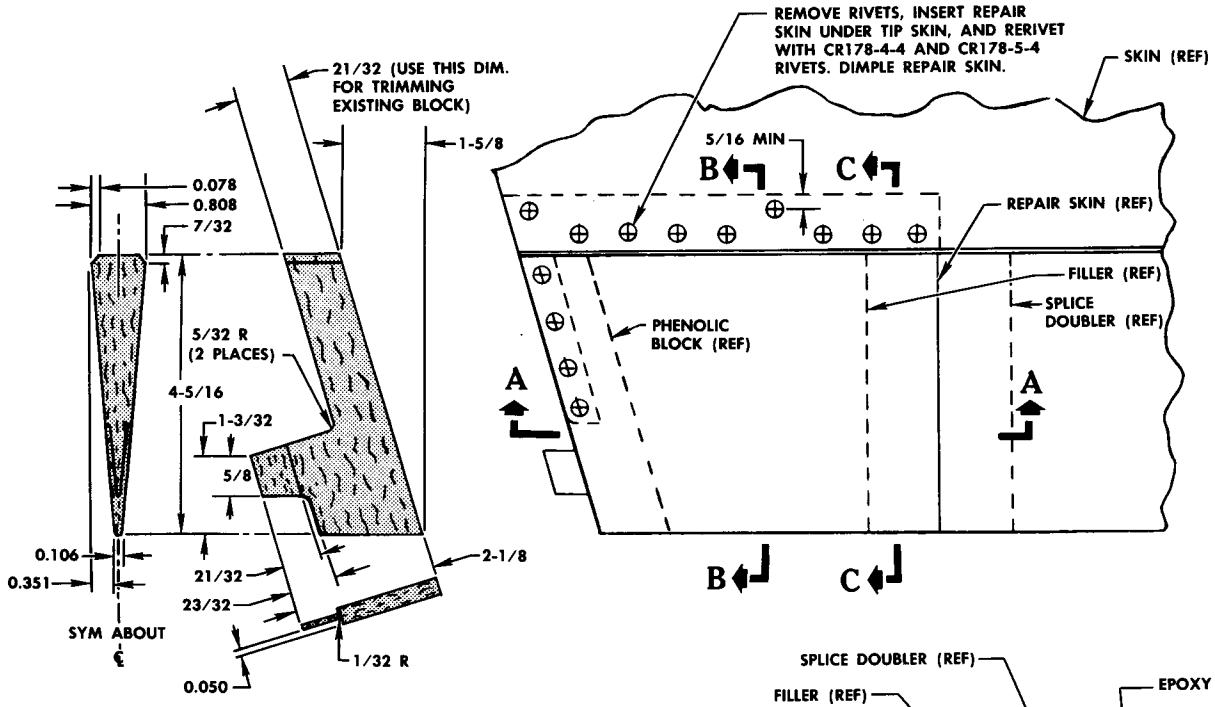
1. Trim out damaged area.
2. Trim out honeycomb core to make room for splice doubler.
3. Fabricate splice doubler and filler. Use balsa wood (Specification MIL-S-7998).
4. Fabricate repair skin. 2024-T4 clad sheet may be substituted for 6061-T6.
5. Apply plenty of adhesive and assemble repair parts. Refer to index for general instructions for repairing honeycomb sandwich panels.
6. Rivet repair skin in place.
7. Clean off excess adhesive.

NOTE For additional information, refer to "Wing Tip Trailing Edge Honeycomb Repair," in text.

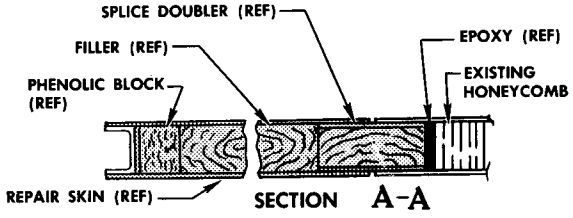
Figure 2-62. Wing Tip Trailing Edge—Honeycomb—Inboard Repair

PERMANENT CATEGORY A

WING



DETAIL-PHENOLIC BLOCK



SECTION A-A

1 Trim out damaged area to aft spar face, and cut out honeycomb core.

2 Fabricate splice doubler, filler, and repair skin. Use balsa wood (Specification MIL-S-7998).

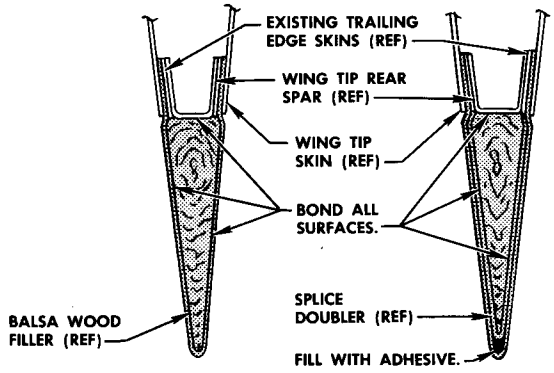
3 Apply plenty of adhesive, and assemble repair parts. Refer to index for general instructions for repairing honeycomb sandwich panels.

4 Rivet repair skin in place.

5 Clean off excess adhesive.

NOTE For additional information, refer to text.

- 2024-T4 clad sheet may be substituted for 6061-T6.



SECTION B-B SECTION C-C

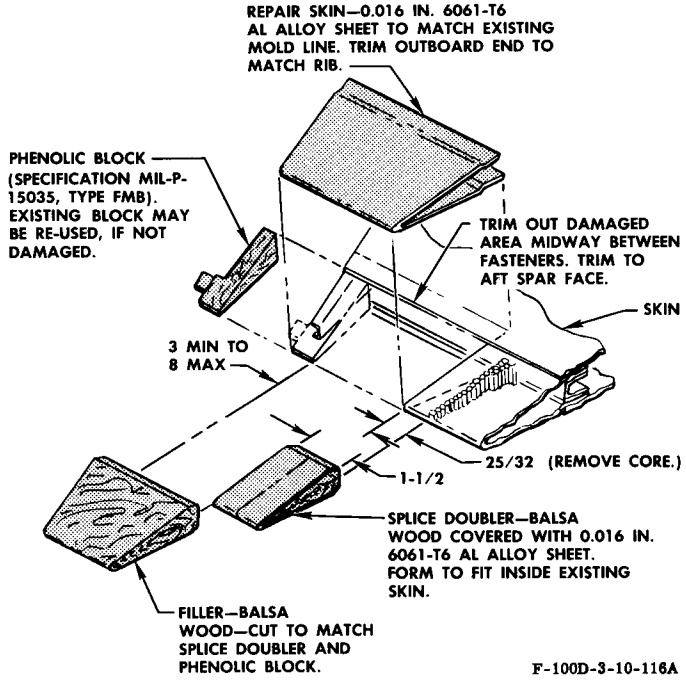
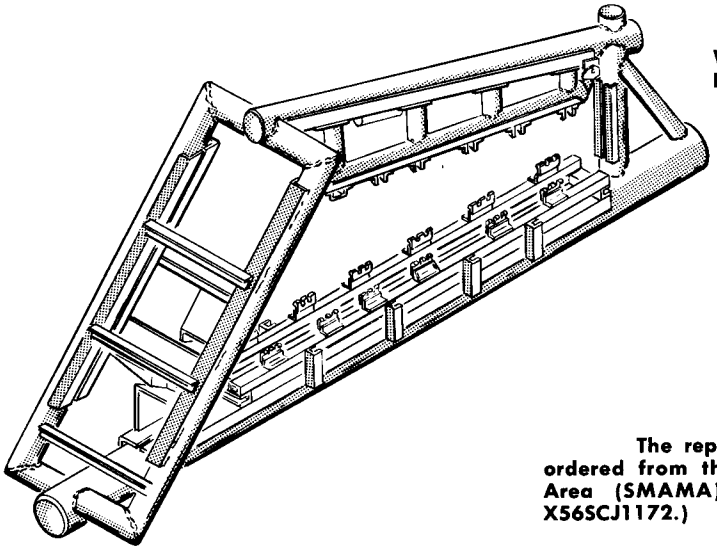


Figure 2-63. Wing Tip Trailing Edge—Honeycomb—Outboard Repair

REPAIR JIG

WING



WING
DRAWING NO. X56SCJ1172

The repair jig for the wing may be ordered from the Sacramento Air Materiel Area (SMAMA). (Refer to Drawing No. X56SCJ1172.)

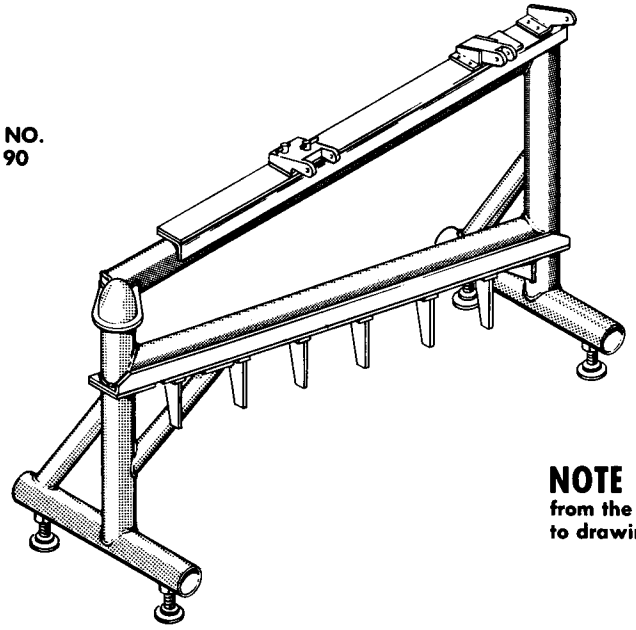
F-100D-3-10-137

Figure 2-64. Wing Repair Jig

FLAP REPAIR JIG

WING

DRAWING NO.
X-56SCJ1190



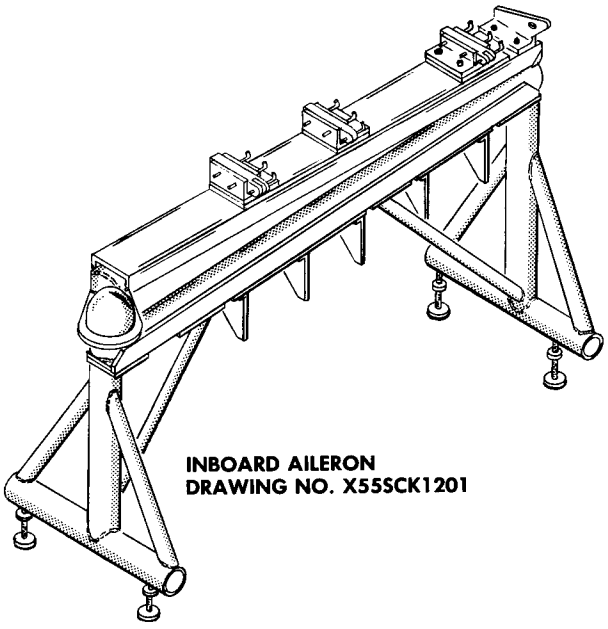
NOTE The repair jig for the flaps may be ordered from the Sacramento Air Materiel Area (SMAMA). Refer to drawing number shown.

F-100D-3-10-136

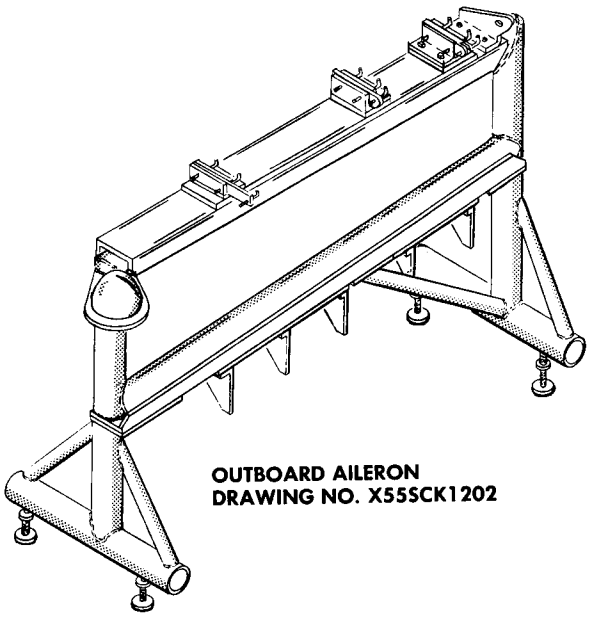
Figure 2-65. Flap Repair Jig

AILERON REPAIR JIG

WING



INBOARD AILERON
DRAWING NO. X55SCK1201



OUTBOARD AILERON
DRAWING NO. X55SCK1202

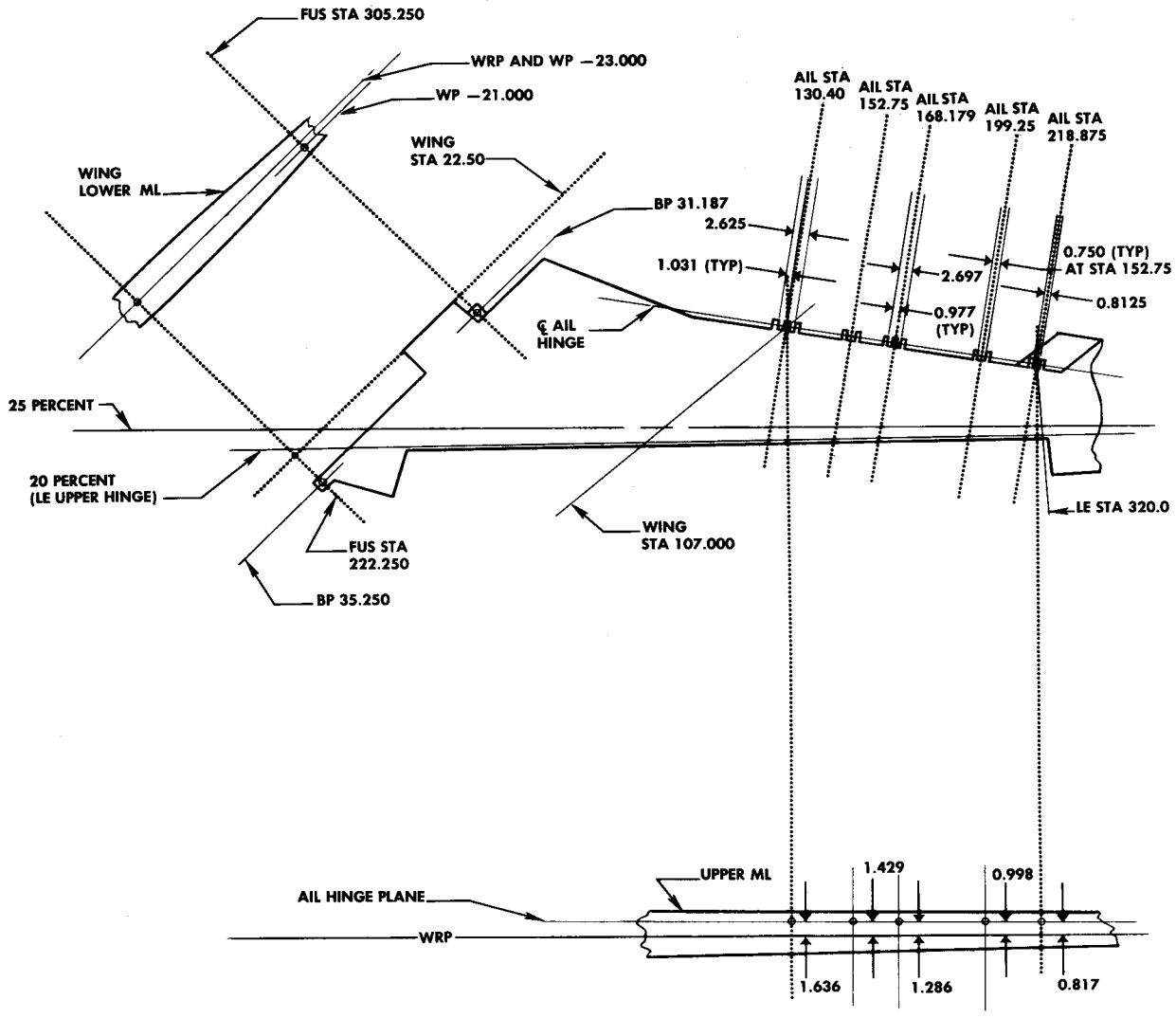
NOTE The repair jigs for the inboard and outboard ailerons may be ordered from the Sacramento Air Materiel Area (SMAMA). (Refer to drawing numbers shown.)

F-100D-3-10-140

Figure 2-66. Aileron Repair Jigs

HOLDING FIXTURE DIMENSIONS

WING



LEGEND

WRP—WING REFERENCE PLANE

WP— WATER PLANE

BP— BUTTOCK PLANE

F-100D-3-10-167

Figure 2-67. Holding Fixture Dimensions—Wing

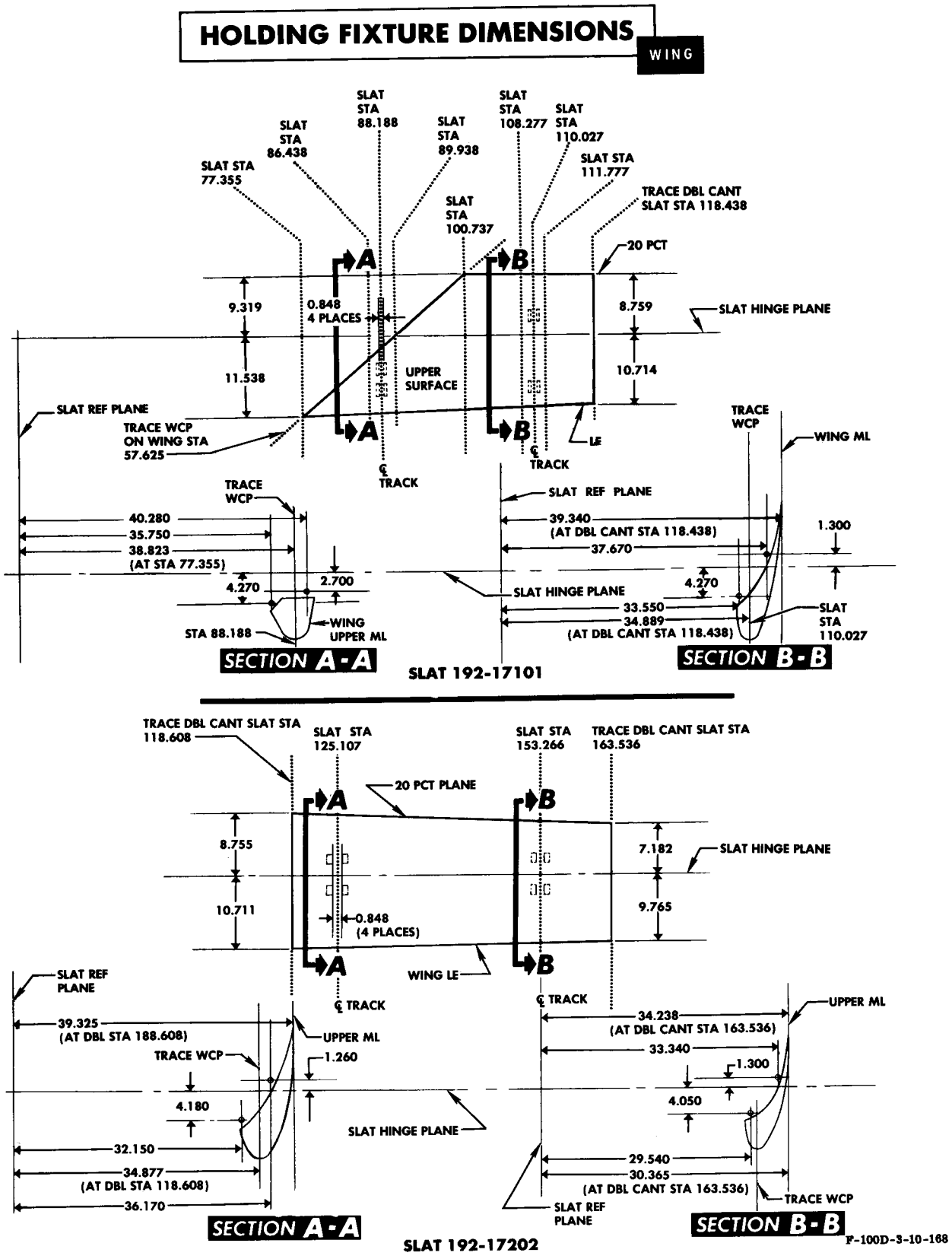
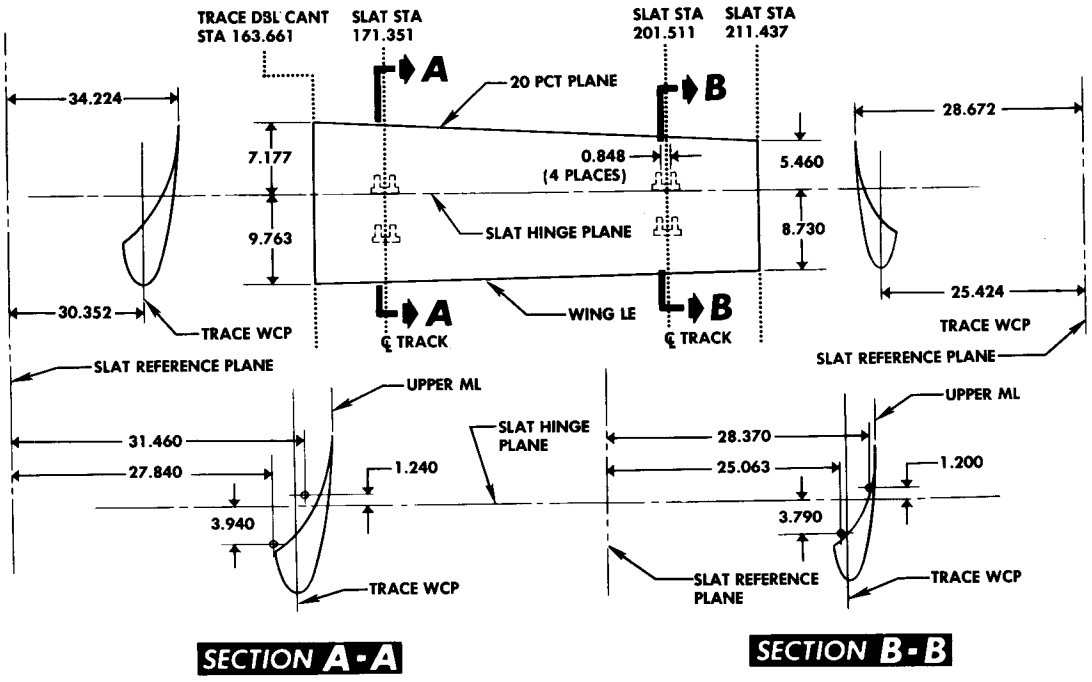


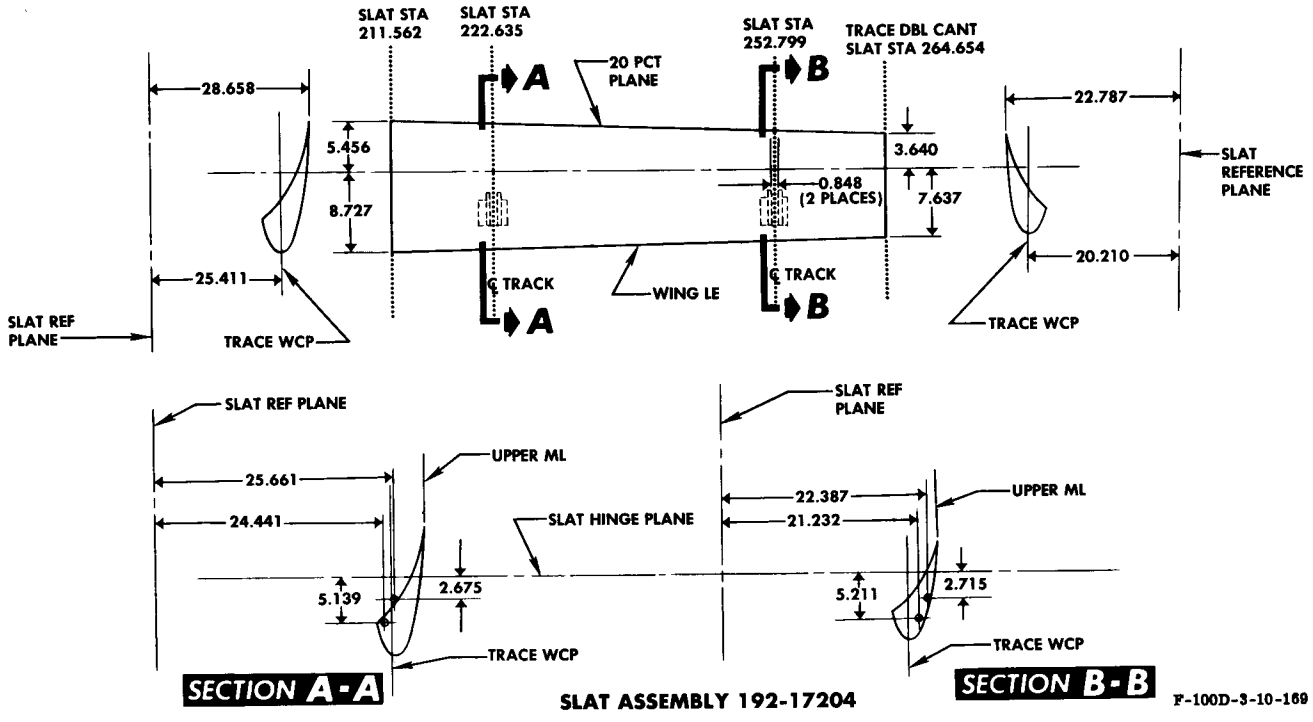
Figure 2-68. Holding Fixture Dimensions—Slats (Sheet 1 of 3)

HOLDING FIXTURE DIMENSIONS

WING



SLAT ASSEMBLY—192-17203

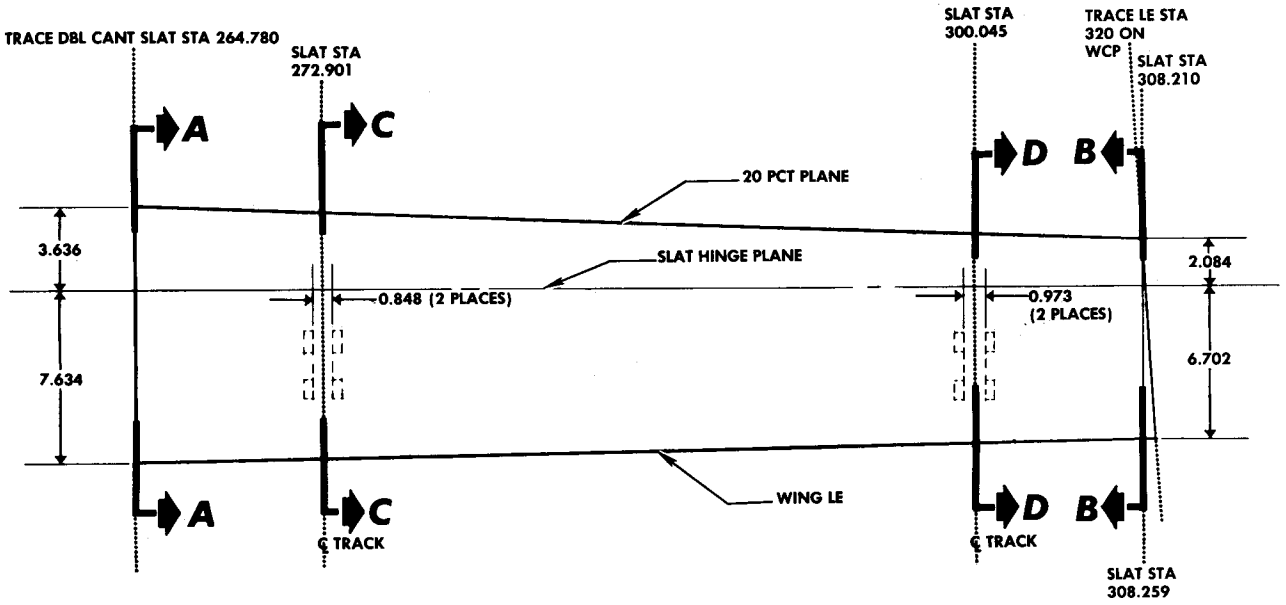


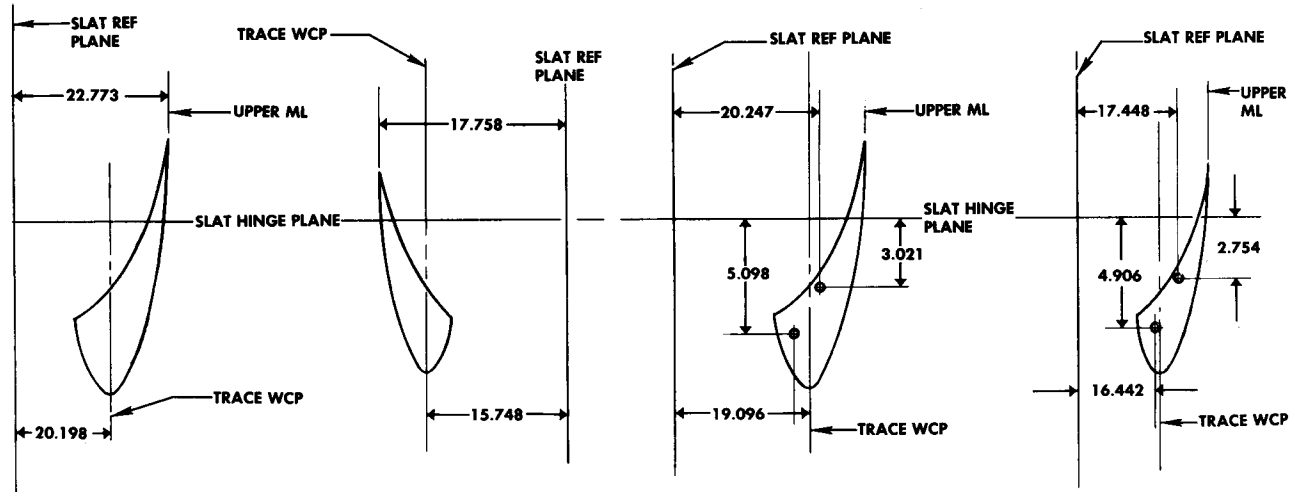
SLAT ASSEMBLY 192-17204

Figure 2-68. Holding Fixture Dimensions—Slats (Sheet 2 of 3)

HOLDING FIXTURE DIMENSIONS

WING





SECTION A-A DBL CANT STA 264.780

SECTION B-B STA 308.259

SECTION C-C STA 272.901

SECTION D-D STA 300.045

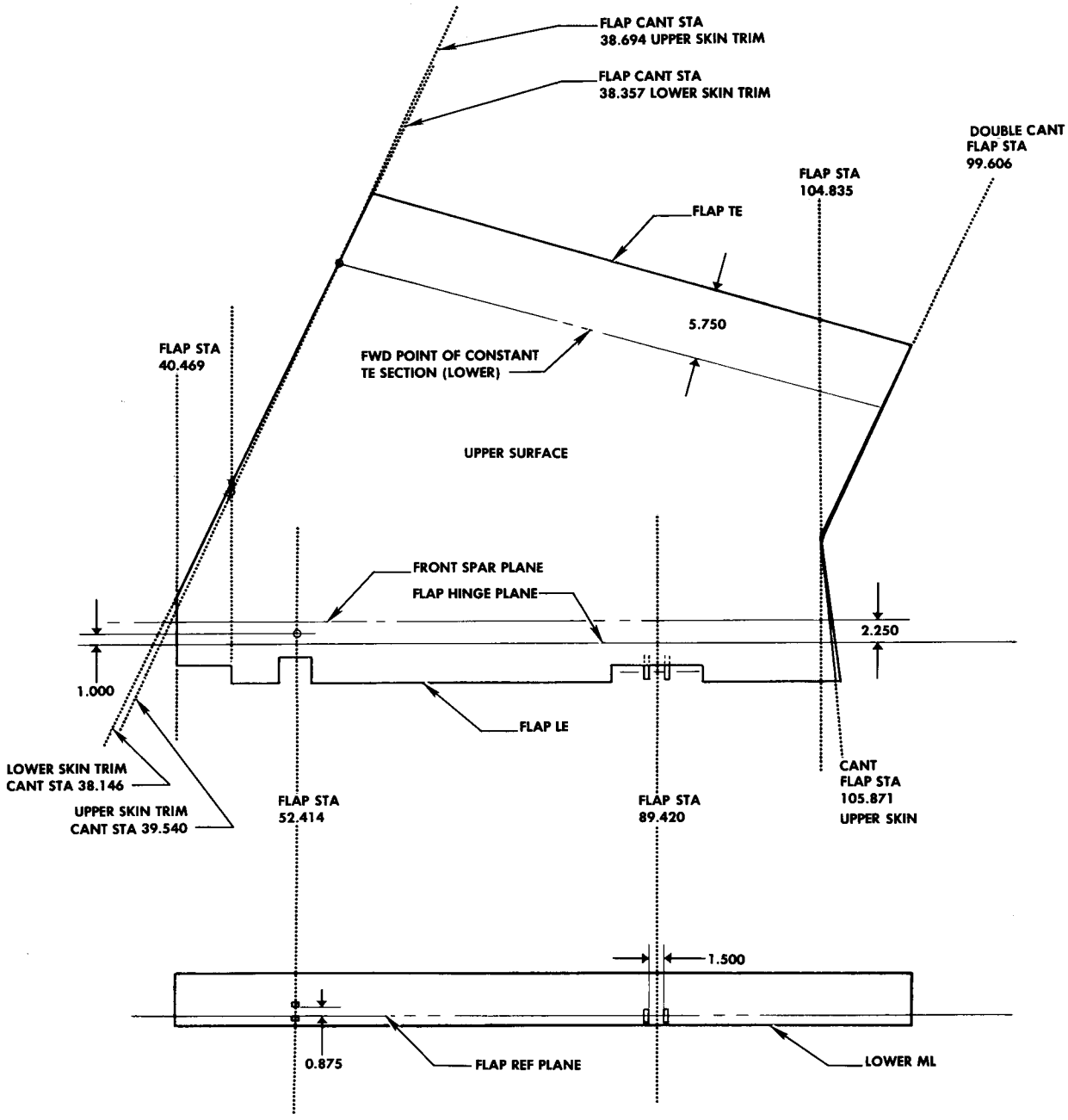
SLAT ASSEMBLY 192-17205

F-100D-3-10-170

Figure 2-68. Holding Fixture Dimensions—Slats (Sheet 3 of 3)

HOLDING FIXTURE DIMENSIONS

WING



F-100D-3-10-171

Figure 2-69. Holding Fixture Dimensions—Flap

HOLDING FIXTURE DIMENSIONS

WING

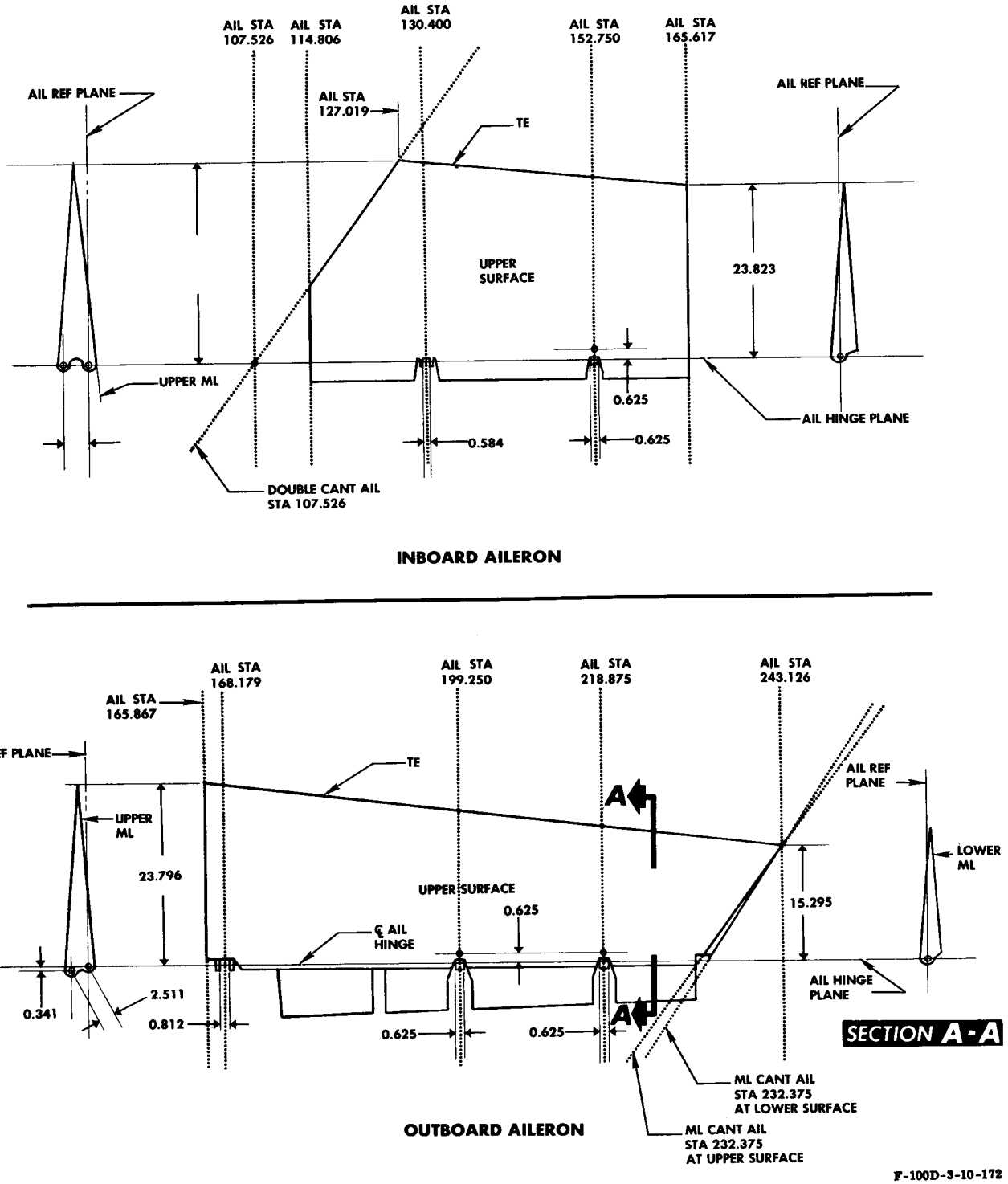
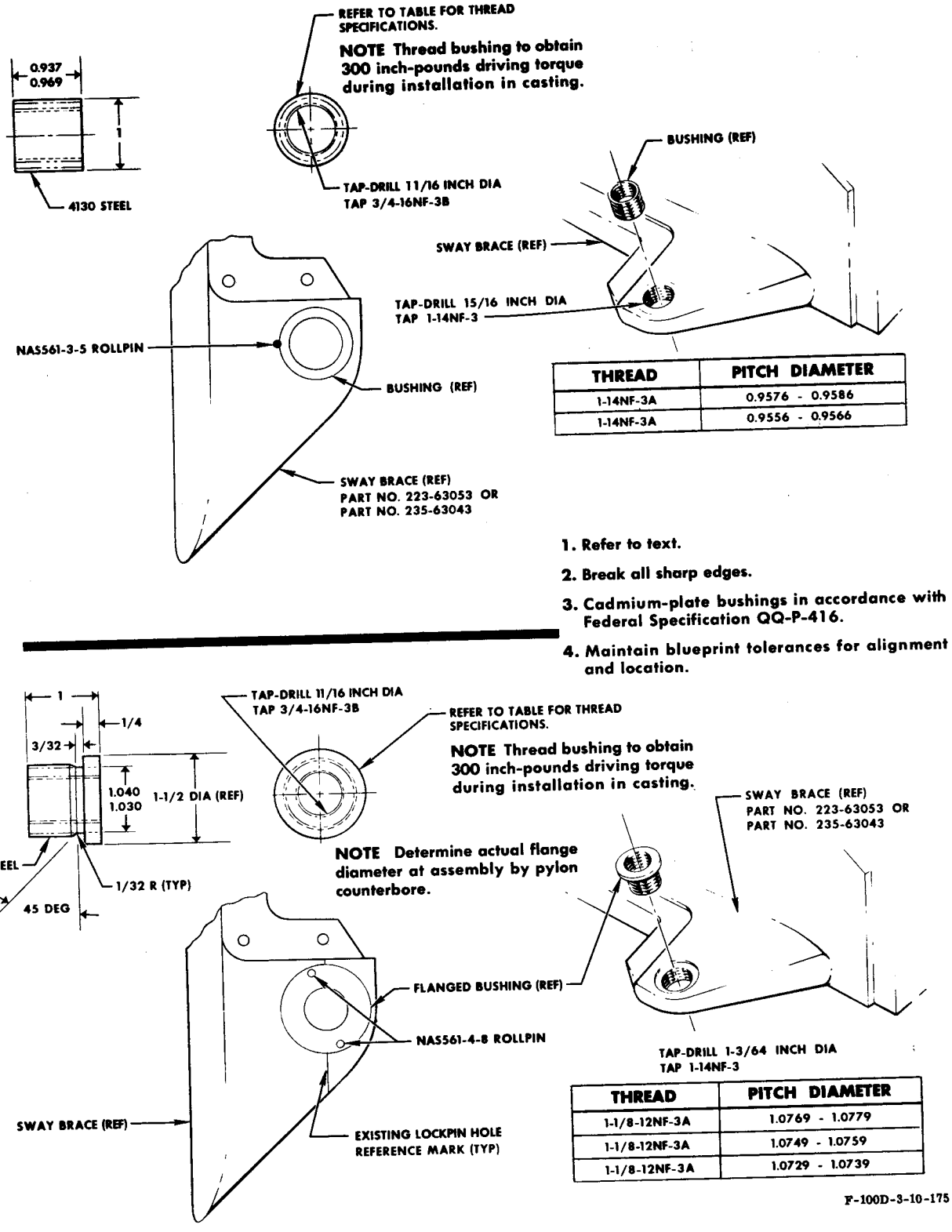


Figure 2-70. Holding Fixture Dimensions—Ailerons

PERMANENT CATEGORY B



THREAD	PITCH DIAMETER
1-14NF-3A	0.9576 - 0.9586
1-14NF-3A	0.9556 - 0.9566

1. Refer to text.
2. Break all sharp edges.
3. Cadmium-plate bushings in accordance with Federal Specification QQ-P-416.
4. Maintain blueprint tolerances for alignment and location.

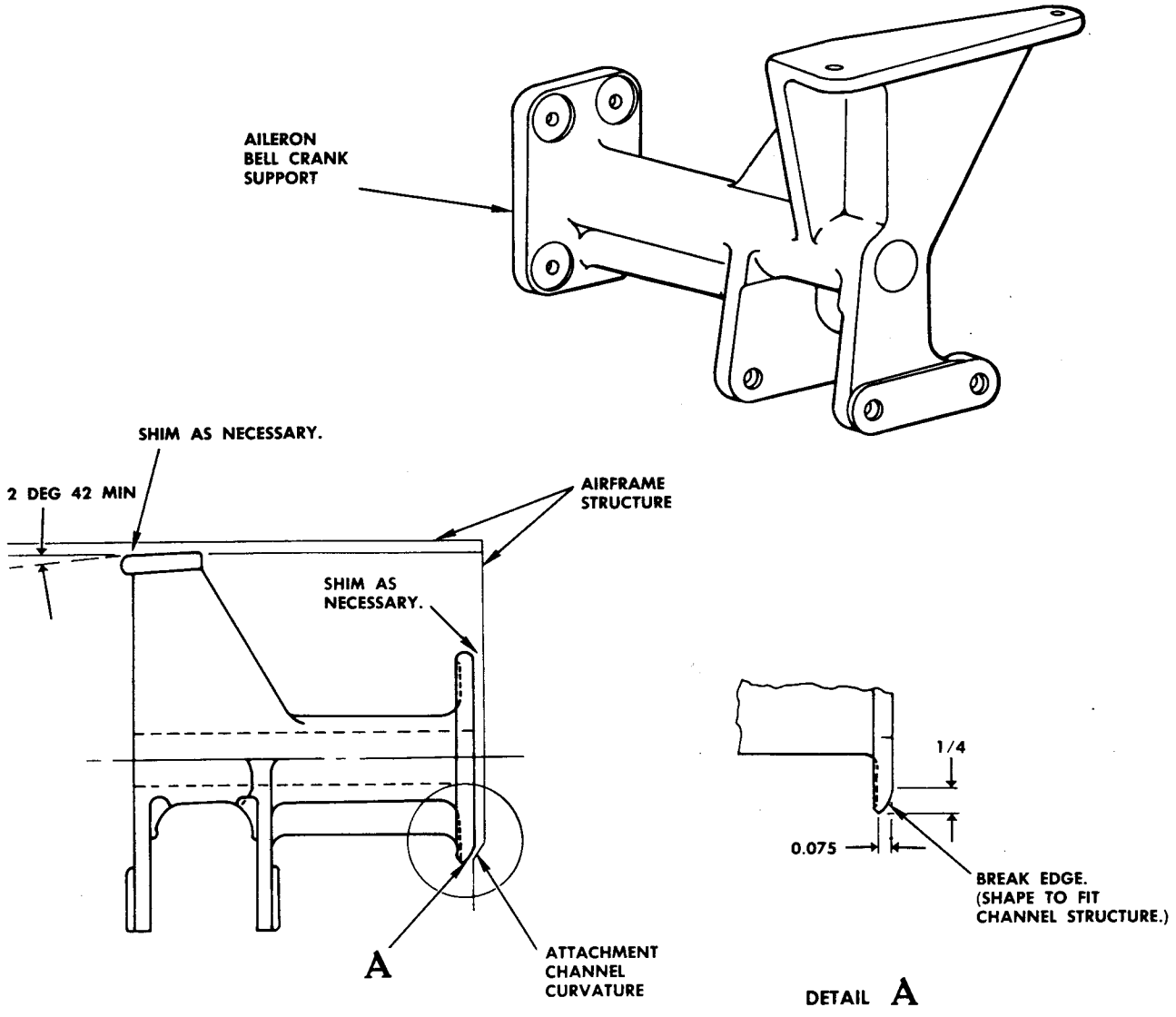
THREAD	PITCH DIAMETER
1-1/8-12NF-3A	1.0769 - 1.0779
1-1/8-12NF-3A	1.0749 - 1.0759
1-1/8-12NF-3A	1.0729 - 1.0739

F-100D-3-10-175

Figure 2-71. Repair of Sway-brace Heli-Coil Inserts—Type I and Type III Pylons

PERMANENT—CATEGORY A

FUSELAGE



NOTE If aileron bell crank supports (left and right link bay areas) do not properly fit the airframe structural member, alignment may be obtained by the following rework.

1 Radius lower portion of support aft flange sufficiently to provide clearance at attachment channel curvature.

2 Install attaching bolts snugly; then check for alignment of support with airframe structural member. Shim as required.

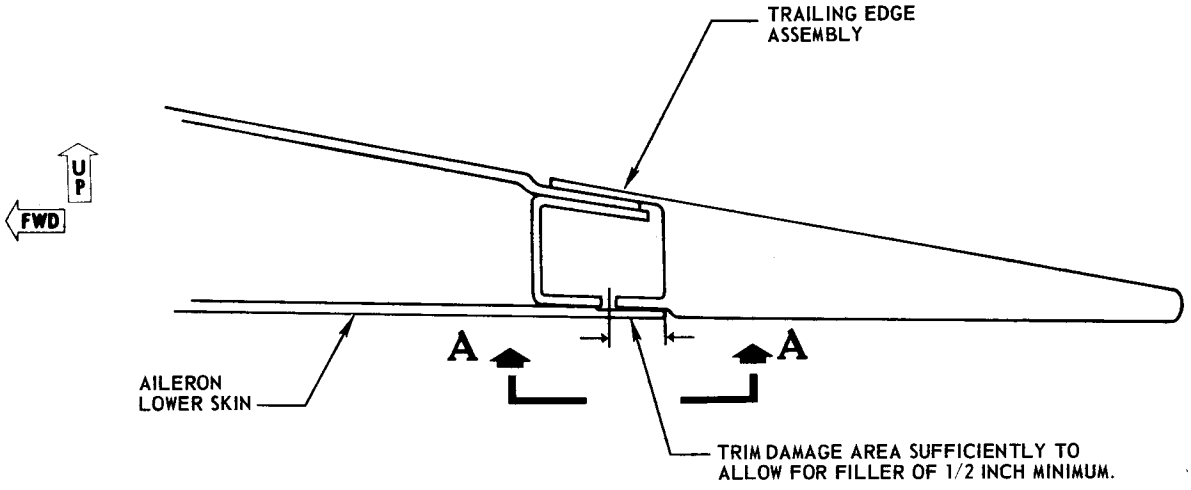
3 Elongate holes in airframe structural member (in the fore and aft direction only), if necessary, to achieve alignment. Maintain 0.050-inch minimum edge distance on hole in attachment angle.

100D-3-10-197

Figure 2-72. Rework of Aileron Bell Crank Support

PERMANENT—CATEGORY A

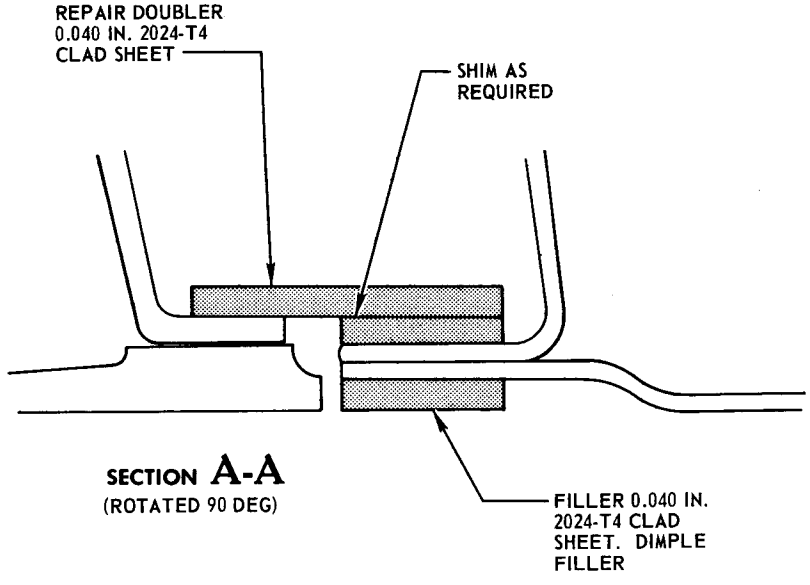
WING



1. Trim damaged area.
2. Fabricate doubler, filler and shim.
3. Assemble repair and install fasteners.

NOTE Fasteners to be same type and diameter as original (CR178). Fasteners to use same pattern as original, maintaining a minimum of 2D edge distance on filler, shim and doubler.

4. Balance aileron. (Refer to Section I).



100A-3-10-159

Figure 2-73. Aileron and Flap Trailing Edge Attachment Repair

Section III

EMPENNAGE GROUP

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DESCRIPTION OF EMPENNAGE GROUP.

HORIZONTAL STABILIZER.

The horizontal stabilizer is a one-piece, all-movable surface which is attached to the airplane by two pivot fittings at fuselage station 485 and water plane -22. The two pivot fittings are equidistant from the centerline of the airplane at buttock plane 25. The angle of incidence of the horizontal stabilizer leading edge is controlled between 7 (+½, −¼) degrees up and 25 (+½, −0) degrees down by a hydraulically operated cylinder.

The horizontal stabilizer is made of two outer panels and a center section beam. The outer panels are swept back at an angle of 45 degrees at the 25-percent line and are interchangeable; each panel may be used as either a right or left surface. The outer panel is a three-cell aluminum alloy structure, with two beams interconnected by closely spaced pressed-type ribs of 2024-T4 clad material.

The main structure box (center cell) consists of the two beams and two integrally stiffened skins that are machined from 7075-T6 aluminum alloy bare plate. At the junction of stiffeners and ribs, the stiffeners are trimmed away so that the ribs may be continuous. Beads are formed in the ribs at these points to provide a rigid tie between the two skins.

Both the front and rear beams are machined from 2014-T6 aluminum alloy "Z" section extrusion and are continuous from the tip section to the root section. At the root section, the shear in the beams is transferred to the center section tube by two machined fittings. The aft beam fitting is 2024-T42 aluminum alloy bar, and the front beam fitting is a 2014-T6 die forging. The forward torque box is provided by the curved leading edge skin which is formed by 0.040-inch 2024-T3 clad sheet and an external 0.016-inch corrosion-resistant steel doubler. The aft torque box is provided by two 0.032-inch 2024-T3 clad sheets and a trailing edge strip machined out of magnesium alloy bar. The front

and rear torque boxes are attached to the main structural box by tabs machined on the main skins. A spanwise skin splice is made on the upper and lower surfaces to make assembly easy. The center section beam is a 4340 steel, flash-welded assembly. It has two end fittings machined from 4340 steel die forgings, a centerpiece with a lug for the hydraulic actuating cylinder (machined from a 4340 steel die forging), and two lengths of 4340 steel tube.

VERTICAL STABILIZER.

The vertical stabilizer is swept back at an angle of about 45 degrees at the 25 percent line. The vertical stabilizer is of all-metal construction with the exception of a laminated glass fabric tip which houses the AN/ARC-34 and AN/APW11A command radio antennas. Above the rudder and aft of the trailing edge of the vertical stabilizer are two position lights, a fuel vent outlet, and the AN/APS-54 antenna. Fairings are used to streamline the vertical stabilizer with the fuselage. The aft section of the vertical stabilizer fairing houses the drag chute support structure and jettison mechanism. The metal section of the vertical stabilizer is a three-cell aluminum structure with two beams interconnected by closely spaced, pressed-type ribs of 2024-T4 clad material. These ribs are continued forward of the front beam to stiffen the leading edge skin, which is made of 0.040-inch 2024-T3 clad material. The box section above the rudder is formed by the rear beam, and the side skins. The main structural box is formed by the front and rear beams, ribs, and two integrally stiffened skins that are machined from 7075-T6 aluminum plate. The ribs are beaded to form a more rigid tie between the two skins. A large tab is machined on the lower edge of each skin. The tab mates with a fuselage fitting to form a joint which can transfer shear and spanwise bending. The attachment is made with six special steel pins. The rear beam is made of 7075-T channel sections.

At vertical stabilizer station 70.5, the rear beam is interrupted to allow passage of the fuel vent tube. The gap between the upper and lower portions of the rear beam is bridged by heavy lands that are machined on the skins in the region of the gap. The vertical stabilizer-to-fuselage aft attachment point is at the root of the rear beam. This tie can transfer shear loads only. The vertical stabilizer tip, which houses the antenna, is made of glass fabric laminate and plastic honeycomb.

RUDDER.

The hydraulically operated splitter-plate type rudder is divided into two sections, upper and lower, with the break at rudder station 25.75. The upper section is attached to the vertical stabilizer with two hinges, one at the upper and one at the lower end of the section. The lower rudder section is also supported at two points. The lower hinge is at the lower end of the rudder torque tube. The upper end of the lower section is hinged to the upper section to provide support for the lower section and to transmit airload torque from the upper rudder section to the lower section. From the root rib to the upper end of the lower section, the structure is a one-cell torque box formed by the leading edge beam, the rear beam, and the side skins. Pressed-type ribs are spaced as needed to stiffen the skin in this area. The bending in the lower section is carried by the leading edge, which is a 356-T6 aluminum alloy permanent-mold casting. Between the root rib and the lower hinge, the bending and torsion are carried by a 4130 normalized steel tube. The upper rudder is also a one-cell torque box, except for the lower three bays, in which there are two torque boxes to accommodate a flutter damper. Bending is carried by the leading edge beam, which is made from a 356-T6 aluminum alloy permanent-mold casting. The side skins are stiffened by closely spaced pressed-type ribs. The skins, ribs, and intercostals are made from 2024-T4 clad material. The aft 8 inches of the rudder chord is a flat plate with integral stiffeners machined into it. This plate, called a splitter plate, is used to reduce flutter. The rear beam is also an integral part of the splitter plate. The splitter plate is machined from 2024-T4 bare aluminum alloy plate.

PRIMARY AND SECONDARY STRUCTURE.

The primary structure in the horizontal stabilizer is all the structure except the last inch of the trailing edge and the horizontal stabilizer tip. The box-type construction of the stabilizer generally distributes the load throughout the entire member. The primary structure in the vertical stabilizer is all the structure forward of the trailing edge beam with the exception of the tip and overboard fuel vent fairing.

The vertical stabilizer is of the same type construction as the horizontal stabilizer and, therefore, has the same load distribution characteristics. The primary structure in the rudder is all of the structure forward of the rudder splitter plate.

ONE-TIME FLIGHT.

Under one-time flight damage conditions, the airplane must be restricted to a maximum of 300 knots airspeed and a G-load of +3 and −1. Figure 3-1 shows the

secondary structure area which may be damaged but which will not prevent the airplane from making a one-time flight without repairs. Figures 3-2 and 3-3 show horizontal and vertical stabilizer primary structure which may have limited damage but which will allow the airplane to be flown to repair facilities under restricted flight conditions. In figures 3-2 and 3-3, the perforations "A" are measured at the greatest dimension. The sum of all the "A's" in a single bay shall not exceed the "L DAMAGE INCHES" value in the graph in figure 3-2. For example: Assume damages "A1," "A2," and "A3" in the main box structure. The sum of the dimensions for these perforations is compared with the value on the graph for the area between stations 118.18 and 125.68; if the sum is less than the value of "L," which is 4¾ inches (taken from the graph), the damage is below the maximum allowed. Damage to the leading and trailing edges is evaluated by using their respective graphs.

When one-time flight repairs are made it must be remembered that permanent repairs will be made later in the same area. When possible, a fastener pattern should be used which will be the same as the permanent repair fastener pattern.

HORIZONTAL STABILIZER, VERTICAL STABILIZER, AND RUDDER SKINS.

Construction of the horizontal stabilizer, vertical stabilizer, and rudder makes it necessary to remove the outer skin if extensive repair is to be done to the internal structure.

REMOVING HORIZONTAL STABILIZER LEADING EDGE SKINS.

1. Remove horizontal stabilizer tip by removing screws and drilling out rivets in leading and trailing edges of tip.
2. Remove forward root fairing on the side of horizontal stabilizer that the leading edge close-out strip is on.
3. Remove leading edge close-out strip by drilling out attaching rivets. Start at horizontal root rib and work out to the tip.
4. Remove leading edge skin.

REMOVING HORIZONTAL STABILIZER TRAILING EDGE SKINS.

1. Remove horizontal stabilizer tip by removing screws and drilling out rivets in leading and trailing edges of tip.
2. Remove aft root fairing on the side of horizontal stabilizer that the trailing edge close-out skin is on.
3. Remove trailing edge close-out strip by drilling out attaching rivets. Start at horizontal root rib and work out to the tip.
4. Remove trailing edge skin.

REMOVING HORIZONTAL STABILIZER MAIN SKINS.

1. Remove horizontal stabilizer from airplane.
2. Remove horizontal stabilizer panel from center section beam.
3. Remove leading edge close-out skin.
4. Remove trailing edge close-out skin.
5. Drill out Jo-bolts from the main skin that is on the same side of the stabilizer as the trailing edge close-out skin.

Note

The main skin may be warmed up to 250°F to help in removing the rivets.

INSTALLING HORIZONTAL STABILIZER SKINS.

The horizontal stabilizer leading edge, main, and trailing edge skins are installed by assembling parts in the reverse order of that used in their disassembly for skin removal. When installing the skins, make sure the required close-tolerance holes are within specifications. Although a jig is desirable, it is possible to reassemble the old leading edge, trailing edge, and main skins without one. When installing new skins, a jig must be used.

REPLACING BLIND FASTENERS IN HORIZONTAL STABILIZER ASSEMBLIES.

Cherry blind rivets CR762 should be used to replace standard Cherry rivets that become loose in horizontal stabilizer assemblies, or when installing horizontal stabilizer skins. (Refer to "Blind Fasteners—CR762 Rivet" in Section I of this manual.)

REMOVING VERTICAL STABILIZER LEADING EDGE SKINS.

1. Remove vertical stabilizer tip.
2. Remove leading edge close-out skin by drilling out the attaching rivets. Start at the vertical stabilizer root rib and work out to the tip.
3. Remove leading edge skin by drilling out attaching rivets.

REMOVING VERTICAL STABILIZER TRAILING EDGE SKINS.

1. Remove vertical stabilizer tip.
2. Remove rudder.
3. Remove fuel overboard vent fairing.
4. Remove trailing edge skin by drilling out rivets attaching trailing edge skin to main skin aft edge.

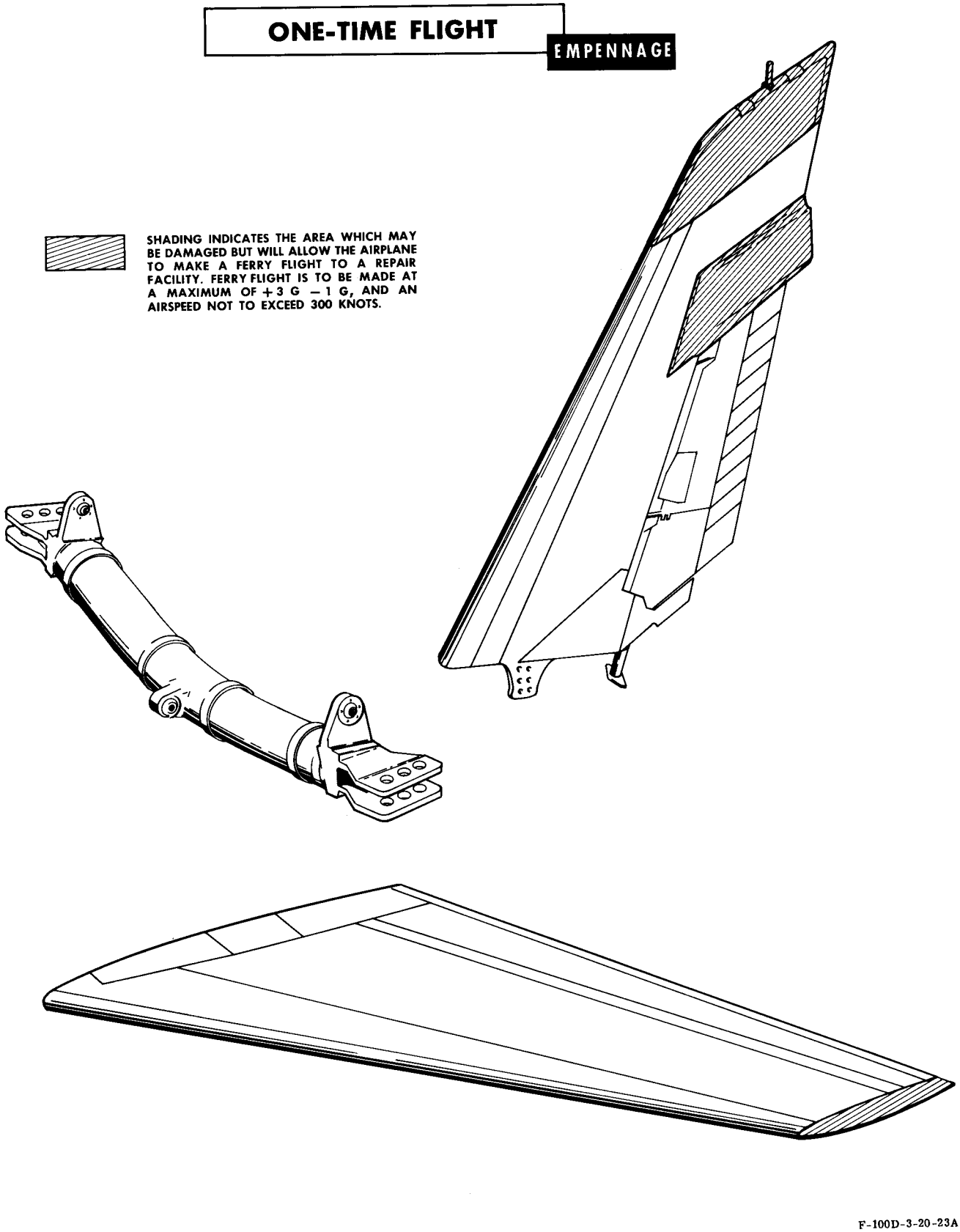

Figure 3-1. Empennage One-time Flight Damage

ONE-TIME FLIGHT DAMAGE

EMPENNAGE

1. Refer to text.
2. One-time flight damages which allow the airplane to make a flight to a repair facility at an airspeed of not more than 300 knots or + 3 G, − 1G.

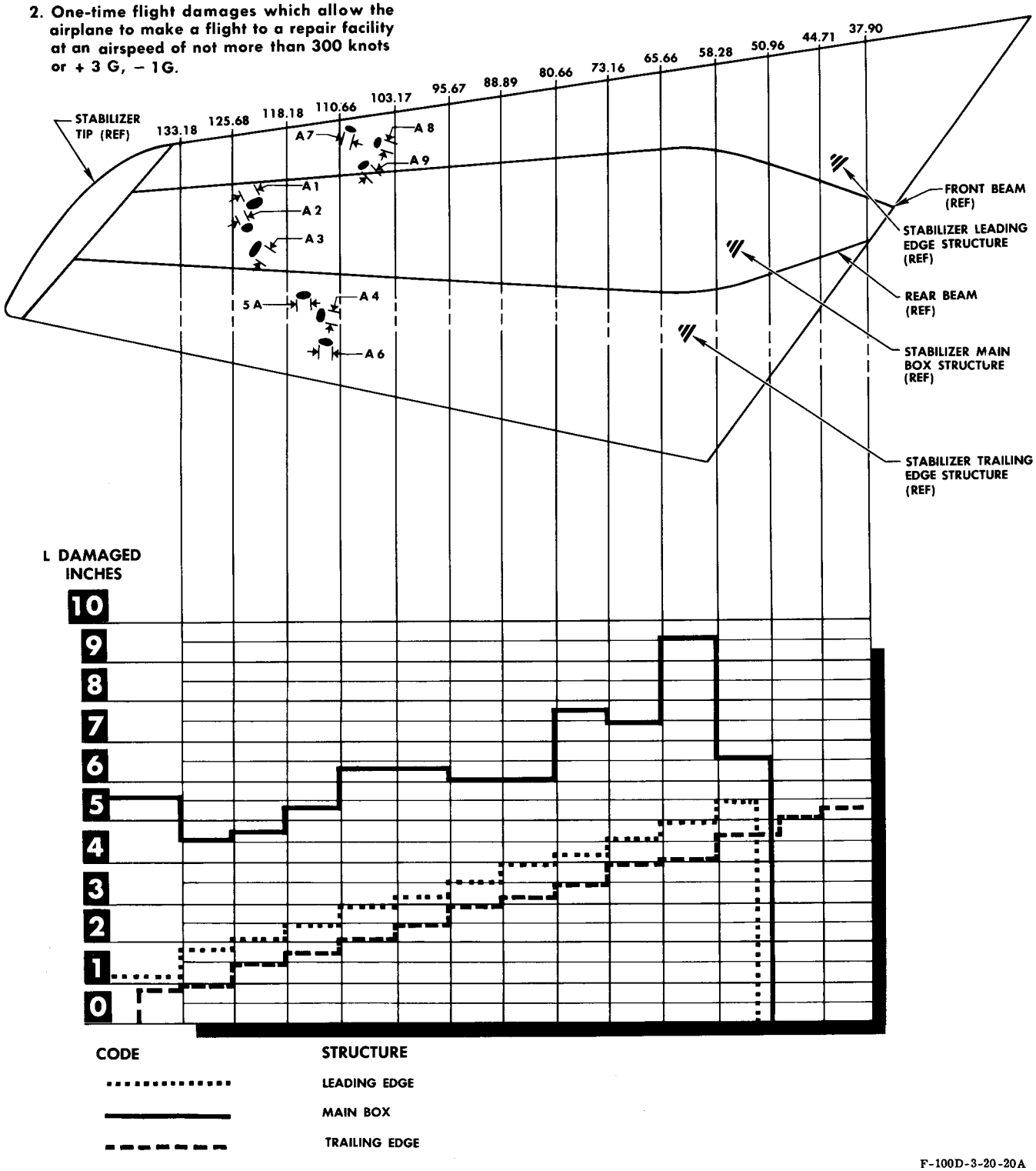

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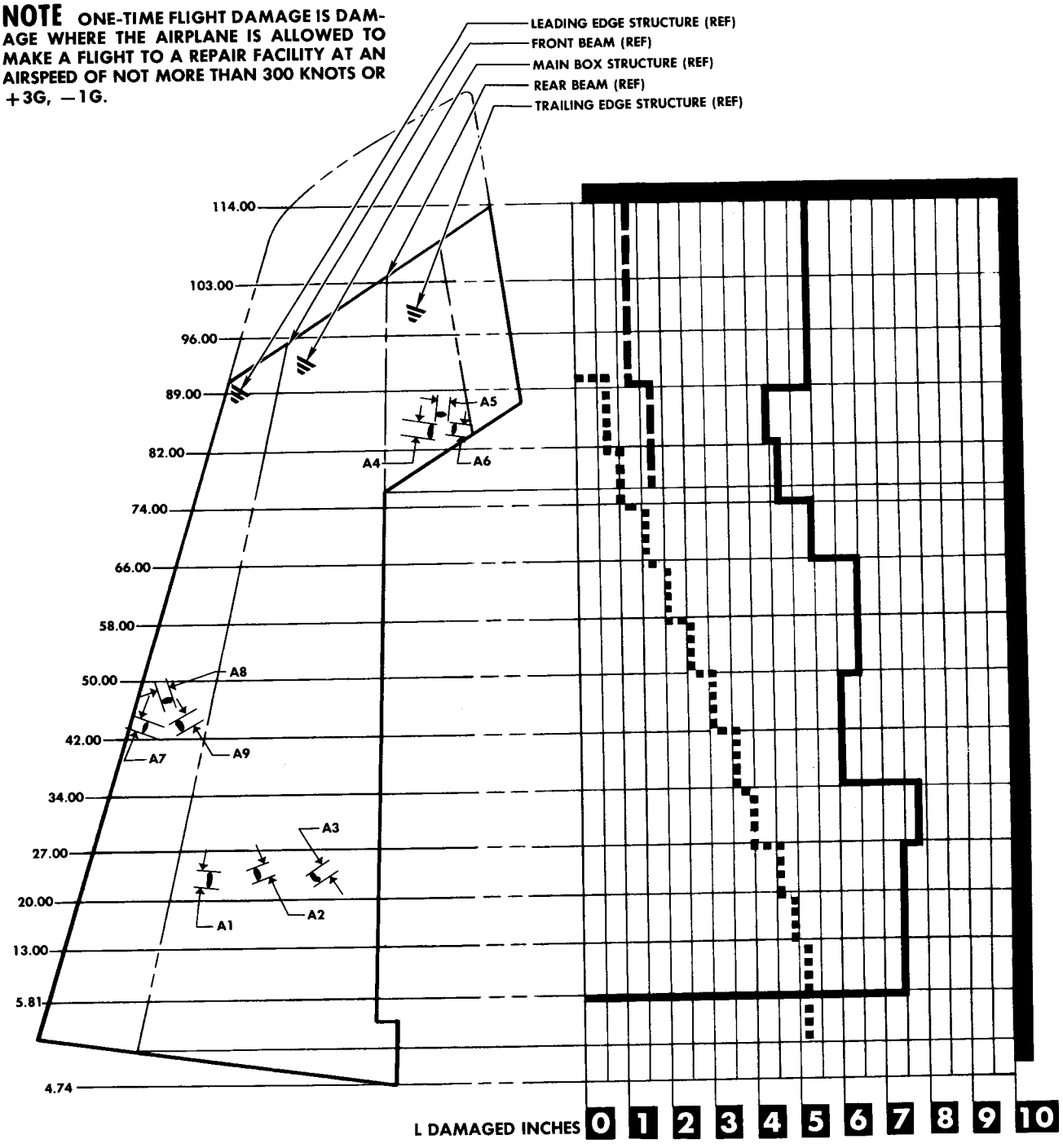
Figure 3-2. Horizontal Stabilizer One-time Flight Damage

ONE-TIME FLIGHT DAMAGE

EMPENNAGE

NOTE ONE-TIME FLIGHT DAMAGE IS DAMAGE WHERE THE AIRPLANE IS ALLOWED TO MAKE A FLIGHT TO A REPAIR FACILITY AT AN AIRSPEED OF NOT MORE THAN 300 KNOTS OR +3G, −1G.



STRUCTURE	CODE
LEADING EDGE	■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■ ■
MAIN BOX	———
TRAILING EDGE	— — — — — — —

F-100D-3-20-55A

Figure 3-3. Vertical Stabilizer One-time Flight Damage

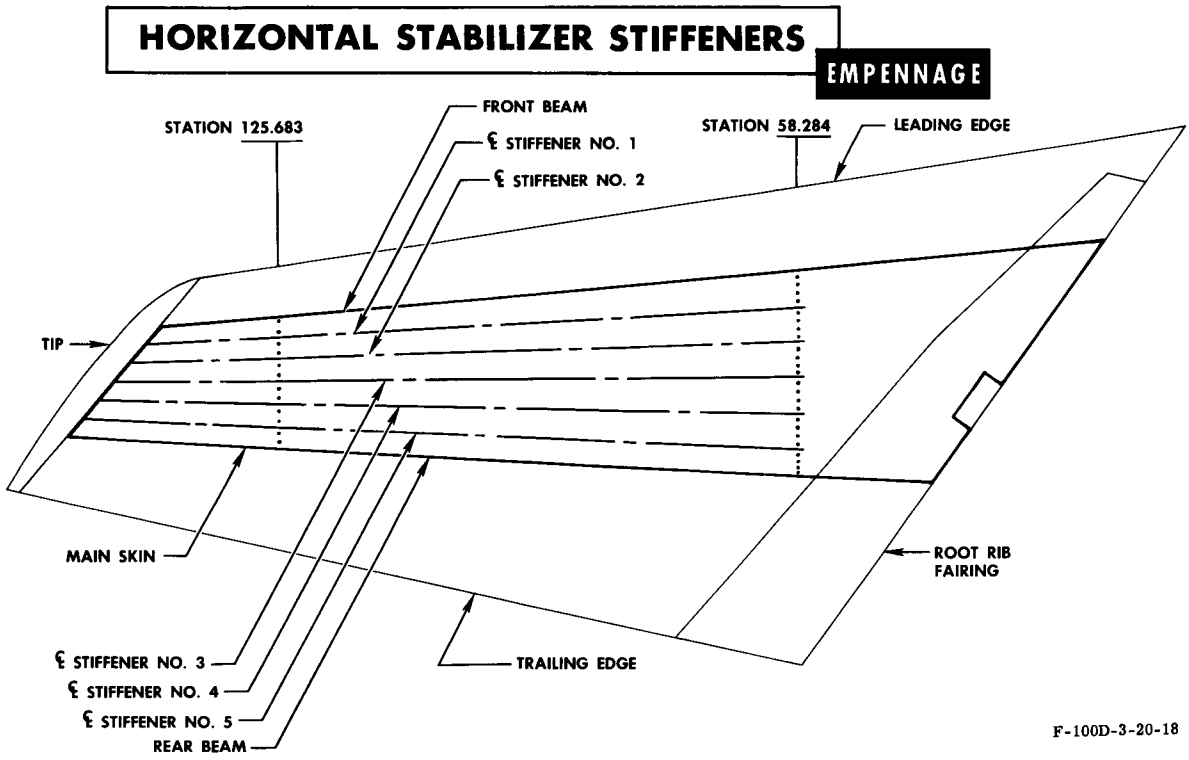


Figure 3-4. Locating Horizontal Stabilizer Integral Stiffeners

REMOVING VERTICAL STABILIZER MAIN SKINS.

When removing fasteners from milled skin, trailing edge skin, and overboard vent fairing, be careful not to damage radar coaxial cables, electrical wiring, or overboard vent tube.

1. Remove vertical stabilizer. (Refer to "F-100D General Airplane," T. O. 1F-100D-2-1.)
2. Remove rudder. (Refer to "F-100D Flight Control Systems," T. O. 1F-100D-2-5.)
3. Remove vertical stabilizer tip. (Refer to "Removing Command Radio Antenna," in "F-100D Radio and Radar," T. O. 1F-100D-2-8.)
4. Remove leading edge close-out skin.
5. Remove left overboard vent fairing.
6. Remove Jo-bolts and Cherry Rivets attaching forward edge of left trailing edge skin to milled skin.

Note

Remove group of four Jo-bolts at lower forward edge of trailing edge skin.

7. Remove left milled skin.

INSTALLING VERTICAL STABILIZER SKINS.

The vertical stabilizer leading edge, trailing edge and main skins are installed by assembling parts in the reverse order of that used in their disassembly for skin removal. When installing close-tolerance fasteners, make sure that the holes are within tolerances specified. Although a jig is desirable, it is possible to reassemble the old leading edge, trailing edge and main skins without a jig. When installing new skins, a jig must be used.

REMOVING UPPER RUDDER SKINS.

1. Remove rudder from airplane.
2. Remove upper rudder from lower rudder.
3. Remove rudder splitter plate.
4. Remove either right or left rudder skin. Drill out rivets at aft edge of skin and work forward.

REMOVING LOWER RUDDER SKINS.

1. Remove rudder from airplane.
2. Remove upper rudder from lower rudder.
3. Remove rudder splitter plate.
4. Remove either right or left rudder skin. Drill out rivets at edge of skin and work forward.

INSTALLING RUDDER SKINS.

The rudder skins are installed by assembling parts in the reverse order of that used in their disassembly for skin removal. When installing close-tolerance fasteners, make sure that the holes are within tolerances specified. Although a jig is desirable it is possible to reassemble the old skins without a jig. When installing new skins, a jig must be used.

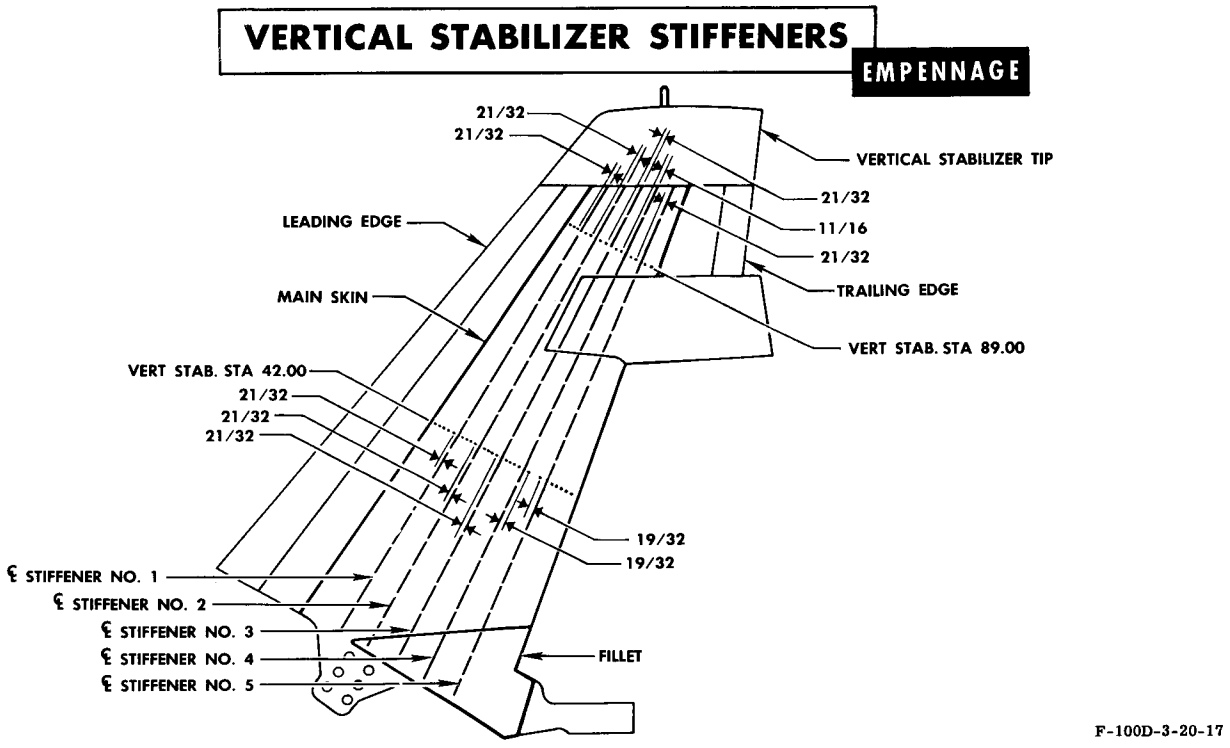


Figure 3-5. Locating Vertical Stabilizer Integral Stiffeners

LOCATING HORIZONTAL STABILIZER INTEGRAL SKIN STIFFENERS.

The horizontal stabilizer upper and lower main skins are attached by rivets on one side and by Jo-bolts on the opposite side. The horizontal stabilizer panels are interchangeable and may be used on either the right or left side. The center horizontal stabilizer main skin stiffeners for the lines are located by joining established points at stations 58.284 and 125.683 with a straight edge (see figure 3-4). The integral stiffeners are 0.156-inch wide. The horizontal stabilizer main skin stiffeners are located by starting with the riveted main skin first. Locate station 58.284 which is the row of rivets about 2.5 inches outboard of the horizontal stabilizer root rib fairing intersection with the aft edge of the main skin. The rivets are AN426-AD5 in groups of four which are preceded and followed by a group of two rivets. The centerline of the stiffeners is midway between the end rivets of adjacent rivet groups. Locate station 125.683, which will be the third row of rib rivets inboard of the tip attach point to the horizontal stabilizer. The rivet pattern will have two rivets in each group. The stiffener centerline will be in the middle of the wide gap between each rivet group. On the horizontal stabilizer main skin where Jo-bolts have been used, the centerline of the stiffeners is found by transferring measurements from the riveted side of the stabilizer. The forward or aft edge of the main skin can be used for transferring the measurements. When making repairs or cutting into the horizontal stabilizer main skin, avoid cutting the main skin stiffeners if possible.

LOCATING VERTICAL STABILIZER INTEGRAL SKIN STIFFENERS.

The vertical stabilizer right main skin is attached by rivets; the left main skin is attached by Jo-bolts. The right vertical stabilizer main skin stiffener centerlines are located by joining established points at vertical stabilizer stations 42.00 and 89.00. (See figure 3-5.) The right vertical stabilizer main skin stiffener centerlines are located by transferring measurements from the left vertical stabilizer main skin. The aft edges of the right and left vertical stabilizer main skins should be used for the zero reference line when transferring measurements for the right main skin stiffener locations. The main skin stiffeners are 0.156-inch wide. All given dimensions for the main skin stiffener centerlines start at the center of the rivet. When making repairs to the main skin, avoid cutting the main skin stiffeners if possible. Care must be used not to cut into the electrical wiring, radio coaxial cables or fuel overboard vent lines inside of the vertical stabilizer when making repairs to the main skin. Station 42.00 location is the tenth

row of rib rivets from the vertical stabilizer tip (the row attaching the plastic tip to stabilizer is omitted).

At station 42.00 on the left of the stabilizer, start with the first forward row of rivets in the milled skin. Stiffener No. 1 is 21/32-inch aft of the fifth rivet in the milled skin. Stiffener No. 2 is 21/32-inch aft of the ninth rivet in the milled skin. Stiffener No. 3 is 21/32-inch aft of the 13th rivet in the milled skin. Stiffener No. 4 is 19/32-inch forward of the seventh rivet from the trailing edge of the milled skin. Stiffener No. 5 is 19/32-inch aft of the sixth rivet from the trailing edge of the milled skin.

At station 89.00 on the left side of the stabilizer, start with the first forward full row of rivets in the milled skin. Stiffener No. 1 is 21/32-inch aft of the third rivet in the milled skin. Stiffener No. 2 is 21/32-inch aft of the fifth rivet in the milled skin. Stiffener No. 3 is 21/32-inch aft of the seventh rivet in the milled skin. Stiffener No. 4 is 11/16-inch aft of the ninth rivet in the milled skin. Stiffener No. 5 is 21/32-inch aft of the 11th rivet in the milled skin. Joining upper and lower locations with a straightedge will give the centerlines of the respective stiffeners.

NEGLIGIBLE DAMAGE.

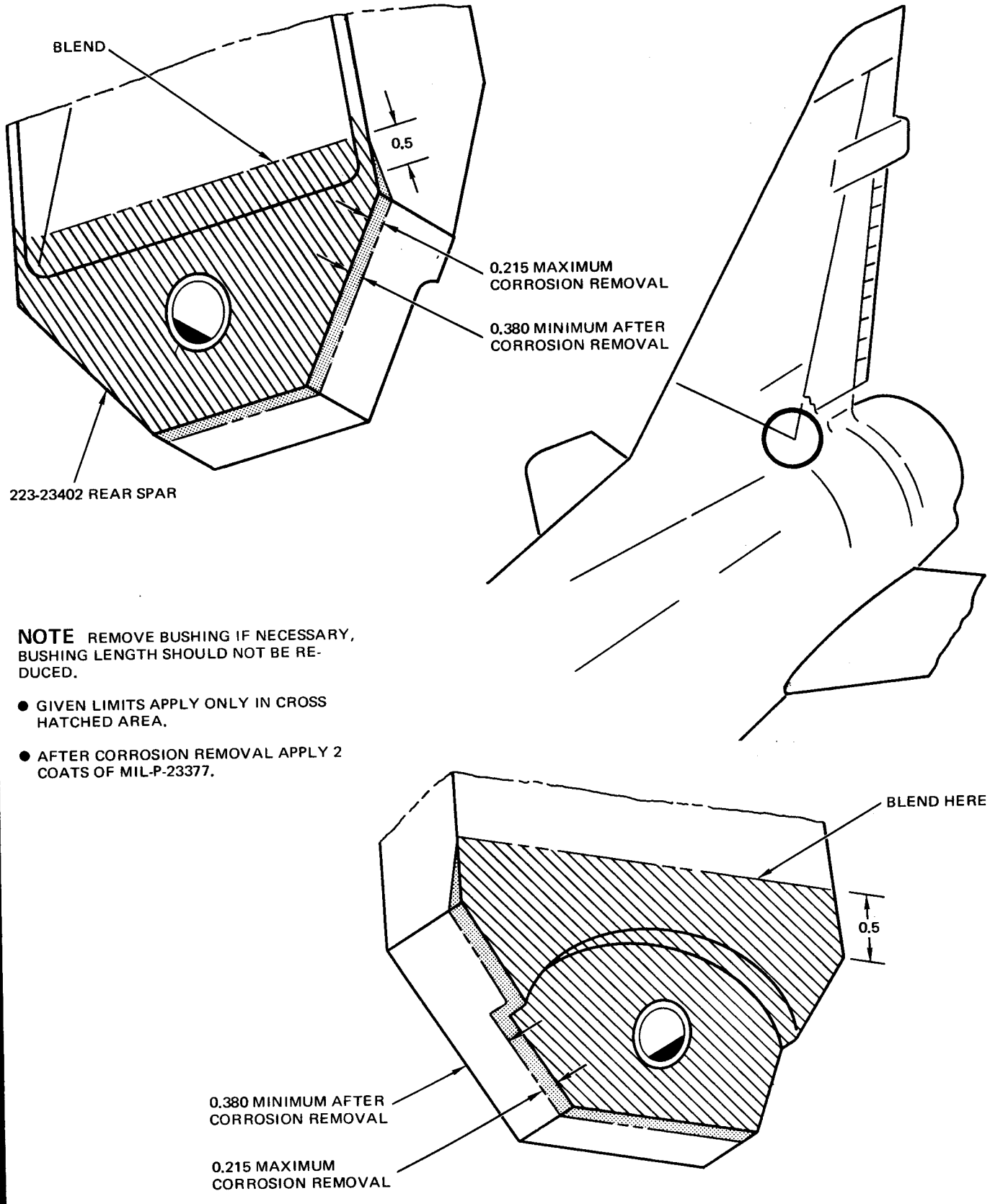
For negligible damage limits for the horizontal stabilizer, see figure 3-7A. For negligible damage limits for the vertical stabilizer, see figure 3-19A. For negligible damage limits for the rudder, see figure 3-34.

EMPENNAGE REPAIRS.

EMPENNAGE REPAIR INDEX.

See figure 3-6.

CORROSION REMOVAL LIMITS FOR REAR SPAR – VERTICAL STABILIZER, FUS ATTACH (PN 223–23402).



NOTE REMOVE BUSHING IF NECESSARY, BUSHING LENGTH SHOULD NOT BE REDUCED.

- GIVEN LIMITS APPLY ONLY IN CROSS HATCHED AREA.
- AFTER CORROSION REMOVAL APPLY 2 COATS OF MIL-P-23377.

Figure 3-5A. Corrosion Removal Limits for Rear Spar-Vertical Stabilizer, Fus Attach (P/N 223-23402).

TEMPORARY REPAIRS.

TEMPORARY REPAIR OF MILLED SKIN.

Temporary (nonflush) repairs may be used on both the horizontal and vertical stabilizer milled skins when time or tooling does not permit the installation of permanent repairs. Figure 3-38 shows areas which may be repaired, the type of repair, and materials to use. To accomplish repair, proceed as follows:

1. Trim out damaged area, keeping hole as small as possible. A rectangular opening is shown in figure 3-38; however, it may be round or oval if those shapes satisfy the repair conditions.

2. With a soft pencil, draw lines on outside of skin around the hole to indicate locations of integral stiffeners and ribs.

3. Tape a piece of heavy paper over hole to represent the doubler. Transfer lines drawn on skin to the paper.

4. Lay out fastener pattern as given in applicable fastener chart (figure 3-38). Maintain spacing given whenever possible. It will be necessary, however, to adjust the spacing somewhat to avoid drilling into integral stiffeners and ribs. Do not space fasteners closer than three times the fastener diameter.

5. Center-punch at least three fastener locations before removing paper pattern. Remove pattern and pilot-drill center-punched locations.

6. Shape material selected for repair doubler to match skin contour.

7. Tape paper pattern to repair doubler and transfer fastener pattern to it with a center punch.

8. Pilot-drill holes in doubler corresponding to those previously drilled in the skin.

9. Fasten doubler in place, pilot-drill and finish drilling all holes through doubler and skin. Also drill the number 17 holes for the repair post fasteners. Remove repair doubler and countersink the holes.

10. Pilot-drill and finish-drill the repair posts. Trim repair posts to length. They must fit tightly between repair doubler and opposite skin.

11. Attach repair posts to repair doubler with temporary fasteners and install assembly over the hole (also with temporary fasteners).

12. Remove fastener from one repair post and line-drill the opposite skin. Replace temporary fastener and repeat operation until all repair posts are drilled; then remove assembly and clean out shavings.

13. If doubler is made of steel, remove posts and heat-treat doubler. Paint with two coats of zinc chromate primer. If bare aluminum alloy was used for the doubler, it should receive two coats of zinc chromate primer.

14. Attach repair posts to repair doubler with permanent fasteners.

15. Install repair assembly in place with permanent fasteners. It is recommended that repair post fasteners be installed first to ensure their alignment.

16. If repair doubler was previously painted with zinc chromate, finish the outside with aluminum lacquer to match the clad skins.

TEMPORARY REPAIR OF LEADING EDGE SKINS.

Temporary repairs to the horizontal and vertical stabilizer leading edge skins are made by using an external patch. When temporary repairs are made, it must be remembered that permanent repairs will replace the temporary repair at a later time. When a temporary repair is made, a fastener pattern that can be used in a permanent repair fastener pattern should be used.

TEMPORARY REPAIR OF HORIZONTAL STABILIZER LEADING EDGE SKINS.

To make temporary leading edge repairs to the horizontal stabilizer see figures 3-8 and 3-9, and then proceed as follows:

1. Trim out damaged area midway between existing rivets.

2. Make repair doubler from 0.020 ½ hard corrosion resistant steel.

   a. For leading edge repair doubler rivet spacing opposite stabilizer close-out skin, see figure 10-10.

   b. For the remaining rivets, see figure 10-18.

   c. Drill and dimple repair doubler.

3. Sub-countersink skin.

4. See figure 10-10. Then rivet the repair doubler to the leading edge opposite close-out skin.

5. See figure 10-18. Then finish riveting repair doubler.

TEMPORARY REPAIR OF VERTICAL STABILIZER LEADING EDGE SKINS.

See figures 3-20 and 3-21.

PERMANENT REPAIRS.

Permanent repairs may be made to the milled skins of the horizontal and vertical stabilizer. The milled

skins will be replaced when damage occurs to the milled skins inboard of station 50 on the horizontal stabilizer or inboard of station 5.81 on the vertical stabilizer.

PERMANENT REPAIR OF HORIZONTAL STABILIZER.

HORIZONTAL STABILIZER SPANWISE BOW TOLERANCE.

The serviceability of horizontal stabilizers with a spanwise bow in the leading edge section, box section, or trailing edge section may be determined by using the following limits and criteria:

Note

When a bow is evident, the necessary inspections should be performed to ensure that the internal structure is not damaged or buckled to a degree that materially affects the structural integrity of the assembly.

1. Leading edge section check (leading edge to forward spar at 25 percent plane).

a. Maximum allowable bow tolerance between horizontal stabilizer stations 30.17 and 127.00 is limited to 3/16-inch deflection and is to flow in an even, gradual curve.

b. The bow tolerance is to be measured vertically from the centerline of the leading edge radius.

Note

Indentations or locally buckled skin should not be classified as a bow, but as structural damage. The serviceability of such damaged components must be classified as either negligible or reparable damage.

2. Trailing edge section check (rear spar at 65 percent plane to extreme trailing edge).

a. The maximum allowable bow tolerance between horizontal stabilizer stations 47.0 and 138.0 is limited to 3/16-inch deflection and is to flow in an even, gradual curve.

b. A maximum of one bow per surface with a maximum length of 30 inches is allowed.

c. No bow tolerances beyond those given on NAA Drawings 192-21001 and 192-21002 are allowed between the following stations:

(1) Horizontal stabilizer tip stations 133.18 and 127.

(2) Horizontal stabilizer root canted station 35.0 and horizontal stabilizer station 30.71.

Note

Indentations or locally buckled skin should not be classified as a bow, but as structural damage. The serviceability of such damaged components must be classified as either negligible or reparable damage.

d. No bow tolerances beyond those given on NAA Drawings 192-21001 and 192-21002 are allowed between the following stations:

(1) Horizontal stabilizer canted tip station 143.8 and horizontal stabilizer station 138.0.

(2) Horizontal stabilizer root canted station 35.0 and horizontal stabilizer station 47.0.

3. Box cell section check (forward spar to rear spar).

Note

Certain bow characteristics may form if the surface is twisted to the maximum allowed on NAA Drawing 192-21002. The bow will form in a diagonal manner from opposite corners of the surface.

a. Bow tolerance for diagonally directed irregularities is 1/4 inch and must be measured from root to tip along the 46.43 percent plane. (Refer to NAA Drawing 192-21002.)

4. Complete horizontal stabilizer panel check.

a. A bow extending through the entire stabilizer panel must have a maximum tolerance of 1/4 inch in any 30-inch section, or 1/2 inch from canted root plane to canted tip plane measured vertically along the 46.43 percent plane.

b. Any bow or twist that results in structural damage or skin deformation must be repaired, regardless of whether box limitations fall into acceptable categories or not.

c. Any stabilizer that has a bowed leading edge, box, or trailing edge, or a combination of all three, must be thoroughly checked for twist and alignment in accordance with applicable directives before bow allowance acceptance.

PERMANENT REPAIR OF HORIZONTAL STABILIZER STATION 88.16 TO 95.67 CHORDWISE MILLED SKIN.

To make a chordwise repair to horizontal stabilizer station 88.16 to 95.67, see figure 3-10 and then proceed as follows:

1. Trim out damaged area in milled skin.

2. Make repair fillers from 7075-T6 bare aluminum alloy plate.

3. Make repair doublers from 0.250-inch 4130 steel plate heat-treated to 140,000-160,000 psi.

a. In one repair doubler drill the holes for Deutsch drive pin rivet wiht a No. 17 drill.

b. Drill an 11/32-inch diameter hole 17/64 inch on center from each of the No. 17 drilled holes.

c. Apply two coats of zinc chromate primer.

4. Make repair post from 5/16-inch diameter 2024-T42 aluminum alloy rod.

a. Drill the rod lengthwise with a No. 17 drill.

b. Repair post will be cut to length when installed.

5. Trim integral skin stiffeners to allow installation of repair doublers.

6. Install repair doublers with F-312 Jo-bolts.

7. Insert repair posts through 11/32-inch diameter hole in drilled repair doubler and slide over to align repair post with rivet holes in drilled repair doubler.

8. Place the repair filler to the undrilled repair doubler.

a. Drill and countersink for CR762-6 rivets.

b. Temporarily attach repair filler to repair doubler.

c. Back drill repair doubler and repair filler with a No. 17 drill through repair post.

9. Remove repair filler and dimple for 6950-S-06 Deutsch drive pin blind rivet and then ream dimple with 3/16-inch drill.

10. Repeat steps 8 and 9 for the remaining filler.

11. Ream out the No. 17 drill holes in repair doublers and repair posts with a 0.192 (±0.001) inch reamer.

12. Subcountersink repair doubler for dimpled repair fillers and apply two coats of zinc chromate primer to subcountersink areas.

13. Place repair fillers into position and install 6950-S-06 Deutsch drive pin blind rivet.

14. Complete repair filler installation to repair doubler by installing CR762-6 rivets.

PERMANENT REPAIR OF VERTICAL STABILIZER.

PERMANENT REPAIR OF VERTICAL STABILIZER STATION 89 CHORDWISE MILLED SKIN.

To make a chordwise repair at vertical stabilizer station 89, see figure 3-22; then proceed as follows:

1. Trim out damaged area in milled skin.

2. Make repair fillers from 7075-T6 bare aluminum alloy plate.

3. Make repair doubler from 7075-T6 bare aluminum alloy plate one gage heavier than thickest part of skin at repair area.

a. In one repair doubler, drill holes for Deutsch drive pin blind rivet with a No. 17 drill.

b. Drill an 11/32-inch diameter hole 17/64 inch on center from each No. 17 drilled hole.

4. Make repair post from 5/16-inch diameter 2024-T42 aluminum alloy rod.

a. Drill the rod lengthwise with a No. 17 drill.

b. Repair post will be cut to length when installed.

5. Trim integral skin stiffeners and intermediate rib to allow installation of repair doublers.

6. Install repair doubler with F260 Jo-bolts.

7. Insert repair posts through 11/32-inch diameter hole in drilled repair doubler and align repair post with rivet holes in repair doubler.

8. Place repair filler to undrilled repair doubler.

a. Drill and countersink for AF462-4 rivet.

b. Temporarily attach repair filler to repair doubler.

c. Back drill repair doubler and repair filler with a No. 17 drill through repair post.

9. Remove repair filler and dimple for 6950-S-06 Deutsch drive pin blind rivet and then ream dimple with 3/16-inch drill.

10. Repeat steps 8 and 9 for the remaining filler.

11. Ream out No. 17 drill holes in repair doublers and repair post with an 0.192 (±0.001) reamer.

12. Subcountersink repair doubler for 6950-S-06 Deutsch drive pin blind rivet.

13. Place repair fillers into position and install 6950-S-06 Deutsch drive pin blind rivets.

14. Complete repair filler installation to repair doubler by installing AF462-4 rivets.

PERMANENT REPAIR OF VERTICAL STABILIZER STATION 89 SPANWISE MILLED SKIN.

To make a spanwise repair at vertical stabilizer station 89, see figure 3-23 and then proceed as follows:

1. Trim out damage area in milled skin.

2. Make repair fillers from 7075-T6 bare aluminum alloy plate.

3. Make repair doubler from 0.063-inch 2024-T4 alclad sheet.

a. In one repair doubler drill holes for Deutsch drive pin blind rivet with a No. 17 drill.

b. Drill an 11/32-inch diameter hole 17/64 inch on center from each No. 17 drilled hole.

4. Make repair post from 5/16-inch diameter 2024-T42 aluminum alloy rod.

a. Drill rod lengthwise with a No. 17 drill.

b. Repair post will be cut to length when installed.

5. Trim integral skin stiffeners and intermediate rib to allow installation of repair doublers.

6. Install repair doublers with AF462-6 rivets.

7. Insert repair posts through 11/32-inch diameter hole in drilled repair doubler and align repair post with rivet holes in repair doubler.

8. Place the repair filler to the undrilled repair doubler.

a. Drill and countersink for AF462-4 rivet.

b. Temporarily attach repair filler to repair doubler.

c. Back drill repair doubler and repair filler with No. 17 drill through repair post.

9. Remove repair filler and dimple for 6950-S-06 Deutsch drive pin blind rivet, and then ream dimple with 3/16-inch drill.

10. Repeat steps 8 and 9 for the remaining filler.

11. Ream out No. 17 drill holes in repair doublers and repair post with a 0.192 (±0.001) inch reamer.

12. Sub-countersink repair doubler for dimpled repair filler.

13. Place repair fillers into position and install 6950-S-06 Deutsch drive pin blind rivets.

14. Complete repair filler installation to repair doubler by installing AF462-4 rivets.

PERMANENT REPAIR OF LEADING EDGE SKINS.

Permanent repairs are made to the leading edge skins of the horizontal and vertical stabilizers by using an internal repair doubler and an external repair filler. Make sure the nose ribs and front beam are not damaged before repairing the leading edge.

PERMANENT REPAIR OF HORIZONTAL STABILIZER LEADING EDGE SKINS.

See figures 3-11 and 3-12, and then proceed as follows:

1. Trim out damaged area midway between existing rivets.

2. Make repair doublers from 0.016-inch half-hard corrosion resistant steel sheet.

a. For leading edge repair doubler rivet spacing, see figure 10-10.

3. Remove enough rivets to allow repair doublers to be installed between leading edge skin and nose rib.

4. Drill leading edge skin and doublers.

5. Countersink leading edge skin for AN426 AD4 rivets.

6. Install repair doublers with AN426 AD4 rivets.

7. Make repair filler from 0.040-inch 2024-T4 aluminum alloy sheet.

a. For repair filler rivet spacing, see figure 10-18.

8. Drill repair filler and repair doubler.

9. Countersink repair filler for AF462-4 rivets.

10. Install repair filler to repair doubler with AF462-4 Cherry rivets.

a. Use aerodynamic filler (PR-341) if the gap between existing skin and repair filler exceeds 1/16 inch.

PERMANENT REPAIR OF VERTICAL STABILIZER LEADING EDGE SKINS.

See figures 3-24 and 3-25.

PERMANENT REPAIR OF TRAILING EDGE SKINS.

PERMANENT REPAIR OF HORIZONTAL STABILIZER TRAILING EDGE SKINS.

See figure 3-13.

PERMANENT REPAIR OF VERTICAL STABILIZER TRAILING EDGE SKINS.

See figures 3-27 through 3-29.

PERMANENT REPAIR OF HORIZONTAL STABILIZER TIP.

Horizontal stabilizer tips are made of weldable aluminum alloy; therefore, much of the repair may be done by welding. The damaged area may be cut out and new sections welded in or regular skin-doubler repairs, shown in figure 3-16, may be used.

PERMANENT REPAIR OF VERTICAL STABILIZER TIP.

The vertical stabilizer tip is made of fiber glass. To repair the vertical stabilizer refer to "Fabrics and Plastics," in Section VII.

PERMANENT REPAIR OF HORIZONTAL AND VERTICAL STABILIZER NOSE RIBS.

To repair the nose ribs for the horizontal stabilizer see figure 3-17, or vertical stabilizer see figure 3-30 and then proceed as follows:

1. Starting at inboard end, remove enough fasteners from the leading edge close-out skin to allow repair work on the rib.

2. See the repair figure for the rib being repaired and then repair the rib.

3. Install close-out skin.

PERMANENT REPAIR OF VERTICAL STABILIZER INTERMEDIATE RIBS.

1. Remove the left milled skin. Do steps 1 through 7 of "Removing Vertical Stabilizer Main Skins."

2. See figure 3-31 and then make repair to the intermediate rib.

3. Install the milled skin.

4. Complete the trailing edge skin installation.

5. Install the left overboard vent fairing.

6. Install the leading edge close-out skin.

7. Install the vertical stabilizer tip. (Refer to "Removing Command Radio Antenna," in "F-100D Radio and Radar," T.O. 1F-100D-2-8.)

8. Install the rudder. (Refer to "F-100D Flight Control Systems," T.O. 1F-100D-2-5.)

9. Install the vertical stabilizer. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

HORIZONTAL AND VERTICAL STABILIZER ATTACH-POINT HOLE REPAIR.

The horizontal and vertical stabilizer attach-point holes may be repaired by installing bushings. (See figures 3-18 and 3-32.)

REWORK AND ADJUSTMENT OF RUDDER LINK AND RUDDER HORN ATTACHMENTS.

When the rudder is installed on the vertical stabilizer, a push-pull check should be made on the trailing edge of the rudder as outlined in "F-100D Flight Control Systems," T.O. 1F-100D-2-5. If the 3½-pound limit called for in this check is exceeded, two possible discrepancies may exist. Check for these discrepancies as follows:

1. Remove the two screws that attach lower hinge link to vertical stabilizer. If link moves forward, causing misalignment of holes in link with holes in vertical stabilizer, rework holes as shown in figure 3-31A. (Make this check with upper and lower attach points secured.)

2. Loosen the two rudder-horn-to-rudder-torque-tube attaching bolts. If horn and tube separate, install a shim as shown in figure 3-31A. (Make this check with upper and lower hinge links secured.)

RUDDER REPAIRS.

See repair index figure 3-33 for rudder repairs.

HOLDING FIXTURE DIMENSIONS.

Figures 3-40 and 3-41 show the necessary dimensions for the construction of fixtures to hold the empennage components while repairs are being made. These dimensions and attach points are not intended for use in manufacturing jigs for the building of a complete component. They are intended only for manufacturing holding fixtures to maintain the proper attach points, and to ensure that the part being repaired will fit when reinstalled.

REWORK OF DRAG CHUTE RELEASE MECHANISM FRAME FILLET.

See figure 3-42.

1. Remove panels F151 and F152.

2. Cut 1¼-inches off the lower end of both right and left hand members of fillet frame Part No. 192-25106.

3. Rivets will be used to fill vacant holes on panels F151 and F152.

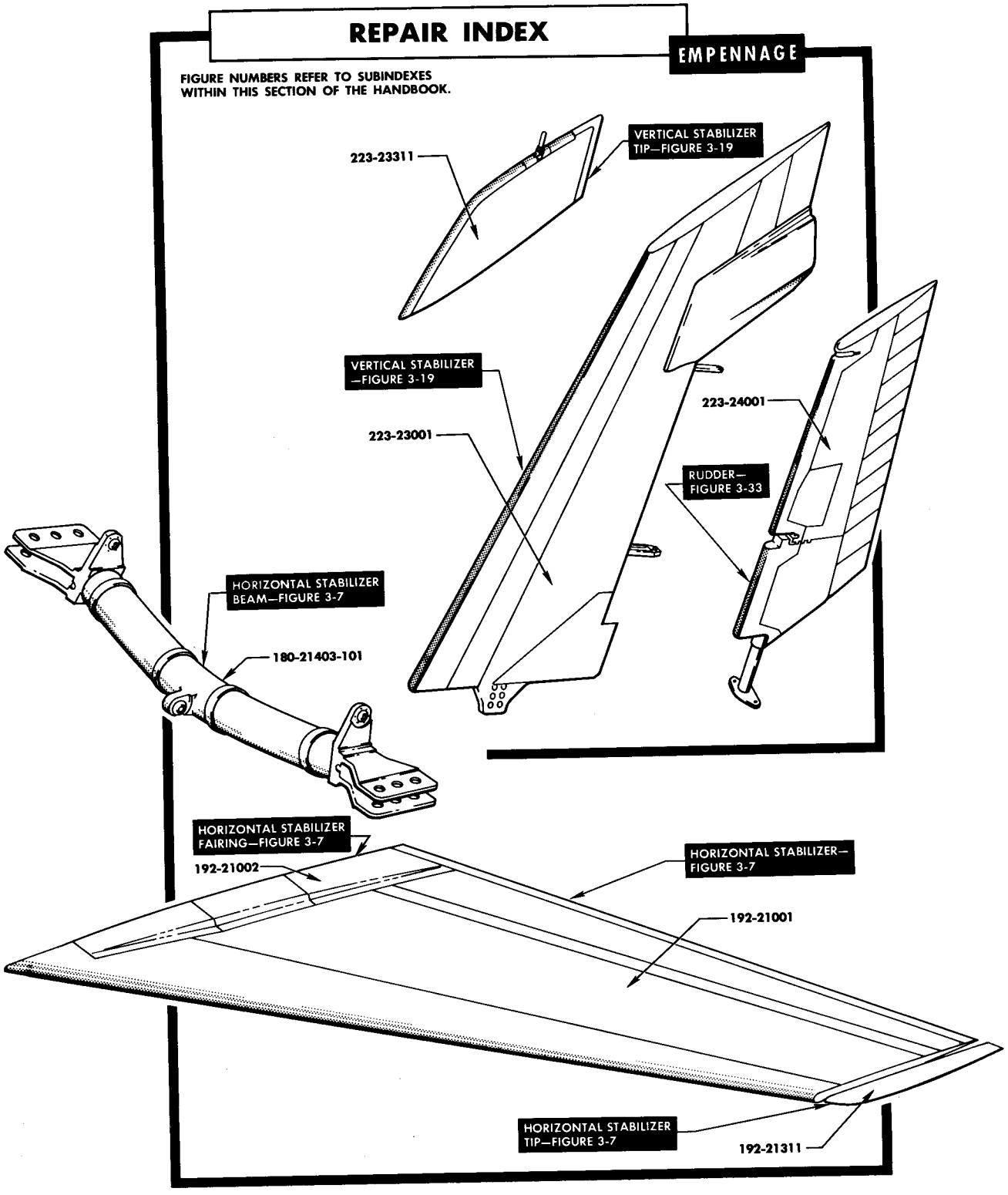


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Figure 3-6. Empennage Repair Index

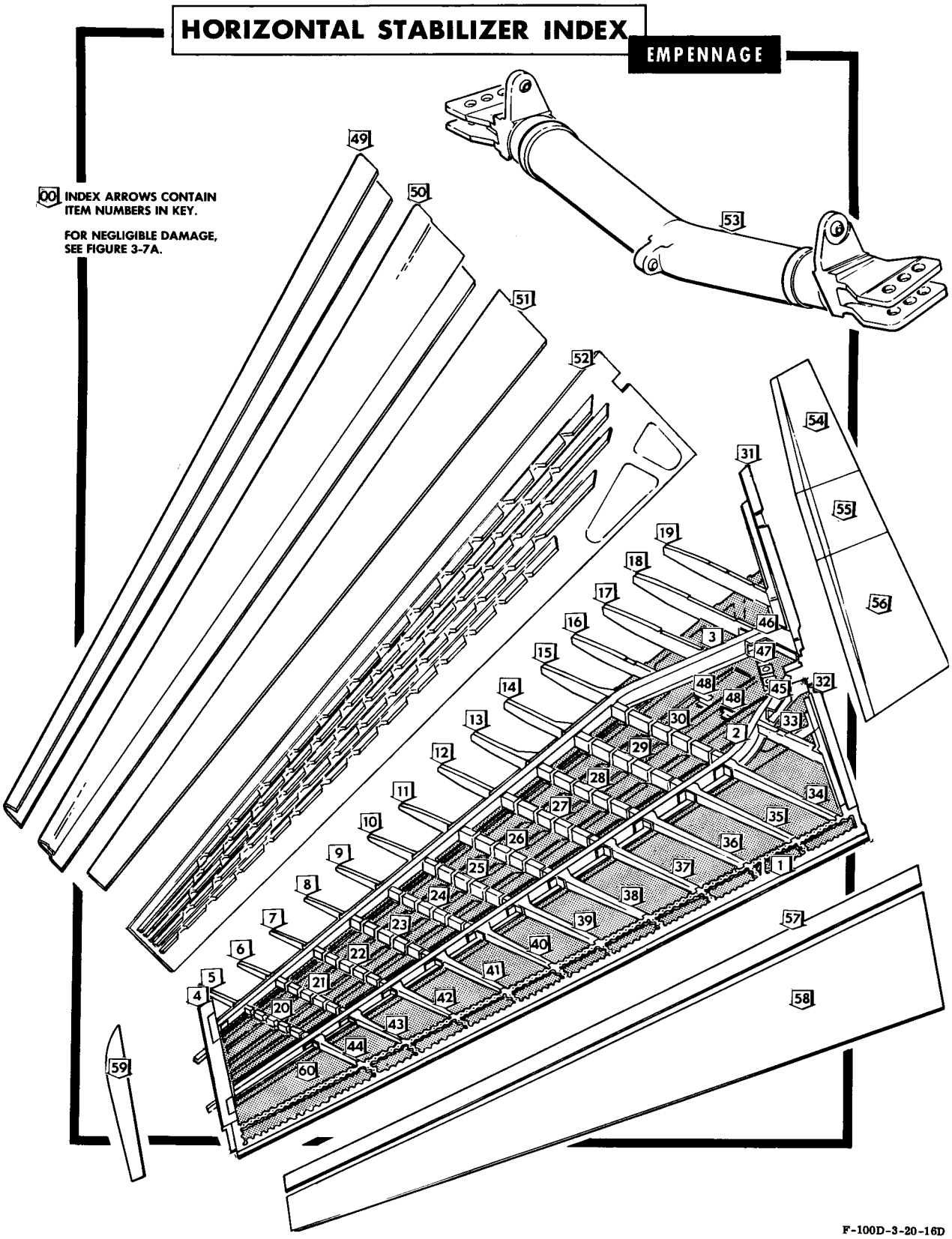


Figure 3-7. Horizontal Stabilizer Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR Figure NO.†
1	192-21420	Beam—TE	Extr	Mag Alloy	3-13
2	192-21414-5	Beam—Rear	Extr	Al Alloy	
3	192-21414-3	Beam—Front	Extr	Al Alloy	
4	192-21139-41	Rib Assy			
5	192-21128-3	Nose Rib	0.032	2024-T4 Clad	
6	192-21120	Nose Rib	0.032	2024-T4 Clad	
7	192-21113	Nose Rib	0.032	2024-T4 Clad	
8	192-21105	Nose Rib	0.032	2024-T4 Clad	
9	192-21098	Nose Rib	0.032	2024-T4 Clad	
10	192-21090	Nose Rib	0.032	2024-T4 Clad	
11	192-21083	Nose Rib	0.032	2024-T4 Clad	
12	192-21075	Nose Rib	0.032	2024-T4 Clad	
13	192-21068	Nose Rib	0.032	2024-T4 Clad	3-17
14	192-21060	Nose Rib	0.032	2024-T4 Clad	3-17
15	192-21053	Nose Rib	0.040	2024-T4 Clad	3-17
16	192-21050	Nose Rib	0.032	2024-T4 Clad	3-17
17	192-21044	Nose Rib	0.032	2024-T4 Clad	3-17
18	192-21037	Nose Rib	0.081	7075-T6 Clad	3-17
19	192-21030	Nose Rib	0.081	2024-T4 Clad	3-17
20	192-21129	Inter Rib	0.032	2024-T4 Clad	
21	192-21121	Inter Rib	0.032	2024-T4 Clad	
22	192-21114	Inter Rib	0.032	2024-T4 Clad	
23	192-21106	Inter Rib	0.040	2024-T4 Clad	
24	192-21099	Inter Rib	0.040	2024-T4 Clad	
25	192-21091	Inter Rib	0.040	2024-T4 Clad	
26	192-21084	Inter Rib	0.040	2024-T4 Clad	
27	192-21076-5	Inter Rib	0.051	2024-T4 Clad	
28	192-21069-5	Inter Rib	0.051	2024-T4 Clad	
29	192-21061-5	Inter Rib	0.064	2024-T4 Clad	
30	192-21054-5	Inter Rib	0.072	2024-T4 Clad	
31	192-21035	Rib Assy			
32	192-21036-3	Rib Assy	0.040	2024-T4 Clad	
33	192-21052	TE Rib	0.064	7075-T6 Clad	
34	192-21058	TE Rib	0.064	7075-T6 Clad	
35	192-21062	Rib	0.064	7075-T6 Clad	
36	192-21070	TE Rib	0.064	7075-T6 Clad	
37	192-21077	TE Rib	0.051	7075-T6 Clad	
38	192-21085	TE Rib	0.051	7075-T6 Clad	
39	192-21092	TE Rib	0.051	7075-T6 Clad	
40	192-21100	TE Rib	0.051	7075-T6 Clad	
41	192-21107	TE Rib	0.051	7075-T6 Clad	
42	192-21115	TE Rib	0.040	2024-T4 Clad	
43	192-21122	TE Rib	0.040	2024-T4 Clad	
44	192-21130	TE Rib	0.040	2024-T4 Clad	
45	192-21408	Attaching Fitting		2024-T42 Bar	
46	192-21423	Attaching Fitting		2014-T6 Forging	
47	192-21415	Attaching Spacer	0.875	2024-T42 Plate	
48	192-21051-5	Inter Rib	0.081	2024-T4 Clad	
49	192-21342-39 ‡192-21002-39	Leading Edge Doubler Skin	0.016	Corrosion-resistant Steel	
50	192-21342-37 ‡192-21002-37	Leading Edge Skin	0.040	2024-T3 Clad	3-8, 3-9, 3-11, 3-12
51	192-21342-33 ‡192-21002-33	Leading Edge Close-Out Skin	0.040	2024-T3 Clad	
52	192-21421-7, 9 ‡192-21003-7	Milled Skin	1.125	7075-T6	3-10, 3-10A, 3-18, 3-38
53	180-21403-101 ‡180-21403-131	Horizontal Stabilizer Beam			3-16A
54	192-21308	Fairing	0.051	2024-T4	
55	192-21309	Fairing—Cover	0.051	2024-T4	
56	192-21310	Fairing	0.051	2024-T4	
57	192-21003-23 ‡192-21002-23	TE Close-Out Skin	0.032	2024-T3 Clad	
58	192-21003-21 ‡192-21002-47	TE Skin	0.032	2024-T3 Clad	
59	192-21311	Tip			3-16

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.
‡ These part numbers are for F-100F-20 Airplanes only.

Key to Figure 3-7

HORIZONTAL STABILIZER NEGLIGIBLE DAMAGE

EMPENNAGE

CRACKS

- Cracks in the leading edge, trailing edge, and tip assembly may be considered negligible damage when the total length of all cracks in any one bay does not exceed one inch. Cracks must be at least 12 inches apart and 3/4 inch from any row of rivets.
- Cracks in the main box structure may be considered negligible when the total length of all cracks is less than 1/2 inch, provided the cracks are at least one inch apart.

NOTE All cracks must be stop drilled except nose rib negligible damage cracks (Refer to figure 3-17) which are not accessible.

- Cracks which restart after stop-drilling are considered major damage.

HOLES

- Holes may be considered negligible damage which meet the same requirements as cracks, when substituting hole diameters for crack length.

NOTE All holes must be smoothed of rough edges.

- All surface holes should be plugged using a flush nonstructural patch.

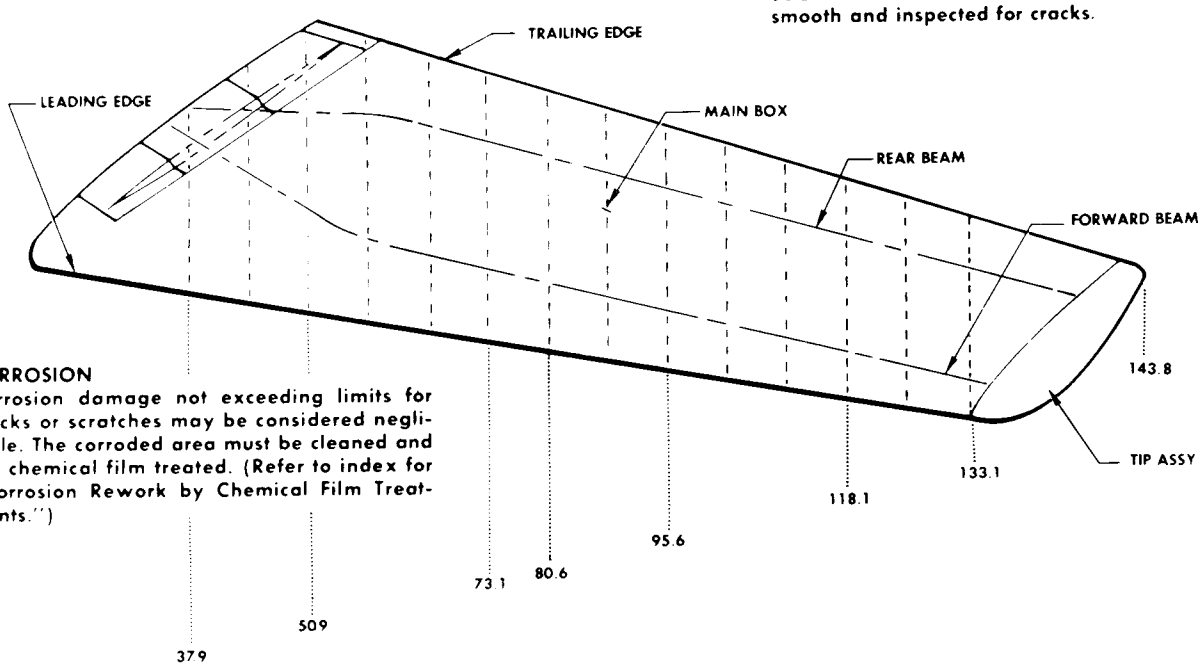
SCRATCHES

- Scratches less than 4-1/2 inches in length which do not exceed 15 percent of the skin thickness may be considered negligible damage.

NOTE All scratches must be polished smooth and inspected for cracks.

CORROSION

Corrosion damage not exceeding limits for cracks or scratches may be considered negligible. The corroded area must be cleaned and the chemical film treated. (Refer to index for "Corrosion Rework by Chemical Film Treatments.")



NEGLIGIBLE DENTS			
STATION	MAX LENGTH CHORDWISE	SPANWISE	TOTAL AREA— ALL DENTS (SQUARE INCHES)
50.9 TO 73.1	1-3/4	5	8-3/4
73.1 TO 80.6	1-1/2	5	7-1/2
80.6 TO 95.6	1-1/16	5	5-1/4
95.6 — 118.1	5/8	5	3-7/16
118.1—133.1	1/2	5	2-1/2
133.1—143.8	1/4	5	1-1/4

NOTE Dents may not exceed 1/16 inch depth.

- Dents in stringers are considered major damage.
- Dents having a radius of less than 0.125 inch are considered a sharp-corner dent (major damage).

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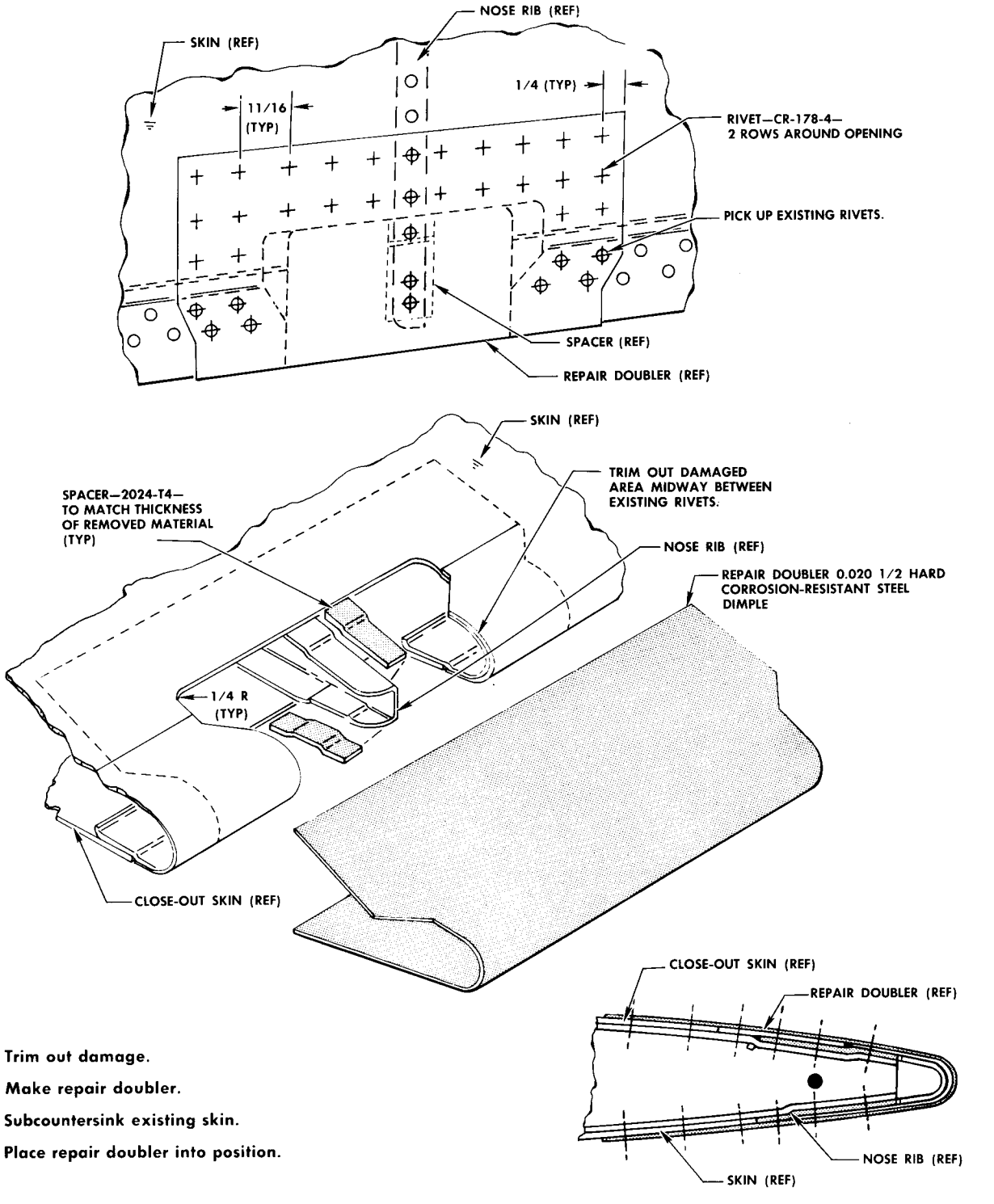
Figure 3-7A. Horizontal Stabilizer Negligible Damage

TEMPORARY—CATEGORY A

EMPENNAGE



1. Trim out damage.
2. Make repair doubler.
3. Subcountersink existing skin.
4. Place repair doubler into position.

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Figure 3-8. Temporary Repair of Horizontal Stabilizer Leading Edge Damaged Across One Rib

TEMPORARY—CATEGORY A

EMPENNAGE

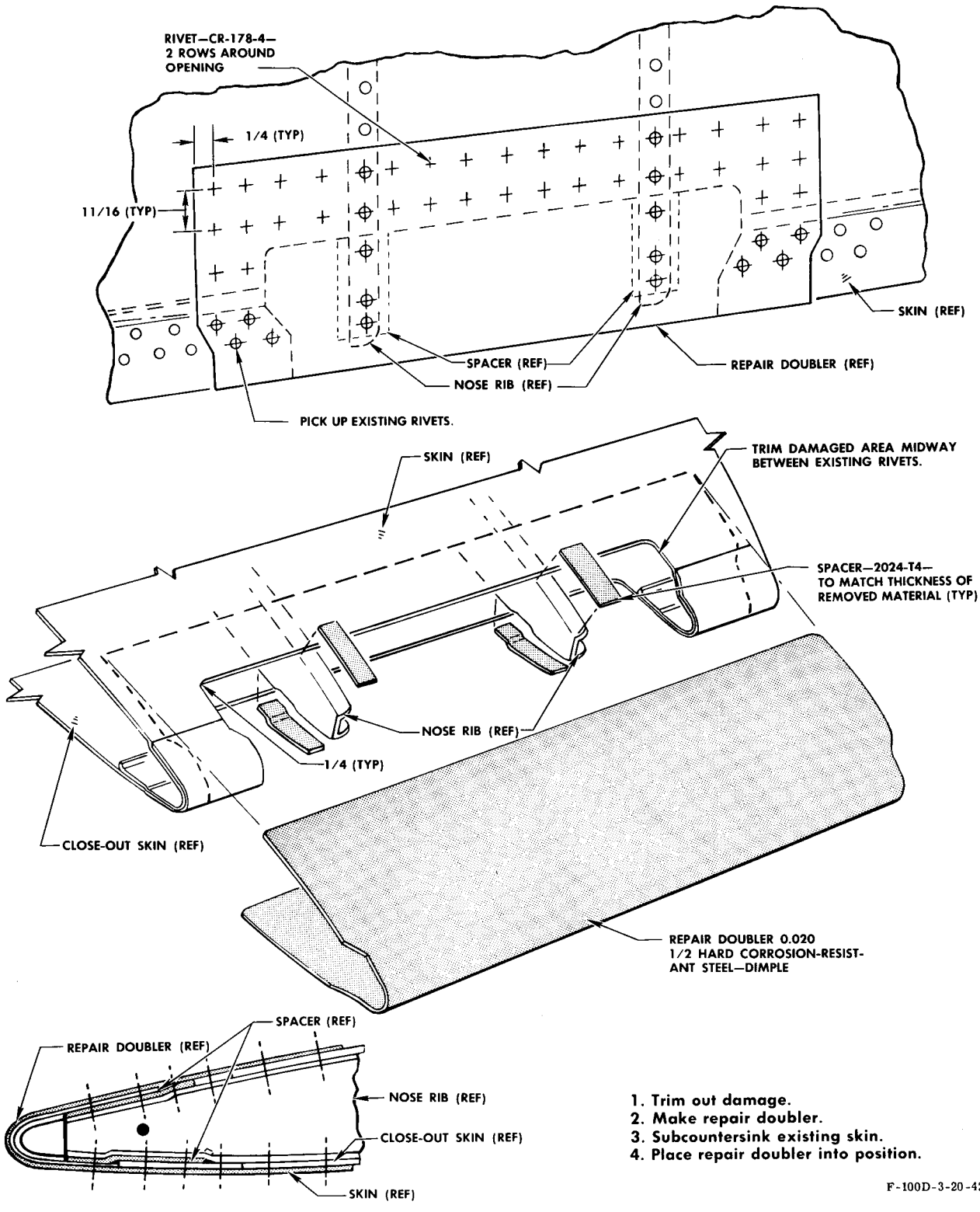
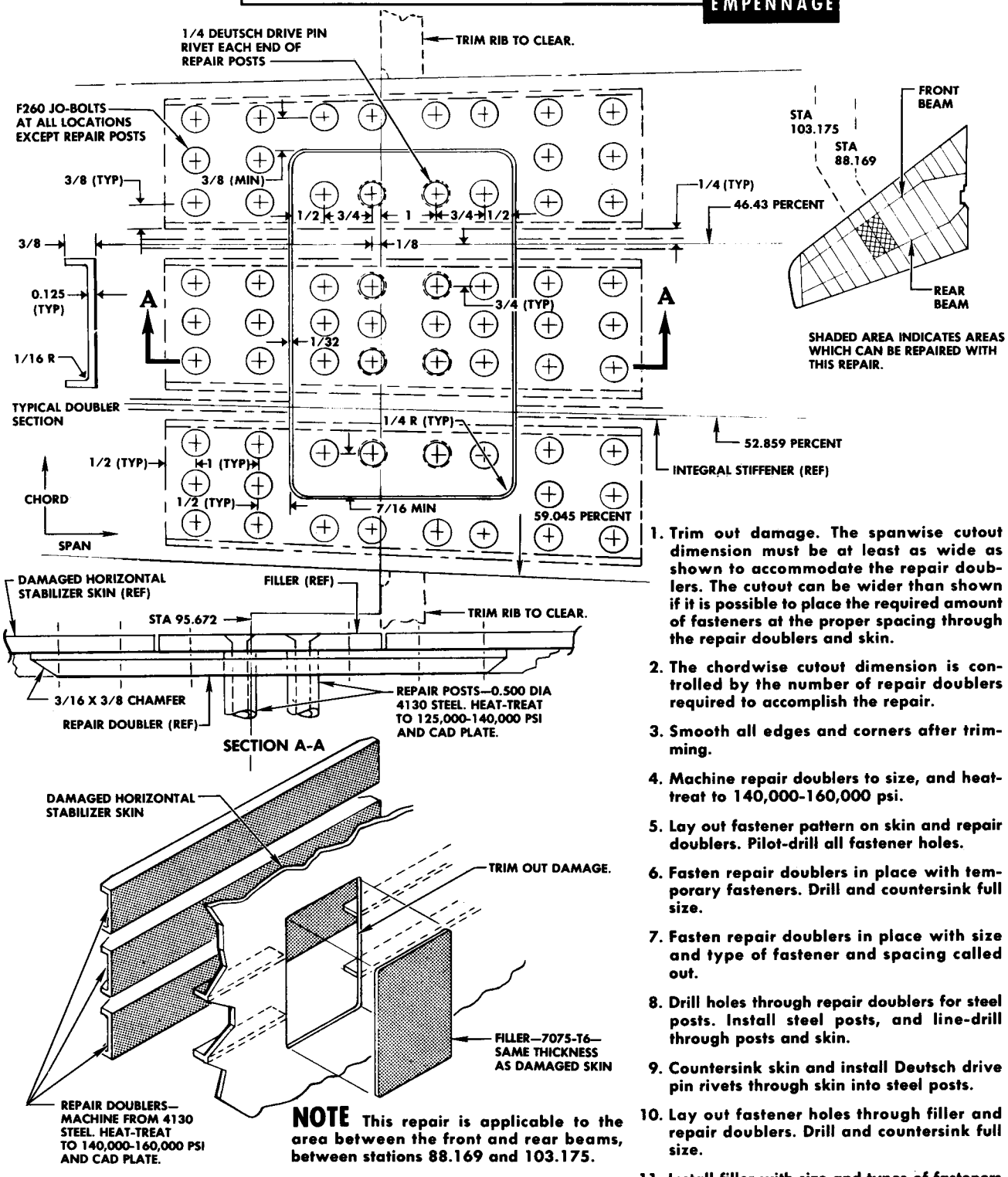


Figure 3-9. Temporary Repair of Horizontal Stabilizer Leading Edge Damaged Across Two Ribs

PERMANENT CATEGORY B — EMPENNAGE



1. Trim out damage. The spanwise cutout dimension must be at least as wide as shown to accommodate the repair doublers. The cutout can be wider than shown if it is possible to place the required amount of fasteners at the proper spacing through the repair doublers and skin.
2. The chordwise cutout dimension is controlled by the number of repair doublers required to accomplish the repair.
3. Smooth all edges and corners after trimming.
4. Machine repair doublers to size, and heat-treat to 140,000-160,000 psi.
5. Lay out fastener pattern on skin and repair doublers. Pilot-drill all fastener holes.
6. Fasten repair doublers in place with temporary fasteners. Drill and countersink full size.
7. Fasten repair doublers in place with size and type of fastener and spacing called out.
8. Drill holes through repair doublers for steel posts. Install steel posts, and line-drill through posts and skin.
9. Countersink skin and install Deutsch drive pin rivets through skin into steel posts.
10. Lay out fastener holes through filler and repair doublers. Drill and countersink full size.
11. Install filler with size and types of fasteners and spacing called out.

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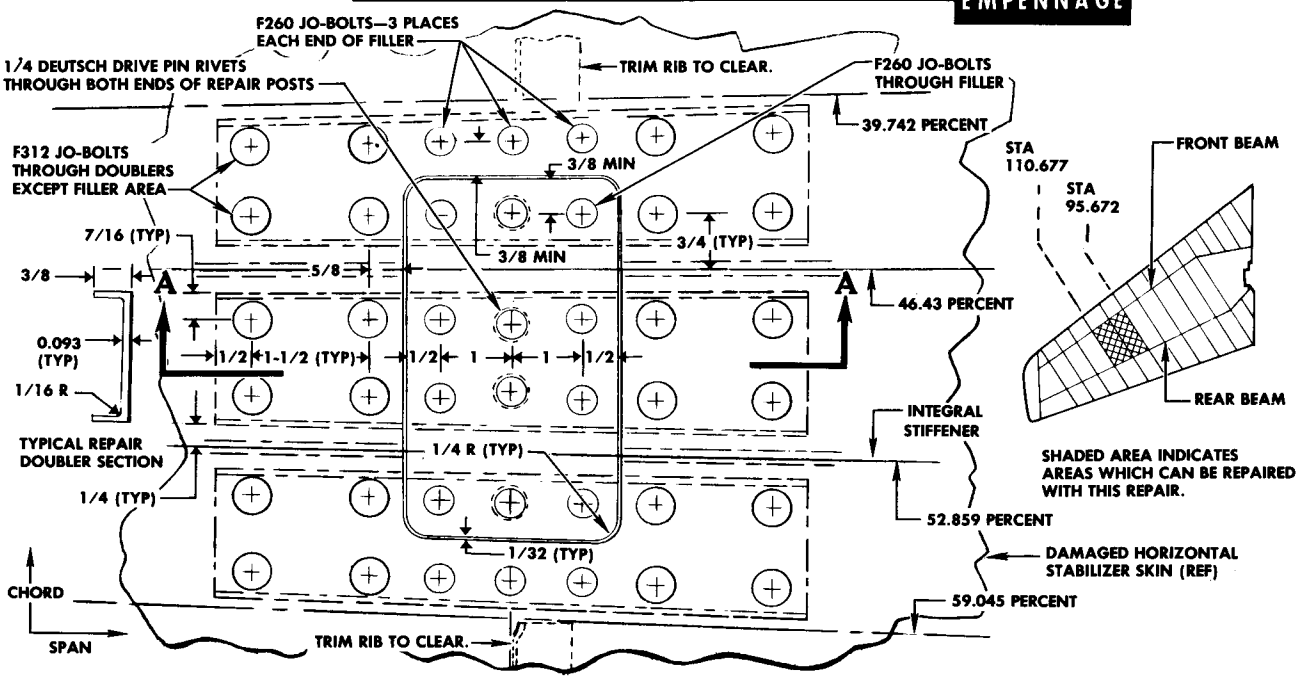
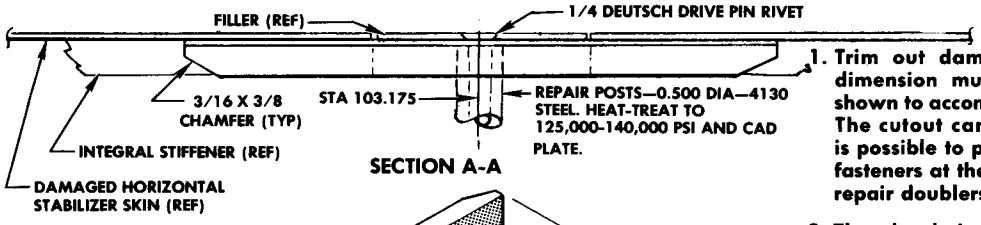
NOTE This repair is applicable to the area between the front and rear beams, between stations 88.169 and 103.175.

- No two adjacent ribs must be cut for these repairs.

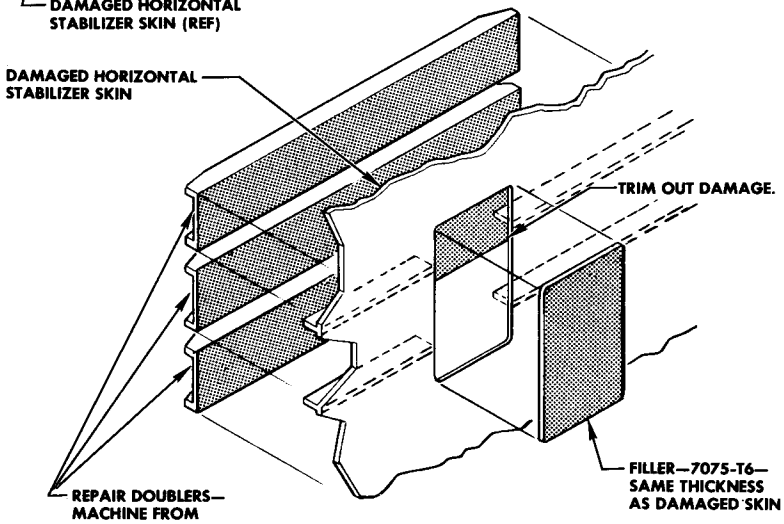
Figure 3-10. Horizontal Stabilizer Milled Skin Repair (Sheet 1 of 4)

PERMANENT CATEGORY B — EMPENNAGE





SECTION A-A



NOTE This repair is applicable to the area between the front and rear beams, between stations 95.672 and 110.677.

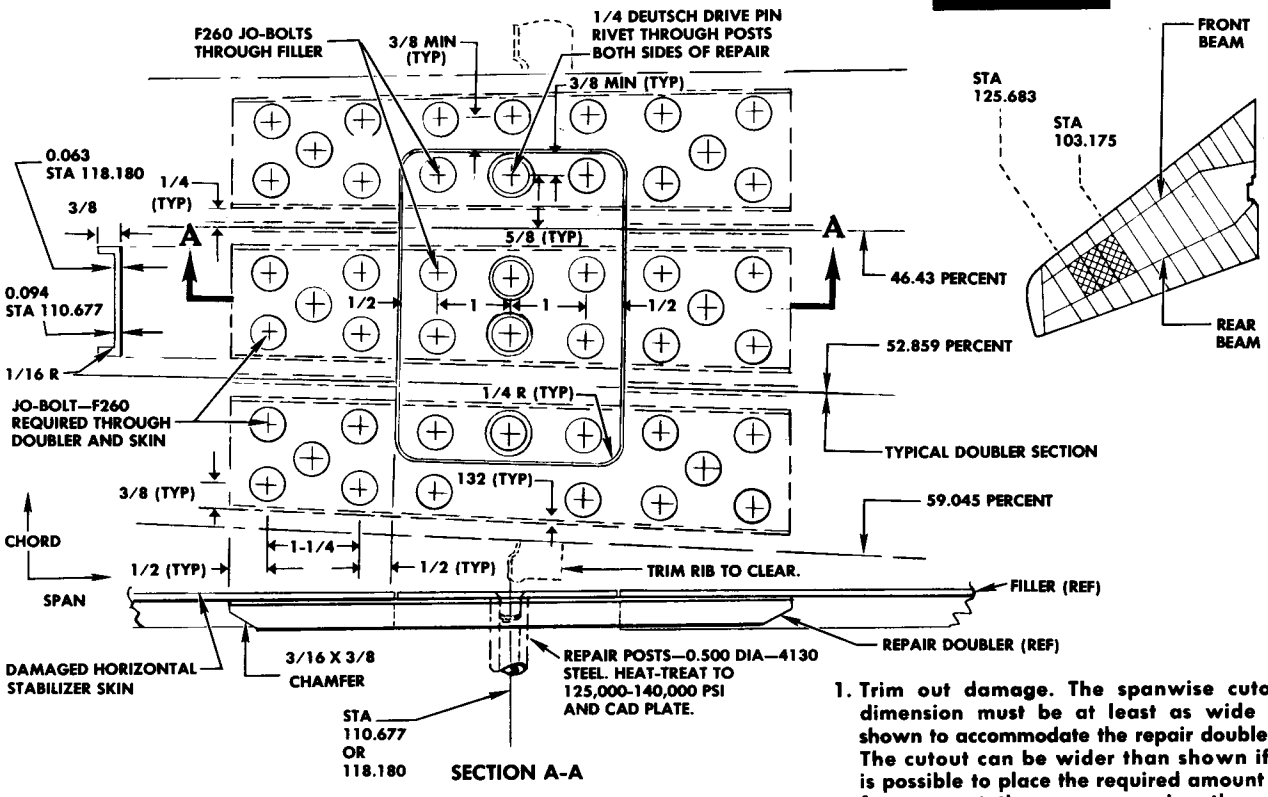
- No two adjacent ribs must be cut for these repairs.

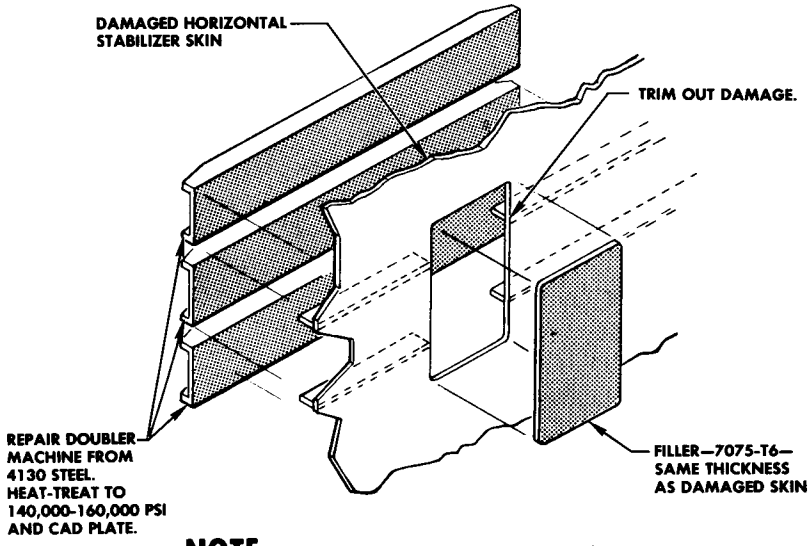
1. Trim out damage. The spanwise cutout dimension must be at least as wide as shown to accommodate the repair doublers. The cutout can be wider than shown if it is possible to place the required amount of fasteners at the proper spacing through the repair doublers and skin.
2. The chordwise cutout dimension is controlled by the number of repair doublers required to accomplish the repair.
3. Smooth all edges and corners after trimming.
4. Machine repair doublers to size, and heat-treat to 140,000-160,000 psi.
5. Lay out fastener pattern on skin and repair doublers. Pilot-drill all fastener holes.
6. Fasten repair doublers in place with temporary fasteners. Drill and countersink full size.
7. Fasten repair doublers in place with size and type of fastener and spacing called out.
8. Drill holes through repair doublers for steel posts. Install steel posts, and line-drill through posts and skin.
9. Countersink skin, and install Deutsch drive pin rivets through skin into steel posts.
10. Lay out fastener holes through filler and repair doublers. Drill and countersink full size.
11. Install filler with size and types of fasteners and spacing called out.

F-100D-3-20-67

Figure 3-10. Horizontal Stabilizer Milled Skin Repair (Sheet 2 of 4)

PERMANENT CATEGORY B — EMPENNAGE





NOTE This repair is applicable to the area between the front and rear beams, between stations 103.175 and 125.683.

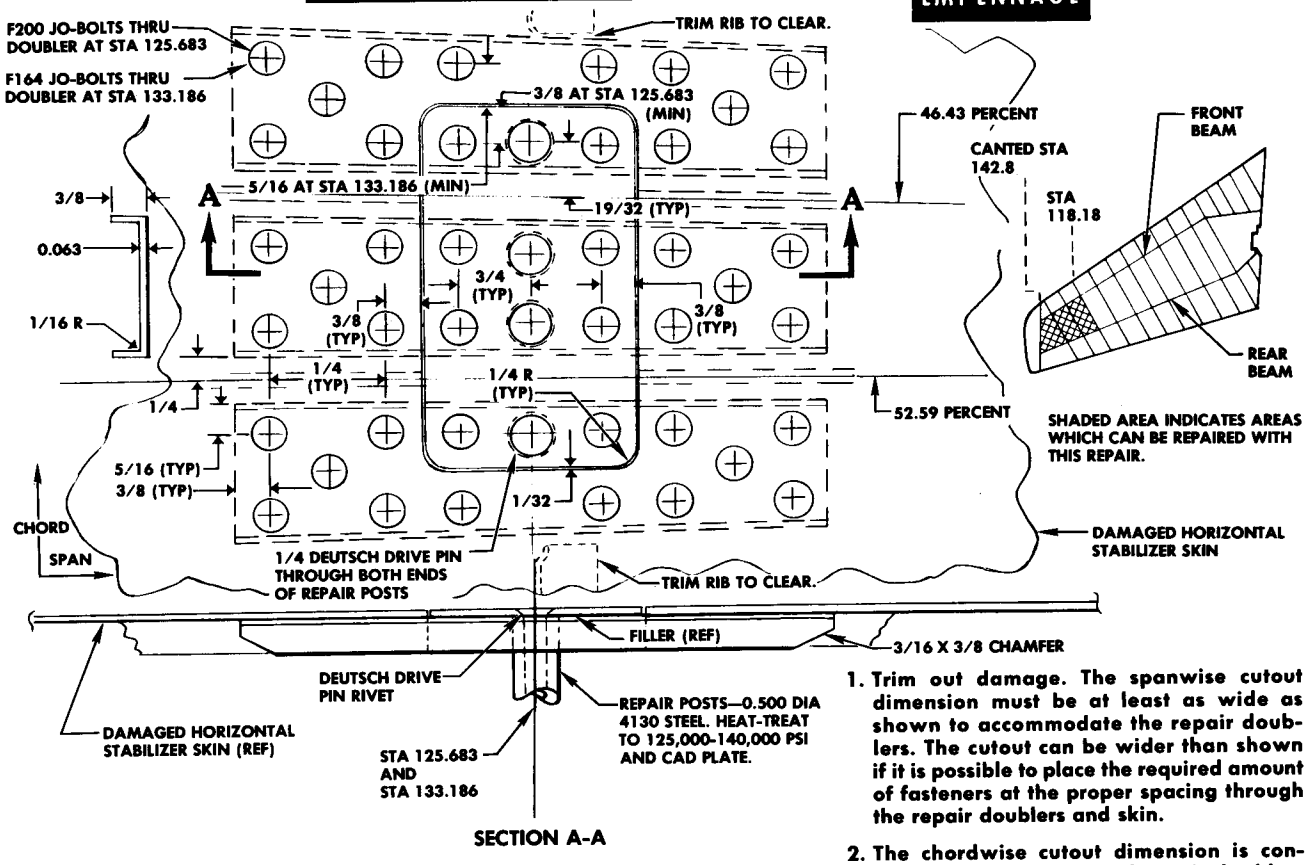
- No two adjacent ribs are to be cut for these repairs.

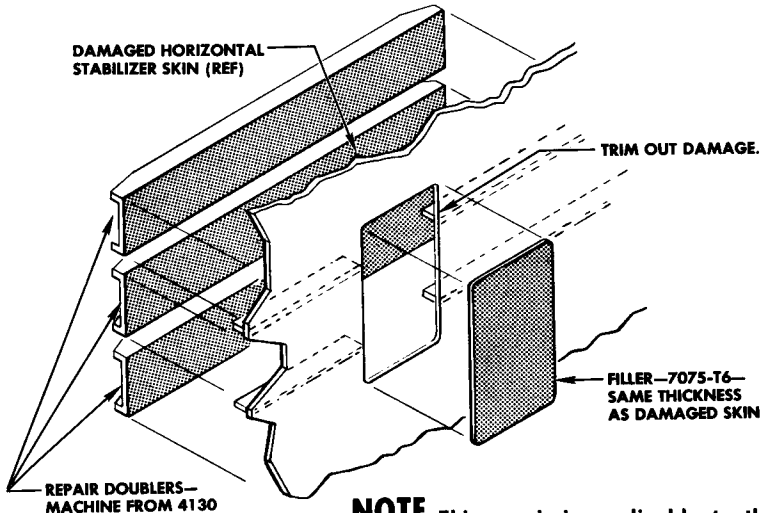
1. Trim out damage. The spanwise cutout dimension must be at least as wide as shown to accommodate the repair doublers. The cutout can be wider than shown if it is possible to place the required amount of fasteners at the proper spacing through the repair doublers and skin.
2. The chordwise cutout dimension is controlled by the number of repair doublers required to accomplish the repair.
3. Smooth all edges and corners after trimming.
4. Machine repair doublers to size, and heat-treat to 140,000-160,000 psi.
5. Lay out fastener pattern on skin and repair doublers. Pilot-drill all fastener holes.
6. Fasten repair doublers in place with temporary fasteners. Drill and countersink full size.
7. Fasten repair doublers in place with the size and type of fastener and spacing called out.
8. Drill holes through repair doublers for steel posts. Install steel posts and line-drill through posts and skin.
9. Countersink skin and install Deutsch drive pin rivets through skin into steel posts.
10. Lay out fastener holes through filler and repair doublers. Drill and countersink full size.
11. Install filler with size and types of fasteners called out.

F-100D-3-20-68

Figure 3-10. Horizontal Stabilizer Milled Skin Repair (Sheet 3 of 4)

PERMANENT CATEGORY B — EMPENNAGE





NOTE This repair is applicable to the area between the front and rear beams, between station 118.18 and canted station 142.8.

- No two adjacent ribs must be cut for these repairs.

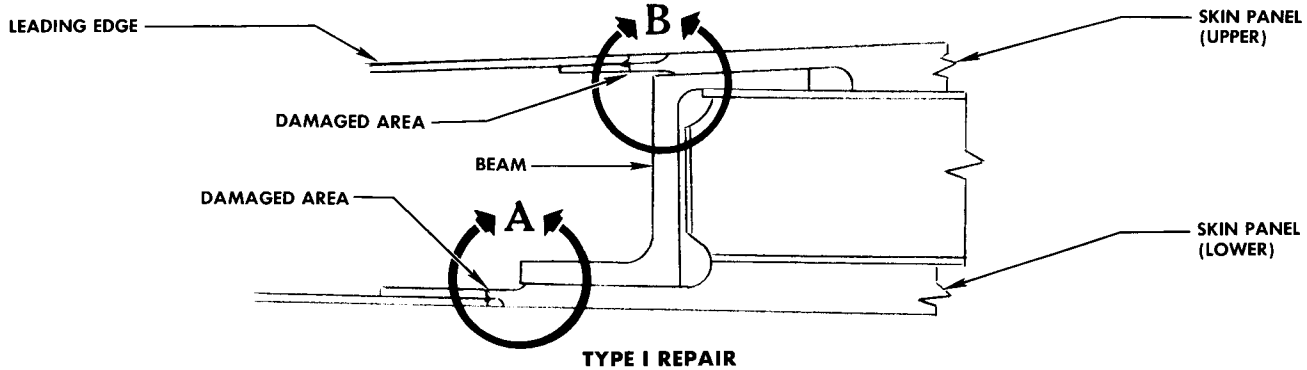
1. Trim out damage. The spanwise cutout dimension must be at least as wide as shown to accommodate the repair doublers. The cutout can be wider than shown if it is possible to place the required amount of fasteners at the proper spacing through the repair doublers and skin.
2. The chordwise cutout dimension is controlled by the number of repair doublers required to accomplish the repair.
3. Smooth all edges and corners after trimming.
4. Machine repair doublers to size, and heat-treat to 140,000-160,000 psi.
5. Lay out fastener pattern on skin and repair doublers. Pilot-drill all fastener holes.
6. Fasten repair doublers in place with temporary fasteners. Drill and countersink full size.
7. Fasten repair doublers in place with size and type of fastener and spacing called out.
8. Drill holes through repair doublers for steel posts. Install steel posts, and line-drill through posts and skin.
9. Countersink skin and install Deutsch drive pin rivets through skin into steel posts.
10. Lay out fastener holes through filler and repair doublers. Drill and countersink full size.
11. Install filler with size and types of fasteners and spacing called out.

F-100D-3-20-69

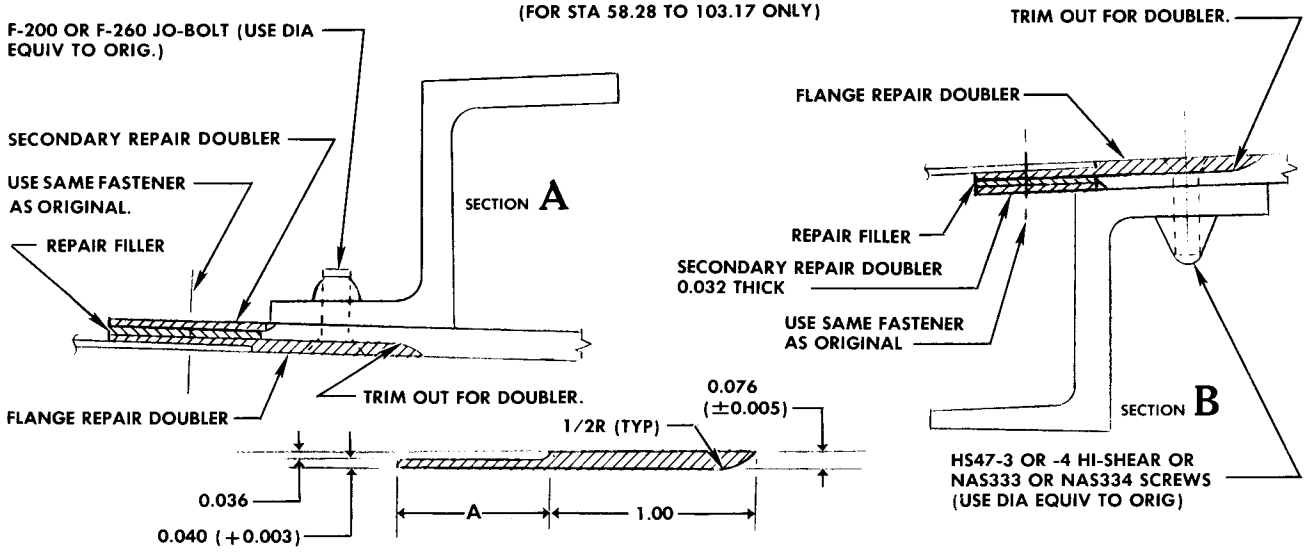
Figure 3-10. Horizontal Stabilizer Milled Skin Repair (Sheet 4 of 4)

PERMANENT—CATEGORY A

EMPENNAGE



TYPE I REPAIR
(FOR STA 58.28 TO 103.17 ONLY)



FLANGE REPAIR DOUBLER DETAIL (TYP)

NOTE No repairs are permitted to the milled skin inboard of station 58.28.

- Type I repair (on sheets 1 and 2) is for damage between stations 58.28 and 103.17 only.
- Type IIA and IIB repairs (on sheets 3 and 4) are for damage outboard of station 103.17.
- Make doublers (except as noted on sheet 4) from 4130 (Specification MIL-S-18729), 4340 (Specification MIL-S-5000), or 301 1/2 H CRES (Specification MIL-S-5059).
- If 4130 or 4340 is used, heat-treat to 180,000 psi and cadmium-plate.
- Finish all doublers, fillers, and repair area with zinc-chromate primer.
- Repairs are applicable to both upper and lower flanges.

TYPE 1 REPAIR PROCEDURE

1. Remove leading edge assembly to gain access to damaged skin flange.
2. Remove damaged portion of skin flange.
3. Machine skin recess to provide for external patch doubler.
4. Make repair doublers and fillers as shown.
5. Make fillers from 0.040-inch 7075-T6 clad.
6. Edge of flange repair doubler may be chamfered to a straight bevel cut in place of the 1/2-inch radius.
7. Dimension A on the flange repair doubler is determined by the width of the skin flange.
8. Overlap flange doubler two rivet spaces and secondary doubler at least three rivet spaces on each end of damaged area.
9. In those areas where a leading edge rib may interfere with installation of secondary doubler, modify rib by rejoggling existing joggle 0.032 inch deeper.

F-100D-3-20-76

Figure 3-10A. Repair of Leading Edge Attaching Flange on Horizontal Stabilizer (Sheet 1 of 4)

PERMANENT—CATEGORY A **EMPENNAGE**

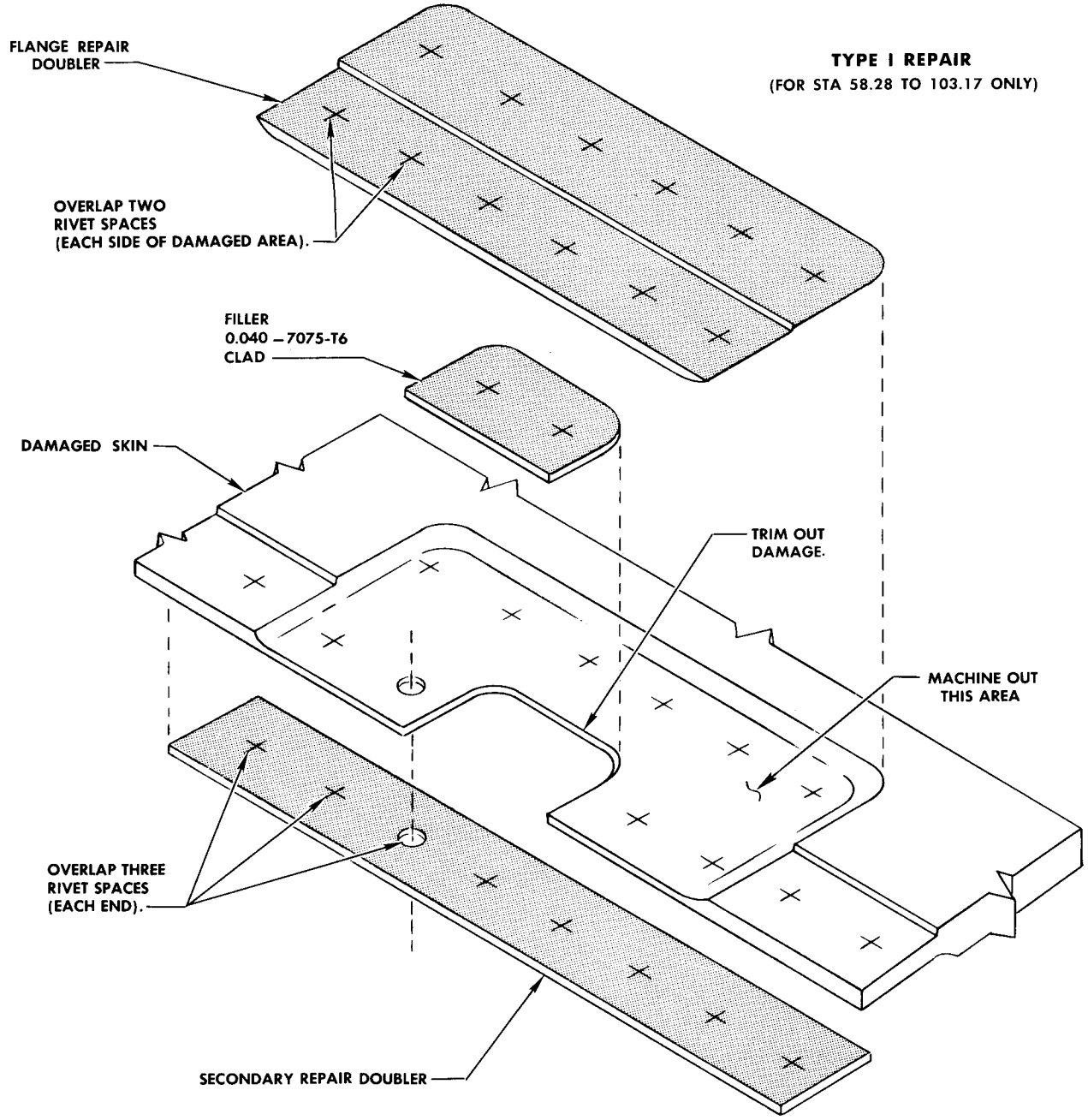
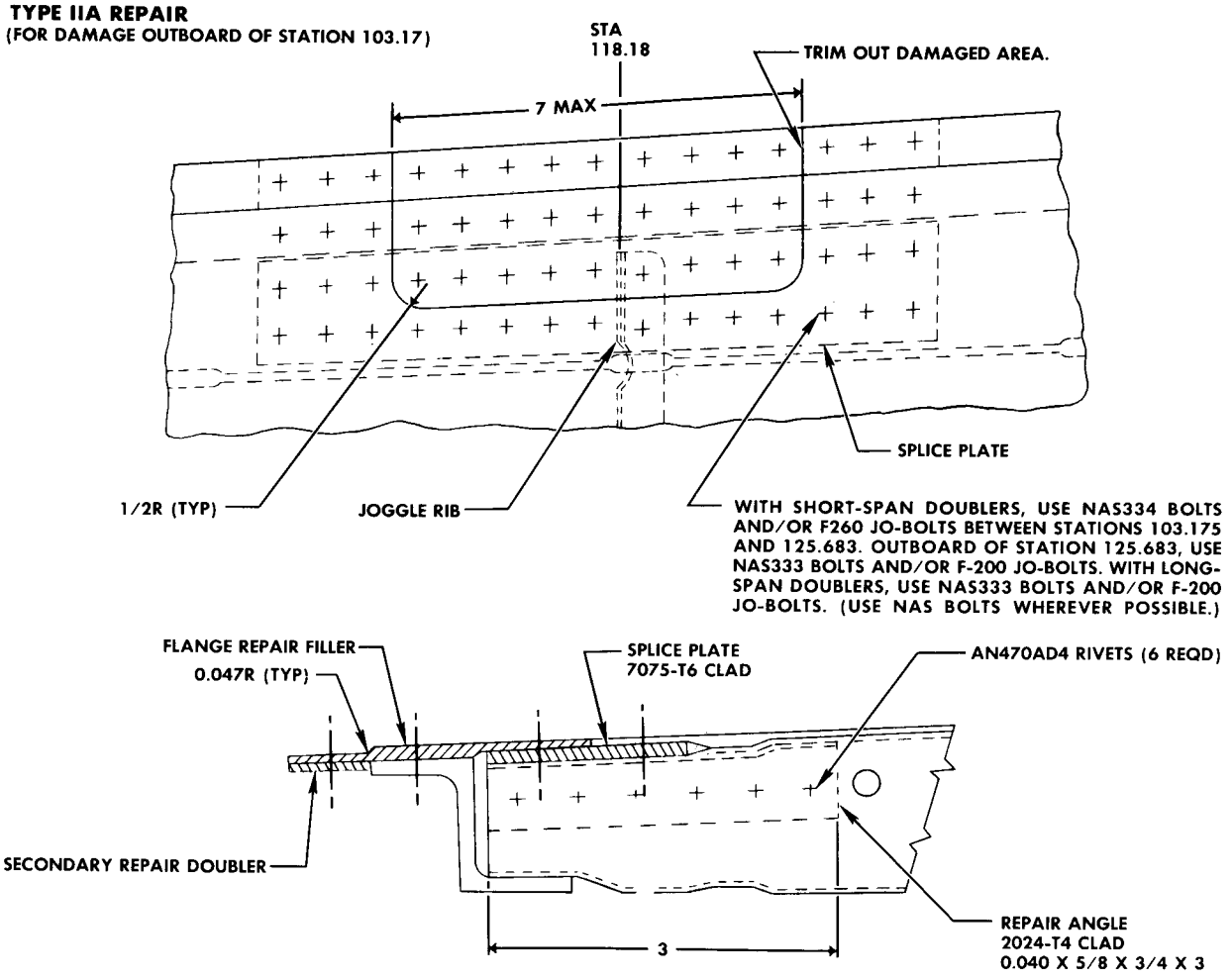


Figure 3-10A. Repair of Leading Edge Attaching Flange on Horizontal Stabilizer (Sheet 2 of 4)

PERMANENT—CATEGORY A

EMPENNAGE

TYPE IIA REPAIR
(FOR DAMAGE OUTBOARD OF STATION 103.17)



NOTE Type IIA and Type IIB repairs are similar. See sheet 4 for Type IIB particulars.

1. Remove leading edge assembly to gain access to damaged skin flange.
2. Remove damaged portion of skin flange and skin.
3. Make doublers and fillers as shown for Type IIA and IIB repairs.
4. A minimum overlap of two rivet spaces at each end of the cut may be used for the repair doubler, provided the entire repair is confined to one rib bay.
5. When repair extends from one rib bay to another rib bay, lap both secondary doubler and splice plate over cutout three rivet spaces on each end.
6. Partial disassembly and removal of the milled skin may be necessary to rework the joggle of the intermediate stabilizer rib. When the damage is in a rib area, rejoggling the rib is required to accommodate the repair doubler. An alternate method is to cut out the existing joggle and install a joggled angle as shown.
7. Lay out fastener hole pattern according to existing rivet pattern through spar flange. (Use AN366F1032 or 428 nut plates with bolts in blind areas.)
8. In those areas where a leading edge rib interferes with installation of secondary doubler, modify rib by rejoggling existing joggle 0.032 inch deeper.

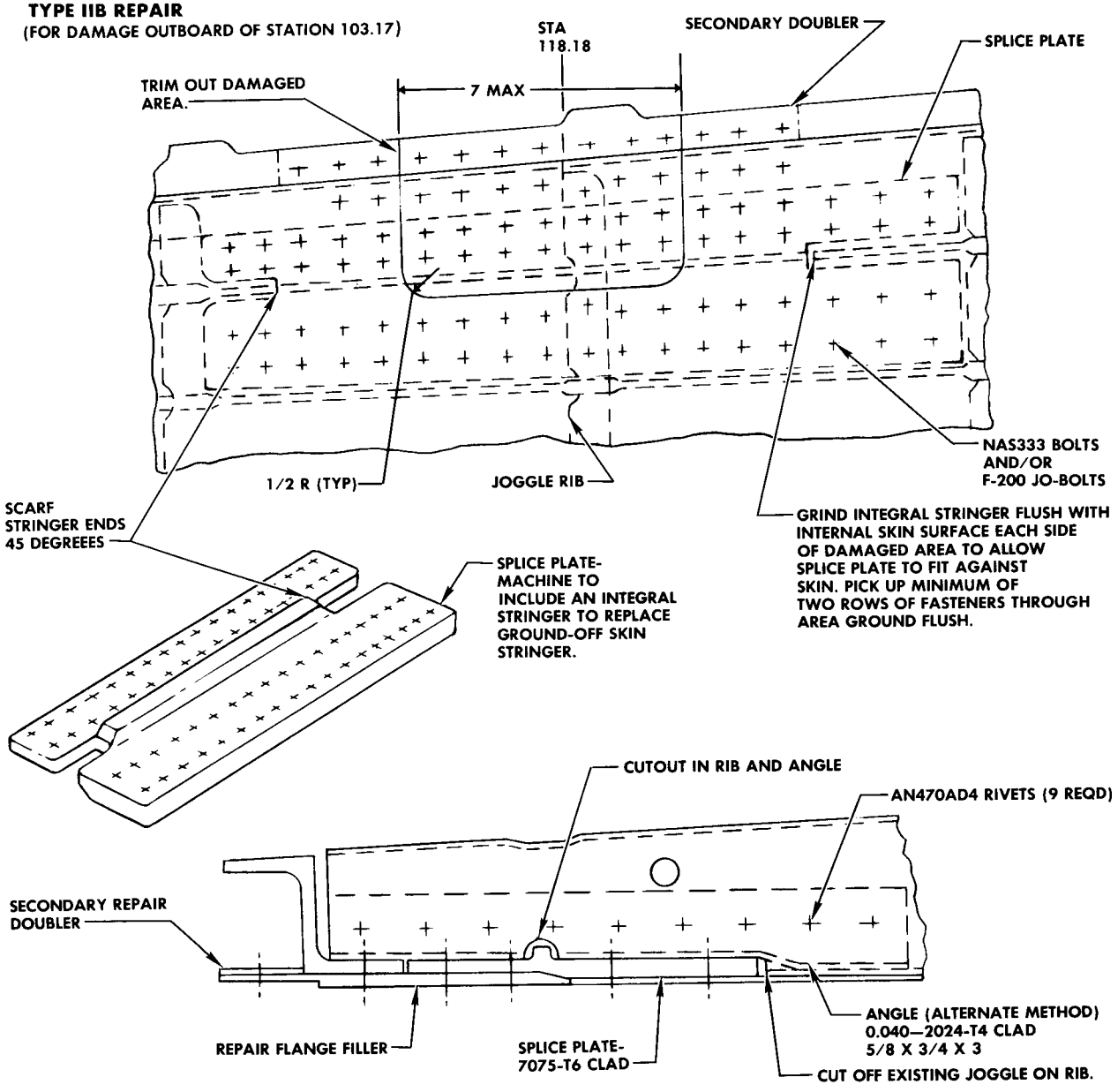
F-100D-3-20-78

Figure 3-10A. Repair of Leading Edge Attaching Flange on Horizontal Stabilizer (Sheet 3 of 4)

PERMANENT—CATEGORY A

EMPENNAGE

TYPE IIB REPAIR
(FOR DAMAGE OUTBOARD OF STATION 103.17)



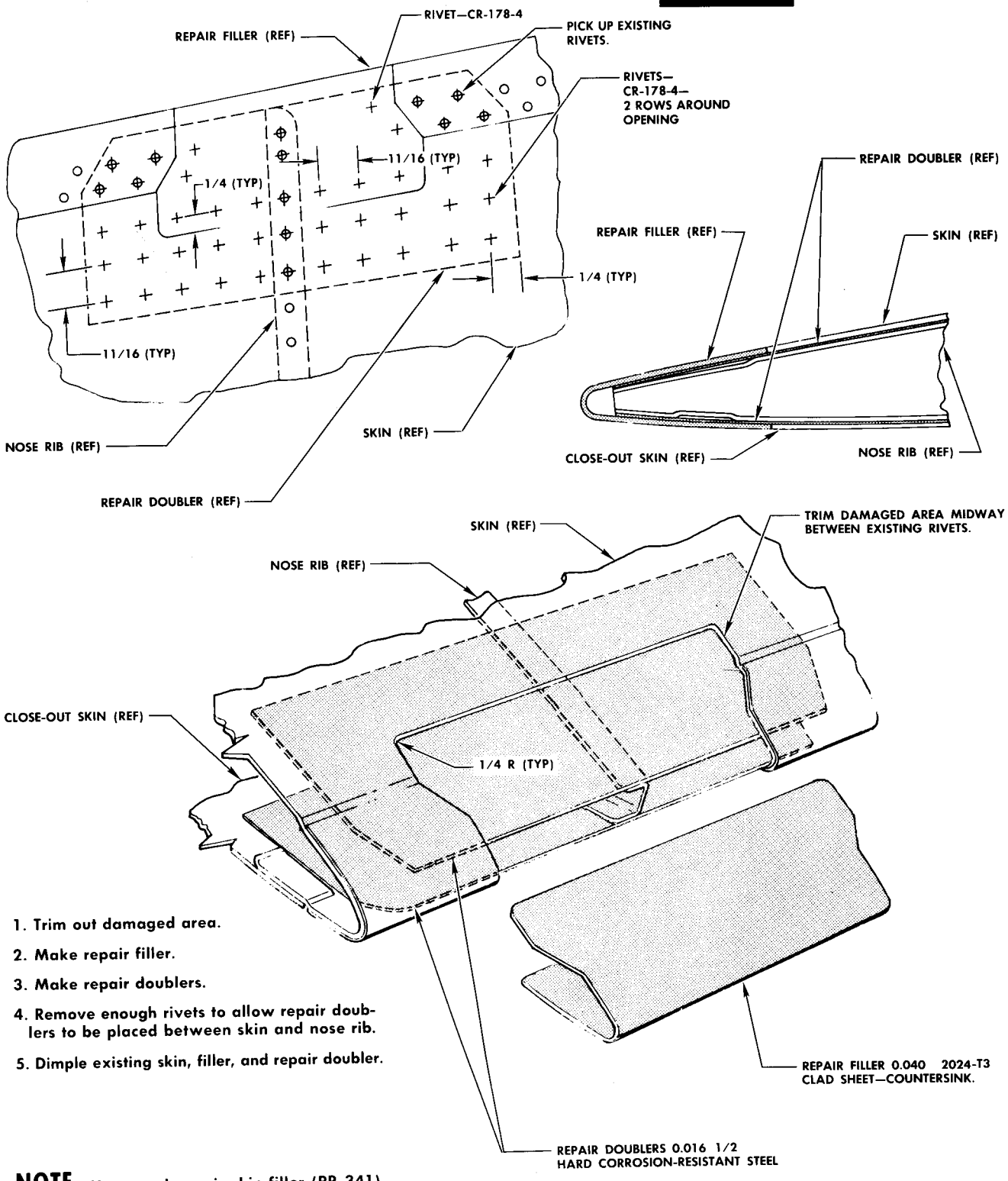
- On applicable Type IIB repairs, grind integral milled skin stringer flush with internal skin surface.
- On Type IIB repairs, the splice plate must extend span- wise on either side of the damage to the adjacent rib.
- Make splice plate from 7075-T6 clad two times skin gage or from steel one gage thicker than skin.

F-100D-3-20-79

Figure 3-10A. Repair of Leading Edge Attaching Flange on Horizontal Stabilizer (Sheet 4 of 4)

PERMANENT - CATEGORY A

EMPENNAGE



1. Trim out damaged area.
2. Make repair filler.
3. Make repair doublers.
4. Remove enough rivets to allow repair doublers to be placed between skin and nose rib.
5. Dimple existing skin, filler, and repair doubler.

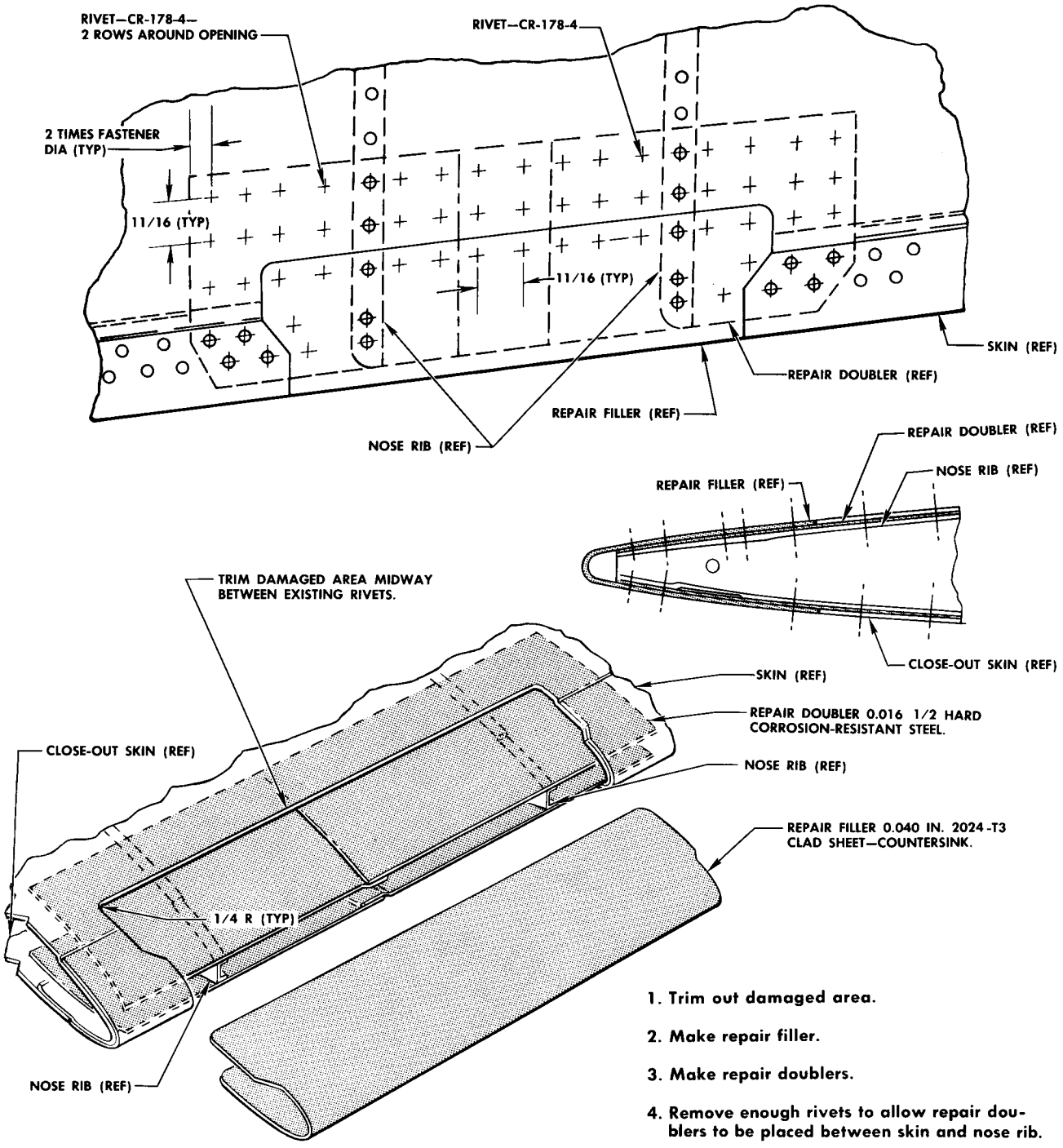
NOTE Use aerodynamic skin filler (PR-341) if gap between existing skin and repair filler is more than 1/16 inch.

F-100D-3-20-40A

Figure 3-11. Permanent Repair of Horizontal Stabilizer Leading Edge Damaged Across One Rib

PERMANENT - CATEGORY A

EMPENNAGE



1. Trim out damaged area.
2. Make repair filler.
3. Make repair doublers.
4. Remove enough rivets to allow repair doublers to be placed between skin and nose rib.
5. Dimple existing skin, filler, and repair doubler.
6. Rivet repair doublers into place.

NOTE

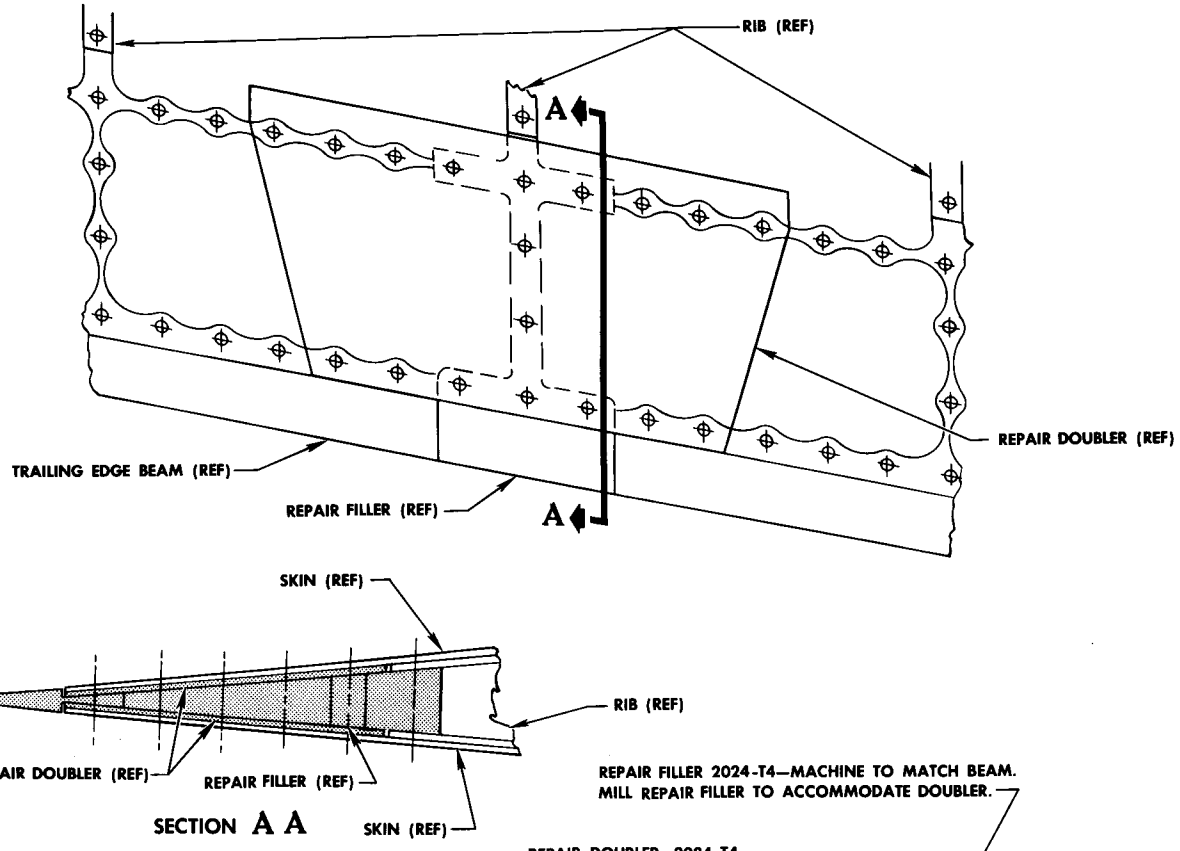
Use aerodynamic skin filler (PR-341) if gap between existing skin and repair filler is more than 1/16 inch.

F-100D-3-20-41A

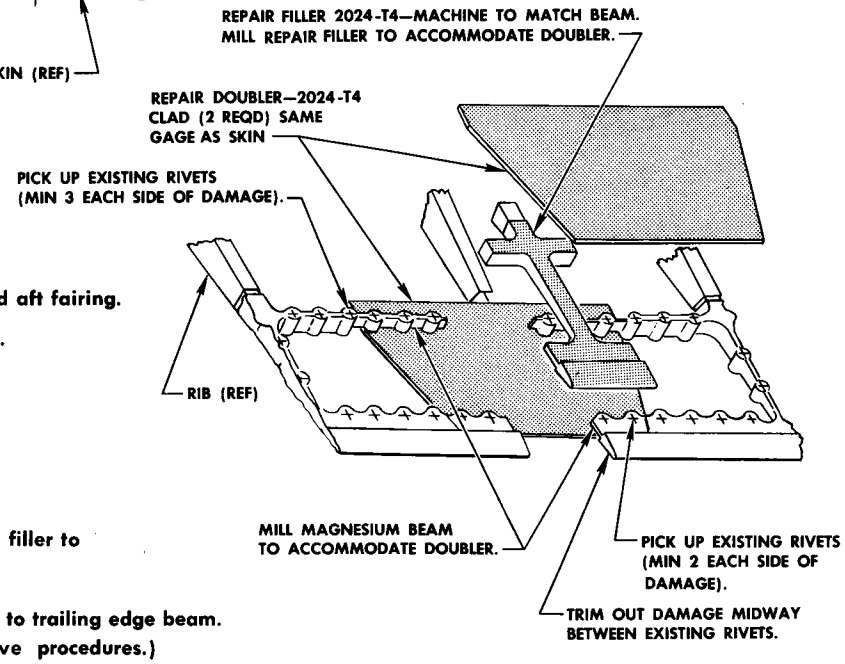
Figure 3-12. Permanent Repair of Horizontal Stabilizer Leading Edge Damaged Across Two Ribs

PERMANENT—CATEGORY A

EMPENNAGE



SECTION A A



1. Refer to text.
2. Remove horizontal stabilizer center and aft fairing.
3. Remove trailing edge skin close-out strip.
4. Remove trailing edge skin.
5. Trim out damaged area.
6. Make repair filler.
7. Make repair doubler.
8. Machine trailing edge beam and repair filler to accommodate doubler.
9. Apply corrosion preventive compound to trailing edge beam. (Refer to index for corrosion preventive procedures.)
10. Place repair filler and repair doubler in position.
11. Install trailing edge skin and close-out strip.
12. Install horizontal stabilizer fairing.

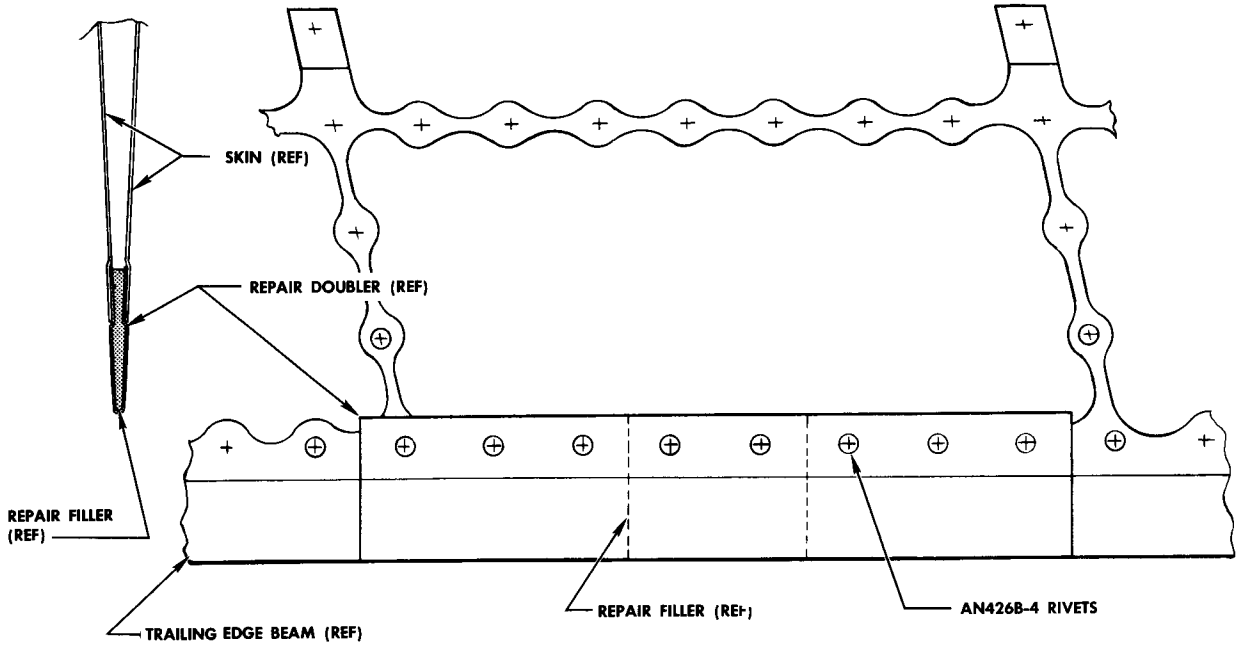
NOTE No more than two repairs may be made per surface. Repairs must be at least 16 inches apart.

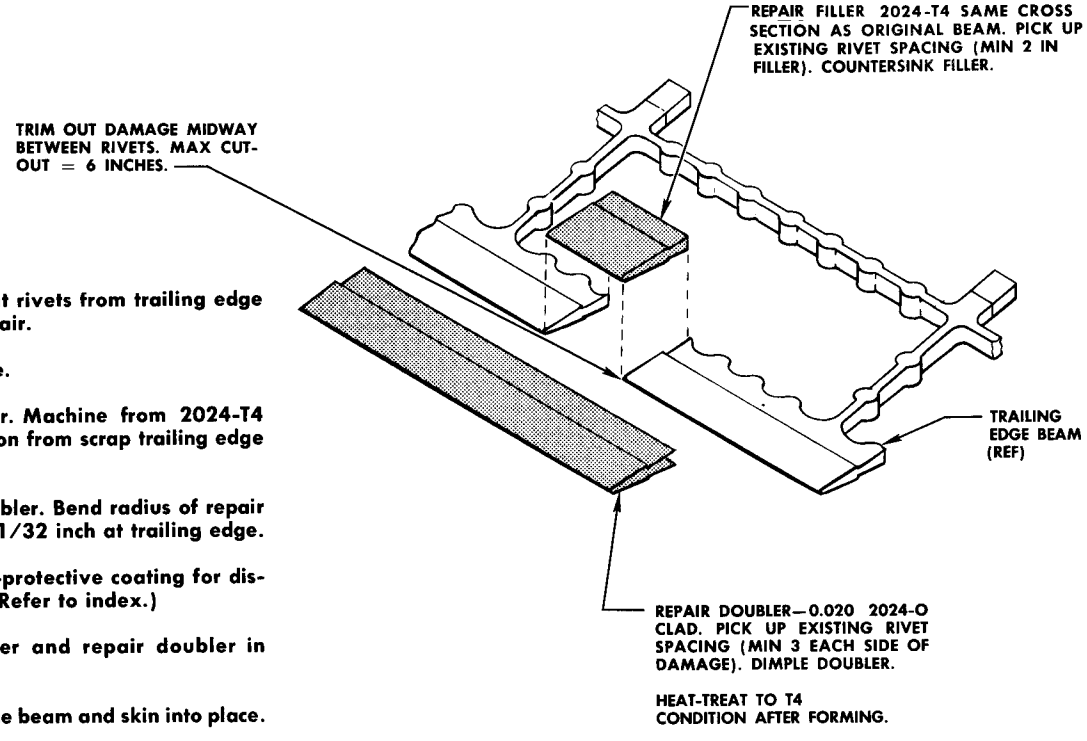
F-100D-3-20-28

Figure 3-13. Horizontal Stabilizer Trailing Edge Repair (Sheet 1 of 5)

PERMANENT—CATEGORY A

EMPENNAGE





1. Remove sufficient rivets from trailing edge skin to make repair.
2. Trim out damage.
3. Make repair filler. Machine from 2024-T4 sheet or cut section from scrap trailing edge beam.
4. Make repair doubler. Bend radius of repair doubler is to be 1/32 inch at trailing edge.
5. Apply corrosion-protective coating for dissimilar metals. (Refer to index.)
6. Place repair filler and repair doubler in position.
7. Rivet trailing edge beam and skin into place.

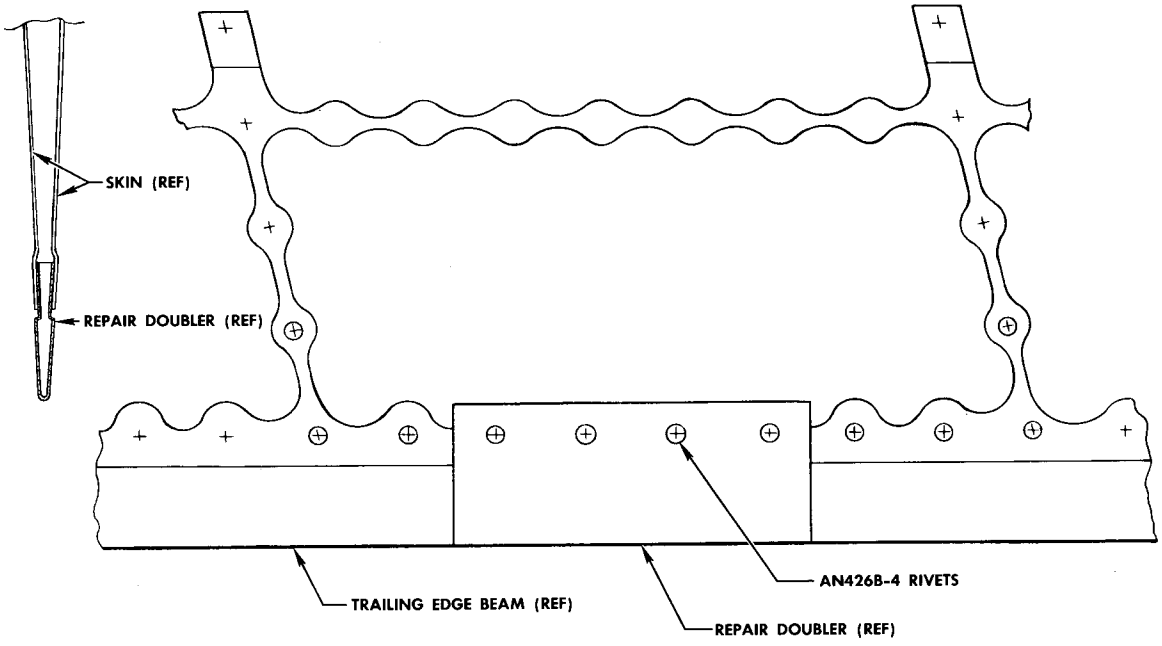
NOTE Max of two repairs per surface. Repairs must be at least 16 inches apart.

F-100D-3-20-27

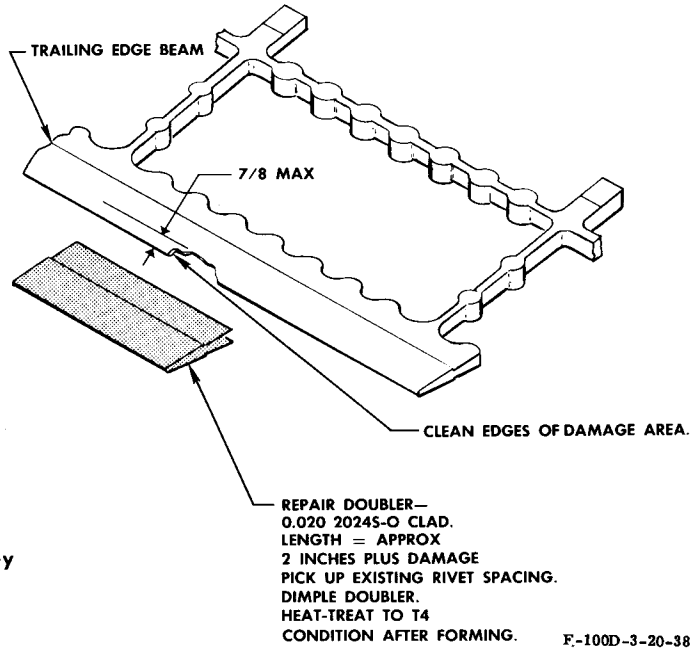
Figure 3-13. Horizontal Stabilizer Trailing Edge Repair (Sheet 2 of 5)

PERMANENT—CATEGORY A

EMPENNAGE



1. Remove sufficient rivets from trailing edge to make repair.
2. Break sharp edges of damage in trailing edge beam.
3. Make repair doubler. Bend radius of repair doubler is to be 1/32 inch at trailing edge.
4. Apply corrosion-protective coating for dissimilar metals. (Refer to index.)
5. Place repair doubler in position.
6. Rivet repair doubler and trailing edge skin.

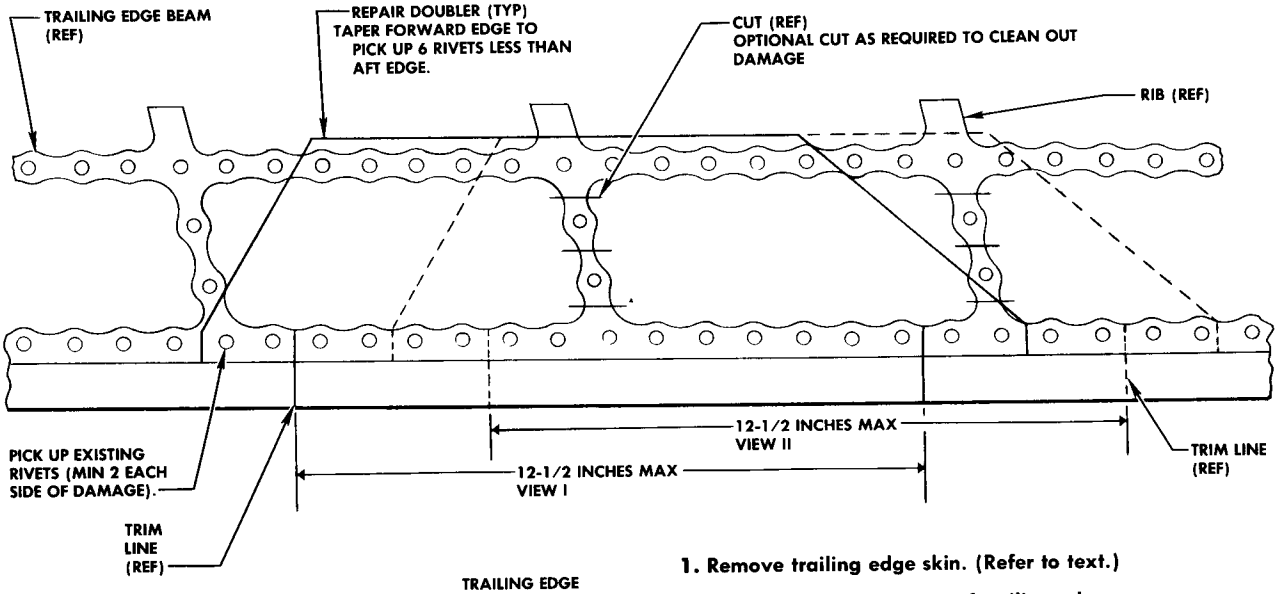


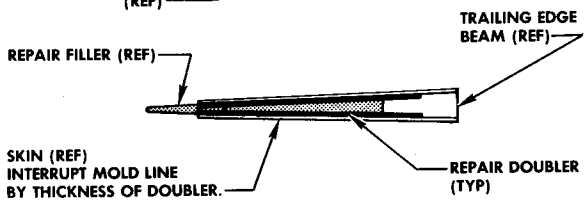
NOTE Max of two repairs per surface; they must be at least 16 inches apart.

Figure 3-13. Horizontal Stabilizer Trailing Edge Repair (Sheet 3 of 5)

PERMANENT—CATEGORY A

EMPENNAGE





NOTE:
This repair may be used with any other trailing edge repair, provided that two undamaged magnesium ribs exist on either side of repair.

- This repair should be used in conjunction with repair of trailing edge skin damage. (Refer to sheet 5 of 5.)

1. Remove trailing edge skin. (Refer to text.)
2. Cut out damaged section of trailing edge.
3. Make repair filler.
4. Make repair doubler.
5. Machine repair filler and trailing edge beam to accommodate doubler.
6. Apply corrosion-preventive compound to trailing edge beam. (Refer to index for corrosion-preventive procedures.)
7. Place repair filler and repair doubler in position.
8. Install trailing edge skin and close-out strip.
9. Install horizontal stabilizer fairing.

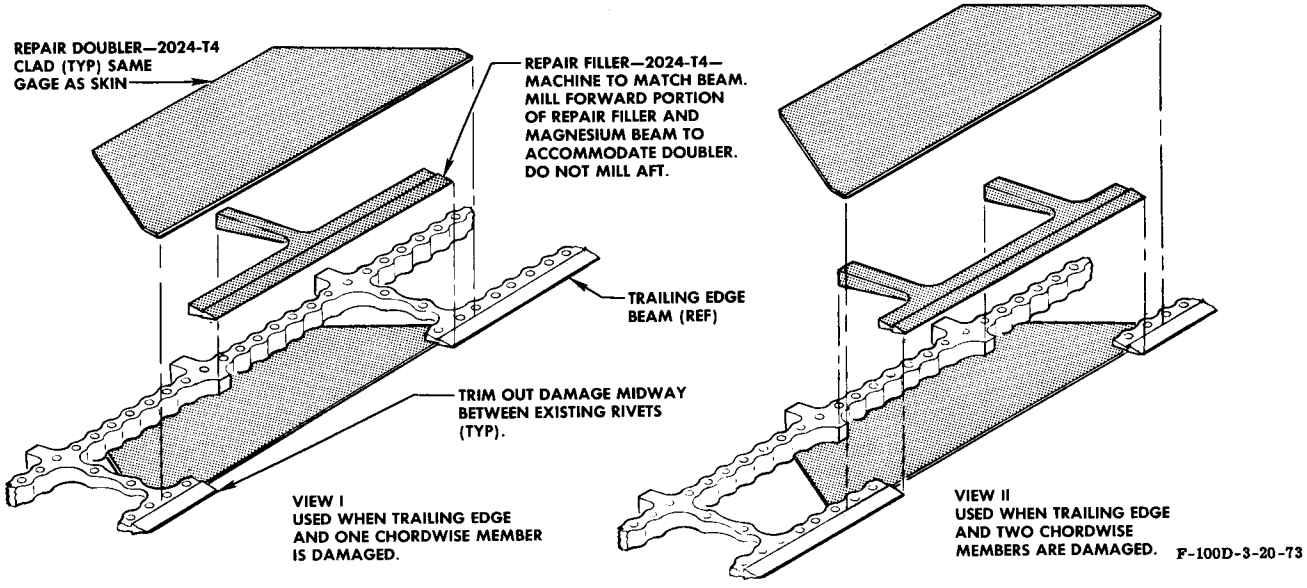
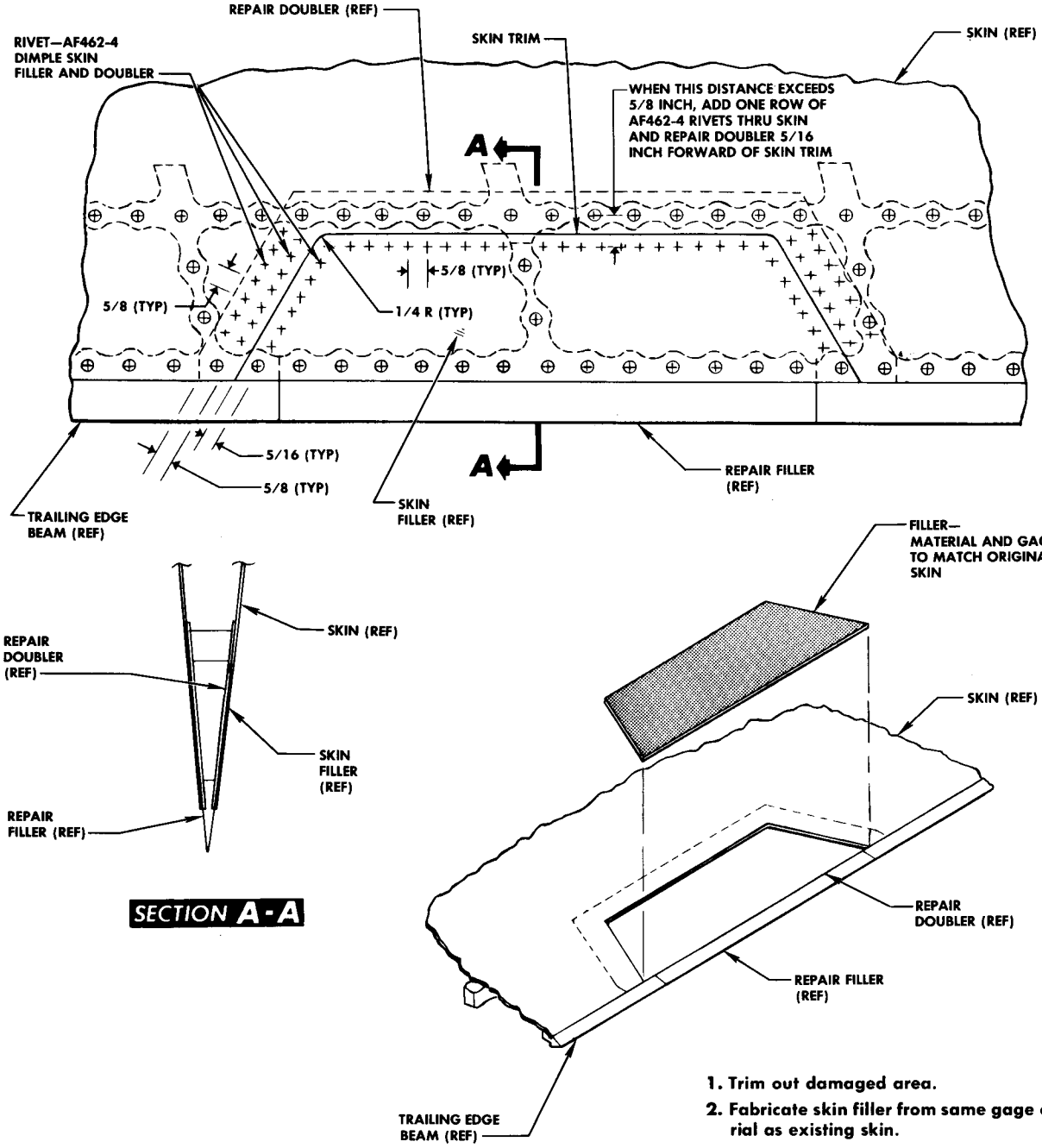


Figure 3-13. Horizontal Stabilizer Trailing Edge Repair (Sheet 4 of 5)

PERMANENT—CATEGORY A

EMPENNAGE



1. Trim out damaged area.
2. Fabricate skin filler from same gage and material as existing skin.
3. Nest skin filler in place. Rivet filler and skin through doubler, using rivet pattern shown and picking up existing rivet locations.

   This repair may be used in conjunction with horizontal stabilizer trailing edge repair (sheet 1 of 5 and sheet 4 of 5).

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Figure 3-13. Horizontal Stabilizer Trailing Edge Repair (Sheet 5 of 5)
(Figures 3-14, 3-15 deleted)

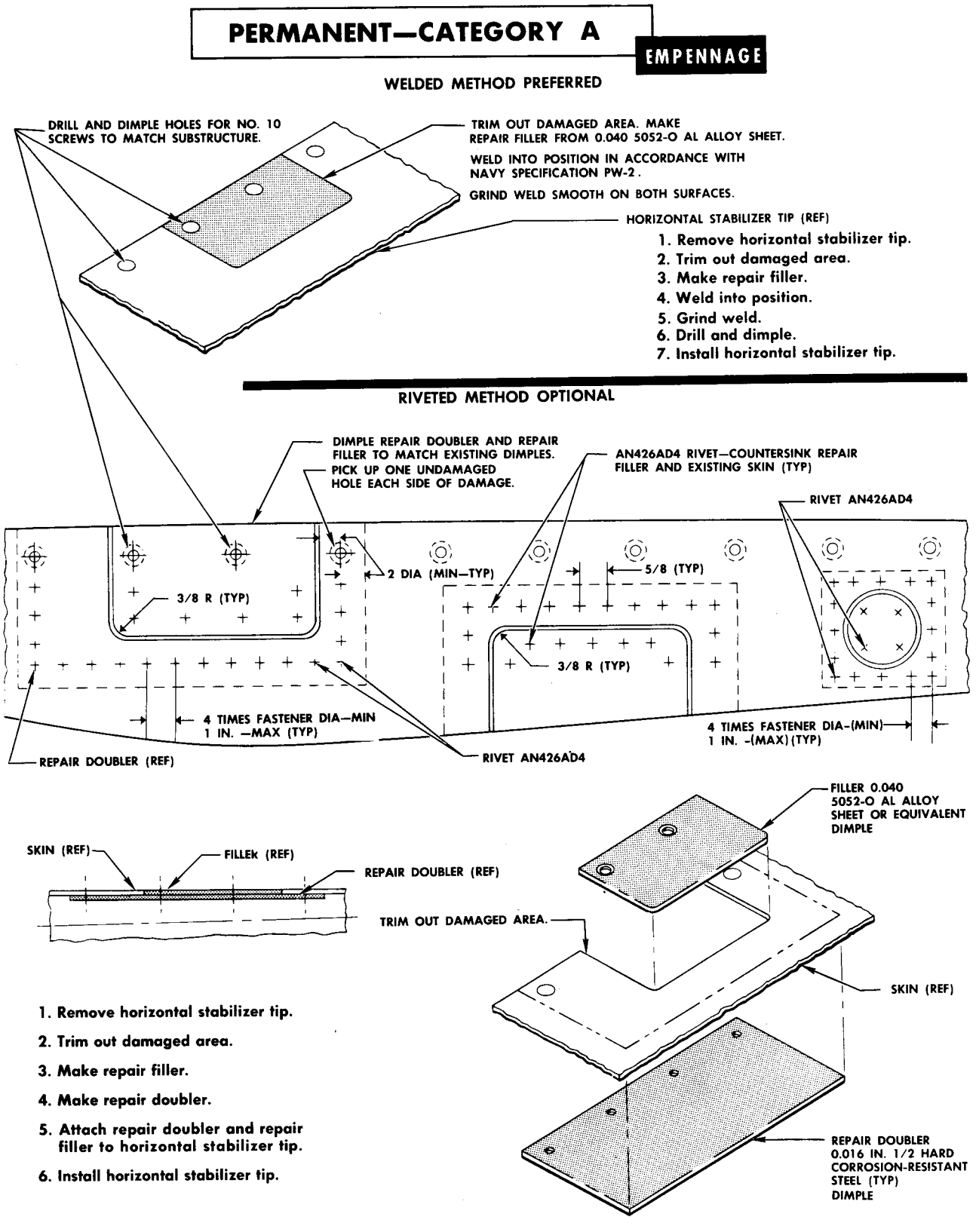
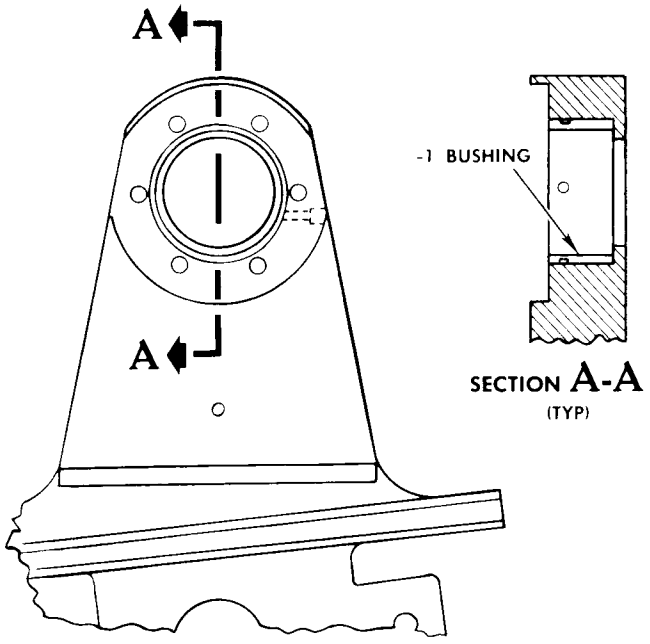


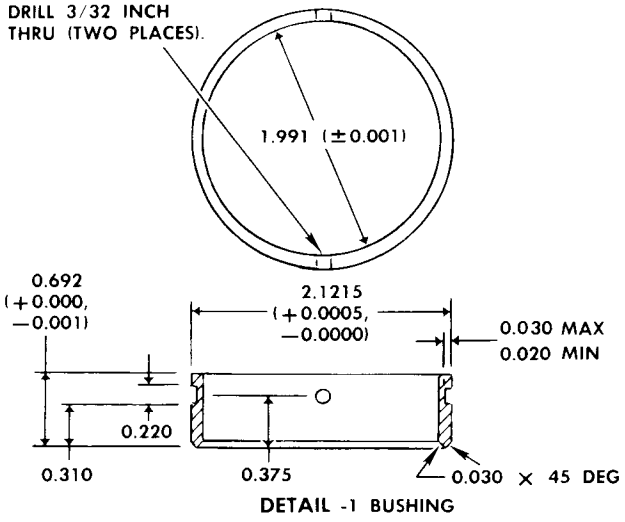
Figure 3-16. Permanent Repair of Horizontal Stabilizer Tip

PERMANENT—CATEGORY A

EMPENNAGE



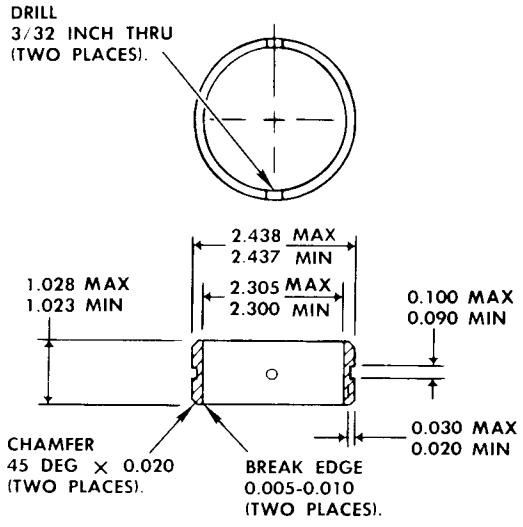
1. Bore 2.120 (±0.0005) inches diameter by 0.692 (+0.000, —0.001 inch deep.
2. Chill and install -1 bushing with chamfer toward center of beam. Maintain 0.001- to 0.003-inch interference fit. Bushing must be flush or below the surface. Angular location of oil holes is not important.
3. Ream installed bushings to 1.9992 (+0.0009, —0.0000) inches.



DETAIL -1 BUSHING

- Dimensions are before plating.
- Material: steel, specification MIL-S-5000 or MIL-S-5626.
- Heat-treat: 160,000-180,000 psi.
- Cadmium-plate all over (0.0001-0.0003 inch thick).

1. Bore 2.4395 (±0.0005) inches diameter.
2. Chill and install a -2 bushing. Maintain 0.001- to 0.003-inch interference fit. Angular location of oil holes is not important.
3. Stake bushing in place on both sides with 1/32-inch minimum radius staking punch.
4. Ream installed bushing to 2.3115 (+0.0010, —0.0000) inches diameter.
5. After bearing installation, carefully stake bearing on both sides. Use -2 bushing material to stake bearing in place.

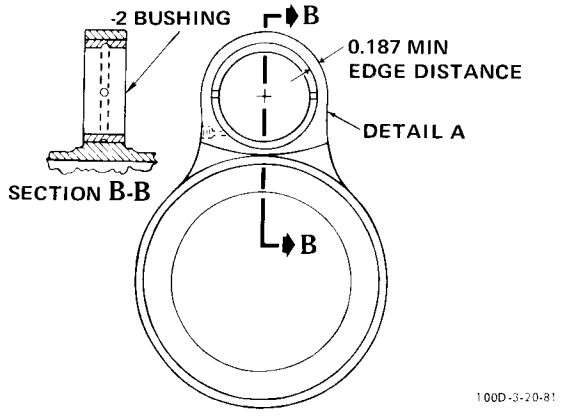


DETAIL -2 BUSHING

- Dimensions are before plating.
- Material: steel, Specification MIL-S-5000 or MIL-S-5626.
- Heat-treat: 160,000-180,000 psi.
- Cadmium-plate all over (0.0001-0.0003 inch thick).

NOTE

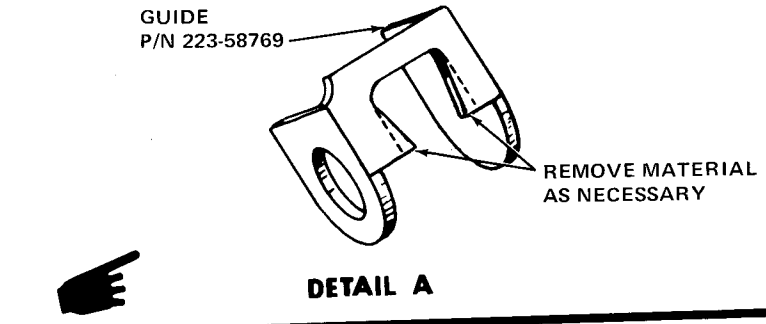
If the bearing cavity edge distance (including the wall thickness of the repair bushing) is under 1/4-inch, the actuator guide, Part No. 223-58769, may tend to bind against and/or ride over the top of the cross beam lug. If this condition is found, the side of the guide which is causing the binding should be reworked as shown.



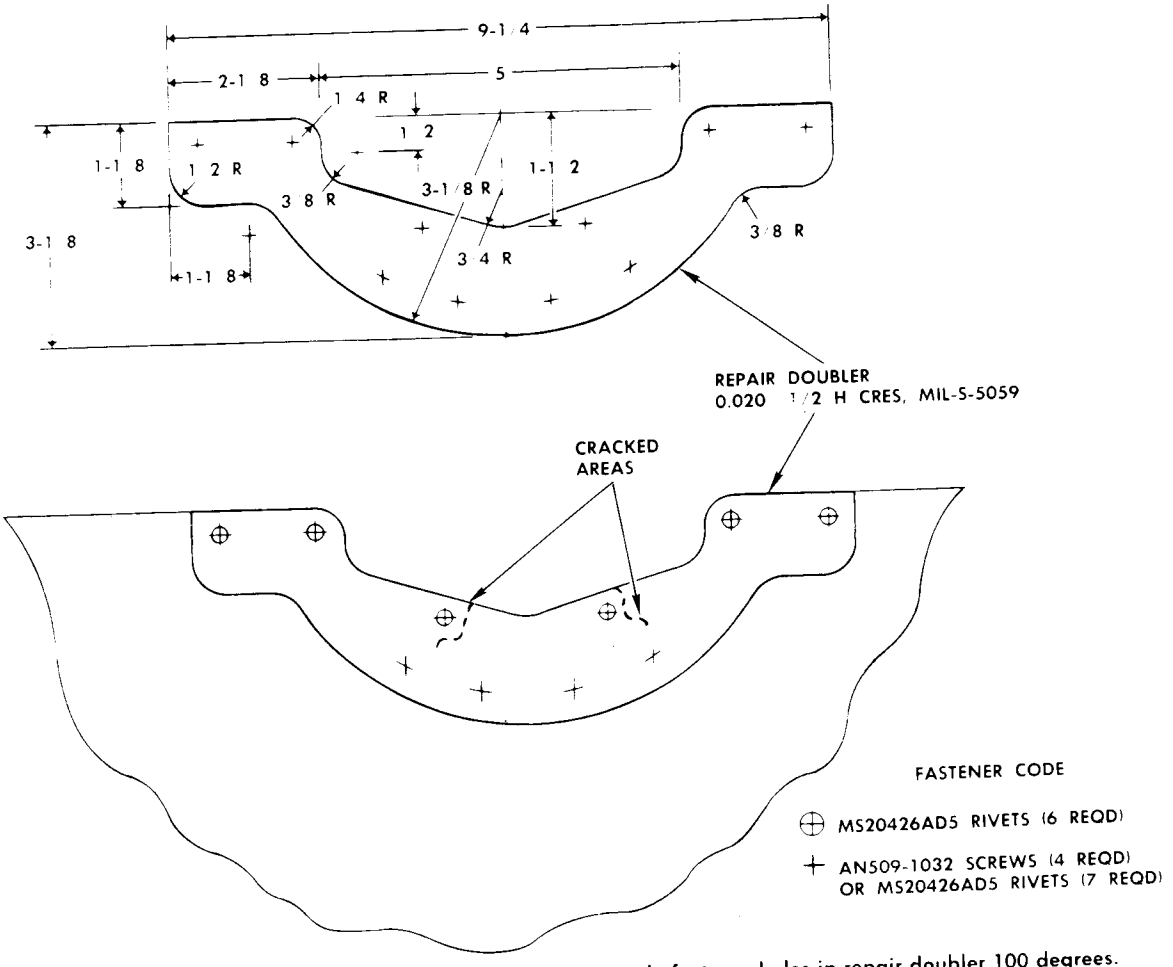
NOTE: SEE DETAIL A PAGE 3-24B

Figure 3-16A. Horizontal Stabilizer Beam Pivot Bearing Seats

PERMANENT—CATEGORY A **EMPENNAGE**



DETAIL A



1. Make repair doubler, using dimensions shown.
2. Remove six existing rivets through upper skin edge of seal assembly.
3. Remove four AN509-1032 screws or seven existing rivets about 2 inches below upper skin edge of seal assembly in center of seal assembly.
4. Place repair doubler in position and pick up existing fastener locations.
5. Dimple fastener holes in repair doubler 100 degrees.
6. Chamfer edges of repair doubler 30 degrees.
7. Finish faying surfaces of seal assembly and repair doubler with zinc-chromate primer (Specification MIL-P-8585).
8. Install repair doubler with same type of fasteners as removed.

Figure 3-16B. Horizontal Stabilizer Trunnion Seal Assembly

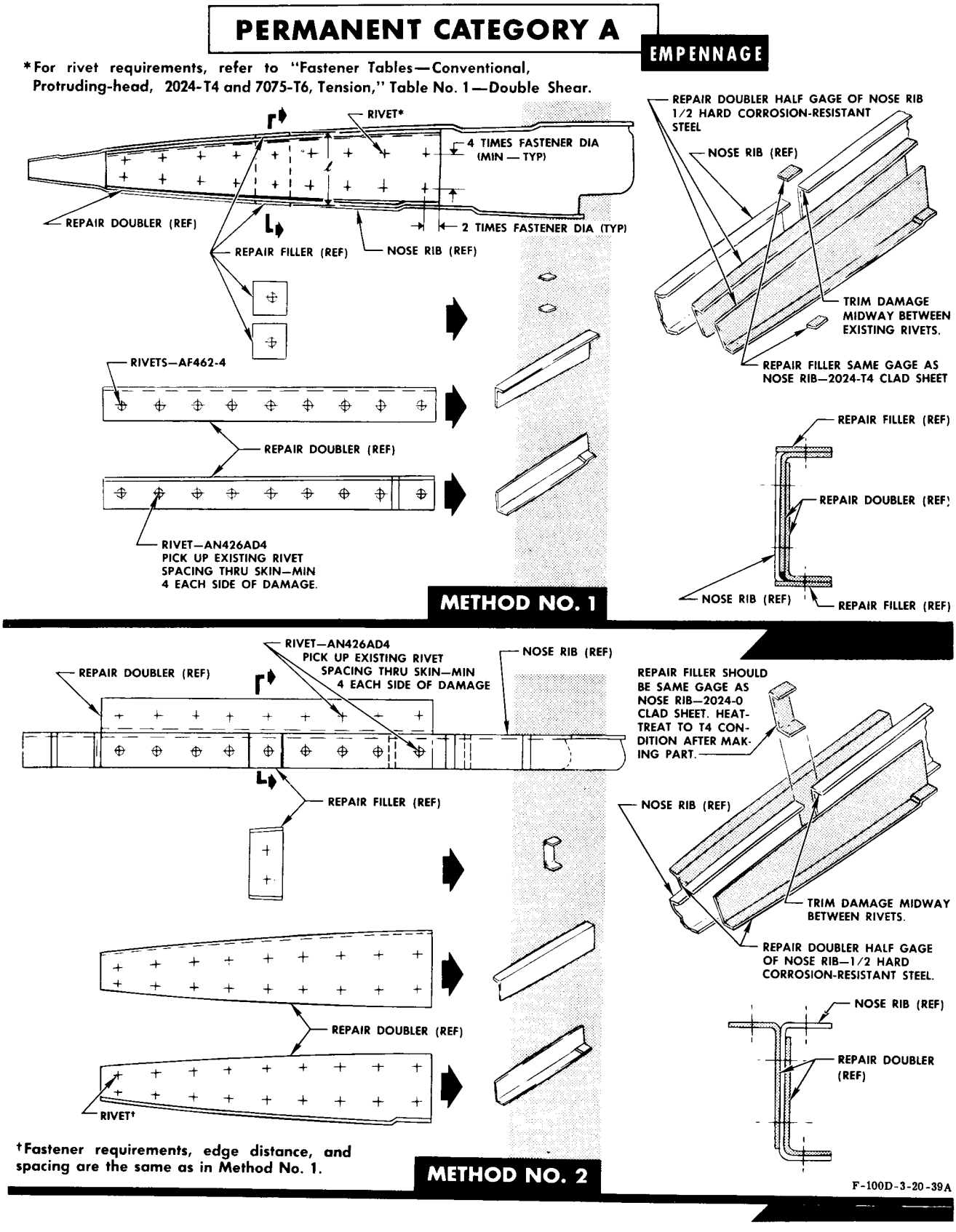
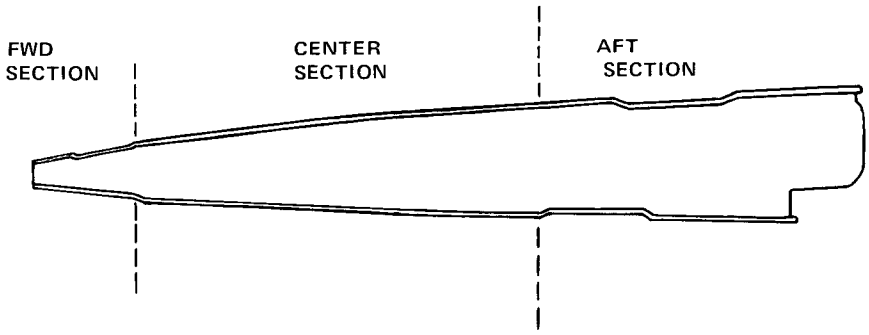


Figure 3-17. Horizontal Stabilizer Nose Rib (Sheet 1 of 2)

PERMANENT—CATEGORY A

EMPENNAGE



THE FOLLOWING REPAIR LIMITS AND PROCEDURES APPLY TO THE FWD AND AFT SECTIONS OF THE NOSE RIBS:

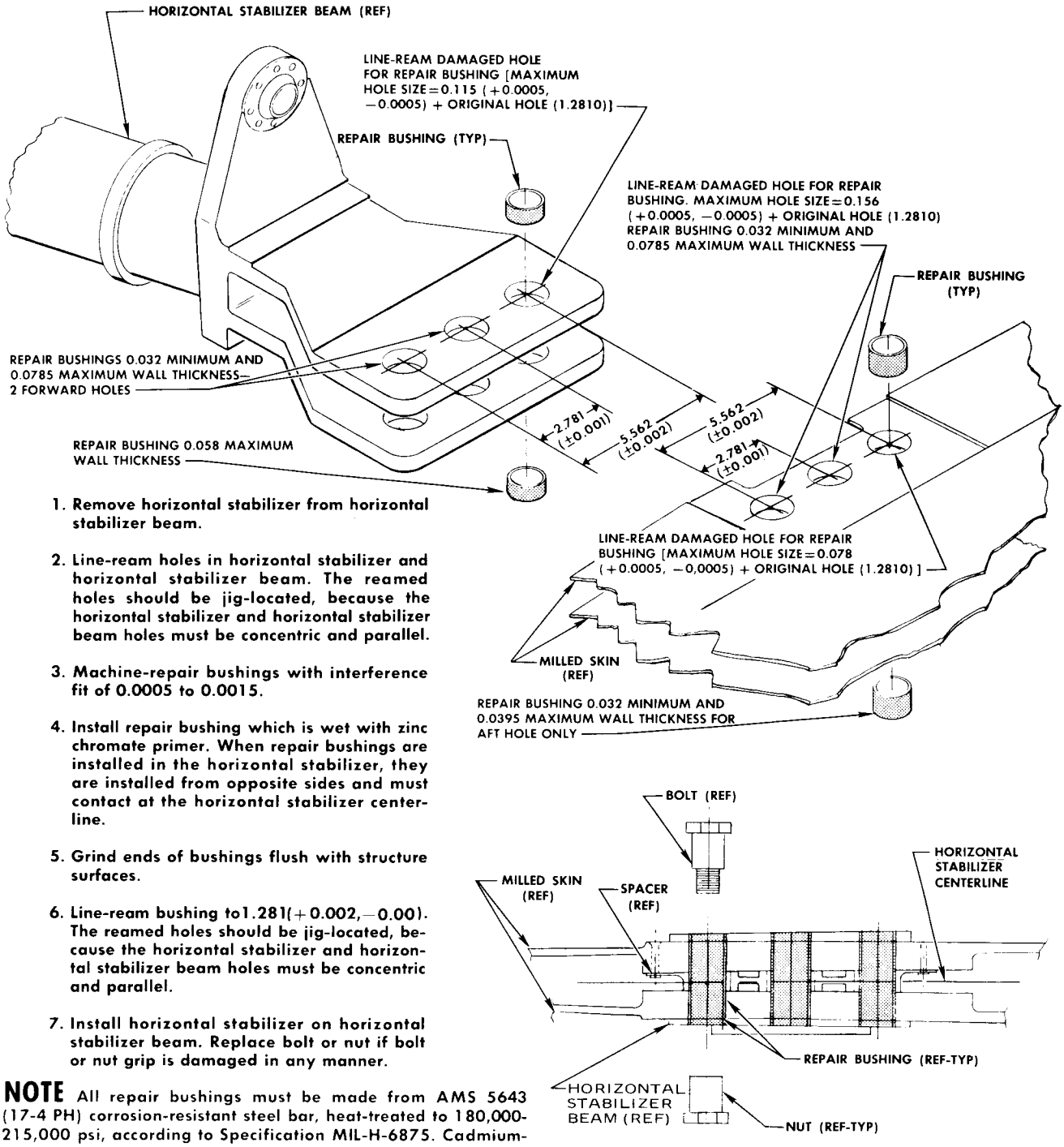
1. Cracks up to 1/2 inch long may be considered neglibible damage. No repair required. They will be stop-drilled if they are accessible.
2. Cracks up to 1-1/4 inches long are repairable. Replace rib if crack is over 1-1/4 inches long.
3. One flange repair is permissible on each end of rib. Replace the rib if both flanges are cracked on the same end of rib.
4. The general repair procedures in section X of this manual may be utilized.

NOTE: Any damage in the center section will require repair or rib replacement.

Figure 3-17. Horizontal Stabilizer Nose Rib (Sheet 2 of 2)

PERMANENT—CATEGORY A

EMPENNAGE



1. Remove horizontal stabilizer from horizontal stabilizer beam.
2. Line-ream holes in horizontal stabilizer and horizontal stabilizer beam. The reamed holes should be jig-located, because the horizontal stabilizer and horizontal stabilizer beam holes must be concentric and parallel.
3. Machine-repair bushings with interference fit of 0.0005 to 0.0015.
4. Install repair bushing which is wet with zinc chromate primer. When repair bushings are installed in the horizontal stabilizer, they are installed from opposite sides and must contact at the horizontal stabilizer centerline.
5. Grind ends of bushings flush with structure surfaces.
6. Line-ream bushing to1.281(+0.002,−0.00). The reamed holes should be jig-located, because the horizontal stabilizer and horizontal stabilizer beam holes must be concentric and parallel.
7. Install horizontal stabilizer on horizontal stabilizer beam. Replace bolt or nut if bolt or nut grip is damaged in any manner.

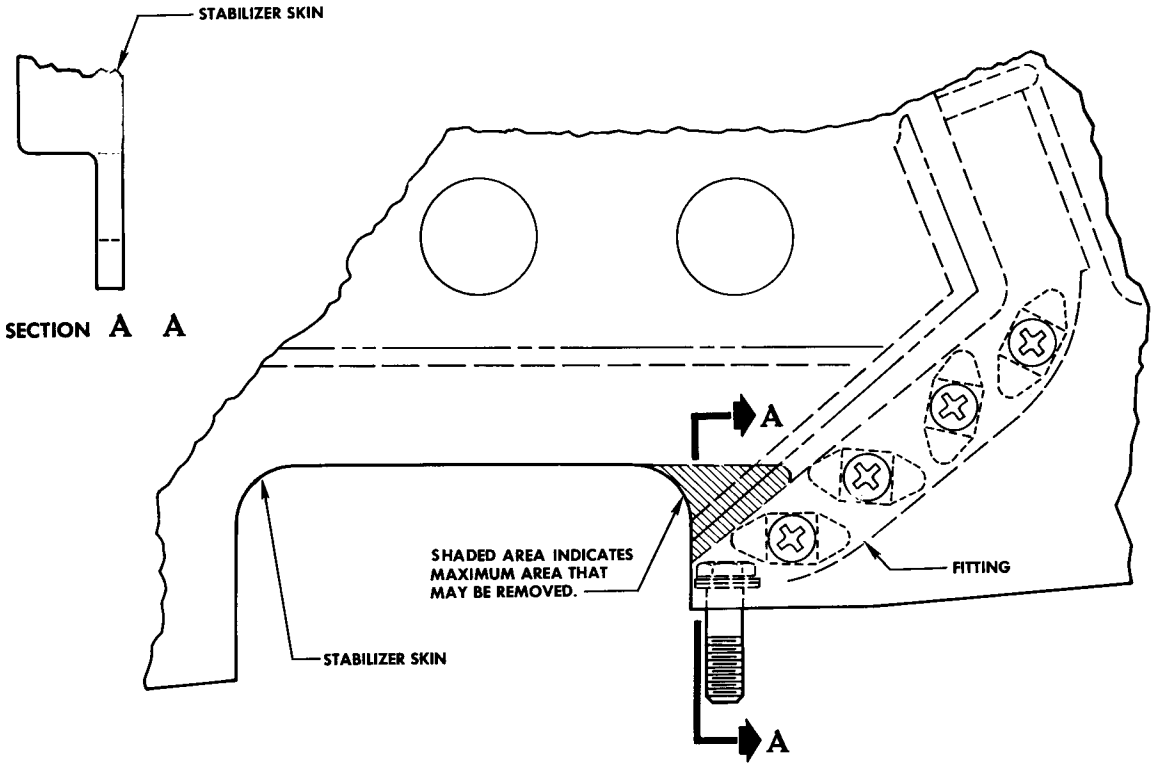
NOTE All repair bushings must be made from AMS 5643 (17-4 PH) corrosion-resistant steel bar, heat-treated to 180,000-215,000 psi, according to Specification MIL-H-6875. Cadmium-plate entire bushing.

F-100D-3-20-53B

Figure 3-18. Repair of Horizontal Stabilizer Attach-point Holes

PERMANENT CATEGORY A

EMPENNAGE



Cracks in the horizontal stabilizer box skins at the forward edge of the cutout for the center section beam assembly may be repaired by the following method:

1. Dye-penetrant check damaged area to determine the extent of crack.

2. Insert a piece of corrosion-resistant steel of about 0.032-inch gage between the front-beam-to-center section attach fitting and the stabilizer skin to prevent damage to the fitting during the grinding operation.

3. Grind out crack, using a rotary file or spiral reamer. Grinding must produce a cut at least 1/4 inch wide. All edges and corners must have a radius of at least 1/8 inch.

4. All areas that have had material removed by grinding must be polished to a 100-microinch finish by using emery cloth (No. 250 grit).

5. After grinding and polishing has been accomplished, remove the stainless steel protective sheet and clean area of all metal particles.

6. Dye-penetrant check to make sure that the crack has been completely removed.

▨ SHADED AREA INDICATES MAXIMUM ALLOWABLE MATERIAL THAT MAY BE REMOVED. IF DAMAGE EXTENDS BEYOND THESE LIMITS, STABILIZER SHOULD BE REPLACED.

F-100D-3-20-65

Figure 3-18A. Horizontal Stabilizer Box Skin Repair

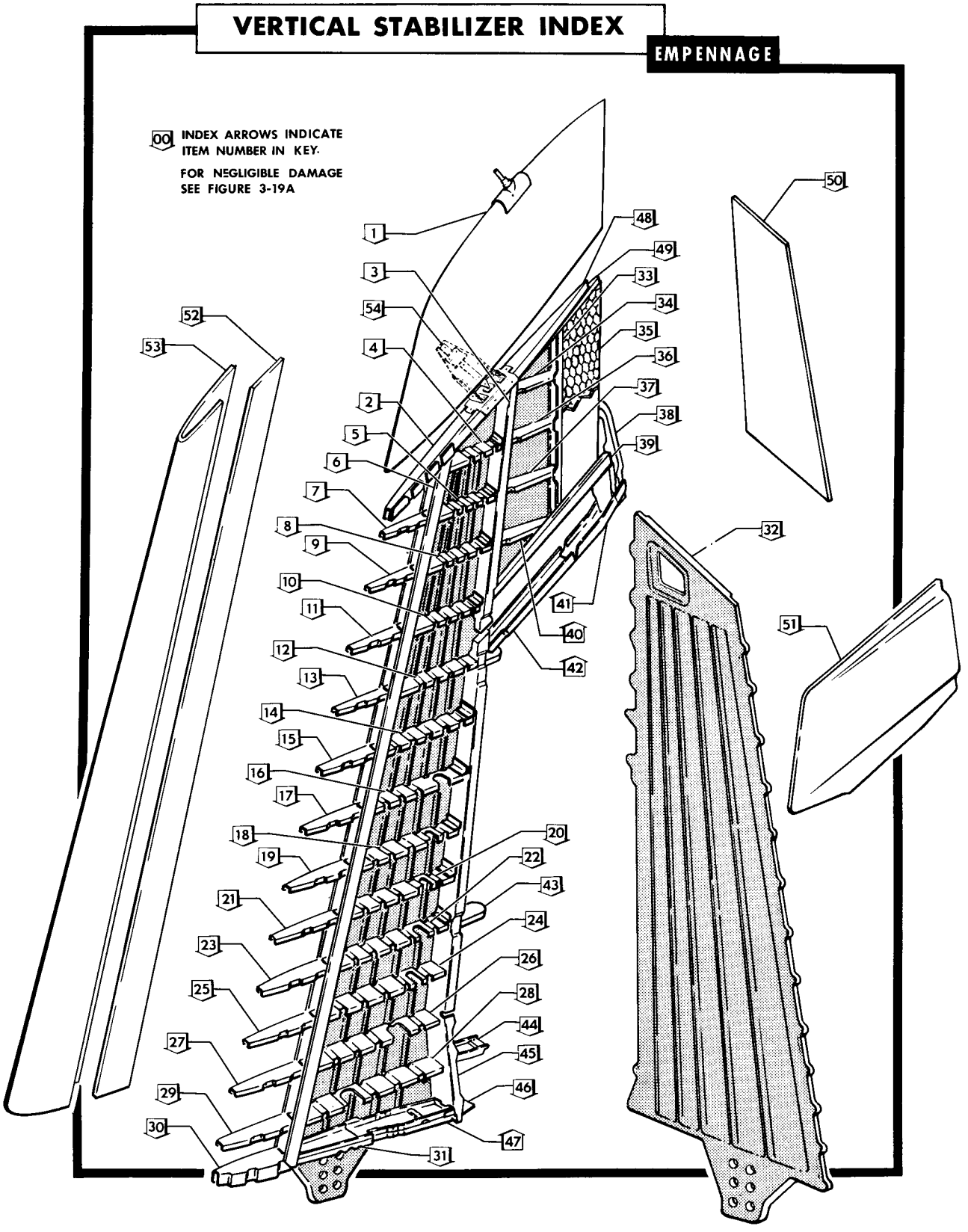


Figure 3-19. Vertical Stabilizer Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-23311	Tip Assy		Fiber Glass	
2	223-23106	Rib	0.050	2024-T4 Clad	
3	223-23303	Rear Beam—Upper		7075-T6 Plate	
4	223-23096	Rib	0.050	2024-T4 Clad	3-31
5	223-23090	Rib	0.050	2024-T4 Clad	3-31
6	223-23401	Front Beam		7075-T6 Plate	
7	223-23089	Nose Rib	0.040	2024-T4 Clad	
8	223-23083	Rib	0.050	2024-T4 Clad	3-31
9	223-23082	Nose Rib	0.040	2024-T4 Clad	
10	223-23075	Rib	0.063	2024-T4 Clad	3-31
11	223-23074	Nose Rib	0.040	2024-T4 Clad	3-30
12	223-23067	Rib	0.063	2024-T4 Clad	3-31
13	223-23066	Nose Rib	0.040	2024-T4 Clad	3-30
14	223-23059	Rib	0.063	2024-T4 Clad	3-31
15	223-23058	Nose Rib	0.040	2024-T4 Clad	3-30
16	223-23051	Rib	0.063	2024-T4 Clad	3-31
17	223-23050	Nose Rib	0.040	2024-T4 Clad	3-30
18	223-23043	Rib	0.063	2024-T4 Clad	3-31
19	223-23042	Nose Rib	0.040	2024-T4 Clad	3-30
20	223-23035	Rib	0.071	2024-T4 Clad	3-31
21	223-23034	Nose Rib	0.040	2024-T4 Clad	3-30
22	223-23028	Rib	0.080	2024-T4 Clad	3-31
23	223-23027	Nose Rib	0.050	2024-T4 Clad	3-30
24	223-23021	Rib	0.090	2024-T4 Clad	3-31
25	223-23020	Nose Rib	0.050	2024-T4 Clad	3-30
26	223-23014	Rib	0.090	2024-T4 Clad	3-31
27	223-23013	Nose Rib	0.050	2024-T4 Clad	3-30
28	223-23006	Rib	0.090	2024-T4 Clad	3-31
29	223-23005	Nose Rib	0.050	2024-T4 Clad	3-30
30	223-23004	Nose Rib	0.080	7075-T6 Clad	3-30
31	223-23409	Rib		7075-T6 Bar	
32	223-23421	Skin		7075-T6 Bare Plate	3-22, 3-23, 3-32, 3-38
33	223-23320	Channel	0.032	2024-T4 Clad	
34	223-23103	Rib	0.032	2024-T4 Clad	
35	223-23420	Te		Honeycomb—0.016 in. 6061-T6 Skin	3-27, 3-28, 3-29
36	223-23097	Rib	0.032	2024-T4 Clad	
37	223-23091	Rib	0.032	2024-T4 Clad	
38	223-23353	Intercostal	0.050	2024-T4 Clad	
39	223-23352	Zee	0.051	2024-T4 Clad	
40	223-23084	Rib	0.032	2024-T4 Clad	
41	223-23069	Rib	0.051	2024-T4 Clad	
42	223-24406	Hinge Link—Rudder			3-31A
43	223-24407	Hinge Link—Rudder			
44	223-25202	Rib	0.040	2024-T4 Clad	
45	223-23402	Rear-Lower Beam		7075-T6 Plate	3-38A
46	223-25220	Rib	0.050	2024-T4 Clad	
47	223-23022	Rib Assembly	0.080	7075-T6 Clad	
48	223-23109	Rib	0.050	2024-T4 Clad	
49	223-23110	Rib	0.050	2024-T4 Clad	
50	223-23001-5	Trailing Edge Skin	0.032	2024-T3 Clad	
51	223-23351	Fairing	0.051	2024-T4 Clad	
52	223-23001-3	Leading Edge Close-out Skin	0.040	2024-T3 Clad	3-26
53	223-23003	Leading Edge Skin	0.040	2024-T3 Clad	3-20, 3-21, 3-24, 3-25, 3-26
54	223-71039	Antenna Mount			3-32C

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 3-19

VERTICAL STABILIZER NEGLIGIBLE DAMAGE

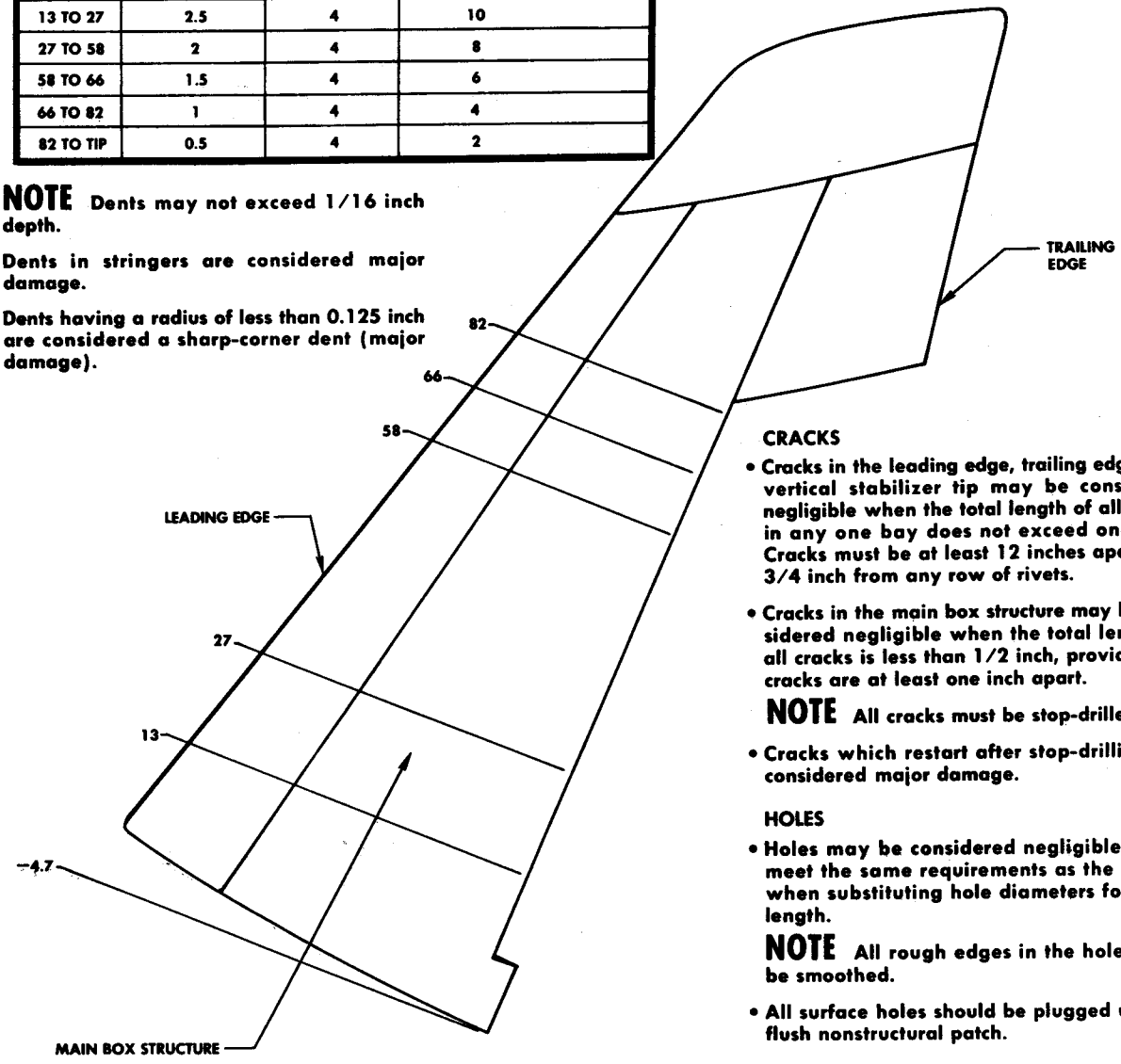
EMPENNAGE

DENTS

STATION	MAXIMUM LENGTH CHORDWISE	SPANWISE	TOTAL AREA—ALL DENTS (SQUARE INCHES)
—4.7 to 13	3.5	8	28
13 TO 27	2.5	4	10
27 TO 58	2	4	8
58 TO 66	1.5	4	6
66 TO 82	1	4	4
82 TO TIP	0.5	4	2

NOTE Dents may not exceed 1/16 inch depth.

- Dents in stringers are considered major damage.
- Dents having a radius of less than 0.125 inch are considered a sharp-corner dent (major damage).



CRACKS

- Cracks in the leading edge, trailing edge, and vertical stabilizer tip may be considered negligible when the total length of all cracks in any one bay does not exceed one inch. Cracks must be at least 12 inches apart and 3/4 inch from any row of rivets.
- Cracks in the main box structure may be considered negligible when the total length of all cracks is less than 1/2 inch, provided the cracks are at least one inch apart.

NOTE All cracks must be stop-drilled.

- Cracks which restart after stop-drilling are considered major damage.

HOLES

- Holes may be considered negligible which meet the same requirements as the cracks, when substituting hole diameters for crack length.

NOTE All rough edges in the holes must be smoothed.

- All surface holes should be plugged using a flush nonstructural patch.

SCRATCHES

- Scratches less than 4-1/2 inches in length which do not exceed 15 percent of the skin thickness may be considered negligible damage.

NOTE All scratches should be polished smooth and inspected for cracks.

CORROSION

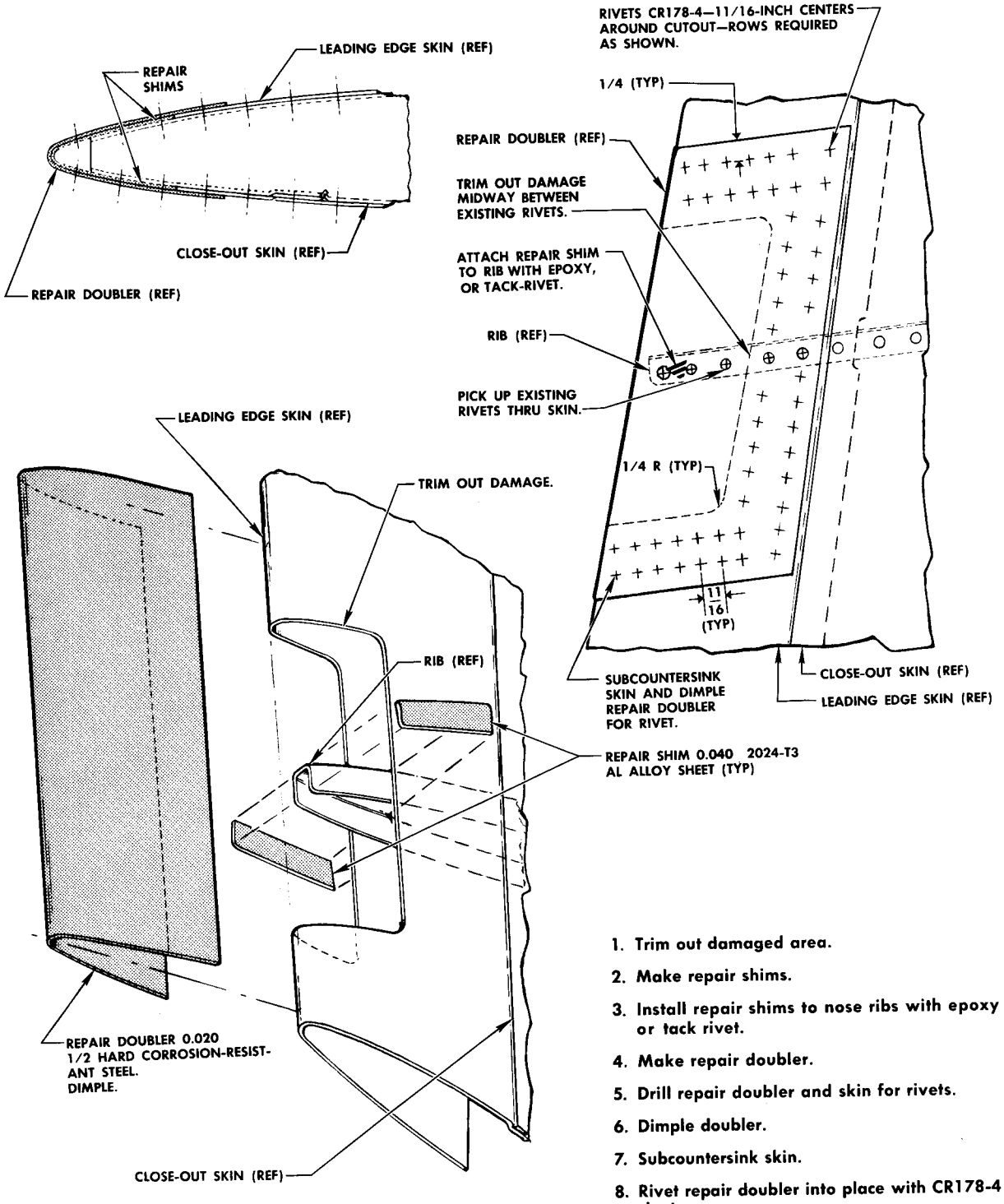
- Corrosion damage not exceeding limits for scratches or cracks may be considered negligible. Corroded area must be cleaned and chemical film treated. (Refer to index for "Corrosion Rework by Chemical Film Treatments.")

F-100D-3-20-58A

Figure 3-19A. Vertical Stabilizer Negligible Damage

TEMPORARY—CATEGORY A

EMPENNAGE



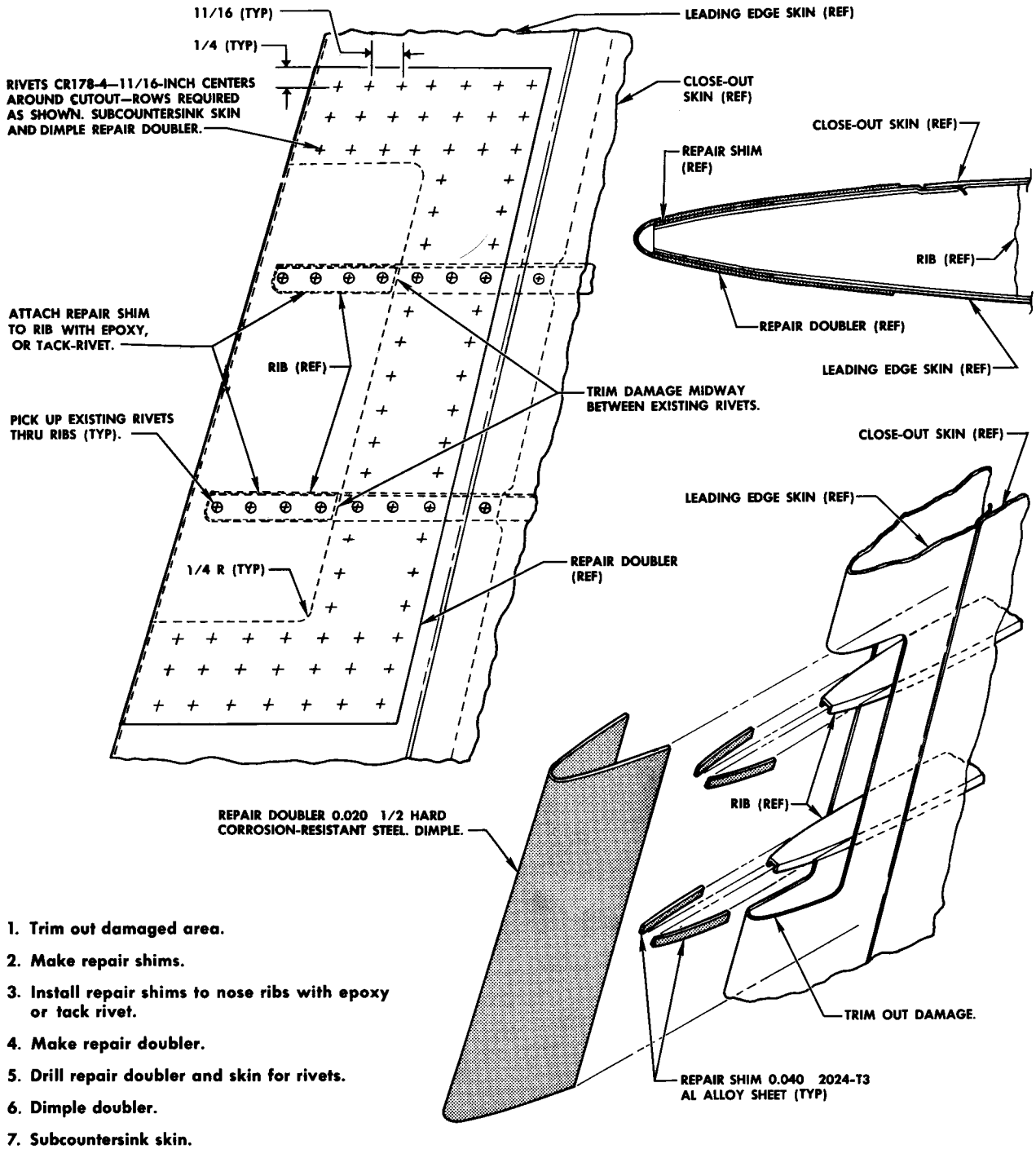
1. Trim out damaged area.
2. Make repair shims.
3. Install repair shims to nose ribs with epoxy or tack rivet.
4. Make repair doubler.
5. Drill repair doubler and skin for rivets.
6. Dimple doubler.
7. Subcountersink skin.
8. Rivet repair doubler into place with CR178-4 rivet.

F-100D-3-20-49

Figure 3-20. Temporary Repair of Vertical Stabilizer Leading Edge Damaged Across One Rib

TEMPORARY—CATEGORY A

EMPENNAGE



1. Trim out damaged area.
2. Make repair shims.
3. Install repair shims to nose ribs with epoxy or tack rivet.
4. Make repair doubler.
5. Drill repair doubler and skin for rivets.
6. Dimple doubler.
7. Subcountersink skin.
8. Rivet repair doubler into place with CR178-4 rivets.

F-100D-3-20-52

Figure 3-21. Temporary Repair of Vertical Stabilizer Leading Edge Damaged Across Two Ribs

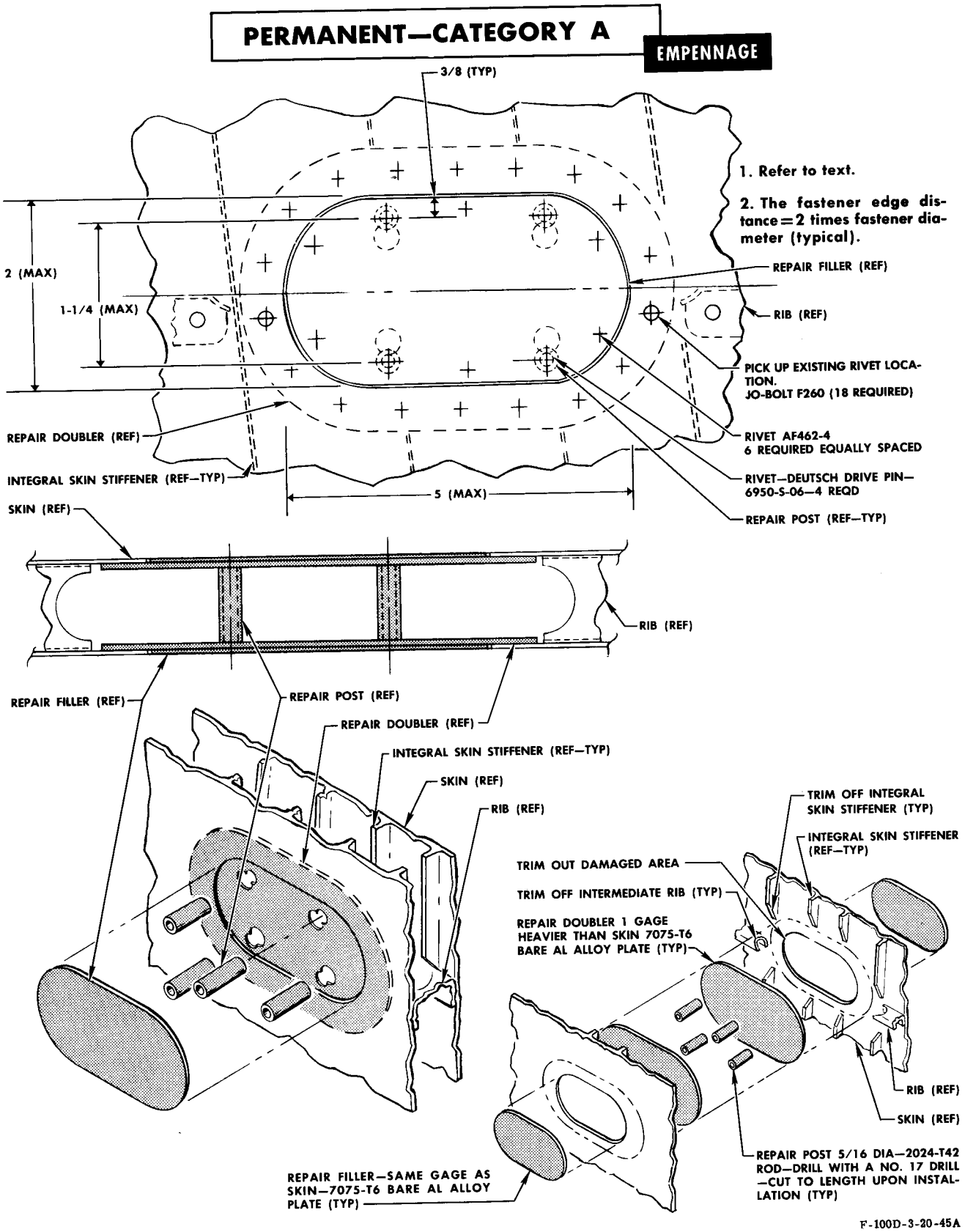


Figure 3-22. Chordwise Repair of Vertical Stabilizer Milled Skin at Station 89

PERMANENT—CATEGORY A

EMPENNAGE

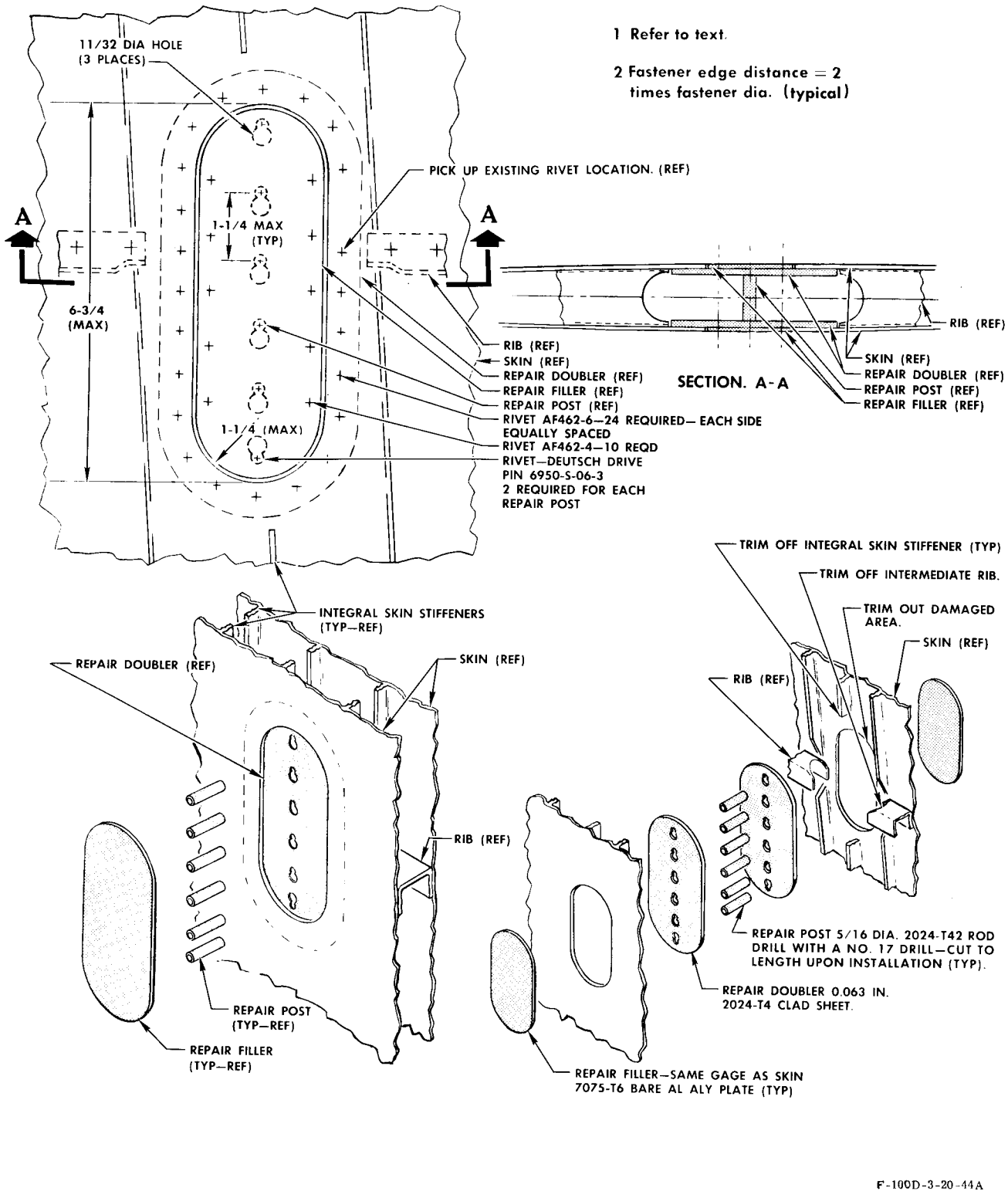
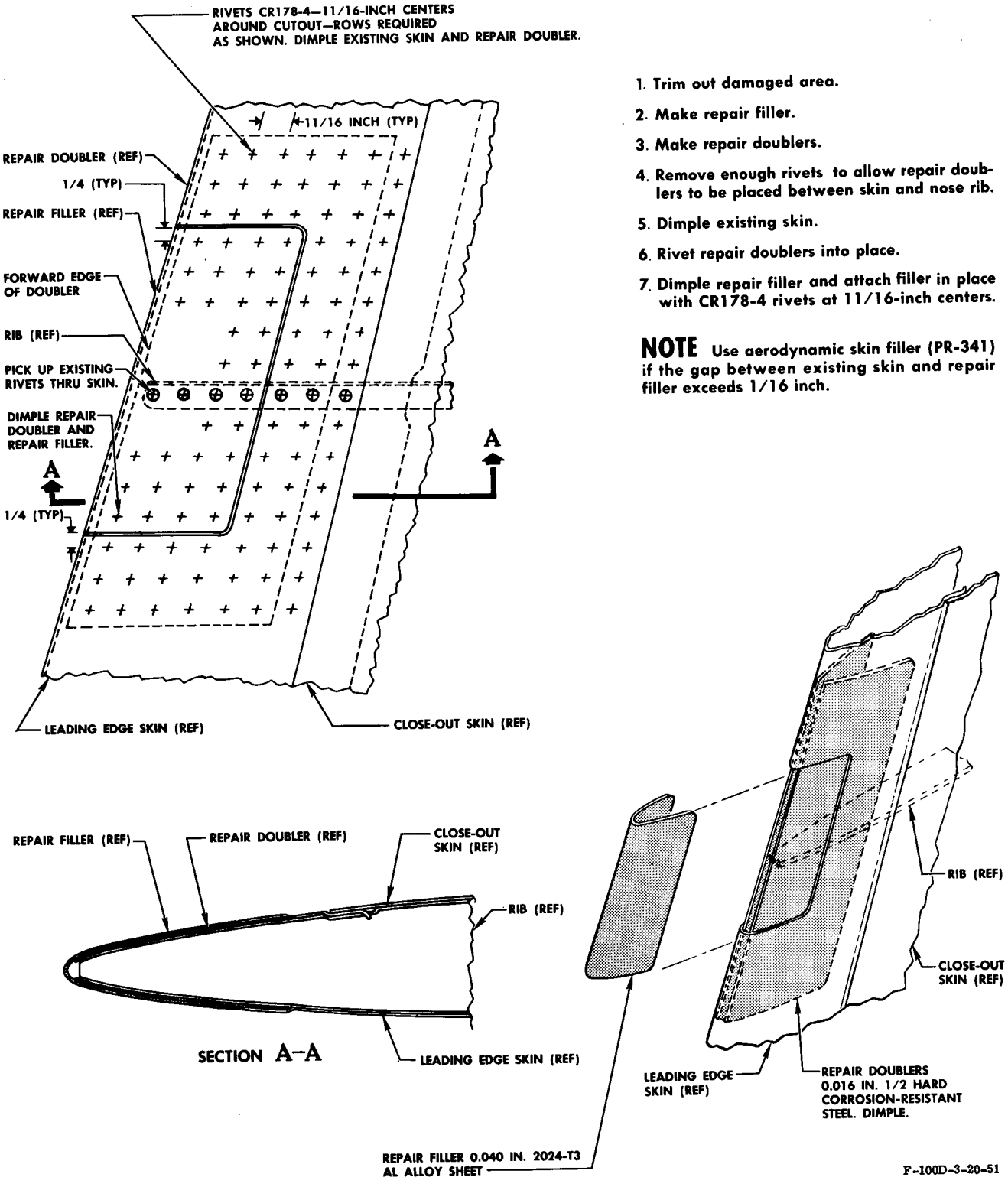


Figure 3-23. Spanwise Repair of Vertical Stabilizer Milled Skin at Station 89

PERMANENT—CATEGORY A

EMPENNAGE



1. Trim out damaged area.
2. Make repair filler.
3. Make repair doublers.
4. Remove enough rivets to allow repair doublers to be placed between skin and nose rib.
5. Dimple existing skin.
6. Rivet repair doublers into place.
7. Dimple repair filler and attach filler in place with CR178-4 rivets at 11/16-inch centers.

NOTE Use aerodynamic skin filler (PR-341) if the gap between existing skin and repair filler exceeds 1/16 inch.

Figure 3-24. Permanent Repair of Vertical Stabilizer Leading Edge Damaged Across One Rib

PERMANENT—CATEGORY A

EMPENNAGE

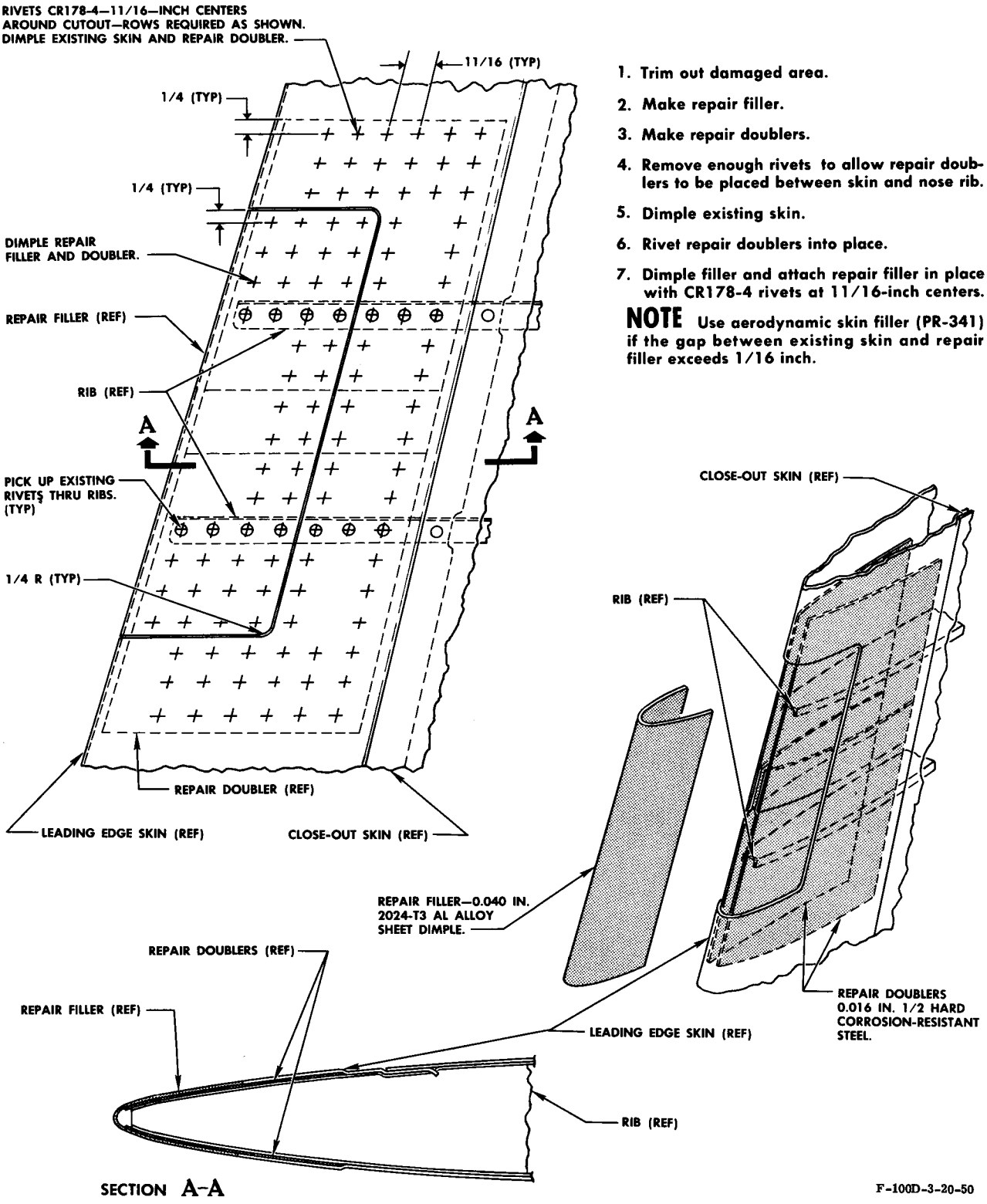
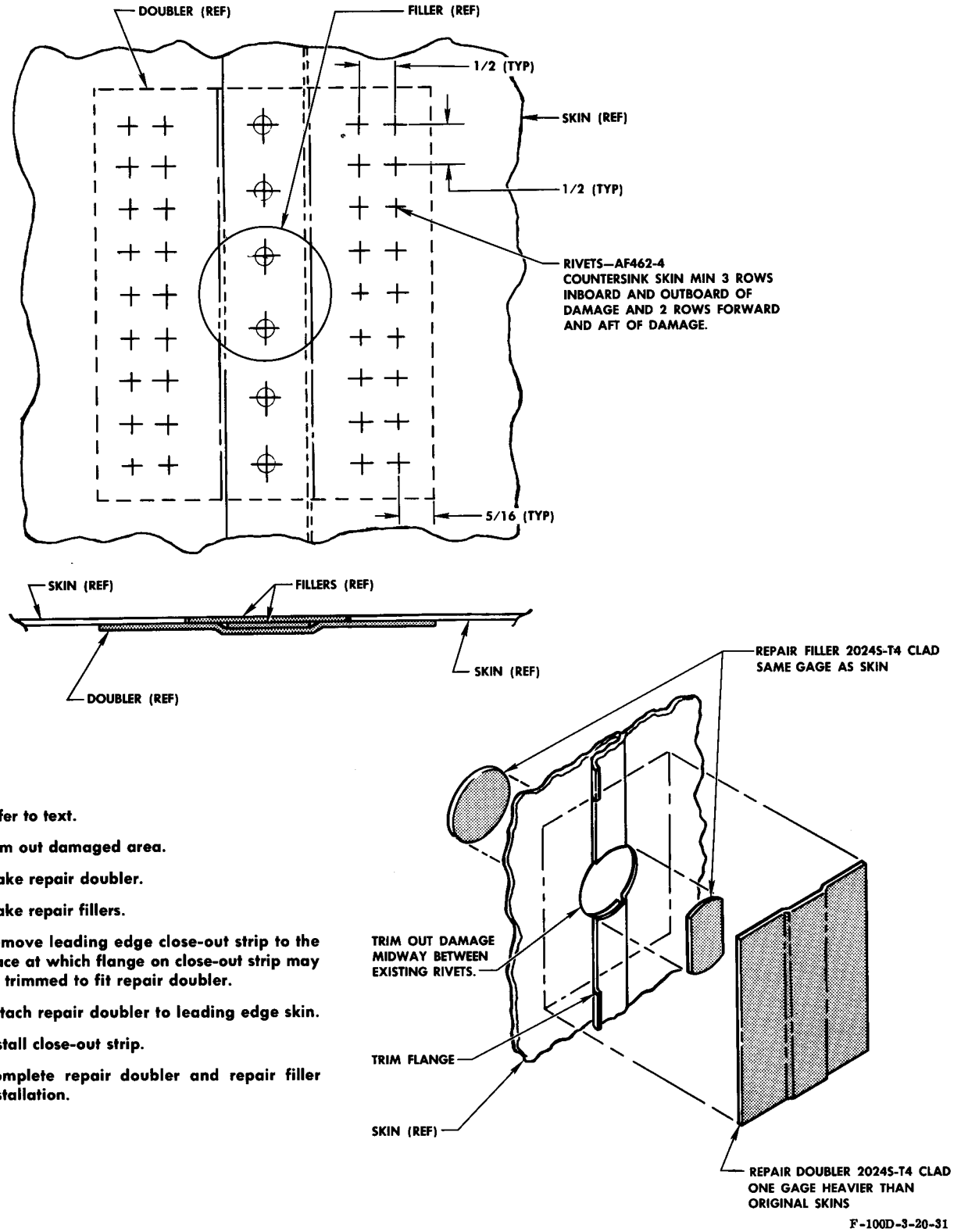


Figure 3-25. Permanent Repair of Vertical Stabilizer Leading Edge Damaged Across Two Ribs

PERMANENT CATEGORY A

EMPENNAGE

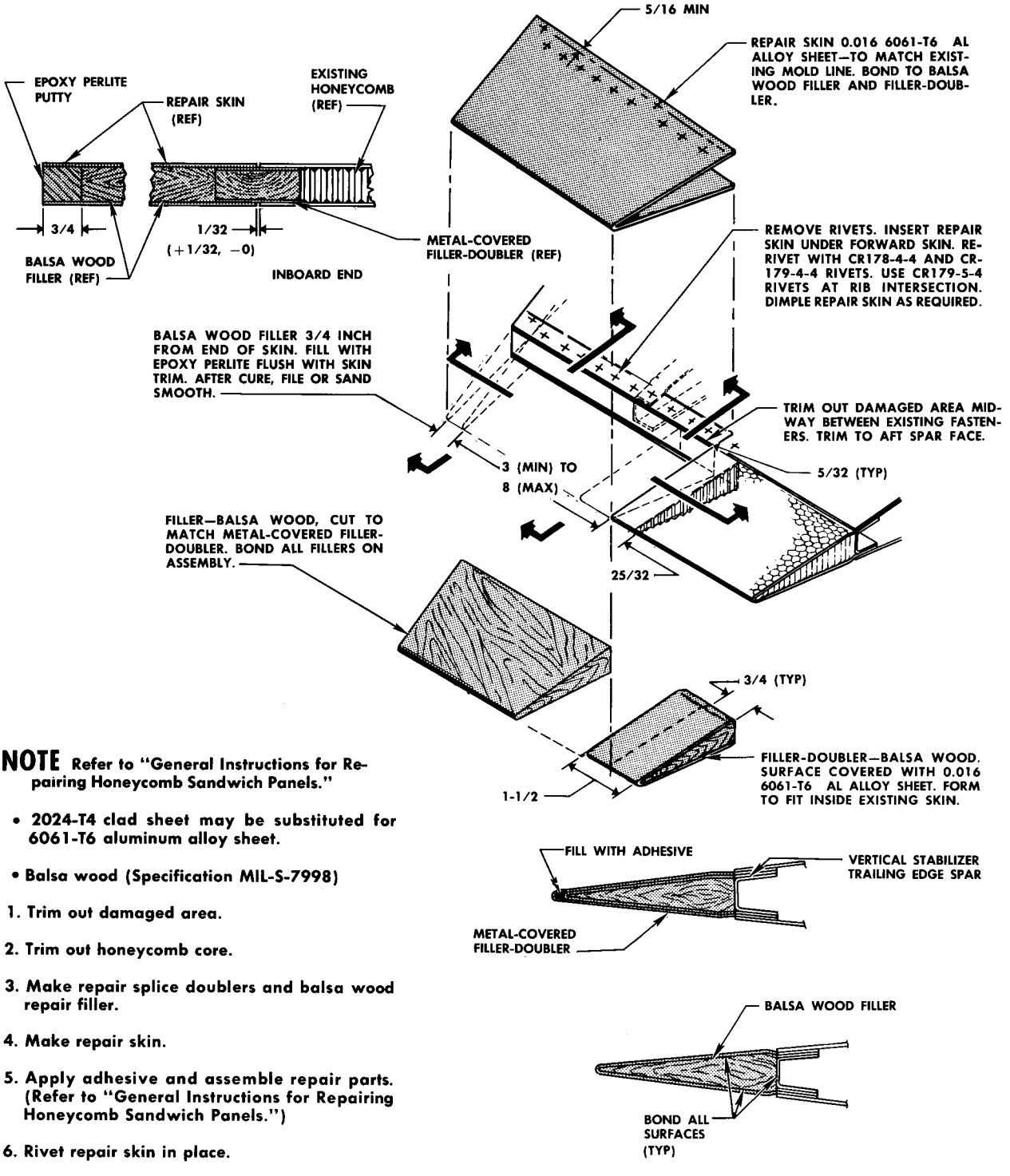


1. Refer to text.
2. Trim out damaged area.
3. Make repair doubler.
4. Make repair fillers.
5. Remove leading edge close-out strip to the place at which flange on close-out strip may be trimmed to fit repair doubler.
6. Attach repair doubler to leading edge skin.
7. Install close-out strip.
8. Complete repair doubler and repair filler installation.

Figure 3-26. Vertical Stabilizer Nose Skin at Close-out Splice Left Side

PERMANENT CATEGORY A

EMPENNAGE



NOTE Refer to "General Instructions for Repairing Honeycomb Sandwich Panels."

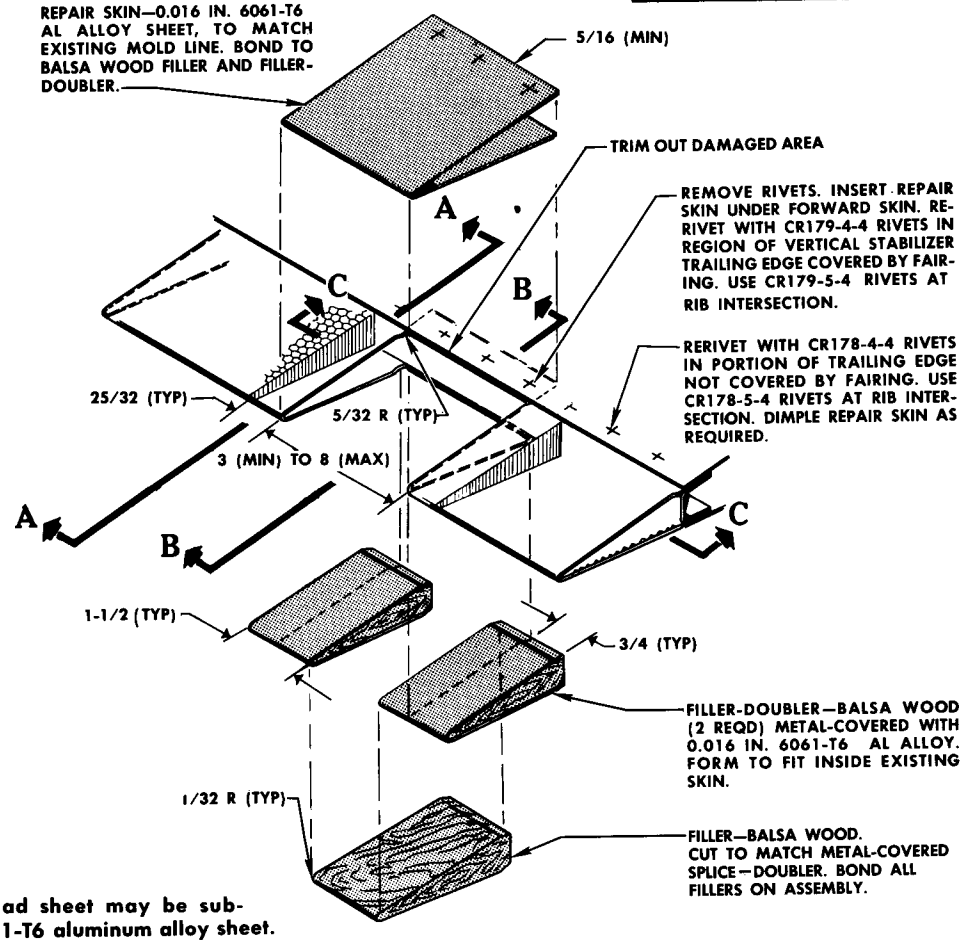
- 2024-T4 clad sheet may be substituted for 6061-T6 aluminum alloy sheet.
- Balsa wood (Specification MIL-S-7998)

1. Trim out damaged area.
2. Trim out honeycomb core.
3. Make repair splice doublers and balsa wood repair filler.
4. Make repair skin.
5. Apply adhesive and assemble repair parts. (Refer to "General Instructions for Repairing Honeycomb Sandwich Panels.")
6. Rivet repair skin in place.
7. Clean off excess adhesive.

Figure 3-27. Inboard End of Vertical Stabilizer Trailing Edge Honeycomb

PERMANENT—CATEGORY A

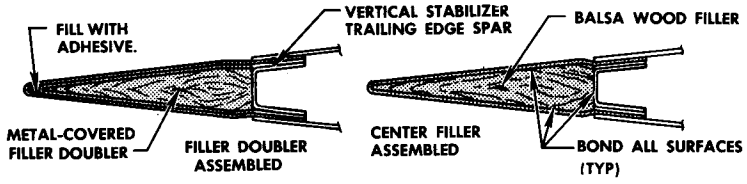
EMPENNAGE



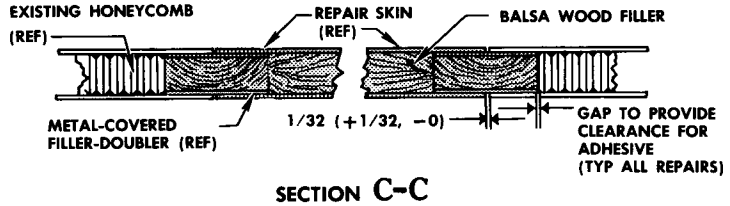
NOTE 2024-T4 clad sheet may be substituted for 6061-T6 aluminum alloy sheet.

- If damage occurs in area where fairing attaches to trailing edge, bond in 2024-T4 filler blocks one inch in diameter.
- Where rivets pass through trailing edge, use a hot curing adhesive.
- Balsa wood (Specification MIL-S-7998)

1. Trim out damaged area midway between existing fasteners to aft spar face.
2. Trim out honeycomb core.
3. Make repair splice doublers and balsa wood repair filler.
4. Make repair skin.
5. Apply adhesive and assemble repair parts. (Refer to "General Instructions for Repairing Honeycomb Sandwich Panels.")
6. Rivet repair skin in place.
7. Clean off excess adhesive.



SECTION A-A SECTION B-B



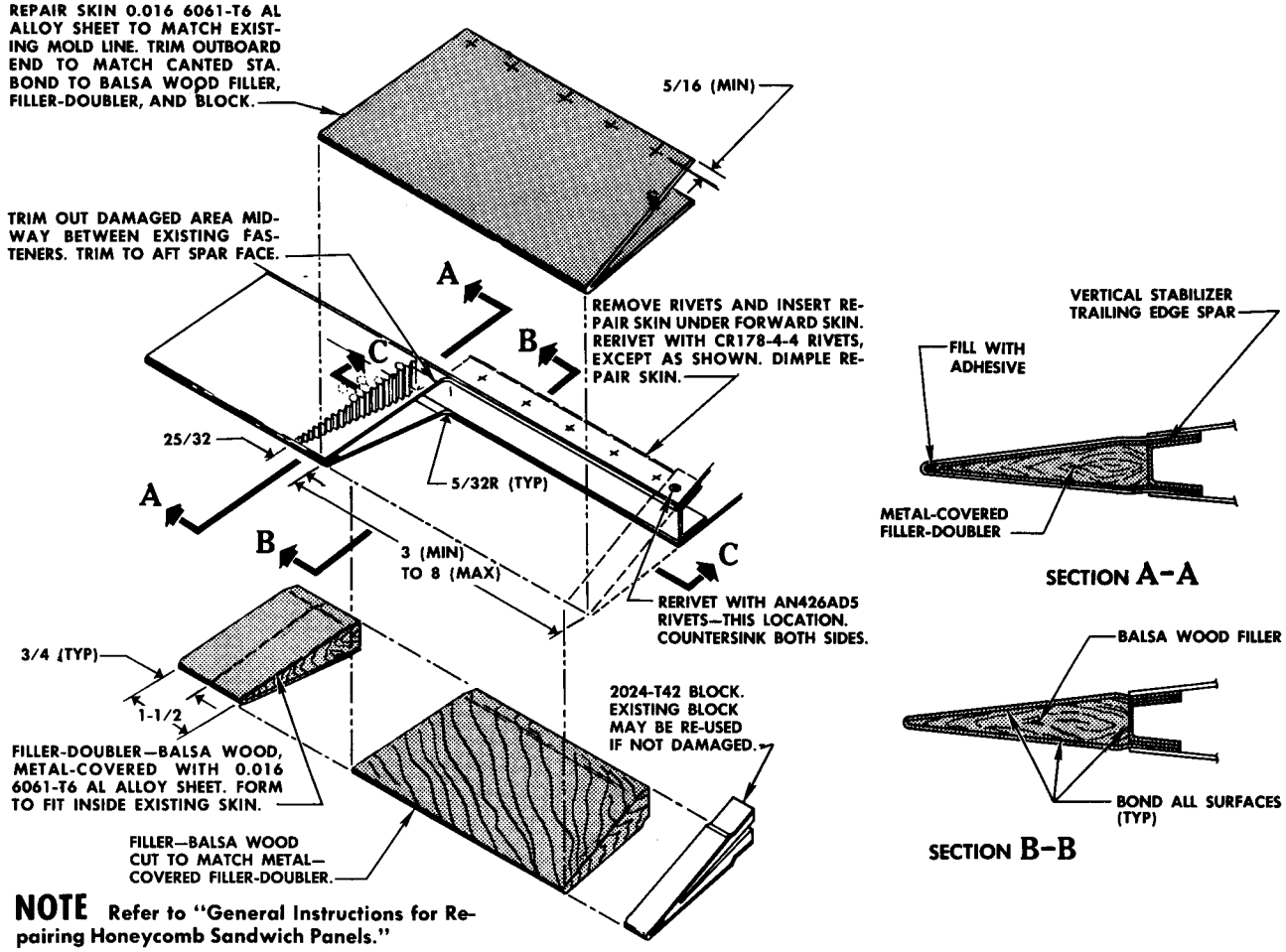
SECTION C-C

F-100D-3-20-33

Figure 3-28. Center of Vertical Stabilizer Trailing Edge Honeycomb

PERMANENT—CATEGORY A

EMPENNAGE



NOTE Refer to "General Instructions for Repairing Honeycomb Sandwich Panels."

- 2024-T4 clad sheet may be substituted for 6061-T6 aluminum alloy sheet.
- This repair may be cold-cured. However, when the 2024-T42 block is being replaced or rebonded, it requires the use of an elevated-temperature cure.
- Balsa wood (Specification MIL-S-7998)

1. Trim out damaged area.
2. Trim out honeycomb core.
3. Make repair splice doublers and balsa wood repair filler.
4. Make repair skin.
5. Apply adhesive and assemble repair parts. (Refer to "General Instructions for Repairing Honeycomb Sandwich Panels.")
6. Rivet repair skin in place.
7. Clean off excessive adhesive.

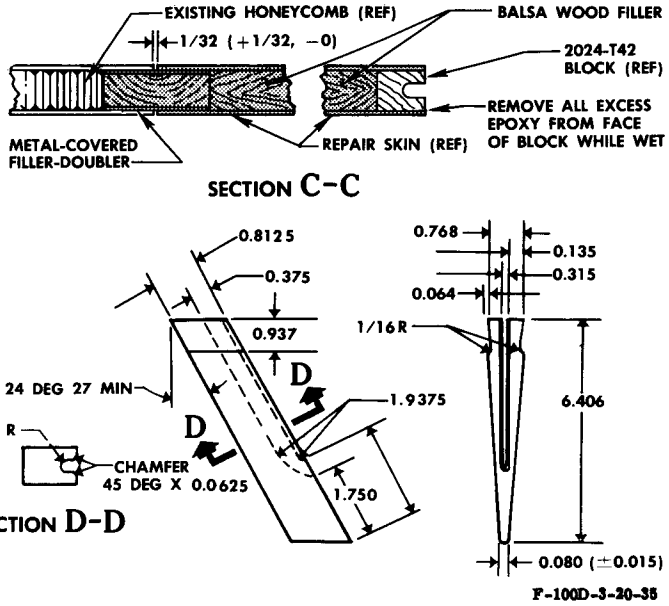


Figure 3-29. Outboard Edge of Vertical Stabilizer Trailing Edge Honeycomb

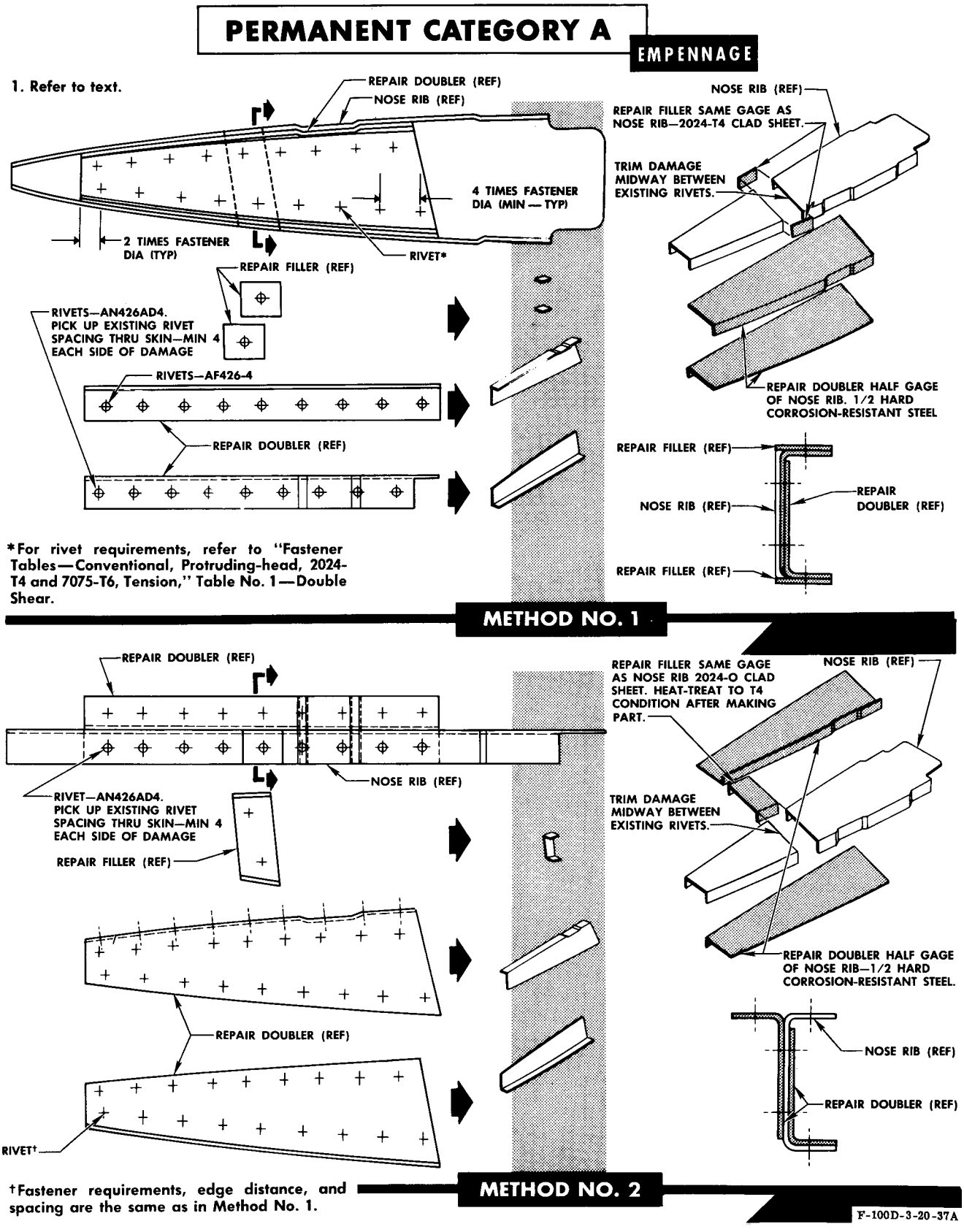


Figure 3-30. Vertical Stabilizer Nose Rib

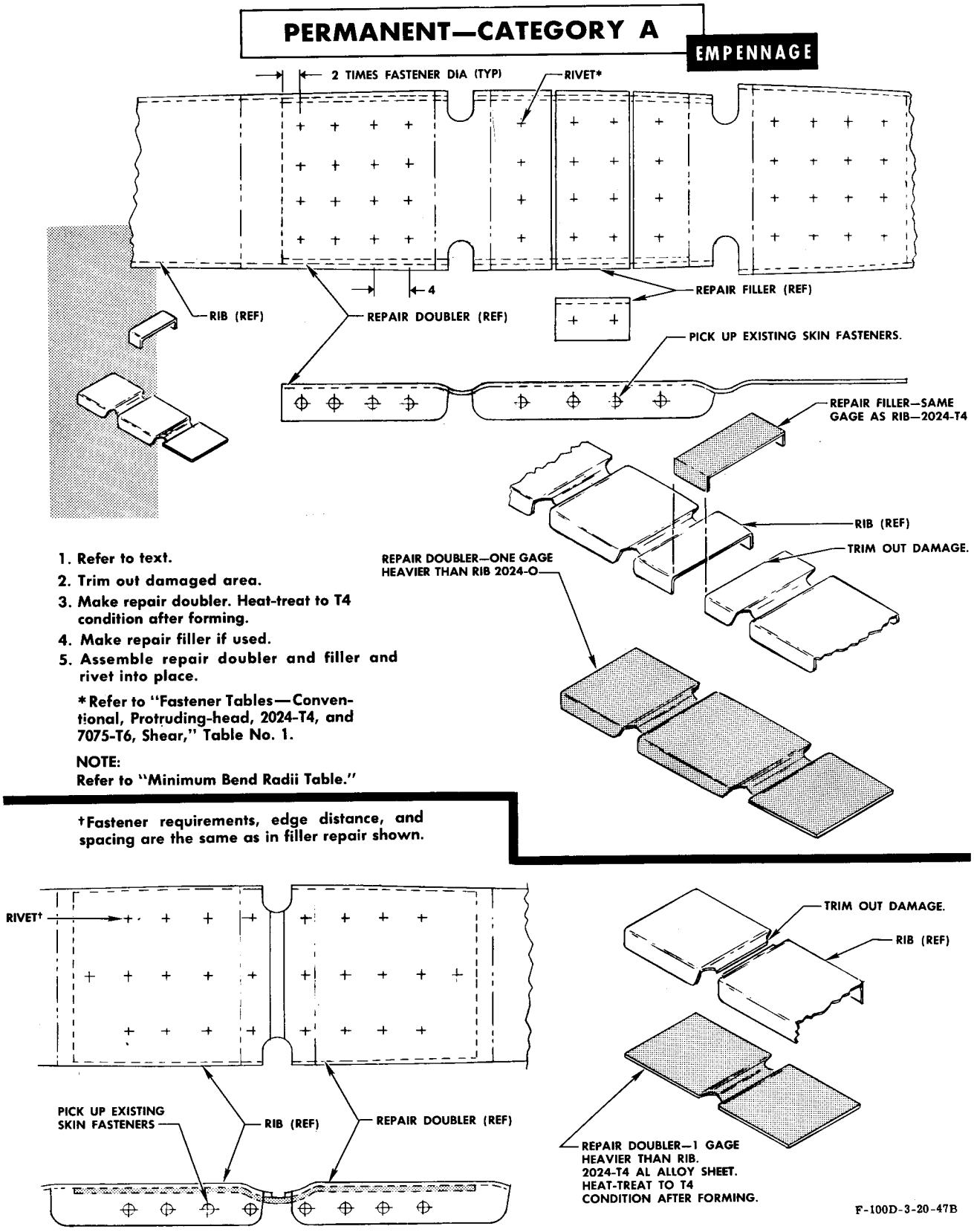


Figure 3-31. Vertical Stabilizer Intermediate Rib

RUDDER LINK AND HORN ATTACHMENTS

EMPENNAGE

Refer to "Rework and Adjustment of Rudder Link and Rudder Horn Attachments" in text for misalignment check procedure.

Correct the misalignment of the lower rudder link attaching holes in the vertical stabilizer by boring and countersinking the holes oversize and in line with the relocated rudder link holes.

1 Line-ream the holes from each side to 0.473 (+0.0000, −0.0005) inch diameter.

2 Enlarge the countersinks to 0.930-0.940 inch diameter.

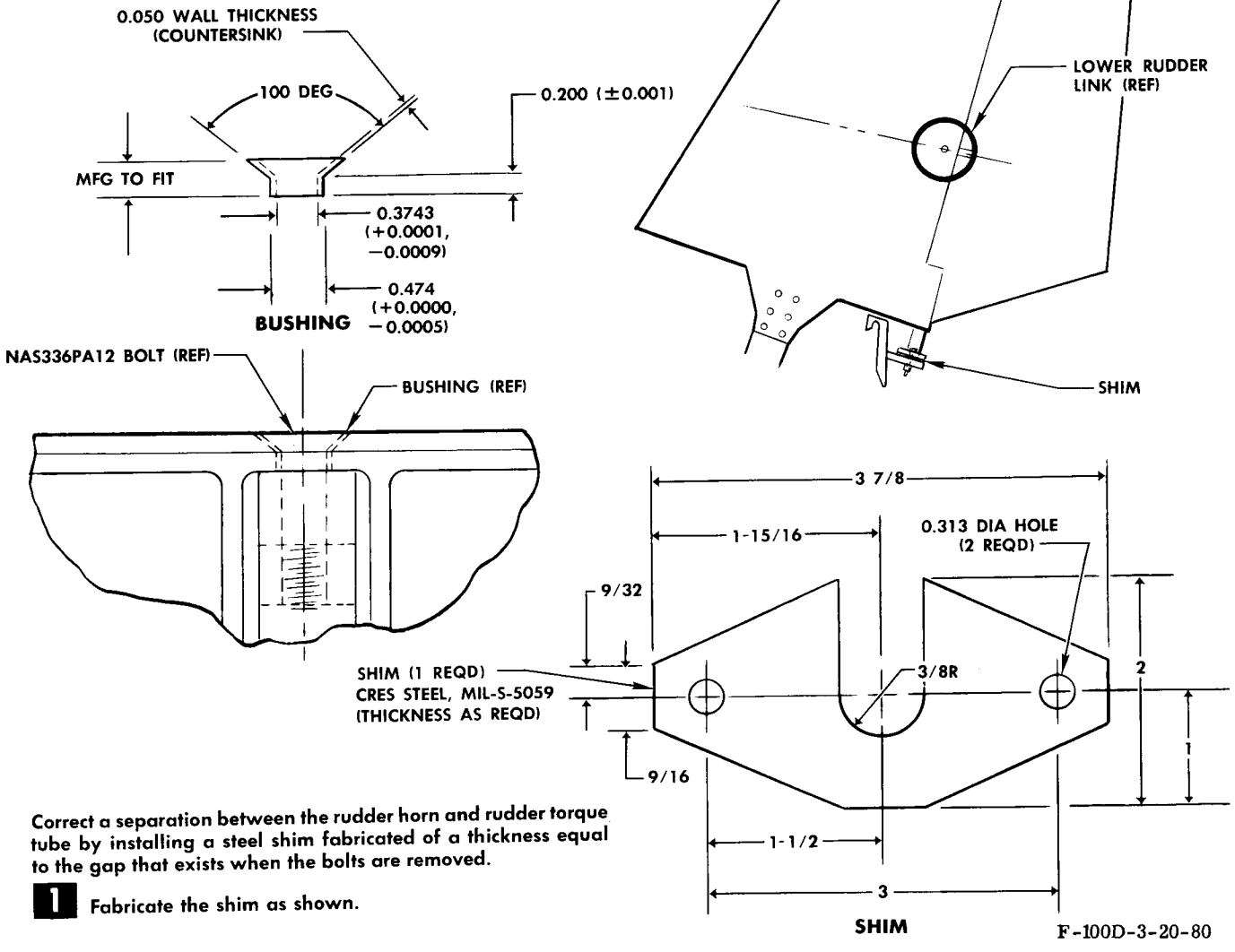
3 Fabricate two bushings as shown.

4 Treat the exposed surfaces with Alodine or Iridite, MIL-D-5541. (Refer to section I.)

5 Prime hole and countersink inside diameter with zinc-chromate primer, MIL-P-8585. Install bushings in the reworked holes while the primer is still wet.

NOTE Make bushing from 4130 steel bar; heat-treat to 125,000-140,000 psi.

- Break sharp edges to 0.015.
- Cad plate.



Correct a separation between the rudder horn and rudder torque tube by installing a steel shim fabricated of a thickness equal to the gap that exists when the bolts are removed.

1 Fabricate the shim as shown.

Figure 3-31A. Rework and Adjustment of Rudder Link and Rudder Horn Attachments

PERMANENT—CATEGORY A

EMPENNAGE

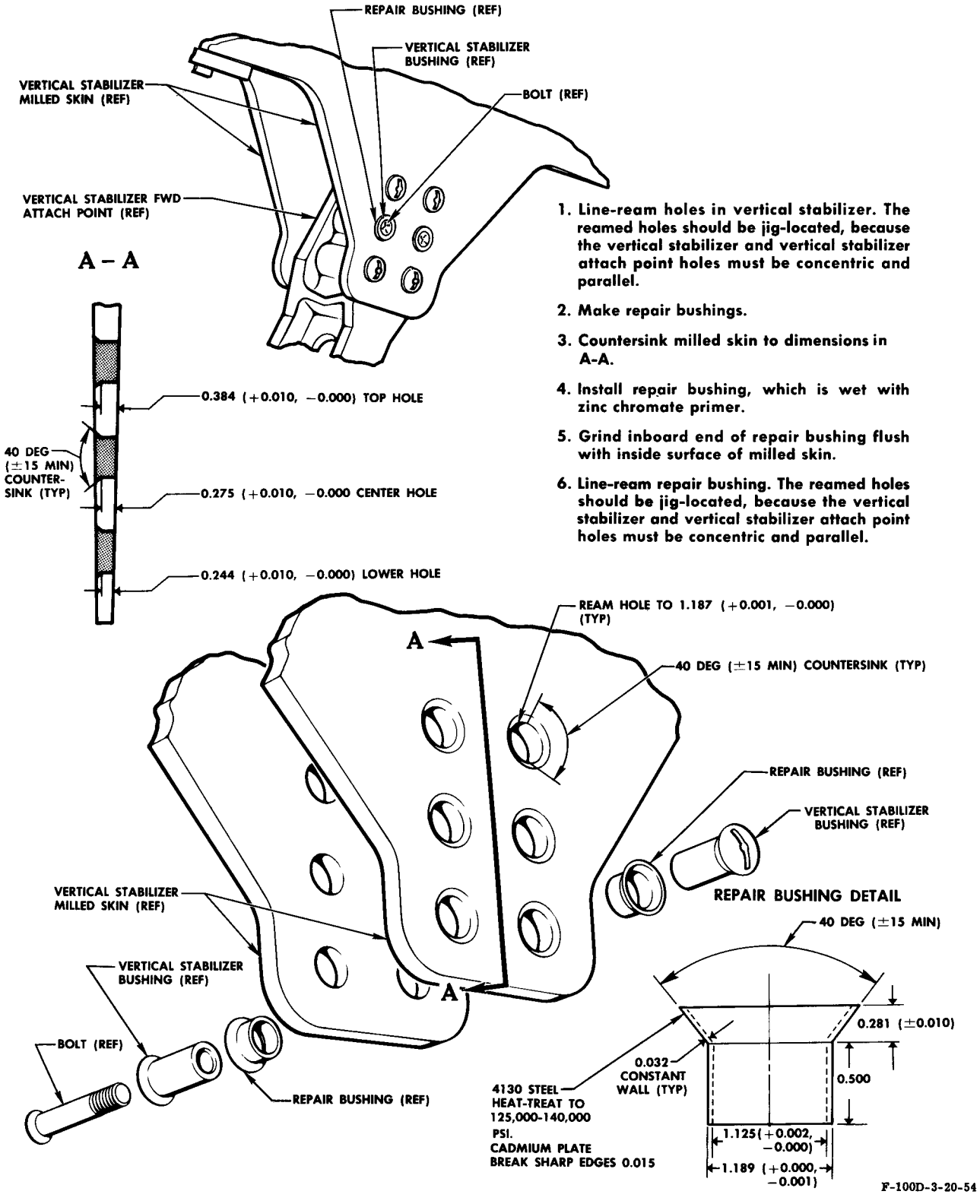
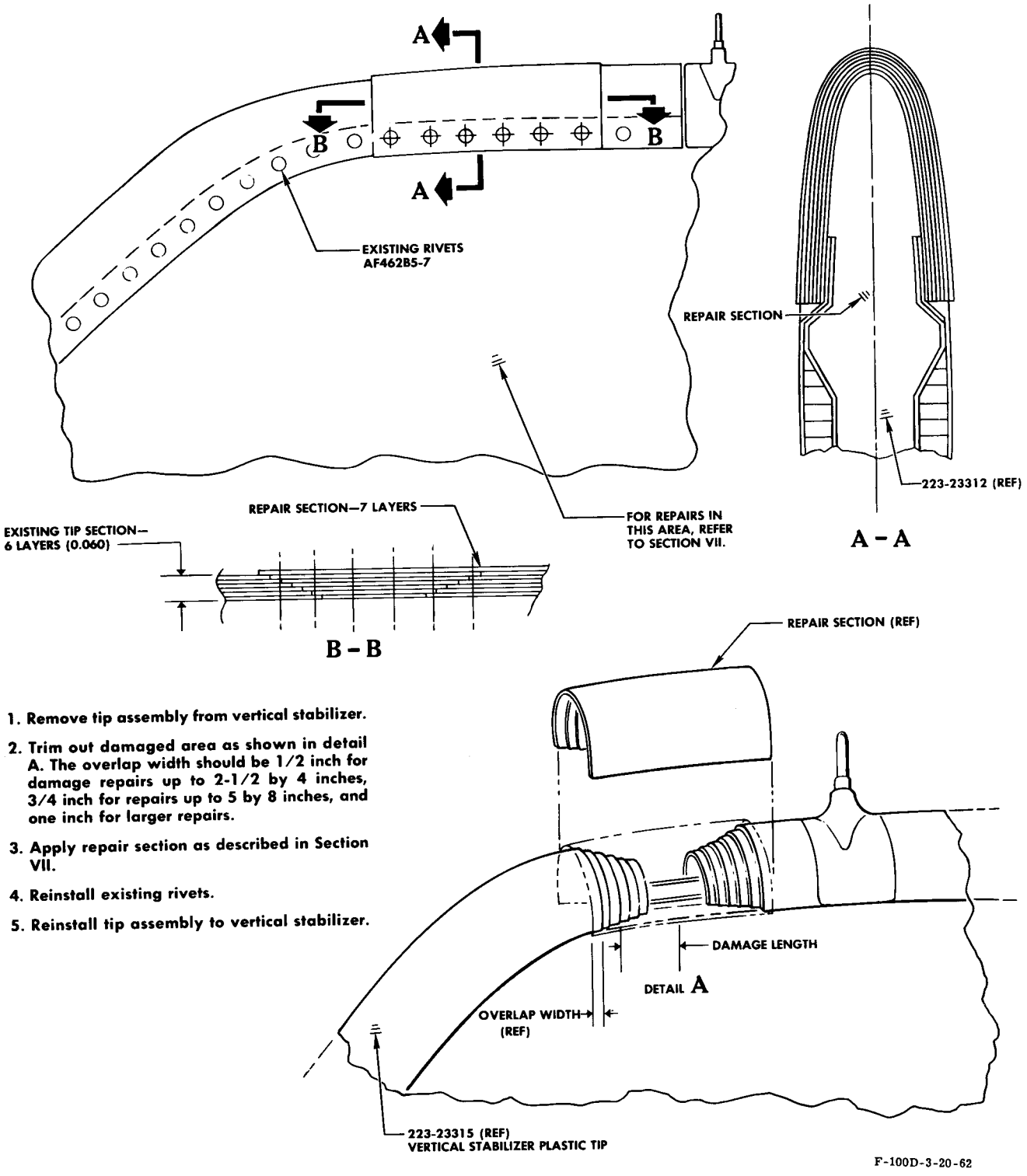


Figure 3-32. Repair of Vertical Stabilizer Attach-point Holes

PERMANENT CATEGORY A — EMPENNAGE

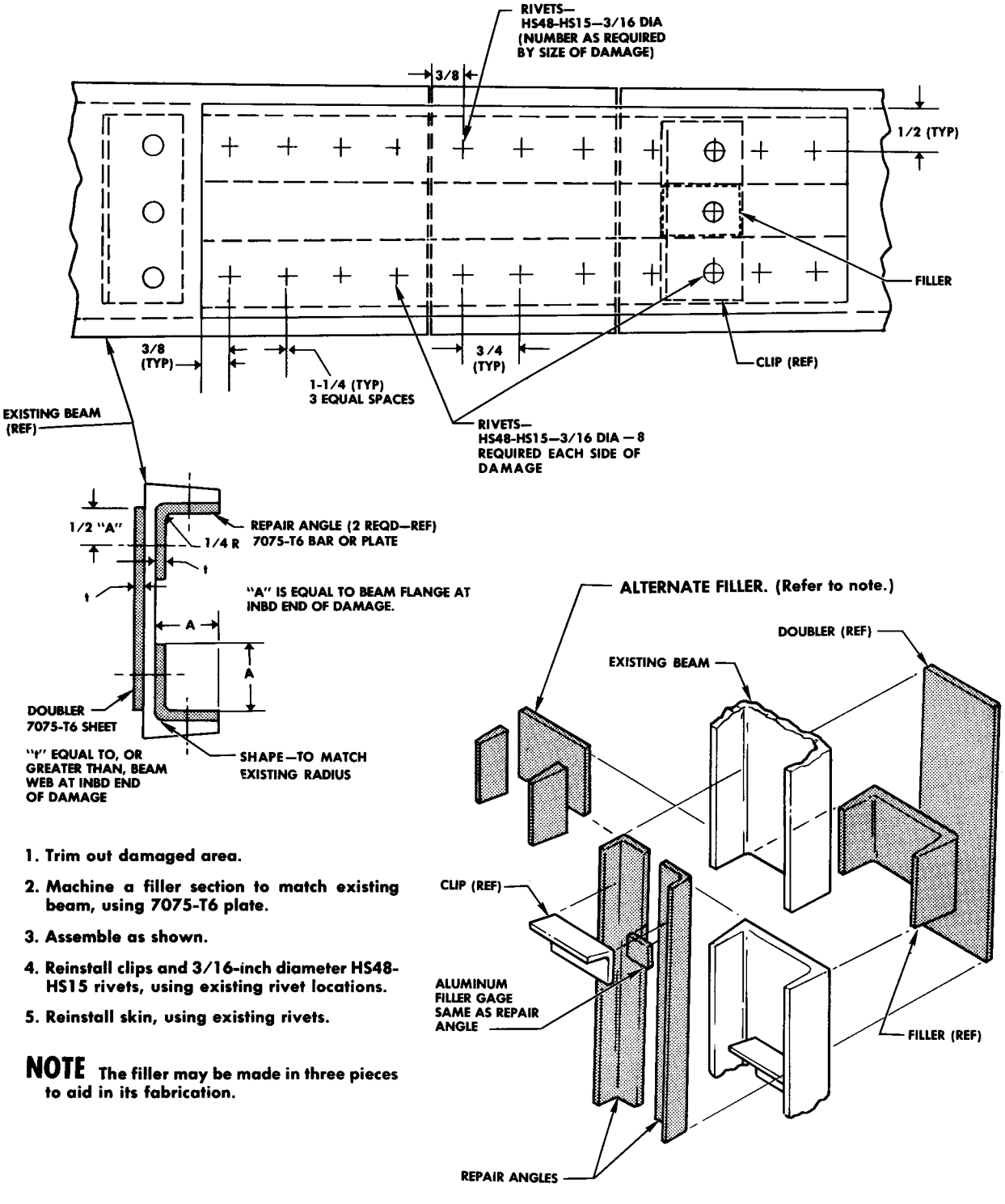


1. Remove tip assembly from vertical stabilizer.
2. Trim out damaged area as shown in detail A. The overlap width should be 1/2 inch for damage repairs up to 2-1/2 by 4 inches, 3/4 inch for repairs up to 5 by 8 inches, and one inch for larger repairs.
3. Apply repair section as described in Section VII.
4. Reinstall existing rivets.
5. Reinstall tip assembly to vertical stabilizer.

Figure 3-32A. Vertical Stabilizer Tip Repair

PERMANENT—CATEGORY B

EMPENNAGE



1. Trim out damaged area.
2. Machine a filler section to match existing beam, using 7075-T6 plate.
3. Assemble as shown.
4. Reinstall clips and 3/16-inch diameter HS48-HS15 rivets, using existing rivet locations.
5. Reinstall skin, using existing rivets.

NOTE The filler may be made in three pieces to aid in its fabrication.

F-100D-3-20-60

Figure 3-32B. Vertical Stabilizer Front and Rear Beam Repair

PERMANENT—CATEGORY A

EMPENNAGE

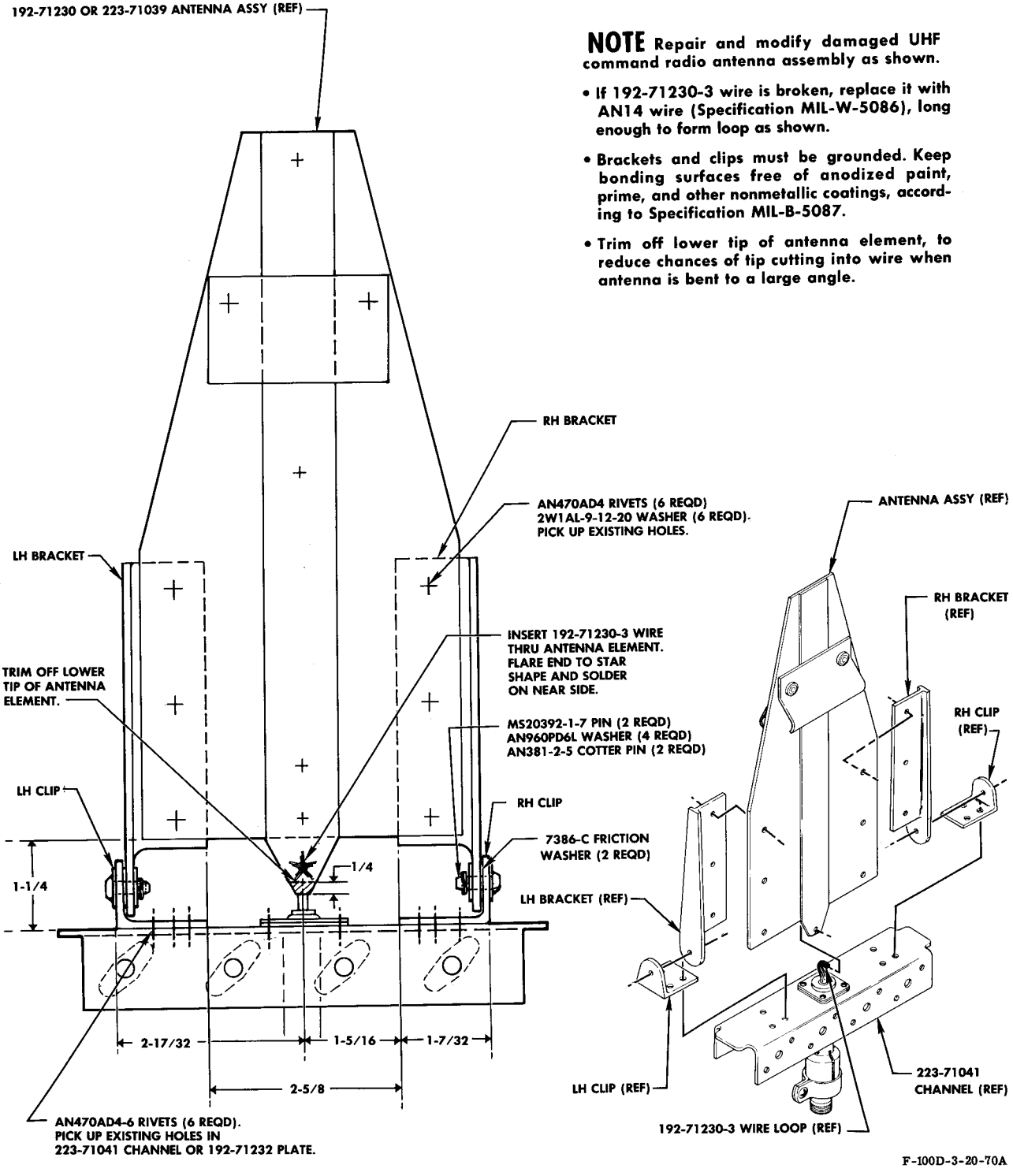
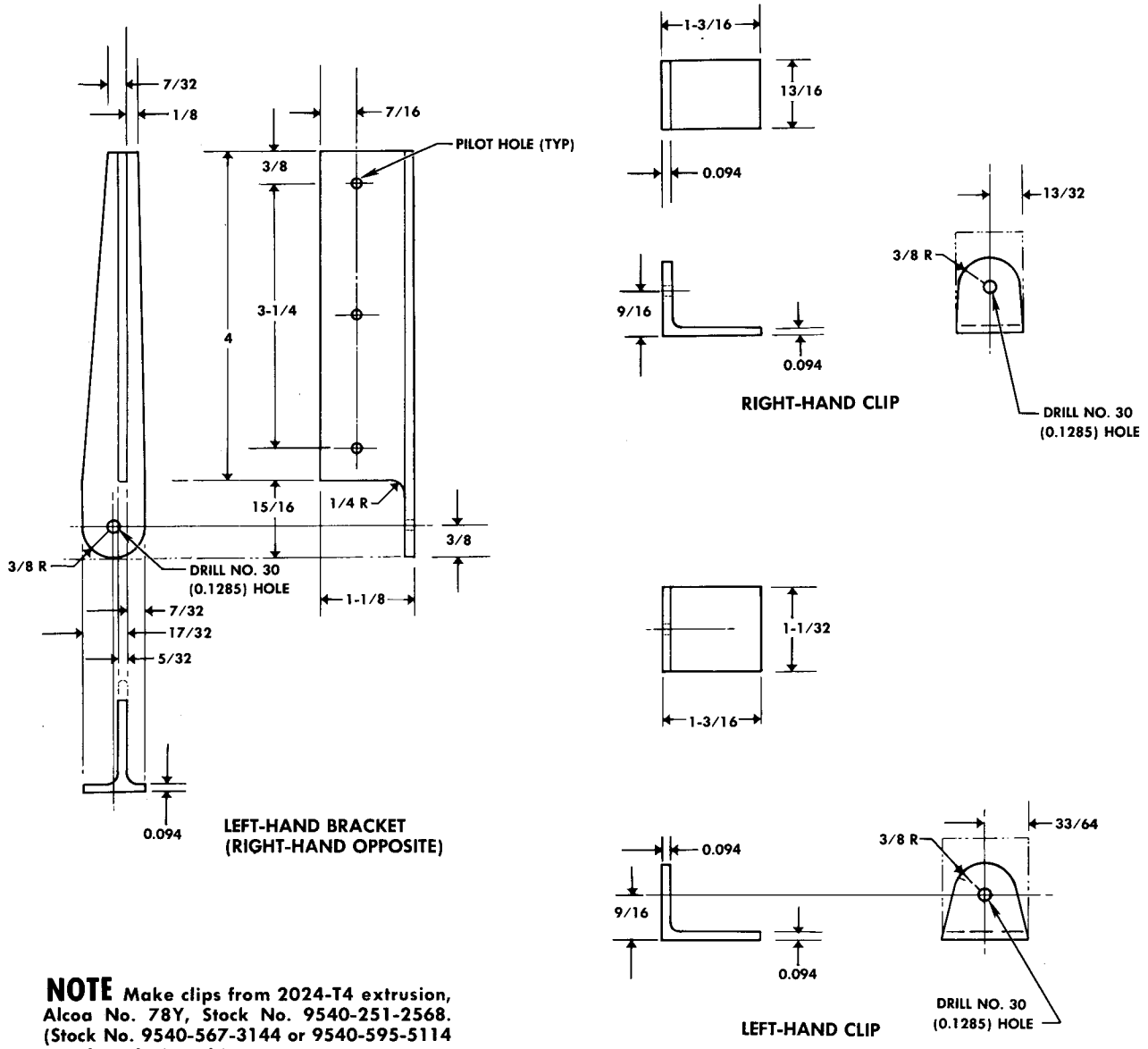


Figure 3-32C. Antenna Mount Rework (Sheet 1 of 2)

PERMANENT—CATEGORY A **EMPENNAGE**

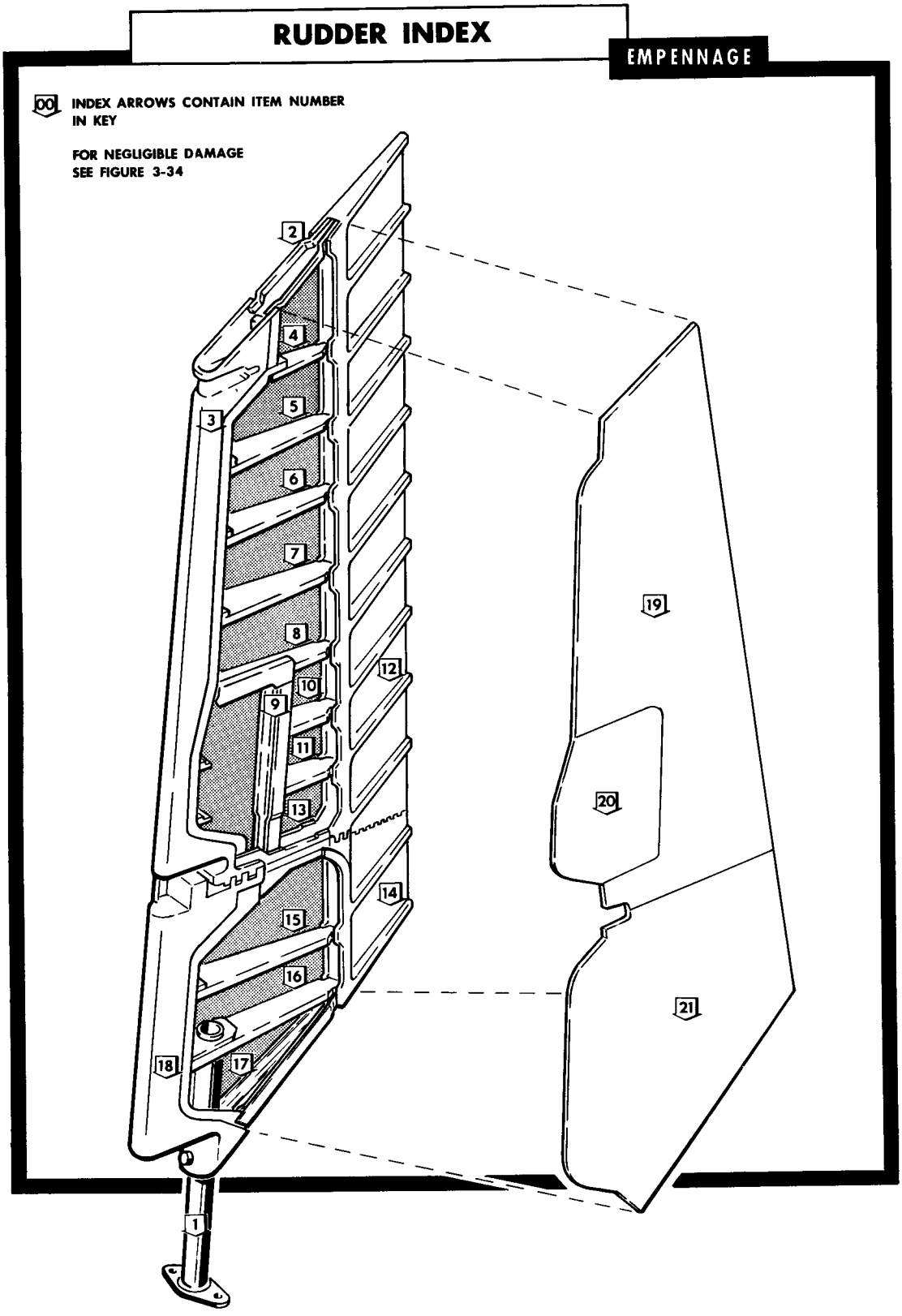


NOTE Make clips from 2024-T4 extrusion, Alcoa No. 78Y, Stock No. 9540-251-2568. (Stock No. 9540-567-3144 or 9540-595-5114 may be substituted.)

- Make brackets from 7075-T6 extrusion, AND10136-1505, Stock No .9540-580-6525.

F-100D-3-20-75

Figure 3-32C. Antenna Mount Rework (Sheet 2 of 2)



F-100D-3-20-25

Figure 3-33. Rudder Repair Index

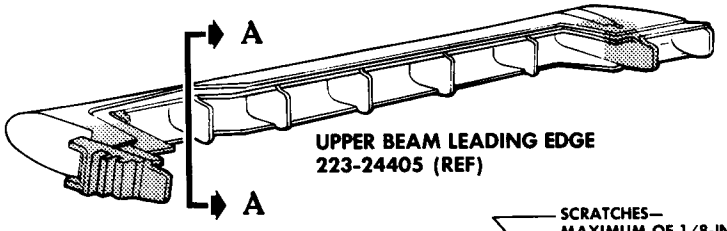
ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR Figure NO.†
1	223-24404	Tube			
2	223-24068	Rib	0.032	2024-T4 Clad	
3	223-24402	Beam Assy		356-T6 Sand Casting	
4	223-24065	Rib	0.040	2024-T4 Clad	
5	223-24059	Rib	0.032	2024-T4 Clad	
6	223-24053	Rib	0.032	2024-T4 Clad	
7	223-24047	Rib	0.032	2024-T4 Clad	
8	223-24042	Rib	0.040	2024-T4 Clad	
9	223-24101	Intercostal	0.032	2024-T4 Clad	
10	223-24036	Rib	0.032	2024-T4 Clad	
11	223-24031	Rib	0.032	2024-T4 Clad	
12	223-24420	Upper Trailing Edge		2024-T4 Bare Plate	3-34, 3-35, 3-36, 3-37
13	223-24026	Rib	0.051	2024-T4 Clad	
14	223-24421	Lower Trailing Edge		2024-T4 Bare Plate	3-34, 3-35, 3-36, 3-37
15	223-24020	Rib	0.032	2024-T4 Clad	
16	223-24015	Rib	0.040	2024-T4 Clad	
17	223-24004	Rib	0.064	2024-T4 Clad	
18	223-24403	Beam TE Lower		356-T6 Perm Mold Casting	
19	223-24002-3, 4	Upper Skin	0.032	2024-T3 Clad	
20	223-24102	Door	0.032	2024-T3 Clad	
21	223-24003-3, 4	Lower Skin	0.032	2024-T3 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 3-33

RUDDER NEGLIGIBLE DAMAGE

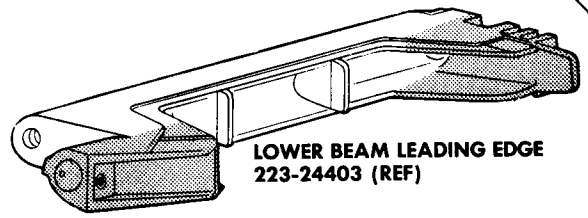
EMPENNAGE

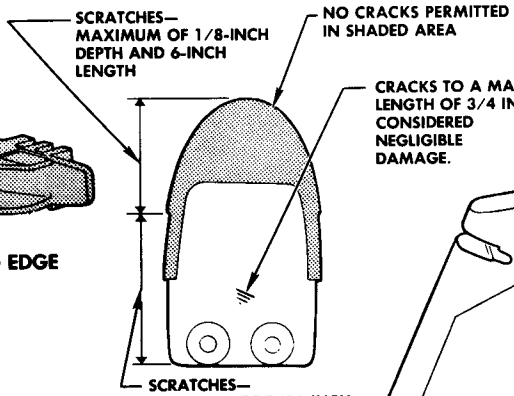


LEADING EDGE MAIN BEAMS

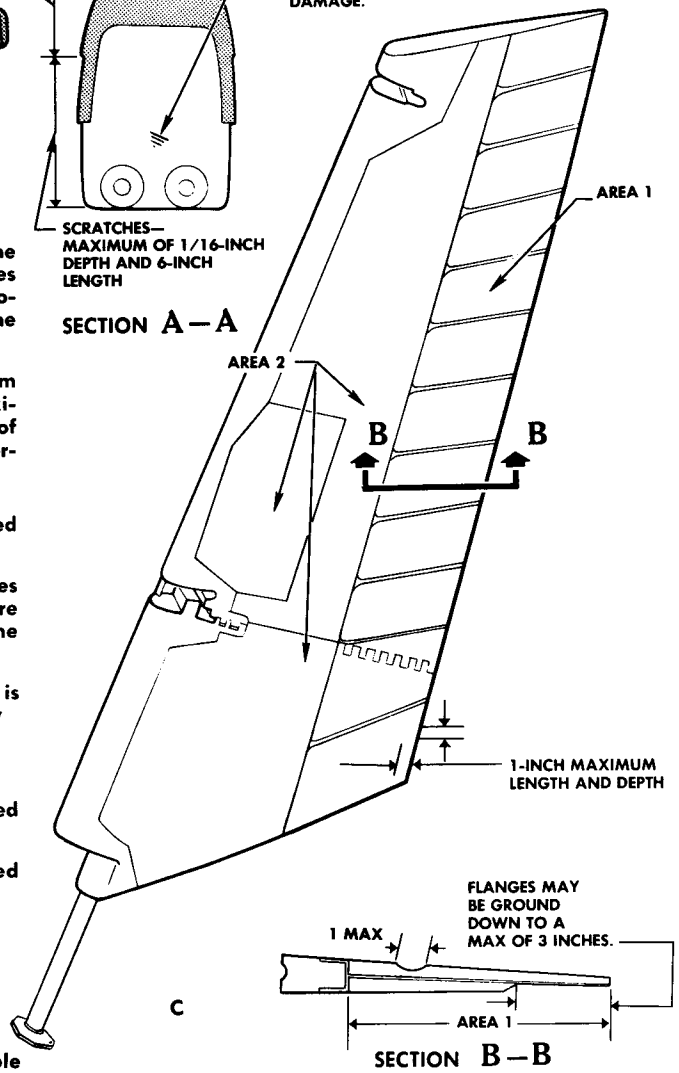
NEGLIGIBLE DAMAGE LIMITS ARE SHOWN IN SECTION A-A.

NO HOLE OR CRACK DAMAGE IS CONSIDERED NEGLIGIBLE IN THIS AREA. SCRATCHES WHICH DO NOT EXCEED A DEPTH OF 0.030 INCH ARE ALLOWED.





SECTION A—A



SECTION B—B

SIDE PANELS AND TRAILING EDGE

AREA 1

- The aft part of the flanges may be removed flush with the web, provided the damage does not extend over 3 inches from the trailing edge, provided the flange on the opposite side is undamaged, and provided there is at least one undamaged flange above or below on either side.
- The forward flanges may be damaged to a maximum depth of 25 percent of the flange height and to a maximum length of one inch. Smooth any rough surfaces of damaged area. Flange damage which is less than 10 percent of the flange height may be left as is.
- Scratches are considered negligible if they do not exceed 0.025-inch depth and 6-inch length.
- Cracks not exceeding one inch spanwise or 2 inches chordwise are considered negligible, provided there are no more than two cracks in any section between the flanges.
- Trailing edge crack or nick damage is negligible if it is less than one inch in length and depth. Smooth any rough surfaces of damaged area.

AREA 2

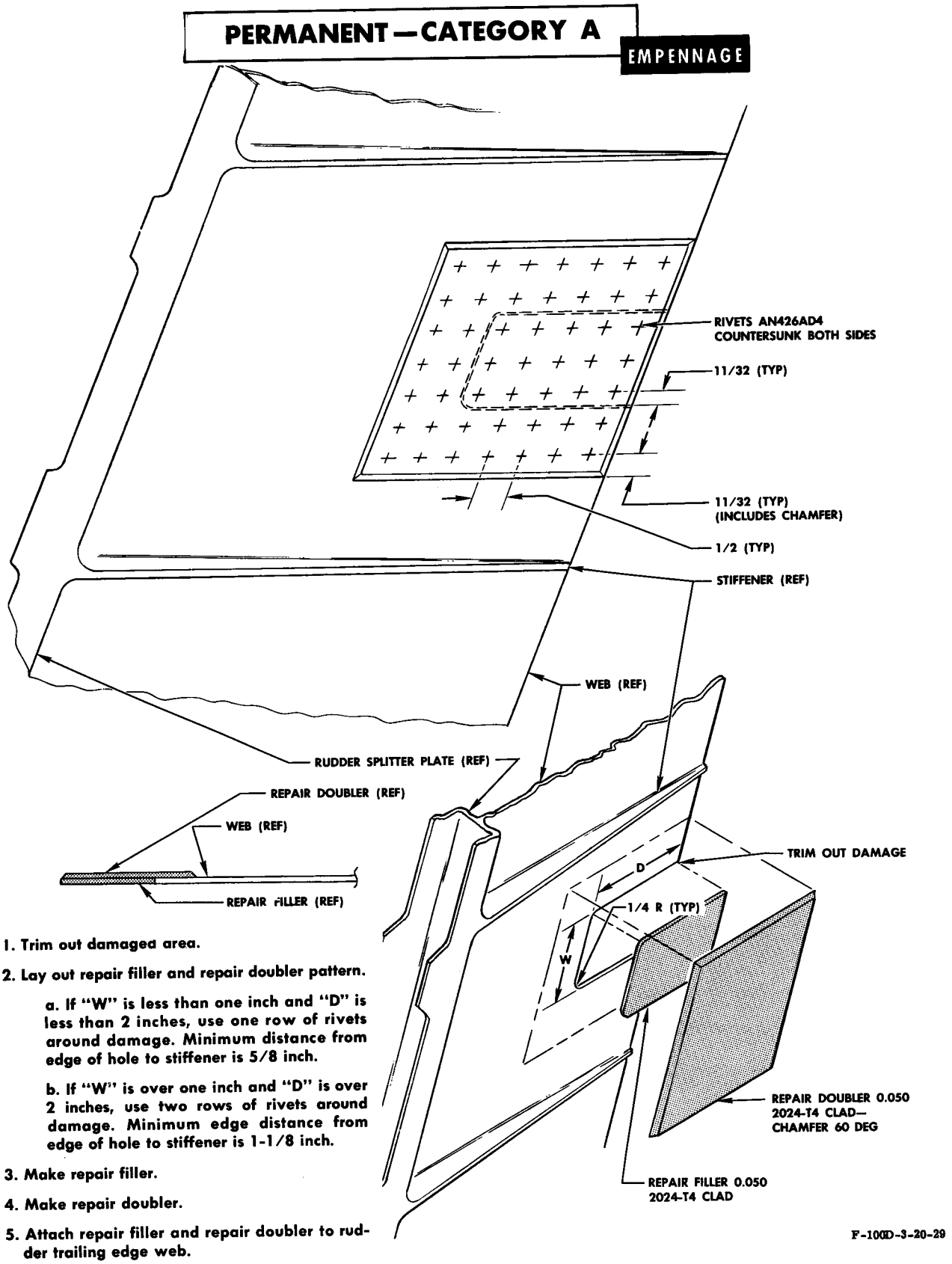
- Scratches are considered negligible if they do not exceed 0.016-inch depth and 6-inch length.
- Cracks which are not in rivet rows may be considered negligible if they are less than 2 inches long.

NOTE

- All cracks must be stop-drilled, using a No. 40 drill.
- Unrepaired hole damage is not considered negligible damage.

F-100D-3-20-64

Figure 3-34. Rudder Negligible Damage



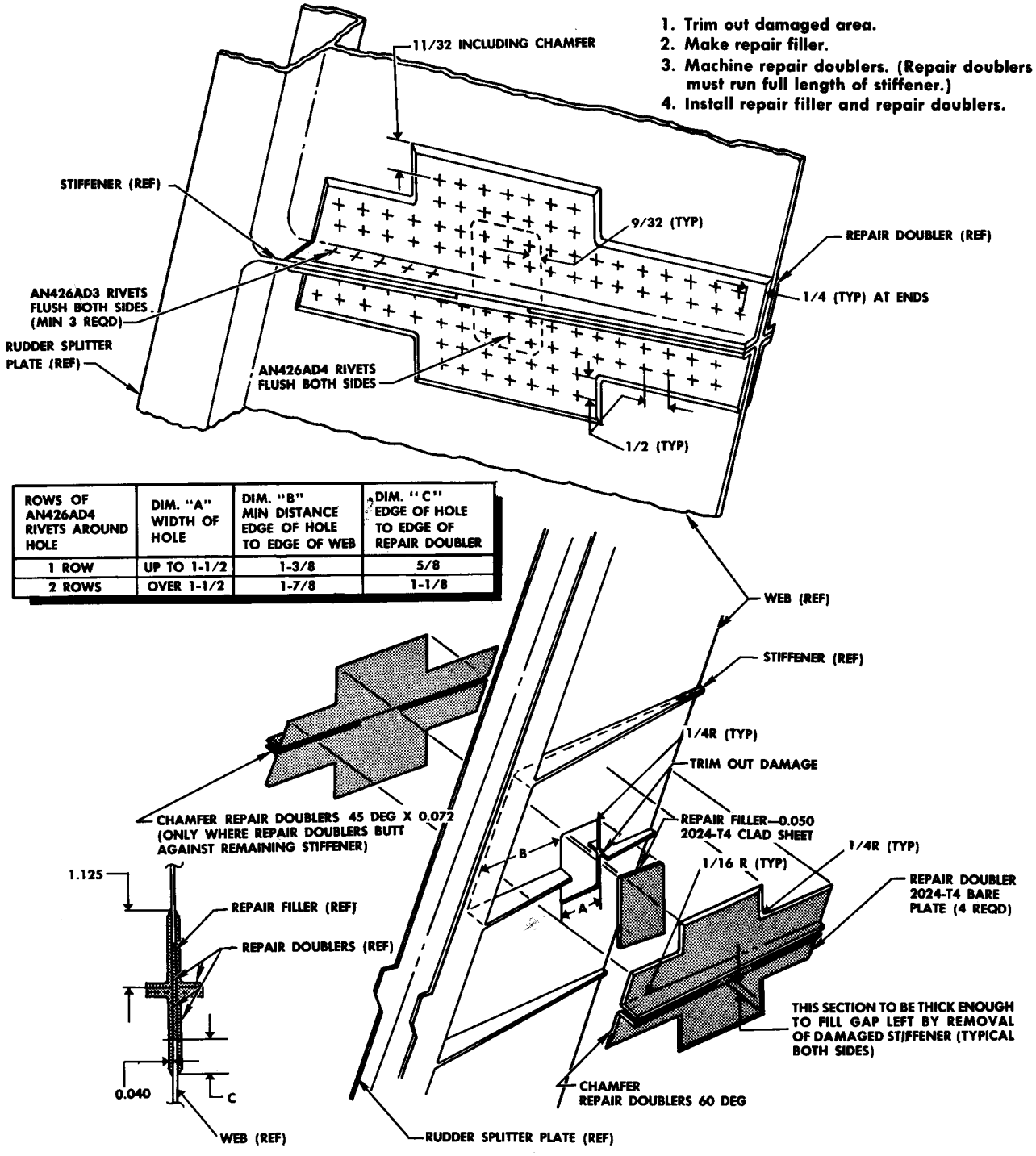
1. Trim out damaged area.
2. Lay out repair filler and repair doubler pattern.
   a. If "W" is less than one inch and "D" is less than 2 inches, use one row of rivets around damage. Minimum distance from edge of hole to stiffener is 5/8 inch.
   b. If "W" is over one inch and "D" is over 2 inches, use two rows of rivets around damage. Minimum edge distance from edge of hole to stiffener is 1-1/8 inch.
3. Make repair filler.
4. Make repair doubler.
5. Attach repair filler and repair doubler to rudder trailing edge web.

Figure 3-35. Rudder Trailing Edge

PERMANENT—CATEGORY B

EMPENNAGE

1. Trim out damaged area.
2. Make repair filler.
3. Machine repair doublers. (Repair doublers must run full length of stiffener.)
4. Install repair filler and repair doublers.

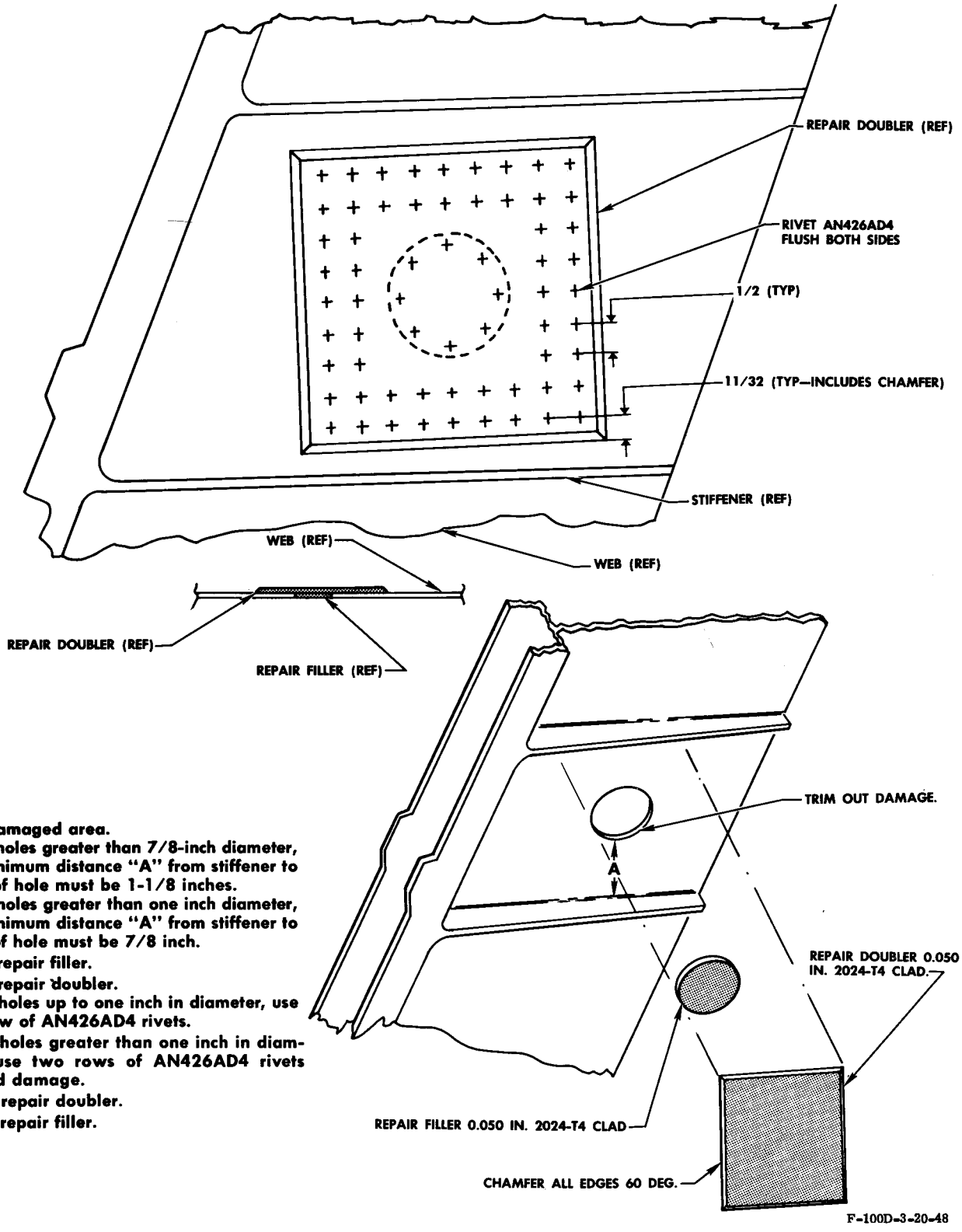


ROWS OF AN426AD4 RIVETS AROUND HOLE	DIM. "A" WIDTH OF HOLE	DIM. "B" MIN DISTANCE EDGE OF HOLE TO EDGE OF WEB	DIM. "C" EDGE OF HOLE TO EDGE OF REPAIR DOUBLER
1 ROW	UP TO 1-1/2	1-3/8	5/8
2 ROWS	OVER 1-1/2	1-7/8	1-1/8

Figure 3-36. Rudder Trailing Edge Stiffeners

PERMANENT—CATEGORY A

EMPENNAGE



1. Trim damaged area.
   a. For holes greater than 7/8-inch diameter, the minimum distance "A" from stiffener to edge of hole must be 1-1/8 inches.
   b. For holes greater than one inch diameter, the minimum distance "A" from stiffener to edge of hole must be 7/8 inch.
2. Make repair filler.
3. Make repair doubler.
   a. For holes up to one inch in diameter, use one row of AN426AD4 rivets.
   b. For holes greater than one inch in diameter, use two rows of AN426AD4 rivets around damage.
4. Install repair doubler.
5. Install repair filler.

Figure 3-37. Rudder Trailing Edge Web

TEMPORARY CATEGORY A

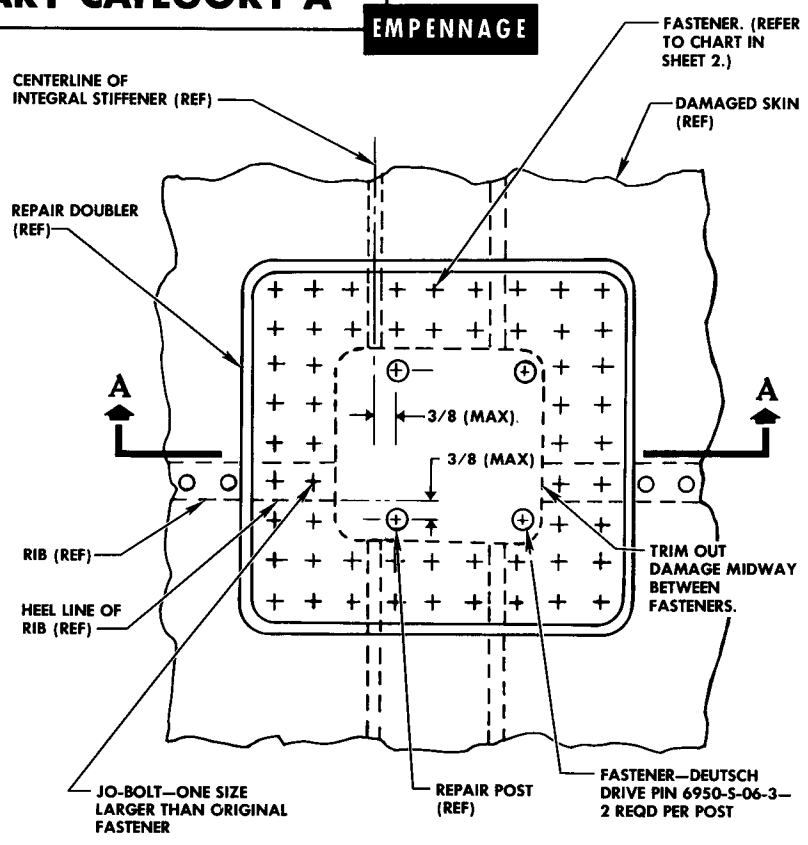
EMPENNAGE

NOTE This repair is applicable to one or both milled skins within the shaded areas shown on sheet 2.

- A 4130 steel doubler heat-treated to 140-16,000 psi may be substituted for 7075-T6 aluminum alloy, using the minimum of 50 percent of the thickness of the aluminum alloy doublers required. All doublers, however, must meet the minimum thickness for countersinking given in the table.
- Repair post spacing not to exceed 3 inches.
- Refer to text for additional fabrication and assembly details.

MINIMUM MATERIAL THICKNESS FOR COUNTERSINKING

FASTENER	MATERIAL
AF462-6 (CHERRY)	0.063
F200 JO-BOLT	0.063
F260 JO-BOLT	0.080
F312 JO-BOLT	0.100



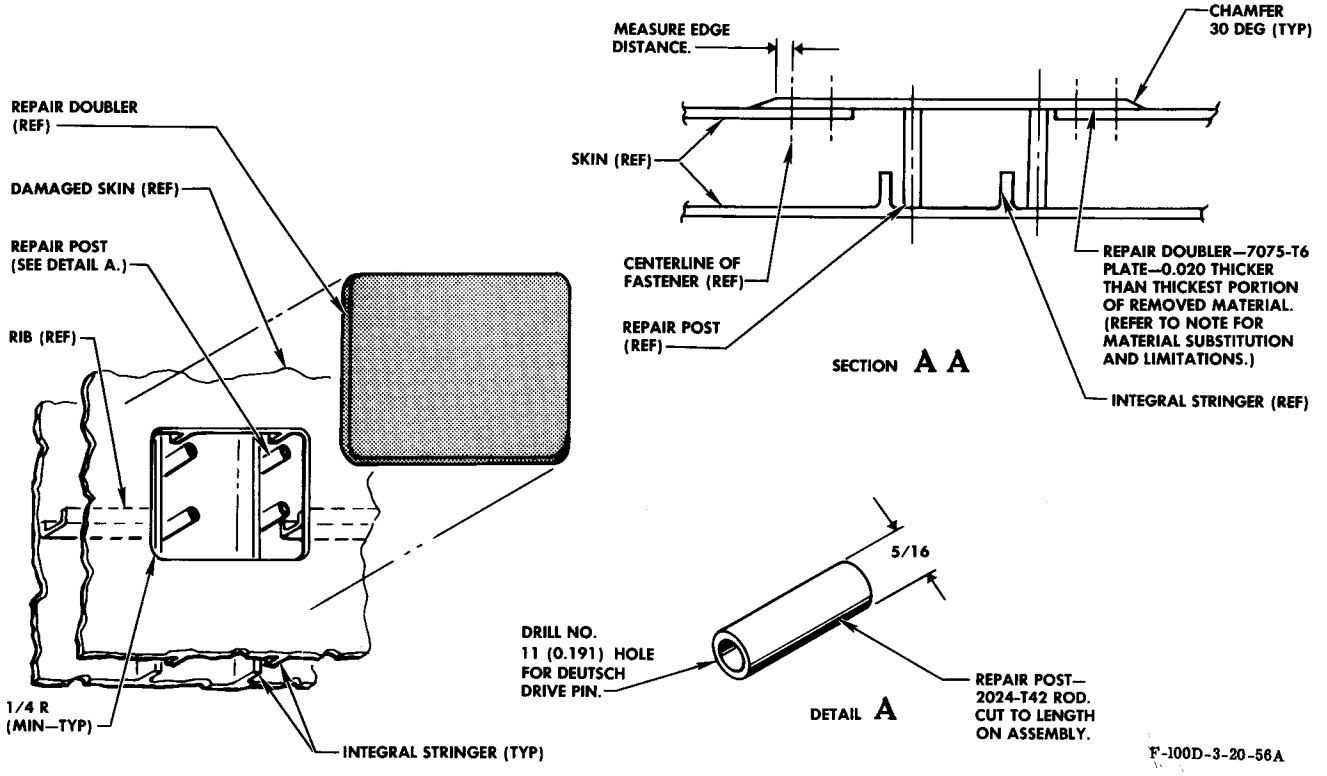


Figure 3-38. Temporary Repair of Horizontal and Vertical Stabilizer Milled Skin (Sheet 1 of 2)

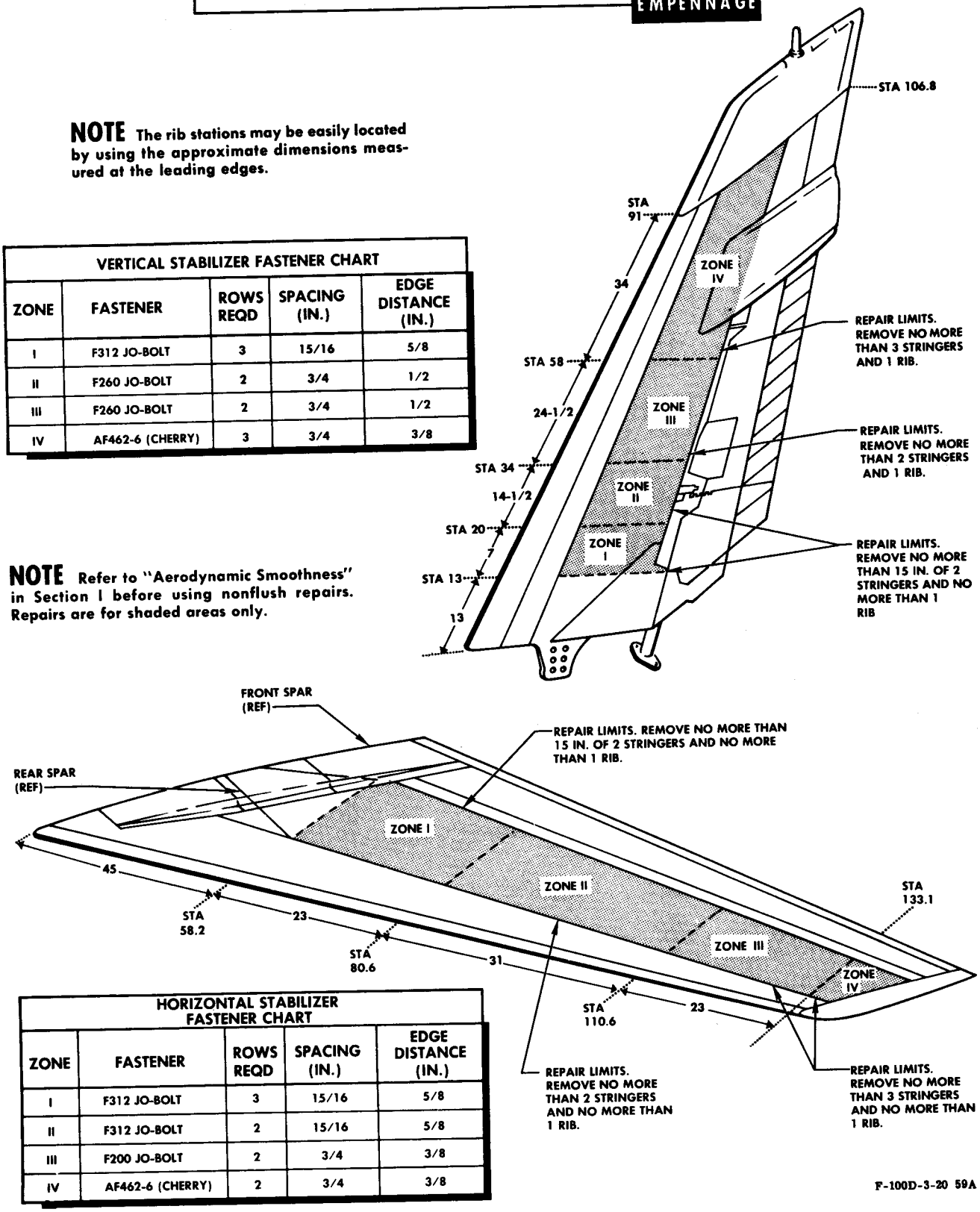
TEMPORARY—CATEGORY A

EMPENNAGE

NOTE The rib stations may be easily located by using the approximate dimensions measured at the leading edges.

VERTICAL STABILIZER FASTENER CHART				
ZONE	FASTENER	ROWS REQD	SPACING (IN.)	EDGE DISTANCE (IN.)
I	F312 JO-BOLT	3	15/16	5/8
II	F260 JO-BOLT	2	3/4	1/2
III	F260 JO-BOLT	2	3/4	1/2
IV	AF462-6 (CHERRY)	3	3/4	3/8

NOTE Refer to "Aerodynamic Smoothness" in Section I before using nonflush repairs. Repairs are for shaded areas only.

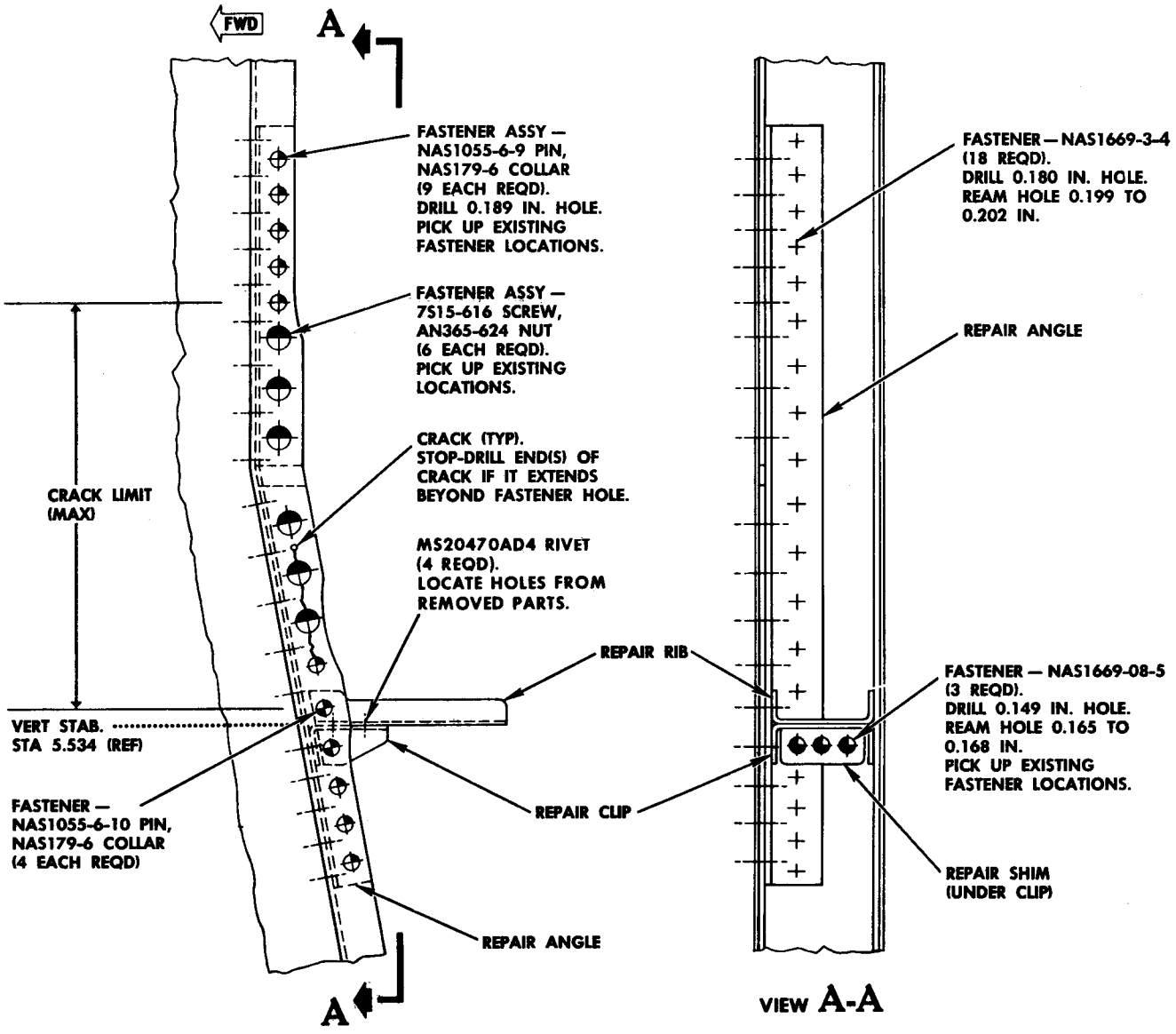


HORIZONTAL STABILIZER FASTENER CHART				
ZONE	FASTENER	ROWS REQD	SPACING (IN.)	EDGE DISTANCE (IN.)
I	F312 JO-BOLT	3	15/16	5/8
II	F312 JO-BOLT	2	15/16	5/8
III	F200 JO-BOLT	2	3/4	3/8
IV	AF462-6 (CHERRY)	2	3/4	3/8

Figure 3-38. Temporary Repair of Horizontal and Vertical Stabilizer Milled Skin (Sheet 2 of 2)

PERMANENT—CATEGORY B

EMPENNAGE



NOTE This repair is applicable to vertical stabilizer rear beams when damage is within limits shown.

1. Remove fairing to gain access to damaged area.
2. Remove rib and clip from aft side of beam at vertical stabilizer station 5.534.
3. Remove fasteners as required for installation of repair angle. Save the six 7S15-616 screws and AN365-624 nuts for reinstallation.
4. Stop-drill crack if applicable.
5. Fabricate repair parts. (See details 1 through 4, on sheet 2.) All bend radii are 0.094 inch unless otherwise noted.
6. Heat-treat repair parts.
7. Chemical-film-treat and zinc-chromate-prime (two coats) repair parts and other bare aluminum alloy surfaces.
8. Install repair parts. Install fasteners wet with zinc-chromate primer (Specification MIL-P-8585).
9. Reinstall fairing.

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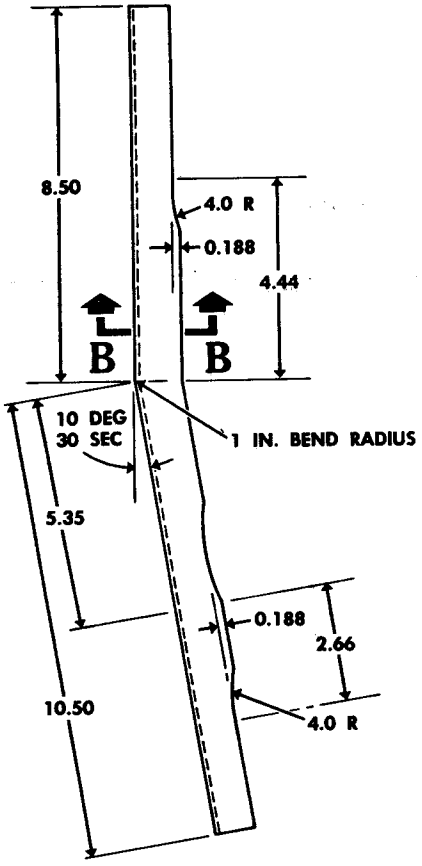
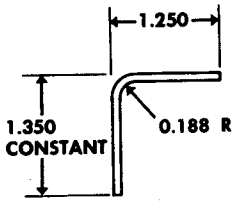
Figure 3-38A. Vertical Stabilizer Rear Beam Repair (Sheet 1 of 2)

PERMANENT—CATEGORY B

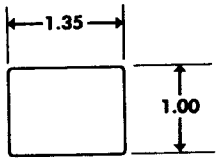
EMPENNAGE



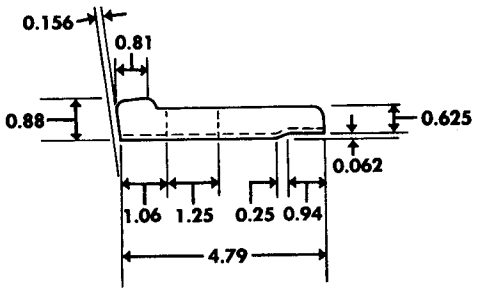
DETAIL 1 — REPAIR ANGLE
(LH SHOWN, RH OPPOSITE)

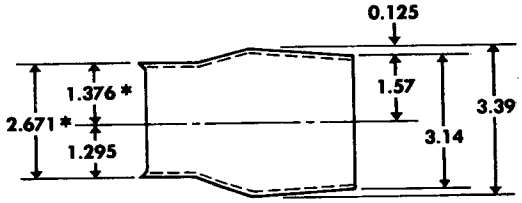


SECTION B-B

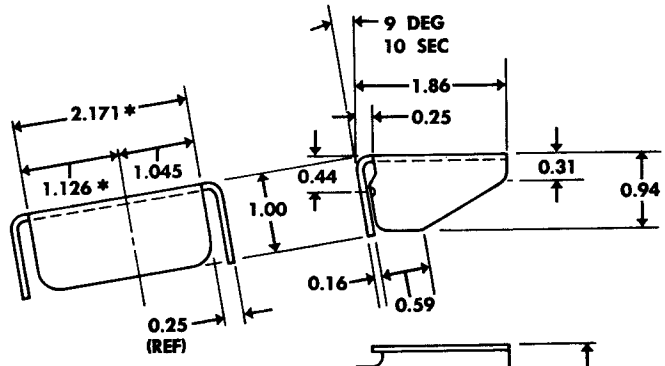


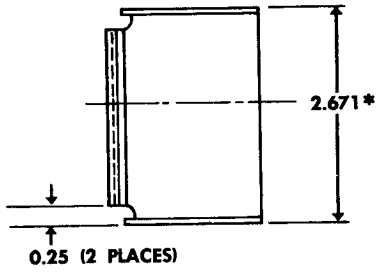
DETAIL 4 — REPAIR SHIM





DETAIL 2 — REPAIR RIB
(LH SHOWN, RH OPPOSITE)





DETAIL 3 — REPAIR CLIP
(LH SHOWN, RH OPPOSITE)

NOMENCLATURE	MATERIAL	GAGE (IN.)	HEAT-TREAT (AFTER FORMING)
REPAIR ANGLE	7075-O CLAD	0.081	7075-T6
REPAIR RIB	2024-O CLAD	0.050	2024-T4
REPAIR CLIP	2024-O CLAD	0.064	2024-T4
REPAIR SHIM †	7075-T6 CLAD	0.081	——

* If repair is installed on both sides of beam, decrease dimension by 0.081 inch.

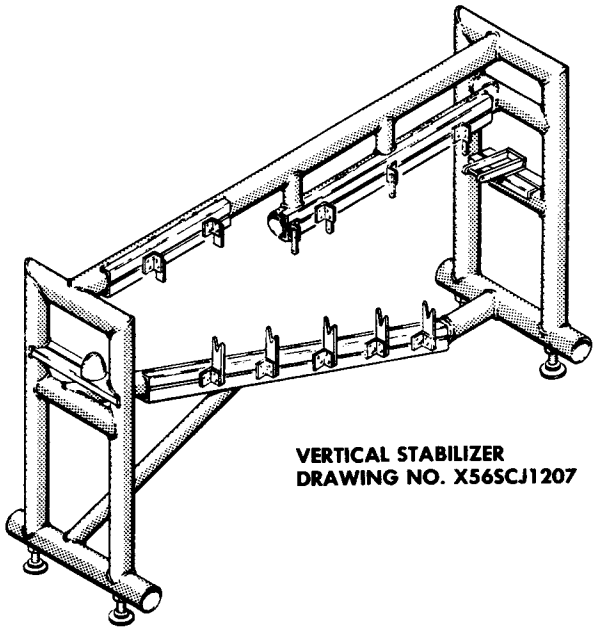
† Not required if repair is installed on both sides of beam.

100D-3-20-84

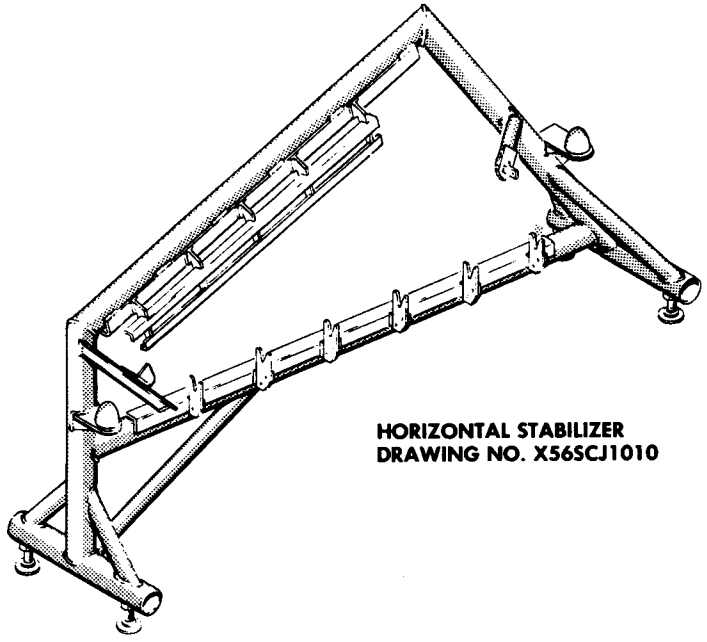
Figure 3-38A. Vertical Stabilizer Rear Beam Repair (Sheet 2 of 2)

REPAIR JIGS

EMPENNAGE



VERTICAL STABILIZER
DRAWING NO. X56SCJ1207



HORIZONTAL STABILIZER
DRAWING NO. X56SCJ1010

NOTE The repair jigs for the horizontal and vertical stabilizer may be ordered from the Sacramento Air Materiel Area (SMAMA). (Refer to drawing numbers shown.)

F-100D-3-20-61

Figure 3-39. Horizontal and Vertical Stabilizer Repair Jigs

HOLDING FIXTURE DIMENSIONS

EMPENNAGE

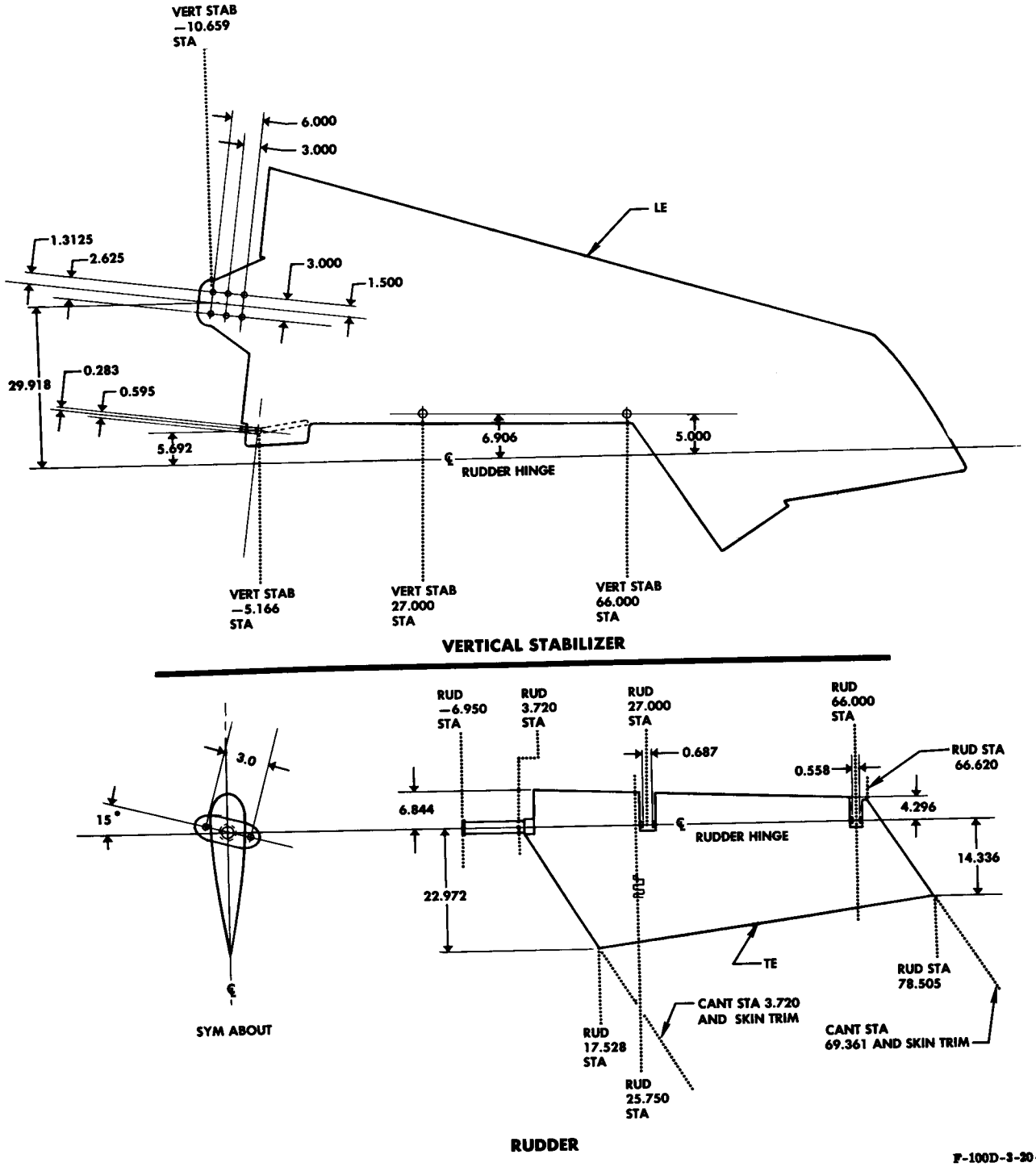


Figure 3-40. Holding Fixture Dimensions—Vertical Stabilizer and Rudder

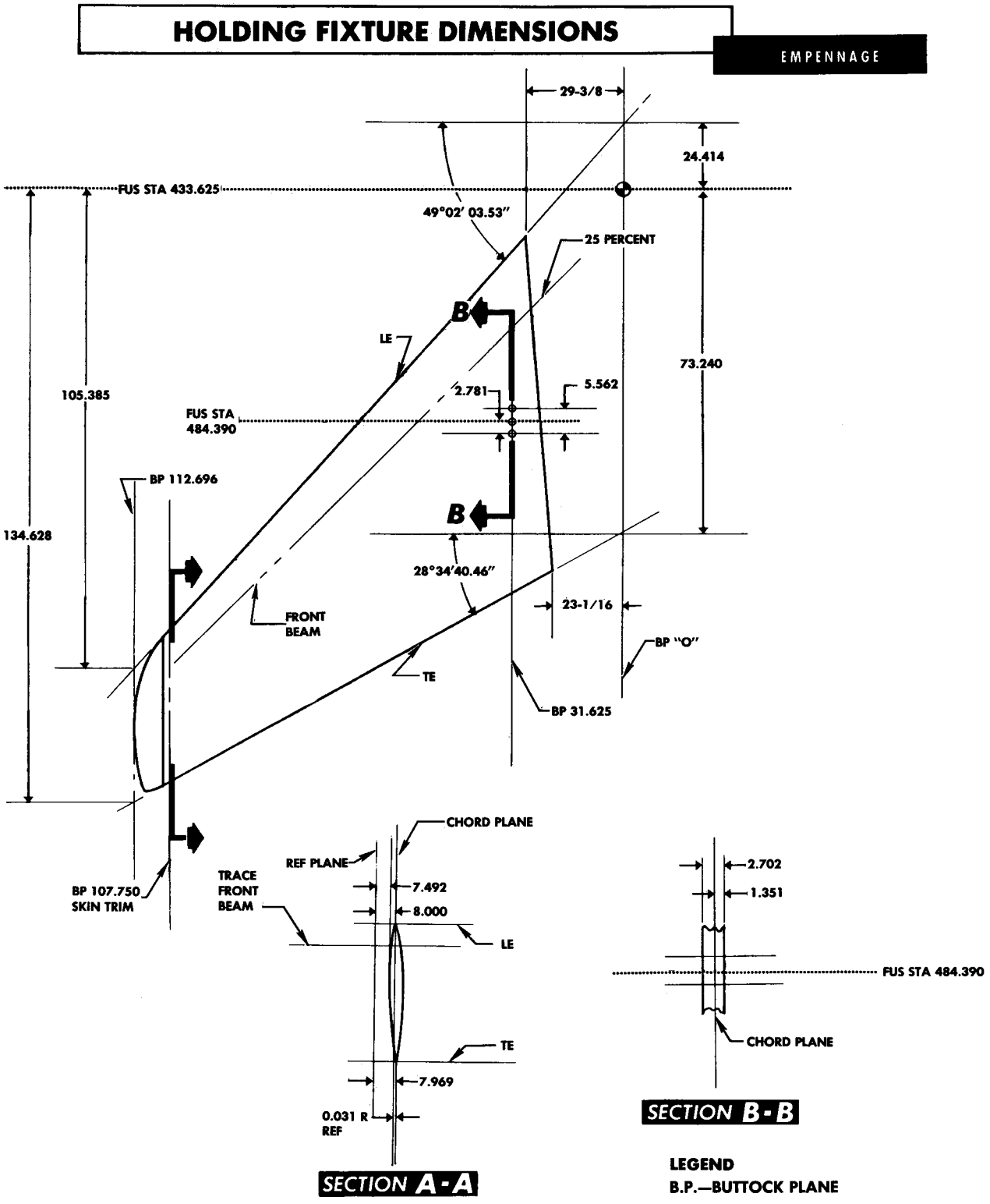


Figure 3-41. Holding Fixture Dimensions—Horizontal Stabilizer

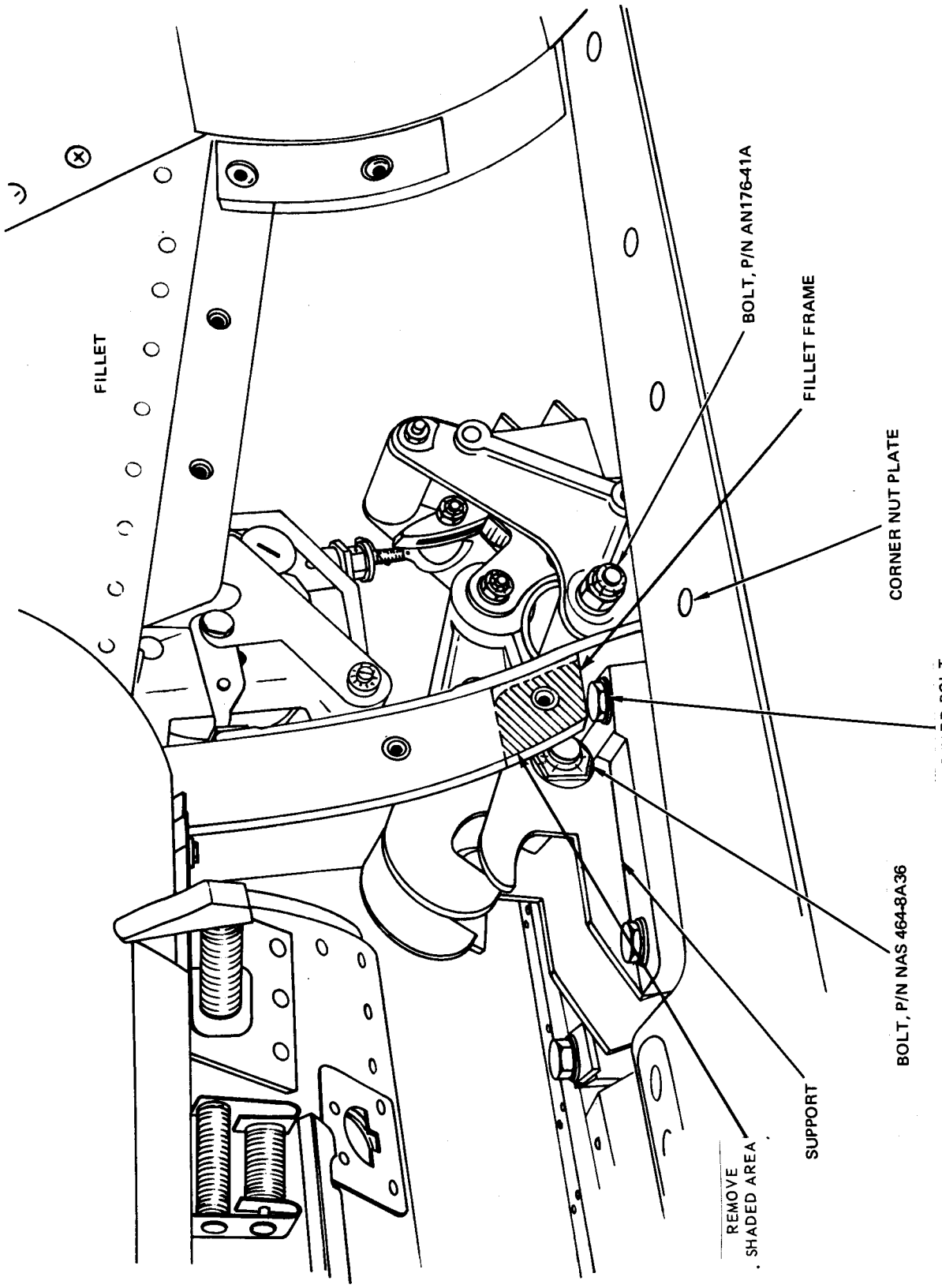


Figure 3-42. Drag Chute-Frame Fillet Rework.

Section IV

FUSELAGE GROUP

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FUSELAGE REPAIR INDEX	4-2C
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FUSELAGE DESCRIPTION.

The complete fuselage is of the semimonocoque type with multiple longerons and a stressed skin to carry the bending and shear load in the forward and forward intermediate sections. The aft intermediate and aft sections have four basic longerons to carry the bending load and a stressed skin which carries the shear load. Frames and bulkheads interconnect with the longerons, stringers, and skins to also help distribute the loads.

The field break for engine removal is at station 389 which separates the aft fuselage from the aft intermediate section of the fuselage.

The basic material used in the forward fuselage section is aluminum alloy. In the aft fuselage section, a combination of aluminum alloy, titanium, and corrosion-resistant steel is used.

Extensive use has been made of Chem-Milled skins in the F-100D and F-100F duct skins. Chem-Milling is a process sculpturing by means of a chemical bath. The part is masked off and immersed in a chemical solution. The chemical etches away the material which is not masked off. The depth of the etch is controlled by the length of time the material is left in the bath. This provides for a skin to be made up with thickened areas where necessary, such as along fastener rows, while the rest of the skin is thin. The use of honeycomb panels is also used in the construction of the fuselage. Honeycomb panels are a sandwich panel with a honeycomb core. The upper and lower surfaces are sheets which are thermobonded to the core. Both metal honeycomb and glass honeycomb are used in the fuselage. This type of structure gives great strength at minimum weight.

Much of the fuselage structure is sealed for various reasons. Pressure sealing, liquid sealing, fume-tight sealing, etc., are found throughout the length of the structure. If damage to the structure is such that the sealing of the fuselage is destroyed or damaged, it will be necessary to reseal. For information outlining the

type of sealant to be used and methods of application, refer to Section I.

FORWARD FUSELAGE SECTION.

The forward fuselage section is made up of three production assemblies. The forwardmost of these, the nose section, begins at station 9.5 and extends to station 106 for the F-100D. For the F-100F the nose station is —25 and extends to station 71. It provides housing for radar and electrical equipment, autopilot equipment if applicable, and forms a part of the engine air intake duct. The radome assemblies in the upper lip of the engine air intake duct and at the top and bottom of the fuselage between stations 46 and 72 on the F-100D and stations 11.5 and 37 on the F-100F are made of laminated glass fabric. The skins are of 2024-T4 and 7075-T6 sheet stock.

The forward intermediate section extends from station 106 to station 222.5 on the F-100D and from station 71 to station 222.5 on the F-100F. The forward intermediate sections of the F-100D and F-100F Airplanes differ in that the F-100F forward intermediate section has been lengthened 35 inches to make room for the rear cockpit. Because of the 35 inches added on the F-100F, the station lines forward of station 222.5 are not the same on the F-100D and F-100F. Aft of station 222.5 the station lines are the same on both airplanes. The forward intermediate section on both the F-100D and the F-100F contain the armament facilities, liquid oxygen equipment, some electrical units, the nose wheel well, the pressurized cockpit and also forms a part of the engine air intake duct. The skin is 7076-T6 sheet stock and Chem-Milled skins. Honeycomb sandwich panels are also used in this area. The pressurized cockpit extends from station 106 aft to canted station 168.118 on the F-100D and from station 71 to canted station 193.501 on the F-100F. The nose gear beam starts at station 106 and continues on back to station 222.5 on the F-100D. On the F-100F the nose gear beam starts at Station 135 and extends to Station 222.5. The nose gear trunnion is located at station 153. The bulkhead at canted station 168.118 on the F-100D supports the pilot's ejection seat. On the F-100F the pilots ejection seats are supported by canted stations 135.118 and 193.501. The canopy is a hinged clamshell type with a pivot point at the rear of the enclosure. It is a one-piece structure fabricated from acrylic plastic. Glass fabric laminations bonded to the edge of the plastic facilitate attachment of the canopy to a frame.

The windshield consists of three laminated glass panels mounted in a frame assembly. The side panels are made up of an inner and an outer semitempered glass pane bonded to a ⅛-inch vinyl plastic core. A metal insert is embedded in the vinyl plastic at the edges to provide attachment strength. The center panel is made up of a series of four semitempered glass panes bonded together with a thin sheet of vinyl plastic between the panes. This panel has no metal insert and is held in place by the retainer strip.

If a windshield is damaged, replacement of the windshield should be determined by the seriousness of the damage. Setting a hard and fast rule as to when a windshield should or should not be replaced is difficult and impractical because of the many variable factors to be considered. It is, therefore, the responsibility of the personnel closely associated with the airplane to make this decision. Factors which must be considered when evaluating damage to windshields are pilot's visibility and efficiency (the windshield area must be kept clear for maximum gun sighting efficiency) and the effects of the damage on the structural strength of the windshield. Scratches may develop into cracks which get progressively larger and eventually result in possible failure. Therefore, any damage considered negligible and permitted to remain "as is" should be checked periodically for progression. A scratch that develops into a crack means the windshield will have to be replaced. A cracked windshield does not have the required structural strength.

A honeycomb glass fabric panel is used as an antenna door assembly in the top of the fuselage between stations 231 and 254.125.

The aft intermediate section extends from station 222.5 to the field break at station 389. It contains hydraulic equipment, air conditioning units, main gear wheel wells, and forms the balance of the engine air intake duct. The duct ends at station 305.25. The compressor section of the engine mates with the duct at this point. The forward support of the engine is located at station 325.6. Wing attachment points are located at stations 222.5 and 305.25. The speed brake is hinged just aft of station 222.5. The forward and center fuel cells, and a small cell mounted below the wing center section, are contained in this section. Each cell is of the nylon bladder type supported by metal liners.

The main gear doors extend from station 300 to station 334, with hinge lines along the lower centerline of the fuselage. The forward hinge and actuating cylinder are located at the station 300 bulkhead. The aft hinge is located at the station 334 bulkhead. Fuselage deformation will not load up the doors.

Because of the many cutouts in the skin of the forward fuselage section, the engine air intake duct is of major structural importance and carries a great portion of the bending and torsion loads. All of the engine air intake duct skins are Chem-Milled.

Because of the many inaccessible areas of the engine air intake duct and of the difficulty in making blind flush repairs on Chem-Milled skins, nonflush repairs are permitted. There are no limits to either the size or number of repairs. Edges of all repair plates must be chamfered 30 degrees and, wherever possible, installed with AN426AD5 rivets. In blind or inaccessible areas, AF463-5 Cherry rivets may be used.

AFT FUSELAGE SECTION.

The aft fuselage section extends from station 389 to 540.6. Four quick-disconnect fittings are provided at station 389 for removal of the aft section. Shear is transmitted through this joint by shear pins in matching holes around the circumference of the two matching frames.

Because of the high temperatures existing in operation, the aft section of the fuselage is made almost entirely of titanium and steel. Only those areas which are locally shrouded from the engine and some of the structure forward of canted station 401, which is the fire wall, are made of aluminum. A special Monel rivet (NAA Part No. 2R14-6) having a 3/16-inch shank and a head the same size as a 3/16-inch Hi-Shear rivet is generally used to fasten titanium skin to structure. Standard heated 3/16-inch Hi-Shear rivet dimple dies are used when dimpling for these special Monel rivets. Rivets should be ordered along with replacement parts. The aft section houses the aft fuel cell, the aft portion of the engine, drag chute, horizontal stabilizer actuating mechanism, and retractable tail skid. It provides attachment for the dorsal fin and vertical and horizontal stabilizers. The aft support for the engine is located at station 440. The vertical stabilizer is mounted on a large arch-shaped steel forging which is canted and extends from station 440 at the centerline of thrust up through the fuselage to the spar of the vertical stabilizer. This connection transmits all bending of the vertical stabilizer to the fuselage. The horizontal tail is attached to the frame at station 485. Because this frame is not a complete frame around the boundary of the fuselage, all unsymmetrical horizontal tail loads are taken as bending in the side panels and transmitted to the frames at station 475 and 495. These frames redistribute the torque into the full fuselage torque section.

PRIMARY AND SECONDARY STRUCTURE.

Primary and secondary structure areas are shown in the typical repair index. (See figure 10-1.)

ONE-TIME FLIGHT.

Allowable unrepaired damage for a ferry flight is shown in figure 4-1. Also, refer to "One-time Flight Damage" in Section I.

SKIN DIAGRAMS.

The skin diagrams show the material type, hardness condition, and thickness of each panel on the fuselage and in the duct.

FUSELAGE SKIN DIAGRAM.

For F-100D Airplanes, see figure 4-2. For F-100F Airplanes, see figure 4-2A.

DUCT SKIN DIAGRAM.

For F-100D and F-100F Airplanes, see figure 4-3.

SKIN GAP TOLERANCES.

The skin gap tolerance drawings show the allowable gaps in the fuselage skin butt joints. (See figures 4-3A and 4-3B.)

FUSELAGE NEGLIGIBLE DAMAGE.

Fuselage negligible damage illustrations show the damage that may exist without structural repairs. For fuselage negligible damage on F-100D Airplanes, see figure 4-4. For fuselage negligible damage on F-100F Airplanes, see figure 4-4A. For treatment or rework of negligible damage, refer to "Negligible Damage" in Section I.

FOLDING PITOT BOOM REPAIR AND TESTING.

Figure 4-5A shows the allowable damage and repairs for the folding pitot boom. In addition to the repairs shown, the static tubes should be tested for leaks as follows: Seal static-pressure openings by any suitable means. Attach a source of vacuum and a manometer or altimeter to the line being tested. Slowly apply suction to simulate 10,000 feet; then close off vacuum source. If after one minute the reading is still 9000 feet or more, leakage is negligible and the unit is serviceable. A pressure drop to less than 9000 feet indicates a leak, and the line must be repaired. To isolate leak, disconnect vacuum lines and connect pressure lines.

CAUTION

Do not apply pressure to static lines with instruments connected.

To find a leak, apply pressure and use a mild soap and water solution. After leak is repaired, retest lines.

Note

A detailed test procedure is given in T.O. 1F-100D-2-7 and T.O. 1F-100F-2-7; the applicable procedure may be used if desired.

WINDSHIELD NEGLIGIBLE DAMAGE.

The windshield is divided into two principal areas, the critical area and the noncritical area. The critical area is that area where undistorted visibility is essential for the satisfactory operation of the sight system and for the pilot's operation of the airplane. The noncritical areas are those areas where visibility is obstructed by other parts of the airplane, such as the nose and the gun sight mount.

Allowable defects in the critical area are well-scattered bubbles not exceeding 1/32 inch in diameter; an occasional hairline scratch not exceeding 3 inches in length, provided visibility is not impaired; and small lint particles that are embedded in the plastic layers or trapped within the laminations.

In addition to the defects allowed in the critical area, the noncritical areas may contain well-scattered bubbles no larger than 1/16 inch; an occasional open bubble (bubbles which have been broken into by grinding, leaving a hemispherical hole in the surface of the glass) not exceeding 1/32 inch in diameter; hairline scratches over 3 inches in length and wavy transparent lines, provided distortion limits are not exceeded.

When delamination of a windshield is discovered, the windshield need not be replaced unless the delamination has progressed more than 3/4 inch into the daylight opening of the windshield or the length of the delamination has exceeded one third of the total length of the side on which it has appeared. (See figure 4-13B.) Small cracks, chips, or nicks in the individual plates of a panel greatly reduce the structural integrity of the panel. Careful consideration should be given in approving such defects for service. The following criteria should be used to determine a serviceable or nonserviceable windshield center or side panel. See figure 4-13B for location of zones.

1. Zones A and B—Scratches 0.002 inch in depth of unlimited length are permissible and a maximum of five scratches is permissible, provided no two scratches are closer than 2 inches from each other. Well-scattered nicks not exceeding 0.002 inch in depth are also permissible.

2. Zone C—A maximum of three scratches 0.001 inch in depth of unlimited length is permissible, provided no two scratches are closer than 3 inches from each other.

3. Zones A, B, and C—A grouping of scratches not exceeding a maximum area of 2 inches in diameter is permissible, provided they do not obstruct visibility.

If there is any question as to strength of a panel, the Structural Repair Officer should be consulted.

CANOPY NEGLIGIBLE DAMAGE.

The following information may be used to determine whether the amount of damage to the canopy glass is permissible or reparable:

DEFINITIONS OF AREAS AND DEFECTS.

PERMISSIBLE DEFECTS.

Permissible defects are those which are acceptable without repair or rework and with which pressurized flight of the airplane is considered safe.

REPARABLE DEFECTS.

Reparable defects are those which can be repaired and which are approved for repair. Defects or damages listed as reparable are considered to be those which, when repaired, would not make the canopy unsafe for flight.

Note

Defects or damage in excess of, and not included in, those defined are cause for rejection.

NICKS.

Nicks are broken indentations or cavities having sharp edges.

DENTS.

Dents are depressions or hollows left by blows or concentrated pressures.

SPALLS.

Spalls are cavities left by the loss of a fragment having at least one feathered edge.

SCRATCHES.

Scratches are tears in the surface made by a pointed object, a shallow mark, groove, furrow, or channel. Those scratches which extend to a depth of 0.001 inch or less are called hairline scratches.

CRACKS.

Cracks are narrow fractures extending deeper than scratches into the sheet and having a pointed bottom and a high depth-to-width ratio. Cracks are, in most instances, the result of unbalanced internal stresses within the plastic surfaces and are usually at right angles to the direction of stress.

CRAZING.

General crazing is a pattern of area of tiny fissures or splits (material separation) in the surface of the plastic. It resembles the cracking of glaze or pottery or surface splits on weathered wood. Crazing which increases in area and/or severity is termed propagating crazing. Crazing in which the area and type remain constant is termed nonpropagating.

Stress crazing is a result of concentrated loads (stresses) and may be caused by improperly installed canopy bubbles. The interior stresses may cause localized crazing in any area.

The effects and degree of crazing can be determined by the reduction of transparency in the plastic. Where severe crazing occurs, the reduction in transparency is visible when examined from any angle, but where slight crazing occurs, the reduction in transparency may be discovered from only a viewing angle. The edges of the crazing fissures cause undesirable highlights and/or reflections dur-

ing night flying. Crazing is a warning that the material has been weakened physically by solvent and/or concentrated or locked-in stresses which have been partially or completely relieved. No satisfactory method for removing crazing is known.

CRITICAL VISION AREA.

The critical vision area is all the vision area except the aft 16 inches of the canopy glass and a ¼-nch border around the periphery of the reinforcing strips.

NONCRITICAL VISION AREA.

The noncritical vision area includes all areas not designated as the critical vision area.

INSPECTION AND REPAIR CRITERIA.

MAXIMUM LIMITS FOR NOTED DEFECTS.

The following are the maximum limits for noted defects:

- Cracks are not permissible in any part of the canopy glass.
- Crazing is not permissible in the critical vision area of the canopy glass.
- Discoloration is permissible up to one inch from the edge of the reinforcing strips.
- Dents and spalls in area of support blocks for sun shields and radio antenna will be acceptable up to 1/16 inch deep.

REPARABLE DEFECTS IN CRITICAL VISION AREA.

Scratches, nicks, dents, and spalls are reparable, provided the depth does not exceed 0.031 (1/32) inch and provided they can be removed without distorting or impairing the view.

Note

Waxing is required to minimize vision impairment.

PERMISSIBLE DEFECTS IN CRITICAL VISION AREA.

The following defects are permissible, provided they do not exceed noted limits.

- Nicks, dents, and spalls
  Length—0.125 inch maximum.
  Width—0.060 inch maximum.
  Depth—0.010 inch maximum.
  Density—Flaws that do not impair vision but cause distraction must be repaired.
- Scratches
  Length—3.000 inches maximum.
  Width—0.010 inch maximum.
  Depth—0.001 inch maximum.
  Density—Scratches that do not impair vision but cause distraction must be repaired.

REPARABLE DEFECTS IN NONCRITICAL VISION AREA.

The following are the maximum limits for noted defects:

- Nicks, dents, and spalls are reparable provided the depth does not exceed 0.063 inch. A slight distortion is permissible after repair.
- Scratches
  Length—12.000 inches maximum.
  Width—0.020 inch maximum.
  Depth—0.031 inch maximum.

Note

A slight distortion is permissible after repair.

PERMISSIBLE DEFECTS IN NONCRITICAL VISION AREA.

The following are the maximum limits for noted defects:

- Nicks, dents, and spalls are permissible, provided the depth does not exceed 0.031 inch.
- Scratches
  Length—12.000 inches maximum.
  Width—0.020 inch maximum.
  Depth—0.010 inch maximum.

Note

The total length of all the scratches must not exceed the longest dimension of the noncritical area containing the scratches.

- Crazing
  Severe crazing—5 percent of total area is permissible if it is not concentrated in any one area.

  Slight crazing—5 percent of total area is permissible.

FUSELAGE REPAIRS.

FUSELAGE REPAIR INDEX.

See figure 4-3C.

HORIZONTAL STABILIZER FORWARD ATTACHING TUBE REPAIR (PERMANENT — CATEGORY A).

Check diameter of bolt hole in forward end of attach-

ing tubes 192-31442-( ). Bolt holes found to be beyond the allowable tolerance of 1.2510 inches must be repaired as instructed in the paragraphs following.

SPECIAL TOOLING REQUIRED.

The following kit can be obtained on loan, when needed, from SMNBT, McClellan AFB, Calif.

64C39445 kit—bushing installation, horizontal stabilizer tube assembly (used to install 64B39443 bushings), consisting of:

64SCB-6055-1 reamers (two)
64SCB-6055-2 reamers (two)
64SCB-6081 reamers (two)
64SCC-6064-2 bushing installation tool
64SCC-6060 tool, consisting of:
64SCB-6059 guide assembly
64SCB-6014 arbor
61SCA-6096 feed nut

REPAIR PROCEDURE.

Ream and bush holes in accordance with the following instructions:

1. Remove hydraulic lines, linkage, brackets, etc., as necessary to facilitate access to area around forward attaching point of tubes.

2. Install guide 64SCB-6059 in either side of tube assembly bolt hole.

3. Secure reamer 64SCB-6055-1 or 64SCB-6055-2 on arbor 64SCB-6014.

4. Place arbor assembly on opposite side of tube assembly bolt hole from guide.

5. Install nut 61SCA-6096 on arbor assembly.

6. Attach right-angle power drive to drive end of reaming tool assembly. (Use a cutting oil while reaming.)

Note

Use a 90-degree angle drive on power unit and maintain a reaming speed of 400 to 600 rpm.

7. Carefully control cutting rate by adjusting feed nut, and control of power drive's speed.

8. Install locally manufactured bushing, 64B39443-5 or 64B39443-3 as required, using bushing installation tool 64SCC-6064-2. (Coat inside reamed hole with zinc chromate; install bushing while chromate is wet.)

9. Reverse tooling and repeat steps 2 through 8 to repair opposite tube assembly.

10. Reassemble tube installations. If the bushings do not readily accept a serviceable bolt, the installed bushing must be reamed, utilizing reamer 64SCB-6081. (Use light machine oil while reaming.)

11. Reinstall those components removed to facilitate access to the area. Retorque and safety as required.

REPAIR OF ENGINE BUMPER SEAL ASSEMBLY.

See figure 4-16B.

REPAIR AND REPLACEMENT OF DRAG CHUTE RETAINER TABS.

See figure 4-28D.

(Deleted)

REPAIR OF CORRODED CANOPY GUIDE RAIL CASTINGS—UPPER.

See figure 4-14C.

REPAIR OF CORRODED CANOPY GUIDE RAIL CASTINGS—LOWER.

See figure 4-14D.

ASPIRATOR BOTTOM SKIN REPAIR.

The repair in figure 4-28B shows a satisfactory method of repairing the bottom skin of an aspirator that has been damaged by a nose-high landing. A hydraulic jack and wooden blocks may be used when it is not possible to restore the original shape with a hammer and dolly bar. When the hydraulic jack is used, some of the strain may be removed by tapping lightly with a hammer while the pressure is still being applied. After the aspirator has been returned to its original shape, a careful inspection should be made to determine the extent of the damage. If the inner skin is cracked or buckled or the stiffening beads have collapsed, a doubler or a series of doublers must be formed to match the original part and to restore the original strength. If the outer skin is cracked or damaged, a nonflush repair may be made by applying a new skin directly over the damaged skin. All cracks must be stop-drilled, but the damaged area need not be removed. In cases where the spot welds have become cracked or separated, NAS508M4 rivets may be used in place of the spot welds.

CONSTANT THICKNESS SKIN REPAIRS.

The eight skin repairs shown in figures 4-31, 4-32, 4-33 are designed to be used on any section of the fuselage where a repair on a non-milled skin is needed. These repairs are not restricted to any size. They may be as large as it is practical to make. Although when the largest part of a skin is damaged it would probably be more practical to replace the complete skin. When replacing a skin or making a partial skin repair, always replace the original fasteners with fasteners of the same

size and type as the ones removed when possible. Blind fasteners should be used only when it is impossible to replace the original type of fastener.

The fastener requirements for the repair splice are relatively simple to determine. If the same size and type of fastener are kept and the same spacing is used, the number required is twice the number of fasteners used in the nearest manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the repair splice requires four rows of fasteners at one inch spacing. For longitudinal repairs, use the nearest longitudinal manufactured splice a (longitudinal splice is approximately parallel with the direction of flight of the airplane). For transverse repairs, use the nearest transverse manufactured splice (a transverse splice crosses or runs into a longitudinal splice at approximately right angles). When the doubler is installed behind the skin and the required number of fasteners is picked up in the doubler, those fasteners which fall outside of the normal fastener pattern must be picked up by enlarging the doubler in that area only, as shown in figures 4-31, 4-32, and 4-33.

REPAIR OF TITANIUM STRUCTURES.

Because of the difficulty in forming titanium, repairs of titanium in the field can be made with ½ hard corrosion-resistant steel. Fillers that are difficult to form can be made from annealed corrosion-resistant steel. Small titanium parts that are damaged beyond repair, such as small stiffening angles, clips, and small brackets may also be made from ½ hard corrosion-resistant steel. Damage length in frame cap members after trimming must not exceed 3 inches when repair is made from corrosion-resistant steel. When only one frame cap is damaged, the total length of steel repair parts must not exceed 10 inches. This is because of the difference rates of expansion of titanium and ½ hard corrosion-resistant steel. If this condition were allowed to exist, the fasteners could be sheared off by the unequal movements of the materials. When both caps are damaged, there are no restrictions to the length of the repair parts when both caps are repaired with ½ hard corrosion-resistant steel. If the damaged length is less than 9/16 inch, the fillers must match the original part.

Because the area being repaired must be made stronger to carry additional stress loads, the Monel rivets that are removed should be replaced with oversize AN bolts, AN screws, Hi-Shear rivets, or Jo-bolts. In some cases (where edge distance and radius clearance permit), the next size larger fastener may be used.

REPLACEMENT OF TITANIUM SKINS.

If a titanium skin on the aft fuselage is damaged enough to require replacement and titanium replacement skins are not available, the accepted repair is to replace the damaged skin with one of equal gage of ½ hard corrosion-resistant steel. When side skins on the aft fuselage are replaced, there are no changes in the fastener requirements with the exception of fuselage station 466¼. Rivets through the side skin splice at fuselage station 466¼ should be changed to HS 47-6 with HS 32-6 collars. On some F-100D and F-100F Airplanes the skin splice at fuselage station 448½ extends to the lower longeron. The skin at this station is the only lower skin that requires a change in the fasteners. If the airplane being repaired has this type of skin splice at station 448½ and the damaged area includes this station, the fasteners should be changed to HS 47-6 with HS 32-6 collars.

USE OF TYPICAL REPAIRS.

In addition to the repairs found in this section of the book, repairs from Section X may be used. These repairs are designed to restore the material to its original strength and may be used whenever they meet requirements as to shape and material for the damaged part.

HOLDING FIXTURE DIMENSIONS.

Figure 4-38 shows the necessary dimensions for the construction of fixtures to hold the fuselage components while repairs are being made. These dimension and attach points are not intended for use in manufacturing jigs for the building of a complete component. They are intended only for manufacturing holding fixtures to maintain the proper attach points, and to ensure that the part being repaired will fit when installed.

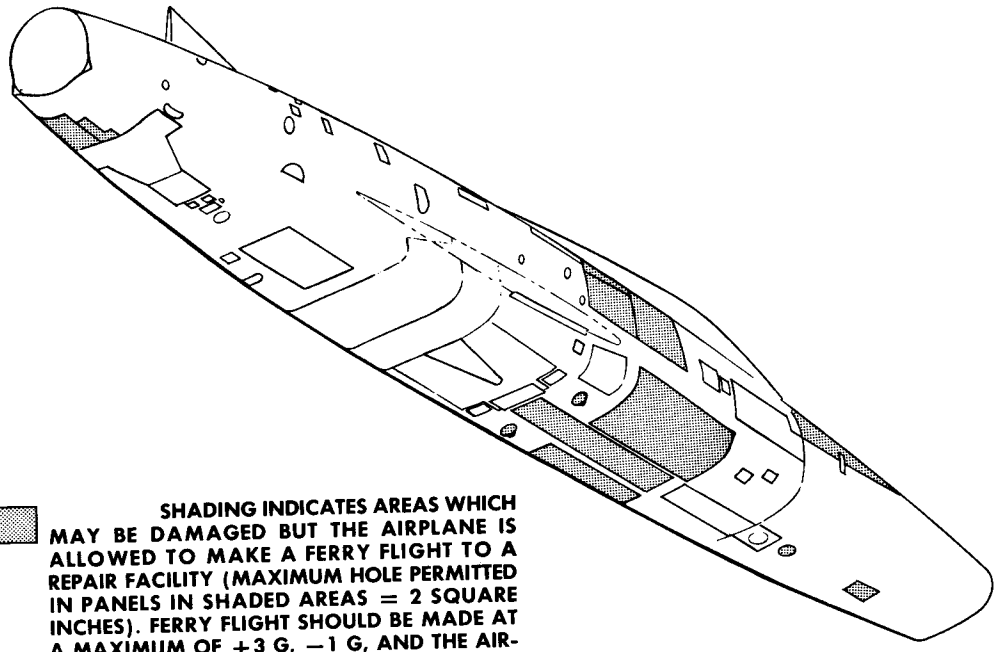
FLIGHT CONTROL ROD ASSEMBLY REPAIR

Loose swaging at the rod end on the following rod assemblies shall be repaired as shown in figure 4-37A.

180-52353-5	192-522128-11
180-52353-9	192-522159-3
180-52383-3	192-522196-3
180-52387-3	192-522514-9
180-52387-7	192-522514-11
180-52395-3	192-52380-3
192-522128-5	223-52226-11
192-522128-9	

ONE-TIME FLIGHT

FUSELAGE



SHADING INDICATES AREAS WHICH MAY BE DAMAGED BUT THE AIRPLANE IS ALLOWED TO MAKE A FERRY FLIGHT TO A REPAIR FACILITY (MAXIMUM HOLE PERMITTED IN PANELS IN SHADED AREAS = 2 SQUARE INCHES). FERRY FLIGHT SHOULD BE MADE AT A MAXIMUM OF +3 G, −1 G, AND THE AIRSPEED IS NOT TO EXCEED 300 KNOTS.

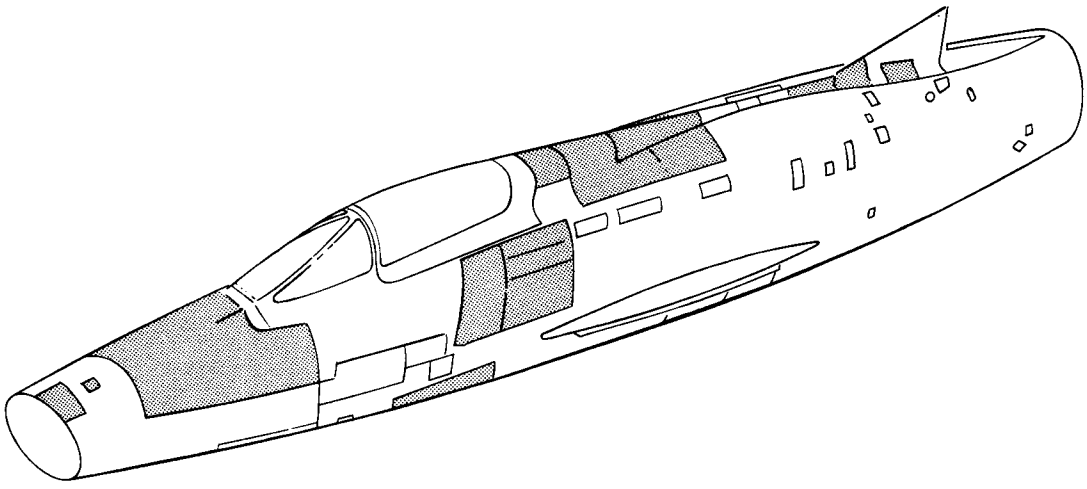


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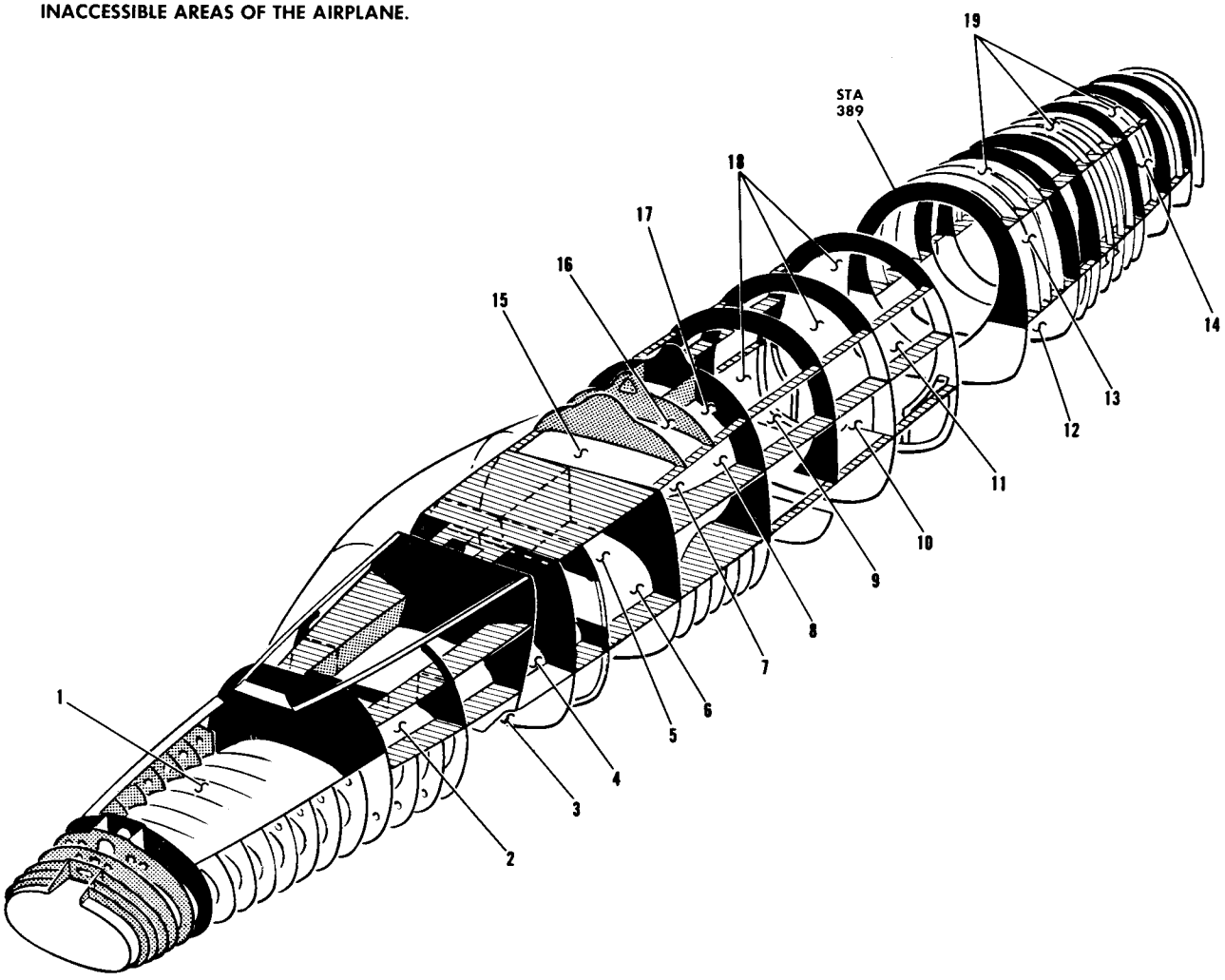
Figure 4-1. One-time Flight

REPAIR ACCESS

FUSELAGE

THIS ILLUSTRATION, USED IN CONJUNCTION WITH FUSELAGE ACCESS PROVISION ILLUSTRATIONS, SHOWS ACCESSIBLE AND INACCESSIBLE AREAS OF THE AIRPLANE.



Access to the numbered areas may be gained as follows:

AREA	POINTS OF ACCESS
1	ELECTRICAL AND RADAR ACCESS HOOD
2	SURFACE CONTROLS, HYDRAULIC BRAKE VALVE ACCESS
3	GUN ACCESS DOOR
4	AMMUNITION ACCESS DOOR
5	BATTERY AND ELECTRICAL ACCESS
6	GENERAL ACCESS DOOR
7	FUEL FITTING ACCESS
8	FLIGHT CONTROL SYSTEM ACCESS
9	AIR TURBINE DOOR
10	FUEL PROBE ACCESS
11	COMPRESSOR BLEED DOOR, FLIGHT CONTROL SYSTEM RETURN FILTER ACCESS
12	AFT FUEL CELL BAY (AFT SECTION AND FUEL CELL REMOVED)
13	DISCONNECT ACCESS DOORS
14	DRAG CHUTE CONTROL BOX ACCESS
15	GENERAL ACCESS DOOR
16	UPPER EQUIPMENT ACCESS
17	HYDRAULIC ACCESS
18	THROUGH ENGINE COMPARTMENT
19	THROUGH ENGINE COMPARTMENT

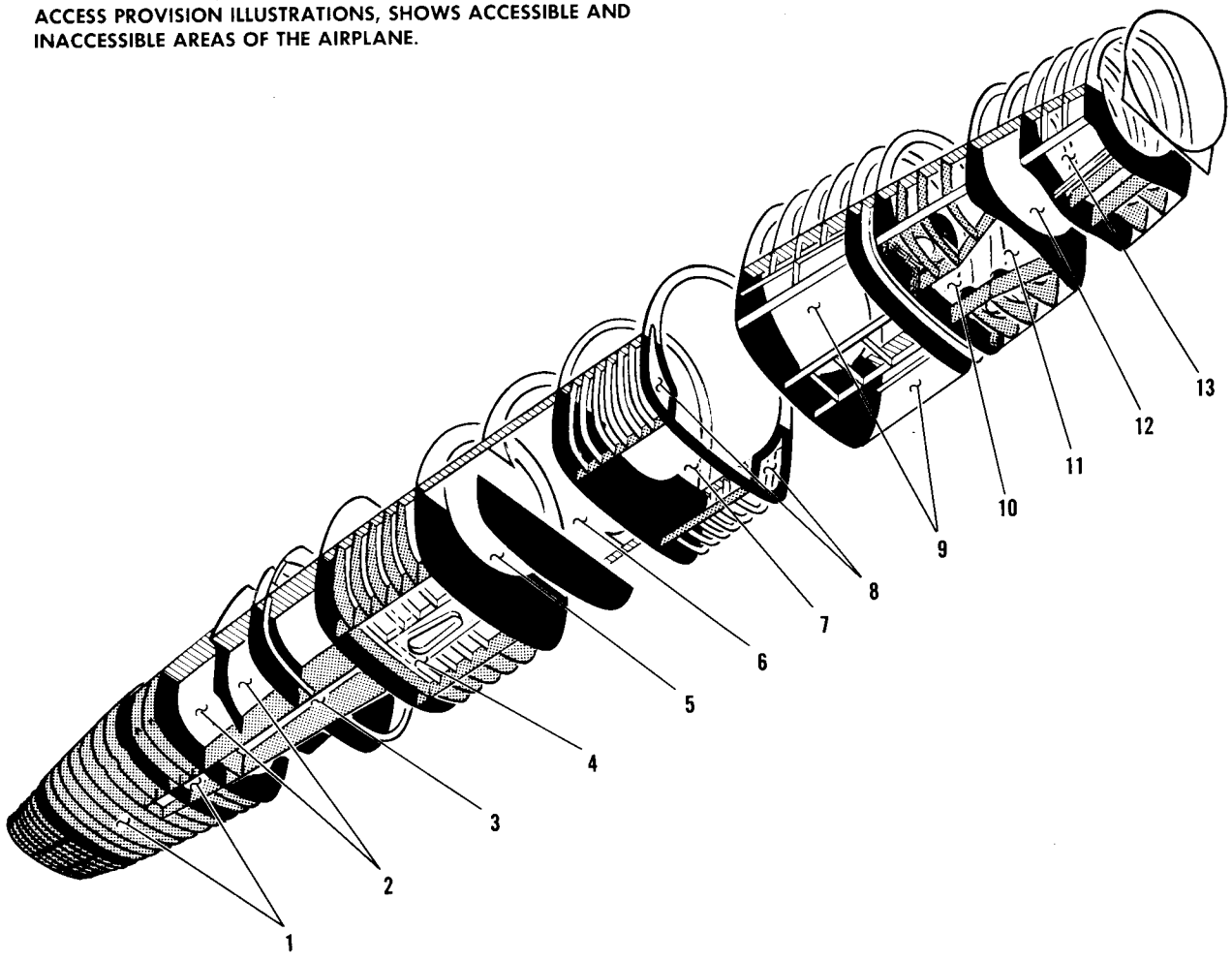
F-100D-3-31-141

Figure 4-1A. Repair Accessibility (Sheet 1 of 2)

REPAIR ACCESS

FUSELAGE

THIS ILLUSTRATION, USED IN CONJUNCTION WITH FUSELAGE ACCESS PROVISION ILLUSTRATIONS, SHOWS ACCESSIBLE AND INACCESSIBLE AREAS OF THE AIRPLANE.



Access to the numbered areas may be gained as follows:

AREA	POINTS OF ACCESS
1	ANTENNA ACCESS, DATA CASE ACCESS
2	GUN ACCESS
3	NOSE WHEEL WELL
4	SPEED BRAKE
5	BOOST PUMP ACCESS, FUEL PROBE ACCESS, HYDRAULIC ACCESS
6	MAIN GEAR DOORS
7	GENERAL ACCESS

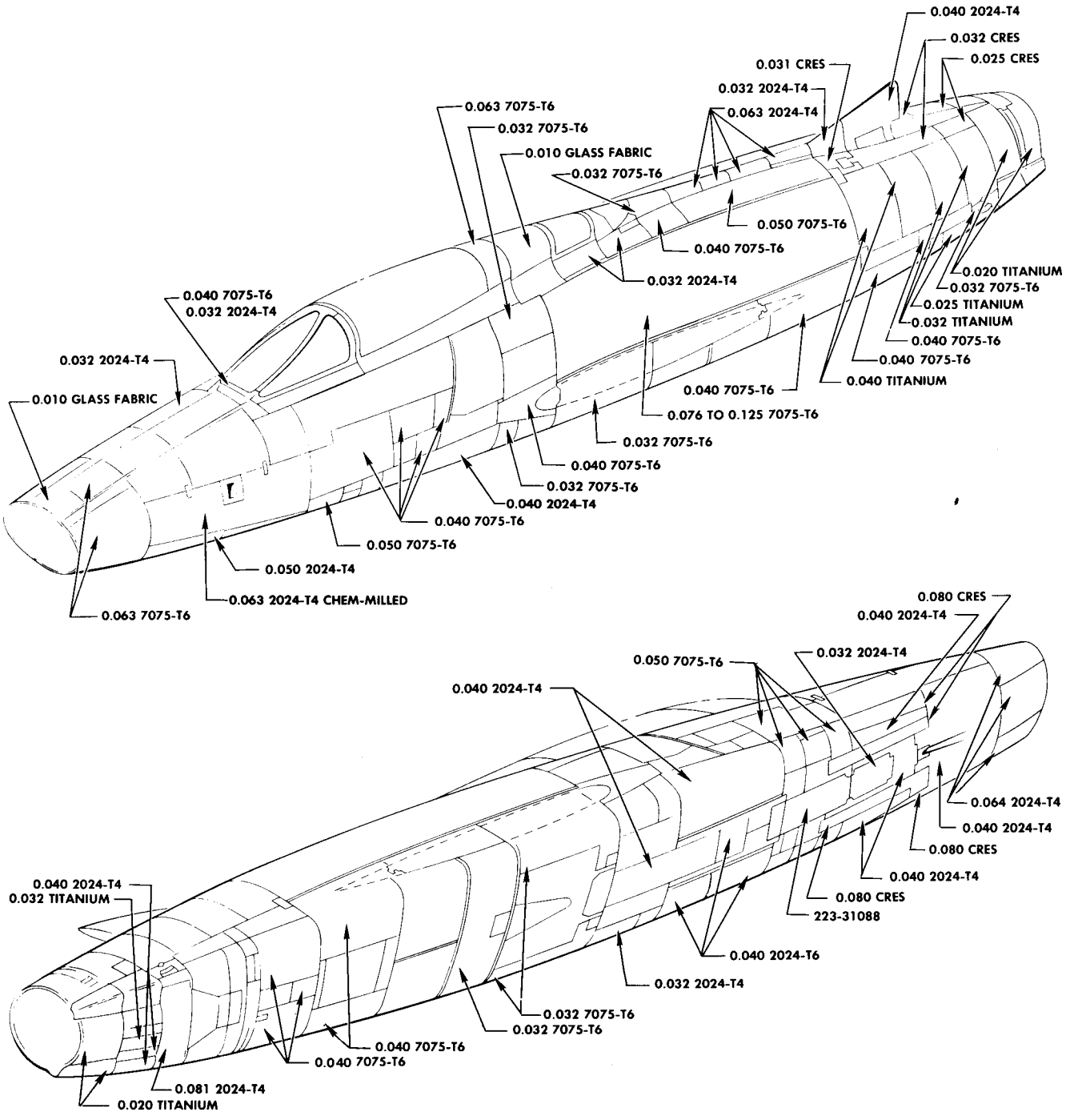
AREA	POINTS OF ACCESS
8	FUEL CELL BAYS (FUEL CELLS REMOVED)
9	AFT FUEL CELL BAY (FUEL CELL REMOVED)
10	HORIZONTAL STABILIZER CONTROL ACCESS
11	INTERMEDIATE CONTROL WELL ACCESS
12	GENERAL ACCESS
13	DRAG CHUTE DOOR

F-100D-3-31-142

Figure 4-1A. Repair Accessibility (Sheet 2 of 2)

SKIN DIAGRAM—F-100D AIRPLANES

FUSELAGE



F-100D-3-31-60C

Figure 4-2. Fuselage Skin Diagram—F-100D Airplanes

SKIN DIAGRAM—F-100F AIRPLANES

FUSELAGE

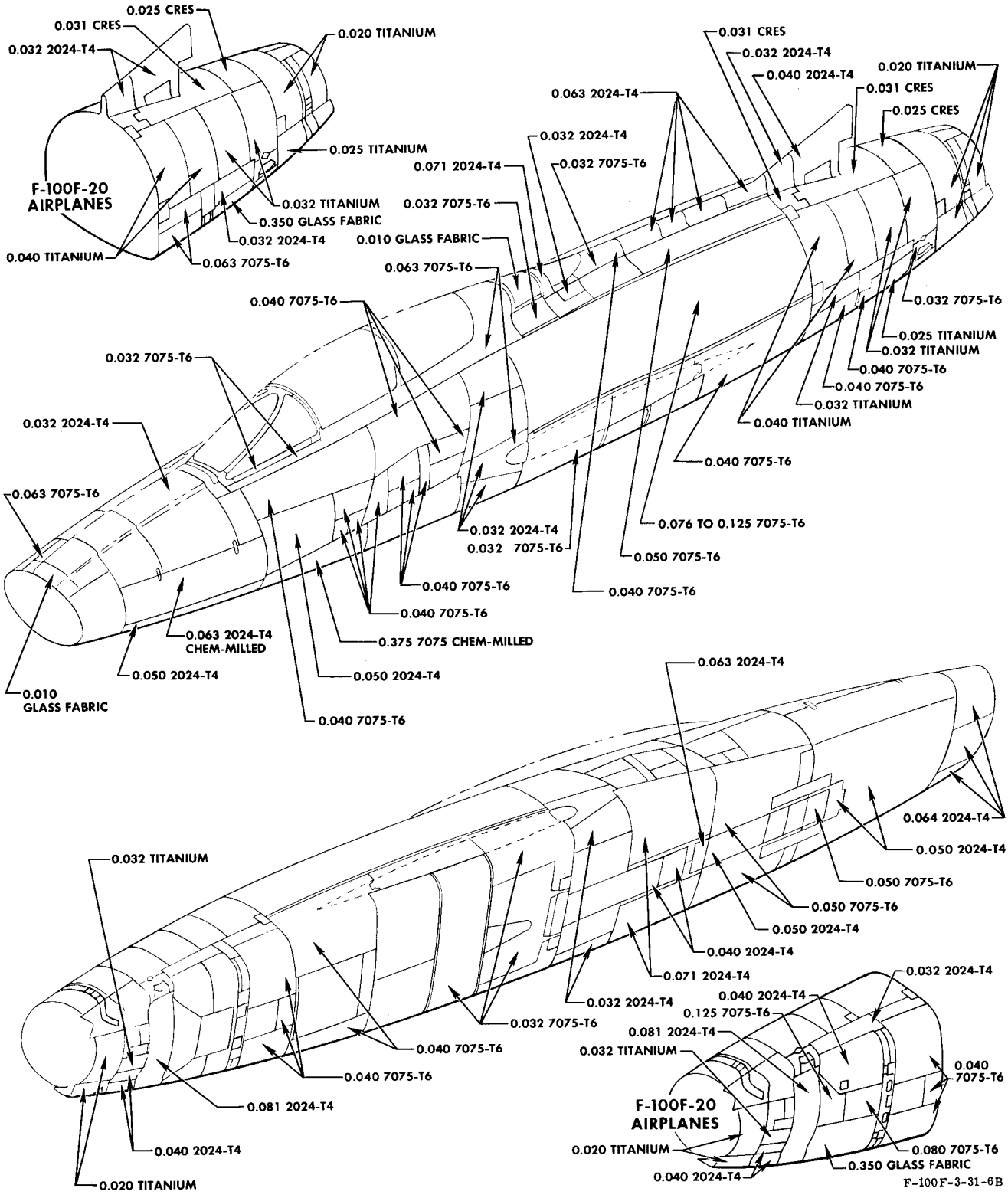


Figure 4-2A. Fuselage Skin Diagram—F-100F Airplanes

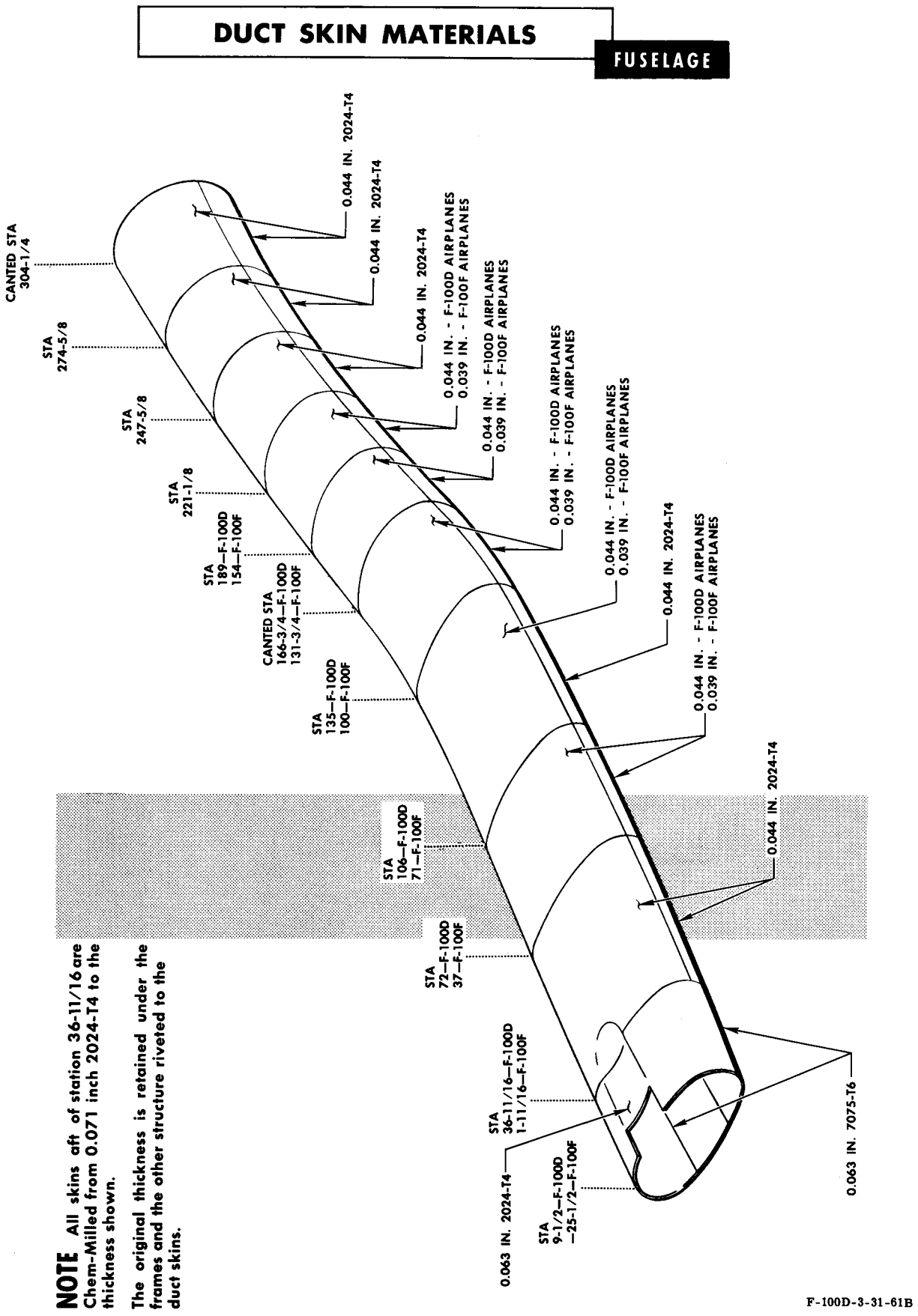


Figure 4-3. Duct Skin Diagram—F-100D and F-100F Airplanes

SKIN GAP TOLERANCES F-100D

FUSELAGE

NOTE All skin gaps and door-to-skin gaps are a minimum of 0.015 inch to a maximum of 0.063 inch, unless noted.

- Longitudinal skin splices in the air intake duct are 1/8 (+1/8, −1/8) inch.

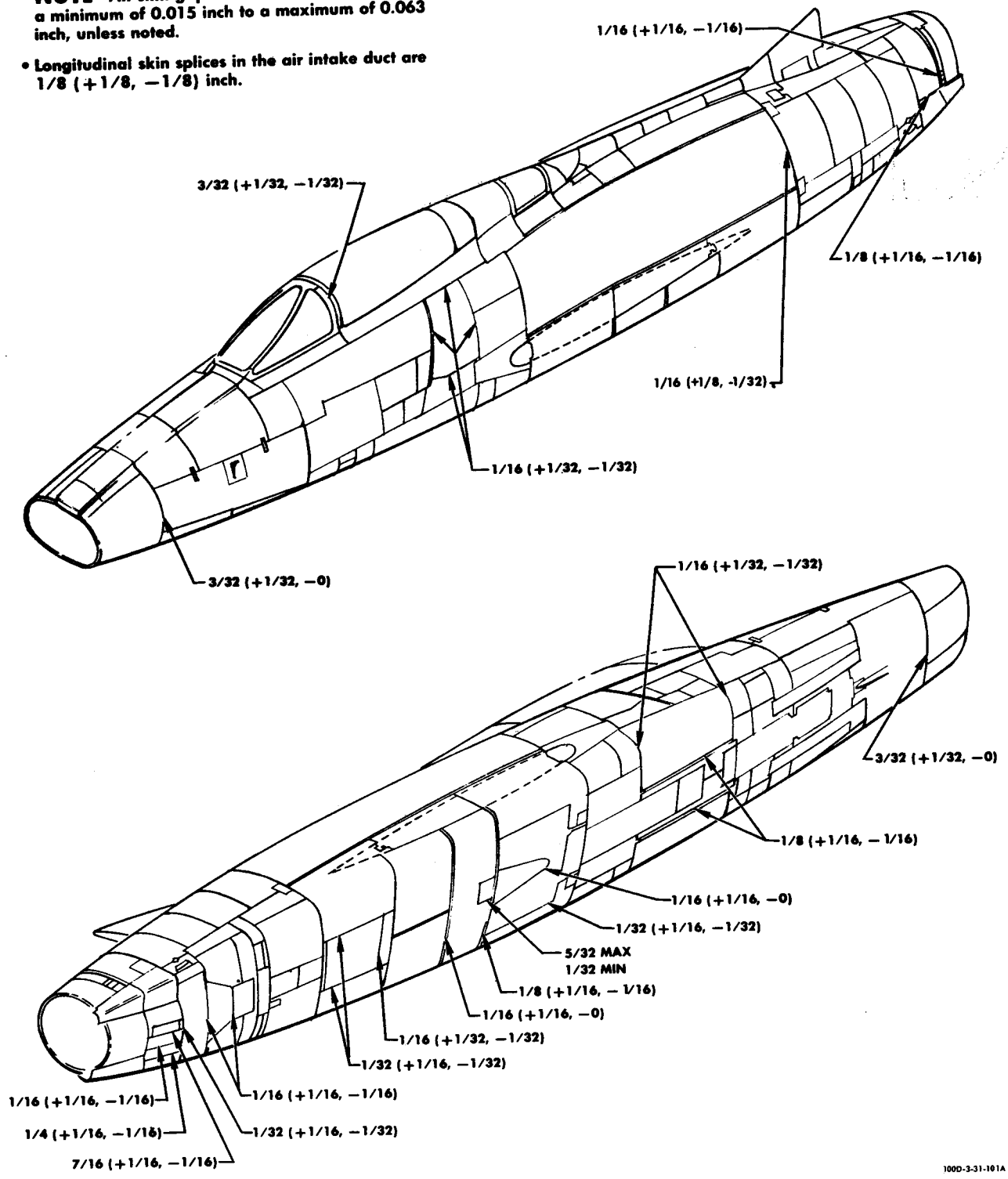


Figure 4-3A. Skin Gap Tolerances—F-100D Airplanes

SKIN GAP TOLERANCES F-100F

FUSELAGE

NOTE All skin gaps and door-to-skin gaps are a minimum of 0.015 inch to a maximum of 0.063 inch, unless noted.

- Longitudinal skin splices in the air intake duct are 1/8 (+1/8, −1/8) inch.

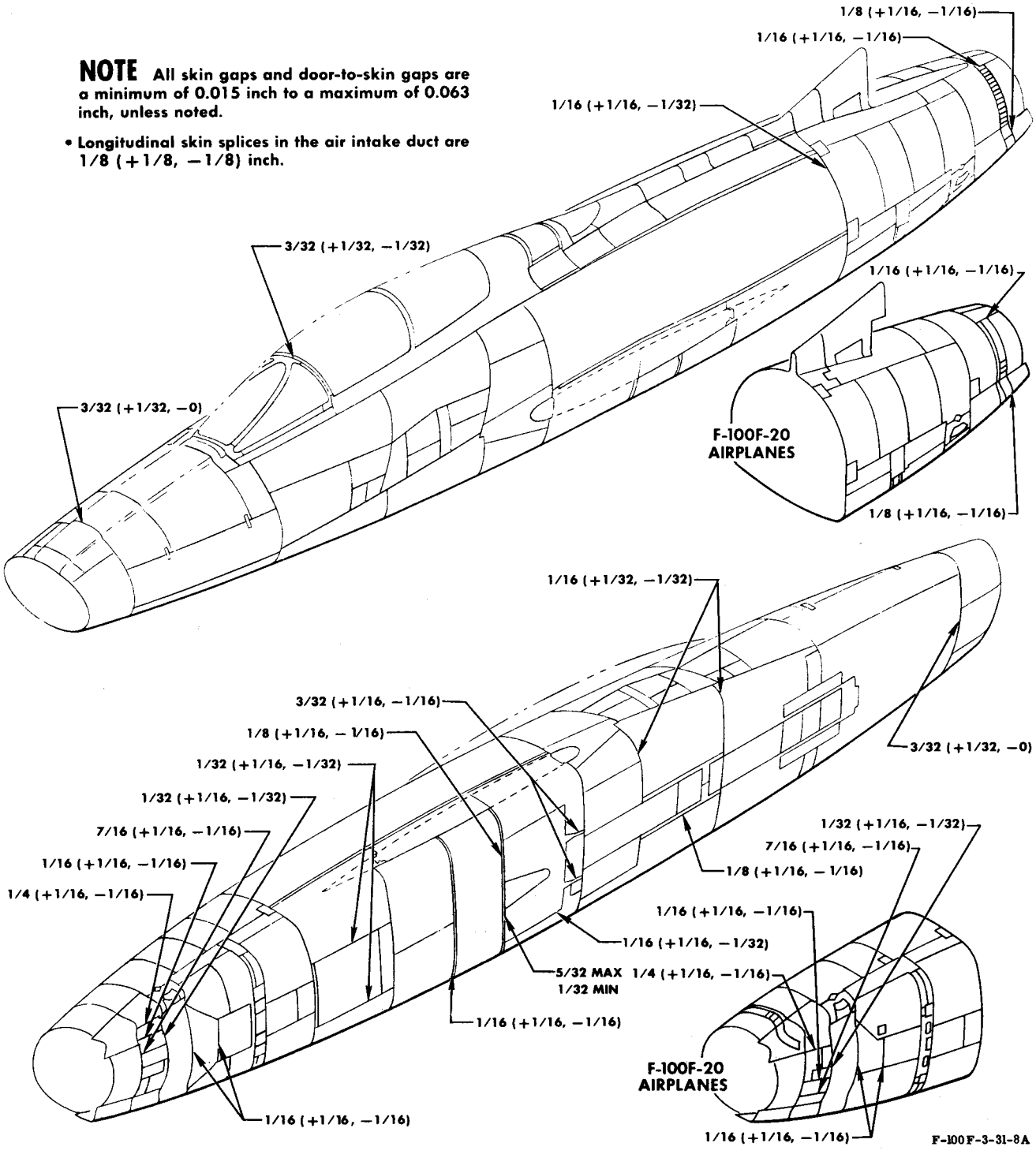


Figure 4-3B. Skin Gap Tolerances—F-100F Airplanes

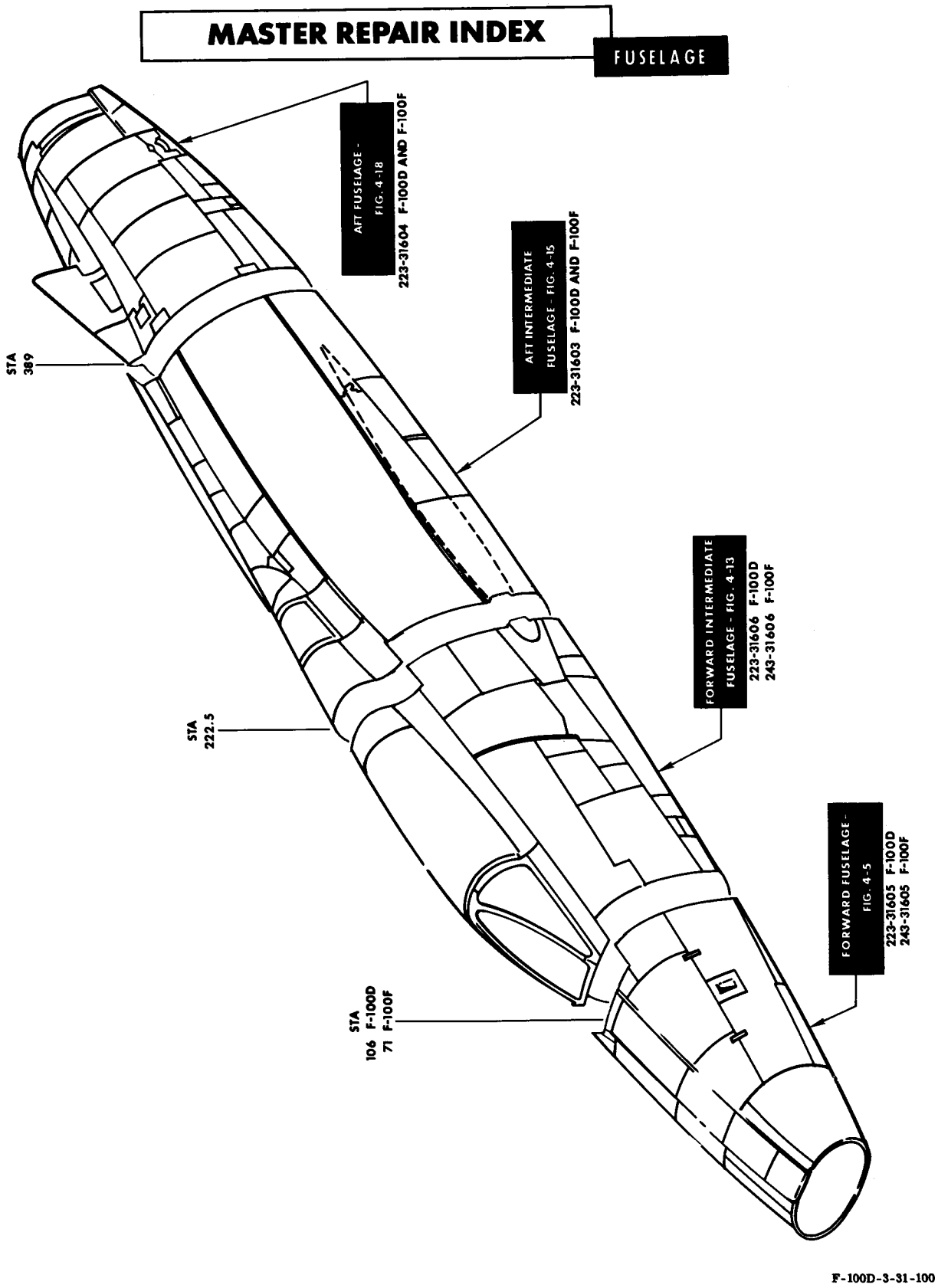


Figure 4-3C. Fuselage Repair Index

NEGLIGIBLE DAMAGE

FUSELAGE

F-100D AIRPLANES

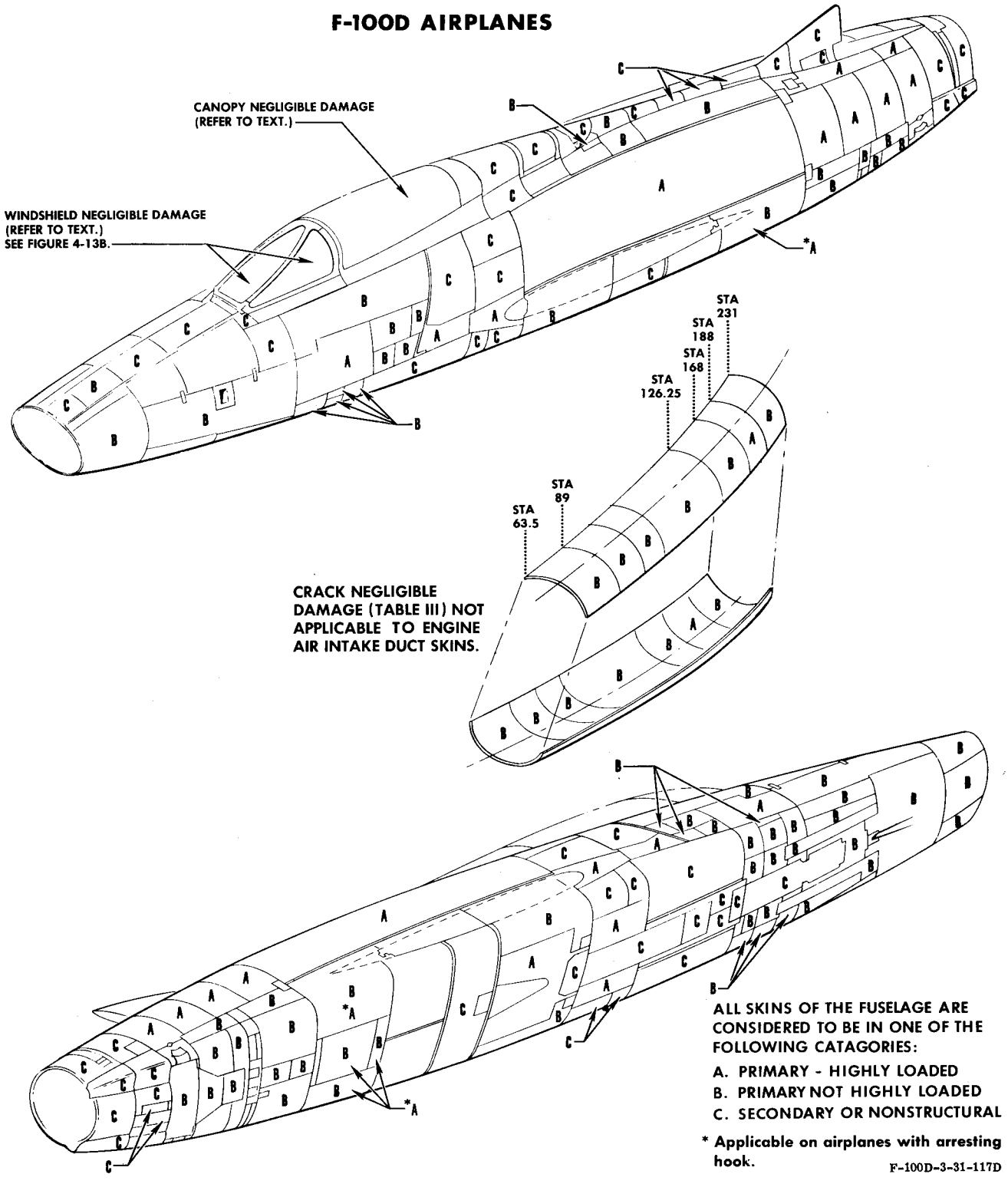


Figure 4-4. Fuselage Negligible Damage (Sheet 1 of 4)

NEGLIGIBLE DAMAGE

FUSELAGE

F-100F AIRPLANES

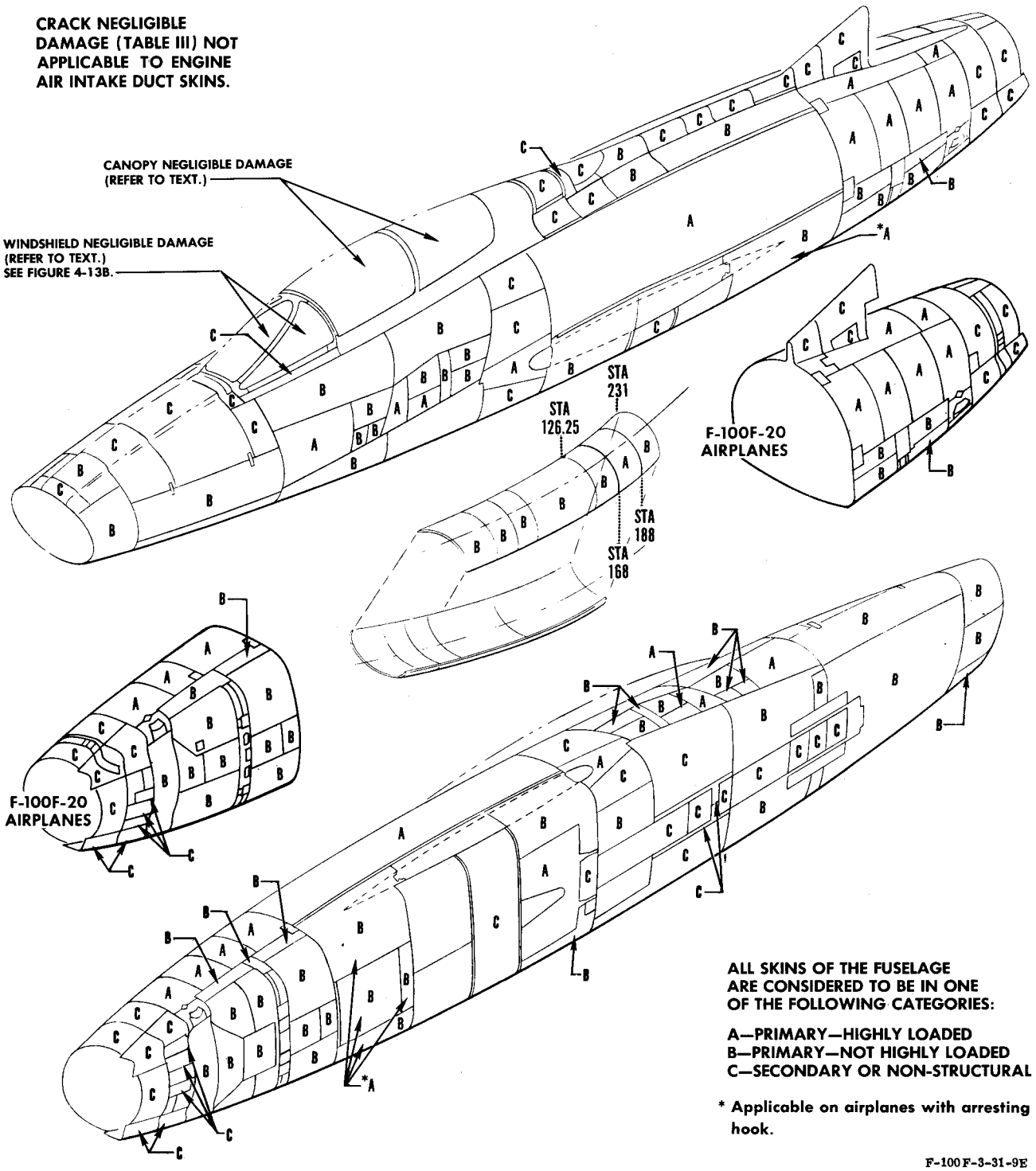


Figure 4-4. Fuselage Negligible Damage (Sheet 2 of 4)

NEGLIGIBLE DAMAGE

FUSELAGE

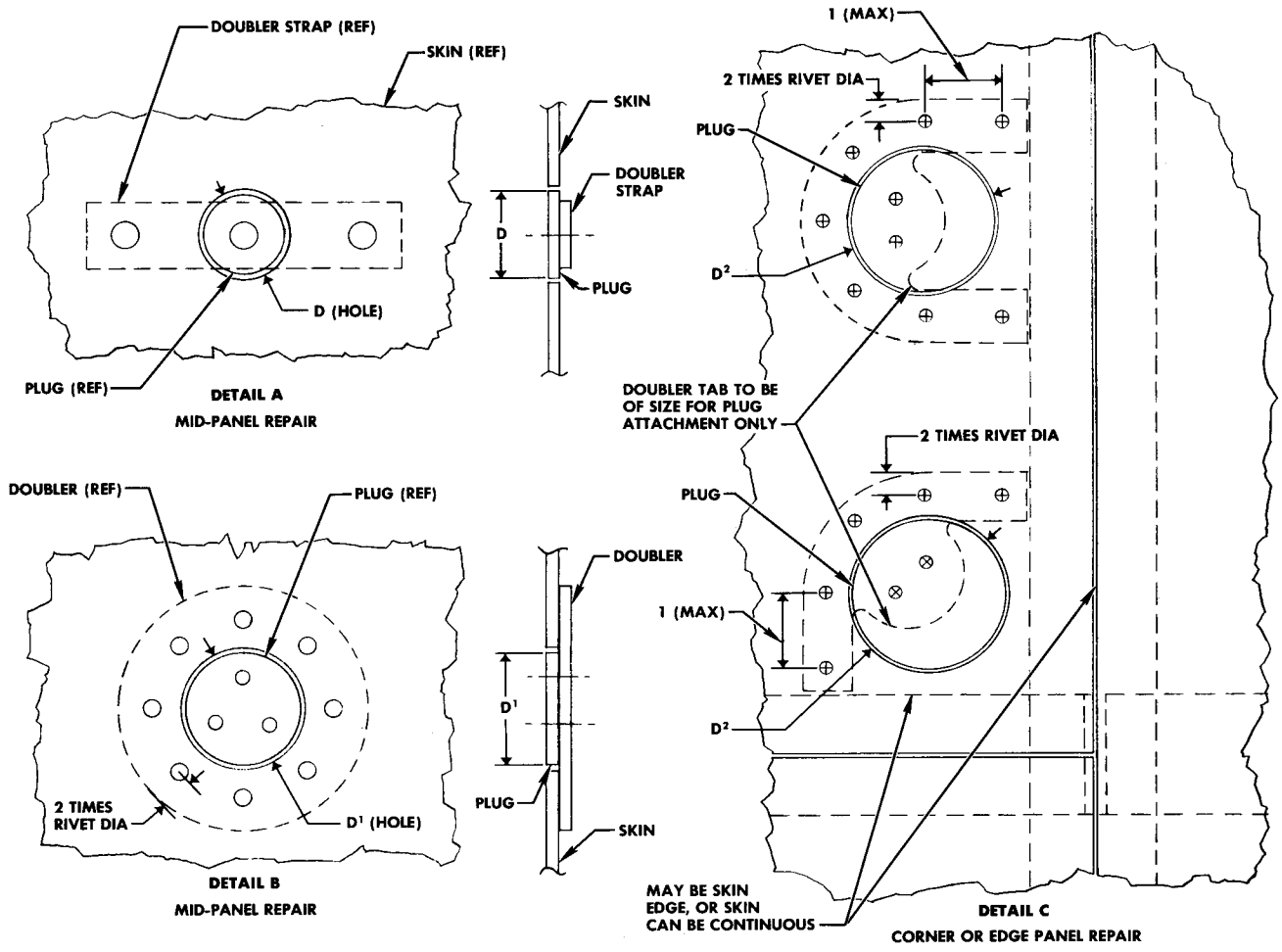


TABLE I

	CATEGORY A	CATEGORY B	CATEGORY C
D	1/2	1	1-1/2
D^1	3/4	1-1/2	4
D^2	3/4	1-1/2	4
DETAIL A	ONE PER PANEL	ONE PER PANEL	TWO PER PANEL
DETAIL B	ONE PER PANEL	TWO PER PANEL	THREE PER PANEL
DETAIL C	ONE PER PANEL	TWO PER PANEL	TWO PER PANEL

NOTE The holes can be rectangular if the maximum dimension does not exceed those noted in Table I. All corners must have a radius of at least 1/4 inch.

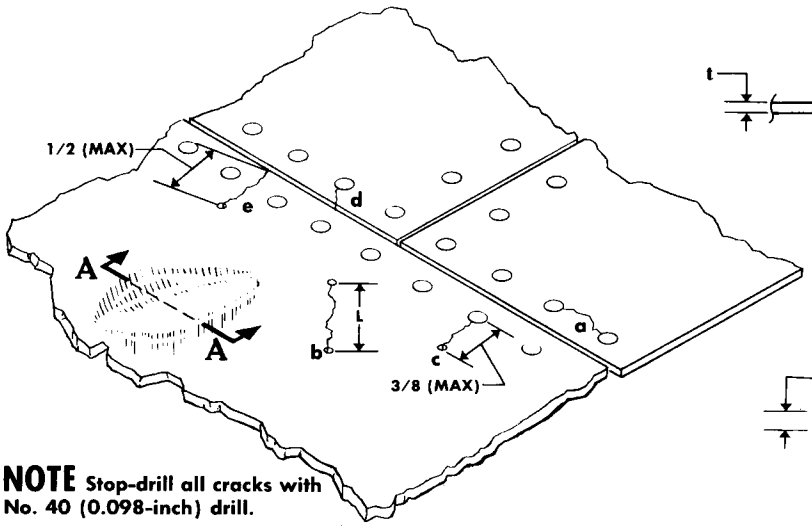
- Make doublers and plug from same gage and material as damaged skin.
- Use AD3 or M3 rivets for skins of 0.032-inch gage.
  Use AD4 or M4 rivets for skins of 0.040-inch gage.
  Use AD5 or M5 rivets for skins of 0.050-inch gage and above.
  Use AD rivets for aluminum skin.
  Use monel rivets for titanium and steel skins.
  All rivets must be countersunk in skins.
- Cherry rivets of the same diameter may be substituted for aluminum or monel rivets.
- A spacing of two times the hole diameter must exist between the edges of the holes when two or more holes are permitted in a panel.

F-100A-3-31-130

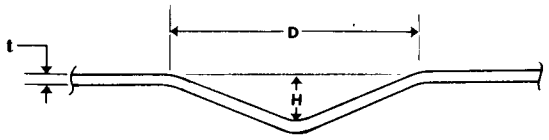
Figure 4-4. Fuselage Negligible Damage (Sheet 3 of 4)

NEGLIGIBLE DAMAGE

FUSELAGE



NOTE Stop-drill all cracks with No. 40 (0.098-inch) drill.



SECTION A - A

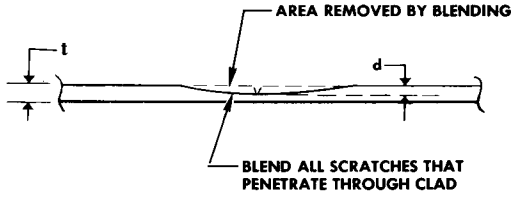


TABLE III CRACKS

	CATEGORY A	CATEGORY B*	CATEGORY C*
l	3/8	3/4	1-1/2
a	ONE PER 20 RIVETS	ONE PER 10 RIVETS	2 PER 10 RIVETS
b	ONE PER PANEL	2 PER PANEL	3 PER PANEL
c	ONE PER PANEL	2 PER PANEL†	3 PER PANEL†
d	ONE PER 10 RIVETS	ONE PER 5 RIVETS	ONE PER 4 RIVETS
e	ONE PER 10 RIVETS	ONE PER 5 RIVETS	ONE PER 4 RIVETS

* Cracks exceeding lengths permitted above, but within "D" values of table I may be repaired as holes.

† 8 good rivets between damage.

A combination of only two of the above items may exist at the same time per panel.

TABLE II SCRATCHES

CATEGORY A		CATEGORY C		CATEGORY B	
d IN PER-CENT OF t	LENGTH (IN.)	d IN PER-CENT OF t	LENGTH (IN.)	d IN PER-CENT OF t	LENGTH (IN.)
3	UNLIMITED	6	UNLIMITED	10	UNLIMITED
25	1-1/2	25	3	25	6
35	1-1/8	35	2-1/4	35	4-1/2
50	3/4	50	1-1/2	50	3

NOTE Any scratch penetrating deeper than 50 percent of the material thickness must be treated as a crack.

• Any bare aluminum alloy exposed by a scratch penetrating either the clad or the painted surface should be painted.

TABLE IV DENTS

D	H=3-1/2 t	H=2-1/2 t	H=1-1/2 t	H= t	H=3/4 t	H=3/8 t	H=5/8 t
1/2 TO 1	.....	0.025	0.032—0.040	0.050	0.063—0.071	0.080—0.090	0.100—0.125
1 TO 1-1/2	.....	0.025—0.032	0.040—0.050	0.063—0.071	0.080—0.100	0.125	
1-1/2 TO 2	.....	0.025—0.040	0.050—0.063	0.071—0.090	0.100—0.125		
2 TO 3	0.025—0.040	0.050—0.063	0.071—0.090	0.100—0.125			
3 TO 4	0.025—0.063	0.071—0.080	0.090—0.125				

NOTE The above table gives the maximum depth (H) of the dent allowed. The maximum values of "D" are as follows: Category A, 1-1/2 inch; category B, 2-1/2 inches; category C, 4 inches. If dents are within restrictions noted in table IV, no repair is required. For titanium, use one gage less than maximum shown in tables (or 0.008 inch less). For steel, use two gages less than maximum shown in tables (or 0.016 inch less).

F-100A-3-31-131

Figure 4-4. Fuselage Negligible Damage (Sheet 4 of 4) (Figures 4-4A, 4-4B deleted)

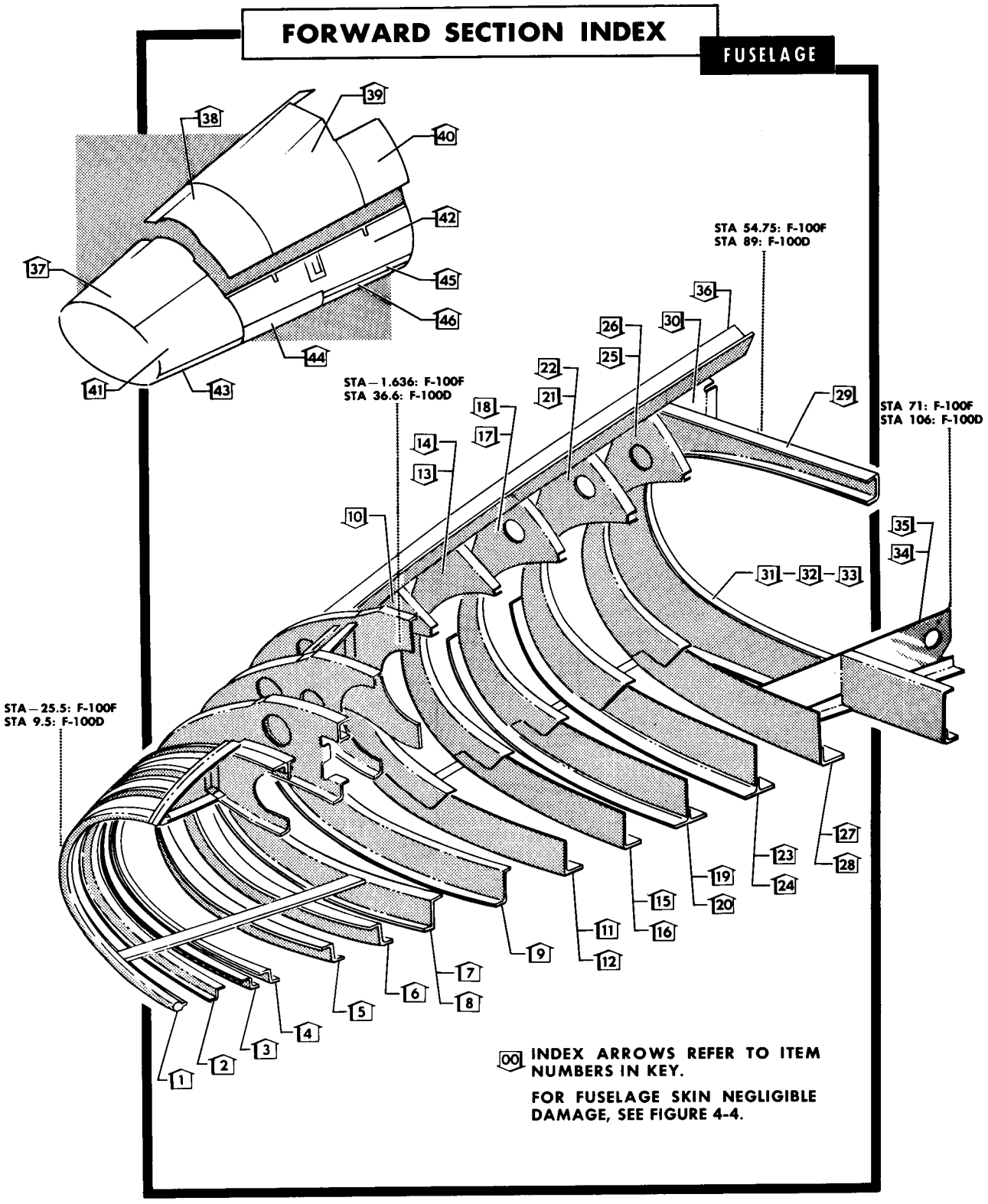


Figure 4-5. Forward Section Repair Index (Sheet 1 of 2)

CRACK REPAIRABILITY LIMITS NOSE SECTION LOWER SKIN SUPPORT ANGLES, FS72 TO 106.

1. Repairable Cracks:

a. Damage Limits:

(1) Max crack size is 1-inch or 1/4 the length of the angle, whichever is smaller.

(2) At either end of a "T" angle, not more than one flange may be cracked.

b. Procedures:

T.O. 1F-100D-3, Section X has general repair procedures.

2. Part Replacement:

a. Reasons:

(1) Crack beyond limits.

(2) Previously repaired crack begins to grow again.

(3) Location of repairable crack prohibits repair part installation.

b. Options:

(1) Install new part.

(2) Manufacture "L" angle substitution, (Table 1).

(a) Only angles listed have substitutions.

(b) Overall dimensions should match part to be replaced.

TABLE 1. SHEET METAL SUBSTITUTES (Ref. 1F-100D-3, Section VIII)

STATION	PART NO.	DASH NO.	EXTRUSION NO.	MATL. THICKNESS/ BEND RADIUS
76.25	223-31103	-55, -56, -57, -117	78J	0.063/0.063
84.75	223-31103	-47 to -54	78J	0.063/0.063
89.0	223-31718	-5	14263	0.090/0.094
		-33 to -36	78F	0.090/0.125
93.25	223-31103	-39 to -47	78J	0.063/0.063
97.5	223-31721	-11	AND10133-0701	0.063/0.094
		-13 to -16	28554	0.090/0.125
101.5	223-31103	-31 to -38	78J	0.063/0.063
		LONGERONS		
N/A	223-31103	-59 to -62	78J	0.063/0.063

NOTE

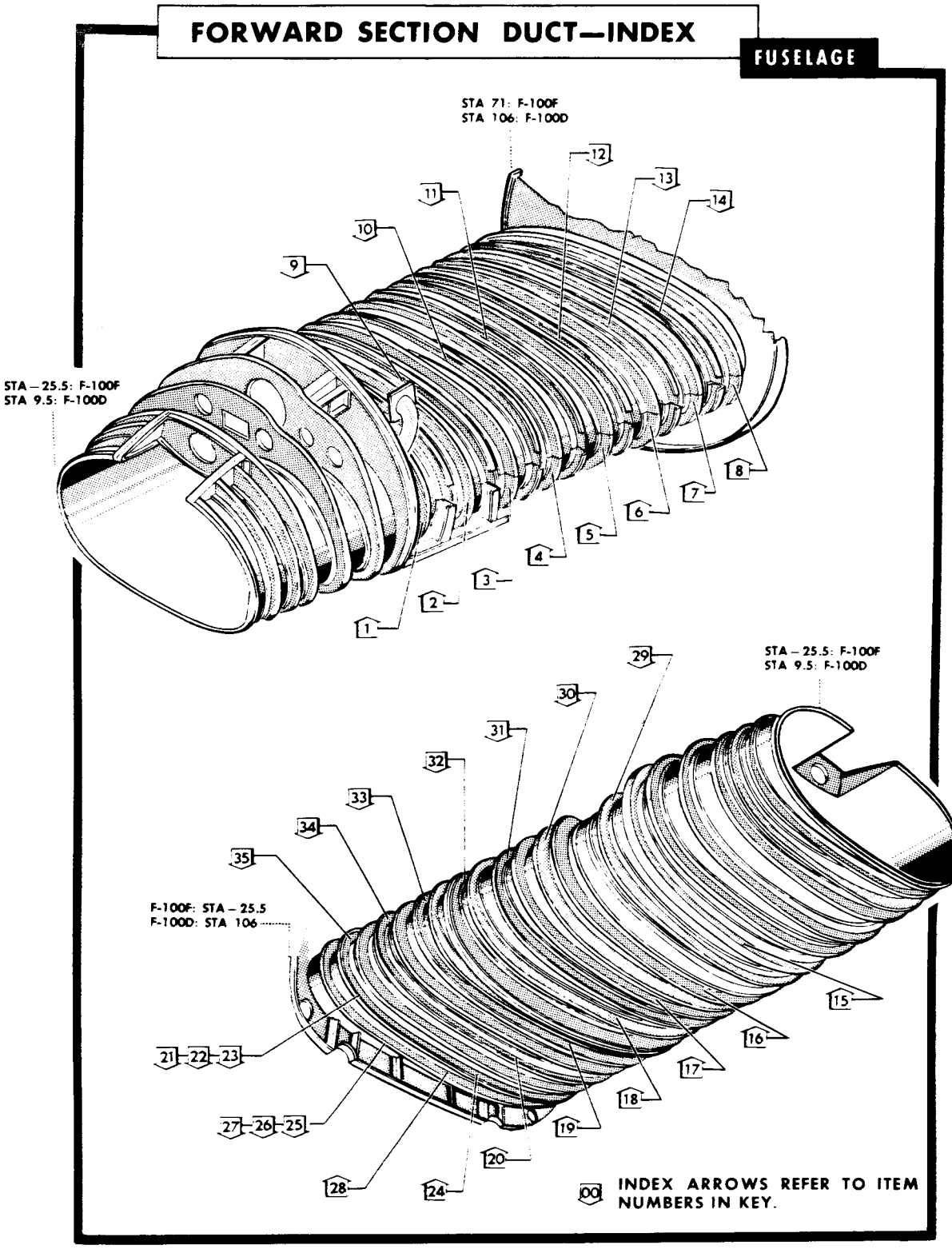
1. Material: 4130 normalized steel. Parts must be heat-formed and then reheat-treat to the normalized condition.
2. Use Section VIII of the -3 to determine the shape and size of the substitution angles. Length and any joggles must be determined from the part to be replaced.
3. Dissimilar metal corrosion prevention procedure as per the -3 should be followed during assembly and installation.

ITEM NO.	DRAWING NO.*	DESCRIPTION	F-100D	F-100F	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31031	Leading Edge	Sta 9.5	Sta 25.5	5E191	7075-T6 Extr	4-8
2	192-31038-11	Frame	Sta 13.5	Sta 21.5	0.063	7075-T6 Clad	4-9
3	192-31039	Frame	Sta 16.5	Sta 18.5	0.063	7075-T6 Clad	4-9
4	192-31040	Frame Assy	Sta 19.5	Sta 15.5	0.063	7075-T6 Clad	4-9
5	192-31041	Frame Assy	Sta 25.5	Sta 9.5	0.063	7075-T6 Clad	4-9
6	192-31042	Frame Assy	Sta 31.5	Sta 3.5	0.063	7075-T6 Clad	4-9
7	192-31701	Bulkhead Assy	Sta 36.7	Sta 1.636	0.063	7075-T6 Clad	4-10A
8	192-31624	Cap Angle	Sta 36.7	Sta 1.636	0.063	7075-T6 Clad	
9	223-31702	Frame Lower Cap	Sta 46.5	Sta 11.5	1E238	7075-T6 Extr	4-30
10	223-31706	Frame Assy—Side	Sta 55	Sta 20	0.063	7075-T6 Clad	
11	223-31704-3	Frame Assy—Lower	Sta 55	Sta 20	0.032	7075-T6 Clad	4-30
12	223-31704-5	Frame Assy—Lower	Sta 55	Sta 20	1E238	7075-T6 Extr	4-30
13	192-31710-3	Frame—Side	Sta 63.5	Sta 28.5	0.071	7075-T6 Clad	4-10A
14	192-31710-5	Frame—Side	Sta 63.5	Sta 28.5	28554	7075-T6 Extr	
15	192-31708-3	Frame Assy—Bottom	Sta 63.5	Sta 28.5	0.032	7075-T6 Clad	4-30
16	192-31708-5	Frame Assy—Bottom	Sta 63.5	Sta 28.5	28554	7075-T6 Extr	
17	192-31713-3	Frame—Side	Sta 72	Sta 37	0.080	7075-T6 Clad	4-10A
18	192-31713-5	Frame—Side	Sta 72	Sta 37	28554	7075-T6 Extr	4-10A
19	223-31714-3	Frame Assy—Lower	Sta 72	Sta 37	0.032	7075-T6 Clad	4-30
20	223-31714-5	Frame Assy—Lower	Sta 72	Sta 37	4E347	7075-T6 Extr	4-30
21	192-31616-3	Frame—Side	Sta 80.5	Sta 45.5	0.080	7075-T6 Clad	4-10A
22	192-31616-5	Frame—Side	Sta 80.5	Sta 45.5	28554	7075-T6 Extr	4-10A
23	223-31715-3	Frame—Lower	Sta 80.5	Sta 45.5	0.032	7075-T6 Clad	4-30
24	223-31715-5	Frame—Lower	Sta 80.5	Sta 45.5	29140	7075-T6 Extr	
25	192-31719-3	Frame Assy—Side	Sta 89	Sta 54	0.080	7075-T6 Clad	
26	192-31719-5	Frame Assy—Side	Sta 89	Sta 54	28554	7075-T6 Extr	
27	192-31718-3	Frame—Lower	Sta 89	Sta 54	0.032	7075-T6 Clad	4-30
28	192-31718-5	Frame—Lower	Sta 89	Sta 54	14263	7075-T6 Extr	4-30
29	192-31697	Frame—Upper	Sta 97.5	Sta 62.5	0.080	7075-T6 Clad	
30	192-31722	Frame—Side	Sta 97.5	Sta 62.5	0.090	7075-T6 Clad	
31	192-31721-3	Frame Assy—Lower	Sta 97.5	Sta 62.5	0.032	7075-T6 Clad	4-30
32	223-31721-9 -11	Frame Assy—Lower	Sta 97.5	Sta 62.5	10131-0701	2024-T42 Extr	4-30
33	223-31721-13 -15 -17	Frame Assy—Lower	Sta 97.5	Sta 62.5	28554	7075-T6 Extr	
34	192-31725	Intercostal Assy	Sta 89-106	Sta 54.71	0.040	7075-T6 Clad	
35	192-31163	Cap Angle	Sta 89-106	Sta 54.71	29140	2024-T42 Extr	4-30
36	192-31110	Upper Longeron	Sta 36-106	Sta 1.71	0.071	7075-T6 Clad	4-11
37	192-31004	Skin			0.063	7075-T6 Clad	4-31, 4-32, 4-33
38	223-31003	Skin			0.032	2024-T4 Clad	4-31, 4-32, 4-33
39	223-31003	Skin			0.032	2024-T4 Clad	4-31, 4-32, 4-33
40	223-31003	Skin			0.032	2024-T4 Clad	4-31, 4-32, 4-33
41	192-31004	Skin			0.063	7075-T6 Clad	4-31, 4-32, 4-33
42	223-31103	Skin			0.040	2024-T4 Clad	4-31, 4-32, 4-33
43	192-31004	Skin			0.063	7075-T6 Clad	4-31, 4-32, 4-33
44	223-31103	Skin			0.050	2024-T4 Clad	4-31, 4-32, 4-33
45	223-61040	Skin			0.080	CRES	4-31, 4-32, 4-33
46	223-31103	Skin			0.040	2024-T4 Clad	4-31, 4-32, 4-33

*** Drawing numbers are for reference only and are not to be used for ordering parts.**

† For additional repairs, refer to Section X.

Key to Figure 4-5 (Sheet 1 of 2)



F-100D-3-31-77B

Figure 4-5. Forward Section Repair Index (Sheet 2 of 2)

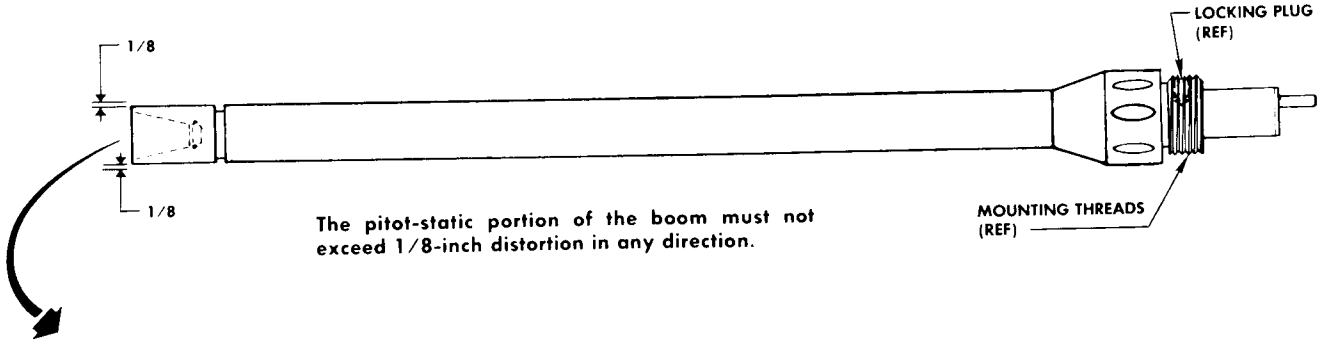
ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-316277	Duct Frame—Upper	Sta 42	0.080	7075-T6 Clad	
2	223-31613	Duct Frame—Upper	Sta 50.75	0.080	7075-T6 Clad	
3	192-31659	Duct Frame—Upper	Sta 59.25	0.080	7075-T6 Clad	
4	192-31667	Duct Frame—Upper	Sta 67.75	0.090	7075-T6 Clad	
5	192-31676	Duct Frame—Upper	Sta 76.25	0.090	7075-T6 Clad	
6	192-31684	Duct Frame—Upper	Sta 84.75	0.090	7075-T6 Clad	
7	192-31693	Duct Frame—Upper	Sta 93.25	0.090	7075-T6 Clad	
8	192-316101	Duct Frame—Upper	Sta 101.75	0.090	7075-T6 Clad	
9	223-31614	Duct Frame—Upper	Sta 46.5	0.063	7075-T6 Clad	
10	223-31705	Duct Frame—Upper	Sta 55	0.063	7075-T6 Clad	
11	192-31709	Duct Frame—Upper	Sta 63.5	0.063	7075-T6 Clad	
12	192-31672	Duct Frame—Upper	Sta 72	0.063	7075-T6 Clad	
13	192-31680	Duct Frame—Upper	Sta 80.5	0.071	7075-T6 Clad	
14	192-31689	Duct Frame—Upper	Sta 89	0.071	7075-T6 Clad	
15	192-316278	Duct Frame—Lower	Sta 42	0.080	7075-T6 Clad	
16	192-31651	Duct Frame—Lower	Sta 50.750	0.080	7075-T6 Clad	
17	192-31660	Duct Frame—Lower	Sta 59.250	0.080	7075-T6 Clad	
18	192-31668	Duct Frame—Lower	Sta 67.750	0.090	7075-T6 Clad	
19	223-31708	Duct Frame—Lower Ctr	Sta 76.250	0.090	7075-T6 Clad	
20	223-31709	Duct Frame—Lower	Sta 84.750	0.090	7075-T6 Clad	
21	192-31694-3	Duct Frame—Lower Side	Sta 93.250	77B	7075-T6 Extr	
22	223-31710-7	Duct Frame—Lower Side	Sta 93.250	0.063	7075-T6 Clad	
23	223-31710-5	Duct Frame—Lower Side	Sta 93.250	78U	7075-T6 Extr	
24	192-31695	Duct Frame—Lower Ctr	Sta 93.250	3E25	7075-T6 Extr	
25	223-31711-3	Duct Frame—Lower Side	Sta 101.750	77B	7075-T6 Extr	
26	223-31711-5	Duct Frame—Lower Side	Sta 101.750	78U	7075-T6 Extr	
27	223-31711-7	Duct Frame—Lower Side	Sta 101.750	0.063	7075-T6 Clad	
28	192-316103	Duct Frame—Lower Ctr	Sta 101.750	0.050	7075-T6 Clad	
29	192-31658	Duct Frame—Lower	Sta 46.5	0.063	7075-T6 Clad	
30	223-31707	Duct Frame—Lower	Sta 55	1E481	7075-T6 Extr	
31	192-31711	Duct Frame—Lower	Sta 63.5	0.080	7075-T6 Clad	
32	192-31614	Duct Frame—Lower	Sta 72	0.080	7075-T6 Clad	
33	192-31717	Duct Frame—Lower	Sta 80.5	0.090	7075-T6 Clad	
34	192-31720	Duct Frame—Lower	Sta 89	0.090	7075-T6 Clad	
35	192-31724	Duct Frame—Lower	Sta 97.5	0.090	7075-T6 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

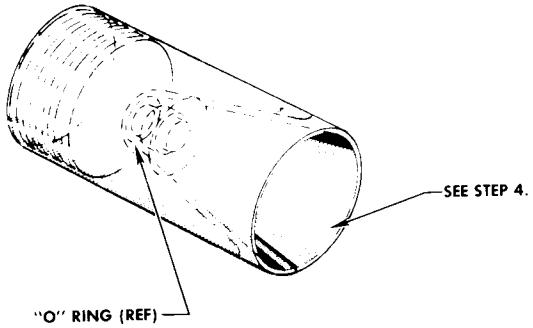
Key to Figure 4-5 (Sheet 2 of 2)

PERMANENT CATEGORY A

FUSELAGE



The pitot-static portion of the boom must not exceed 1/8-inch distortion in any direction.



NOTE

1 Check mounting threads for serviceability.

2 Replace nylon locking plug at time of each overhaul or repair.

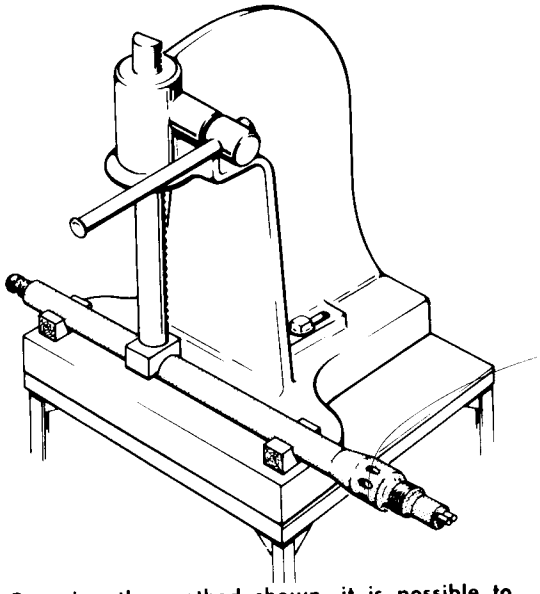
3 Check outer surfaces of tube assembly and pitot head for dents, nicks, and scratches. Smooth nicks, scratches, and sharp dents to blend with surrounding surface. Defects to 0.008 inch deep after smoothing are acceptable except in the area within 2 inches forward of the static-pressure ports. Flattened areas in the tube are acceptable up to a maximum of 0.020 inch depth except in an area within 2 inches either forward or aft of the static-pressure ports.

4 Check pitot head assembly for damage to inner surface of cone and leading edge. Replace pitot head assembly if either of these areas is damaged or distorted.

5 Replace 0227-5 "O" ring in pitot-static head.

6 No defects are permissible in the pitot tube within the pitot head.

7 The external portion of the pitot and static tubes must be free of nicks, burrs, and dents. Minor damage to the ends of these tubes may be repaired by shortening the protruding part of the tubes to 15/16 inch minimum. See sheet 2 for repair of more extensive damage.



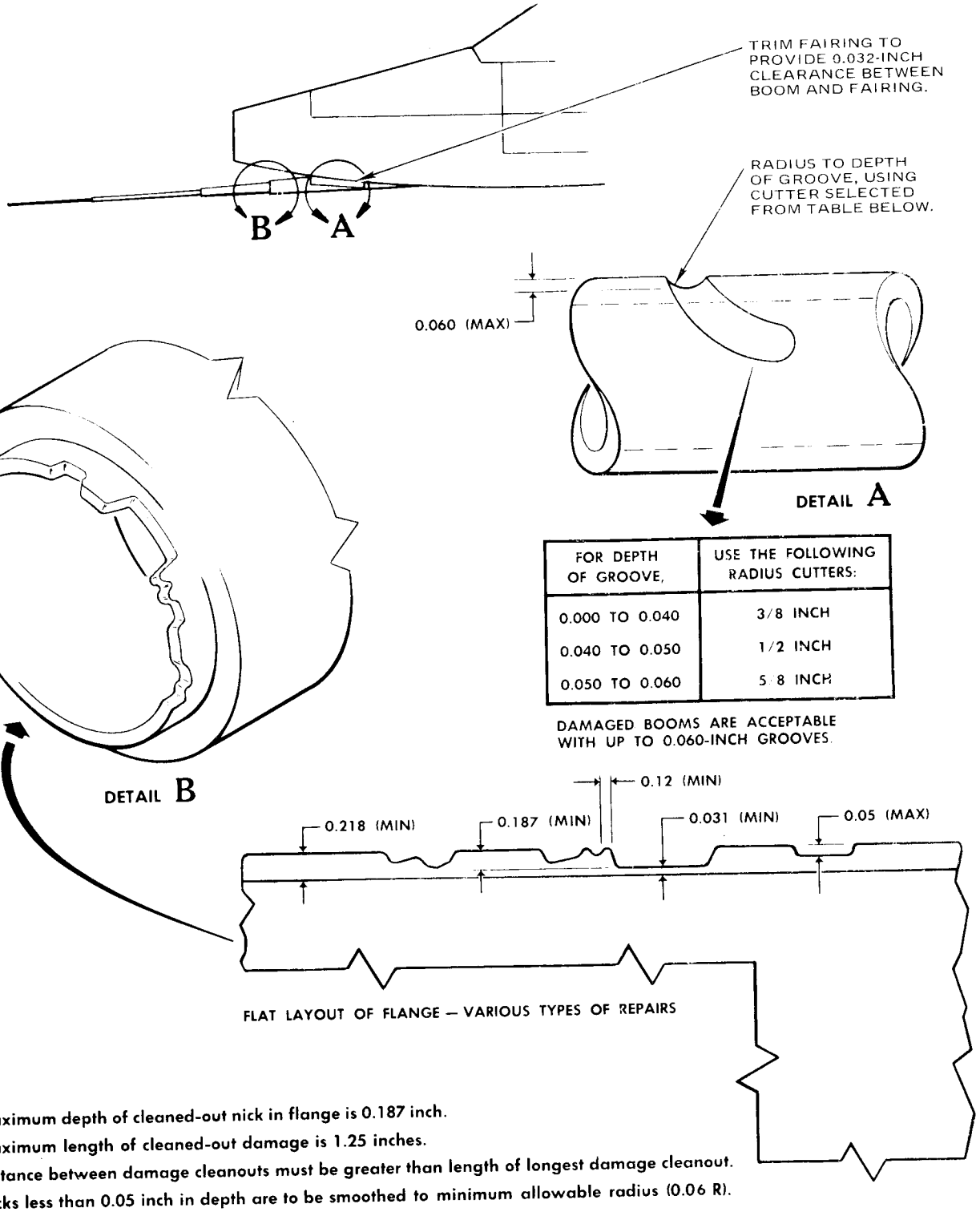
By using the method shown, it is possible to straighten the sections of the pitot-static boom. The holding fixtures should be made of wood to prevent marring the tube.

100D-3-31-97C

Figure 4-5A. Folding Pitot Boom Repair (Sheet 1 of 4)

PERMANENT—CATEGORY A

FUSELAGE



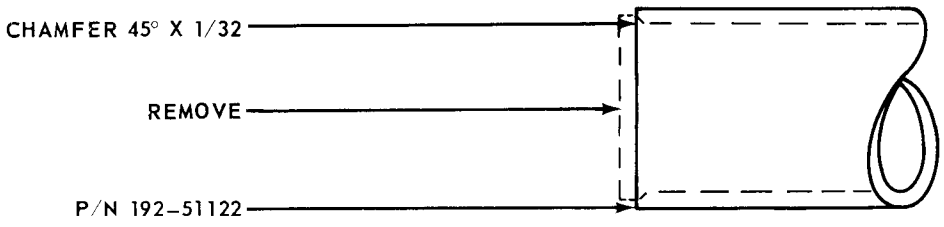
FOR DEPTH OF GROOVE,	USE THE FOLLOWING RADIUS CUTTERS:
0.000 TO 0.040	3/8 INCH
0.040 TO 0.050	1/2 INCH
0.050 TO 0.060	5/8 INCH

DAMAGED BOOMS ARE ACCEPTABLE WITH UP TO 0.060-INCH GROOVES.

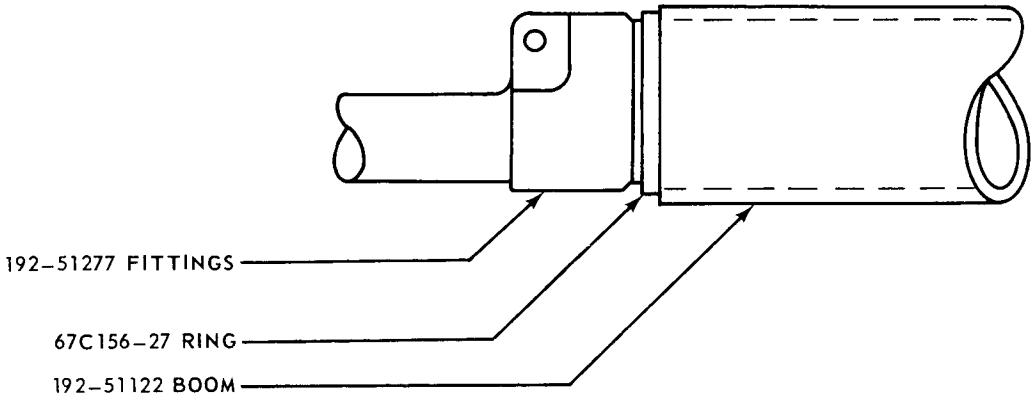
1. Maximum depth of cleaned-out nick in flange is 0.187 inch.
2. Maximum length of cleaned-out damage is 1.25 inches.
3. Distance between damage cleanouts must be greater than length of longest damage cleanout.
4. Nicks less than 0.05 inch in depth are to be smoothed to minimum allowable radius (0.06 R).
5. Total length of all cleaned-up damage must not exceed 2.73 inches.

Figure 4-5A. Folding Pitot Boom Repair (Sheet 2 of 4)

NOTE: WHEN FLANGE REPAIR LIMITS OF T.O. 1F-100D-3, FIGURE 4-5A, ARE EXCEEDED, REMOVE FLANGE FLUSH TO SHOULDER OF BOOM, AND CHAMFER AS SHOWN.



REASSEMBLE WITH RING , PART NO. 67C156-27 AS SHOWN.



MANUFACTURE RING, PART NO. 67C156-27 FROM 4130 STEEL, HEAT-TREAT TO 150,000 TO 160,000 PSI CADMIUM PLATE QQ-P-416, CLASS 2, TYPE 2.

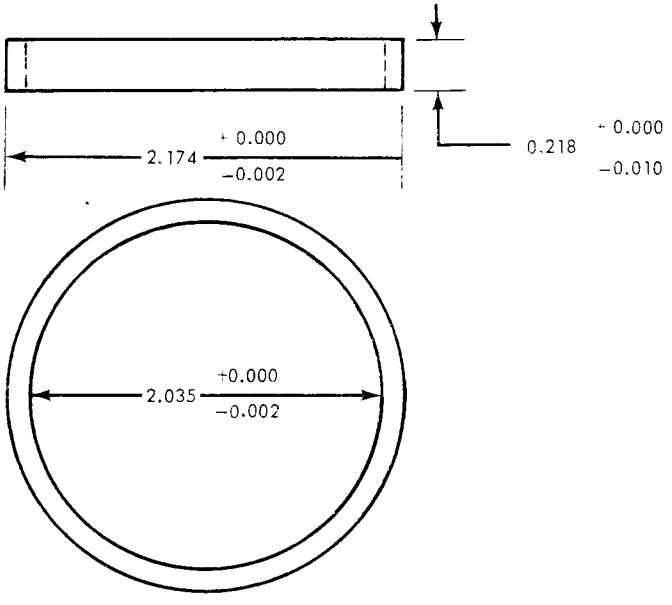
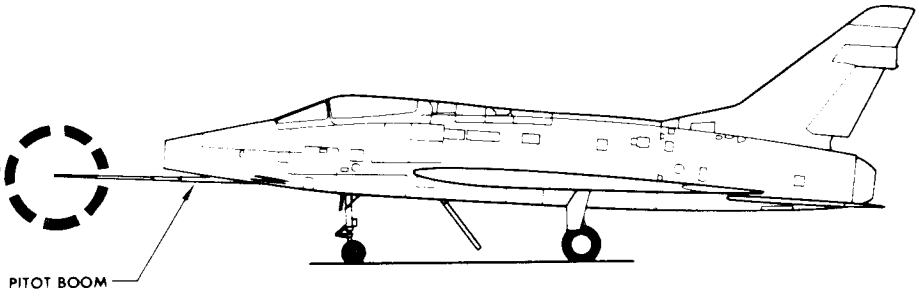
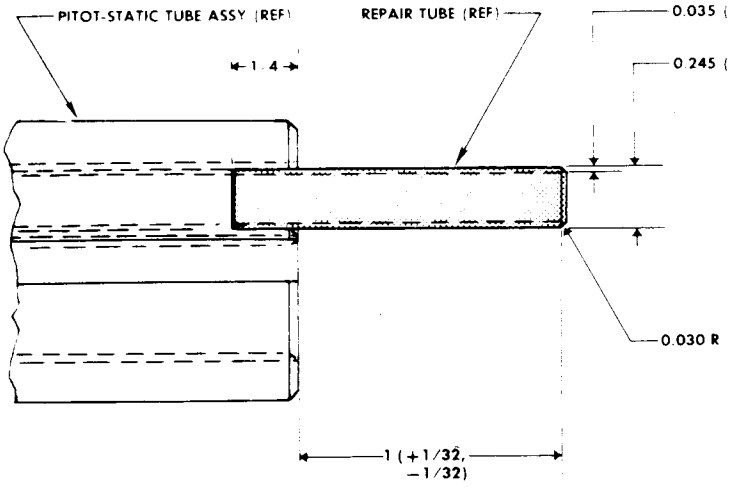


Figure 4-5A. Folding Pitot Boom Repair (Sheet 3 of 4)

PERMANENT—CATEGORY A — FUSELAGE

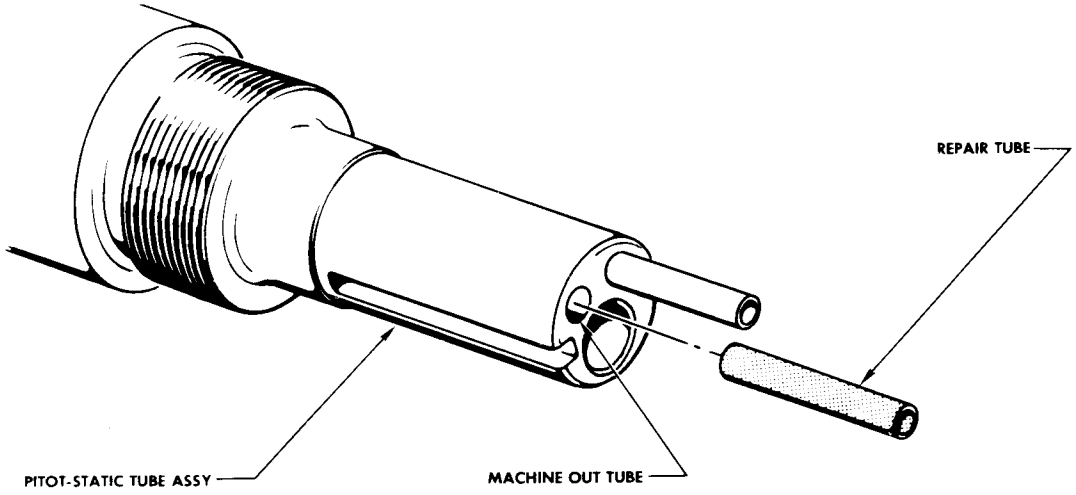
REPAIR OF DAMAGED TUBES OR TUBES THAT ARE BROKEN OFF FLUSH





NOTE Machine out broken-off tubes to a depth of 1/4 inch.

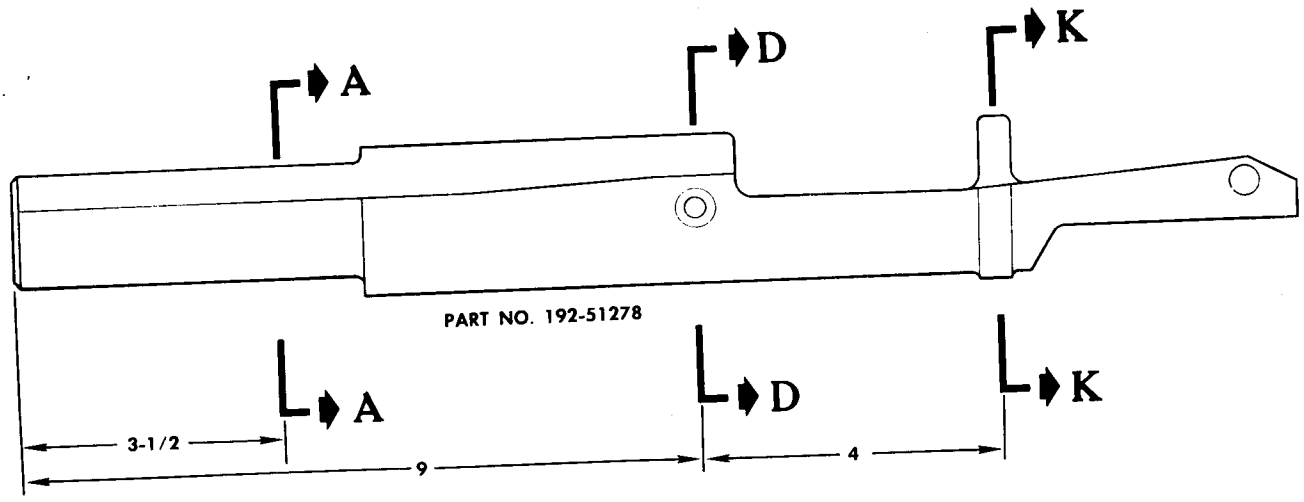
- Make repair tubes from corrosion-resistant steel tubing (Specification MIL-T-8606 or, as a substitute, MIL-T-6845), using dimensions shown.
- Repair tube must align with centerline of existing tube.
- The end of the repair tube must be free of all nicks and burrs.
- Inside of tubes must be clean and free of chips or filings.
- Silver-braze in position according to Specification MIL-B-7883.



F-100D-3-31-138B

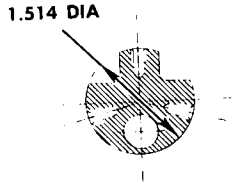
Figure 4-5A. Folding Pitot Boom Repair (Sheet 4 of 4)

PERMANENT—CATEGORY A — FUSELAGE

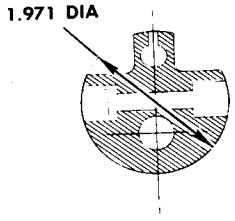


1 Grind nickel plating off damaged areas to minimum dimensions shown below.

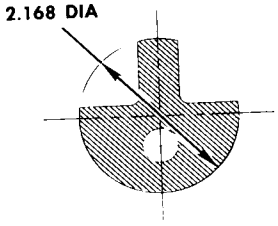
NOTE It is not necessary to strip the nickel plating from surfaces not requiring rework.



SECTION A-A



SECTION D-D



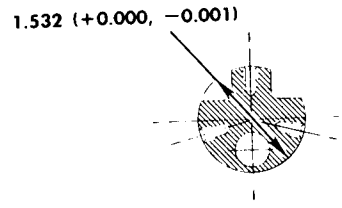
SECTION K-K

(See sheet 2 of 3 for repair of this area.)

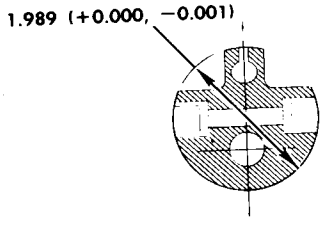
2 Build up ground areas with hard chromium plating in accordance with T.O. 43C2-1-7. Mask off nickel-plated areas as required.

3 Cadmium-plate nickel/chromium-plated fitting all over to effect a corrosion seal at joints between plating and all other exposed surfaces. Remove excess cadmium from nickel/chromium surface by blowing, wiping, or brushing.

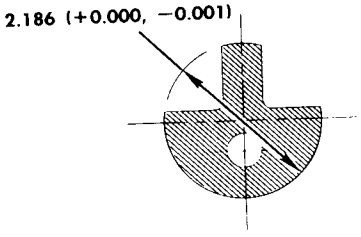
4 Grind chromium-plated areas to dimensions shown below.



SECTION A-A



SECTION D-D



SECTION K-K

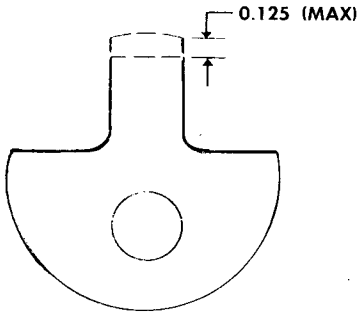
100D-3-31-156A

Figure 4-5B. Pitot Boom Rework (Sheet 1 of 3)

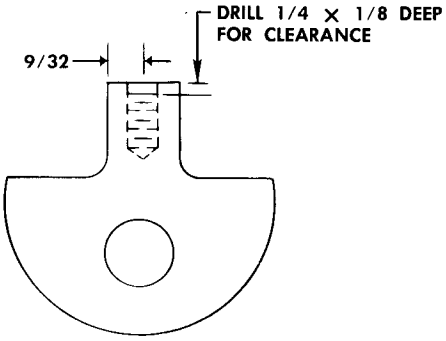
PERMANENT—CATEGORY A

FUSELAGE

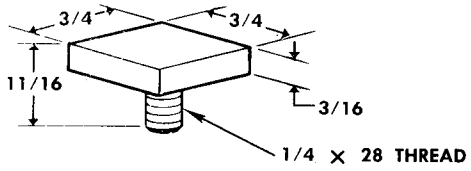
REWORK OF SECTION K-K



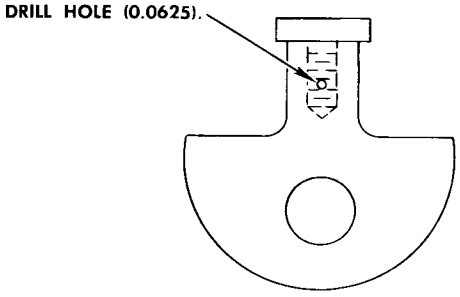
1. Trim off approximately 1/8 inch from top of 7/16 x 9/16 inch projection.



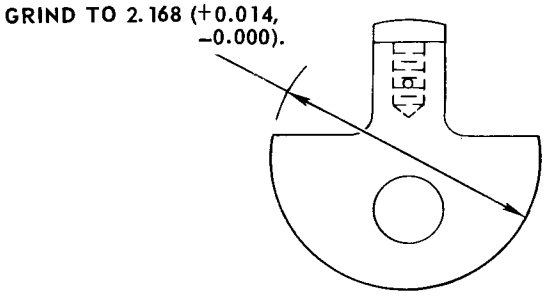
2. In exact center of upper projection, drill with No. 3 (0.213-inch) drill 5/8 inch deep. Tap with 1/4-28 tap.



3. Make cap from 3/4 x 3/4 x 11/16 4340 steel bar, Cond C. Heat-treat to 160,000-180,000 psi per Specification MIL-H-6875.



4. Install cap. Drill (0.0625-inch) hole through fitting and cap threaded portion. Install roll pin (0.0625-inch diameter) through fitting and cap.



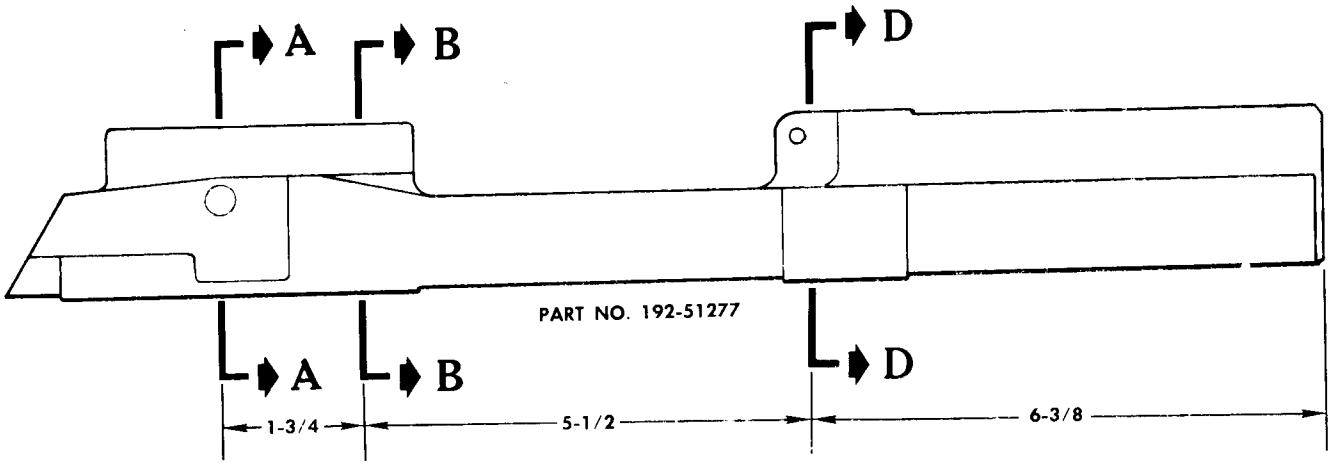
5. Mill excess material from cap and grind to 2.168 inches diameter for plating. After plating, grind to dimensions shown in step 4 of sheet 1.

100D-3-31-164

Figure 4-5B. Pitot Boom Rework (Sheet 2 of 3)

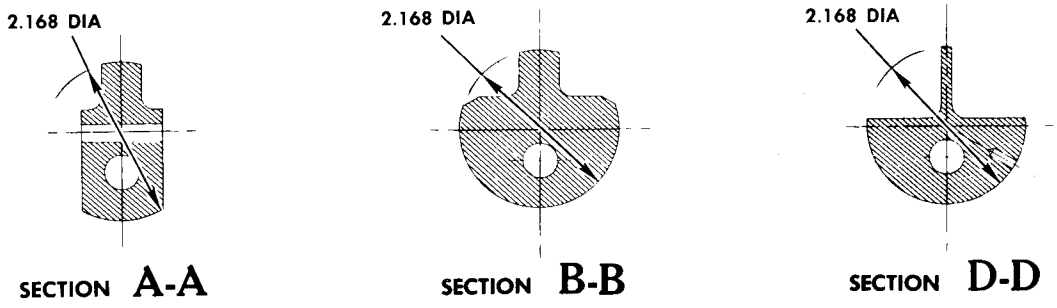
PERMANENT—CATEGORY A

FUSELAGE



1 Grind nickel plating off damaged areas to minimum dimensions shown below.

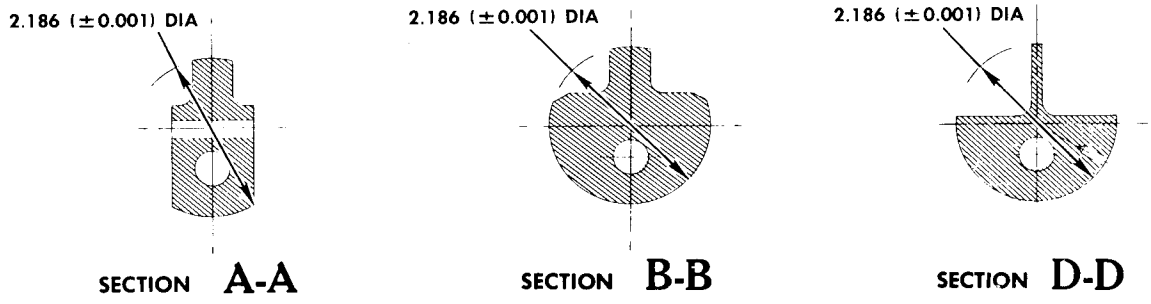
NOTE It is not necessary to strip the nickel plating from surfaces not requiring rework.



2 Build up ground areas with hard chromium plating in accordance with T.O. 42C2-1-7. Mask off nickel-plated areas as required.

3 Cadmium-plate nickel/chromium-plated fitting all over to effect a corrosion seal at joints between plating and all other exposed surfaces. Remove excess cadmium from nickel/chromium surface by blowing, wiping, or brushing.

4 Grind chromium-plated areas to dimensions shown below.



F-100D-3-31-157

Figure 4-5B. Pitot Boom Rework (Sheet 3 of 3)

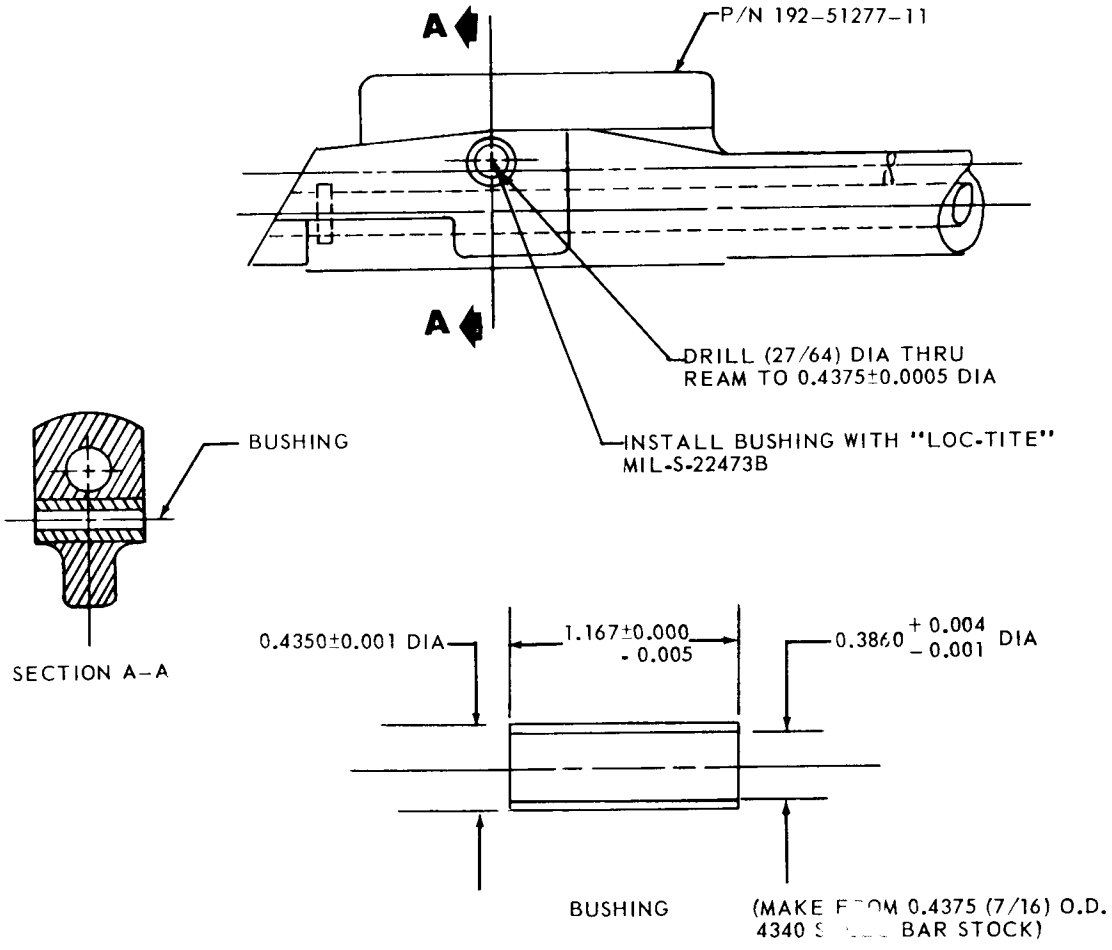
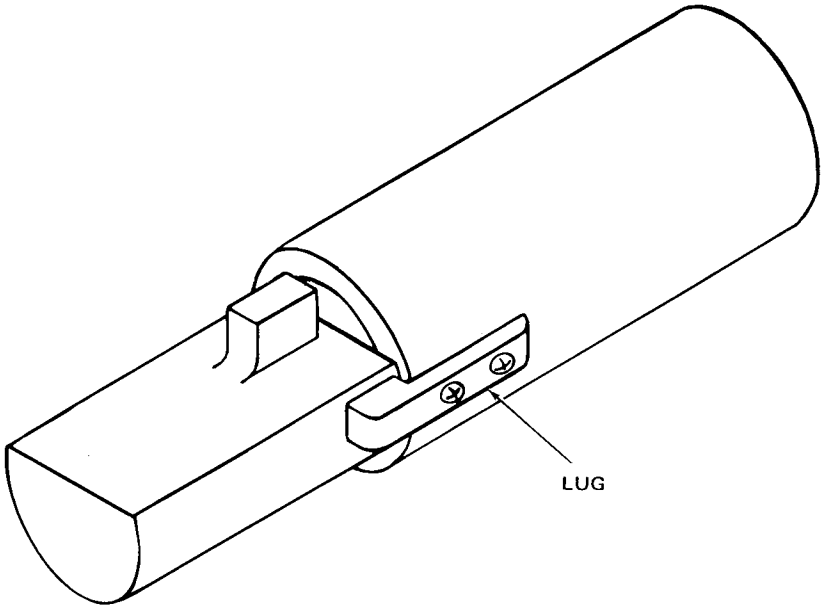
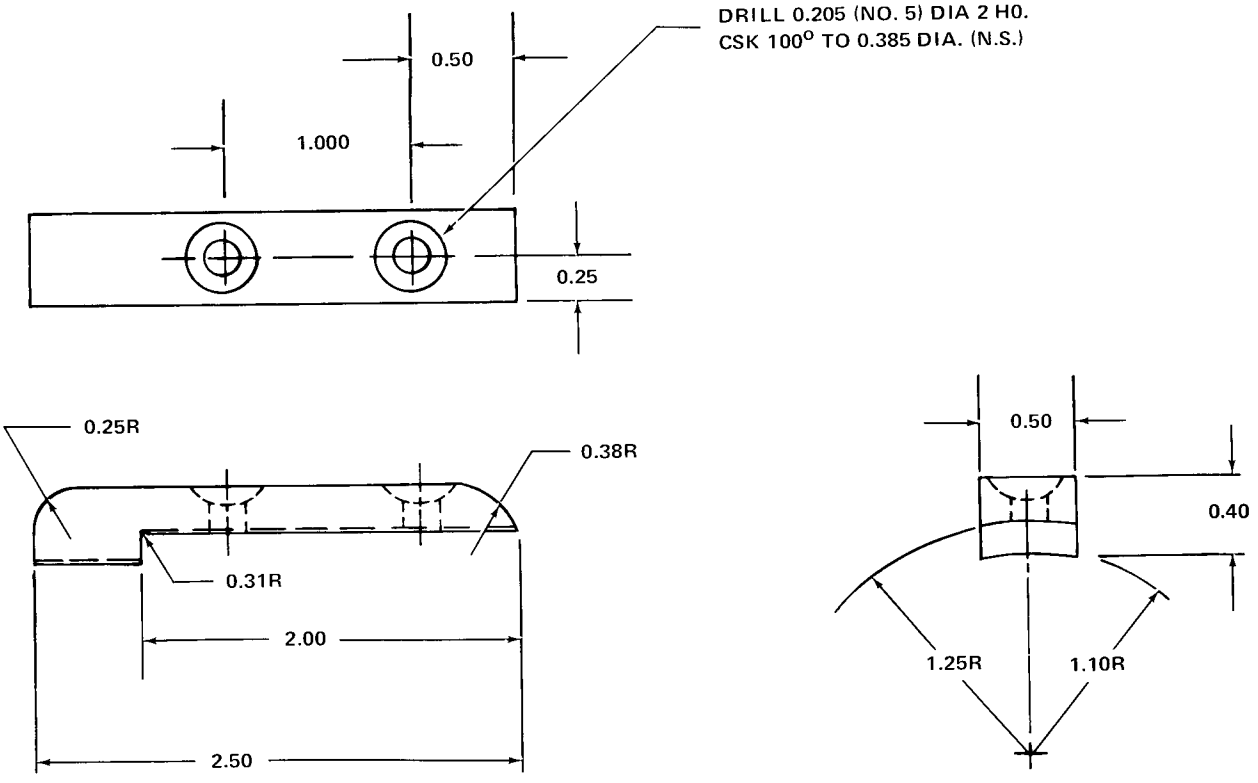


Figure 4-5C. Pitot Boom Rework, Hinge Pin Hole





DETAIL OF LUG

MATL: CRES STEEL BAR 0.50 X 0.50 X 2.50

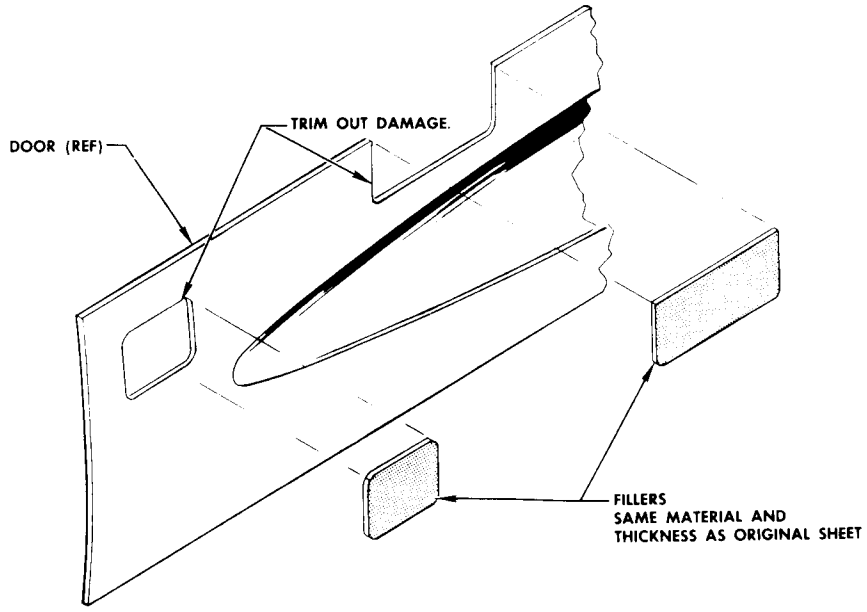
SPEC: MIL-S-7720 TYPE 303 CONDITION A

FINISH: MACHINE ALL OVER 250

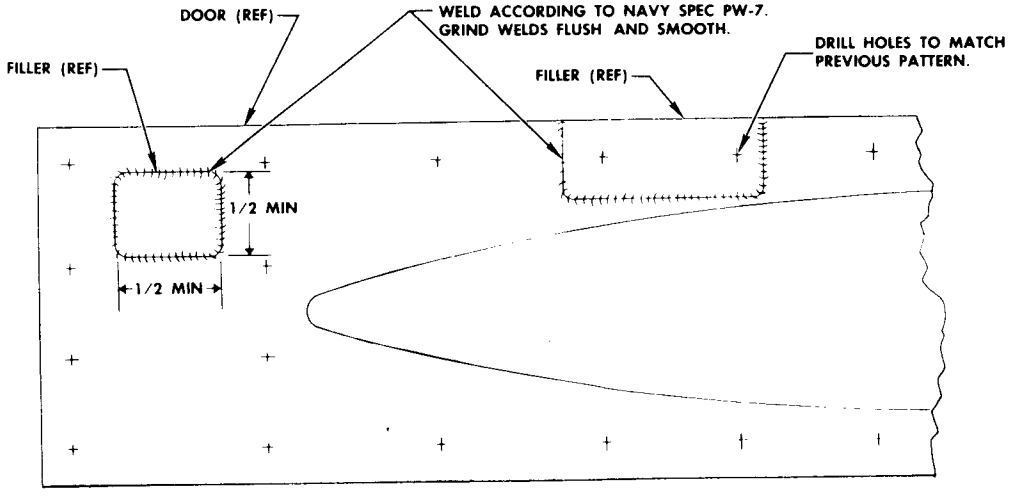
Figure 4-5D. Pitot Boom Alignment Lug Instructions and Mfg.

PERMANENT—CATEGORY A

FUSELAGE



NOTE If major damage is sustained in the aft section of the door, replace the damaged door.



F-100D-3-31-62

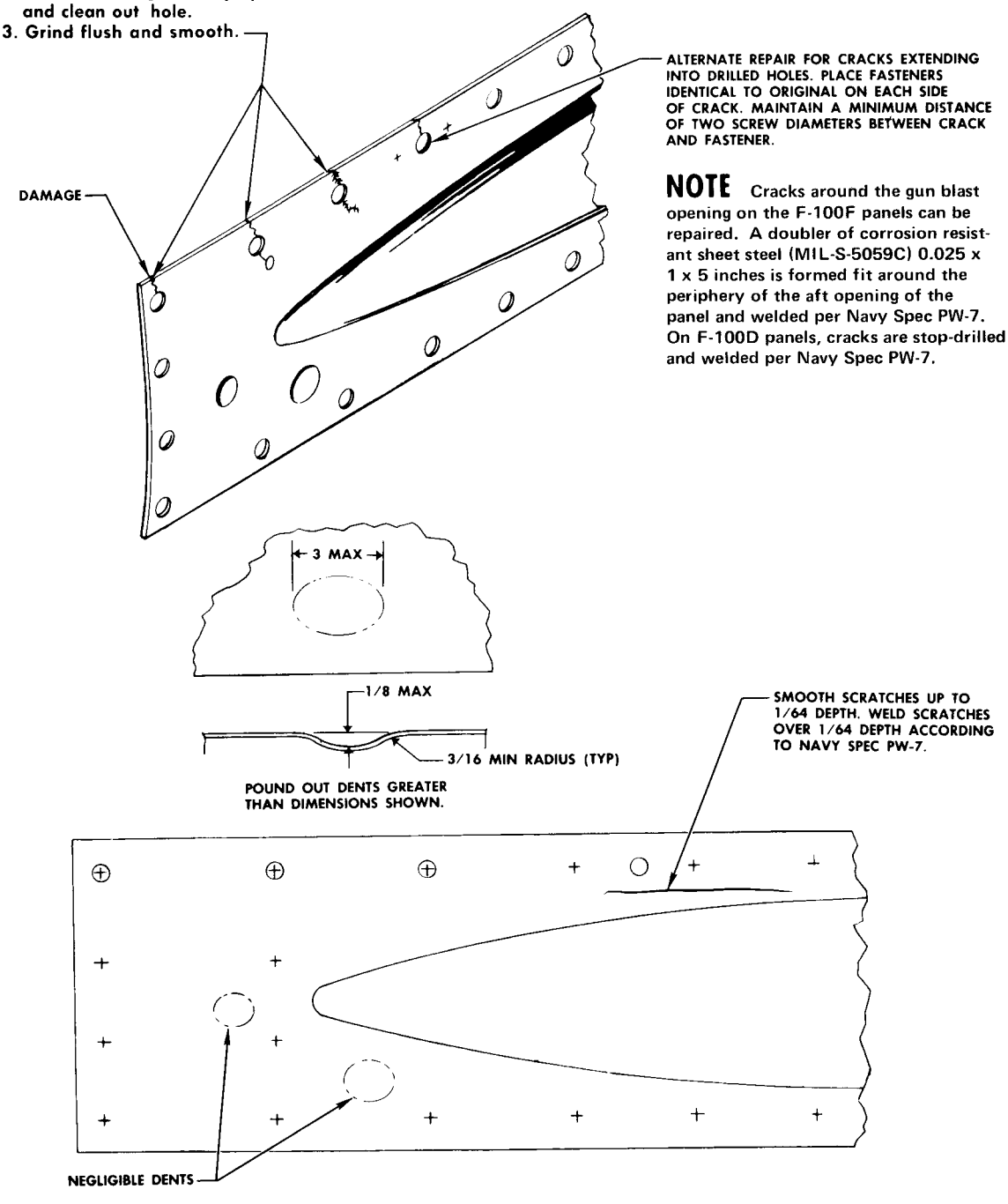
Figure 4-6. Gun Blast Panel

NEGLIGIBLE DAMAGE GUN BLAST DOOR

FUSELAGE

REPAIR STEP

1. Stop-drill.
2. Weld according to Navy Specification PW-7, and clean out hole.
3. Grind flush and smooth.



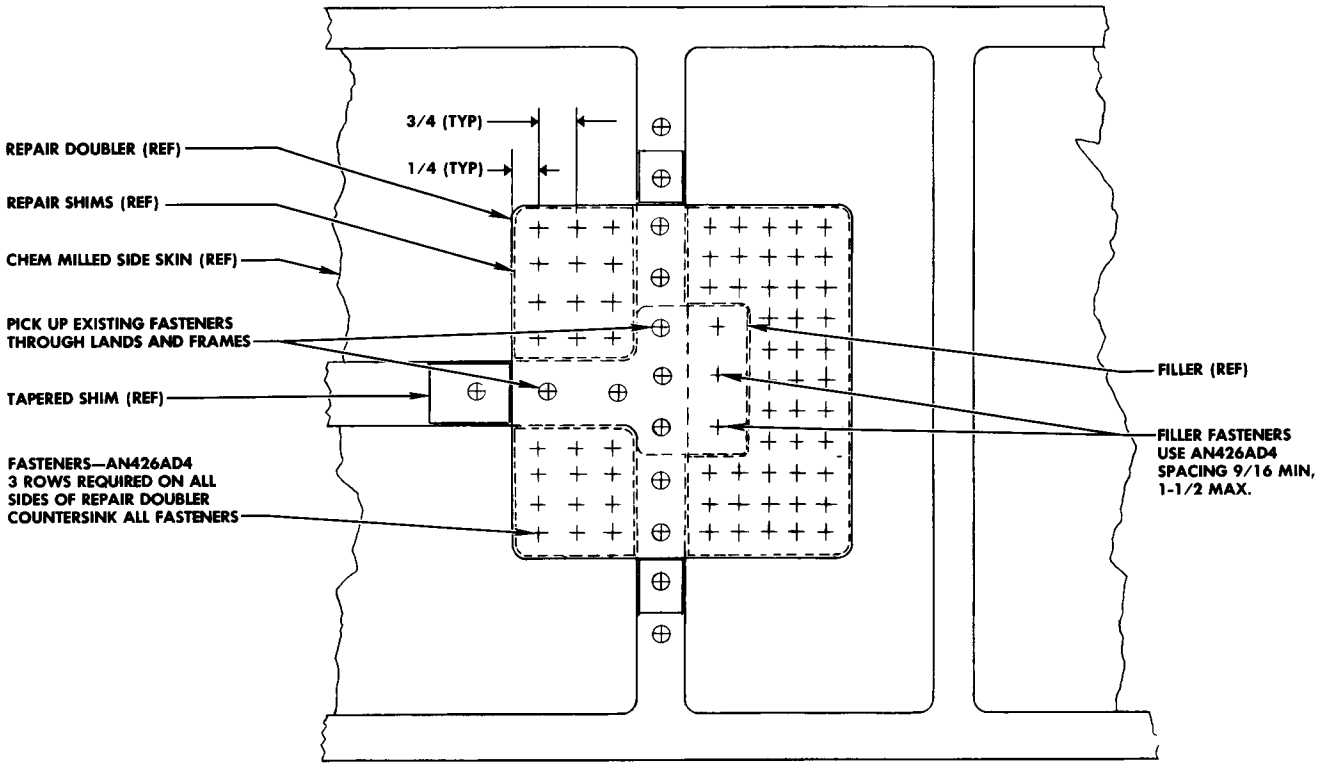
NOTE Cracks around the gun blast opening on the F-100F panels can be repaired. A doubler of corrosion resistant sheet steel (MIL-S-5059C) 0.025 x 1 x 5 inches is formed fit around the periphery of the aft opening of the panel and welded per Navy Spec PW-7. On F-100D panels, cracks are stop-drilled and welded per Navy Spec PW-7.

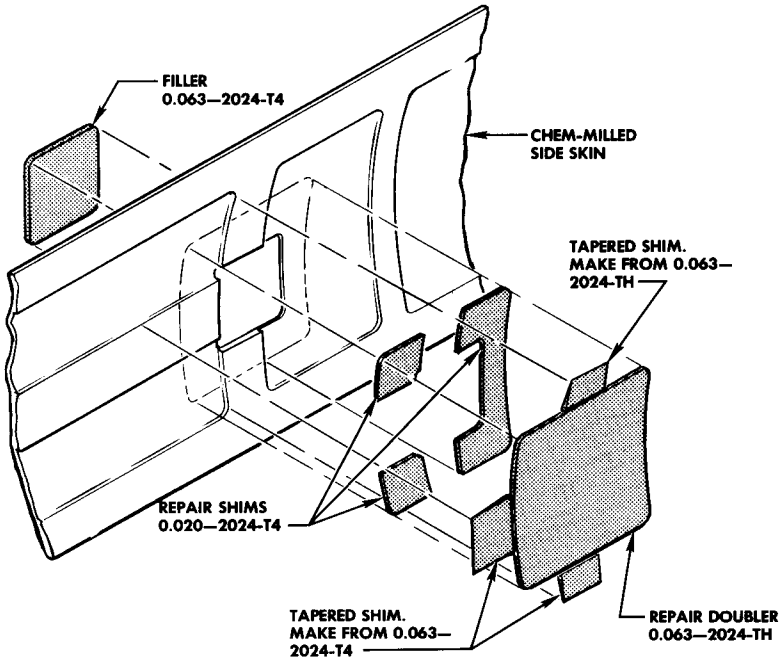
F-100D-3-31-63

Figure 4-7. Negligible Damage—Gun Blast Panel

PERMANENT CATEGORY A

FUSELAGE





1. Trim out damage midway between existing fasteners if damage crosses fastener rows.
2. Smooth all edges and corners after trimming.
3. Make repair doublers, repair shims, and tapered shims.
4. Fasten all repair parts in place, using size and type of fastener and spacing called out.

NOTE All frames having been omitted for clarity.

The tapered shims are used to eliminate jogging frames.

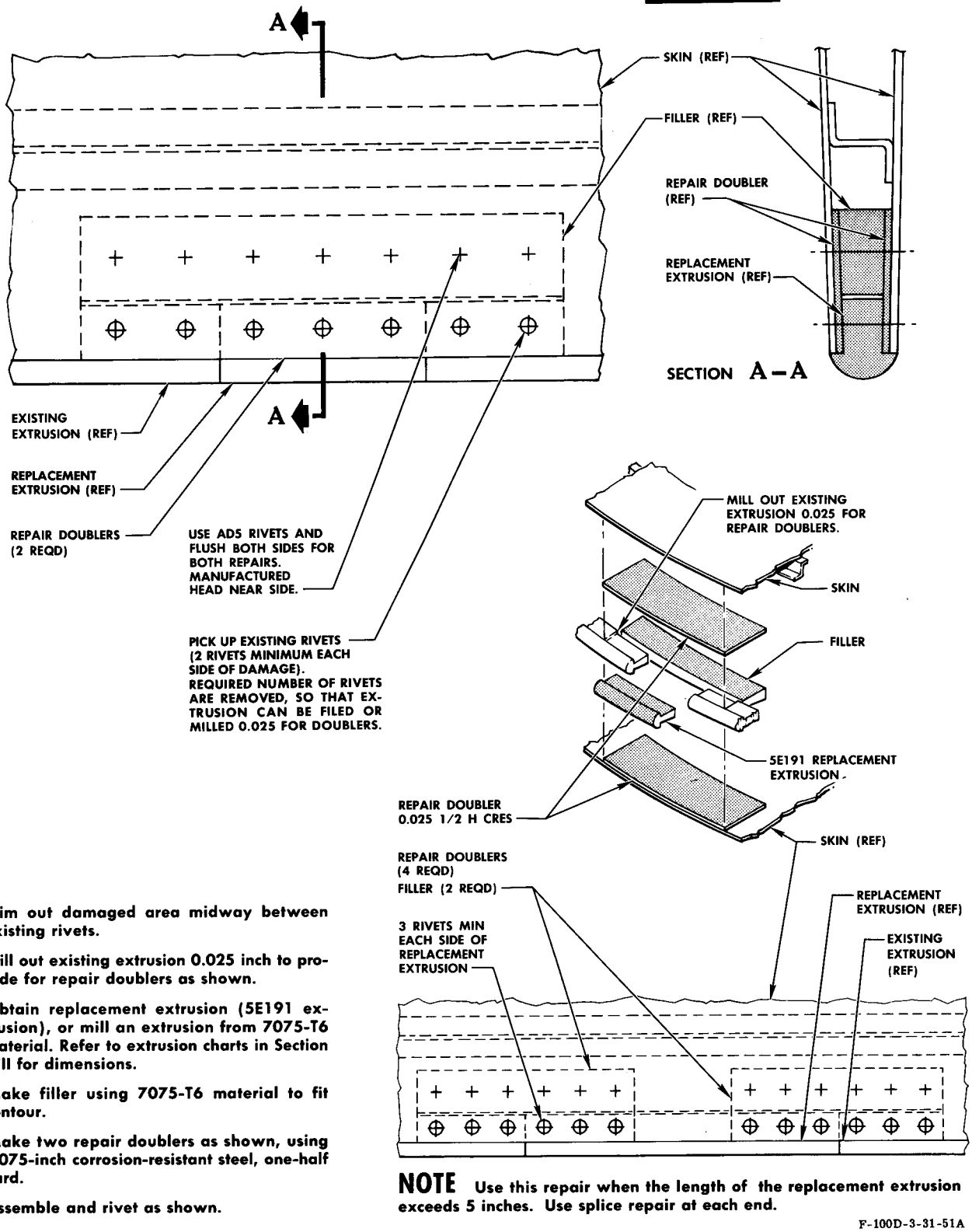
This repair is applicable only to the chem-milled side skin between station 36.623 and station 106 on the F-100D airplane, and between station 1.6 and station 71 on F-100F airplanes.

F-100D-3-31-119

Figure 4-7A. Repair of Chem-Milled Side Skins—Station 36.623 to 106

PERMANENT—CATEGORY A

FUSELAGE



1. Trim out damaged area midway between existing rivets.
2. Mill out existing extrusion 0.025 inch to provide for repair doublers as shown.
3. Obtain replacement extrusion (5E191 extrusion), or mill an extrusion from 7075-T6 material. Refer to extrusion charts in Section VIII for dimensions.
4. Make filler using 7075-T6 material to fit contour.
5. Make two repair doublers as shown, using 0.075-inch corrosion-resistant steel, one-half hard.
6. Assemble and rivet as shown.

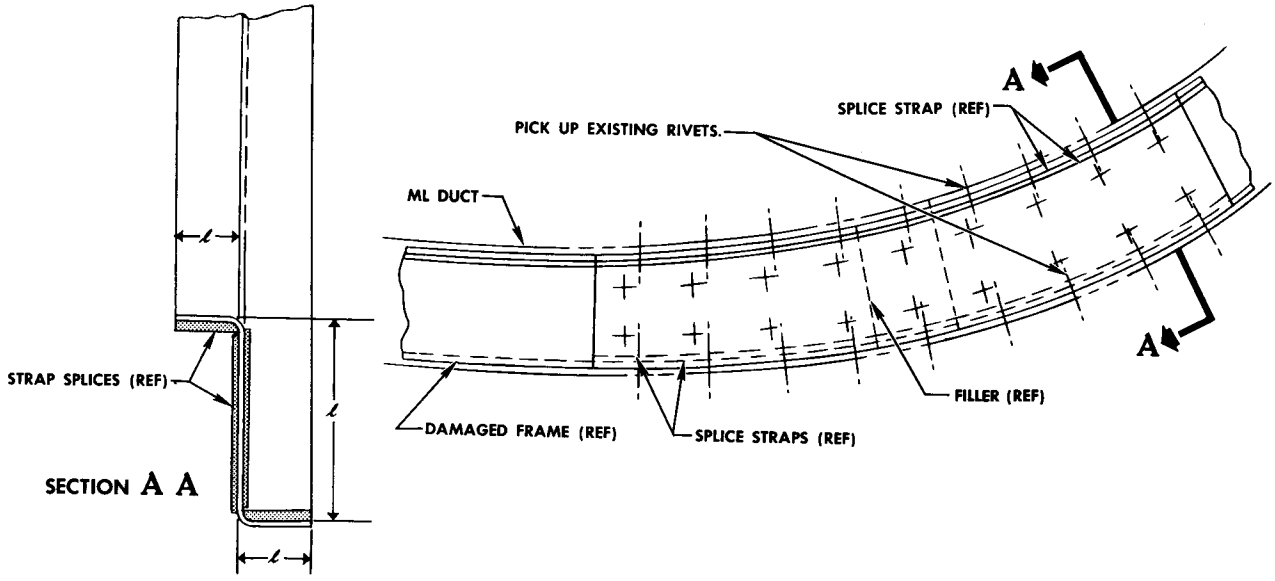
NOTE Use this repair when the length of the replacement extrusion exceeds 5 inches. Use splice repair at each end.

F-100D-3-31-51A

Figure 4-8. Duct Leading Edge Extrusion

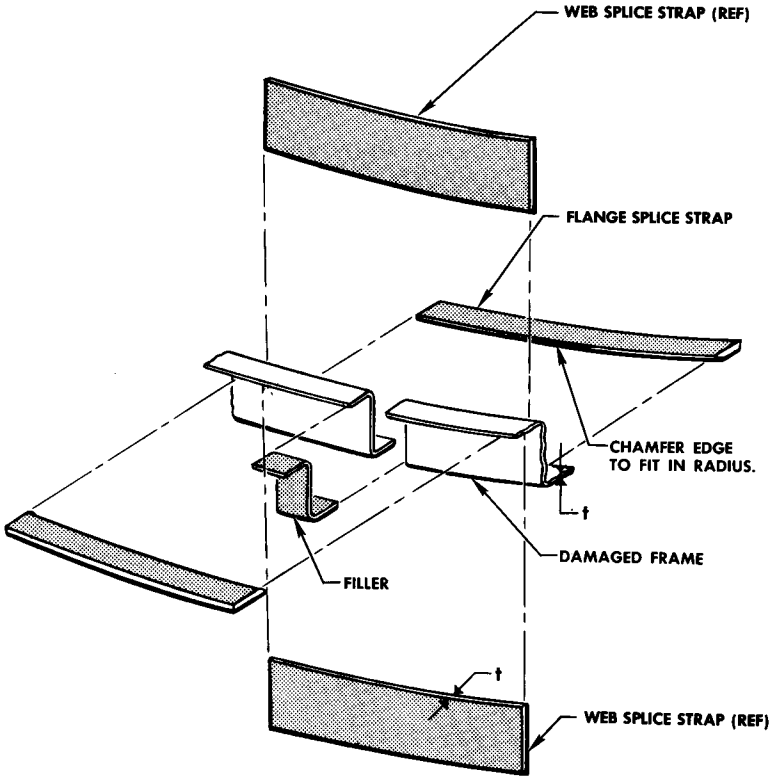
PERMANENT—CATEGORY A

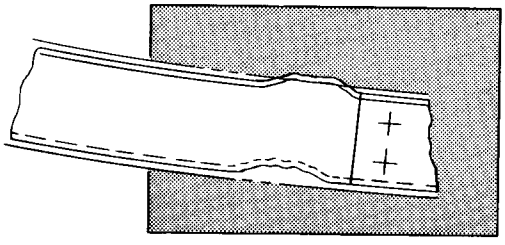
FUSELAGE



1. Trim out damaged area midway between existing rivets.
2. Make filler and web splice straps of same gage and material as damaged frame.
3. Make flange splice strap of same material and 1-1/2 times the gage of damaged frame.
4. Assemble and rivet as shown.

NOTE Refer to the fastener tables in Section X for fastener requirments.





IN CASES WHERE FRAME IS DAMAGED BY BENDING ONLY, FRAME MAY BE CONTOURED BACK TO SHAPE. IF CRACKS APPEAR, USE REPAIR SHOWN

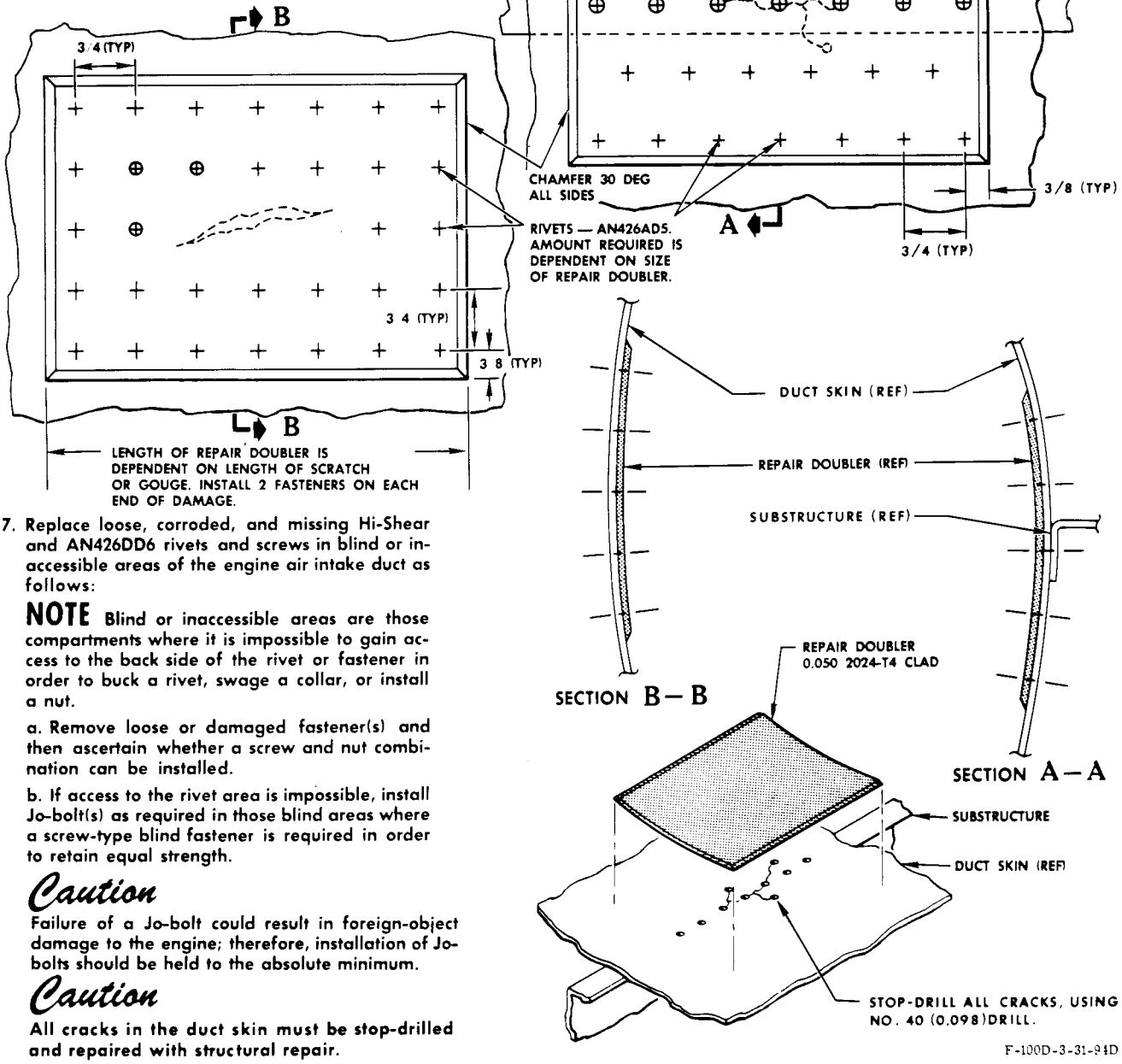
F-100D-3-31-69A

Figure 4-9. Duct Frames—Lower and Side—Stations 16.5, 19.5, 22.5, and 31.5

PERMANENT–CATEGORY A

FUSELAGE

1. Stop-drill all cracks with a No. 40 (0.098-inch) drill.
2. Make repair doubler from 0.050-inch 2024-T4 clad sheet.
3. Make repair doubler large enough to pick up two fasteners at each end of damage and two rows of fasteners on each side of damage (fore and aft). Use 3/4-inch spacing.
4. Dimple repair doubler to match existing holes. Countersink for added fasteners.
5. Prime the concealed surfaces.
6. Fasten repair doubler in position. AF462-5 cherry rivets may be substituted for AN426AD5 rivets in blind or inaccessible areas.



7. Replace loose, corroded, and missing Hi-Shear and AN426DD6 rivets and screws in blind or inaccessible areas of the engine air intake duct as follows:

NOTE Blind or inaccessible areas are those compartments where it is impossible to gain access to the back side of the rivet or fastener in order to buck a rivet, swage a collar, or install a nut.

a. Remove loose or damaged fastener(s) and then ascertain whether a screw and nut combination can be installed.

b. If access to the rivet area is impossible, install Jo-bolt(s) as required in those blind areas where a screw-type blind fastener is required in order to retain equal strength.

Caution

Failure of a Jo-bolt could result in foreign-object damage to the engine; therefore, installation of Jo-bolts should be held to the absolute minimum.

Caution

All cracks in the duct skin must be stop-drilled and repaired with structural repair.

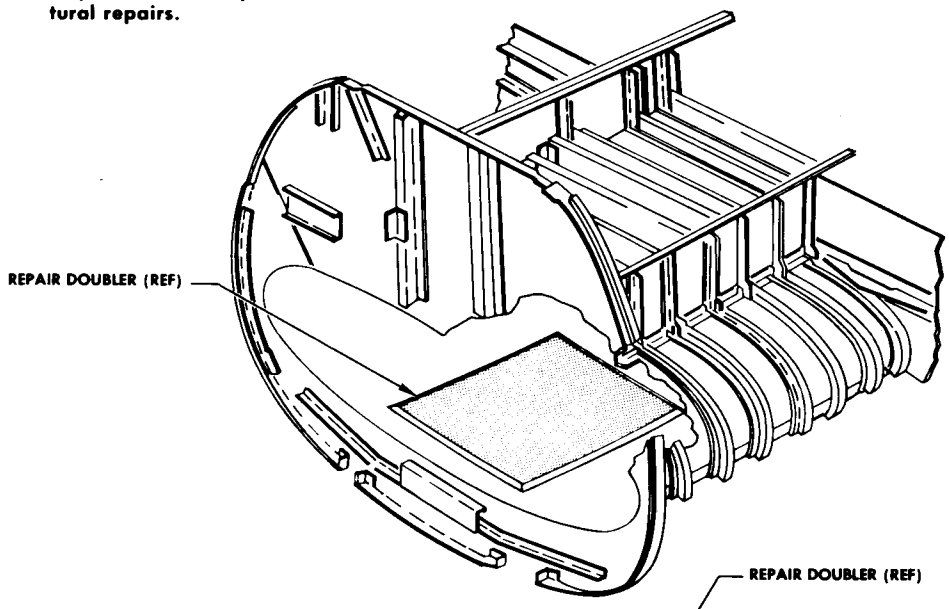
Figure 4-9A. Duct Skin Repair and Rivet Replacement

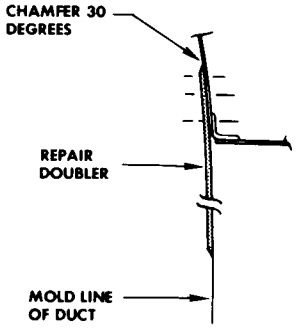
PERMANENT – CATEGORY A

FUSELAGE

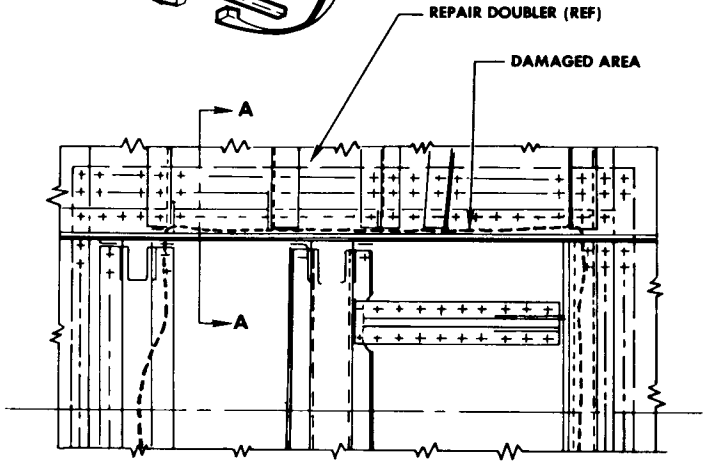
Caution

All cracks in the duct skin must be stop-drilled and repaired with structural repairs.





VIEW A-A



This repair is applicable to any section of the duct. The size is limited only by damage.

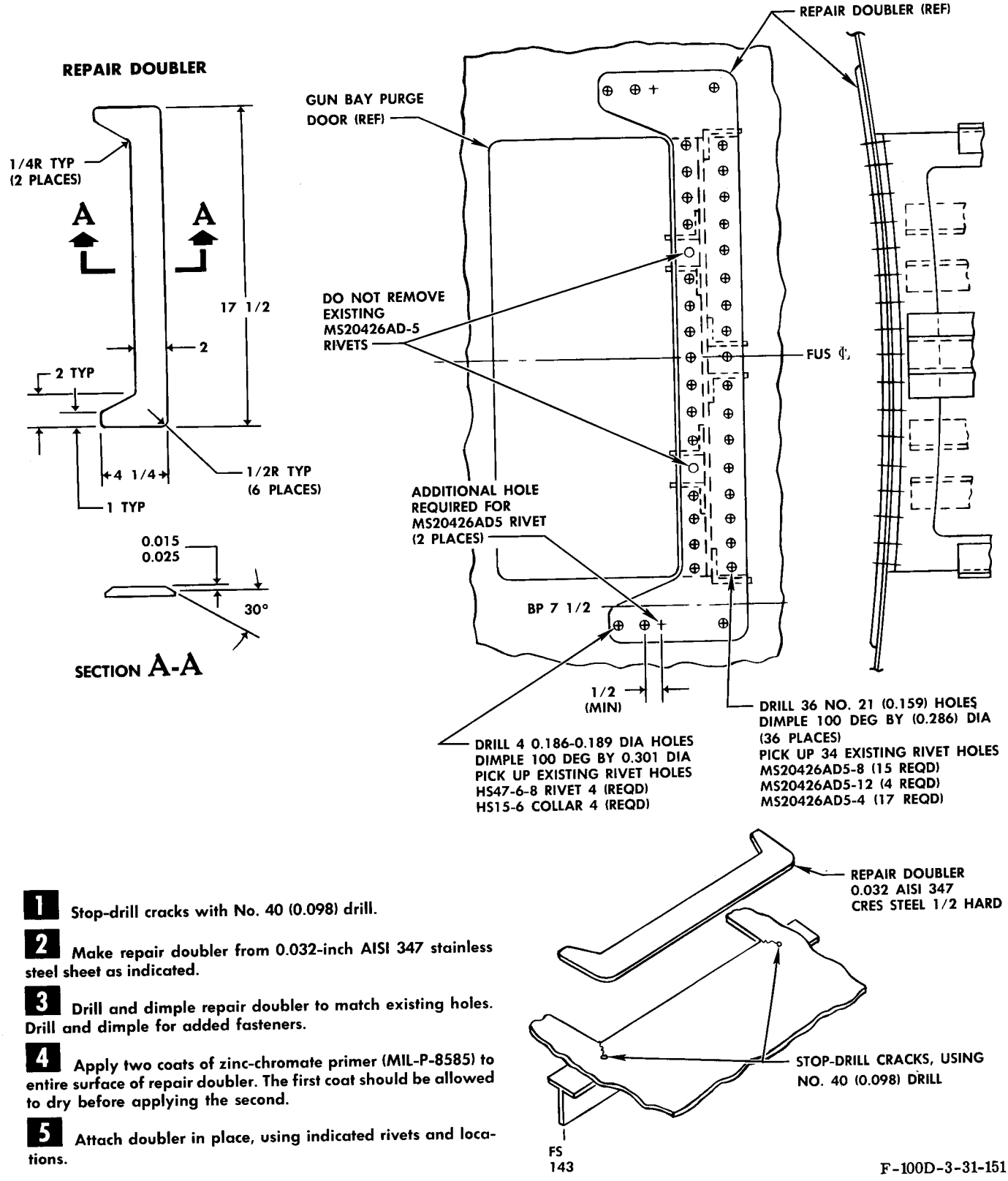
- All repair doublers are to be 0.050 inch 2024-T4 condition. All edges must be chamfered 30 degrees.
- All damaged structure must be repaired or replaced.
- Fillers must be used under the structure where the damaged skin has been removed. Make fillers from same gage as removed skin.
- Fastener requirements are two rows of AN426AD5 fasteners on all sides of repair doublers, except between the upper and lower stringers at station 222.5 to 231 and station 265 to 275.5 where three rows of AN426AD5 fasteners are required on all sides of repair doublers.
- Fastener spacing will be 3/4 inch with 3/8-inch edge distance. The existing fastener spacing through a stiffener may be used for one row of fasteners. Pick up existing fasteners.

F-100D-3-31-98A

Figure 4-9B. Large Area Duct Skin Repair

PERMANENT—CATEGORY A

FUSELAGE

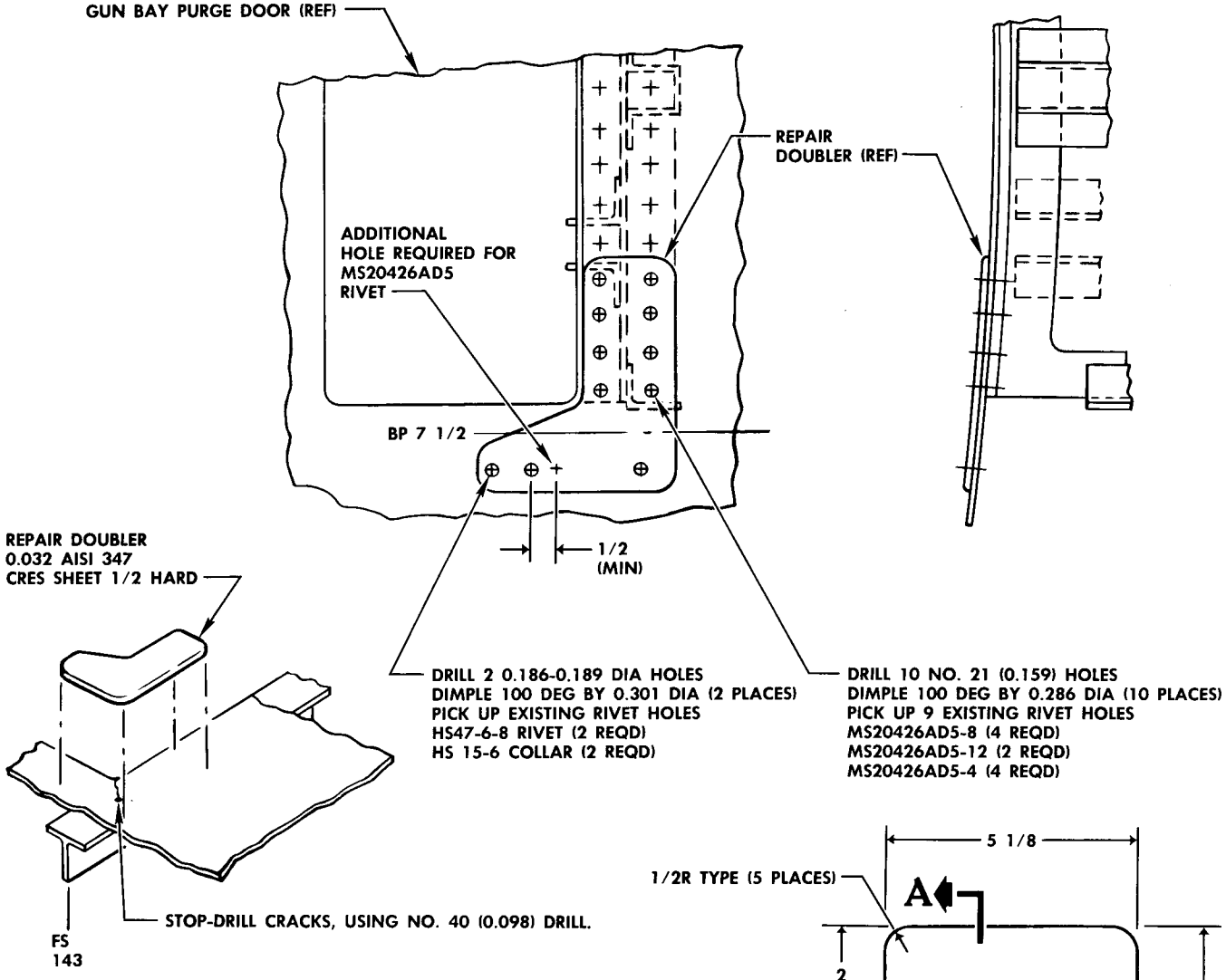


1. Stop-drill cracks with No. 40 (0.098) drill.
2. Make repair doubler from 0.032-inch AISI 347 stainless steel sheet as indicated.
3. Drill and dimple repair doubler to match existing holes. Drill and dimple for added fasteners.
4. Apply two coats of zinc-chromate primer (MIL-P-8585) to entire surface of repair doubler. The first coat should be allowed to dry before applying the second.
5. Attach doubler in place, using indicated rivets and locations.

Figure 4-9C. Gun Bay Purge Door Cutout Repair for Two Corner Cracks

PERMANENT—CATEGORY A

FUSELAGE



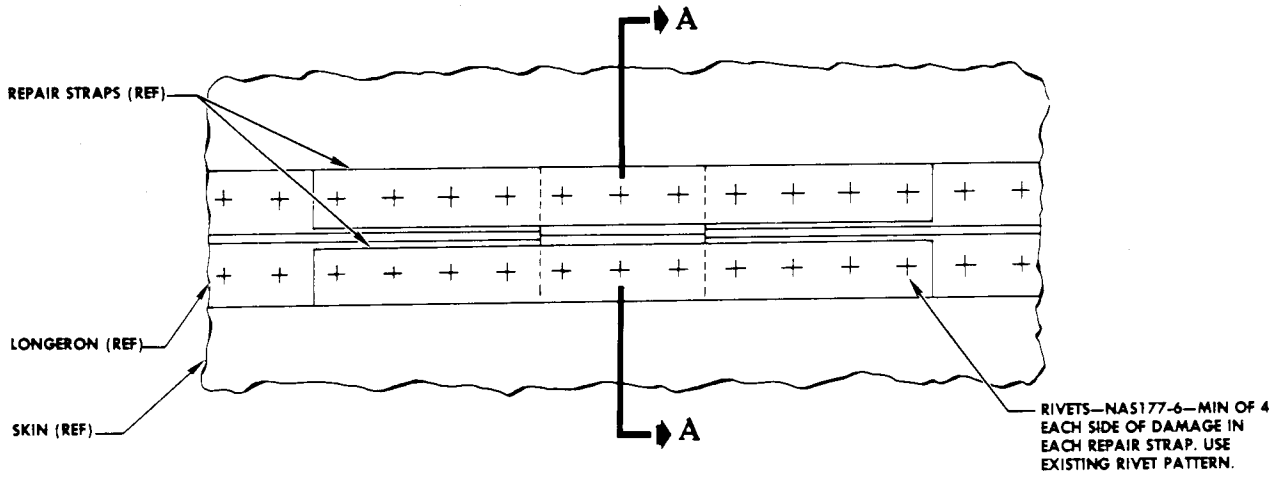
1. Stop-drill cracks with No. 40 (0.098) drill.
2. Make repair doubler from 0.032-inch AISI 347 stainless steel sheet as indicated.
3. Drill and dimple repair doubler to match existing holes. Drill and dimple for added fasteners.
4. Apply two coats of zinc-chromate primer (MIL-P-8585) to entire surface of repair doubler. The first coat should be allowed to dry before applying the second.
5. Attach doubler in place, using indicated rivets and locations.

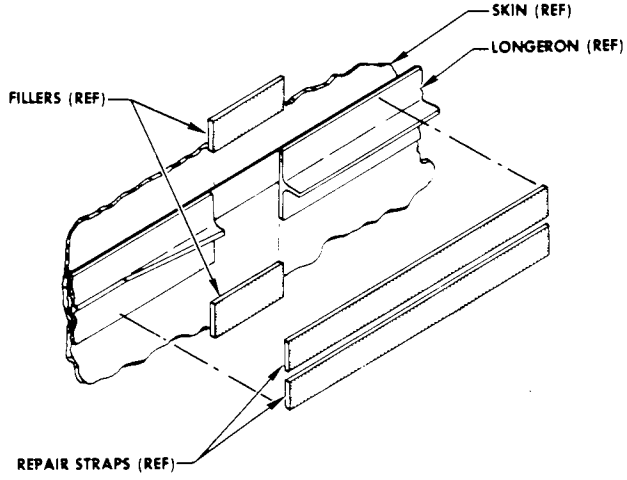
F-100D-3-31-152

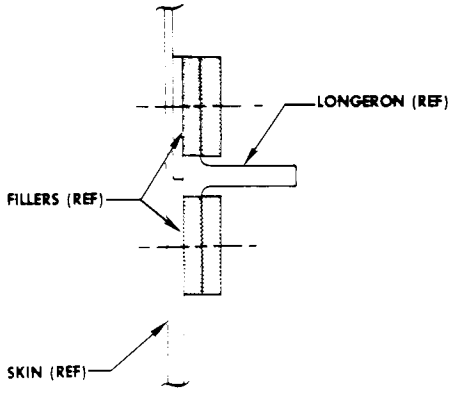
Figure 4-9D. Gun Bay Purge Door Cutout Repair for One Corner Crack

PERMANENT CATEGORY A

FUSELAGE







SECTION A-A

1. Trim out damage area midway between existing rivets.
2. Make fillers using same material and gage as longeron.
3. Make repair straps using 0.190-inch 7075-T6 clad or 0.125-inch corrosion-resistant steel, one-half hard (Specification MIL-S-5059). The length should be about 7 inches plus the width of the damage.
4. Assemble and rivet as shown.

NOTE When damage extends through intersection of frame and longeron, replace with original longeron or equivalent member, and splice between frames.

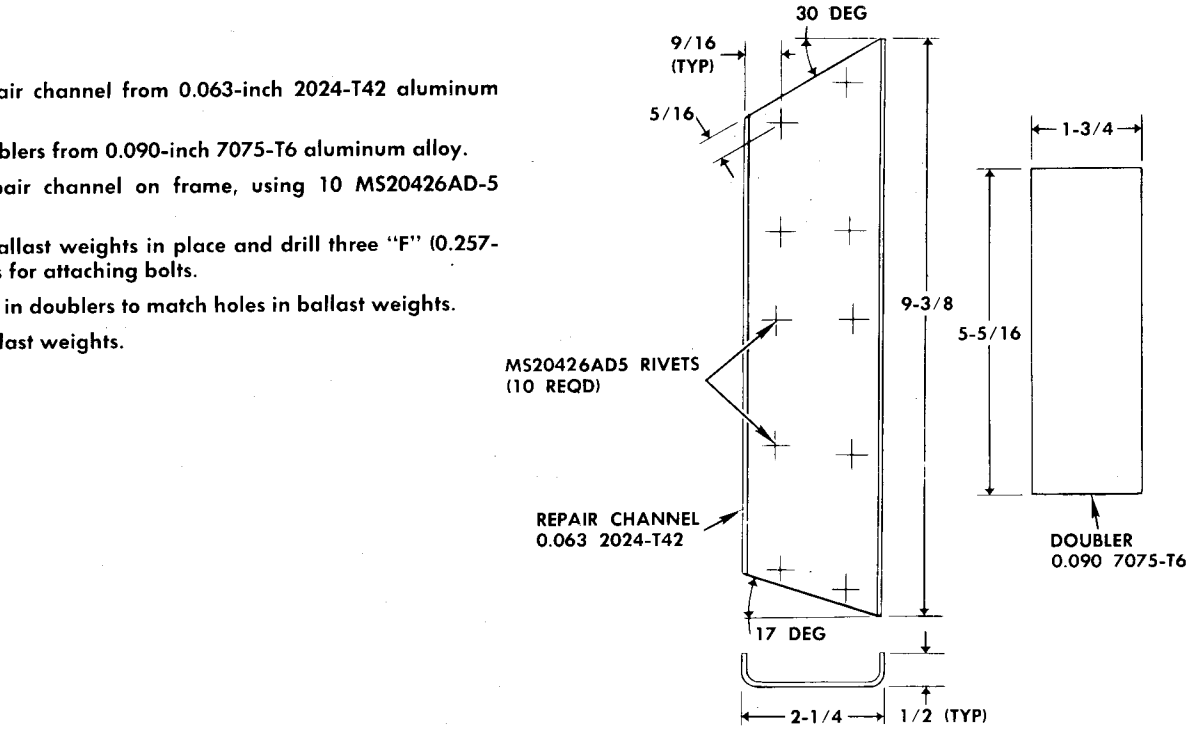
F-100D-3-31-38A

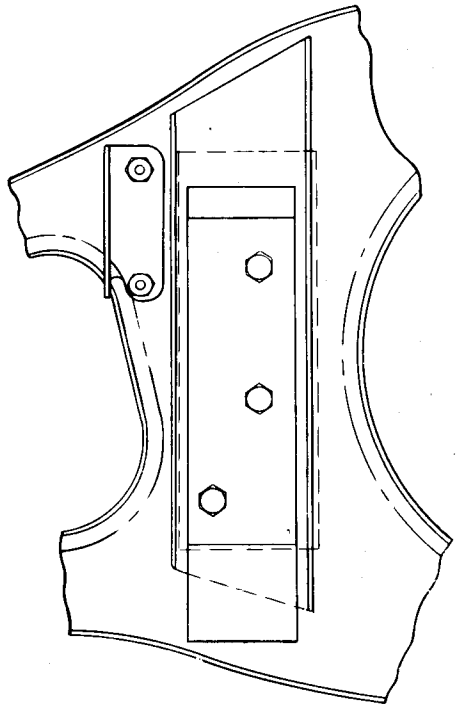
Figure 4-10. Lower Longeron—Station 36 to 89

PERMANENT—CATEGORY A

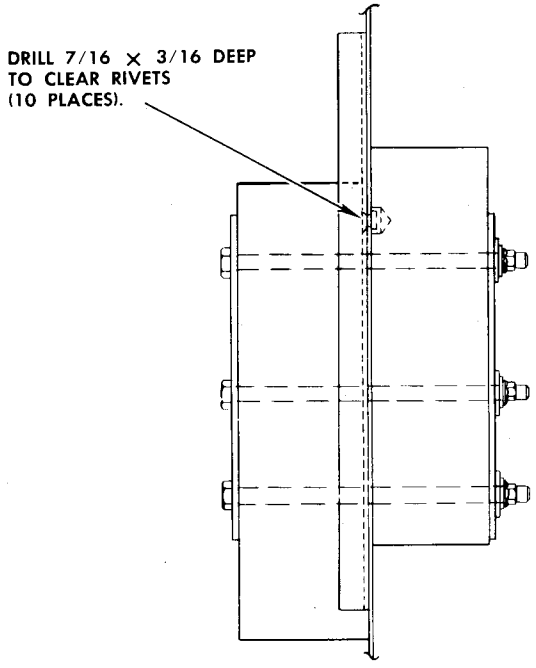
FUSELAGE

1. Make repair channel from 0.063-inch 2024-T42 aluminum alloy.
2. Make doublers from 0.090-inch 7075-T6 aluminum alloy.
3. Install repair channel on frame, using 10 MS20426AD-5 rivets.
4. Position ballast weights in place and drill three "F" (0.257-inch) holes for attaching bolts.
5. Drill holes in doublers to match holes in ballast weights.
6. Install ballast weights.





TYPICAL INSTALLATION
FUSELAGE STA 31.5 AND 36.7.
THIS REPAIR IS ALSO APPLICABLE
TO OTHER BALLAST INSTALLATIONS
IN FORWARD FUSELAGE.

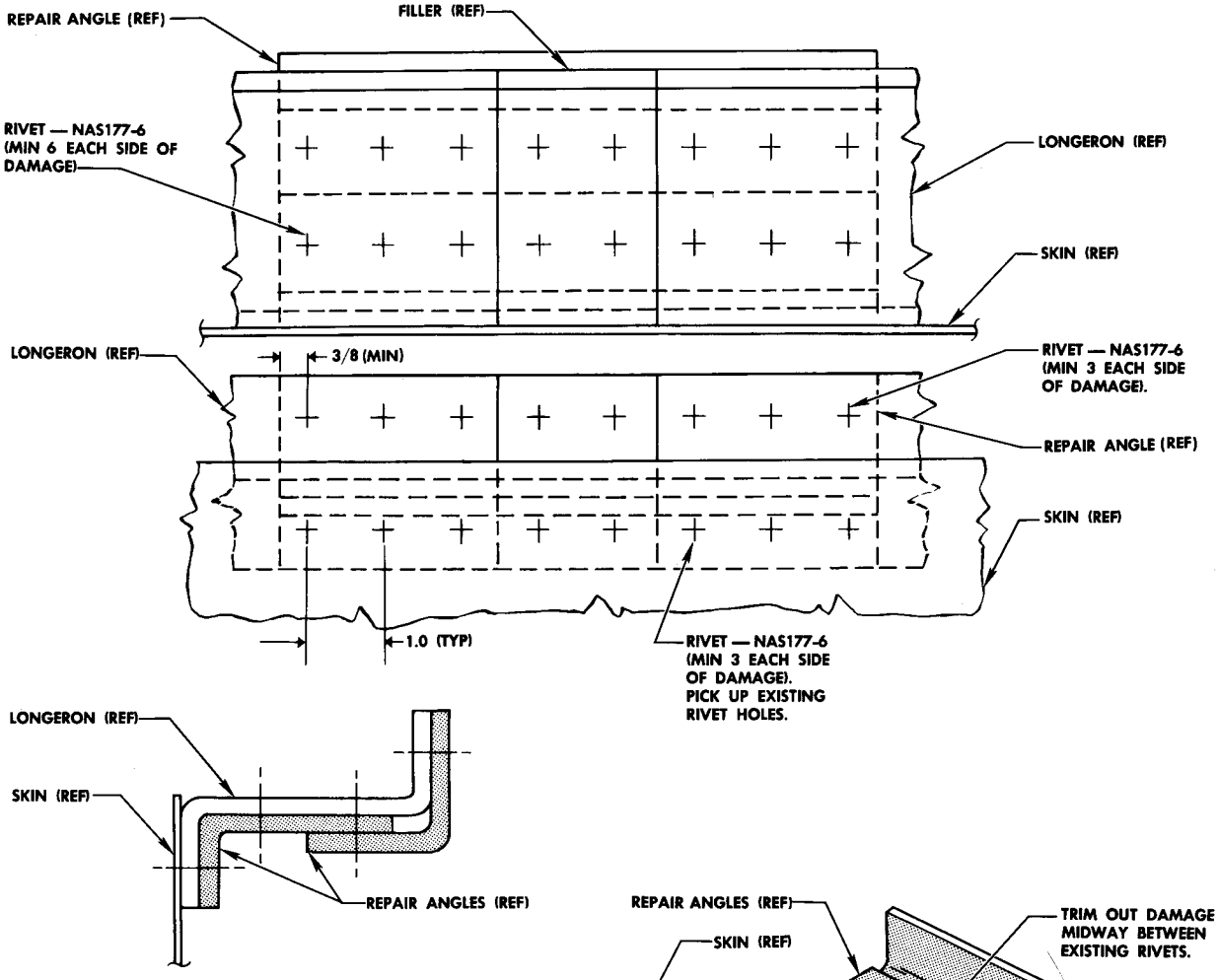


100D-3-31-165

Figure 4-10A. Ballast Weight Installation Repair

PERMANENT—CATEGORY A

FUSELAGE



1. Trim out damaged area midway between existing rivets.
2. Make filler to match removed section of longeron using 0.071-inch 7075 clad material heat-treated to T6 condition after forming. Make repair angles, using 0.080-inch 7075 clad material heat-treated to T6 condition after forming. The length of the repair angles should be about 6 inches plus the length of the damage.
3. Assemble and rivet repair, using rivets as indicated.

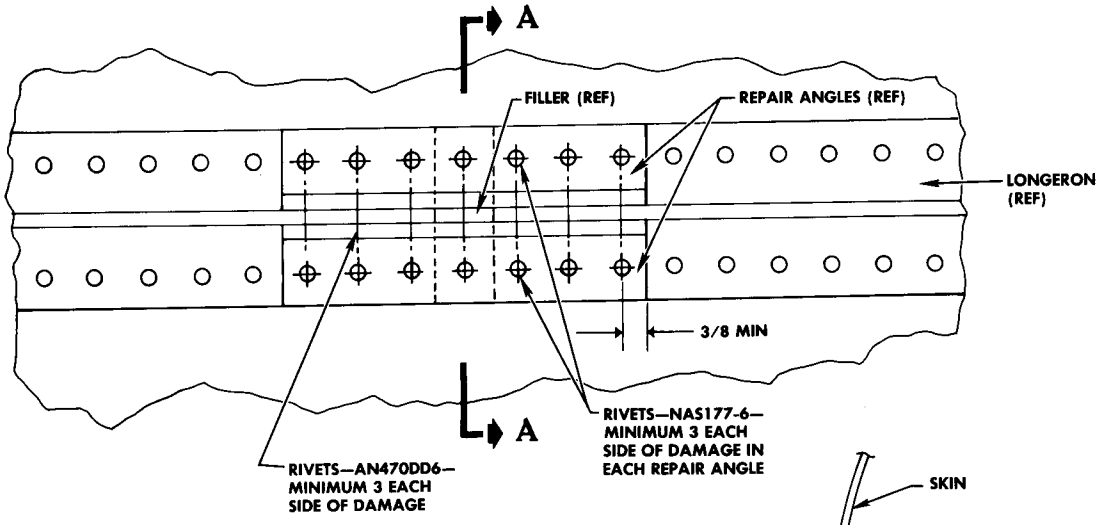
NOTE For damage extending through intersection of frame and longeron, replace with original longeron or equivalent, and splice between frames.

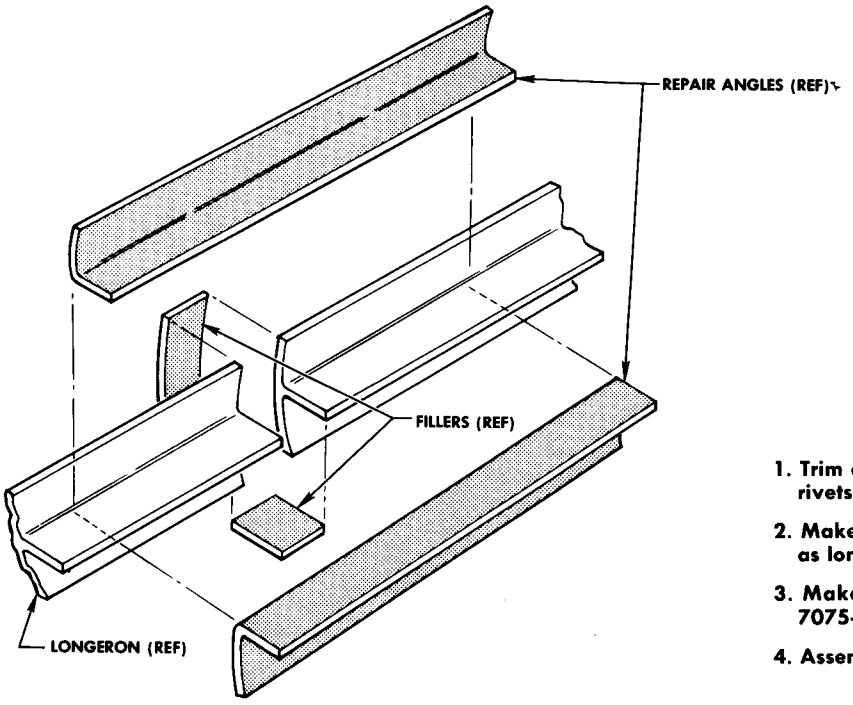
F-100D-3-31-46A

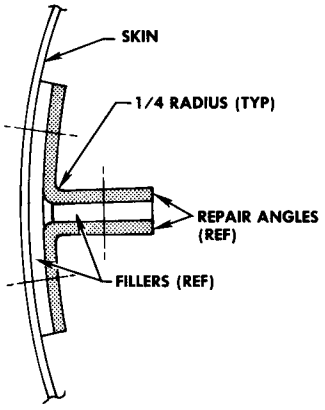
Figure 4-11. Upper Longeron—Station 36 to 106

PERMANENT—CATEGORY A

FUSELAGE







SECTION A-A

1. Trim out damage midway between existing rivets.
2. Make fillers using same material and gage as longeron.
3. Make two repair angles using 0.090-inch 7075-O clad material.
4. Assemble and rivet as shown.

NOTE Heat-treat repair angles to T6 condition after forming. The repair angles should be about 6 inches plus the width of the damaged area.

F-100D-3-31-39A

Figure 4-12. Lower Longeron—Station 89 to 106

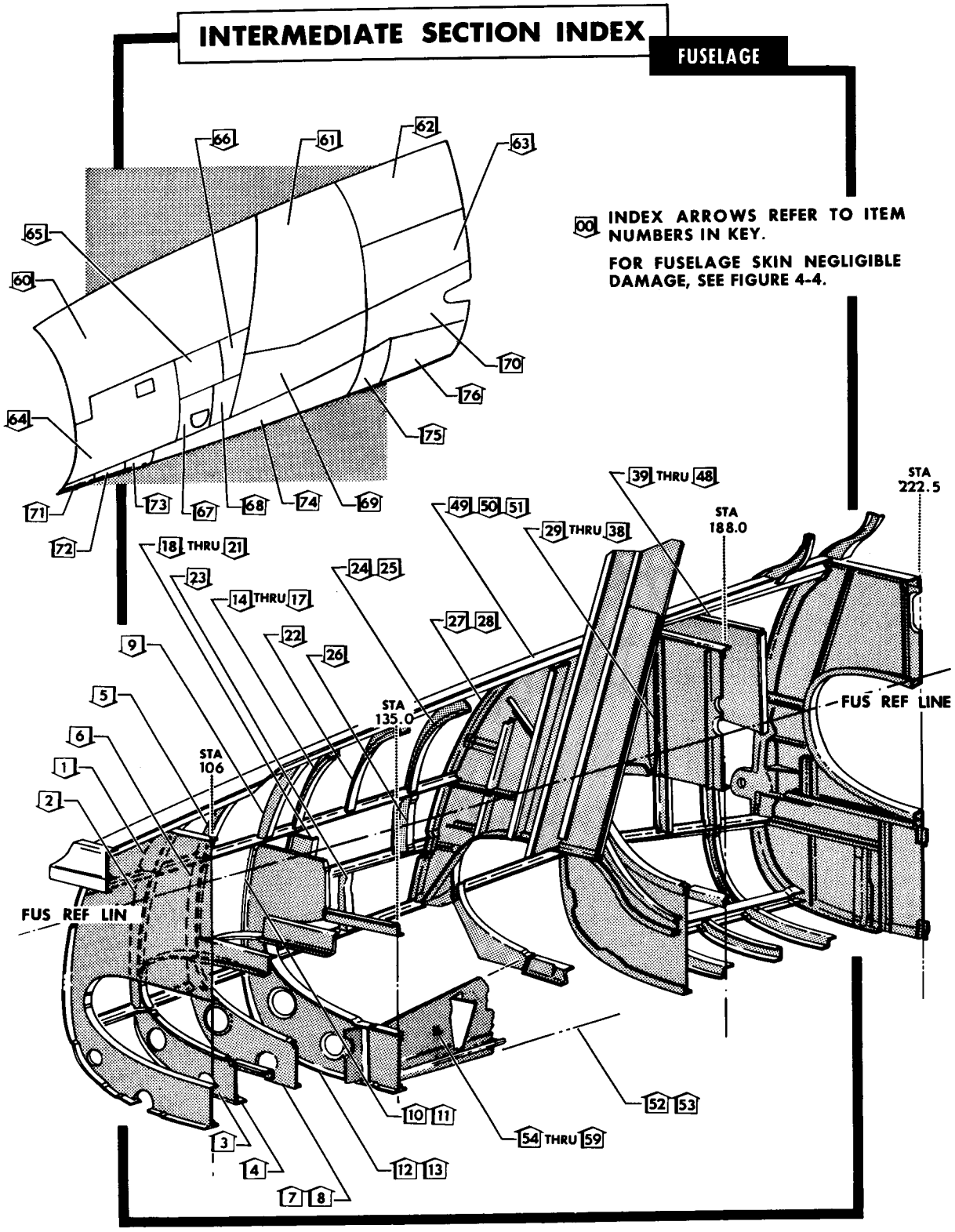


Figure 4-13. Forward Intermediate Section Repair Index—F-100D Airplane (Sheet 1 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31116	Frame Assy—Upper Ctr	Sta 116.125	0.051	7075-T6 Clad	
2	192-31119	Frame Assy—Side	Sta 116.125	0.050 0.063	7075-T6 Clad	
3	223-31125-3	Frame Assy—Lower	Sta 116.125 (Web)	0.063	7075-T6 Clad	4-30
4	223-31125-5	Frame Assy—Lower	Sta 116.125	AND 10134-2009	7075-T6 Clad	4-30
5	192-31126	Frame—Upper	Sta 126.250	0.050	7075-T6 Clad	
6	192-31139	Frame—Side	Sta 126.250	0.050	7075-T6 Clad	
7	192-31140-3	Frame Assy—Lower	Sta 126.250 (Web)	0.063	2024-T4 Clad	4-30
8	192-31140-5	Frame Assy—Lower	Sta 126.250	34610	7075-T6 Extr	4-30
9	223-31115	Frame—Upper	Sta 135		14S-T6 Forging	
10	223-31143	Bulkhead Assy	Sta 135 (Web)	0.125	7075-T6 Clad	4-30
11	223-31298	Frame—Intermediate	Sta 135	4E578	7075-T6 Extr	
12	223-31299-1	Frame—Lower	Sta 135	AND 10136-2402	7075-T6 Extr	4-30
13	223-31299-5	Frame—Lower	Sta 135	0.050	2024-T4 Clad	4-30
14	223-31136	Frame Assy	Sta 143	0.071	7075-T6 Clad	4-30
15	223-31137-3	Frame Assy—Side	Sta 143	0.063	7075-T6 Clad	
16	223-31137-5	Frame Assy—Side	Sta 143	4E347	7075-T6 Extr	
17	223-31138	Frame Assy—Lower	Sta 143	10136-2005	2024-T42 Extr	4-30
18	223-31139-3	Web Assy—Lower Ctr	Sta 143	0.063	2024-T42 Extr	4-30
19	223-31139-5	Web Assy—Lower	Sta 143	0.063	2024-T42 Extr	4-30
20	223-31139-7	Web Assy—Lower	Sta 143	10136-2005	2024-T42 Extr	4-30
21	223-31139-9	Web Assy—Lower	Sta 143	1E129	2024-T42 Extr	4-30
22	223-31151	Frame Assy—Upper	Sta 152	0.050	7075-T6 Clad	
23	223-31152	Frame—Intermediate	Sta 152	0.050	7075-T6 Clad	
24	223-31161-3	Frame Assy—Upper	Sta 162	0.080	7075-T6 Clad	
25	223-31161-5	Frame Assy—Upper	Sta 162	1E68	7075-T6 Extr	
26	223-31162	Frame—Intermediate	Sta 162	4E183	7075-T6 Extr	
27	223-31663-3	Beam Assy—Duct Side Canted	Sta 168.118	0.090	CRES (17-7PH)	
28	223-31663-5	Beam Assy—Duct Side Canted	Sta 168.118	0.090	CRES (17-7PH)	
29	223-31196-3	Bulkhead Assy—Ctr	Sta 188 (Web)	0.040	7075-T6 Clad	
30	223-31196-5	Bulkhead Assy—Ctr	Sta 188	22885	7075-T6 Extr	
31	223-31196-7	Bulkhead Assy—Ctr	Sta 188	0.063	7075-T6 Clad	
32	223-31196-9	Bulkhead Assy—Ctr	Sta 188	0.090	7075-T6 Clad	
33	223-31196-11	Bulkhead Assy—Ctr	Sta 188	12871	7075-T6 Extr	
34	223-31196-13	Bulkhead Assy—Ctr	Sta 188	3E52	7075-T6 Extr	
35	223-31196-43	Bulkhead Assy—Ctr	Sta 188	0.063	7075-T6 Clad	
36	223-31195-3	Bulkhead Assy—Side	Sta 188 (Web)	0.063	7075-T6 Clad	
37	223-31195-5	Bulkhead Assy—Side	Sta 188 (Web)	0.063	7075-T6 Clad	
38	223-31195-13	Bulkhead Assy—Side	Sta 188	1E321	7075-T6 Extr	
39	223-31232-3	Frame Assy—Upper LH	Sta 196 (Web)	0.040	2024-T4 Clad	
40	223-31232-5	Frame Assy—Upper LH	Sta 196	78F	2024-T42 Extr	
41	223-31232-7	Frame Assy—Upper LH	Sta 196	Alcoa 22969	2024-T42 Extr	
42	223-31232-9	Frame Assy—Upper LH	Sta 196	78J	2024-T42 Extr	
43	223-31233-25	Frame Assy—Upper RH	Sta 196 (Web)	0.063	2024-T4 Clad	
44	223-31233-17	Frame Assy—Upper RH	Sta 196	78J	2024-T42 Extr	
45	223-31233-19	Frame Assy—Upper RH	Sta 196	78F	2024-T42 Extr	
46	223-31206	Frame—Side	Sta 196	0.040	7075-T6 Clad	
47	223-31207	Frame—Lower	Sta 196	4E183	7075-T6 Extr	4-30
48	223-31208	Frame Assy—Bottom	Sta 196	0.064	7075-T6 Clad	
49	223-31112-3	Upper Longeron Assy	Sta 89 to 222.5 (Web)	0.064	7075-T6 Clad	4-14
50	223-31112-5	Upper Longeron Assy	Sta 89 to 222.5	1E391	7075-T6 Extr	4-14
51	223-3112-7 & -11	Upper Longeron Assy	Sta 89 to 222.5	1E392	7075-T6 Extr	4-14
52	223-31147-3	Beam Assy—Gun Door	Sta 135 to 188	0.064	7075-T6 Clad	
53	223-31147-5	Beam Assy—Gun Door	Sta 135 to 188	6E567	7075-T6 Extr	
54	223-31175-3	Beam Assy—Nose Gear LH	Sta 135 to 222.5 (Web)	0.071	7075-T6 Clad	
55	223-31175-5	Beam Assy—Nose Gear LH	Sta 135 to 222.5 (Web)	0.032	7075-T6 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-13 (Sheet 1 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
56	223-31175-7	Beam Assy—Nose Gear LH	Sta 135 to 222.5	1E408	7075-T6 Extr	4-30
57	223-31176-3	Beam Assy—Nose Gear RH	Sta 135 to 222.5 (Web)	0.071	7075-T6 Clad	
58	223-31176-5	Beam Assy—Nose Gear RH	Sta 135 to 222.5 (Web)	0.032	7075-T6 Clad	
59	223-31176-7	Beam Assy—Nose Gear RH	Sta 135 to 222.5	1E408	7075-T6 Extr	4-30
60	223-31105	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
61	223-31014	Door—Ammunition Access		0.032	7075-T6 Clad	
62	223-31015	Door—Battery and Electrical		0.032	7075-T6 Clad	
63	223-31016	Door—General Access		0.032	7075-T6 Clad	
64	223-31034	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
65	223-31017	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
66	223-31105	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
67	192-31060	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
68	223-31095	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
69	223-31155	Skin—Side (Chem-Milled)		0.375	7075-T6 Alloy	
70	223-31107	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
71	223-31113	Skin—Lower		0.050	7075-T6 Clad	4-31, 4-32, 4-33
72	223-31047	Door—Gun Barrel Stabilizer		0.063	7075-T6 Clad	
73	223-31113	Skin—Lower		0.050	7075-T6 Clad	4-31, 4-32, 4-33
74	223-31013	Door—Gun Bay Access		0.040	2024-T4 Clad	
75	223-31107	Skin		0.032	7075-T6 Clad	4-31, 4-32, 4-33
76	223-31019	Door—Link Compartment		0.032	2024-T4 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-13 (Sheet 1 of 3 Continued)

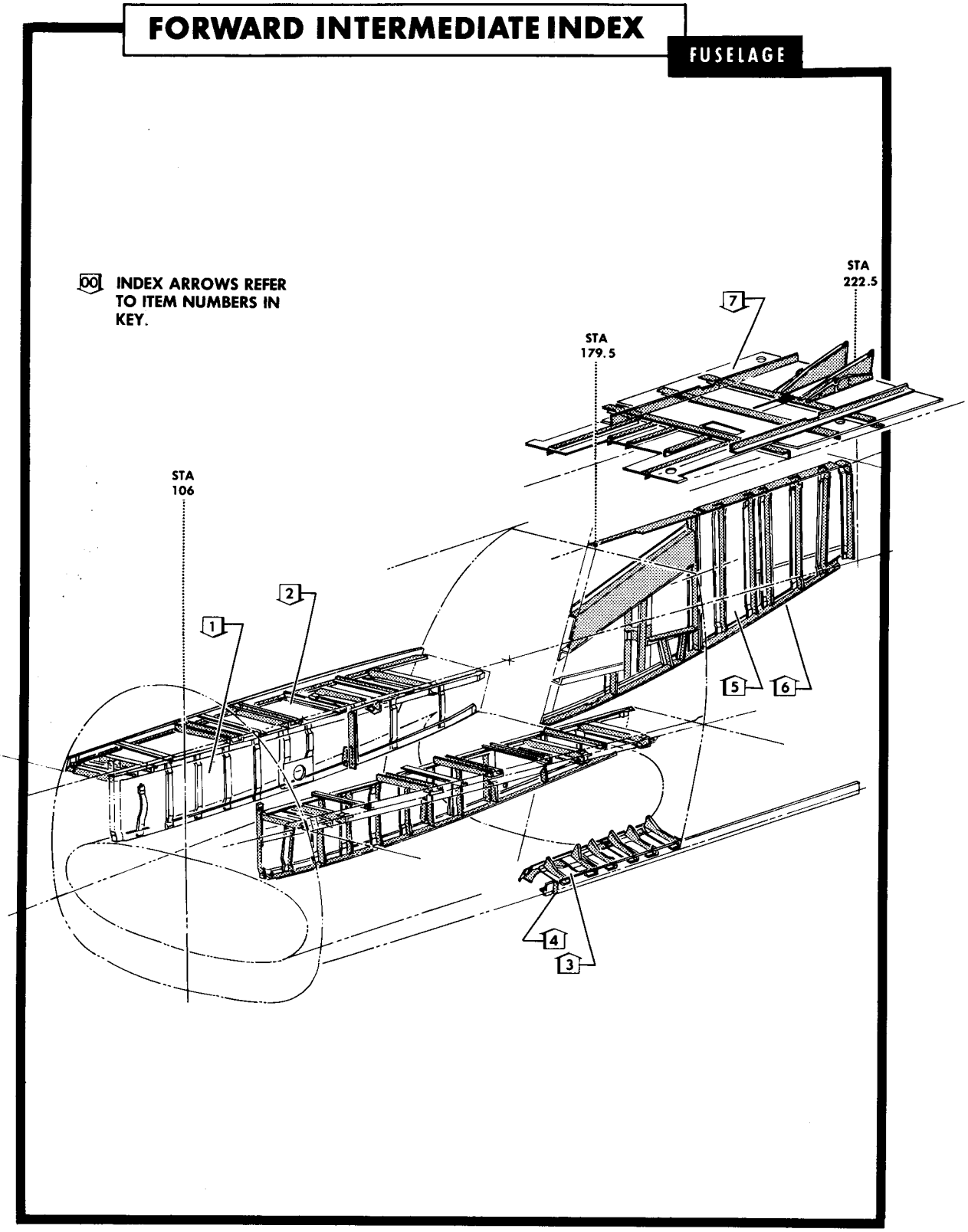


Figure 4-13. Forward Intermediate Section Repair Index—F-100D Airplane (Sheet 2 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31726	Lower Longeron, Web	0.040	2024-T4 Clad	
2	223-31153	Horizontal Pressure, Web	0.040	7075-T6 Clad	
3	223-31159-3	Web Assembly, Horizontal	0.040	2024-T4 Clad	
4	223-31159-5	Web Assembly, Horizontal	4E309	2024-T42 Extr	
5	223-31146-3	Web Assembly, Vertical	0.040	7075-T6 Clad	
6	223-31146-11	Web Assembly, Vertical	1E214	7075-T6 Extr	
7	223-31160	Web Assembly, Horizontal	0.032	2024-T4 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-13 (Sheet 2 of 3)

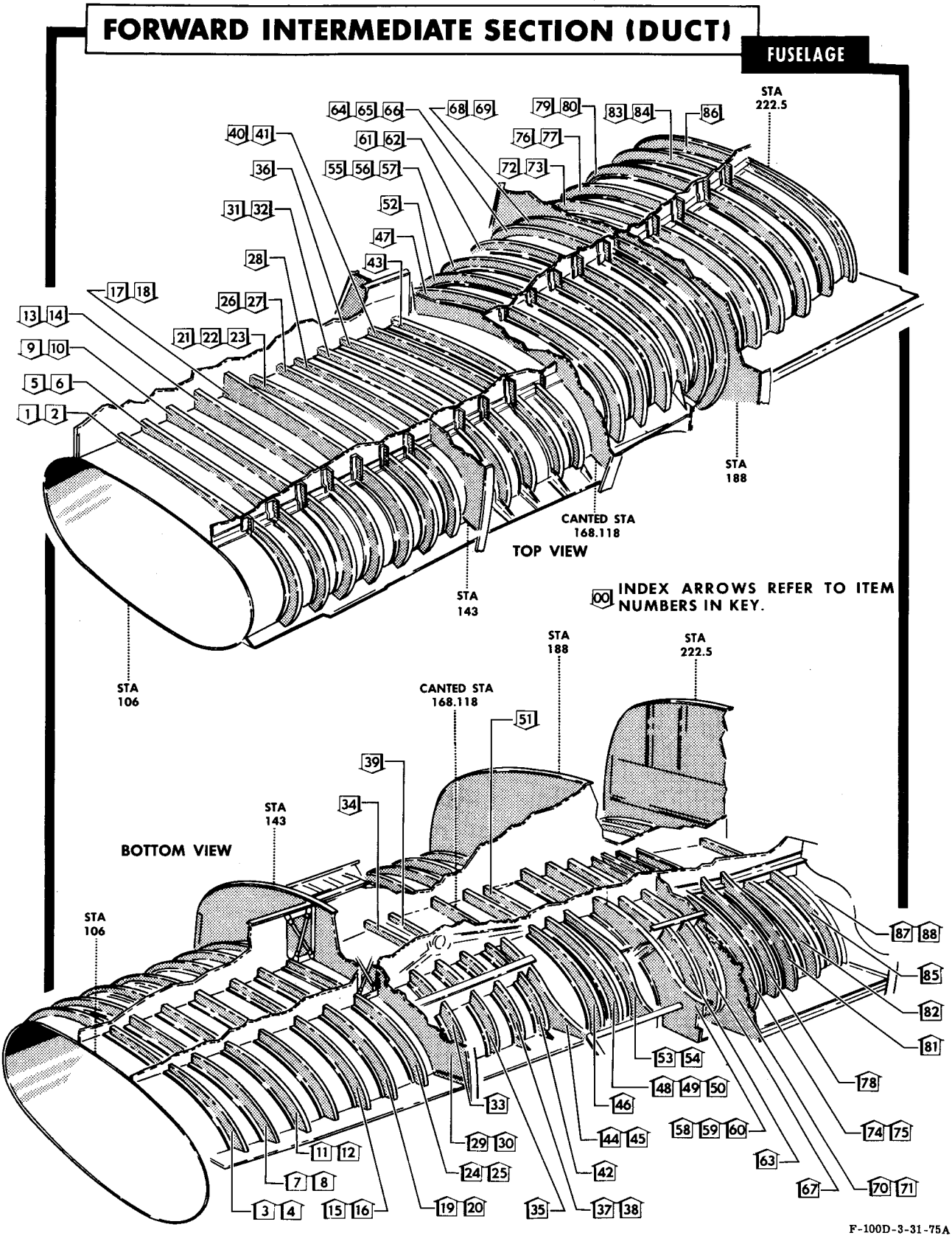


Figure 4-13. Forward Intermediate Section Repair Index—F-100D Airplane (Sheet 3 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-316111	Duct Frame—Upper	Sta 111	6E496	7075-T6 Extr	
2	192-316112	Duct Frame—Side	Sta 111	6E524	2024-T42 Extr	
3	223-31712	Duct Frame—Lower	Sta 111	6E524	7075-T6 Extr	
4	192-316114	Duct Frame—Lower Ctr	Sta 111	0.050	7075-T6 Clad	
5	223-31654	Duct Frame—Upper Ctr	Sta 116.125	0.050	7075-T6 Clad	
6	192-316117	Duct Frame—Upper Side	Sta 116.125	6E524	2024-T42 Extr	
7	192-316118	Duct Frame—Lower Side	Sta 116.125	0.063	2024-T4 Clad	
8	223-31685	Duct Frame—Lower Ctr	Sta 116.125	0.050	7075-T6 Clad	
9	192-316121	Duct Frame—Upper	Sta 121.125	0.050	7075-T6 Clad	
10	192-316122	Duct Frame—Upper Side	Sta 121.125	6E524	2024-T42 Extr	
11	192-316123	Duct Frame—Lower Side	Sta 121.125	6E524	2024-T42 Extr	
12	192-316124	Duct Frame—Lower Ctr	Sta 121.125	0.050	7075-T6 Clad	
13	192-316126	Duct Frame—Lower Ctr	Sta 125.5	0.050	7075-T6 Clad	
14	192-316127	Duct Frame—Upper Side	Sta 126.25	6E524	2024-T42 Extr	
15	192-316128	Duct Frame—Lower Side	Sta 126.25	0.063	2024-T4 Clad	
16	192-316129	Duct Frame—Lower Ctr	Sta 126.25	0.050	7075-T6 Clad	
17	223-31651	Duct Frame—Upper Ctr	Sta 131.375	0.050	7075-T6 Clad	
18	192-316132	Duct Frame—Upper Side	Sta 131.375	6E524	2024-T42 Extr	
19	192-316133	Duct Frame—Lower	Sta 131.375	6E524	2024-T42 Extr	
20	192-316134	Duct Frame—Lower Ctr	Sta 131.375	0.050	7075-T6 Clad	
21	223-317139-3	Duct Frame—Upper	Sta 135	0.050	7075-T6 Clad	
22	223-317139-5	Duct Frame—Upper	Sta 135	1E520	7075-T6 Extr	4-30
23	223-31625	Duct Frame—Side	Sta 135	34610	7075-T6 Extr	
24	223-31623	Duct Frame—Lower	Sta 135	AND10132-2406	7075-T6 Extr	
25	223-31115	Frame—Upper	Sta 135		14ST-6 Forging	
26	223-31644	Duct Frame—Upper Side	Sta 139	6E524	2024-T42 Extr	
27	192-316140	Duct Frame—Upper Ctr	Sta 140.5	6E496	7075-T6 Extr	
28	223-31657	Duct Frame—Upper Side	Sta 143	0.071	2024-T4 Clad	
29	223-31655	Duct Frame—Lower Side	Sta 143	0.063	2024-T4 Clad	
30	223-31656	Duct Frame—Lower Ctr		6E524	2024-T42 Extr	
31	192-316148	Duct Frame—Upper Side	Sta 147.5	0.050	7075-T6 Clad	
32	223-31658	Duct Frame—Upper Side	Sta 147.5	6E524	7075-T6 Extr	
33	223-31659	Duct Frame—Lower Side	Sta 147.5	0.063	2024-T4 Clad	
34	192-316150	Duct Frame—Lower Ctr	Sta 149	0.050	7075-T6 Clad	
35	223-31682	Duct Frame—Lower Inbd	Sta 150.250	AND101394-2003	2024-T42 Extr	
36	223-31660	Duct Frame—Upper Side	Sta 152	6E524	2024-T42 Extr	
37	223-31679	Duct Frame—Lower Inbd	Sta 152	6E524	7075-T6 Extr	
38	223-31667	Duct Frame—Lower Side	Sta 152.281	0.050	7075-T6 Clad	
39	192-316160	Duct Frame—Lower Ctr	Sta 154.5	0.063	7075-T6 Clad	
40	223-31641	Duct Frame—Upper	Sta 156.5	0.100	7075-T6 Clad	
41	223-31665	Duct Frame—Upper Side	Sta 157.125	6E524	2024-T42 Extr	
42	223-31645	Duct Frame—Lower Side	Sta 157.125	6E524	2024-T42 Extr	
43	192-316163	Duct Frame—Upper Side	Sta 162	6E524	2024-T42 Extr	
44	223-31663-3	Beam Assy—Duct Side—Canted	Sta 168.118	0.090	CRES-(17-7PH)	
45	223-31663-5	Beam Assy—Duct Side—Canted	Sta 168.118	0.090	CRES-(17-7PH)	
46	223-31675	Duct Frame—Lower	Sta 171	2E279	2024-T42 Extr	
47	223-31680	Duct Frame—Upper Canted	Sta 173.952	0.063	7075-T6 Clad	
48	223-31672	Duct Frame—Lower Side	Sta 175.250	0.063	2024-T4 Clad	
49	223-31677	Duct Frame—Lower Side	Sta 175.250	0.125	2024-T4 Clad	
50	223-31621	Duct Frame Assy—Lower Canted	Sta 176.968	0.063	7075-T6 Clad	
51	223-31684	Duct Frame	Sta 177.034	0.063	7075-T6 Clad	
52	223-31640	Duct Frame	Sta 180	0.063	7075-T6 Clad	
53	223-31681	Duct Frame	Sta 180	0.063	2024-T4 Clad	
54	223-31642	Duct Frame	Sta 180	0.063	2024-T4 Clad	
55	223-31678	Duct Frame—Canted	Sta 186.096	0.063	7075-T6 Clad	
56	223-31689	Duct Frame—Canted	Sta 186.096	0.063	2024-T4 Clad	
57	223-31691	Duct Frame—Lower Inbd Canted	Sta 186.096	0.050	2024-T4 Clad	
58	223-31696	Duct Frame (Cap Upper)	Sta 188	1E456	7075-T6 Extr	
59	223-31695	Duct Frame—Side	Sta 188	1E456	7075-T6 Extr	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-13 (Sheet 3 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
60	223-316197	Duct Frame	Sta 188			
61	223-31692	Duct Frame—Upper RH	Sta 192	0.063	7075-T6 Clad	
62	223-31693	Duct Frame—Upper LH	Sta 192	0.063	7075-T6 Clad	
63	223-31699	Duct Frame—Lower	Sta 192	0.050	7075-T6 Clad	
64	223-31650	Duct Frame—Lower	Sta 195	0.063	2024-T4 Clad	
65	192-31676	Duct Frame—Upper RH	Sta 196	0.063	7075-T6 Clad	
66	223-31687	Duct Frame—Upper LH	Sta 196	0.063	7075-T6 Clad	
67	223-31674	Duct Frame—Lower	Sta 196	0.050	7075-T6 Clad	
68	192-316201	Duct—RH	Sta 200.5	0.050	7075-T6 Clad	
69	192-316200	Duct—LH	Sta 200.5	0.050	7075-T6 Clad	
70	192-316203	Duct	Sta 200.5	0.064	7075-T6 Clad	
71	192-316202	Duct	Sta 200.5	0.050	7075-T6 Clad	
72	192-316204	Duct	Sta 204.5	0.090	7075-T6 Clad	
73	192-316206	Duct RH	Sta 205.750	0.050	7075-T6 Clad	
74	192-316205	Duct LH	Sta 205.750	0.050	7075-T6 Clad	
75	192-316207	Duct	Sta 205.750	0.050	7075-T6 Clad	
76	192-31610	Duct RH	Sta 209.250	0.050	7075-T6 Clad	
77	192-31609	Duct LH	Sta 209.250	0.050	7075-T6 Clad	
78	192-316211	Duct	Sta 209.250	0.050	7075-T6 Clad	
79	192-316214	Duct RH	Sta 213.750	0.050	7075-T6 Clad	
80	192-316213	Duct LH	Sta 213.750	0.050	7075-T6 Clad	
81	192-316215	Duct	Sta 213.750	0.050	7075-T6 Clad	
82	192-316216	Duct	Sta 216			
83	192-316219	Duct RH	Sta 218	0.050	7075-T6 Clad	
84	192-316218	Duct LH	Sta 218	0.050	7075-T6 Clad	
85	192-316220	Duct	Sta 218	0.050	7075-T6 Clad	
86	192-31269	Duct Angle—Upper	Sta 222.5	0.063	7075-T6 Clad	
87	192-31270	Duct Attach Angle—Lower	Sta 222.5	0.063	7075-T6 Clad	
88	192-316106	Duct Angle—Aft	Sta 221.125	0.090	2024-T4 Clad	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-13 (Sheet 3 of 3 Cont)

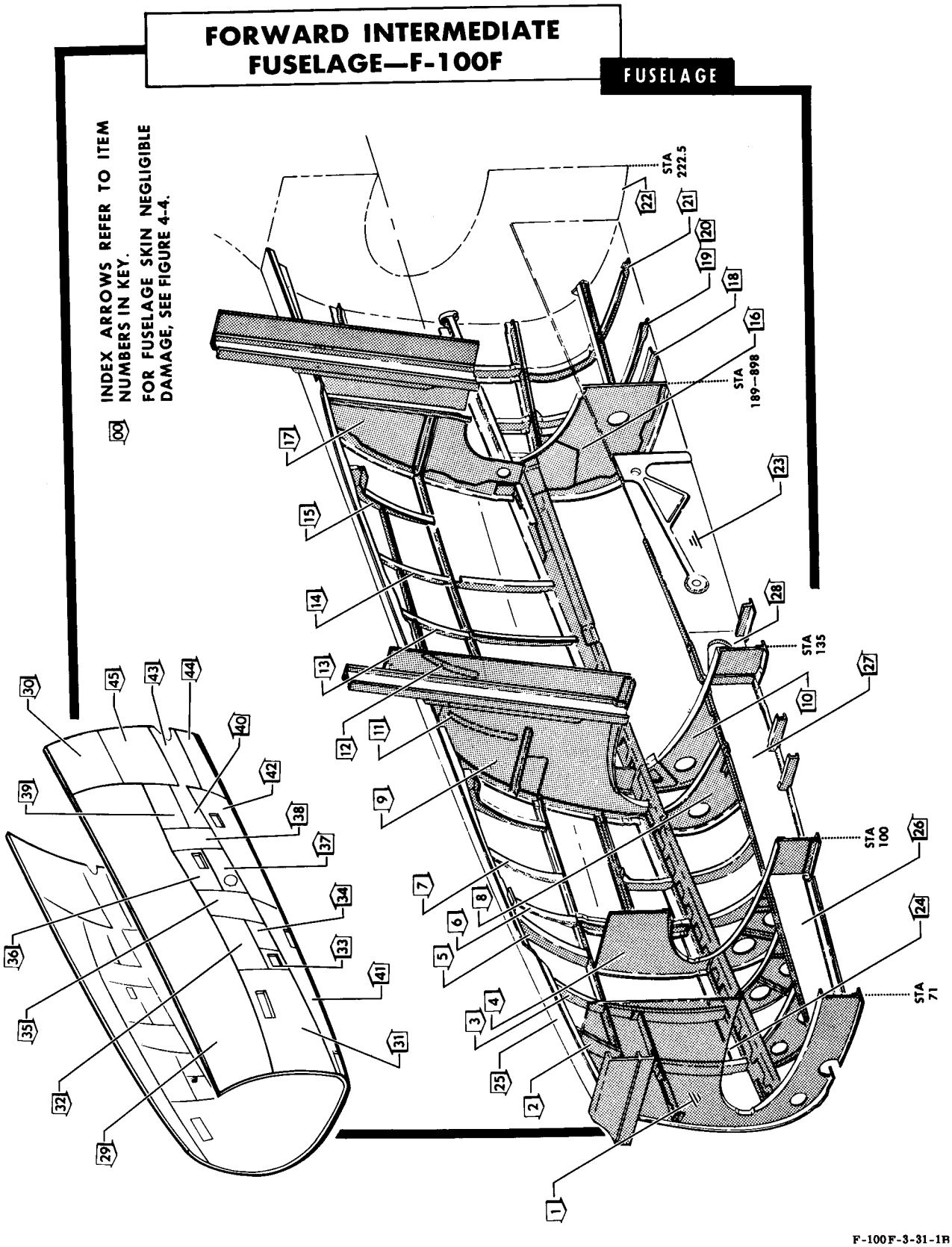


Figure 4-13A. Forward Intermediate Section Repair Index—F-100F Airplane (Sheet 1 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	243-31106	Bulkhead Assy	Sta 71	0.040	7075-T6 Clad	4-30
2	243-31125	Frame Assy	Sta 81.125	0.063	2024-T4 Clad	4-30
3	243-31140	Frame Assy	Sta 91.250	0.063	2024-T4 Clad	4-30
4	243-31216	Bulkhead Assy	Sta 100	0.100	7075-T6 Clad	4-30
5	243-31115	Frame Assy	Canted Sta 100.653	Forging	2014-T6 Clad	
6	243-31287	Frame	Sta 108	0.090	7075-T6 Clad	
7	243-31288	Frame—Lower Side	Sta 116.125	0.090	7075-T6 Clad	
8	243-31289	Bulkhead Assy	Sta 126.250	0.063	2024-T4 Clad	
9	243-31163	Bulkhead Assy	Canted Sta 133.118			4-30
10	243-31743	Bulkhead Assy	Sta 135	0.125	7075-T6 Clad	4-30
11	243-31166	Frame Assy	Sta 143	0.063	7075-T6 Clad	
12	243-31227	Frame—Side	Sta 154	0.050	7075-T6 Clad	
13	243-31273	Frame—Side	Sta 162	0.050	7075-T6 Clad	
14	243-31271	Frame—Side	Sta 172	0.050	7075-T6 Clad	
15	243-31275	Frame—Side	Sta 183.250	0.063	7075-T6 Clad	
16	243-31195	Bulkhead Assy	Sta 189.898	0.063	7075-T6 Clad	4-30
17	243-31173	Bulkhead Assy	Canted Sta 193.501	0.063	7075-T6 Clad	
18	243-31208	Frame Assy	Sta 196	0.063	7075-T6 Clad	
19	243-31205	Frame Assy	Sta 201.375	0.063	7075-T6 Clad	
20	243-31221	Frame	Sta 202.625	0.063	7075-T6 Clad	
21	243-31209	Frame Assy	Sta 212.250	0.063	7075-T6 Clad	
22	243-31211	Frame Assy	Sta 212.5	0.063	7075-T6 Clad	4-30
23	243-34117	Beam Assy	Sta 135 to 176.875	Forging	2014-T6 Alloy	
24	243-31124	Lower Longeron	Sta 71 to 222.5	2E362	7075-T6 Extr	
25	243-31112	Upper Longeron	Sta 54 to 222.5	0.090	7075-T6 Clad	4-14
26	243-31149	Beam Assy	Sta 71 to 100	0.040	7075-T6 Clad	
27	243-31177	Beam Assy	Sta 100 to 135	0.040	7075-T6 Clad	
28	243-31147	Beam Assy	Sta 135 to 189.898	0.063	7075-T6 Clad	
29	243-31105	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
30	243-31015	Door—Battery Access		0.032	2024-T4 Clad	
31	243-31034	Door—Surface Control and Hyd Brake Valve Access		0.050	2024-T4 Clad	
32	243-31017	Door—Flight and Throttle Control Access		0.040	7075-T6 Clad	
33	243-31105	Kick Step		0.040	7075-T6 Clad	
34	243-31095	Door—Lower Access		0.040	7075-T6 Clad	
35	243-31014	Door—Surface Control Access		0.040	7075-T6 Clad	
36	243-31030	Door—Heat and Vent Access		0.040	7075-T6 Clad	
37	243-31105	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
38	243-31047	Door—Surface Control Access		0.040	7075-T6 Clad	
39	243-31023	Door—Surface Control Access		0.040	7075-T6 Clad	
40	243-31278	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
41	243-31113	Skin—Bottom		0.050	7075-T6 Clad	4-31, 4-32, 4-33
42	243-31013	Door—Gun Access		0 375	7075-T6 (Bare)	
43	243-31107	Skin—Wing Root		0.063	7075-T6 Clad	4-31, 4-32, 4-33, 4-33A
44	243-31019	Door—General Access		0.032	2024-T4 Clad	
45	243-31016	Door—General Access		0.032	2024-T4 Clad	

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Key to Figure 4-13A (Sheet 1 of 3)

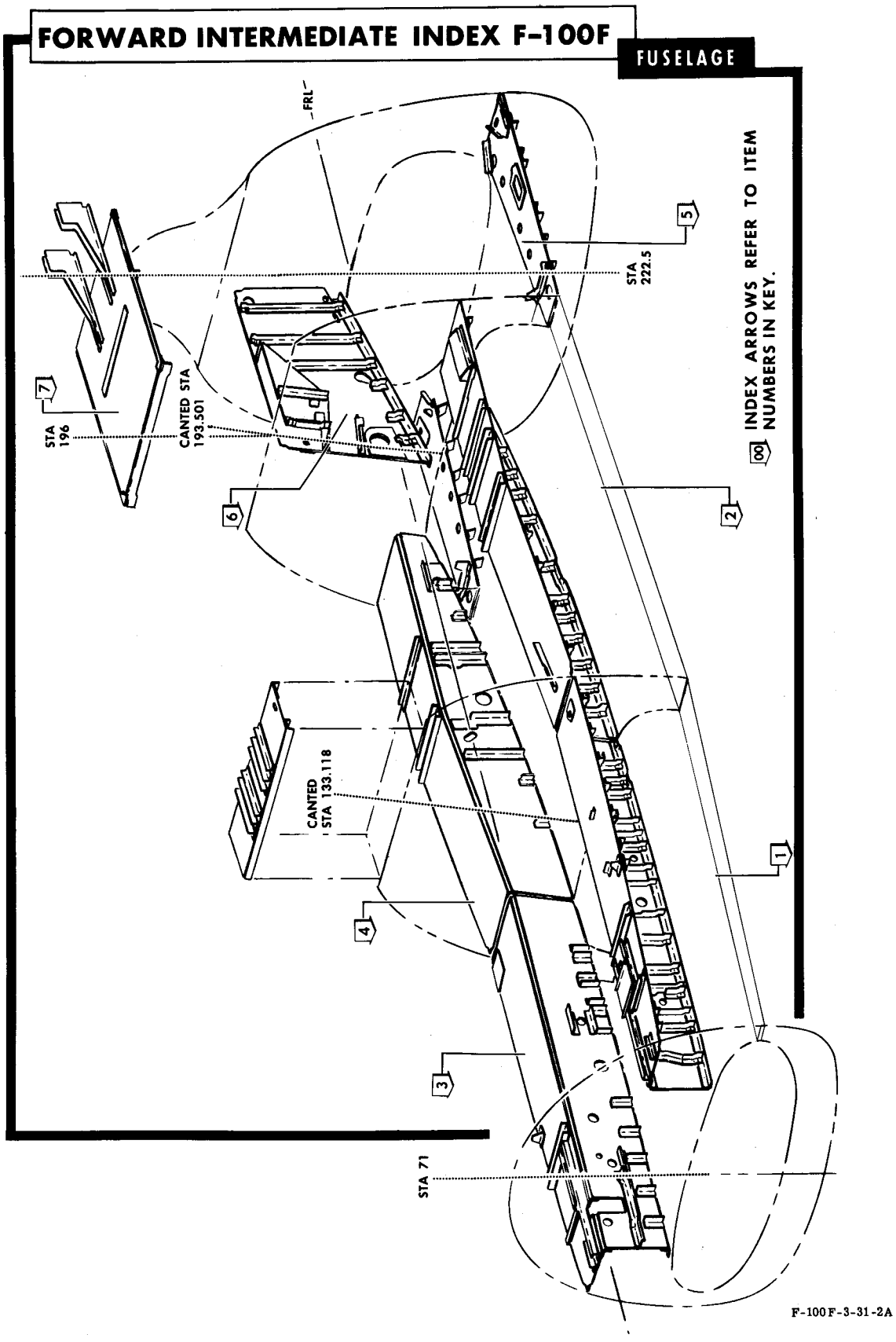

Figure 4-13A. Forward Intermediate Section Repair Index—F-100F Airplane (Sheet 2 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	243-31677	Duct Close-out Web Assy Sta 71-135			
2	243-31678	Duct Close-out Web Assy Sta 135-189.898			
3	243-31153	Hor Press. Web Assy Sta 71—Cant Sta 133.118			
4	243-31183	Hor Press. Web Assy— Canted Sta 133.118—Canted Sta 193.501			
5	243-31160	Web Assy Canted Sta 193.501-222.5			
6	243-31145	Vert Shear Web Canted Sta 193.501-222.5			
7	243-31158	Deck—Canted Sta 193.501-231			

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 4-13A (Sheet 2 of 3)

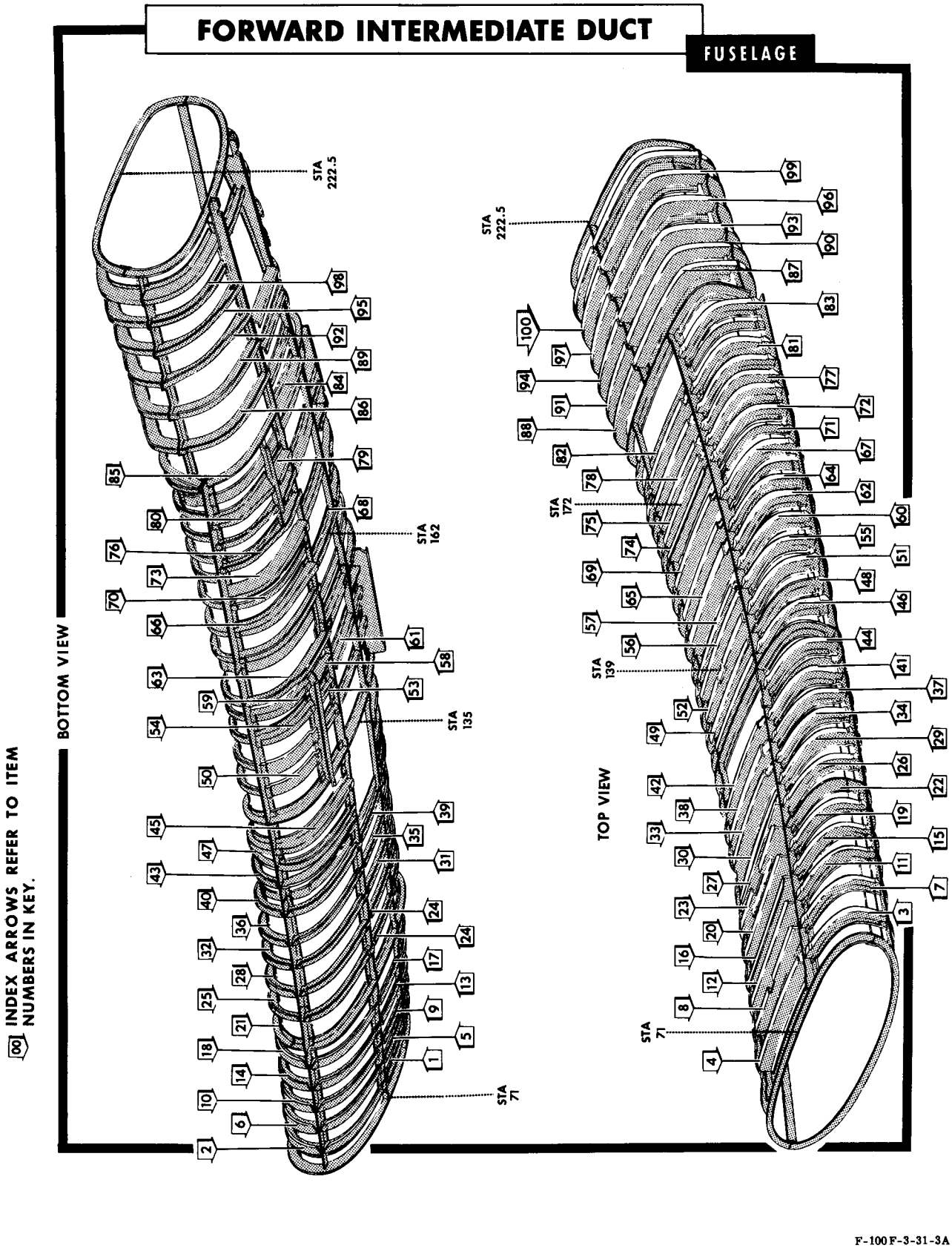

Figure 4-13A. Forward Intermediate Section Repair Index—F-100F Airplane (Sheet 3 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	243-316216	Duct Frame—Lower Center Sta 76			
2	243-316203	Duct Frame—Lower Side Sta 76			
3	243-316129	Duct Frame—Upper Side Sta 76			
4	243-316128	Duct Frame—Upper Center Sta 76			
5	243-316217	Duct Frame—Lower Center Sta 81.125			
6	243-316205	Duct Frame—Lower Side Sta 81.125			
7	243-316131	Duct Frame—Upper Side Sta 81.125			
8	243-316130	Duct Frame—Upper Center Sta 81.125			
9	243-316218	Duct Frame—Lower Center Sta 86.125			
10	243-316207	Duct Frame—Lower Side Sta 86.125			
11	243-316133	Duct Frame—Upper Side Sta 86.125			
12	243-316132	Duct Frame—Upper Center Sta 86.125			
13	243-316219	Duct Frame—Lower Center Sta 91.25			
14	243-316209	Duct Frame—Lower Side Sta 91.25			
15	243-316135	Duct Frame—Upper Side Sta 91.25			
16	243-316134	Duct Frame—Upper Center Sta 91.25			
17	243-316220	Duct Frame—Lower Center Sta 96.375			
18	243-316211	Duct Frame—Lower Side Sta 96.375			
19	243-316137	Duct Frame—Upper Side Sta 96.375			
20	243-316136	Duct Frame—Upper Center Sta 96.375			
21	243-316276	Duct Frame—Lower Sta 100			
22	243-316141	Duct Frame Sta 100			
23	243-316140	Duct Frame—Upper Center Sta 100			
24	243-316224	Duct Frame—Lower Center Sta 104 & 108		7075-T6 Extr	
25	243-316189	Duct Frame—Lower Side Sta 104		7075-T6 Extr	
26	243-316143	Duct Frame—Upper Side Sta 104		2024-T42 Extr	
27	243-316142	Duct Frame—Upper Center Sta 105.187	0.070	7075-T6 Clad	
28	243-316191	Duct Frame—Lower Side Sta 108		7075-T6 Extr	
29	243-316145	Duct Frame—Upper Side Sta 108		2024-T42 Extr	
30	243-316144	Stiffener—Duct Upper Center Sta 108		7075-T6 Extr	
31	243-316226	Duct Frame—Lower Center Sta 112.063	0.050	7075-T6 Clad	
32	243-316193	Duct Frame—Lower Side Sta 112.063		7075-T6 Extr	
33	243-316146	Duct Frame—Upper Center Sta 112.063	0.050	7075-T6 Clad	
34	243-316147	Duct Frame—Upper Side Sta 112.063		7075-T6 Extr	
35	243-316227	Duct Frame—Lower Center Sta 116.125	0.050	7075-T6 Clad	
36	243-316195	Duct Frame—Lower Side Sta 116.125		7075-T6 Extr	
37	243-316149	Duct Frame—Upper Side Sta 116.125		2024-T42 Extr	
38	243-316148	Duct Frame—Upper Center Sta 116.125	0.050	7075-T6 Clad	
39	243-316228	Duct Frame—Lower Center Sta 121.125	0.050	7075-T6 Clad	
40	243-316197	Duct Frame—Lower Side Sta 121.125		7075-T6 Extr	
41	243-316151	Duct Frame—Upper Side Sta 121.125		2024-T42 Extr	
42	243-316150	Duct Frame—Upper Center Sta 126.25	0.050	7075-T6 Clad	
43	243-316233	Duct Frame—Lower Side Sta 126.25	78J	7075-T6 Extr	
44	243-316235	Duct Frame—Upper Side Sta 126.25	6E524	7075-T6 Extr	
45	243-316239	Duct Frame—Lower Side Sta 131.375	6E524	7075-T6 Extr	
46	243-316236	Duct Frame—Upper Side Sta 131.375	6E524	2024-T42 Extr	
47	243-31662	Duct Frame—Lower Side Canted Sta 133.118	0.071	7075-T6 Clad	
48	243-316231	Duct Frame—Upper Side Sta 135	34610	7075-T6 Extr	
49	243-316232	Duct Frame—Upper Center Sta 135	6E549	7075-T6 Extr	
50	243-316242	Strap—Lower Side Sta 139	0.312	7075-T6 Clad	
51	243-316243	Duct Frame—Upper Side Sta 139	6E524	2024-T42 Extr	
52	243-316246	Duct Frame—Upper Center Sta 139	6E496	7075-T6 Extr	
53	243-316248	Duct Frame—Lower Inbd Sta 143	6E524	7075-T6 Extr	
54	243-316223	Duct Frame—Lower Side Sta 143	0.063	2024-T4 Clad	
55	243-316153	Duct Frame—Upper Side Sta 143	0.071	7075-T6 Clad	
56	243-316152	Duct Frame—Upper Center Sta 143			

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† For additional repairs, refer to Section X.

Key to Figure 4-13A (Sheet 3 of 3)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
57	243-316234	Duct Frame—Upper Center Sta 145.5	0.050	7075-T6 Clad	
58	223-31658	Duct Frame—Lower Inbd Sta 147.5	6E524	7075-T6 Extr	
59	243-316230	Duct Frame—Lower Side Sta 147.5	0.063	2024-T4 Clad	
60	243-316154	Duct Frame—Upper Side Sta 147.5	6E524	2024-T42 Extr	
61	243-316244	Duct Frame—Lower Center Sta 149	0.050	7075-T6 Clad	
62	243-316158	Duct Frame—Upper Side Sta 151	6E524	7075-T6 Extr	
63	243-316253	Duct Frame—Lower Side Sta 154	0.090	CRES	
64	243-316157	Duct Frame—Upper Side Sta 154	6E524	7075-T6 Extr	
65	243-316161	Duct Frame—Upper Center Sta 156.5	0.050	7075-T6 Clad	
66	243-316254	Duct Frame—Sta 159.75	6E524	7075-T6 Extr	
67	243-316162	Duct Frame—Upper Side Sta 159.75	0.071	7075-T6 Clad	
68	243-316266	Duct Frame—Lower Center Sta 162			
69	243-316238	Duct Frame—Upper Center Sta 162	0.063	7075-T6 Clad	
70	243-316255	Duct Frame—Lower Side Sta 163.375	6E524	7075-T6 Extr	
71	243-316164	Duct Frame—Upper Side Sta 163.375	6E524	7075-T6 Extr	
72	243-316273	Duct Frame—Upper Side Sta 167	6E524	7075-T6 Extr	
73	243-316274	Duct Frame—Lower Side Sta 167	0.050	7075-T6 Clad	
74	243-316163	Duct Frame—Upper Center Sta 167.75	0.051	7075-T6 Clad	
75	243-316165	Duct Frame—Upper Center Sta 172	0.071	2024-T4 Clad	
76	243-316263	Duct Frame—Lower Side Sta 175.25	6E524	7075-T6 Extr	
77	243-316168	Duct Frame—Upper Side Sta 175.25	0.050	7075-T6 Clad	
78	243-316167	Duct Frame—Upper Center Sta 175.25	0.063	2024-T4 Clad	
79	243-316270	Duct Frame—Lower Inbd Sta 179	0.063	2024-T4 Clad	
80	243-316269	Duct Frame—Lower Side Sta 179	6E524	7075-T6 Extr	
81	243-316264	Duct Frame—Upper Side Sta 179	0.050	7075-T6 Clad	
82	243-316257	Duct Frame—Upper Center Sta 179	6E524	2024-T42 Extr	
83	243-31661	Duct Frame—Upper Side Sta 188.25	0.080	17-7 PH CRES	
84	243-316260	Duct Frame—Lower Center Sta 189.898	1E456	7075-T6 Extr	
85	243-316259	Duct Frame—Lower Side Sta 189.898	0.050	7075-T6 Clad	
86	223-31674	Duct Frame—Lower Sta 196	0.063	7075-T6 Clad	
87	223-31687	Duct Frame—Upper LH Sta 196	0.063	7075-T6 Clad	
88	223-31676	Duct Frame—Upper RH Sta 196	0.063	7075-T6 Clad	
89	192-316203	Duct Frame Sta 200.5	0.080	7075-T6 Clad	
90	243-316271	Duct Frame—Upper LH Sta 200.5	0.051	7075-T6 Clad	
91	243-316201	Duct Frame—Upper RH Sta 200.5	0.050	7075-T6 Clad	
92	243-316206	Duct Frame—Lower Side Sta 205.75	0.050	7075-T6 Clad	
93	243-316205	Duct Frame—LH Sta 205.75	0.050	7075-T6 Clad	
94	192-316206	Duct Frame—RH Sta 205.75	0.050	7075-T6 Clad	
95	243-316210	Duct Frame—Lower Sta 209.5	0.050	7075-T6 Clad	
96	192-316209	Duct Frame—Upper LH Sta 209.5	0.050	7075-T6 Clad	
97	192-316210	Duct Frame—Upper RH Sta 209.5	0.050	7075-T6 Clad	
98	192-316215	Duct Frame Sta 213.75	0.050	7075-T6 Clad	
99	192-316213	Duct Frame—LH Sta 213.75	0.050	7075-T6 Clad	
100	192-316214	Duct Frame—RH Sta 213.75			

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

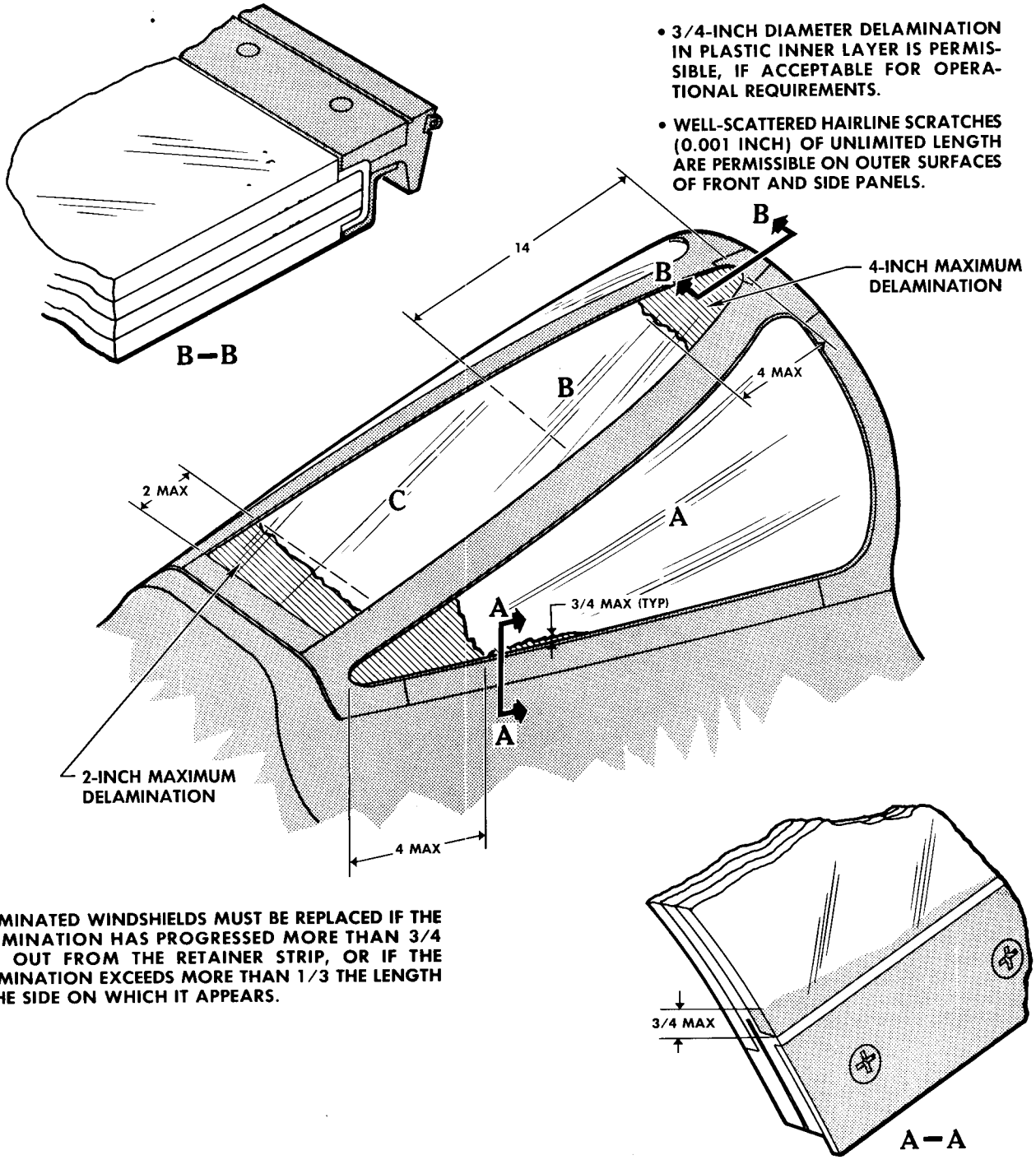
Key to Figure 4-13A (Sheet 3 of 3 Cont)

WINDSHIELD NEGLIGIBLE DAMAGE

FUSELAGE

SMALL, WELL-SCATTERED BUBBLES ARE ALLOWED, PROVIDED THE BUBBLES DO NOT EXCEED 1/16 INCH IN DIAMETER.

- 3/4-INCH DIAMETER DELAMINATION IN PLASTIC INNER LAYER IS PERMISSIBLE, IF ACCEPTABLE FOR OPERATIONAL REQUIREMENTS.
- WELL-SCATTERED HAIRLINE SCRATCHES (0.001 INCH) OF UNLIMITED LENGTH ARE PERMISSIBLE ON OUTER SURFACES OF FRONT AND SIDE PANELS.



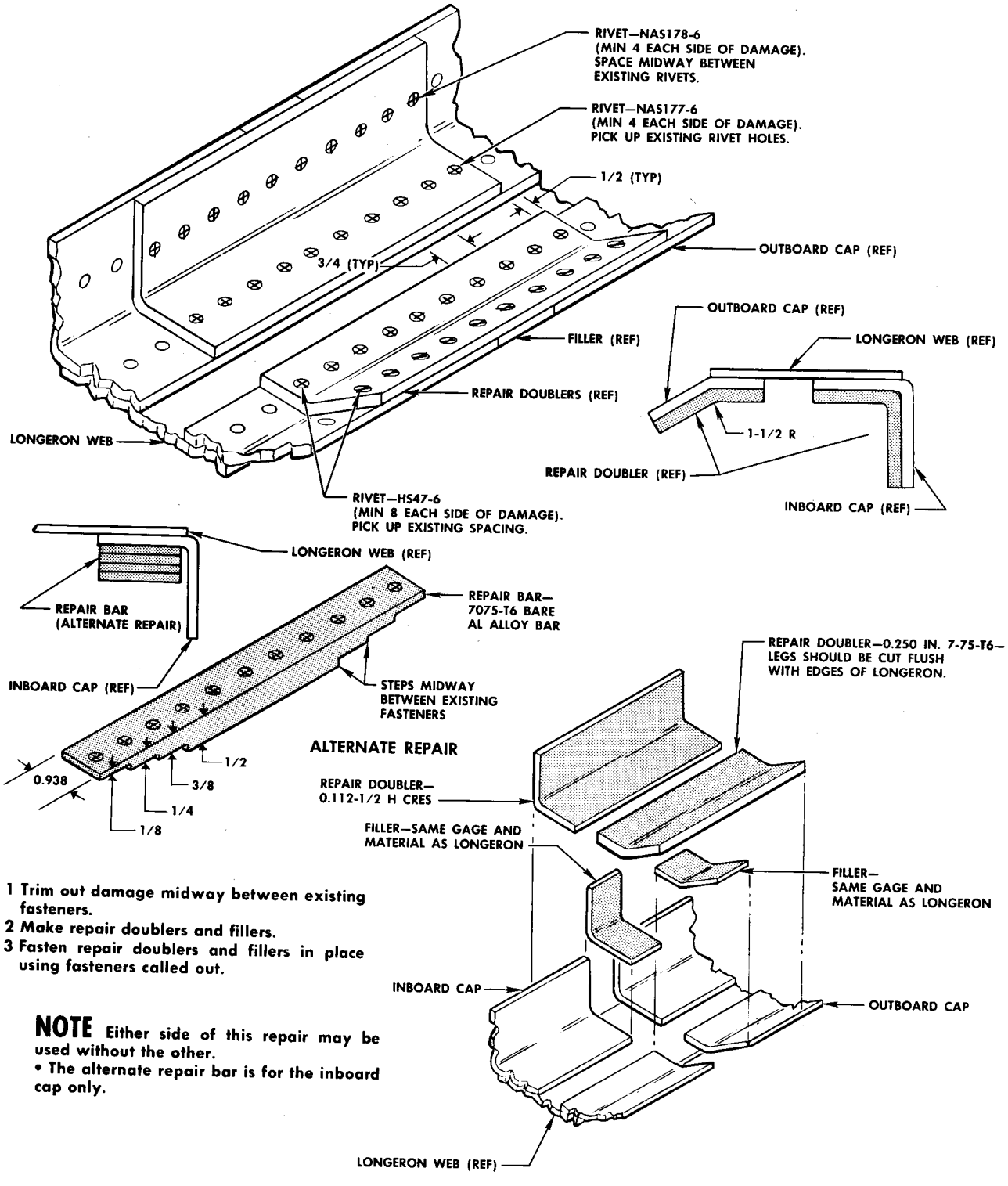
DELAMINATED WINDSHIELDS MUST BE REPLACED IF THE DELAMINATION HAS PROGRESSED MORE THAN 3/4 INCH OUT FROM THE RETAINER STRIP, OR IF THE DELAMINATION EXCEEDS MORE THAN 1/3 THE LENGTH OF THE SIDE ON WHICH IT APPEARS.

NOTE Refer to "Windshield Negligible Damage."

F-100D-3-31-99B

Figure 4-13B. Windshield Negligible Damage

PERMANENT CATEGORY A — FUSELAGE



1 Trim out damage midway between existing fasteners.
2 Make repair doublers and fillers.
3 Fasten repair doublers and fillers in place using fasteners called out.

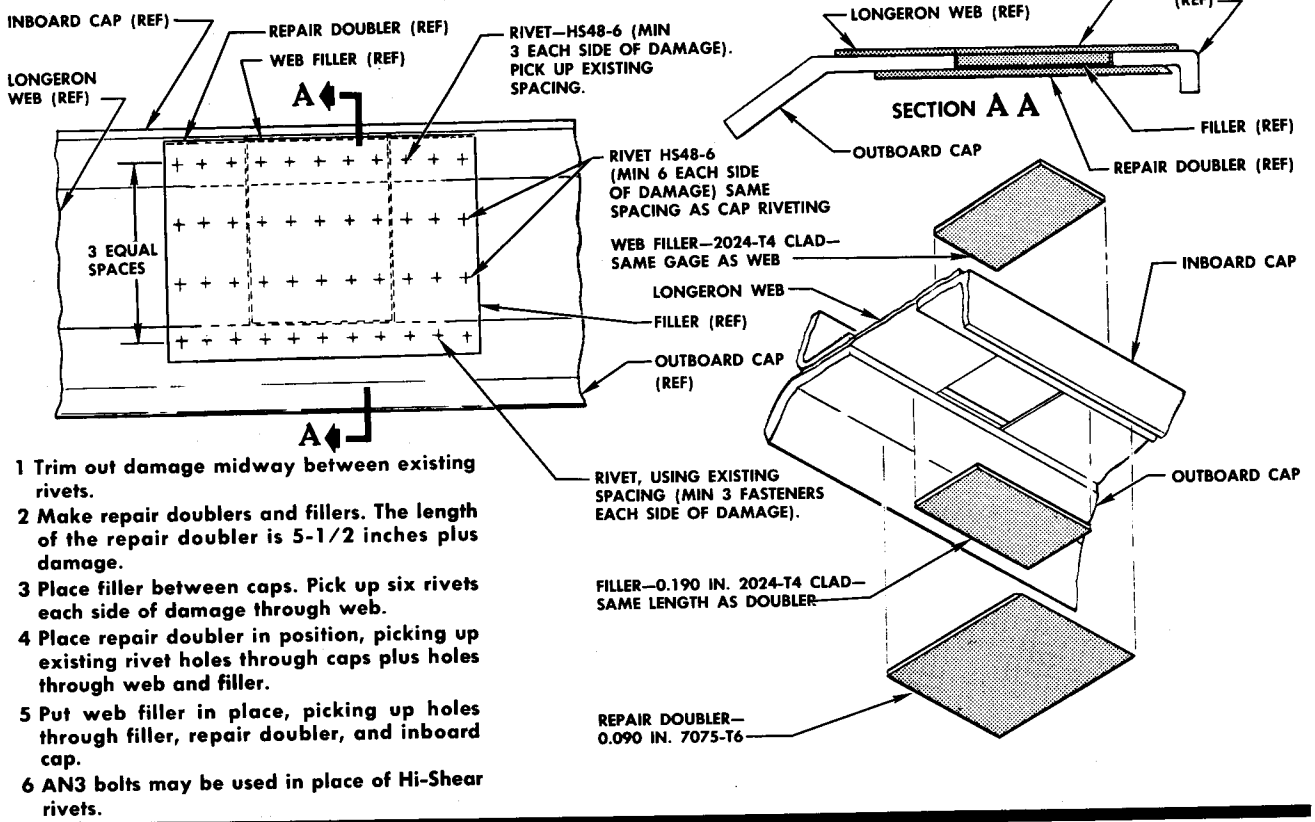
NOTE Either side of this repair may be used without the other.
• The alternate repair bar is for the inboard cap only.

F-100D-3-31-34A

Figure 4-14. Longeron Assembly (Sheet 1 of 2)

PERMANENT CATEGORY A

FUSELAGE



1 Trim out damage midway between existing rivets.

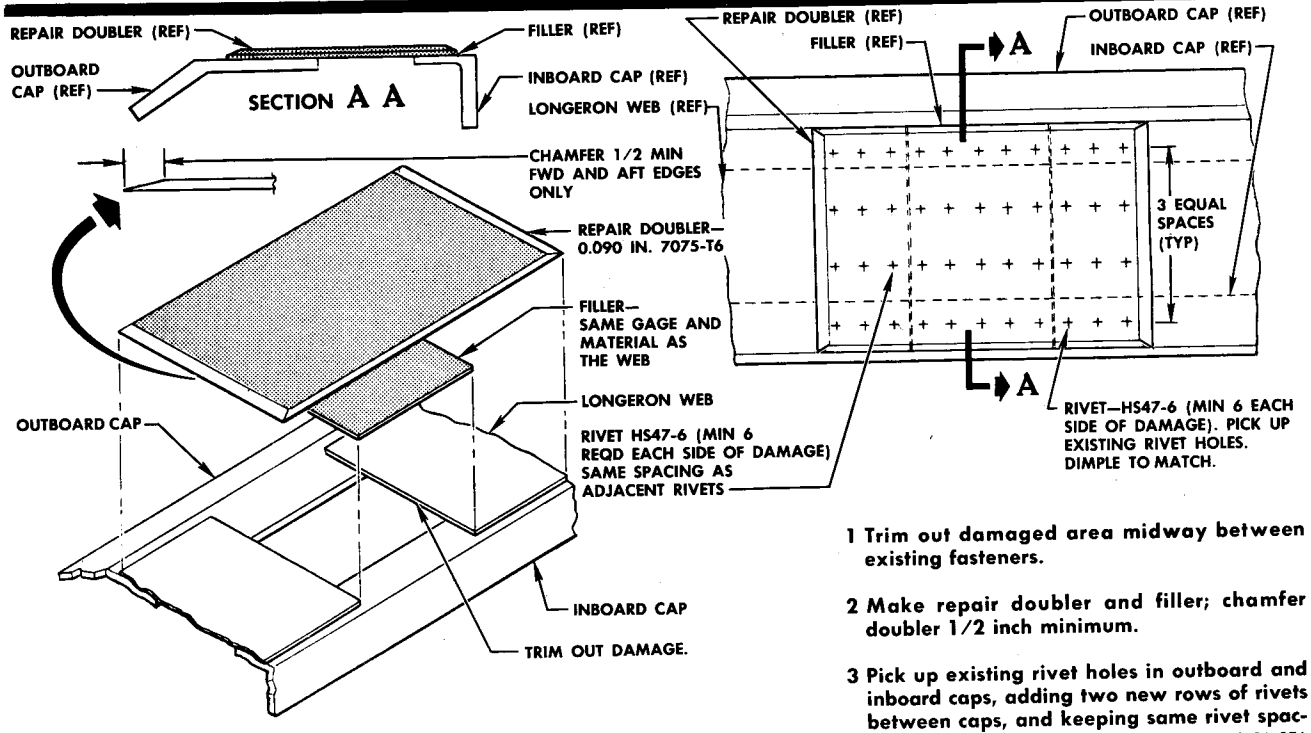
2 Make repair doublers and fillers. The length of the repair doubler is 5-1/2 inches plus damage.

3 Place filler between caps. Pick up six rivets each side of damage through web.

4 Place repair doubler in position, picking up existing rivet holes through caps plus holes through web and filler.

5 Put web filler in place, picking up holes through filler, repair doubler, and inboard cap.

6 AN3 bolts may be used in place of Hi-Shear rivets.



1 Trim out damaged area midway between existing fasteners.

2 Make repair doubler and filler; chamfer doubler 1/2 inch minimum.

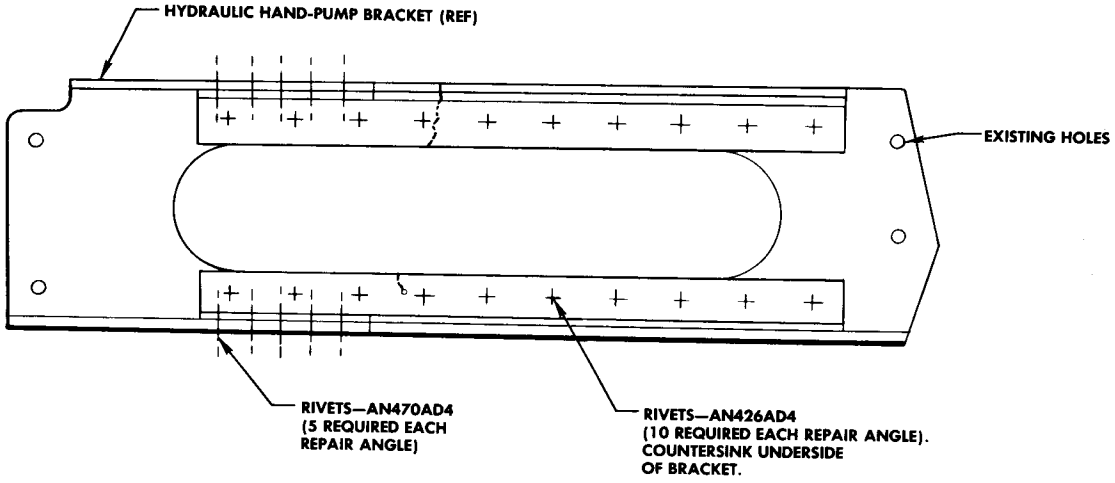
3 Pick up existing rivet holes in outboard and inboard caps, adding two new rows of rivets between caps, and keeping same rivet spacing.

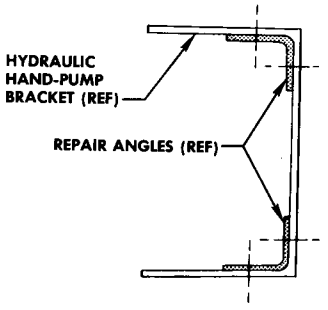
F-100D-3-31-37A

Figure 4-14. Longeron Assembly (Sheet 2 of 2)

PERMANENT CATEGORY A

FUSELAGE

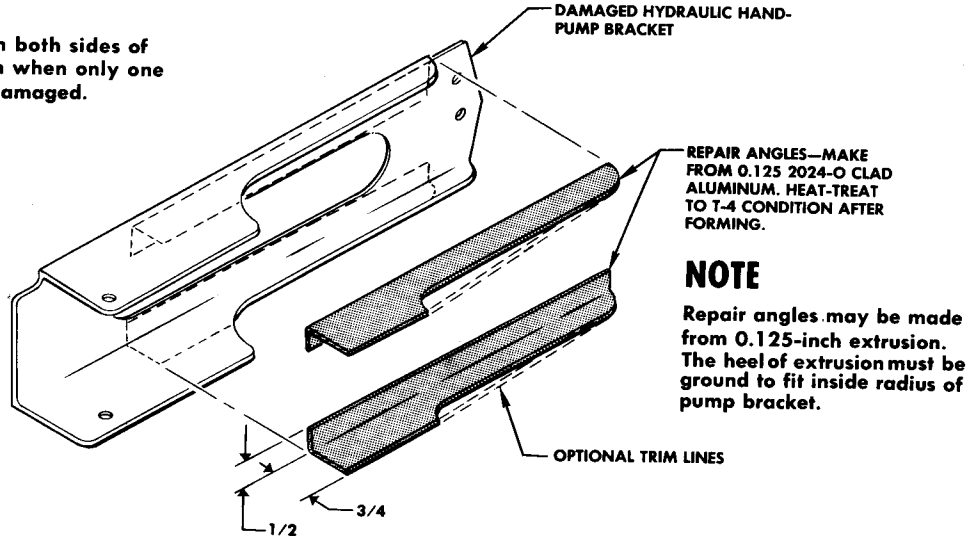




1. Stop-drill all cracks.
2. Make repair angles from either 0.125-inch 2024-T4 sheet metal or 0.125-inch thick extrusion.
3. Fasten repair angles in place using AN426 flush-head rivets through the base of the bracket and AN470 universal-head rivets through the side of the bracket.

NOTE

Place repair angles on both sides of damaged bracket even when only one side of the bracket is damaged.



NOTE

Repair angles may be made from 0.125-inch extrusion. The heel of extrusion must be ground to fit inside radius of pump bracket.

F-100D-3-31-122

Figure 4-14A. Hydraulic Hand-pump Bracket Repair

PERMANENT—CATEGORY A

FUSELAGE

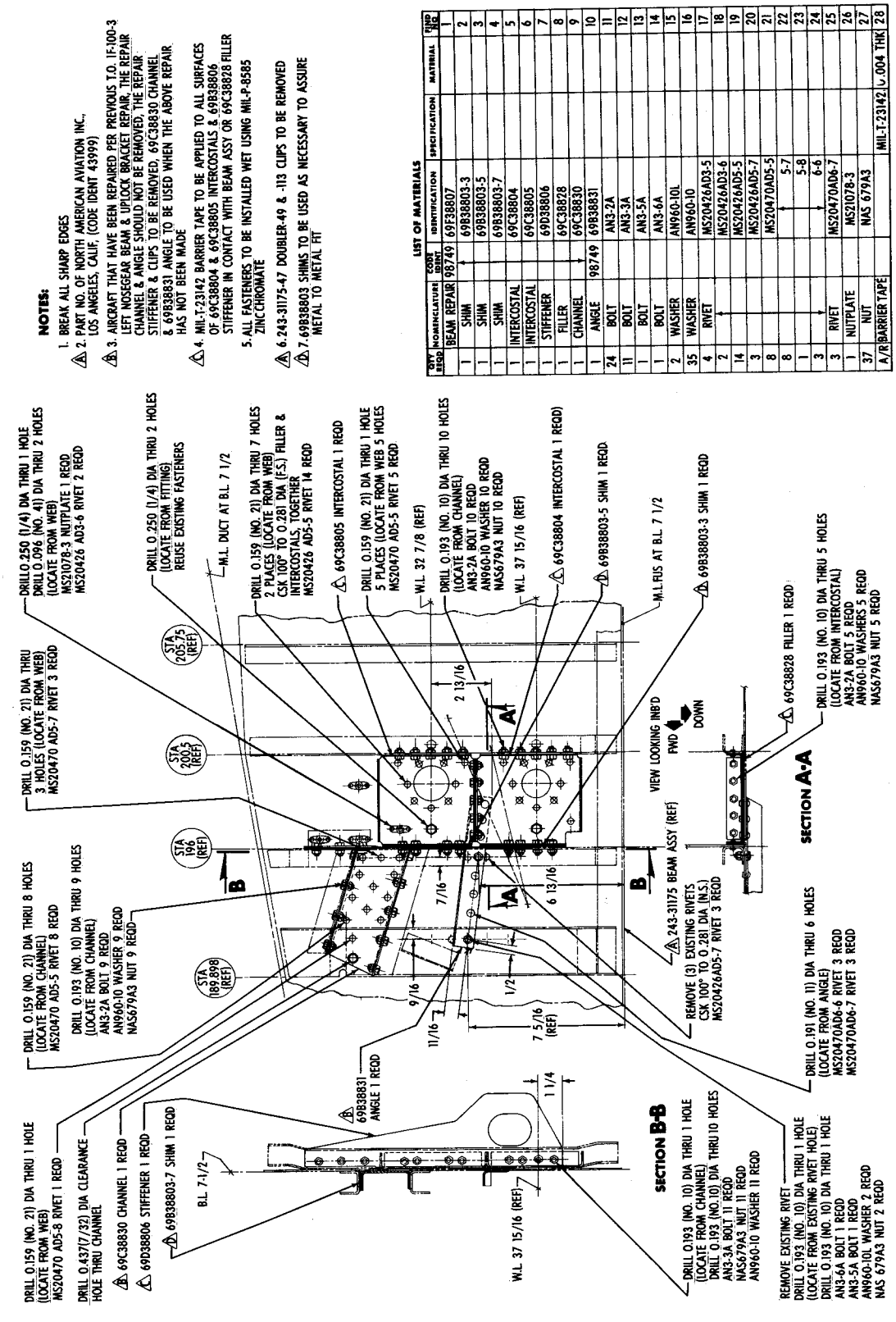
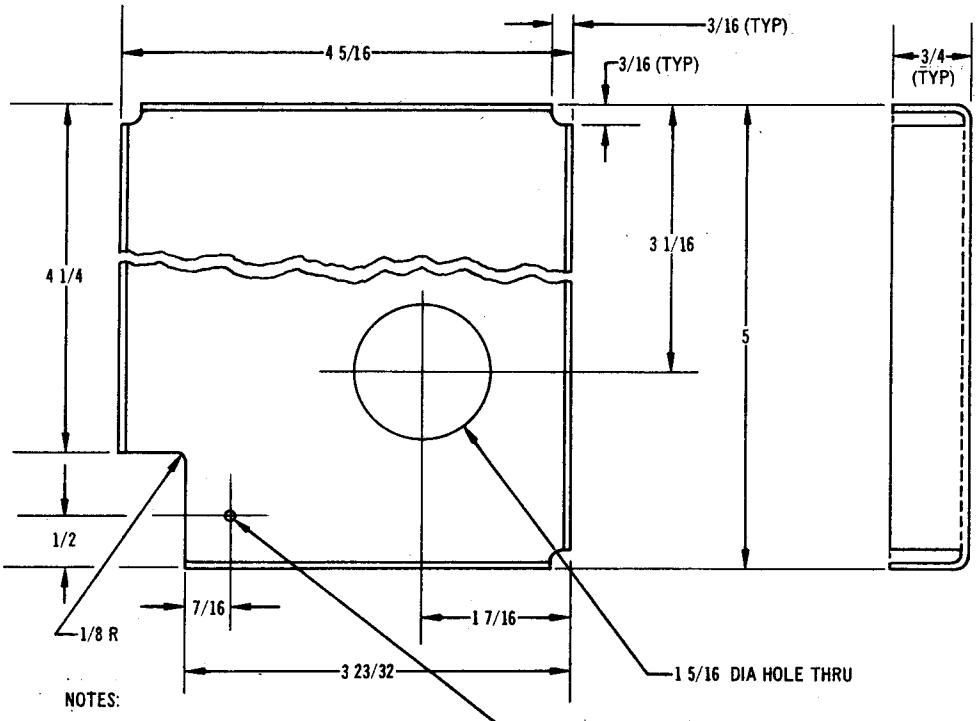


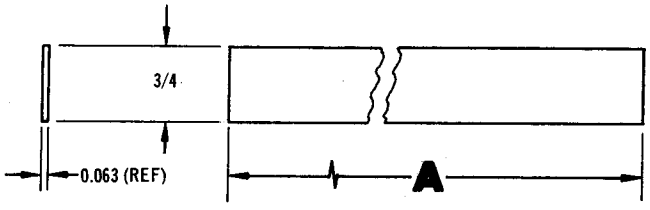
Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 1 of 6)

PERMANENT CATEGORY A FUSELAGE



NOTES:
1. BREAK ALL SHARP EDGES
2. BEND RADII AND BEND RELIEF RADII =3/32
3. MATERIAL: MIL-S-5059, COMP 302 AND CONDITION (ANNEALED)
4. FINISH: CADMIUM PLATE QQ-P-416, TYPE II, CLASS 3
5. APPLY PART NO. PER MIL-STD-130

INTERCOSTAL BEAM NOSE GEAR



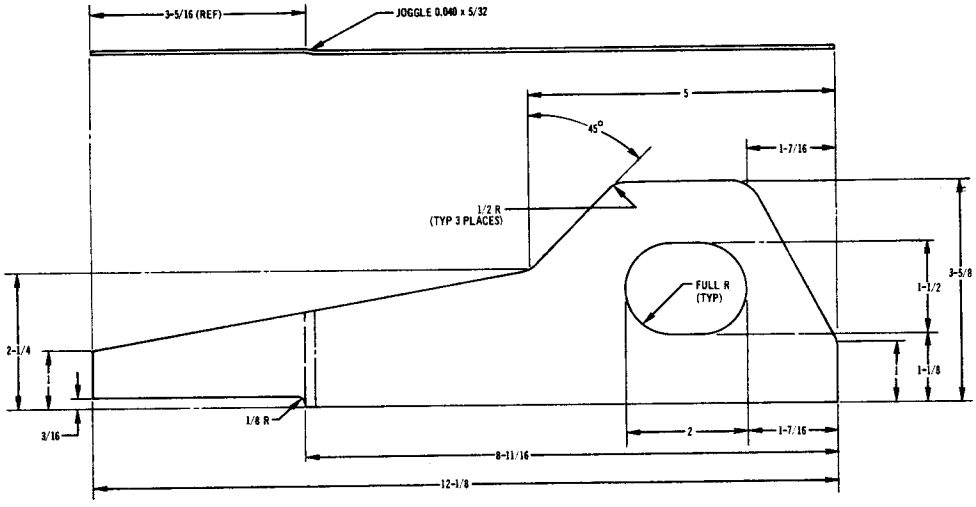
NOTES:
1. BREAK ALL SHARP EDGES
2. MATERIAL: MIL-S-22499 COMP. 1, CLASS 2 AND TYPE I LAMINATED SHIM STOCK
3. APPLY PART NOS. PER MIL-STD-130

PART NO.	A DIM		
-3	3 1/4		
-5	4 5/8		
-7	5 3/16		

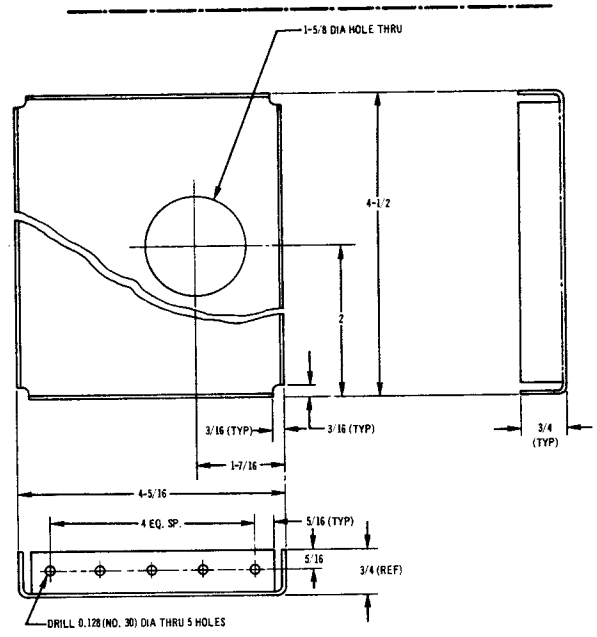
	LIST OF MATERIALS								
QTY REQD	SYM	NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL	UNIT WT	ZONE	FIND NO.
		SHIM	98749	69B38803-5	SEE NOTE 2	0.063 x 0.75 x 3.30			1
		SHIM	98749	69B38803-3		0.063 x 0.75 x 4.70			2
		SHIM	98749	69B38803-7		0.063 x 0.75 x 5.20			3
		INTERCOSTAL	98749	69C38804	SEE NOTE 3	0.040 x 6.00 x 6.50 CRES SH			1

Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 2 of 6)

PERMANENT—CATEGORY A FUSELAGE



LIST OF MATERIALS				
NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL
STIFFENER	98749	69D38806	SEE NOTE 3	0.050 x 4.50 x 12.13 CRES SH



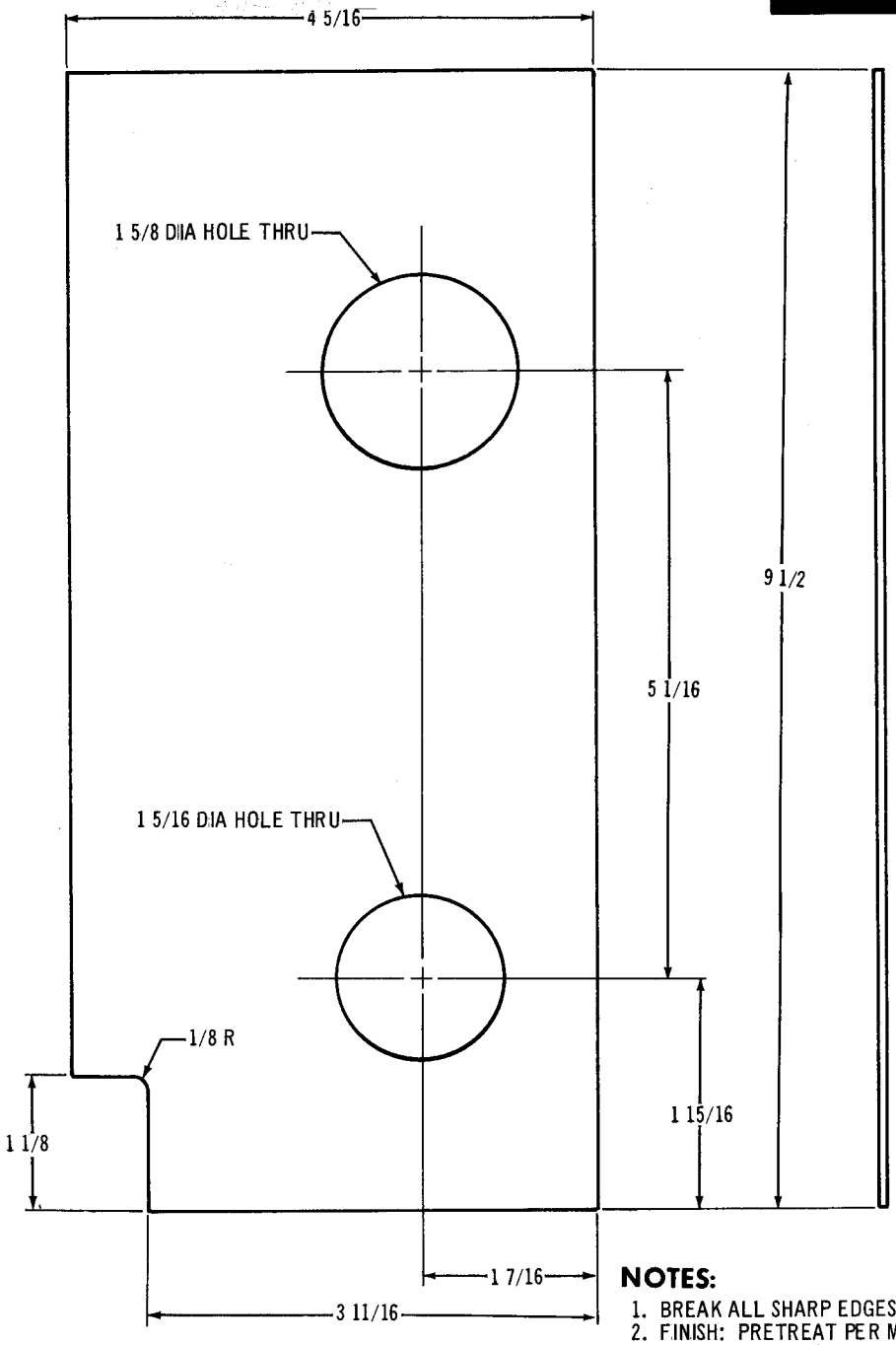
NOTES:

1. BREAK ALL SHARP EDGES.
2. BEND RADII & BEND RELIEF RADII = 3/32.
3. MATERIAL: MIL-S-5059, COMP. 302 & CONDITION (ANNEALED).
4. FINISH: CADMIUM PLATE QQ-P-416, TYPE II, CLASS 3.
5. APPLY PART NO. PER MIL-STD-130.

LIST OF MATERIALS				
NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL
INTERCOSTAL	98749	69C38805	SEE NOTE 3	0.040 X 5.90 X 6.0 CRES SH

Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 3 of 6)

PERMANENT—CATEGORY A **FUSELAGE**



NOTES:

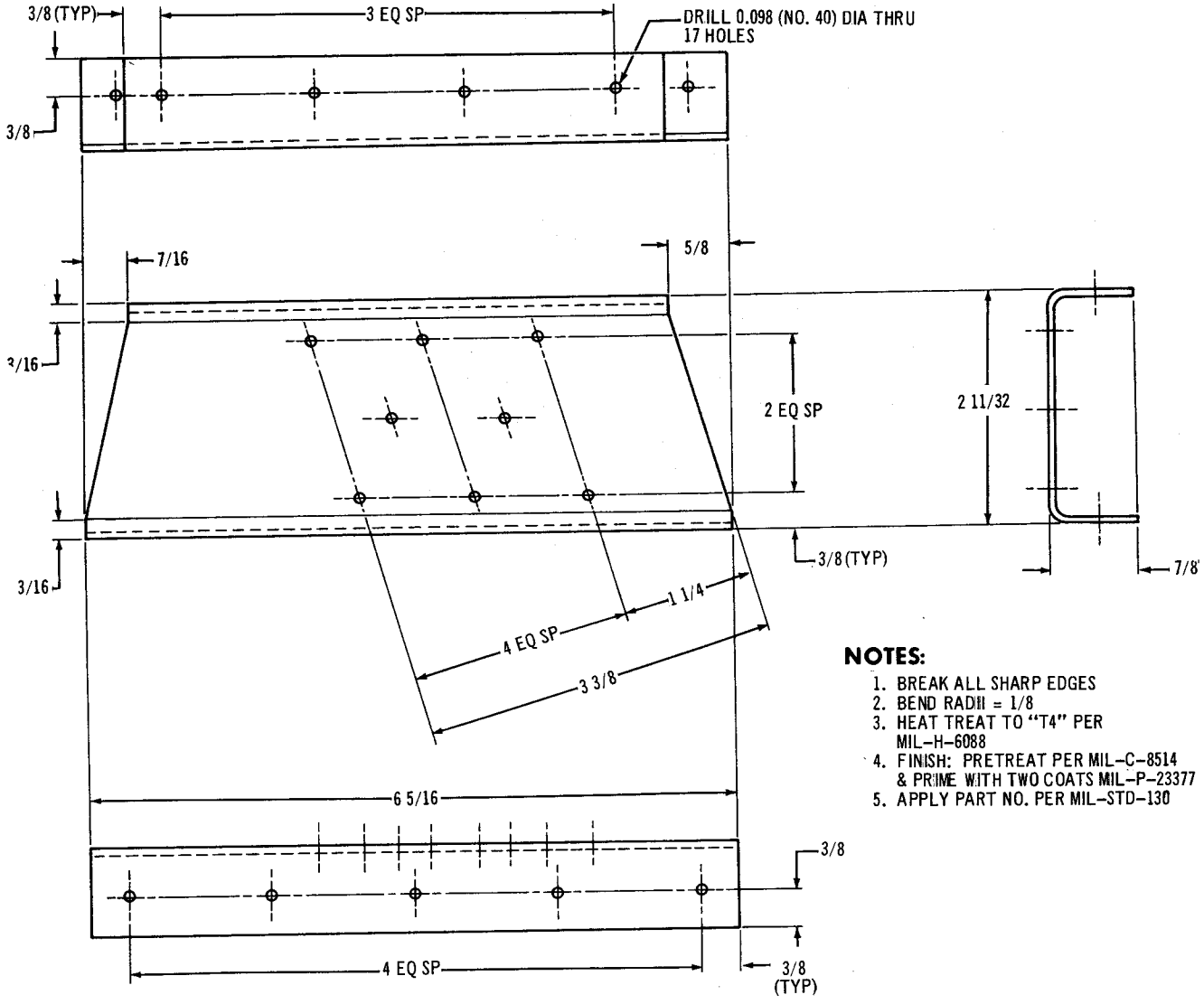
1. BREAK ALL SHARP EDGES
2. FINISH: PRETREAT PER MIL-C-8514 & PRIME WITH TWO COATS MIL-P-23377
3. APPLY PART NO. PER MIL-STD-130

LIST OF MATERIALS

NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL	UNIT WT	ZONE
FILLER	98749	69C38828	QQ-A-250/5 TEMP "T3"	0.032 x 4.50 x 9.70 AL. ALY. S		

Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 4 of 6)

PERMANENT—CATEGORY A FUSELAGE



NOTES:

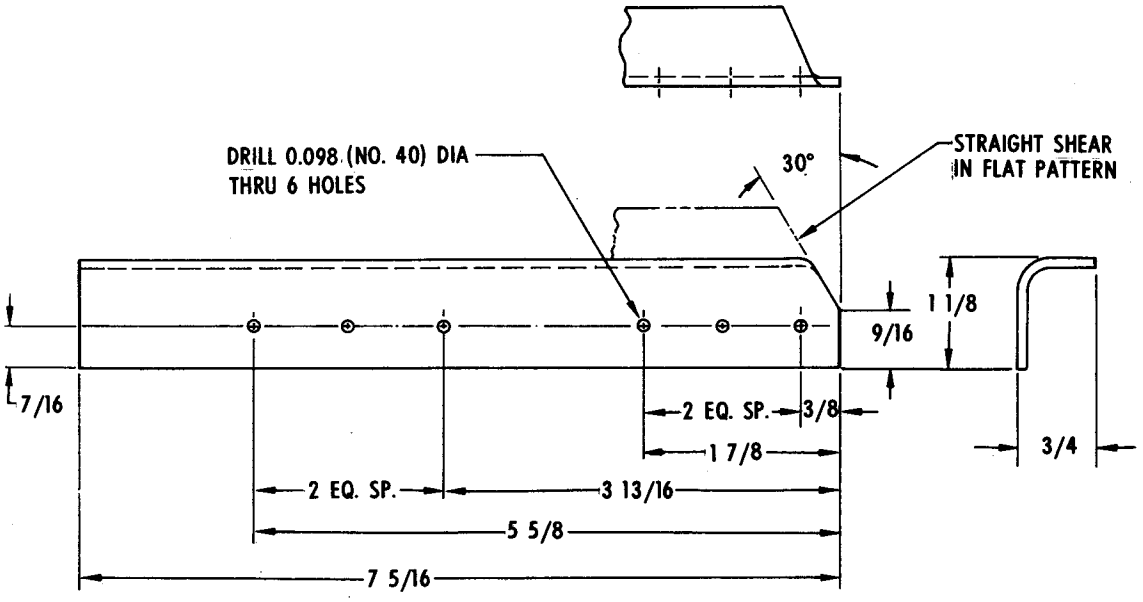
1. BREAK ALL SHARP EDGES
2. BEND RADII = 1/8
3. HEAT TREAT TO "T4" PER MIL-H-6088
4. FINISH: PRETREAT PER MIL-C-8514 & PRIME WITH TWO COATS MIL-P-23377
5. APPLY PART NO. PER MIL-STD-130

LIST OF MATERIALS

NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL	UNIT WT	ZONE
CHANNEL	98749	69C38830	QQ-A-150/5 SEE NOTE 3	0.063 x 4.10 x 6.35 AL. ALY. SH		

Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 5 of 6)

PERMANENT—CATEGORY A FUSELAGE



NOTES:
1. BREAK ALL SHARP EDGES
2. BEND RADIUS = 9/32
3. APPLY PART NO. PER MIL-STD-130
4. FINISH PRETREAT PER MIL-C-8514 & PRIME WITH TWO COATS MIL-P-23377

NOMENCLATURE	CODE IDENT	IDENTIFYING NO.	SPECIFICATION	MATERIAL	UNIT WT	ZONE	FIND NO.
ANGLE	98749	69B38831	QQ-A--250/5 TEMP 'T3'	.090× 1.9 × 7.4 AL. ALY SH.			1

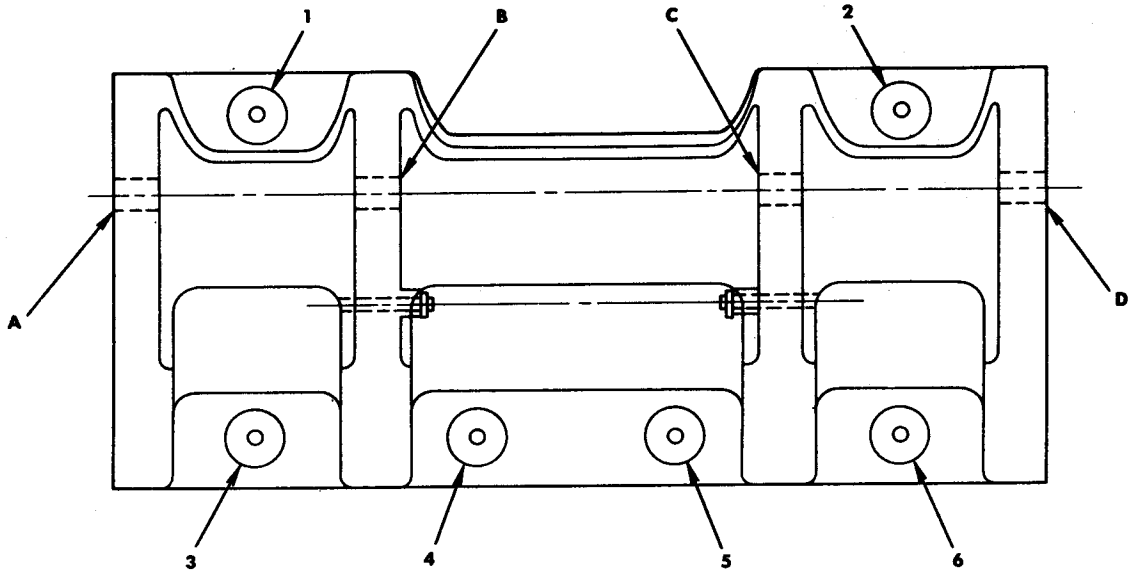
Figure 4-14A-1. Left Nose Gear Beam and Uplock Bracket Repair, F-100D/F (Sheet 6 of 6)

(All data on page 4-30G, including Figure 4-14B is deleted)

PERMANENT—CATEGORY A

FUSELAGE

CORROSION DEPTH REMOVAL LIMITS (223-31889)



HOLE NO.	REMOVAL LIMIT (INCHES)
1—6	0.080
A—D	0.150

1 Limits are for total of both sides of each hole.

2 Remove corrosion by increasing spot-face depth.

3 Where no spot face exists, remove material smoothly from entire surface.

4 Refinish per instructions in "Corrosion Protection for Magnesium Alloys" in Section I.

5 Reinstall part with vinyl barrier tape between all magnesium and aluminum faying surfaces. (Refer to Section I.)

100D-3-31-167

Figure 4-14C. Repair of Corroded Canopy Guide Rail Castings—Upper

REPAIR CAPABILITY A

FUSELAGE

CORROSION DEPTH REMOVAL LIMITS

If corrosion has been determined to be severe or if heavy pitting exists as stated in Table 6-1 of T.O. 1-1-2, either of two actions are necessary:

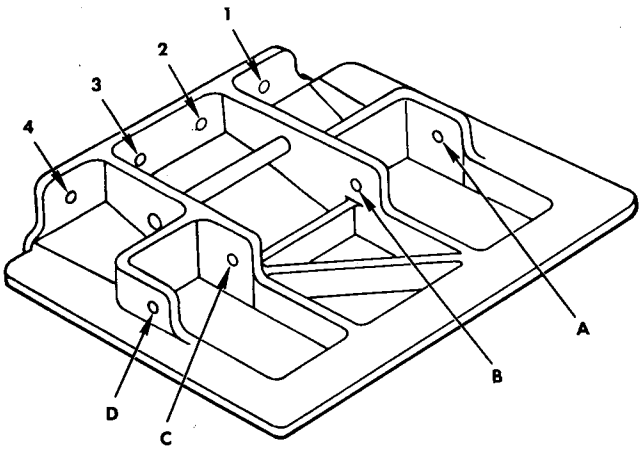
1. Rework the rails in accordance with the above figure provided the 5/32 inch limit is not exceeded.
2. If limits are exceeded, replace with new part.

Figure 4-14D. Corrosion Removal Limits and Locations 223-31835-3 or -4

PERMANENT—CATEGORY A

FUSELAGE

CORROSION DEPTH REMOVAL LIMITS (223-31890)



HOLE NO.	REMOVAL LIMIT (INCHES)	
	WITH ORIGINAL HOLE DIAMETER	WITH 1/16 INCH INCREASE IN HOLE DIAMETER
A	0.060	
B	0.080	
C	0.080	
D	0.060	
1	0.100	
2	0.080	
3	0.060	
4*	0.019	0.031

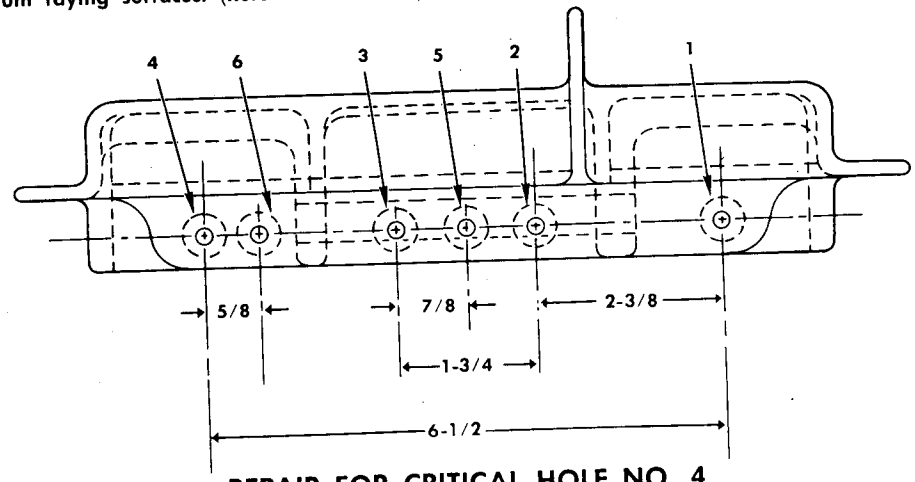
1 Limits are for total of both sides of each hole.

2 Remove corrosion by increasing spot-face depth.

3 Refinish per instructions in "Corrosion Protection for Magnesium Alloys" in Section I.

4 Reinstall part with vinyl barrier tape between all magnesium and aluminum faying surfaces. (Refer to Section I.)

* If corrosion depth is greater than 0.031 inch, but less than 0.060 inch, refer to repair for critical hole No. 4.



REPAIR FOR CRITICAL HOLE NO. 4

1 Drill two new holes, 5 and 6 (0.257-inch diameter), located as shown.

2 Drill two holes to match in bulkhead plate (223-31173).

3 Spot-face holes 5 and 6 in the same manner as holes 1 through 4. Spot-face diameter 5/8 inch with 0.062-inch fillet radius.

4 Use AN4-10A bolts, 2W18-416 washers, and AN365-428 nuts as in holes 1 through 4.

100D-3-31-168

Figure 4-14E. Repair of Corroded Canopy Guide Rail Castings—Lower

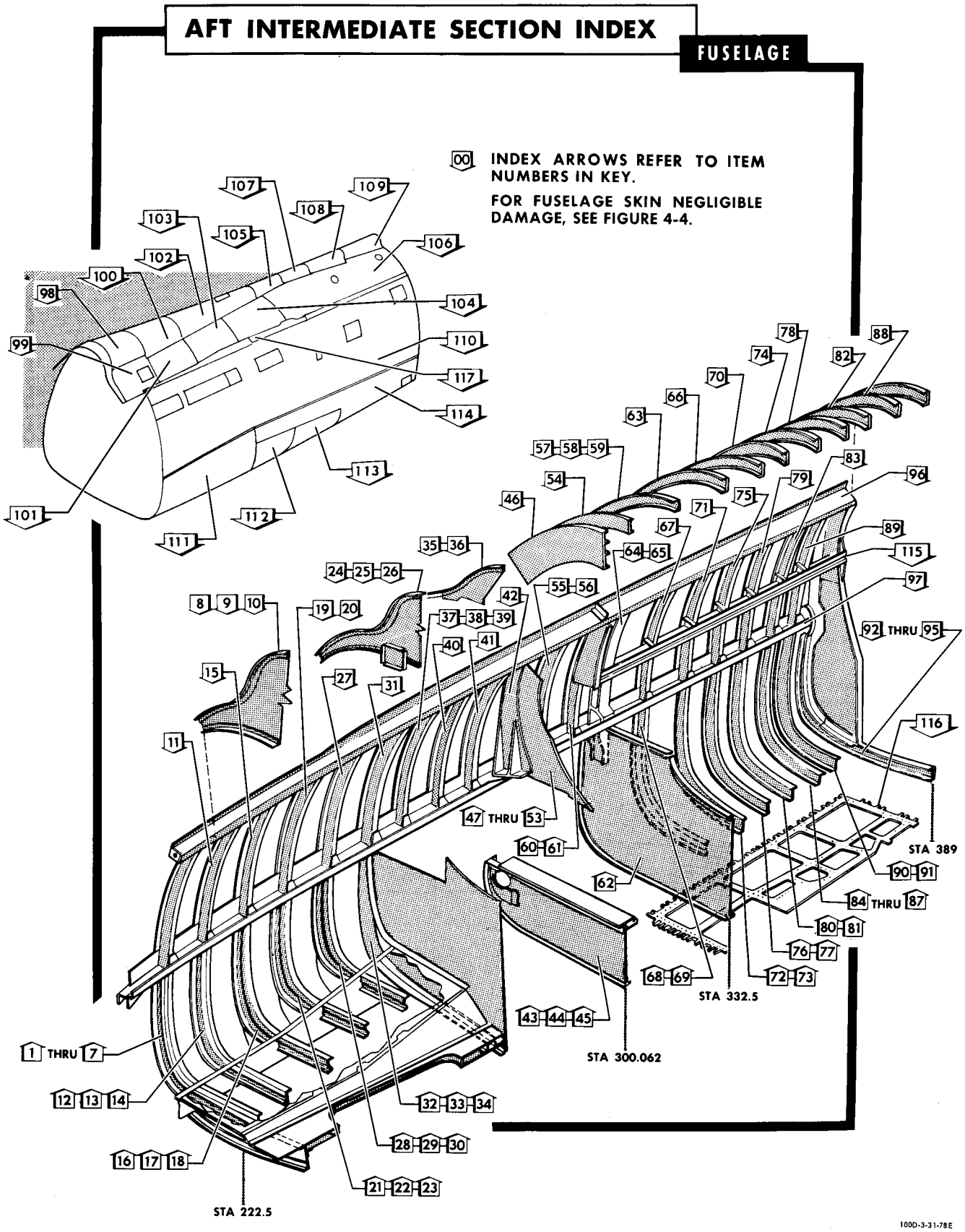


Figure 4-15. Aft Intermediate Section Repair Index (Sheet 1 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31222	Bulkhead Assy	Sta 222½			
2	192-31244	Bulkhead Assy—Lwr	Sta 222½	0.040	7075-T6 Clad	4-30
3	192-31252	Bulkhead Assy—Up.	Sta 222½	0.040	7075-T6 Clad	
4	192-31227	Cap Up.	Sta 222½	4E356	7075-T6 Extr	
5	192-31268	Cap Lwr	Sta 222½	AND10136-2006	7075-T6 Extr	4-30
6	192-31265	Channel Assy	Sta 222½	2E221	7075-T6 Extr	
7	192-31237	Angle	Sta 222½	0.040	2024-T4 Clad	
8	192-31230	Frame Assy—Up.	Sta 231	0.032	7075-T6 Clad	
9	192-31230	Frame Assy—Up.	Sta 231	14044	7075-T6 Extr	
10	192-31271	F.	Sta 231	4E307	7075-T6 Extr	
11	192-31231	Frame Assy—Side	Sta 231	6E514	7075-T6 Extr	4-17
12	192-31245	Frame Assy—Lwr Side	Sta 231	0.063	7075-T6 Clad	
13	192-31245	Frame Assy—Lwr Side	Sta 231	1E409	7075-T6 Extr	
14	192-31245	Frame Assy—Lwr Side	Sta 231	E129	7075-T6 Extr	4-17
15	192-31240	Frame Assy—Side	Sta 240	6E514	7075-T6 Extr	4-16
16	192-31246	Frame Assy—Lwr	Sta 240	0.063	7075-T6 Clad	
17	192-31246	Frame Assy	Sta 240	1E409	7075-T6 Extr	
18	192-31246	Frame Assy	Sta 240	1E129	7075-T6 Extr	4-17
19	192-31249	Frame Assy—Side	Sta 249	6E514	7075-T6 Extr	4-16, 4-17
20	192-31249	Frame Assy—Side	Sta 249	2E185	7075-T6 Extr	
21	192-31247	Frame Assy—Lwr Side	Sta 249	0.063	7075-T6 Clad	
22	192-31247	Frame Assy—Lwr Side F.	Sta 249	1E409	7075-T6 Extr	
23	192-31247	Frame Assy—Lwr Side	Sta 249	1E129	7075-T6 Clad	4-17
24	192-31254	Frame Assy—Up.	Sta 254⅛	0.032	7075-T6 Clad	
25	192-31293	Cap Up.	Sta 254⅛	0.063	7075-T6 Clad	
26	192-31294	Cap Lwr	Sta 254⅛	0.063	7075-T6 Clad	4-30
27	192-31258	Frame Assy—Side	Sta 258	6E514	7075-T6 Extr	4-16, 4-17
28	192-31248	Frame Assy—Lwr Side	Sta 258	0.064	7075-T6 Clad	
29	192-31248	Frame Assy—Lwr Side	Sta 258	1E129	7075-T6 Clad	4-17
30	192-31248	Frame Assy—Lwr Side	Sta 258	1E409	7075-T6 Extr	
31	192-31267	Frame Assy—Side	Sta 267	6E514	7075-T6 Extr	4-16, 4-17
32	192-31251	Frame Assy—Lwr Side	Sta 267	0.063	7075-T6 Clad	4-30
33	192-31251	Frame Assy—Lwr Side	Sta 267	1E410	7075-T6 Extr	4-30
34	192-31251	Frame Assy—Lwr Side	Sta 267	1E129	7075-T6 Extr	4-17, 4-30
35	192-31274	Frame Assy—Up.	Sta 275½	0.040	7075-T6 Clad	
36	192-31292	Frame Cap—Up.	Sta 275½	4E345	7075-T6 Extr	
37	192-312153	Frame Assy—Side RH	Sta 275½	0.064	7075-T6 Clad	
38	192-312152	Frame Assy—Side LH	Sta 275½	0.064	7075-T6 Clad	
39	192-312151	Angle Side	Sta 275½	6E580	7075-T6 Extr	
40	192-31283	Frame Assy—Side	Sta 283	6E514	7075-T6 Extr	4-16
41	192-31291	Frame Assy—Side	Sta 291	6E514	7075-T6 Extr	4-16, 4-17
42	192-31298	Frame Assy—Side	Sta 298	6E514	7075-T6 Extr	4-16, 4-17
43	192-31300	Frame Assy—Bottom	Sta 300 1/16	0.063	7075-T6 Clad	4-17B-4, 4-30
44	192-31300	Frame Assy—Bottom	Sta 300 1/16	10134-1206	7075-T6 Extr	4-30
45	192-312202	Cap Assy	Sta 300 1/16	0.072	7075-T6 Clad	4-30
46	192-31328	Frame Top	Sta 305¼	Forging	14S-T6 Alloy	
47	192-31304	Frame Assy—Side	Sta 305¼	4E362	7075-T6 Extr	
48	192-31304	Frame Assy—Side	Sta 305¼	10133-0702	7075-T6 Extr	
49	192-31304	Frame Assy—Side	Sta 305¼	78J	7075-T6 Extr	
50	192-31304	Frame Assy—Side	Sta 305¼	1E520	7075-T6 Extr	
51	192-31303	Web Side—Frame	Sta 305¼	0.250	7075-T6 Plate	
52	192-31305	Frame Side	Sta 305¼	Forging	14S-T6 Alloy	
53	192-313132	Frame Assy—Up.	Sta 305¼	0.050	7075-T6 Clad	
54	192-31313	Frame Assy—Up.	Sta 314⅝	0.063	7075-T6 Clad	
55	192-31314	Frame Assy—LH Side	Sta 314⅝	6E579	7075-T6 Extr	4-16, 4-17
56	192-31315	Frame Assy—RH Side	Sta 314⅝	6E514	7075-T6 Extr	4-16, 4-17
57	192-31324	Frame Assy—Up.	Sta 325⅝	0.071	7075-T6 Clad	
58	192-31324	Frame Assy—Up.	Sta 325⅝	1E14	7075-T6 Extr	
59	192-31324	Frame Assy—Up.	Sta 325⅝	1E409	7075-T6 Extr	
60	192-31325	Frame Assy—LH Side	Sta 325⅝	0.050	7075-T6 Clad	
61	192-31323	Frame Assy—RH Side	Sta 325⅝	6E514	7075-T6 Extr	
62	223-31368	Bhd Assy—Lwr	Sta 332½			4-30
	-3	Web		0.032	7075-T6 Clad	4-30
	-5	Cap		4E299	7075-T6 Extr	4-30
	-9	Angle		0.090	7075-T6 Clad	
	-11 & -12	Hat		0.071	7075-T6 Clad	
	-15 & -16	Hat		0.051	7075-T6 Clad	
	-17 & -18	Angle		0.051	7075-T6 Clad	
	-39 & -40	Angle		78F	7075-T6 Extr	

*Drawing numbers are for reference only and are not to be used for ordering parts.
†For additional repairs, refer to Section X.

Key to Figure 4-15 (Sheet 1 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
	-45 & -46	Hat		6E172	7075-T6 Extr	
	-51	Angle		0.125	7075-T6 Extr	
	-57	Beam		3E45	7075-T6 Extr	
63	192-31333	Frame Assy—Up.	Sta 334	3E37	7075-T6 Extr	
64	192-31334	Frame Assy—Side	Sta 334	3E37	7075-T6 Extr	
65	192-31335	Fitting Side	Sta 334		14S-76 Forging	
66	192-31340	Frame Assy—Up.	Sta 341	0.063	7075-T6 Clad	4-16, 4-17
67	192-31341	Frame Assy—Side	Sta 341	6E579	7075-T6 Extr	
68	223-31342	Frame Assy—Lwr	Sta 341	1E428	7075-T6 Extr	4-30
69	223-31342	Frame Assy—Lwr	Sta 341	2E202	7075-T6 Extr	
70	192-31349	Frame Assy—Up.	Sta 350	0.063	7075-T6 Clad	
71	192-31350	Frame Assy—Side	Sta 350	6E579	7075-T6 Extr	4-16, 4-17
72	192-31351	Frame Assy—Lwr	Sta 350	2E202	7075-T6 Extr	
73	192-31351	Frame Assy—Lwr	Sta 350	1E428	7075-T6 Extr	
74	192-31356	Frame Assy—Up.	Sta 357	0.071	7075-T6 Clad	
75	192-31357	Frame Assy—Side	Sta 357	6E579	7075-T6 Extr	4-16, 4-17
76	192-31358	Frame Assy—Lwr	Sta 357	2E202	7075-T6 Clad	4-30
77	192-31358	Frame Assy—Lwr	Sta 357	1E428	7075-T6 Extr	4-30
78	192-31364	Frame Assy—Up.	Sta 365	0.063	7075-T6 Clad	
79	192-31365	Frame Assy—Side	Sta 365	6E579	7075-T6 Extr	4-16, 4-17
80	192-31366	Frame Assy—Lwr	Sta 365	2E202	7075-T6 Extr	
81	192-31366	Frame Assy—Lwr	Sta 365	1E428	7075-T6 Extr	
82	192-31372	Frame Assy—Up.	Sta 372¾	0.063	7075-T6 Clad	
83	192-31373	Frame Assy—Side	Sta 372¾	6E579	7075-T6 Extr	4-16, 4-17
84	192-31375	Frame—Lwr RH	Sta 372¾	2E202	7075-T6 Extr	
85	192-31375	Frame Assy—Lwr RH	Sta 372¾	1E428	7075-T6 Extr	
86	192-31374	Frame—Lwr LH	Sta 373¼	2E202	7075-T6 Extr	
87	192-31374	Frame Assy—Lwr LH	Sta 373¼	1E428	7075-T6 Extr	
88	192-31380	Frame Assy—Up.	Sta 381	0.063	7075-T6 Clad	
89	192-31381	Frame Assy—Side	Sta 381	6E579	7075-T6 Extr	4-16, 4-17
90	192-31382	Frame Assy—Lwr LH	Sta 381	2E202	7075-T6 Extr	
91	192-31383	Frame Assy—Lwr RH	Sta 381	1E429	7075-T6 Extr	
92	192-31389	Bhd Assy	Sta 389	77B	7075-T6 Extr	
93	192-31389	Bhd Assy	Sta 389	1E433	7075-T6 Extr	
94	192-31389	Bhd Assy	Sta 389	14263	7075-T6 Extr	
95	192-31389	Bhd Assy	Sta 389	0.050	7075-T6 Clad	
96	192-31223	Longn Assy	Sta 222½ to 389 Up.	6E523	7075-T6 Extr	4-16
97	192-31224	Longn Assy	Sta 222½ to 389 Lwr	6E522	7075-T6 Extr	4-17
98	223-31061	Skin—Top		0.032	2024-T4 Clad	
99	223-31214	Skin—Side		0.032	7075-T6 Clad	4-31, 4-32, 4-33
100	193-310101	Door—Antenna		0.010	Glass Fabric	
101	192-310225	Skin—Side		0.032	7075-T6 Clad	4-31, 4-32, 4-33
102	192-31068	Door—Accumulator Access		0.032	7075-T6 Clad	
103	192-31066	Door—Hydraulic Access		0.032	2024-T4 Clad	
104	192-31067	Door—Hydraulic Access		0.040	7075-T6 Clad	
105	192-31007	Skin—Top		0.063	2024-T4 Clad	
106	192-31403	Skin—Top		0.050	7075-T6 Clad	
107	192-31008	Skin—Top		0.063	2024-T4 Clad	
108	192-31009	Skin—Top		0.063	2024-T4 Clad	
109	192-31010	Skin—Top		0.063	2024-T4 Clad	
110	192-31406	Skin—Side (Machined)		0.125	7075-T6 Clad	4-15A
111	192-31404	Skin—Lower		0.032	7075-T6 Clad	4-31, 4-32, 4-33
112	192-31076	Door—Fuel Bay Access		0.032	7075-T6 Clad	4-16A
113	317-33302	Doors—Wheel Well		0.125	2024-T4 Clad	
114	223-31387	Skin—Bottom		0.040	7075-T6 Clad	4-31, 4-32, 4-33
115	192-31327	Track—Engine Support	Sta 305¼ to 389	6E591	7075-T6 Extr	4-16A
116	7415-313013	Door Assy				4-17B-2, 4-17B-3
117	2629-31011	Doubler—Longn Repair	Sta 291 to 326		301 CRES	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.
‡ Airplanes changed by T.O. 1F-100-1053.

Key to Figure 4-15 (Sheet 1 of 2 Continued)

AFLC SMAMA JAN 70, 1600

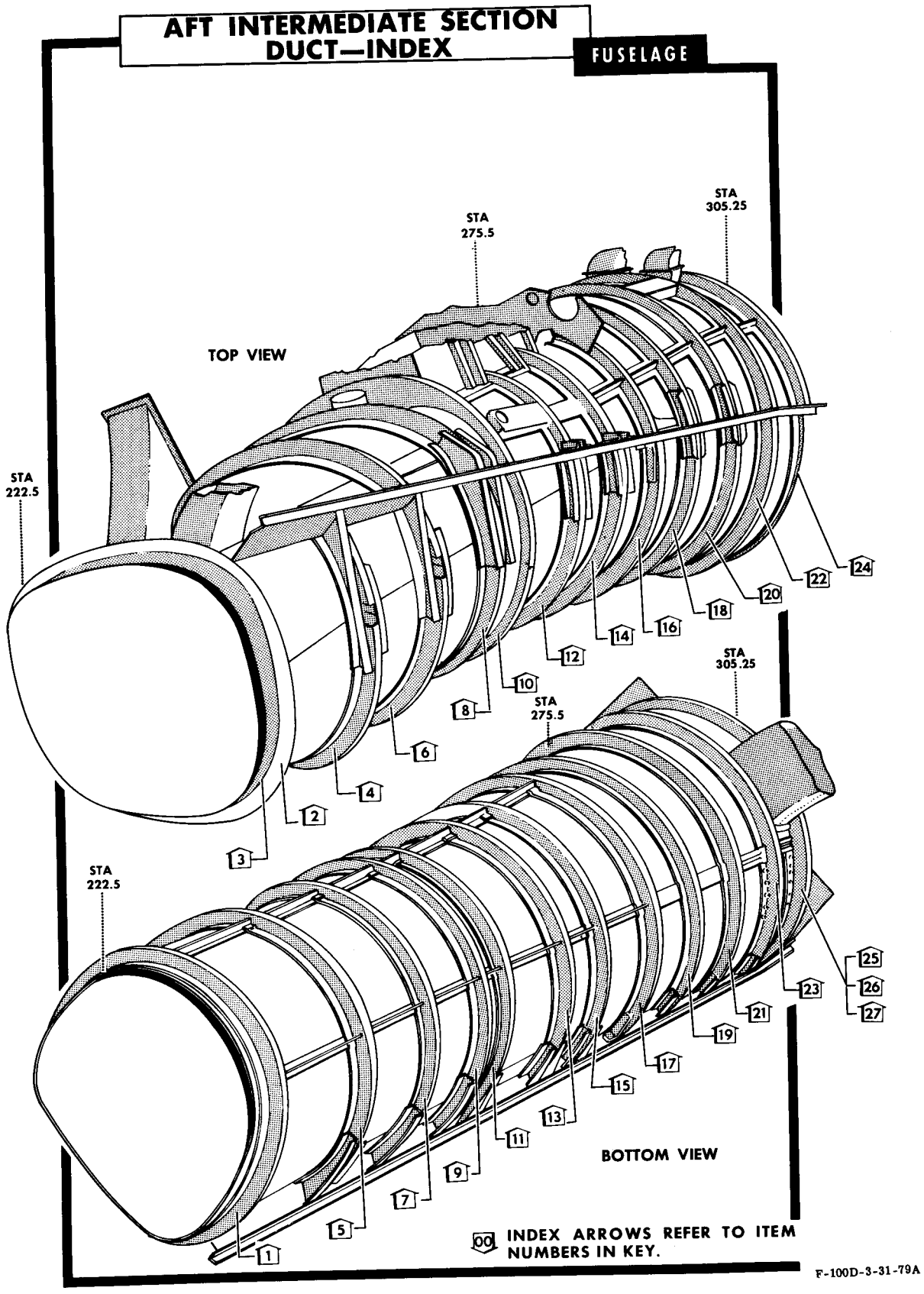


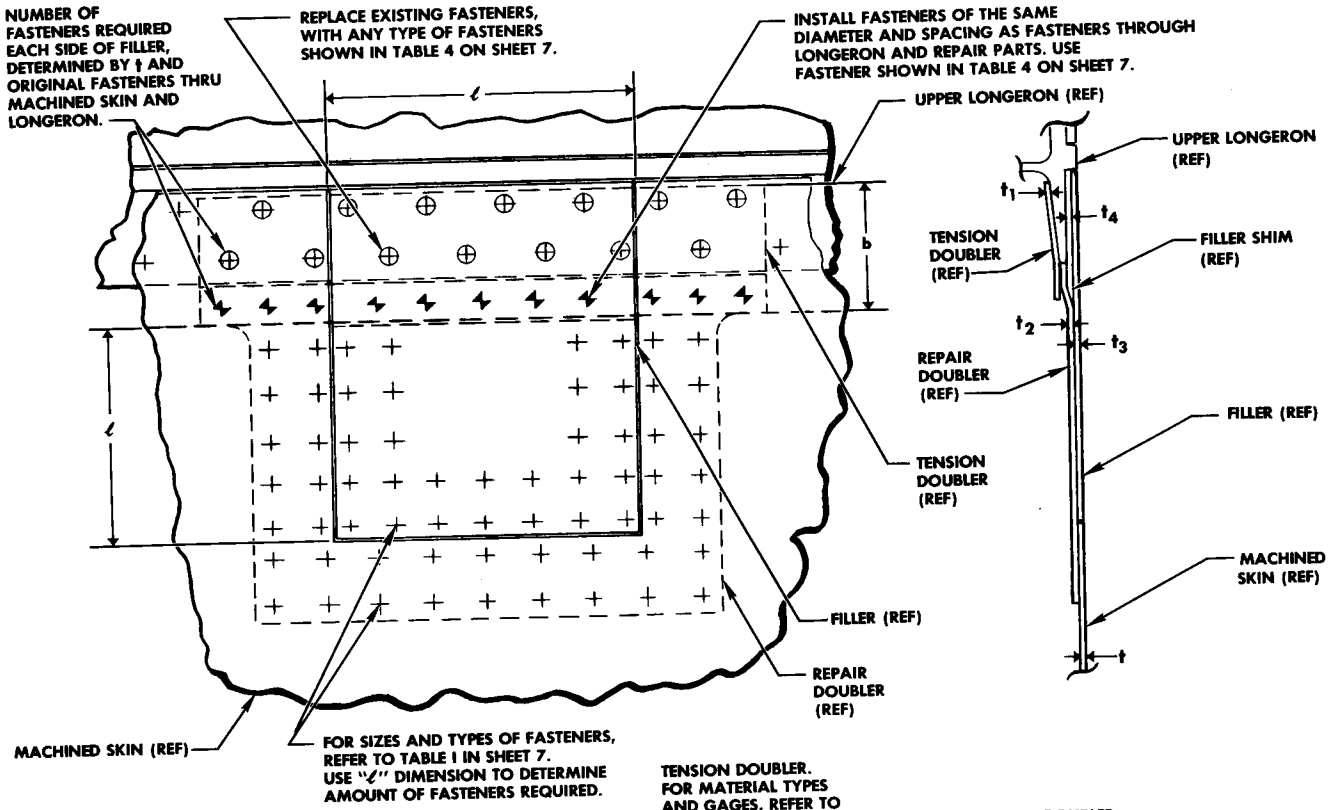
Figure 4-15. Aft Intermediate Section Repair Index (Sheet 2 of 2)

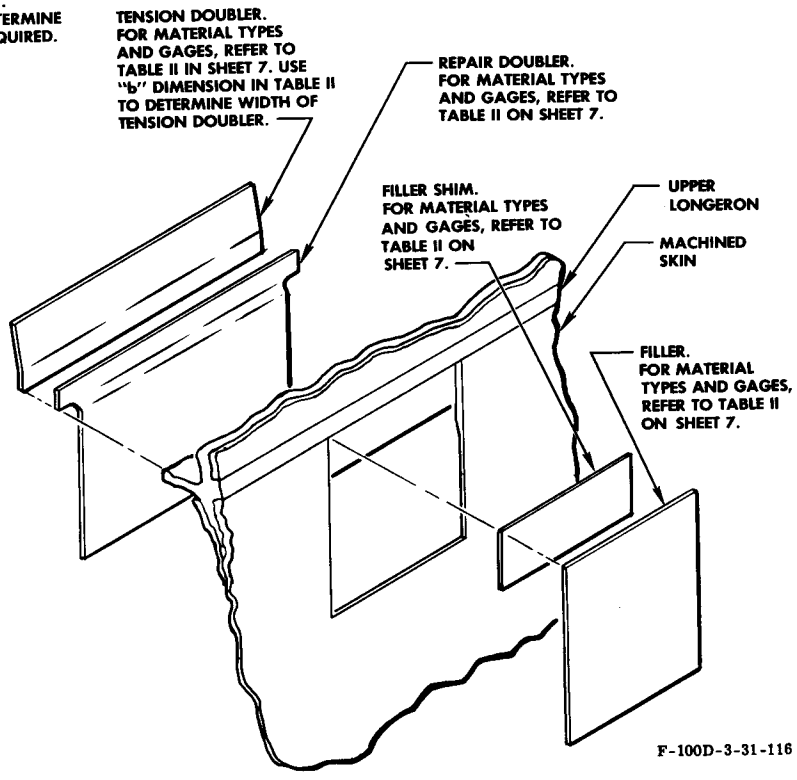
ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31270	Duct Angle—Lwr	Sta 222½	0.063	7075-T6 Clad	
2	192-31269	Duct Angle—Up.	Sta 222½	0.063	7075-T6 Clad	
3	192-316106	Duct Angle	Sta 221⅛	0.090	2024-T4 Clad	
4	192-316231	Duct Frame—Up.	Sta 231	0.080	7075-T6 Clad	
5	192-316232	Duct Frame—Lwr	Sta 231	0.090	7075-T6 Clad	
6	192-316239	Duct Frame Assy—Up.	Sta 239¾	0.071	7075-T6 Clad	
7	192-316240	Duct Frame Assy—Lwr	Sta 239¾	0.071	7075-T6 Clad	
8	192-316248	Duct Frame Assy—Up.	Sta 248½	0.071	7075-T6 Clad	
9	192-316249	Duct Frame Assy—Lwr	Sta 248½	0.071	7075-T6 Clad	
10	192-316254	Duct Frame—Up.	Sta 254⅛	0.040	7075-T6 Clad	
11	192-316255	Duct Frame—Lwr	Sta 254⅛	0.040	7075-T6 Clad	
12	192-316261	Duct Frame—Up.	Sta 261½	0.050	7075-T6 Clad	
13	192-316262	Duct Frame—Lwr	Sta 261½	0.050	7075-T6 Clad	
14	192-316268	Duct Frame Assy—Side	Sta 268½	0.050	7075-T6 Clad	
15	192-316269	Duct Frame Assy—Lwr	Sta 268½	0.050	7075-T6 Clad	
16	192-312160	Duct Angle—Up.	Sta 275½	4E344	7075-T6 Extr	
17	192-312161	Duct Frame—Lwr	Sta 275½	0.072 0.040	7075-T6 Clad	
18	192-316282	Duct Frame Assy—Up.	Sta 282¼	0.040	7075-T6 Clad	
19	192-316283	Duct Frame Assy—Lwr	Sta 282¼	0.040	7075-T6 Clad	
20	192-316290	Duct Frame Assy—Upr	Sta 290⅛	0.050	7075-T6 Clad	
21	192-316291	Duct Frame Assy—Lwr	Sta 290⅛	0.050	7075-T6 Clad	
22	192-316298	Duct Frame Assy	Sta 298	0.063	7075-T6 Clad	
23	192-316299	Duct Frame Assy—Lwr	Sta 298	0.063	7075-T6 Clad	
24	192-313124	Duct Frame—Up.	Sta 305¼	0.063	7075-T6 Clad	
25	192-313125	Duct Frame—Lwr	Sta 305¼	0.063	7075-T6 Clad	
26	192-313125	Duct Frame—Lwr	Sta 305¼	77B	7075-T6 Extr	
27	192-313125	Duct Frame—Lwr	Sta 305¼	1E233	7075-T6 Extr	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-15 (Sheet 2 of 2)

PERMANENT CATEGORY A — FUSELAGE





1. Trim out damage to a square or rectangular shape.
2. Smooth all corners and edges after trimming.
3. Make repair doubler, tension doubler, filler, and filler shim. Refer to Table II on sheet 7 for material types and gages. The amount of fasteners required determines the size of the repair doubler. Use "ℓ" dimension and Table I on sheet 7 to determine amount of fasteners required.
4. Fasten all repair parts in place. When spacing is not called out, the "ℓ" dimension and the amount of fasteners required determines the fastener spacing.

NOTE

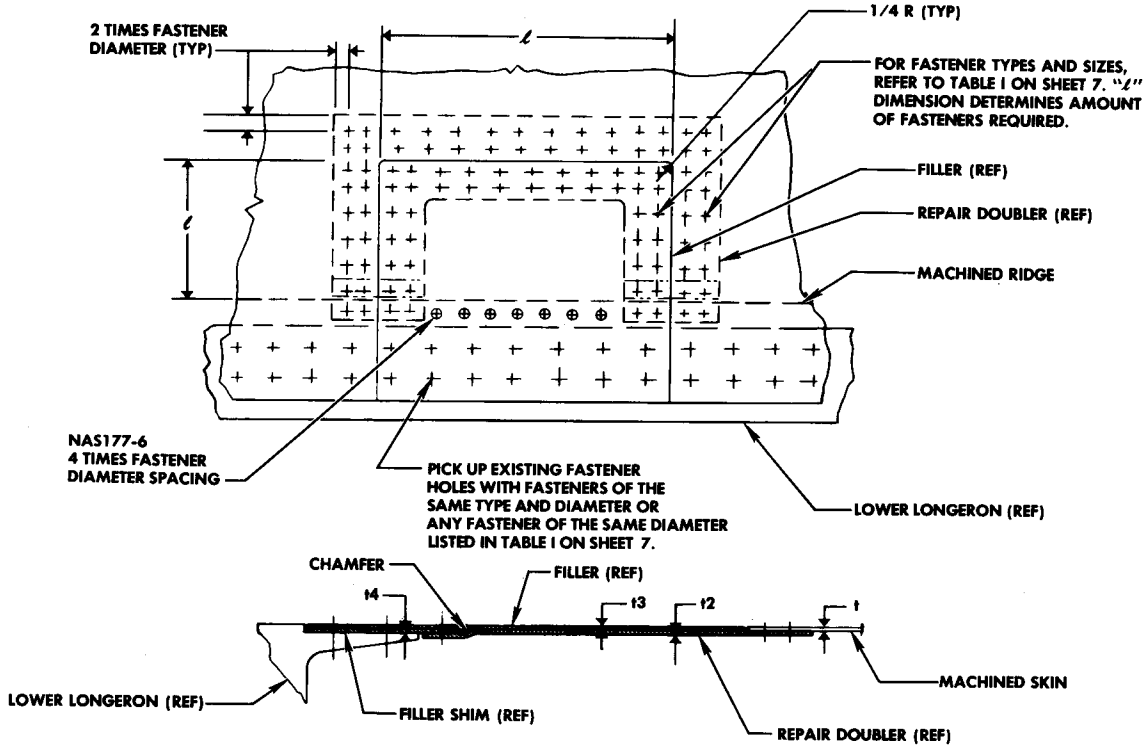
The repair is for the upper longeron only.

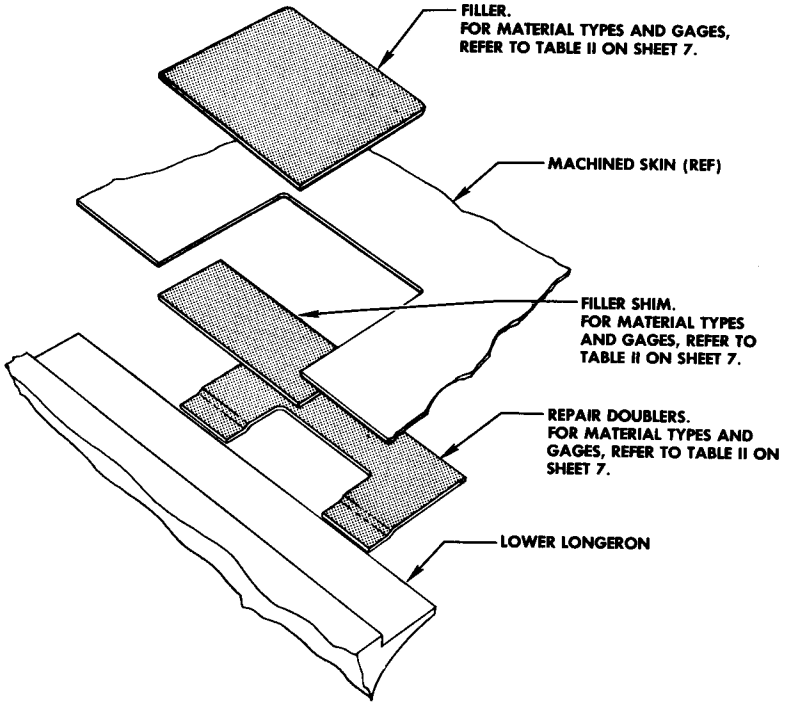
- Tension-type fasteners coded ϟ are not to be included in the shear repair fastener pattern.
- The minimum edge distance is two times the fastener diameter.
- The minimum fastener spacing is four times the fastener diameter.

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 1 of 7)

PERMANENT CATEGORY A

FUSELAGE





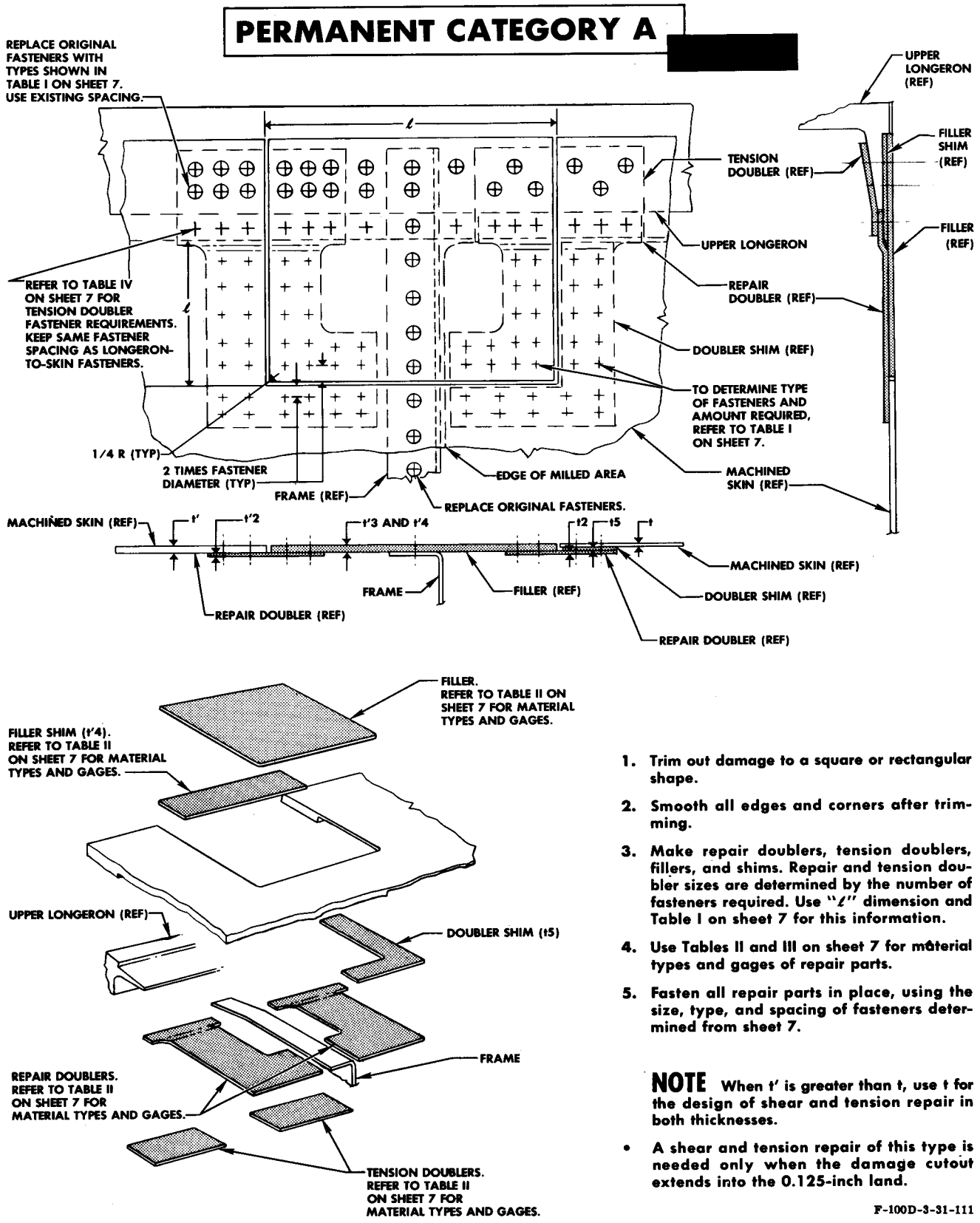
1. Trim out damage to a square or rectangular shape.
2. Smooth all edges and corners after trimming.
3. Make repair doubler, filler, and filler shim. The amount of fasteners required determines the size of the repair doubler. Use "ℓ" dimension and Table I on sheet 7 to determine number of fasteners required.
4. Fasten all repair parts in place, using fasteners selected from Table I on sheet 7 and fasteners called out on this page. When fastener spacing is not called out, the number of fasteners to be used and "ℓ" dimension determines the fastener spacing.

NOTE The minimum edge distance is two times the fastener diameter.

- The minimum fastener spacing is four times the fastener diameter.

F-100D-3-31-110

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 2 of 7)



1. Trim out damage to a square or rectangular shape.
2. Smooth all edges and corners after trimming.
3. Make repair doublers, tension doublers, fillers, and shims. Repair and tension doubler sizes are determined by the number of fasteners required. Use "ℓ" dimension and Table I on sheet 7 for this information.
4. Use Tables II and III on sheet 7 for material types and gages of repair parts.
5. Fasten all repair parts in place, using the size, type, and spacing of fasteners determined from sheet 7.

NOTE When t' is greater than t, use t for the design of shear and tension repair in both thicknesses.

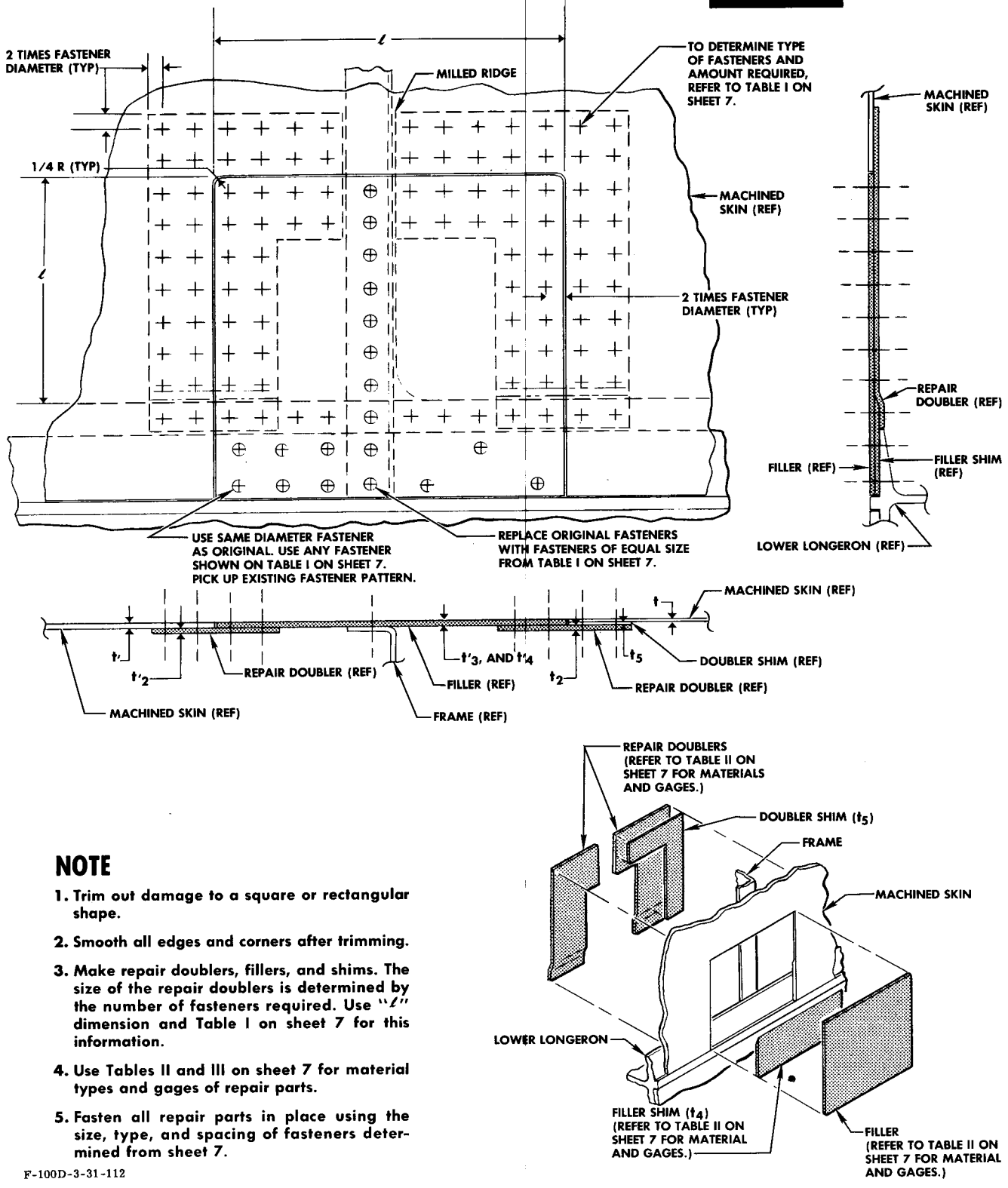
- A shear and tension repair of this type is needed only when the damage cutout extends into the 0.125-inch land.

F-100D-3-31-111

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 3 of 7)

PERMANENT CATEGORY A

FUSELAGE



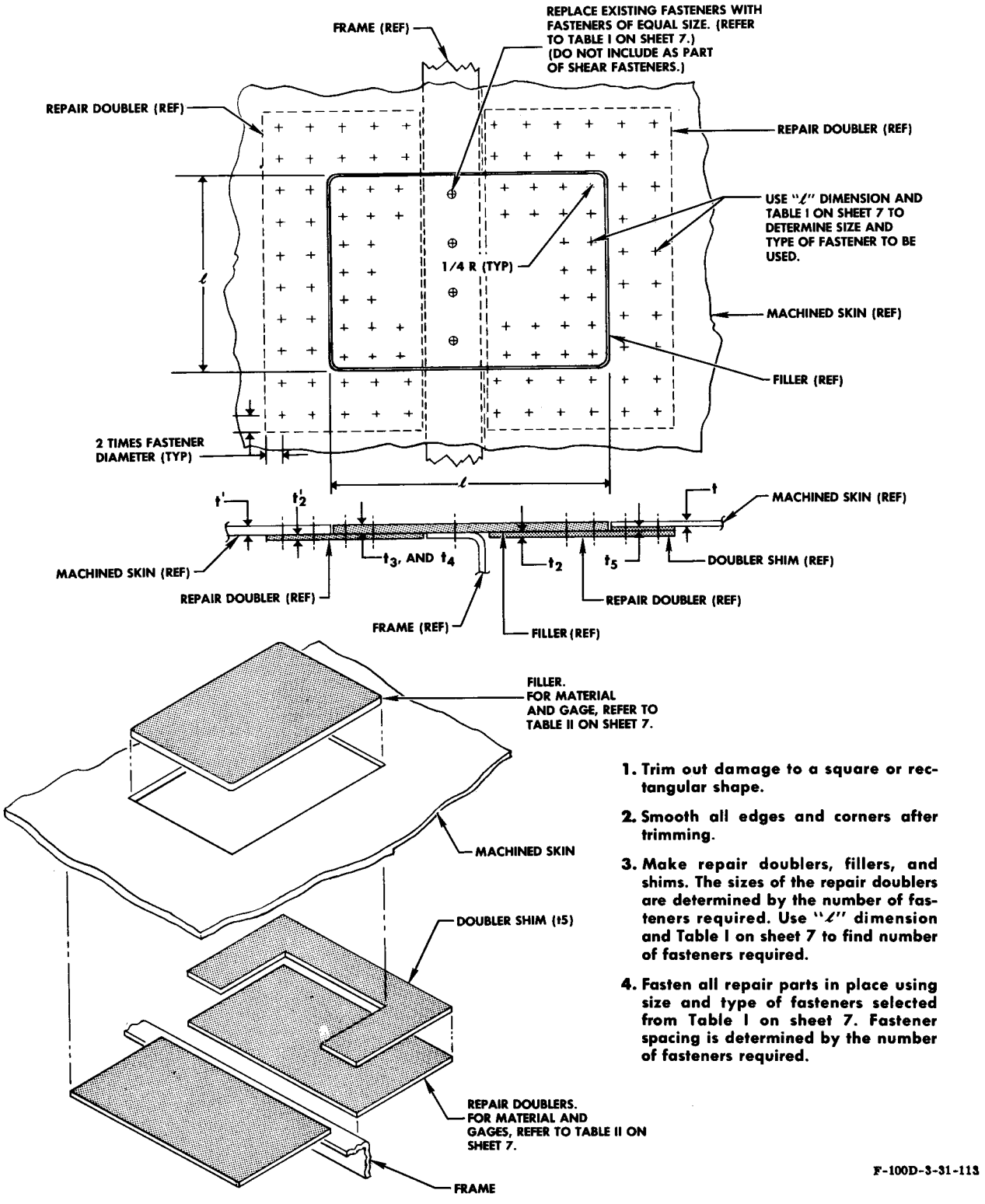
NOTE

1. Trim out damage to a square or rectangular shape.
2. Smooth all edges and corners after trimming.
3. Make repair doublers, fillers, and shims. The size of the repair doublers is determined by the number of fasteners required. Use "ℓ" dimension and Table I on sheet 7 for this information.
4. Use Tables II and III on sheet 7 for material types and gages of repair parts.
5. Fasten all repair parts in place using the size, type, and spacing of fasteners determined from sheet 7.

F-100D-3-31-112

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 4 of 7)

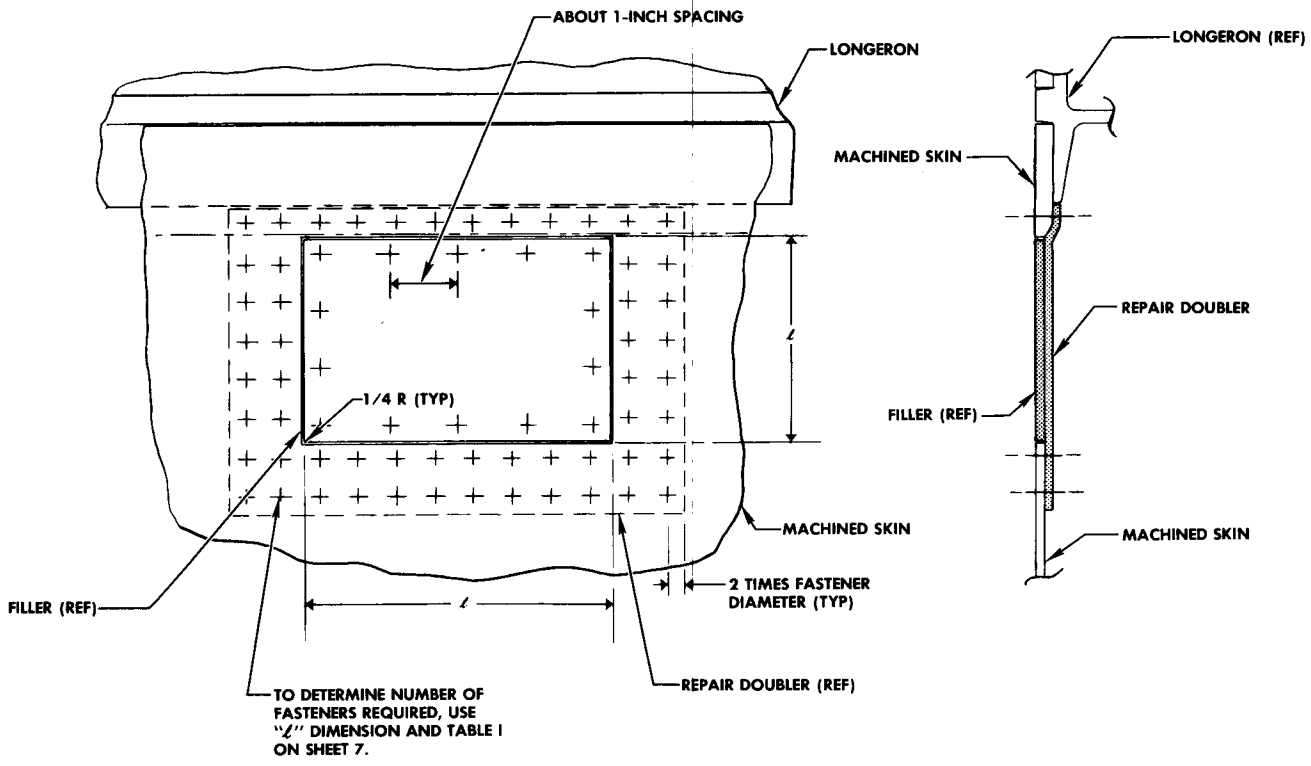
PERMANENT CATEGORY A — FUSELAGE



1. Trim out damage to a square or rectangular shape.
2. Smooth all edges and corners after trimming.
3. Make repair doublers, fillers, and shims. The sizes of the repair doublers are determined by the number of fasteners required. Use "ℓ" dimension and Table I on sheet 7 to find number of fasteners required.
4. Fasten all repair parts in place using size and type of fasteners selected from Table I on sheet 7. Fastener spacing is determined by the number of fasteners required.

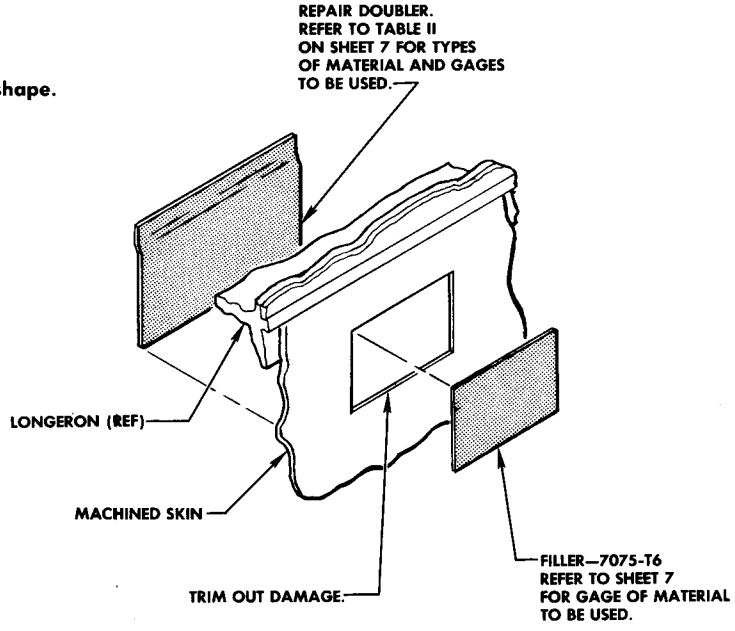
Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 5 of 7)

PERMANENT CATEGORY A — FUSELAGE



NOTE Trim out damage to a square or rectangular shape.

- Smooth all edges and corners after trimming.
- Make repair doubler. The dimensions of the repair doubler will depend on the number of fasteners to be used and the rows of fasteners required. Refer to Table I on sheet 7 for the fastener requirements.
- Make filler using information furnished in Table II on sheet 7 for material thickness.
- Fasten repair doubler and filler in place. The filler fasteners may be either conventional or blind fasteners 5/32 inch in diameter, with about one-inch spacing.



F-100D-3-31-114

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 6 of 7)

PERMANENT CATEGORY A — FUSELAGE

TABLE I

	ℓ	FASTENERS REQUIRED											
		1/2			1			2			4		
RIVET	t	0.076	0.098	0.125	0.076	0.098	0.125	0.076	0.088	0.125	0.076	0.098	0.125
DD6		2	2	2	3	3	3	5	5	6	10	10	12
DD8		—	1	1	—	2	2	—	4	4	—	7	7
NAS177-6		1	1	1	2	2	2	3	3	3	6	5	6
NAS177-8		1	1	1	1	1	1	2	2	2	4	4	4
NAS177-10		1	1	1	1	1	1	2	2	2	4	3	3
HS47-6		1	1	1	2	2	2	3	3	3	6	5	5
AN509-10		1	1	1	2	2	2	4	3	3	7	7	6
AN509-416		1	1	1	2	2	2	3	3	3	6	5	5
AN509-516		—	1	1	—	1	1	—	2	2	—	5	4
F164		1	1	1	2	2	2	4	4	4	8	7	8
F200		1	1	1	2	2	2	4	3	3	7	7	6
F260	BLIND	1	1	1	2	2	2	3	3	3	6	5	5
F312		—	1	1	—	1	1	—	2	2	—	5	4
FA200		1	1	1	2	2	2	4	4	5	8	8	9
FA262		1	1	1	2	2	2	3	3	3	6	5	6

TABLE II

REPAIR MATERIALS AND THICKNESSES				
	MATL t	0.076	0.098	0.125
t1	7075-T6 CLAD	0.0125	0.125	0.125
	1/2 H 301 CRES	0.050	0.050	0.050
t2	7075-T6 CLAD	0.063	0.080	0.125
	1/2 H 301 CRES	0.036	0.040	0.050
t3	7075-T6 CLAD	0.080	0.100	0.125
t4	7075-T6 CLAD	0.050	0.025	NONE
b		2-13/16	2-13/16	3-5/16

TABLE III

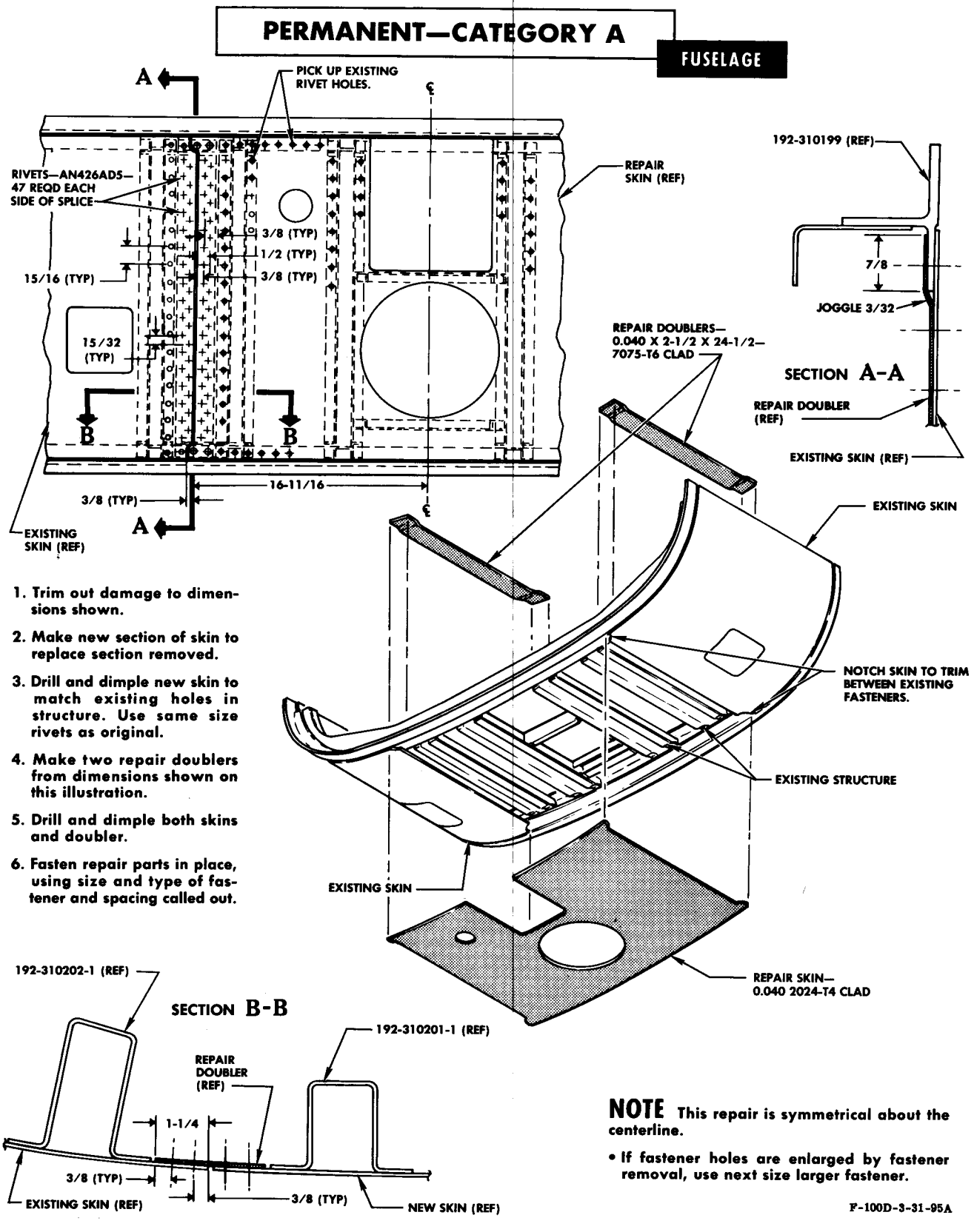
SHIM 2024-T4 CLAD		
	t5	
t' \ t	0.076	0.098
0.125	0.050	0.025
0.098	0.025	

TABLE IV

TENSION AND SHEAR FASTENERS REQUIRED												
*SINGLE ROW												
t	0.076				0.098				0.125			
ORIGINAL DIA	3/16		1/4		3/16		1/4		3/16		1/4	
ROWS	LONGN	SKIN	LONGN	SKIN	LONGN	SKIN	LONGN	SKIN	LONGN	SKIN	LONGN	SKIN
HS47-6 NAS333	6	—		—	6	—		—	6	6		6
HS47-8 NAS334		3*	3	3*		3*	3	3*		—	3	—
F200 AN509-10	8	—		—	8	—		—	6	6		8
F260 AN509-416		4*	5	5*		4*	6	6*		—	4	—

F-100D-3-31-115

Figure 4-15A. Machined Side Skin Repair—Station 222.5 to 389 (Sheet 7 of 7)



1. Trim out damage to dimensions shown.
2. Make new section of skin to replace section removed.
3. Drill and dimple new skin to match existing holes in structure. Use same size rivets as original.
4. Make two repair doublers from dimensions shown on this illustration.
5. Drill and dimple both skins and doubler.
6. Fasten repair parts in place, using size and type of fastener and spacing called out.

NOTE This repair is symmetrical about the centerline.

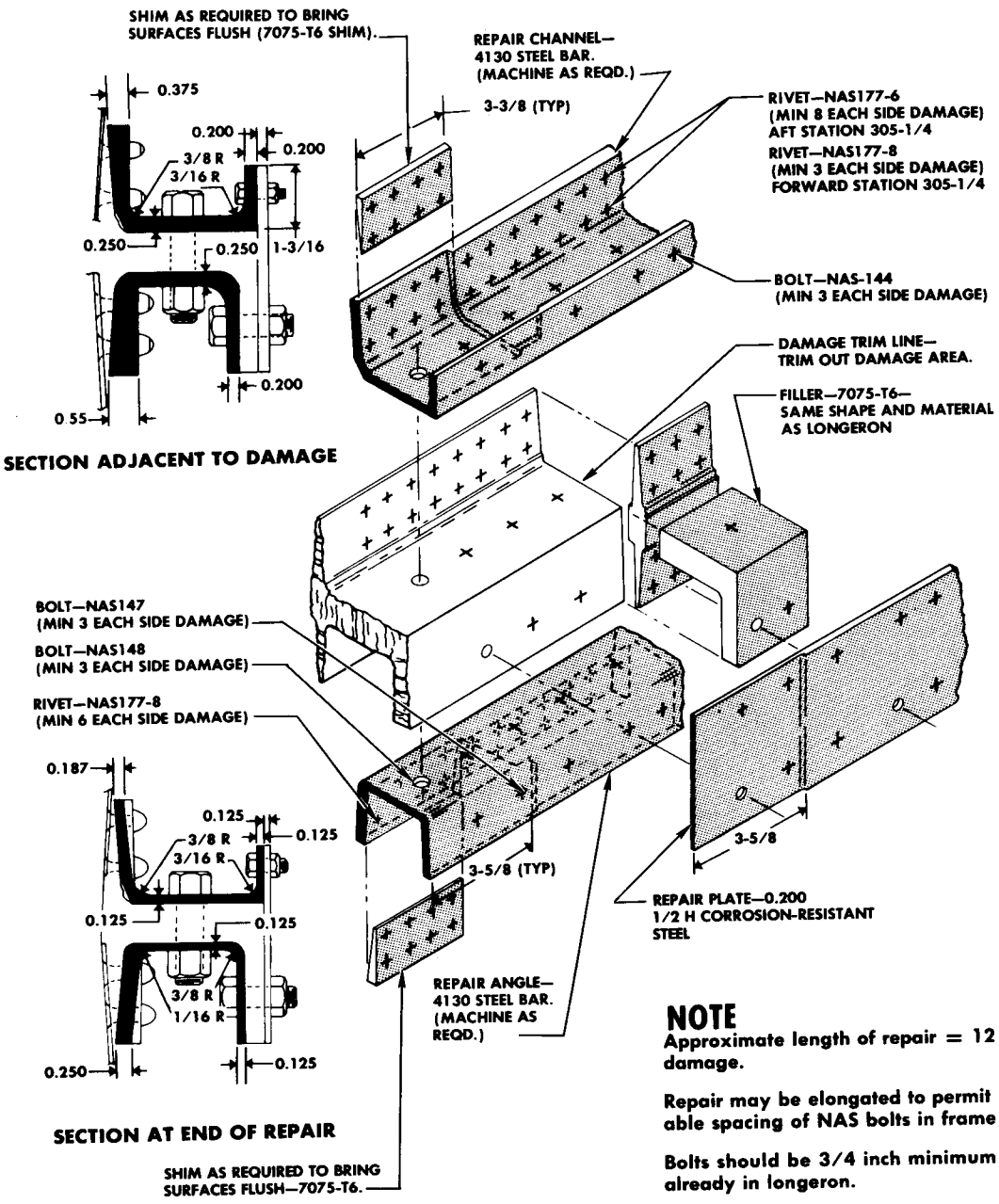
- If fastener holes are enlarged by fastener removal, use next size larger fastener.

F-100D-3-31-95A

Figure 4-15B. Skin Cracks—Fuel Bay Access Door

PERMANENT—CATEGORY B

FUSELAGE



NOTE

Approximate length of repair = 12 inches plus damage.

Repair may be elongated to permit more desirable spacing of NAS bolts in frame areas.

Bolts should be 3/4 inch minimum from holes already in longeron.

Heat-treat 4130 repair channels to 125,000 to 140,000 psi after milling.

Only half of repair shown; symmetrical each side of damage.

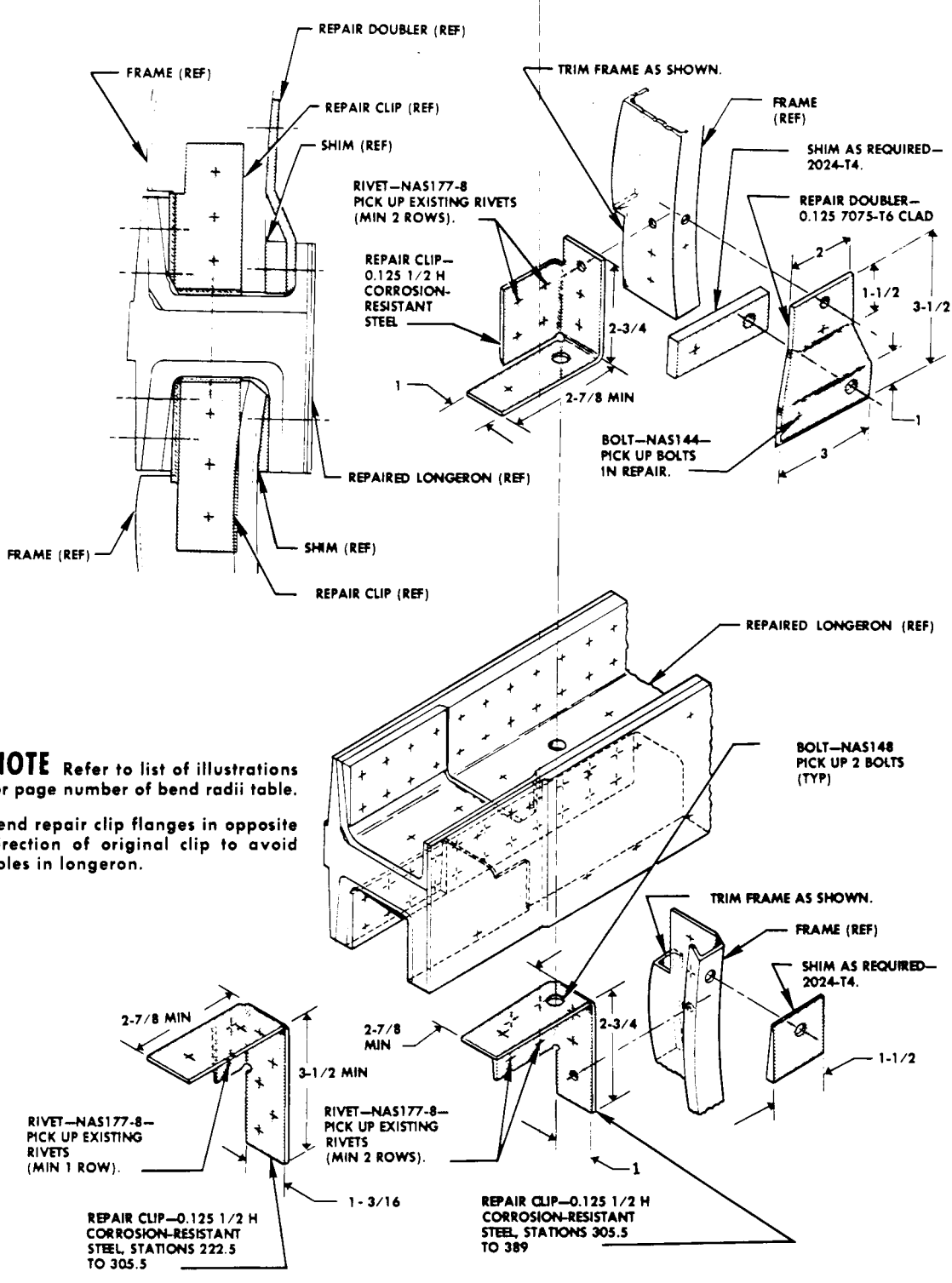
Do not run repair forward of station 233, aft of station 381, or through stations 275-1/2, 305-1/4, 325-5/8, and 334.

F-100D-3-31-41A

Figure 4-16. Upper Longeron—Station 222.5 to 389 (Sheet 1 of 2)

PERMANENT—CATEGORY B

FUSELAGE



NOTE Refer to list of illustrations for page number of bend radii table.

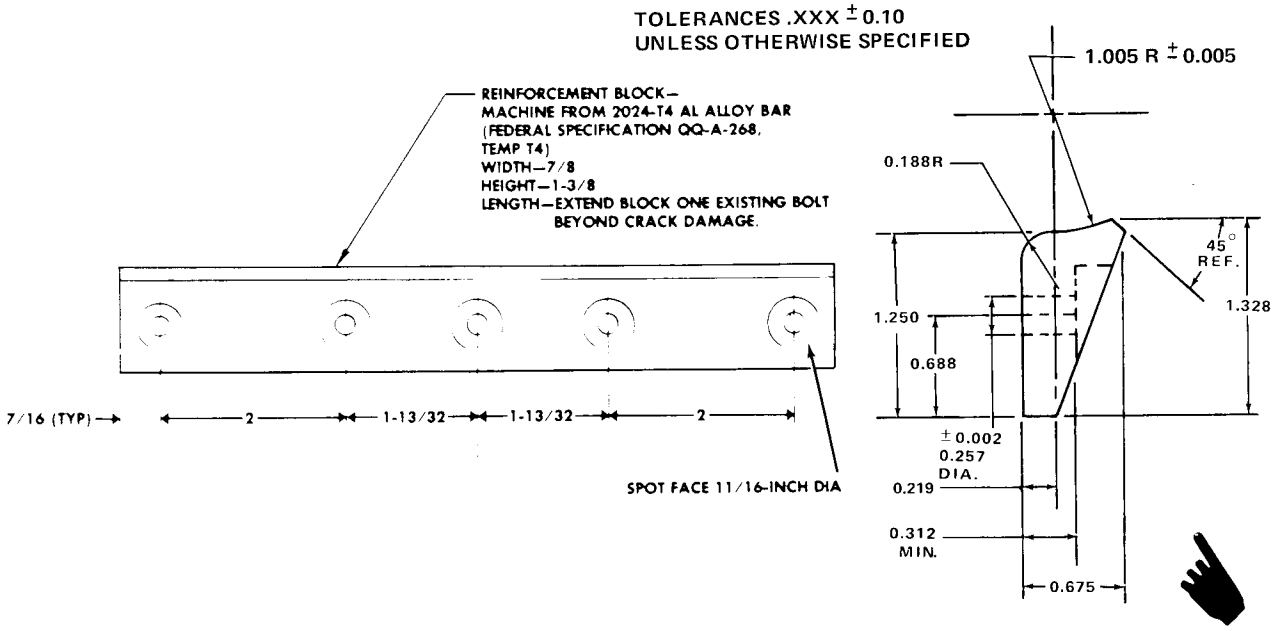
Bend repair clip flanges in opposite direction of original clip to avoid holes in longeron.

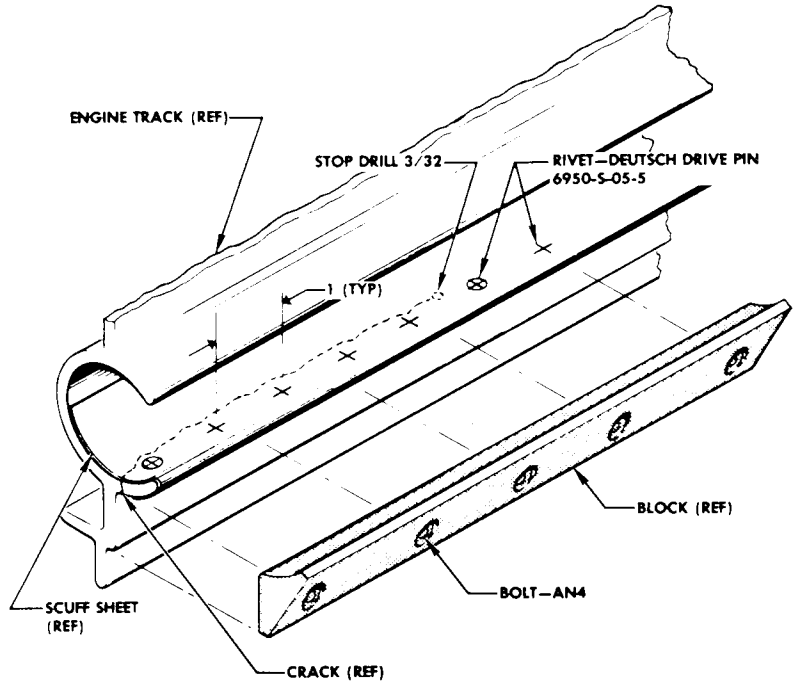
F-100D-3-31-42A

Figure 4-16. Upper Longeron—Station 222.5 to 389 (Sheet 2 of 2)

PERMANENT CATEGORY A

FUSELAGE





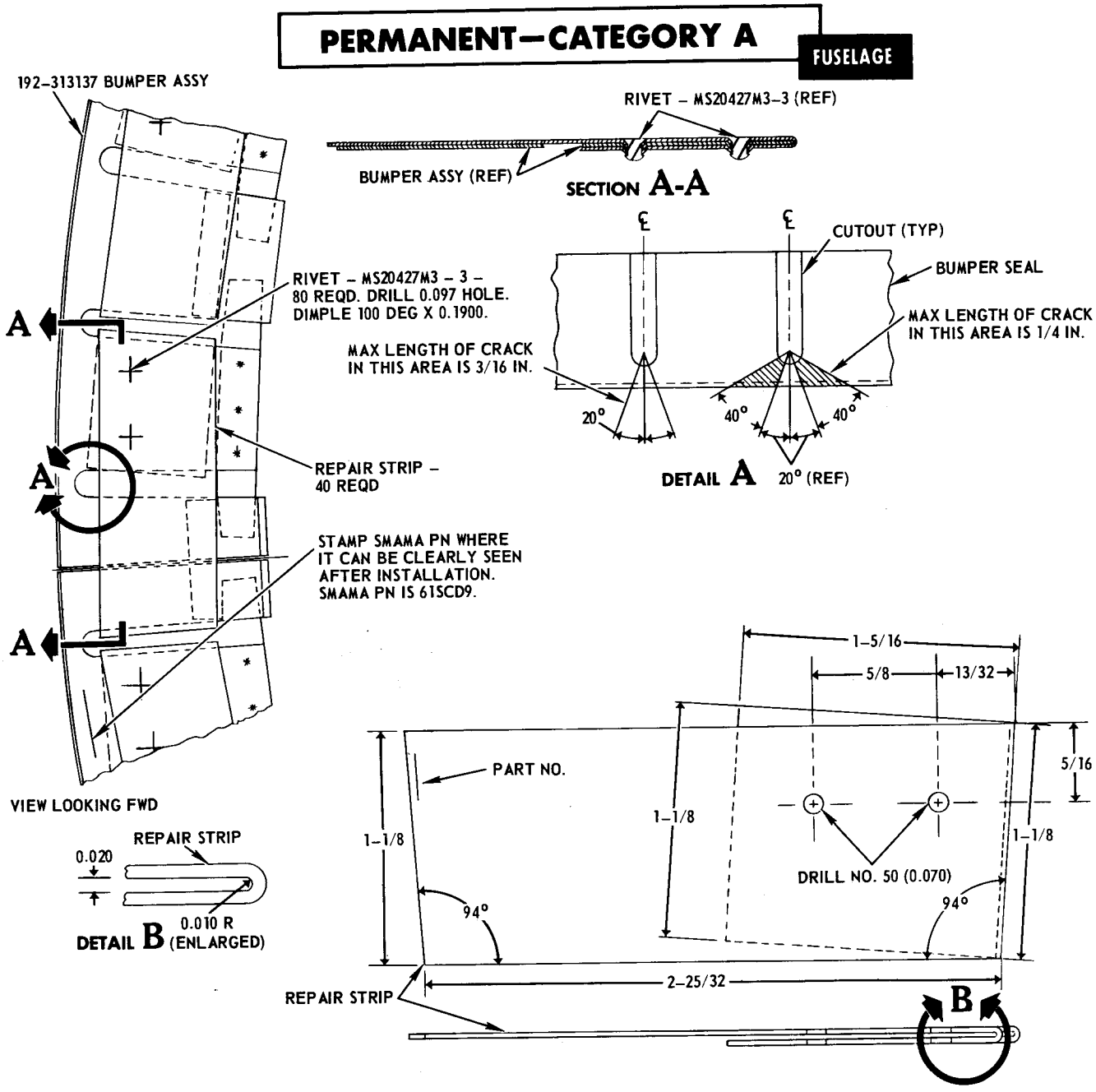
1. **Stop-drill crack 3/32-inch diameter. Fabricate reinforcing block. Machine block as required to nest it to track.**
2. **Drill block and spot-face it 11/16-inch diameter. Maintain 5/16-inch minimum thickness after spot-facing.**
3. During installation of block, drill 0.125 diameter holes approximately 0.062 deep, to provide clearance for the scuff sheet rivets in the block. This will permit block to be placed against the curved part of the track.
4. Install block, picking up AN4 bolt pattern as shown. Drill and countersink track. Dimple stainless-steel scuff sheet and install rivets, maintaining approximately one-inch equal spacing. Pick up existing rivets.

NOTE

Replace track where crack damage exceeds 10 inches in length.

F-100D-3-31-134

Figure 4-16A. Engine Support Track Repair



Engine bumper seal replacement and repair criteria are as follows:

1. Bumper assemblies with one or more missing tabs must be removed and replaced with like serviceable parts.
2. Bumper assemblies with worn tabs that are not chafed through more than 50 percent of the metal thickness and are not wider than 3/8 inch across any one tab are acceptable as is.

NOTE The wear area is approximately in the center of the tab and may cover a short arc, or the complete circumference, depending on the engine rubbing area.

3. Bumper assemblies that have exceeded the wear tolerance noted must be repaired as shown.
4. Bumper assemblies with small cracks emanating from the tab cutout radii may be acceptable under the following conditions:
   - Length of cracks must not exceed limits shown in Detail A.
   - Cracks must be stop-drilled with a No. 40 drill. Exercise care when drilling so as not to drill into airplane structure.

F-100D-3-31-162

Figure 4-16B. Engine Bumper Seal Assembly Repair—Station 308.87

PERMANENT CATEGORY B

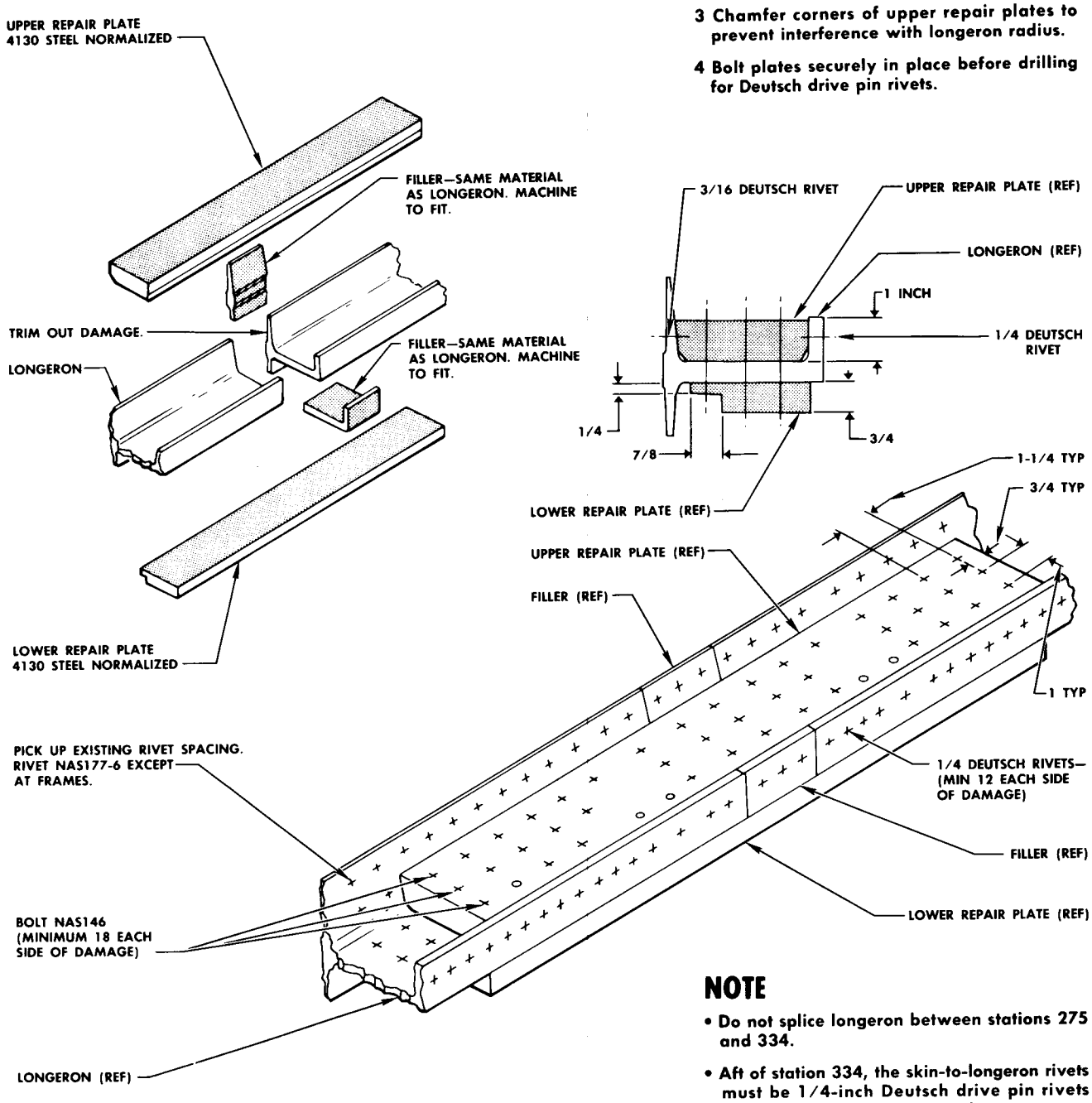
FUSELAGE

1 Trim out damage midway between existing fasteners.

2 Make fillers and repair plates. The length of the damage is 16 inches plus the damage.

3 Chamfer corners of upper repair plates to prevent interference with longeron radius.

4 Bolt plates securely in place before drilling for Deutsch drive pin rivets.



NOTE

- Do not splice longeron between stations 275 and 334.
- Aft of station 334, the skin-to-longeron rivets must be 1/4-inch Deutsch drive pin rivets where they can not be bucked.

F-100D-3-31-43A

Figure 4-17. Lower Longerons—Station 222.5 to 389 (Sheet 1 of 2)

PERMANENT CATEGORY A — FUSELAGE

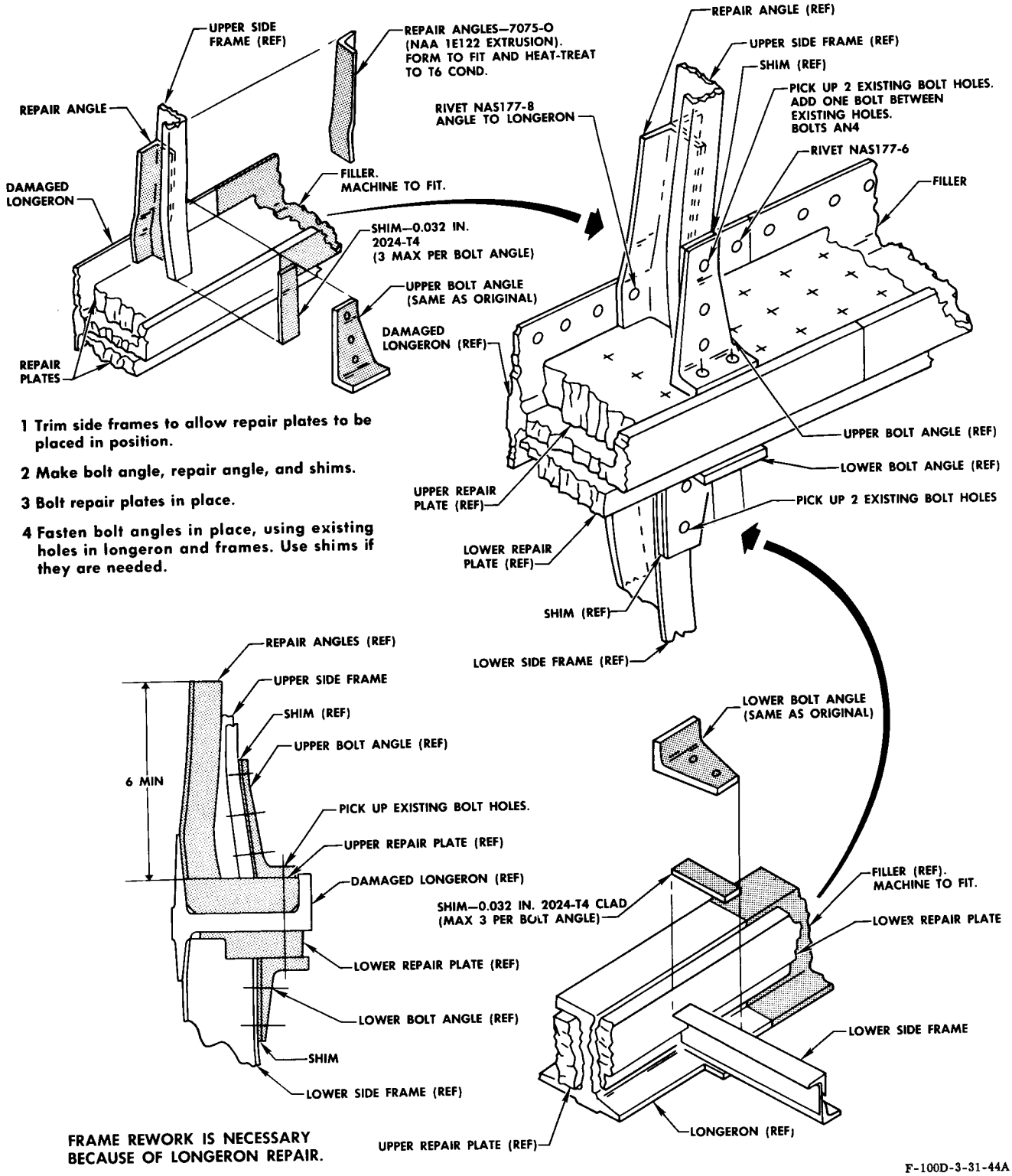


Figure 4-17. Lower Longerons—Station 222.5 to 389 (Sheet 2 of 2)

PERMANENT - CATEGORY A

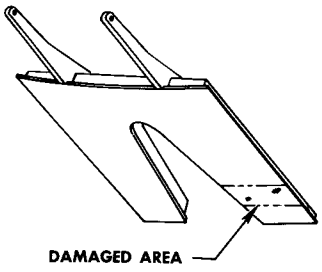
FUSELAGE

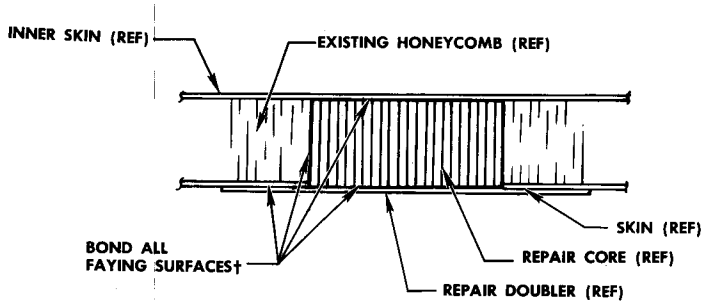
* If damage is such that edge distance is less than overlap "D" (sheet 2), extend doubler to edge of brake. Remove original rivets along edge and attach doubler to brake with adhesive and rivets through original holes.

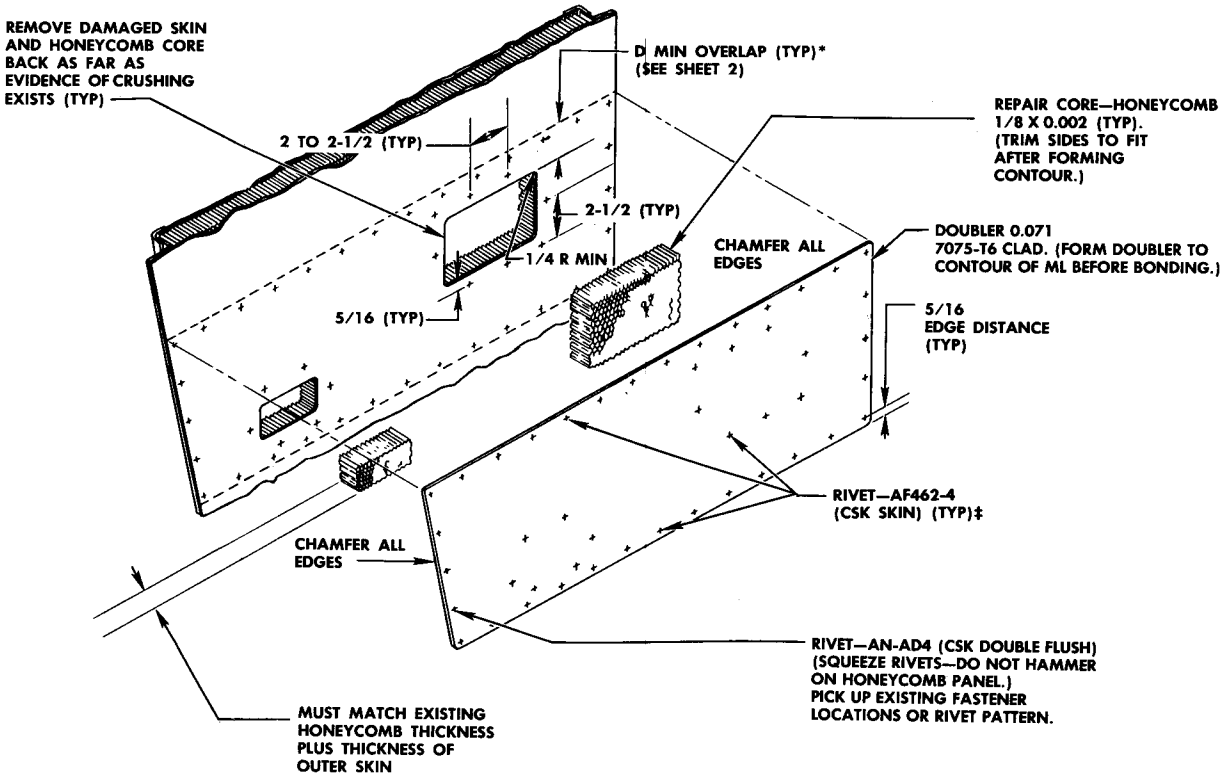
† Bond according to instructions contained in "Honeycomb Sandwich Panels." If scratches go through cladding, blend scratches and extend doubler in accordance with overlap "D" (sheet 2).

‡ All riveting is to be done before cure. This repair is applicable to the inner and/or the outer skin and core.

NOTE Maximum skin cutout must not exceed 8 inches.
Ribbon direction of repair core must match ribbon direction of damaged area.





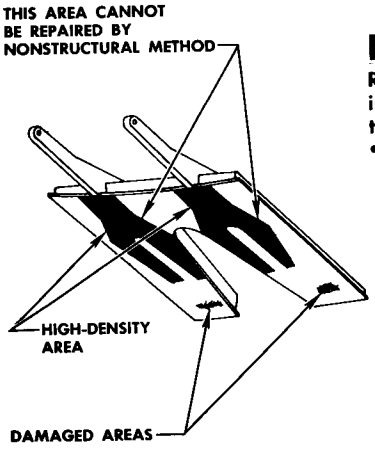


F-100D-3-31-19B

Figure 4-17A. Speed Brake—Honeycomb (Sheet 1 of 4)

PERMANENT—CATEGORY A

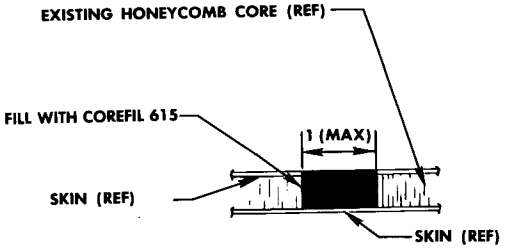
FUSELAGE



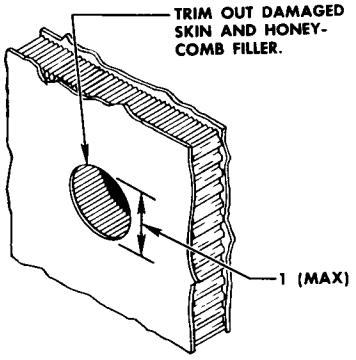
NOTE

Refer to "Honeycomb Sandwich Panels." For information on specific repair procedure, refer to "Nonstructural Repair."

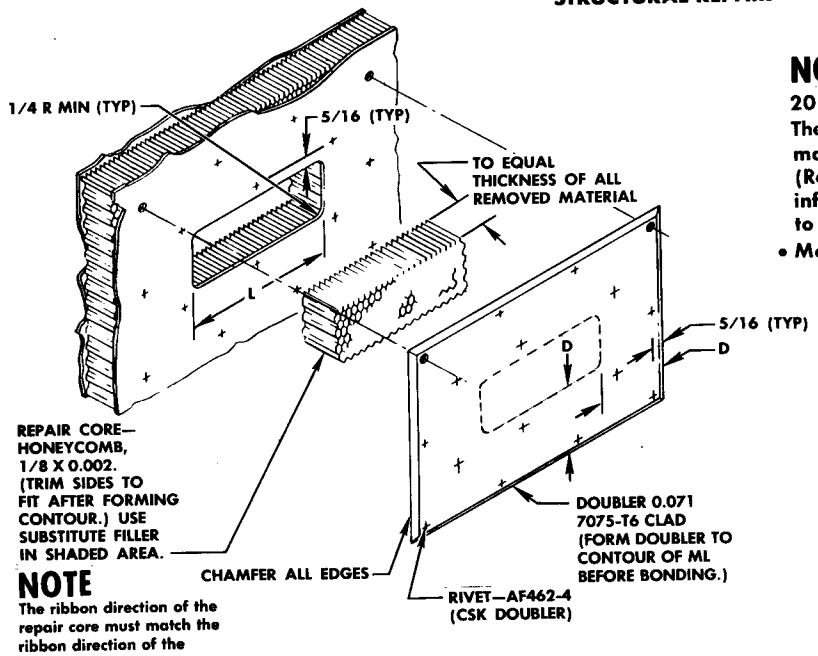
- There must be a minimum of 12 inches between this repair and any other nonstructural repair.



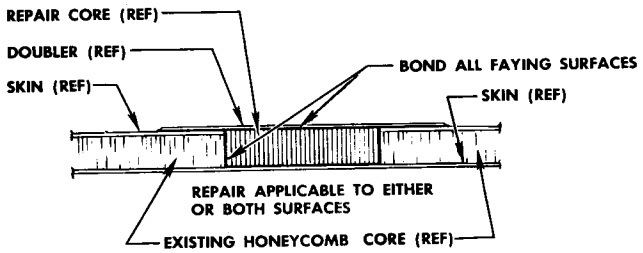
NONSTRUCTURAL REPAIR



STRUCTURAL REPAIR

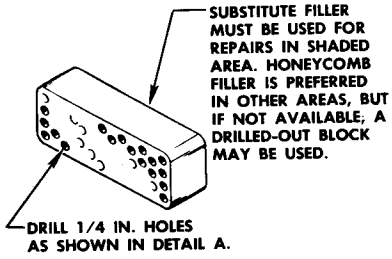


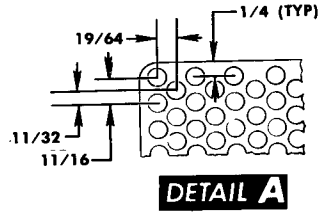
NOTE The ribbon direction of the repair core must match the ribbon direction of the damaged area.



NOTE The filler block may be made from 2014-T6, 2024-T4, 2024-T42, or 7075-T6 material. The holes should be drilled before the block is machined to match the contour of the part. (Refer to "Honeycomb Sandwich Panels.") For information on specific repair procedure, refer to "Structural Repair."

- Maximum skin cutout must not exceed 8 inches





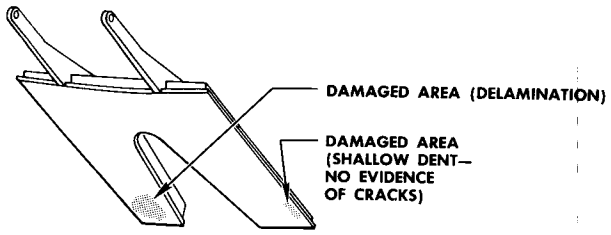
HOLE "L"	2 IN.	4 IN.	6 IN.	8 IN.
OVERLAP "D"	2 IN.	2.5 IN.	2.75 IN.	3 IN.

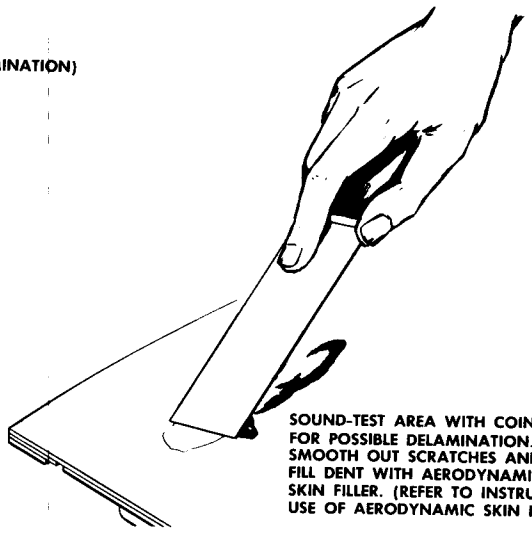
F-100D-3-31-20B

Figure 4-17A. Speed Brake—Honeycomb (Sheet 2 of 4)

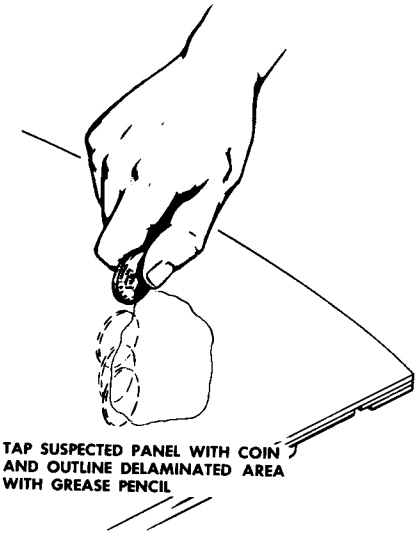
PERMANENT-CATEGORY A

FUSELAGE





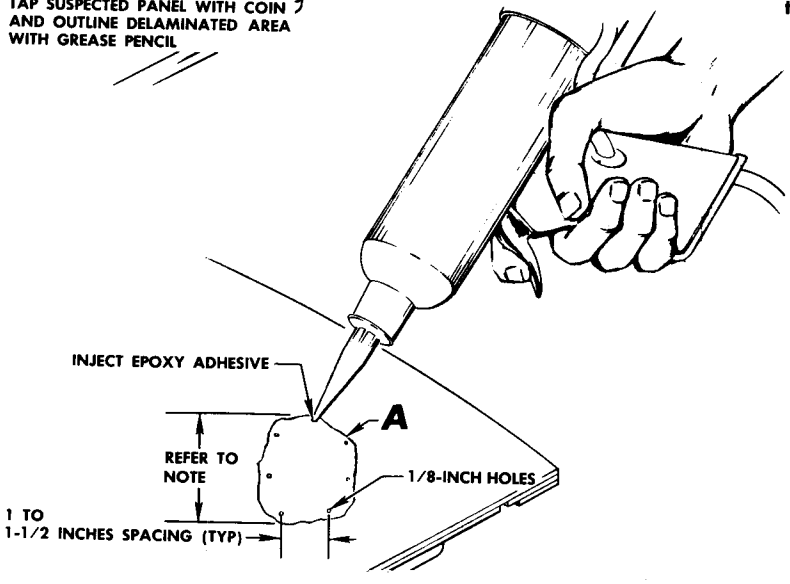
SOUND-TEST AREA WITH COIN FOR POSSIBLE DELAMINATION. SMOOTH OUT SCRATCHES AND FILL DENT WITH AERODYNAMIC SKIN FILLER. (REFER TO INSTRUCTIONS FOR USE OF AERODYNAMIC SKIN FILLER.)

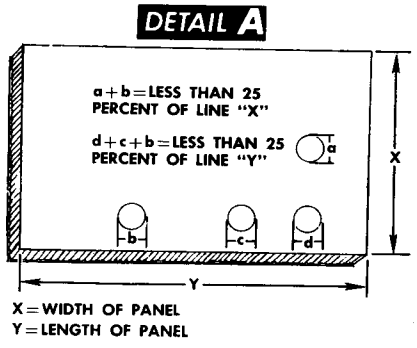


TAP SUSPECTED PANEL WITH COIN AND OUTLINE DELAMINATED AREA WITH GREASE PENCIL

NOTE

Refer to "Honeycomb Sandwich Panels." For information on specific repair procedure, refer to "Repair of Class IV Damages." The maximum reparable delamination is 10 square inches or a circle about 3 inches in diameter. The length of one delamination, or the total lengths of several, may not exceed 25 percent of the dimensions of the panel in any direction. (See detail A.)



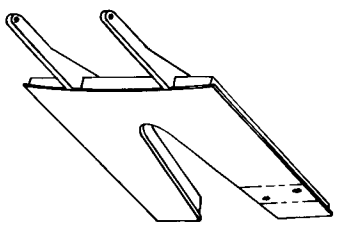


F-100D-3-31-21B

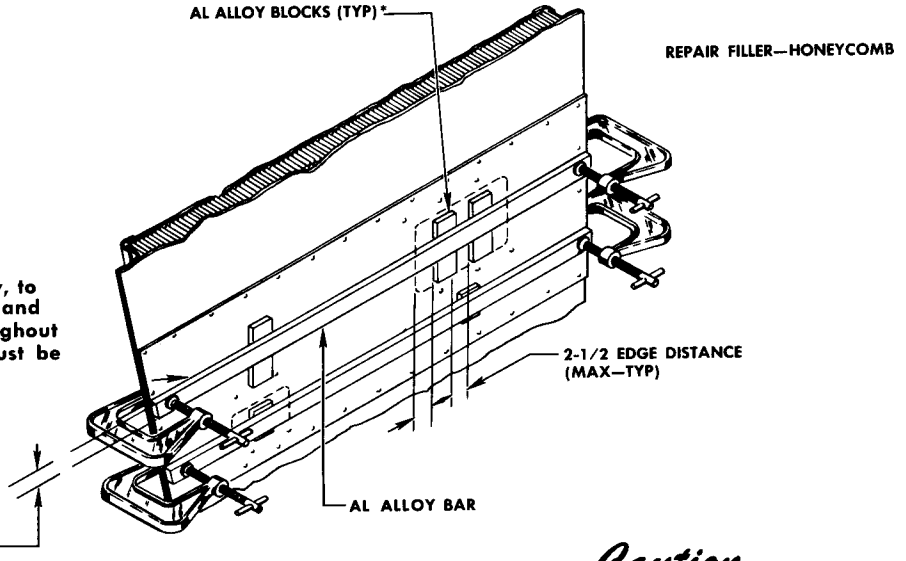
Figure 4-17A. Speed Brake—Honeycomb (Sheet 3 of 4)

PERMANENT—CATEGORY A

FUSELAGE



NOTE *Blocks are used, as necessary, to ensure contact with the original skin and honeycomb repair filler. Contact throughout the glue line is mandatory. Blocks must be contoured to match mold line.



Caution

Do not exceed 180°F when using heat lamps on adhesive repairs.

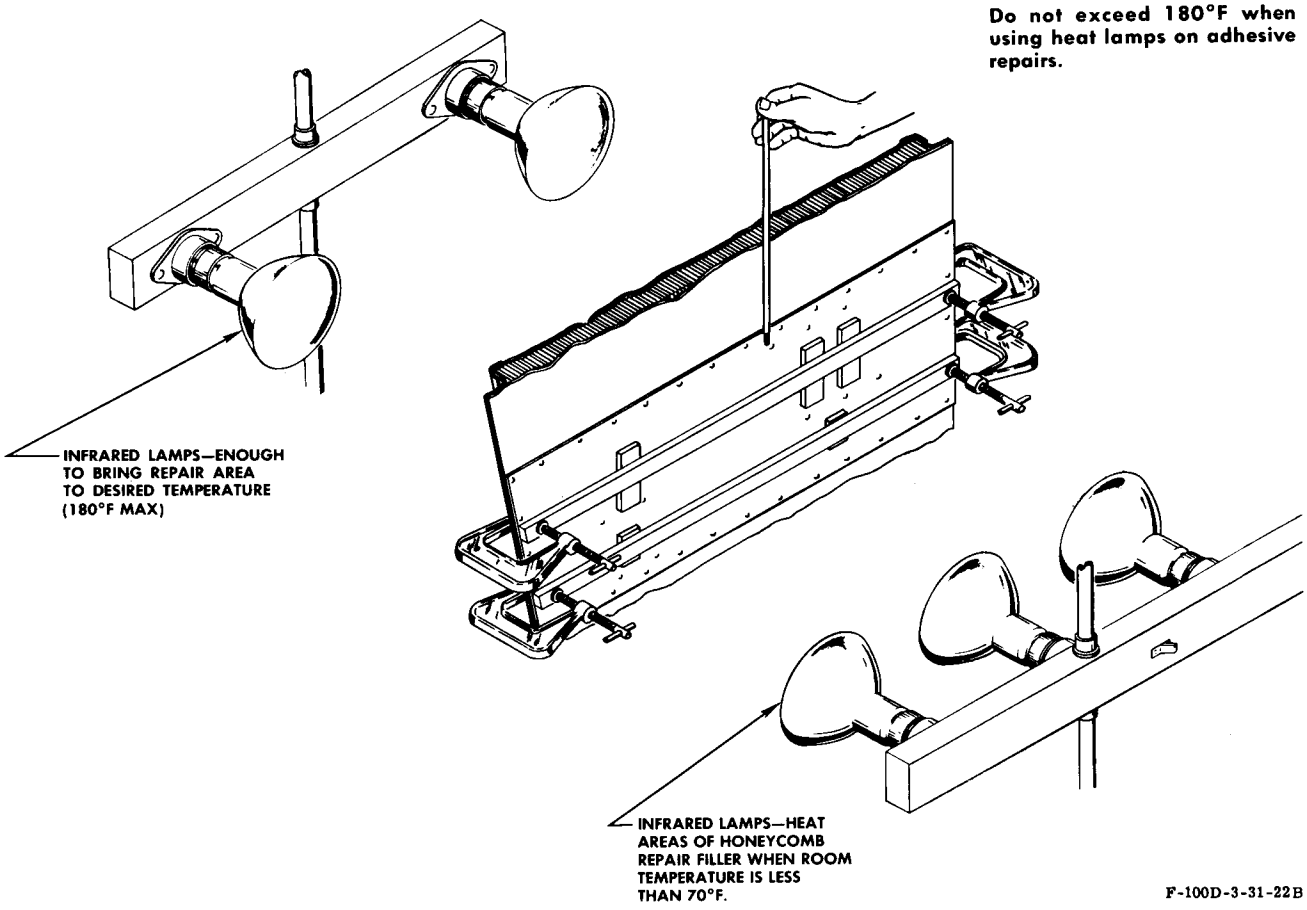
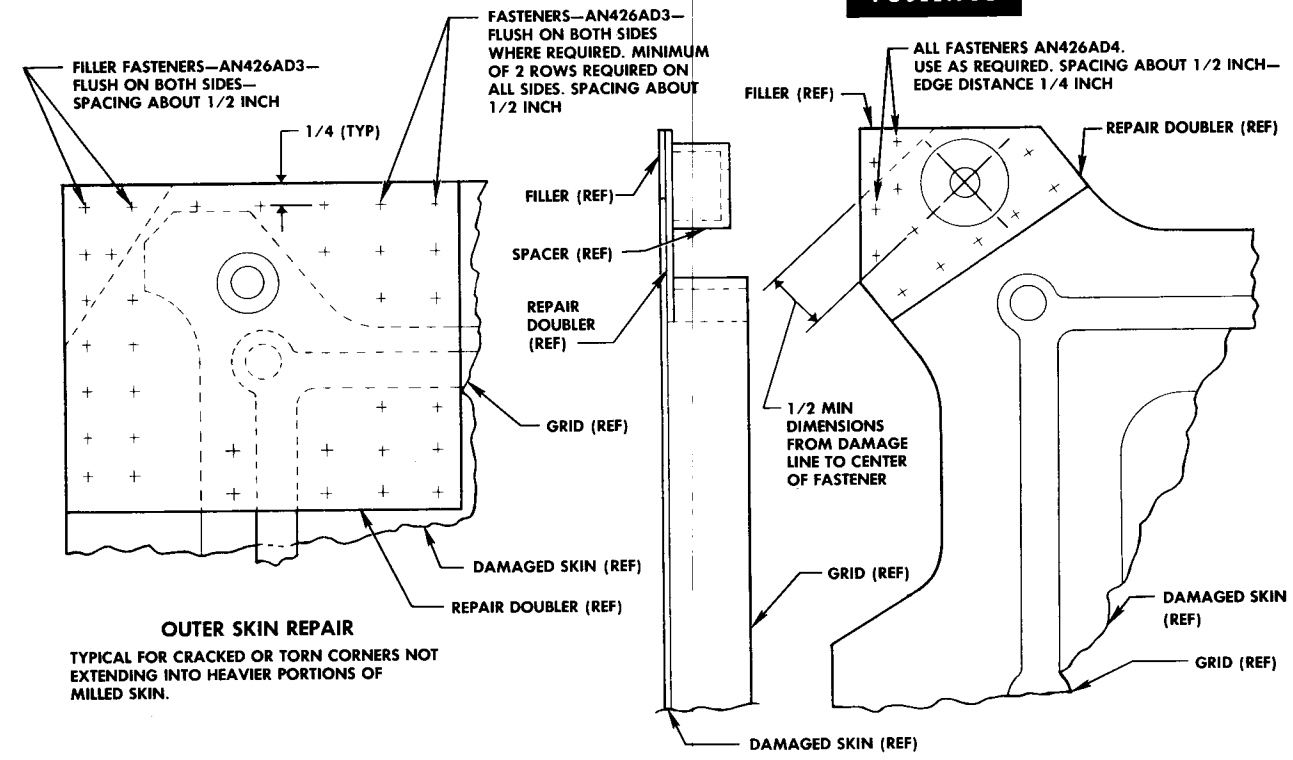
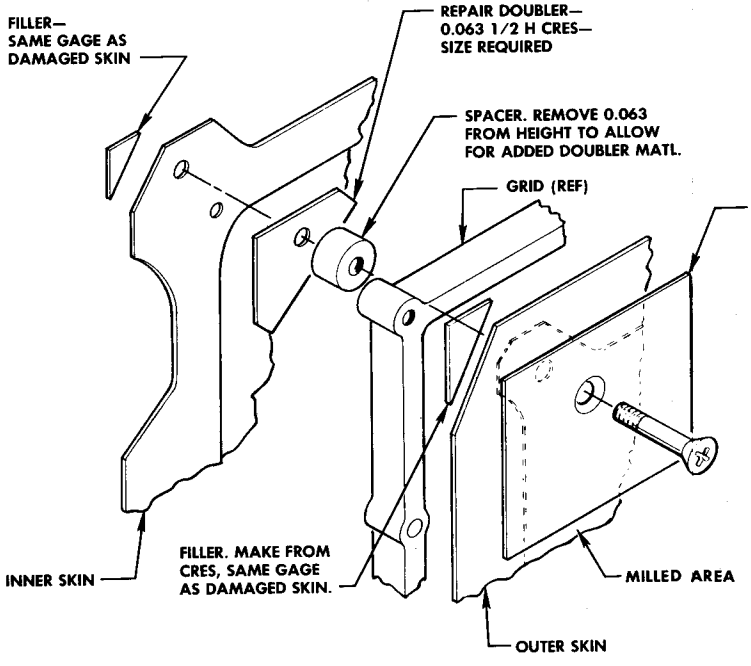


Figure 4-17A. Speed Brake—Honeycomb (Sheet 4 of 4)

PERMANENT—CATEGORY A

FUSELAGE





1. Trim off damage and smooth all edges.
2. Make repair doublers and fillers.
3. If repair is for internal skin, remove thickness of doubler from spacer.
4. Fasten repair parts in place, using size and type called out.
5. Fastener spacing is about 1/2 inch. Edge distance is 1/4 inch. Double-flush fasteners where required.

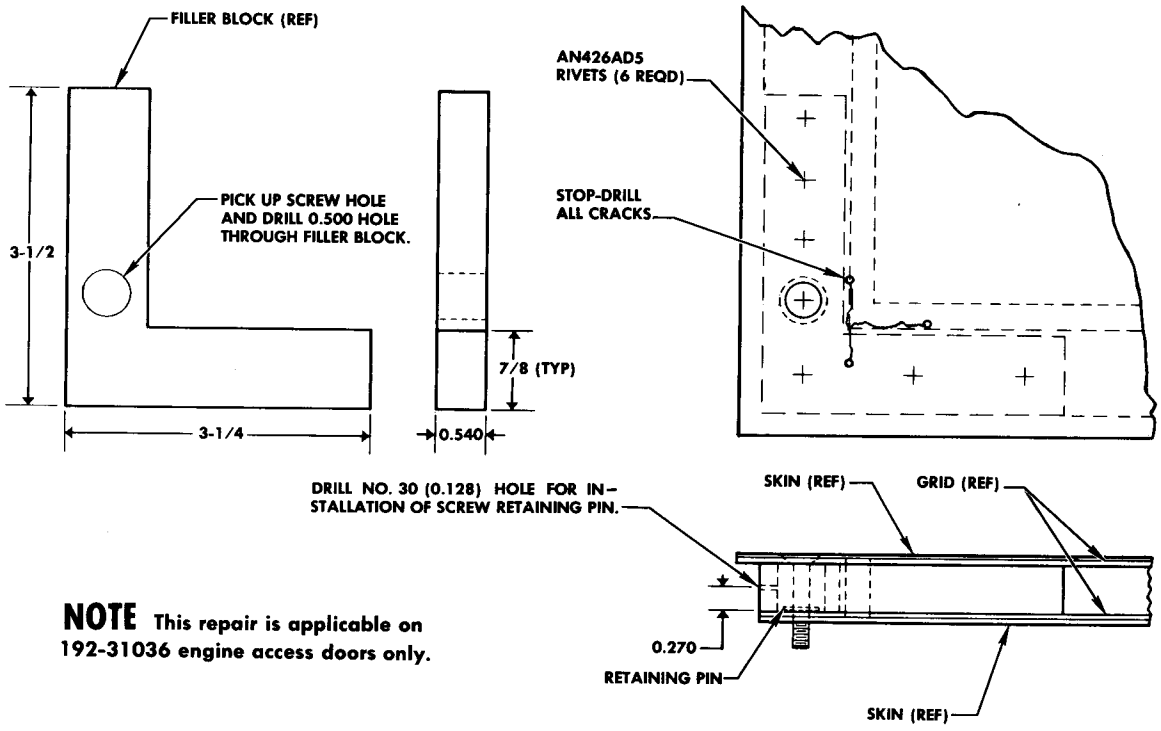
NOTE These repairs may be used individually or together, depending upon which skins are damaged.

THIS REPAIR IS APPLICABLE ON AIRPLANES WITHOUT ARRESTING HOOK.

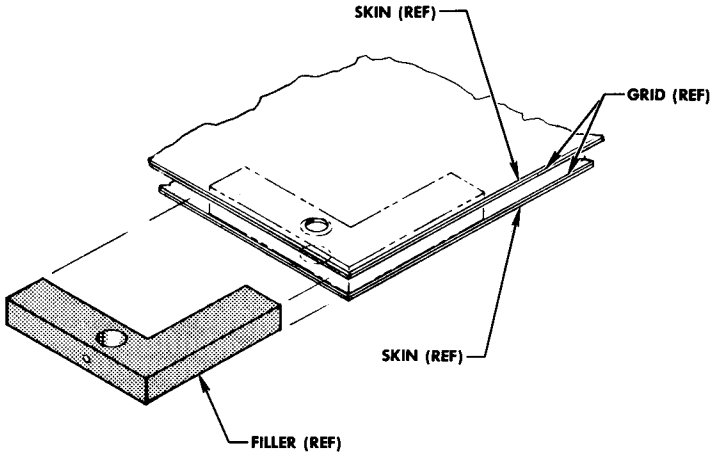
F-100D-3-31-118A

Figure 4-17B. Repair of Damaged Corners on Engine Access Doors (Sheet 1 of 2)

PERMANENT CATEGORY A FUSELAGE



NOTE This repair is applicable on 192-31036 engine access doors only.



1 Stop-drill all cracks in grid and skin with No. 40 (0.098-inch) drill.

2 Straighten all flanges.

3 Make filler block from 2024-T3 aluminum alloy, and anodize.

4 Double-flush existing skin rivets, and drill new rivet holes through skins, grid, and filler.

5 Install filler with six AN426AD5 rivets.

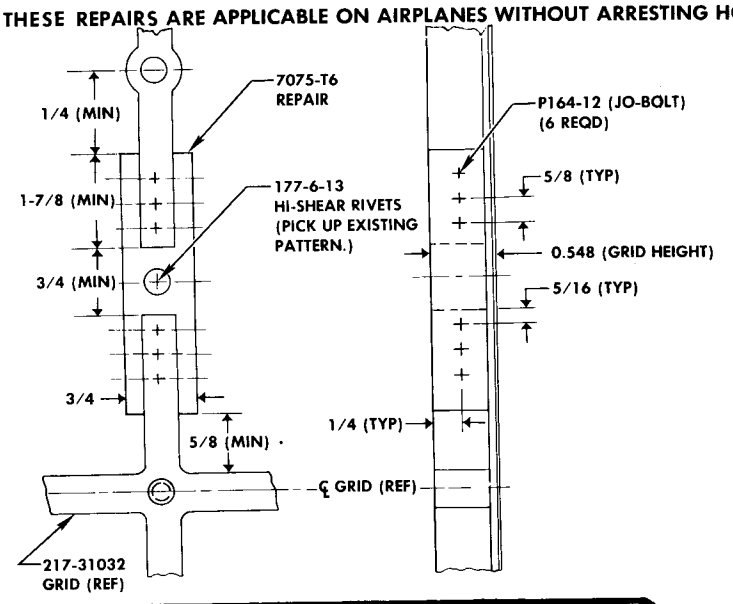
F-100D-3-31-129B

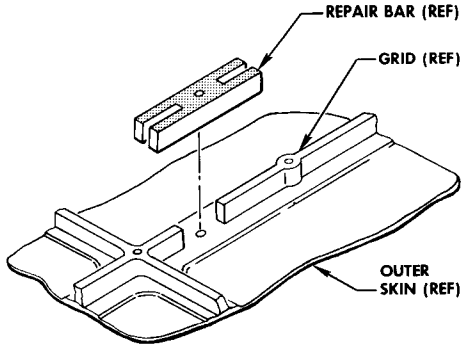
Figure 4-17B. Repair of Damaged Corners on Engine Access Doors (Sheet 2 of 2)

PERMANENT—CATEGORY A

FUSELAGE

THESE REPAIRS ARE APPLICABLE ON AIRPLANES WITHOUT ARRESTING HOOK.



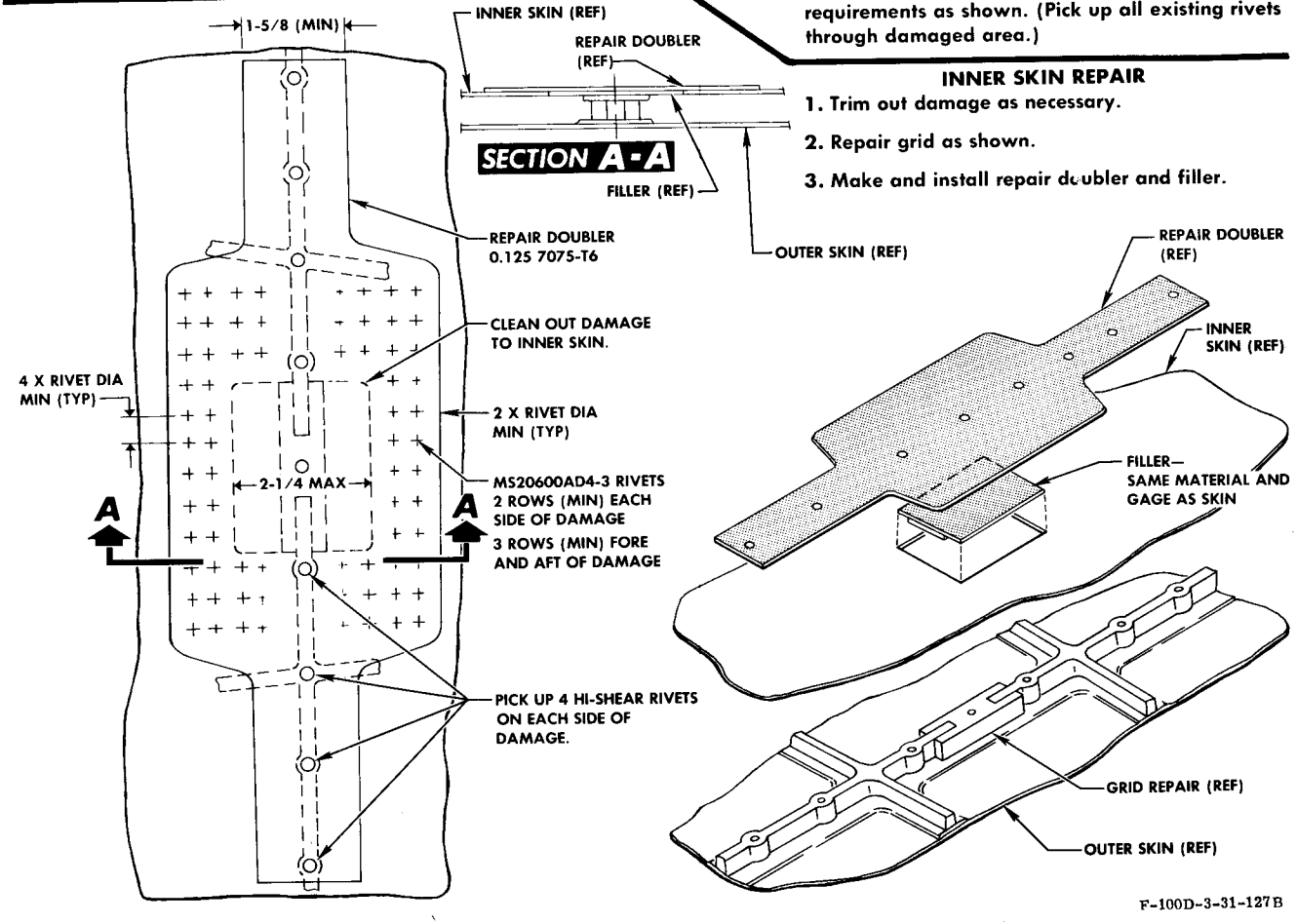


GRID REPAIR

1. Trim out damage as necessary.
2. Make and install repair bar as shown.

NOTE

The repair may extend from one grid intersection to another, provided there is enough material to attach the repair bar to meet the minimum requirements as shown. (Pick up all existing rivets through damaged area.)



INNER SKIN REPAIR

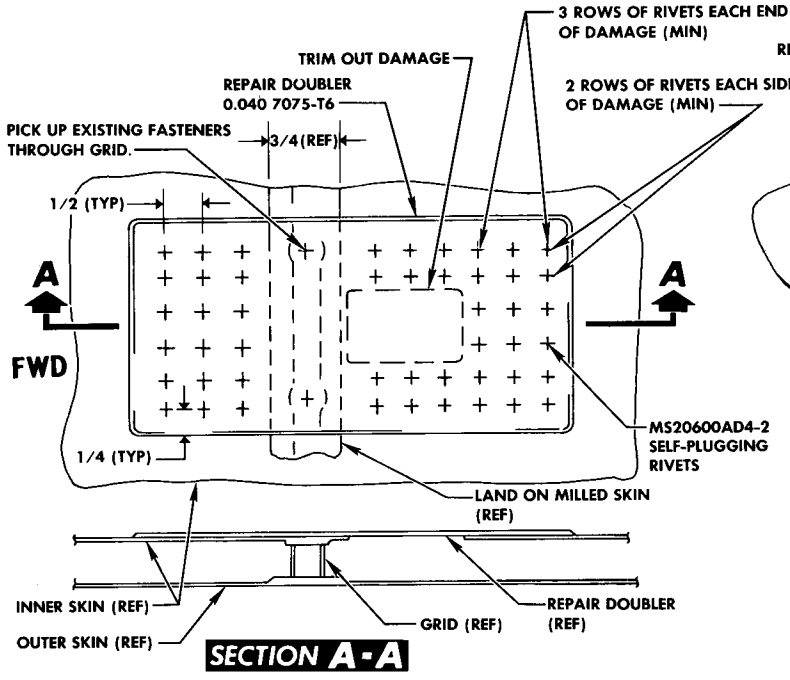
1. Trim out damage as necessary.
2. Repair grid as shown.
3. Make and install repair doubler and filler.

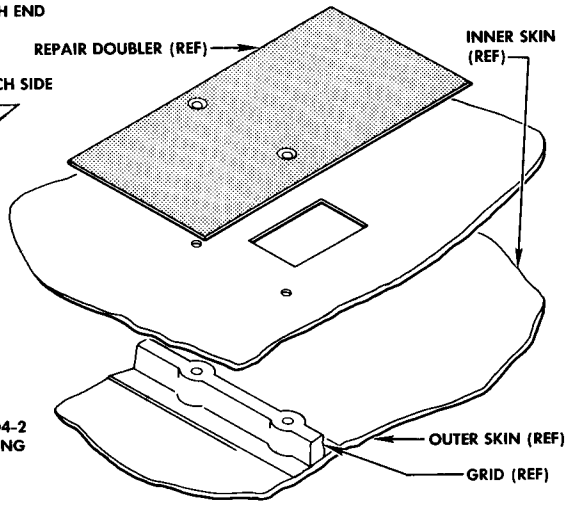
Figure 4-17B-1. Engine Access Doors (Sheet 1 of 3)

PERMANENT AND TEMPORARY— CATEGORY A

FUSELAGE

THIS REPAIR IS APPLICABLE ON AIRPLANES WITHOUT ARRESTING HOOK.



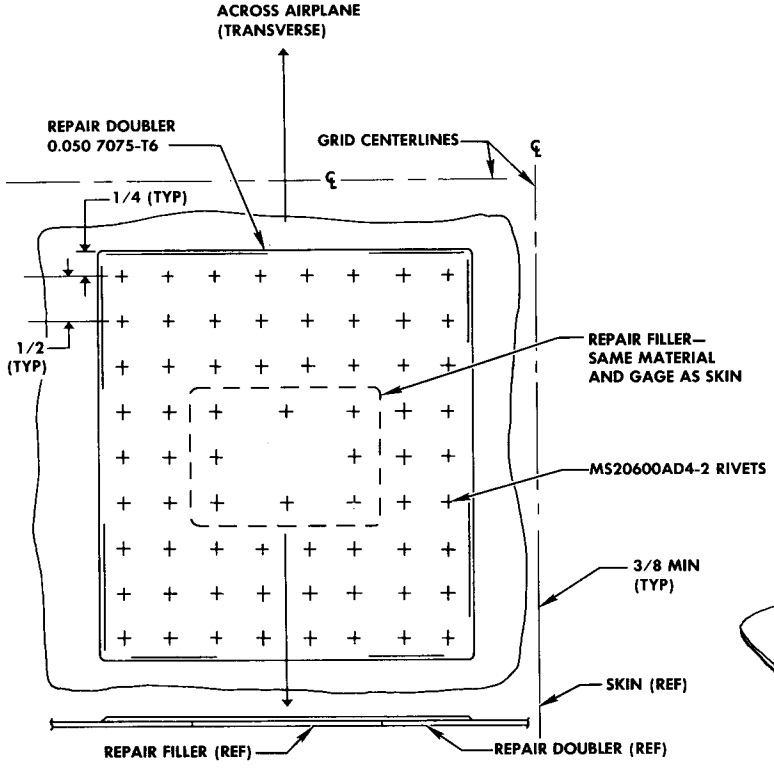


1. Trim out damage as necessary.
2. Make and install repair doubler (30-degree chamfer on edges).
3. Use a minimum of three rows of rivets at each end of repair across airplane (transverse). Use a minimum of two rows of rivets forward and aft of damage.

NOTE

Inner skin repairs may be classified in the permanent repair category. However, where outer skin damage occurs, the repair category must be considered to be in the temporary class.

The outer skin may be lifted and the repair doubler installed inside, provided skin damage is near the edge of the door, and the grid section will accommodate the doubler. Use filler in cutout.



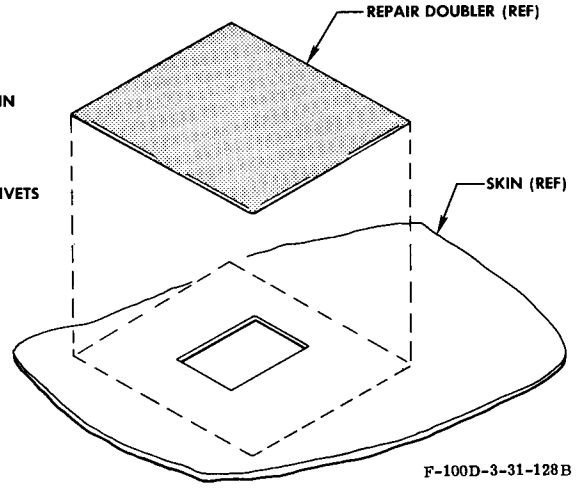
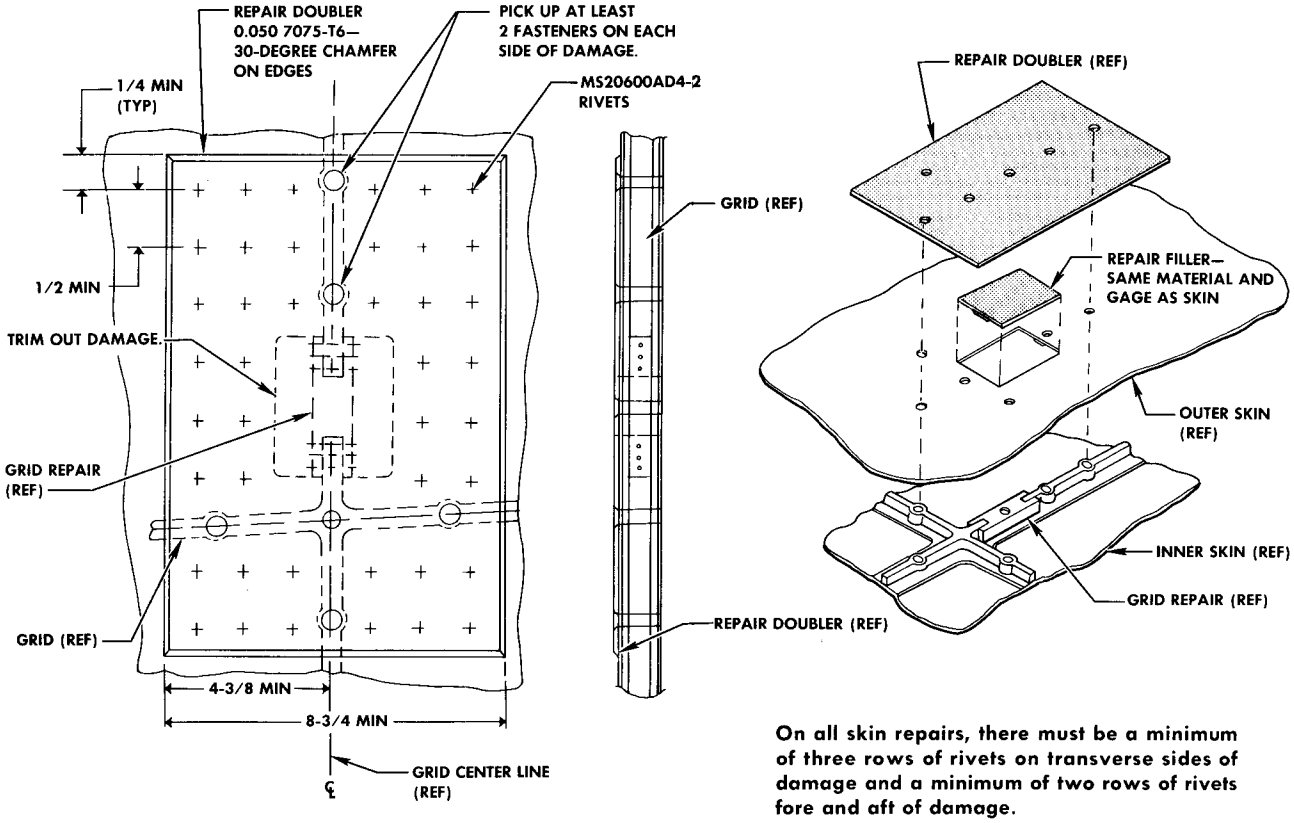


Figure 4-17B-1. Engine Access Doors (Sheet 2 of 3)

PERMANENT AND TEMPORARY—CATEGORY A

FUSELAGE

THESE REPAIRS ARE APPLICABLE ON AIRPLANES WITHOUT ARRESTING HOOK.



INNER OR OUTER SKIN REPAIR—GRID AREA

On all skin repairs, there must be a minimum of three rows of rivets on transverse sides of damage and a minimum of two rows of rivets fore and aft of damage.

1. Trim out damaged area.
2. Repair grid as shown.
3. Make and install filler and repair doublers.
4. Pick up existing fasteners through grid.

NOTE

Inner skin edge repairs may be classified in the permanent repair category. However, where outer skin edge damage occurs, the repair category must be considered to be in the temporary class.

EDGE REPAIR

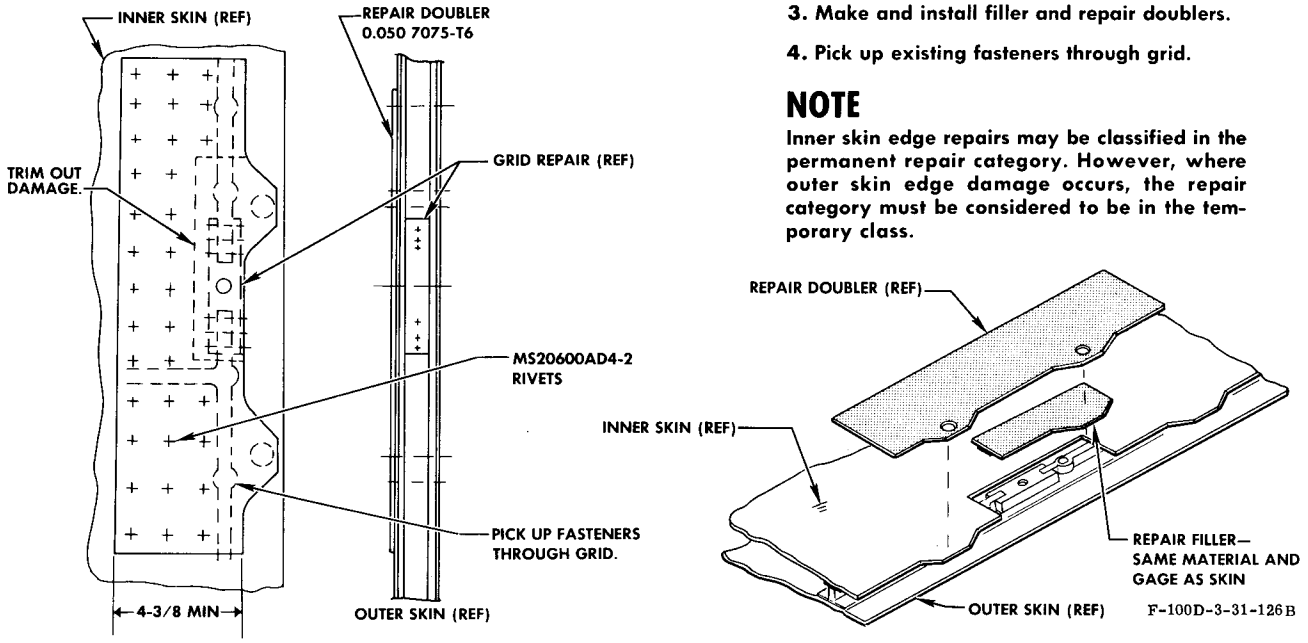


Figure 4-17B-1. Engine Access Doors (Sheet 3 of 3)

TAB REPAIR—7415-313013 ENGINE ACCESS DOOR

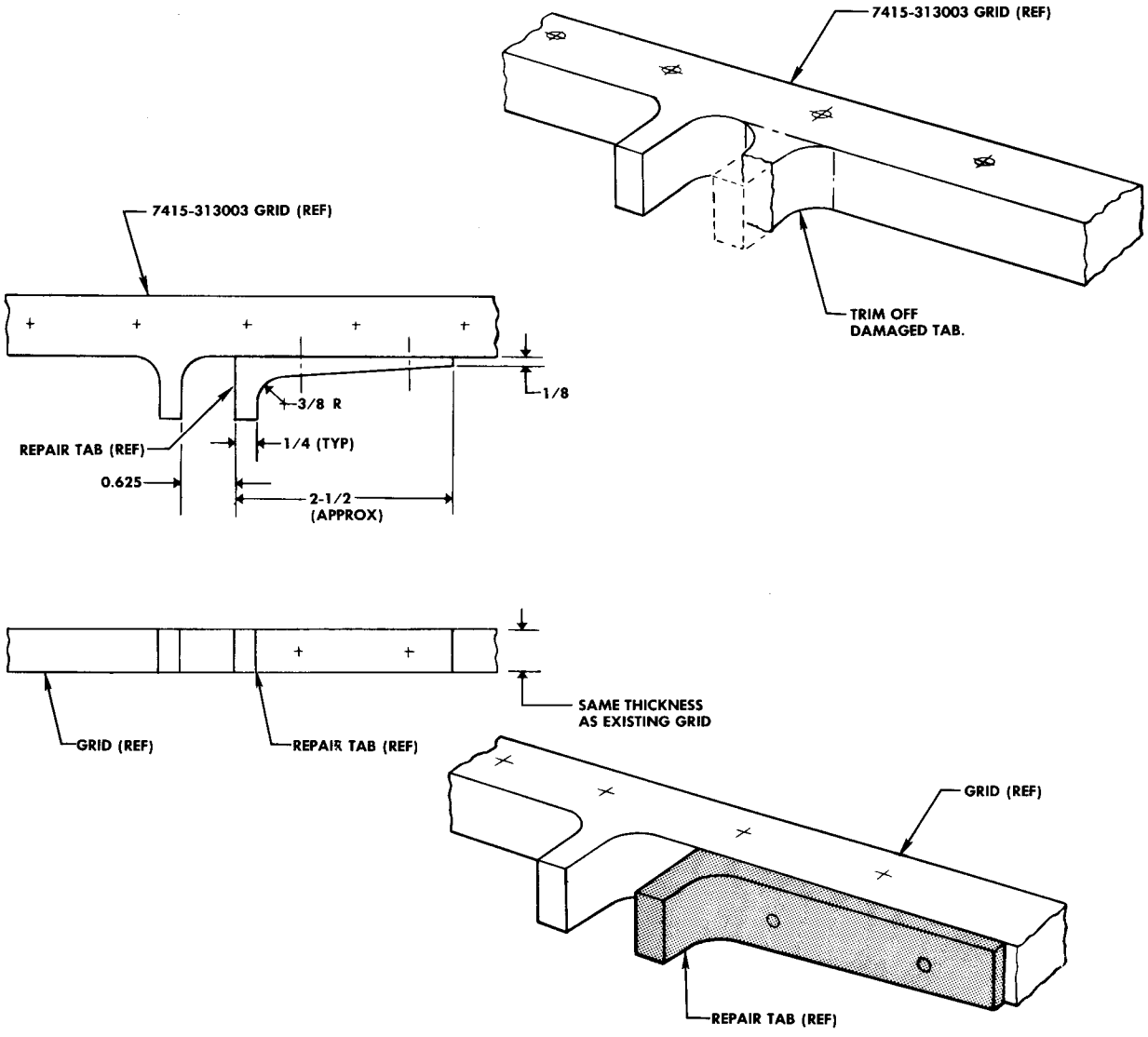
FUSELAGE

NOTE After damaged tab has been trimmed off, inspect trimmed surface for cracks. Refer to "Penetrant inspection method" in section I for inspection procedure.

Attach repair tab to grid with AN470DD6 rivets or with 3/16-inch Deutsch drive pins (6951-S-06-6).

If space is limited, the repair tab may be shorter than 2-1/2 inches and may be attached with one fastener.

Make repair tab from 7075-T6 extrusion or bar stock.

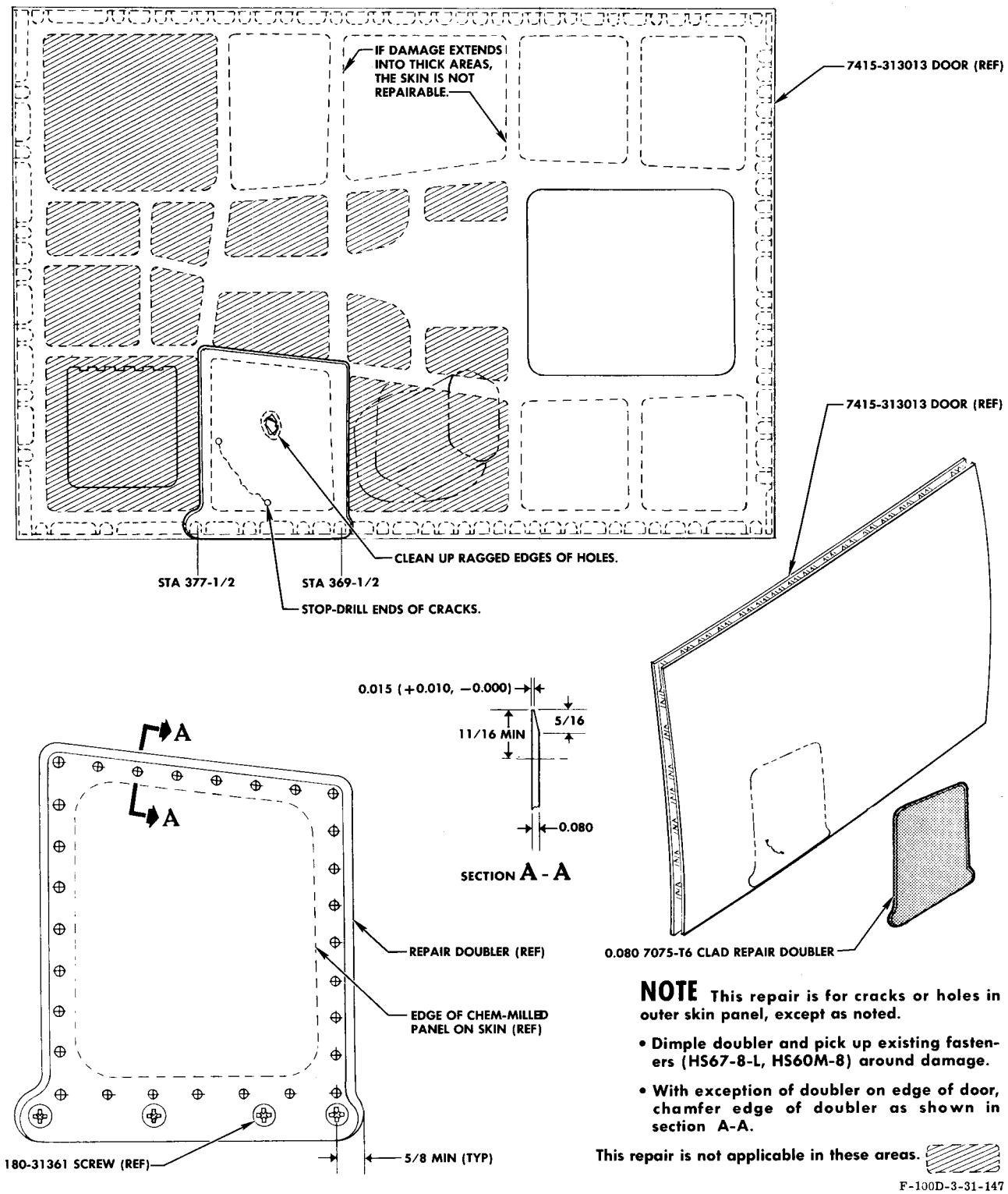


F-100D-3-31-143

Figure 4-17B-2. Tab Repair—Engine Access Door

TEMPORARY—CATEGORY A

FUSELAGE



NOTE This repair is for cracks or holes in outer skin panel, except as noted.

- Dimple doubler and pick up existing fasteners (HS67-8-L, HS60M-8) around damage.
- With exception of doubler on edge of door, chamfer edge of doubler as shown in section A-A.

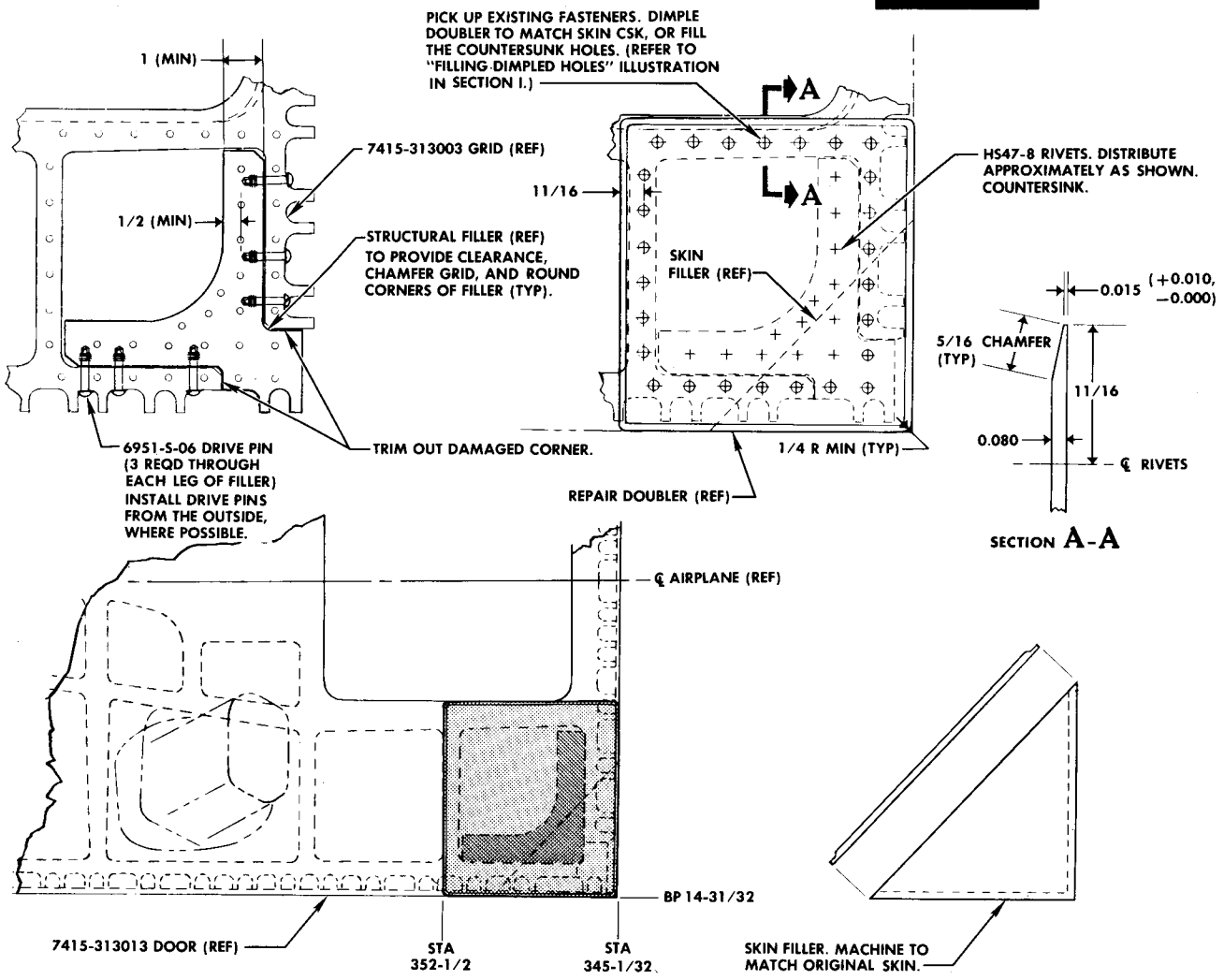
This repair is not applicable in these areas.

F-100D-3-31-147

Figure 4-17B-3. Engine Access Door — 7415-313013 (Sheet 1 of 2)

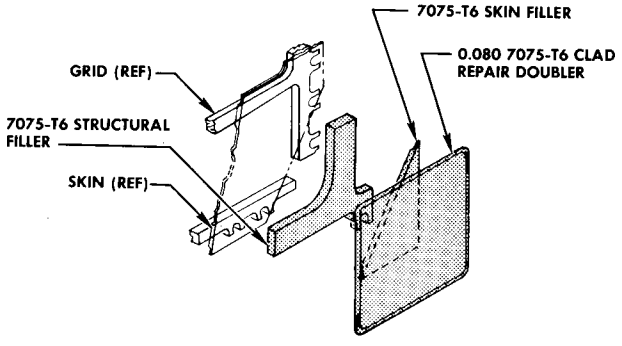
PERMANENT—CATEGORY A

FUSELAGE



NOTE This repair is applicable to the forward corners of the outer skin.

1. Remove enough skin rivets to gain access to damaged corner. Trim out damaged corner of grid. Make structural filler to replace the damaged corner as shown, and install filler, using Deutsch drive pins.
2. If necessary, add shims between structural filler and skins.
3. Trim off damaged skin. Replace trimmed-off section with filler, machined to fit.
4. Make and install repair doubler. Pick up existing fasteners through grid and add like fasteners through repair doubler and structural filler as shown.

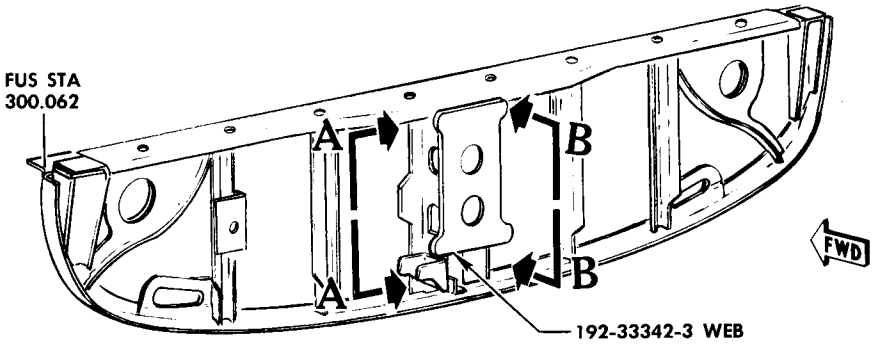


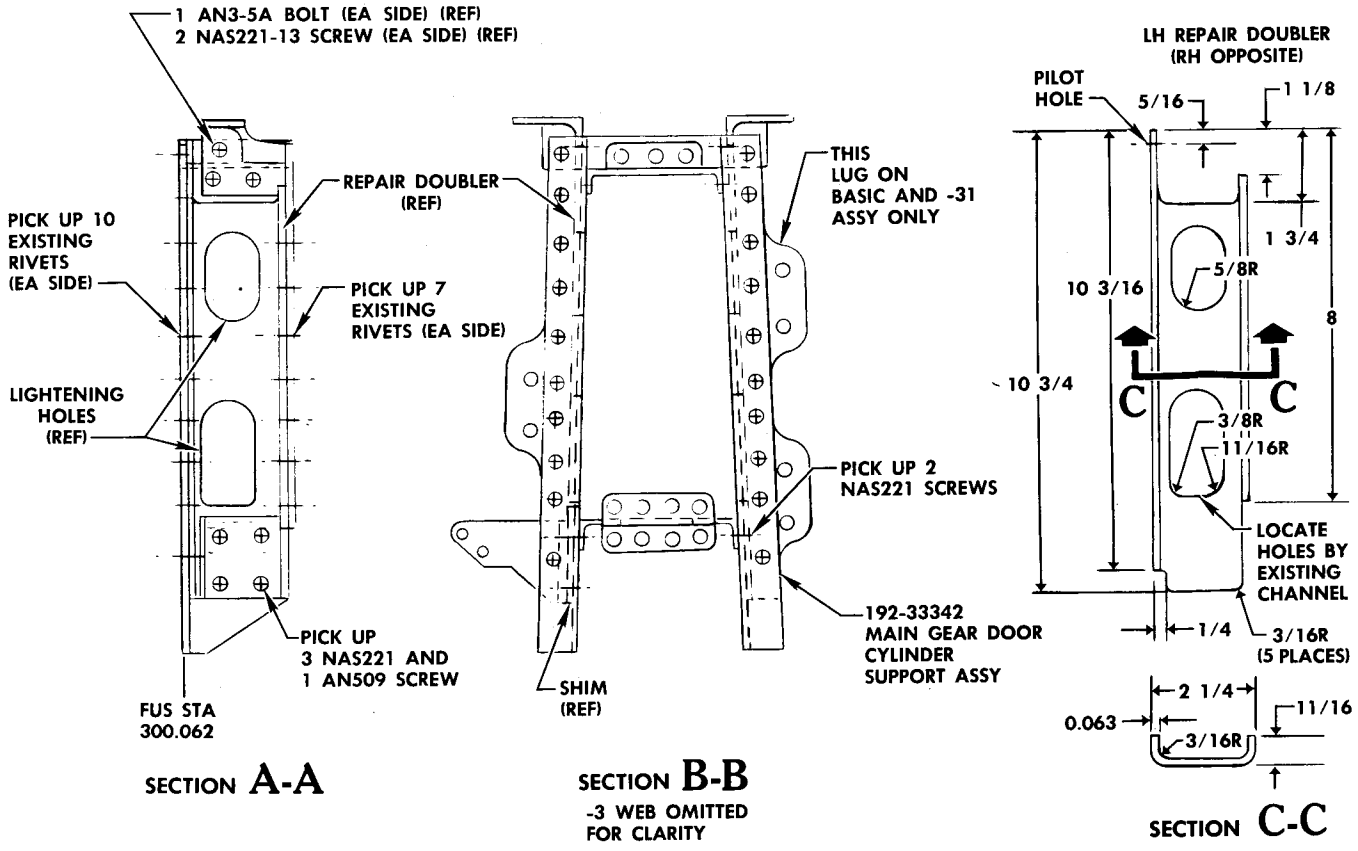
F-100D-3-31-146

Figure 4-17B-3. Engine Access Door—7415-313013 (Sheet 2 of 2)

PERMANENT—CATEGORY A

FUSELAGE





NOTE This repair is applicable to F-100D-1, -5, -10, -15, and -35 Airplanes.

1 Grind out cracks in lightening hole(s) until they are smooth and have no sharp corners. All corner radii of reworked lightening holes should not be less than 3/8 inch. Rework damaged area no closer than 5/16 inch to either outer flange or adjacent structure.

2 Make repair doubler from 0.063 1/4 hard AISI 301 stainless steel sheet as indicated.

3 Make substitute shim from 2024-T4 clad aluminum sheet, 0.063, 1-13/16 by 1-15/16 inches.

4 Before installing repair doubler, apply two coats of zinc-chromate primer (MIL-P-8585) over entire surface, allowing first coat to dry before applying the second coat. Dip all rivets and screws for attaching repair doubler into zinc-chromate primer and apply wet.

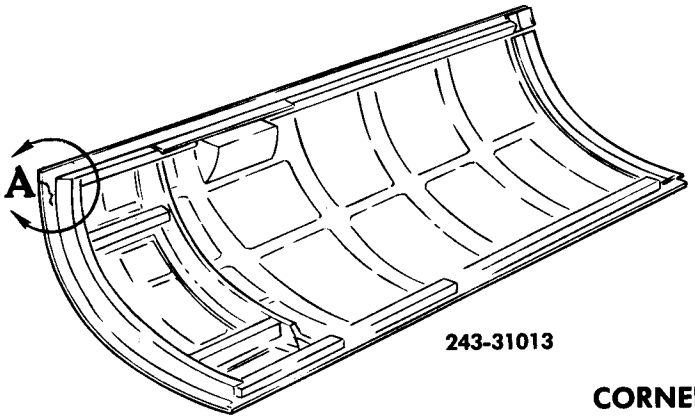
5 Install repair doubler(s) and replacement shim as indicated in Section A-A.

F-100D-3-31-154

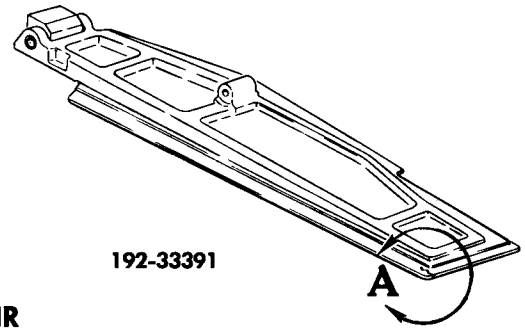
Figure 4-17B-4. Repair of Cracked Lightening Holes on Main Gear Door Cylinder Support Assembly

PERMANENT—CATEGORY A

FUSELAGE



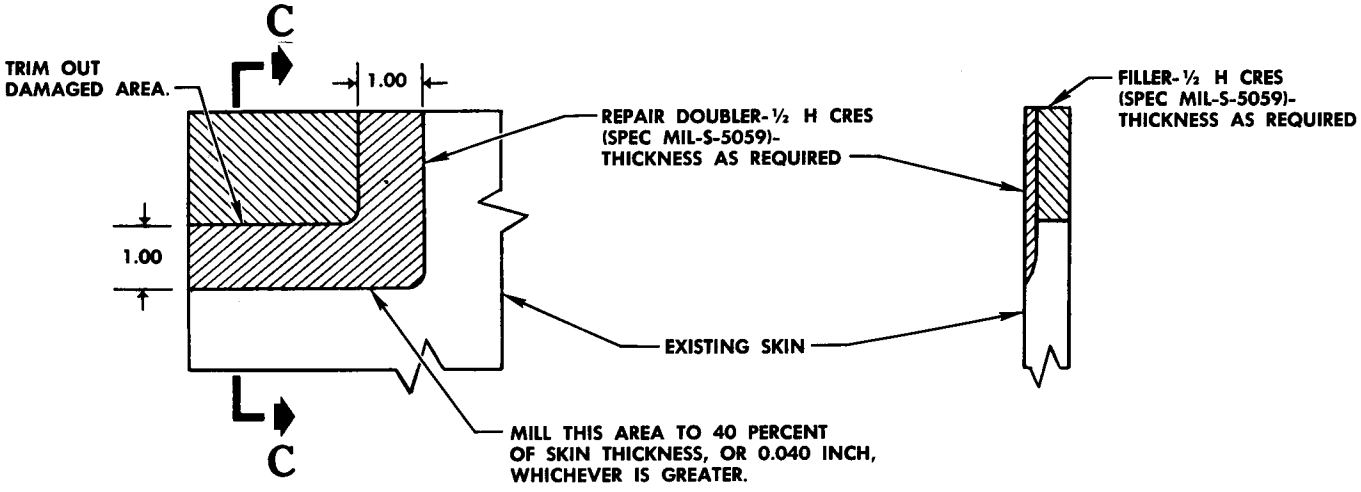
243-31013



192-33391

CORNER REPAIR

NOTE These repairs are for edges and corners of milled doors. Locations and dimensions illustrate the type of repair. Notched or stepped shaped areas may be repaired, using these procedures.



VIEW A

SECTION C-C

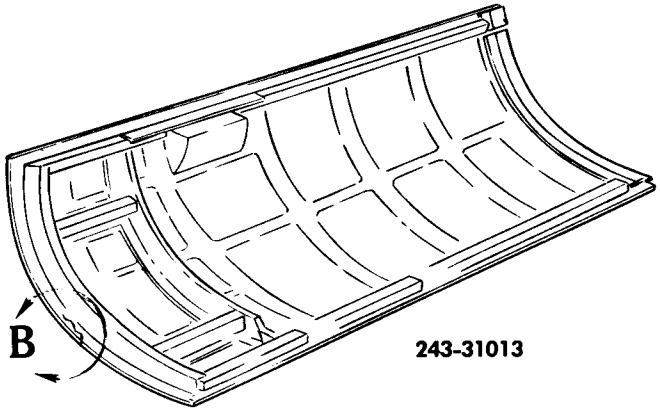
1. Mill out damaged area to eliminate cracks, breakouts, and deformed metal.
2. Mill to a depth of 40 percent of outside surface thickness, or 0.040 inch, whichever is greater, and of sufficient area to allow for rivet pattern of repair doubler.
3. Maintain ¼-inch corner radii and 0.030-inch fillet radii. Break all sharp corners.
4. Treat machined areas with Alodine/Iridite according to Specification MIL-C-5541.
5. Fabricate a doubler and filler from ½ hard CRES (Specification MIL-S-5059) to fit milled areas. Extend edge of doubler one inch beyond edge of filler.
6. Prime milled area and repair parts with zinc-chromate primer (Specification MIL-P-8585).
7. Attach repairt parts to door with MS20426AD4 rivets, flush both sides. 6950-S-06 Deutsch rivets may be used in thick sections. Refer to tables in Section X to determine number of rivets required.
8. Fill gaps between repair parts and door with aerodynamic filler.
9. Paint door according to existing requirements.

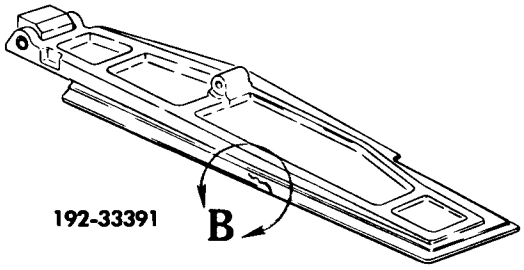
F-100D-3-31-158

Figure 4-17B-5. Gun Access Door and Landing Gear Strut Door Repair (Sheet 1 of 2)

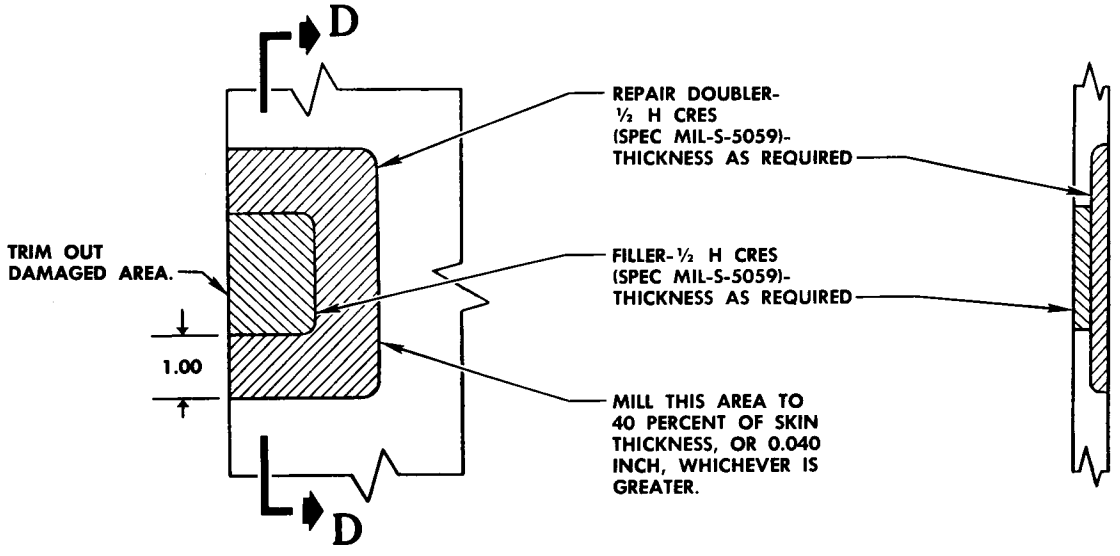
PERMANENT—CATEGORY A

FUSELAGE





EDGE REPAIR



VIEW B

SECTION D-D

1 Mill out damaged area to eliminate cracks, breakouts, and deformed metal.

2 Mill to a depth of 40 percent of outside surface thickness, or 0.040 inch, whichever is greater, and of sufficient area to allow for rivet pattern of repair doubler.

3 Maintain ¼-inch corner radii and 0.030-inch fillet radii. Break all sharp corners.

4 Treat machined areas with Alodine/Iridite according to Specification MIL-C-5541.

5 Fabricate a doubler and filler from ½ hard CRES (Specification MIL-S-5059) to fit milled areas. Extend edge of doubler one inch beyond edge of filler.

6 Prime milled area and repair parts with zinc-chromate primer (Specification MIL-P-8585).

7 Attach repair parts to door with MS20426AD4 rivets, flush both sides. 6950-S-06 Deutsch rivets may be used in thick sections. Refer to tables in Section X to determine number of rivets required.

8 Fill gaps between repair parts and door with aerodynamic filler.

9 Paint door according to existing requirements.

F-100D-3-31-159

Figure 4-17B-5. Gun Access Door and Landing Gear Strut Door Repair (Sheet 2 of 2)

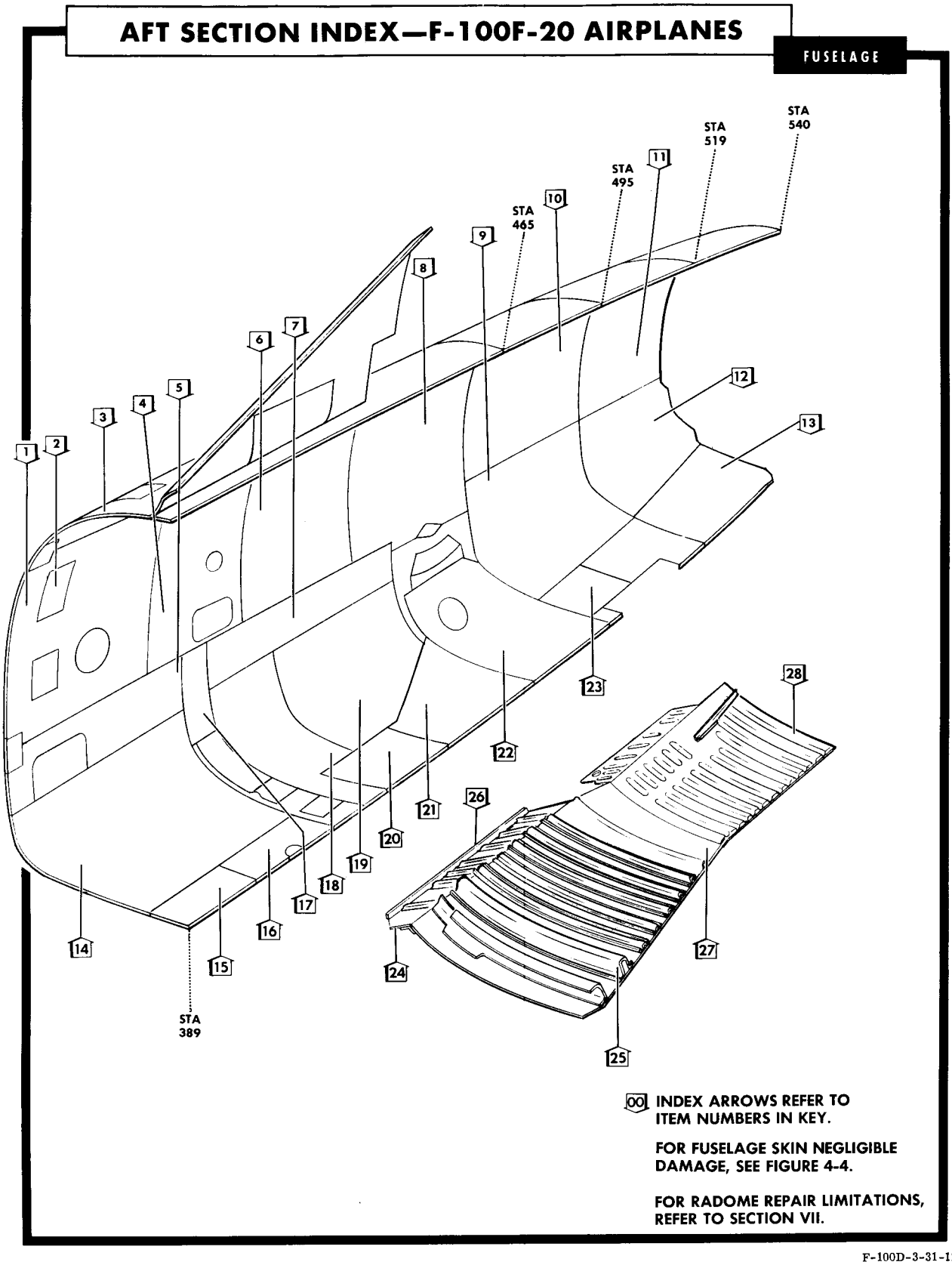


Figure 4-17C. Aft Section Repair Index—F-100F-20 Airplanes (Sheet 1 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-314175	Skin	0.040	Titanium Aly Sh	4-31, 4-32, 4-33
2	192-314149	Doubler	0.071	Titanium Aly Sh	4-31, 4-32, 4-33
3	192-31504	Skin—Top	0.032	1/4 Hard CRES	4-31, 4-32, 4-33
4	192-31504	Skin—Top	0.040	Titanium Aly Sh	4-31, 4-32, 4-33
5	235-31460-33	Skin—Side	0.063	7075-T6	4-31, 4-32, 4-33
6	192-31509-3	Skin—Side	0.080	7075-T6	4-31, 4-32, 4-33
7	255-314097	Skin—Side	0.032	Titanium Aly Sh	4-31, 4-32, 4-33
8	192-31509-5	Skin—Side	0.032	Titanium Aly Sh	4-31, 4-32, 4-33
9	255-315006	Skin—Lower	0.020	Titanium	4-31, 4-32, 4-33
10	255-315009	Skin—Side	0.020	Titanium	4-31, 4-32, 4-33
11	223-31538-5	Aspirator Skin	0.020	Titanium	4-31, 4-32, 4-33
12	223-31538-15	Aspirator Skin	0.032	Titanium	4-31, 4-32, 4-33
13	223-31538-11	Aspirator Skin	0.020	Titanium	4-28B, 4-31, 4-32, 4-33
14	235-31460-5	Skin—Lower	0.063	7075-T6	4-31, 4-32, 4-33
15	235-31528	Skin—Lower	0.080	7075-T6	
16	235-31460	Skin—Lower	0.063	7075-T6	4-31, 4-32, 4-33
17	255-315006	Skin—Side	0.040	7075-T6	
18	235-31460	Skin—Lower			
19	255-314077	Skin—Side	0.032	Titanium Aly Sh	4-31, 4-32, 4-33
20	255-314097	Door—Lower	0.080	7075-T6	
21	255-314098	Door—Lower	0.125	7075-T6	
22	192-310307	Skin—Bottom	0.080	2024-T4	4-31, 4-32, 4-33
23	223-31503	Skin—Bottom	0.020	Titanium	4-31, 4-32, 4-33
24	192-31414	Web	0.040	Titanium Aly Sh	
25	192-31596	Seal	0.040	Titanium	
26	192-31557	Angle—Fuel Shelf	0.040	Titanium Aly Sh	
27	192-31408	Shroud	0.016	Titanium	
28	255-314190	Shroud	0.020	Titanium	

*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.

Key to Figure 4-17C (Sheet 1 of 2)

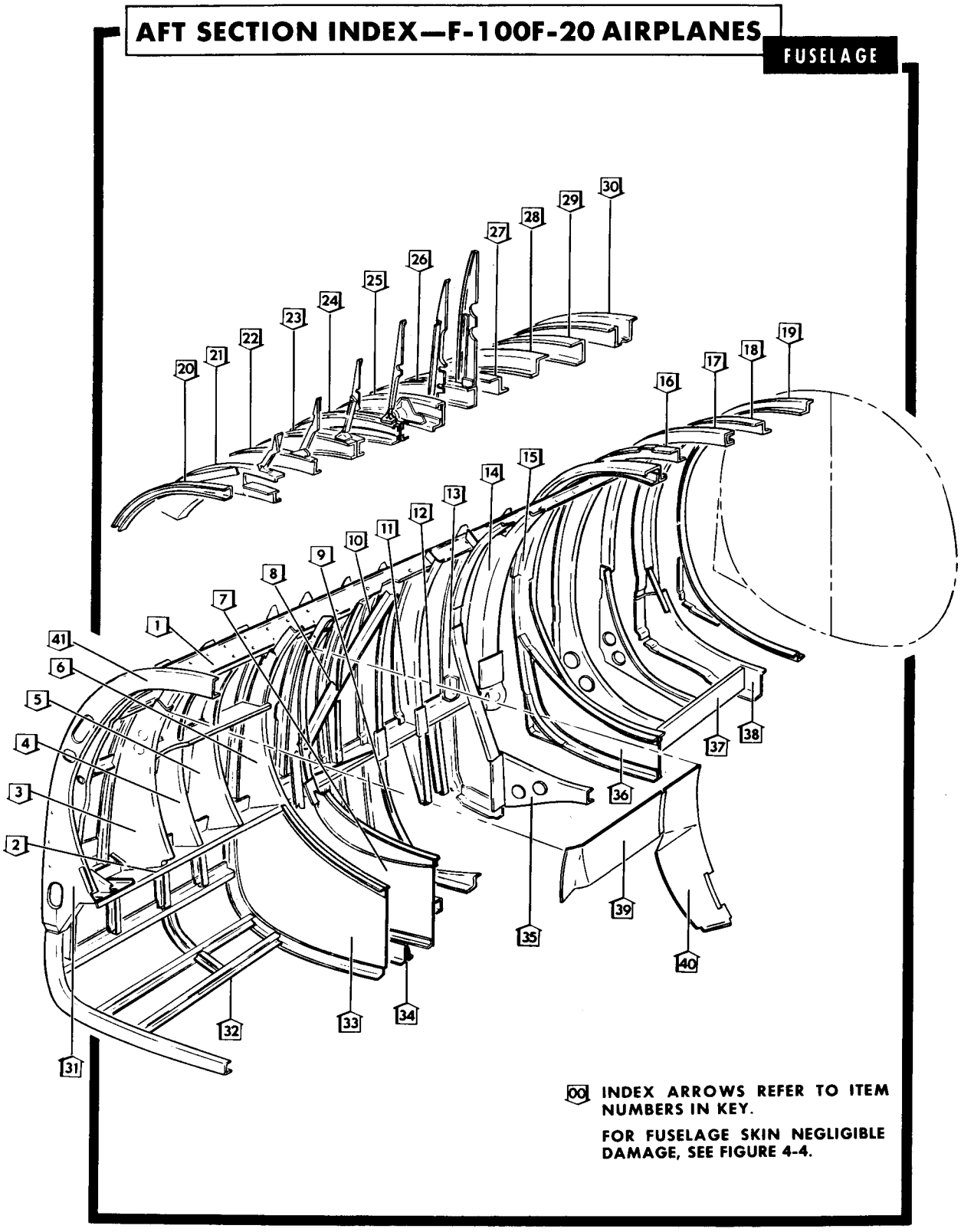


Figure 4-17C. Aft Section Repair Index—F-100F-20 Airplanes (Sheet 2 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION	STA	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31420	Longeron Assy—Upper		0.071	Titanium Aly Sh	
2	192-31421	Longeron Assy—Lower		0.080	Titanium Aly Sh	4-19, 4-20
3	217-31400	Frame Assy	401.5	0.040	Titanium Aly Sh	4-21, 4-22, 4-23
4	217-31411	Frame Assy	410.5	0.032	Titanium Aly Sh	4-21, 4-22, 4-23
5	217-31425	Frame Assy	420	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
6	235-31460	Bulkhead Assy	429.5	0.040	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
7	255-314041	Bulkhead Assy	440	0.032	301 CRES Sh	4-21, 4-22, 4-23
8	217-31447	Frame Assy	448	0.090	Titanium Aly Sh	4-21, 4-22, 4-23
9	255-31441	Frame Assy	454	0.050	7056-T6	
10	192-31520	Angle	440-456	0.063	Titanium Aly Sh	4-21, 4-22, 4-23
11	255-314092	Frame Assy	466	0.025	Titanium Aly Sh	
12	255-31443	Frame Assy	469	0.050	7075-T6	
13	255-314075	Frame Assy	475	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
14	192-31485	Frame Assy	485	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
15	192-310227	Angle—Upper	495	0.063	Titanium Aly Sh	
16	192-315171	Frame—Upper	503.2	0.025	Titanium	4-21, 4-22, 4-23
17	192-315172	Frame—Upper	511.5	0.025	Titanium	4-21, 4-22, 4-23
18	223-31522	Frame Assy	519	0.090	17-7 PH CRES	4-21, 4-22, 4-23, 4-46B
19	192-31530	Frame Assy	532	0.032	Titanium	4-21, 4-22, 4-23
20	192-31392	Frame Assy	395	0.064	7075-T6	4-21, 4-22
21	217-31399	Frame Assy—Upper	401	0.040	Titanium Aly Sh	4-21, 4-22, 4-23
22	217-31409	Frame Assy—Upper	410	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
23	192-31424	Frame Assy—Upper	420	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
24	235-31464	Frame Assy—Upper	429	0.025	17-7 PH CRES	4-21, 4-22, 4-23
25	192-31439	Frame Assy—Upper	440	0.063	½ Hard CRES	4-21, 4-22, 4-23
26	217-31447	Frame Assy—Upper	448	0.090	Titanium Aly Sh	4-21, 4-22, 4-23
27	192-31451	Frame Assy—Upper	456	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
28	192-31465	Frame Assy—Upper	466	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
29	192-31474	Frame Assy—Upper	475	0.030	Titanium Aly Sh	4-21, 4-22, 4-23
30	192-31484	Frame Assy—Upper	485	0.056	Titanium Aly Sh	4-21, 4-22, 4-23
31	255-314151	Frame Assy	389	0.063	7075-T6	4-21, 4-23
32	235-31460	Stringer		6E521	7075-T6	4-27
33	235-31462	Bulkhead Assy	429.5	0.040	Titanium Aly Sh	4-30
34	192-31434	Frame Assy	434.75	0.040	Titanium Aly Sh	
35	255-314076	Frame Assy	475	0.045	Titanium Aly Sh	4-21, 4-23
36	192-315195	Frame Assy	495	0.040	Titanium Aly Sh	4-21, 4-22, 4-23
37	192-315170	Beam Assy		0.050	Titanium Aly Sh	
38	192-31522	Frame Assy	519	0.032	Titanium Aly Sh	4-21, 4-23
39	192-31588	Shroud—Cover		0.016	Titanium	
40	192-31397	Shroud—Cont Bay		0.016	Titanium	
41	192-31390	Bulkhead	389	0.063	7075-T6	4-18C

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 4-17C (Sheet 2 of 2)

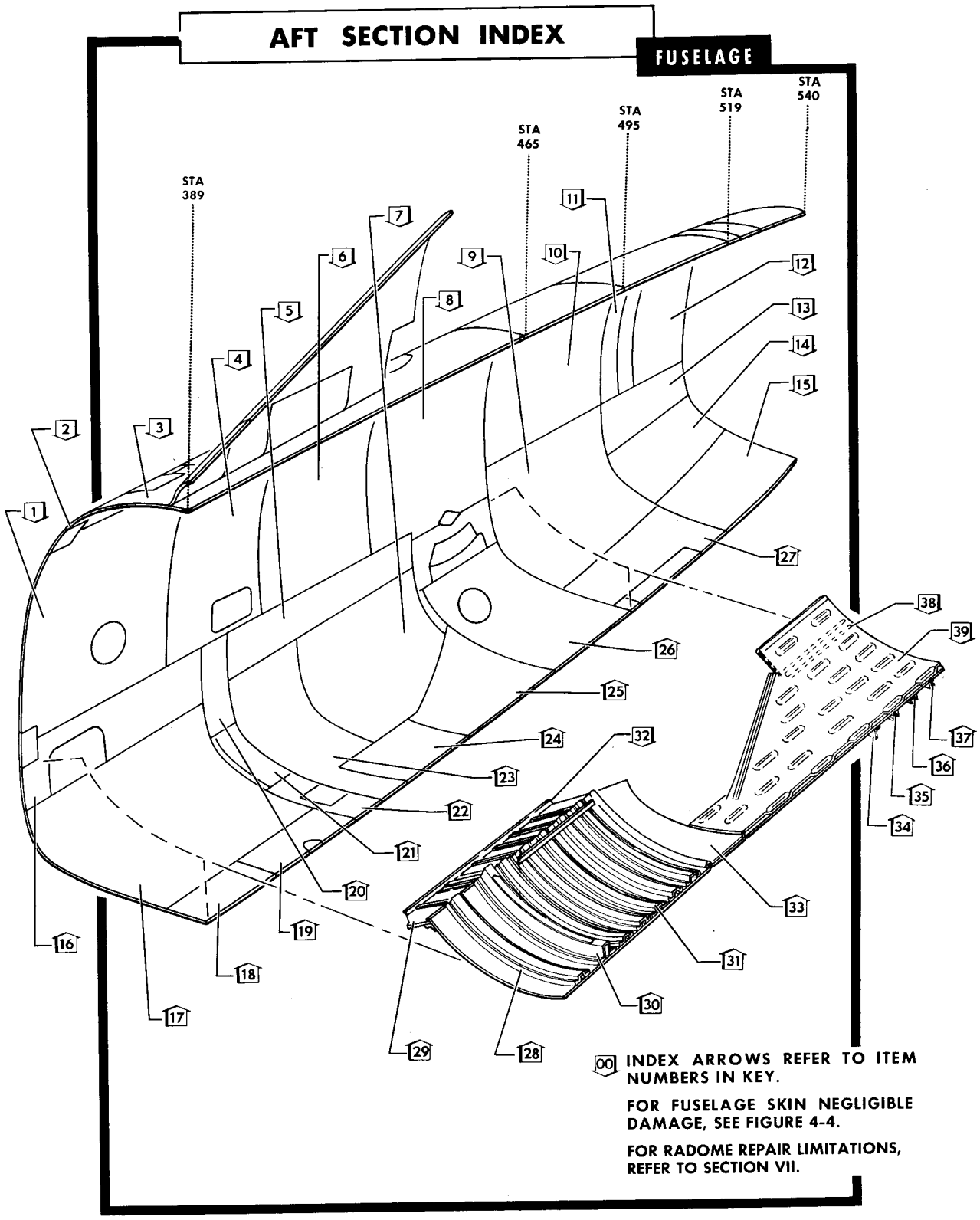
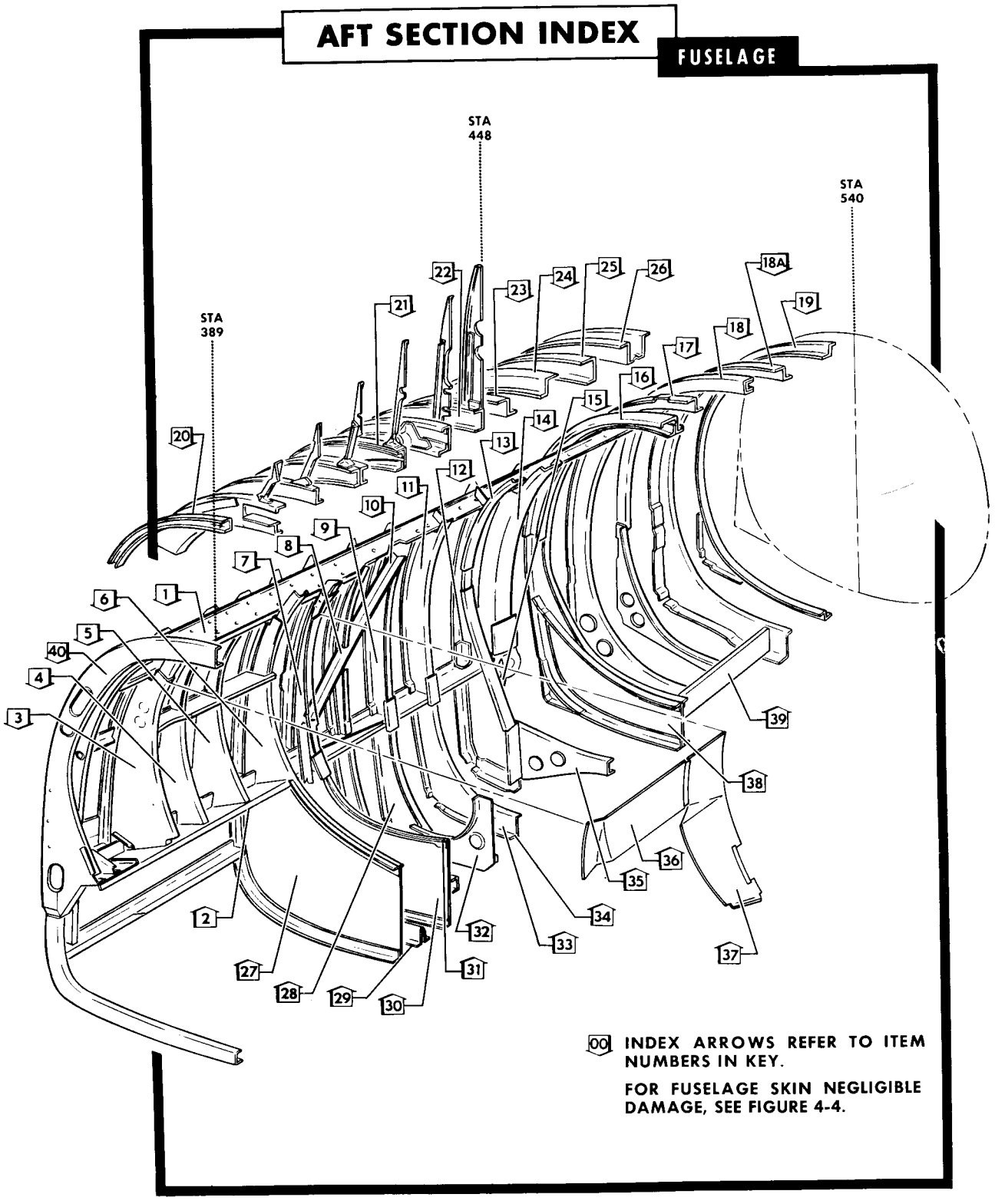


Figure 4-18. Aft Section Repair Index—F-100D Airplanes and F-100F-1 Through F-100F-15 Airplanes (Sheet 1 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-314175	Skin		0.040	Titanium Aly Sh	4-31, 4-32, 4-33
2	192-314149	Doubler		0.071	Titanium Aly Sh	4-31, 4-32, 4-33
3	192-31504	Skin—Top		0.032	¼H CRES	4-31, 4-32, 4-33
4	192-314175	Skin—Top		0.040	Titanium Aly Sh	4-31, 4-32, 4-33
5	223-31506	Skin—Side		0.025	Titanium	4-31, 4-32, 4-33
6	192-31509	Skin—Side		0.032	Titanium Aly Sh	4-31, 4-32, 4-33
7	192-31506	Skin—Lower		0.032	Titanium	4-31, 4-32, 4-33
8	192-31509	Skin—Side		0.032	Titanium Aly Sh	4-31, 4-32, 4-33
9	192-315173-3	Skin—Lower		0.020	Titanium	4-31, 4-32, 4-33
10	192-315173-5	Skin—Side		0.020	Titanium	4-31, 4-32, 4-33
11	192-315184	Skin—Side		0.020	Titanium	4-31, 4-32, 4-33
12	192-315184-5	Skin—Aspirator		0.020	Titanium	4-28, 4-28A, 4-31, 4-32, 4-33
13	192-315184-15	Skin—Aspirator		0.020	Titanium	4-28, 4-28A, 4-31, 4-32, 4-33
14	192-315184-11	Skin—Aspirator		0.020	Titanium	4-28, 4-28A, 4-31, 4-32, 4-33
15	192-315184-63	Skin—Aspirator		0.020	Titanium	4-28B
16	192-310287	Skin—Side		0.032	7075-T6 Clad	4-31, 4-32, 4-33
17	223-31391	Skin—Lower		0.040	7075-T6 Clad	4-31, 4-32, 4-33
18	223-31391	Skin—Bottom		0.032	7075-T6 Clad	4-31, 4-32, 4-33
19	223-31026	Skin—Bottom		0.032	7075-T6 Clad	4-31, 4-32, 4-33
20	223-31506-15	Skin—Side		0.040	7075-T6 Clad	4-31, 4-32, 4-33
21	223-315261	Skin—Lower		0.040	7075-T6 Clad	4-31, 4-32, 4-33
22	223-31506-83	Skin—Bottom		0.040	7075-T6 Clad	4-31, 4-32, 4-33
23	223-31506-3	Skin—Bottom		0.025	Titanium Aly Sh	4-31, 4-32, 4-33
24	223-31036	Skin—Bottom		0.080	7075-T6 Clad	4-31, 4-32, 4-33
25	223-31037	Skin—Bottom		0.125	7075-T6 Clad	4-31, 4-32, 4-33
26	192-310307	Skin—Bottom		0.080	2024-T4 Clad	4-31, 4-32, 4-33
27	192-310280	Drag Chute Doors		0.040	2024-T4 Clad	
28	192-31415	Stiffener		1S220	Titanium Aly Sh	
29	192-31414	Web		0.040	Titanium Aly Sh	
30	192-31596	Seal		0.040	Titanium	
31	192-31408	Retainer		0.025	Titanium	
32	192-31557	Angle—Fuel Shelf		0.040	Titanium Aly Sh	
33	192-31398	Shroud		0.016	Titanium	
34	192-310273	Angle	Sta 475	0.040	Titanium Aly Sh	
35	192-310264	Channel	Sta 480.531	0.032	Titanium	
36	192-310265	Channel	Sta 485.968	0.032	Titanium	
37	192-310266	Channel	Sta 491.750	0.032	Titanium	
38	192-310268	Stiffener	Sta 475 to 495	0.040	Titanium	
39	223-31460	Shroud Assy		0.046	6061-O	4-34A

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 4-18 (Sheet 1 of 2)



F-100D-3-31-81D

Figure 4-18. Aft Section Repair Index—F-100D Airplanes and F-100F-1 Through F-100F-15 Airplanes (Sheet 2 of 2)

(Pages 4-45 through 4-48 deleted)

ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-31420	Longeron Assy—Upper		0.071	Titanium Aly Sh	
2	192-31421	Longeron Assy—Lower		0.080	Titanium Aly Sh	4-19, 4-20
3	217-314000	Frame Assy	Sta 392	0.040	Titanium Aly Sh	
4	217-31411	Frame Assy	Sta 401	0.032	Titanium Aly Sh	4-21, 4-22, 4-23
5	217-31425	Frame Assy	Sta 410.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
6	192-31429	Bulkhead Assy	Sta 429.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
7	192-31440	Frame Assy	Sta 440	0.063	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
8	192-31448	Frame Assy	Sta 448.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
9	192-31457	Frame Assy	Sta 456.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
10	192-31520	Angle	Sta 440 to 456.5	0.063	Titanium Aly Sh	
11	192-31466	Frame Assy	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
12	192-31475	Frame Assy	Sta 475	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
13	192-315132	Angle	Sta 475	0.090	Titanium Aly Sh	4-21, 4-22, 4-23
14	192-31485	Frame Assy	Sta 485	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
15	192-31512	Seal—Cont Well	Sta 485	0.020	Titanium	
16	192-310227	Angle	Sta 495	0.063	Titanium Aly Sh	
17	192-315171	Frame Assy	Sta 503.25	0.025	Titanium	4-21, 4-22, 4-23
18	192-315172	Frame Assy	Sta 511.5	0.025	Titanium	4-21, 4-22, 4-23
18A	223-31522	Frame Assy	Sta 519.875	0.090	301 CRES, ½ Hard	4-26B
19	192-31530	Frame Assy	Sta 532	0.032	Titanium	
20	192-31409	Upper Frame	Sta 410.5	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
21	192-31787	Upper Frame Assy	Sta 420	0.025	Titanium	4-21, 4-22, 4-23
22	192-31439	Upper Frame Assy	Sta 440	0.032	Titanium Aly Sh	4-21, 4-22, 4-23
23	192-31451	Upper Frame Assy	Sta 456.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
24	192-31465	Upper Frame Assy	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
25	192-31474	Upper Frame Assy	Sta 475	0.030	Titanium Aly Sh	4-21, 4-22, 4-23
26	192-31484	Upper Frame Assy	Sta 485	0.056	Titanium Aly Sh	4-21, 4-22, 4-23
27	223-31430	Bulkhead Assy	Sta 429.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
28	192-314174	Frame Assy	Sta 456.5	0.032	Titanium	4-21, 4-22, 4-23
29	192-31434	Frame Assy	Sta 434.75	0.040	Titanium Aly Sh	
30	192-31449	Bulkhead Assy	Sta 448.5	0.025	Titanium	4-21, 4-22, 4-23, 4-30
31	192-31455	Support	Sta 448.5	0.050	Titanium Aly Sh	
32	192-31458	Frame—Lower	Sta 456.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
33	102-31467	Frame—Lower	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
34	192-31467	Frame—Lower	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
35	192-31476	Frame Assy		0.045	Titanium Aly Sh	
36	192-31588	Cover—Shroud	Sta 401 to 429.5	0.016	Titanium	
37	192-31397	Shroud—Cont Bay		0.016	Titanium	
38	192-315195	Frame Assy	Sta 495	0.040	Titanium Alloy	4-22
39	192-315170	Beam Assy—Tail Skid		0.050	Titanium Alloy	
40	192-31390	Bulkhead	Sta 389	0.063	7075-T6	4-18C

***Drawing numbers are for reference only and are not to be used for ordering parts.**

†For additional repairs, refer to Section X.

Key to Figure 4-18 (Sheet 2 of 2)

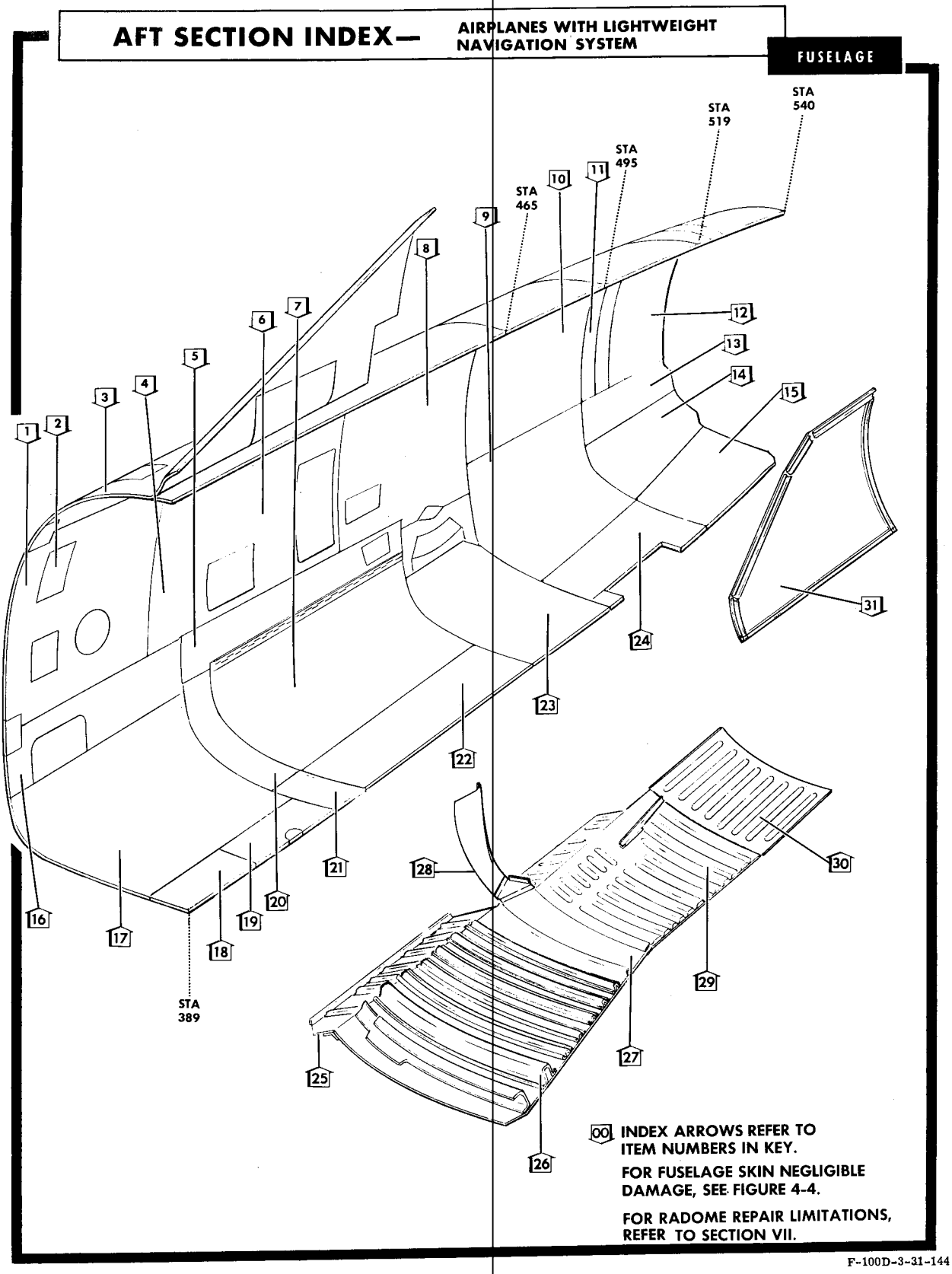
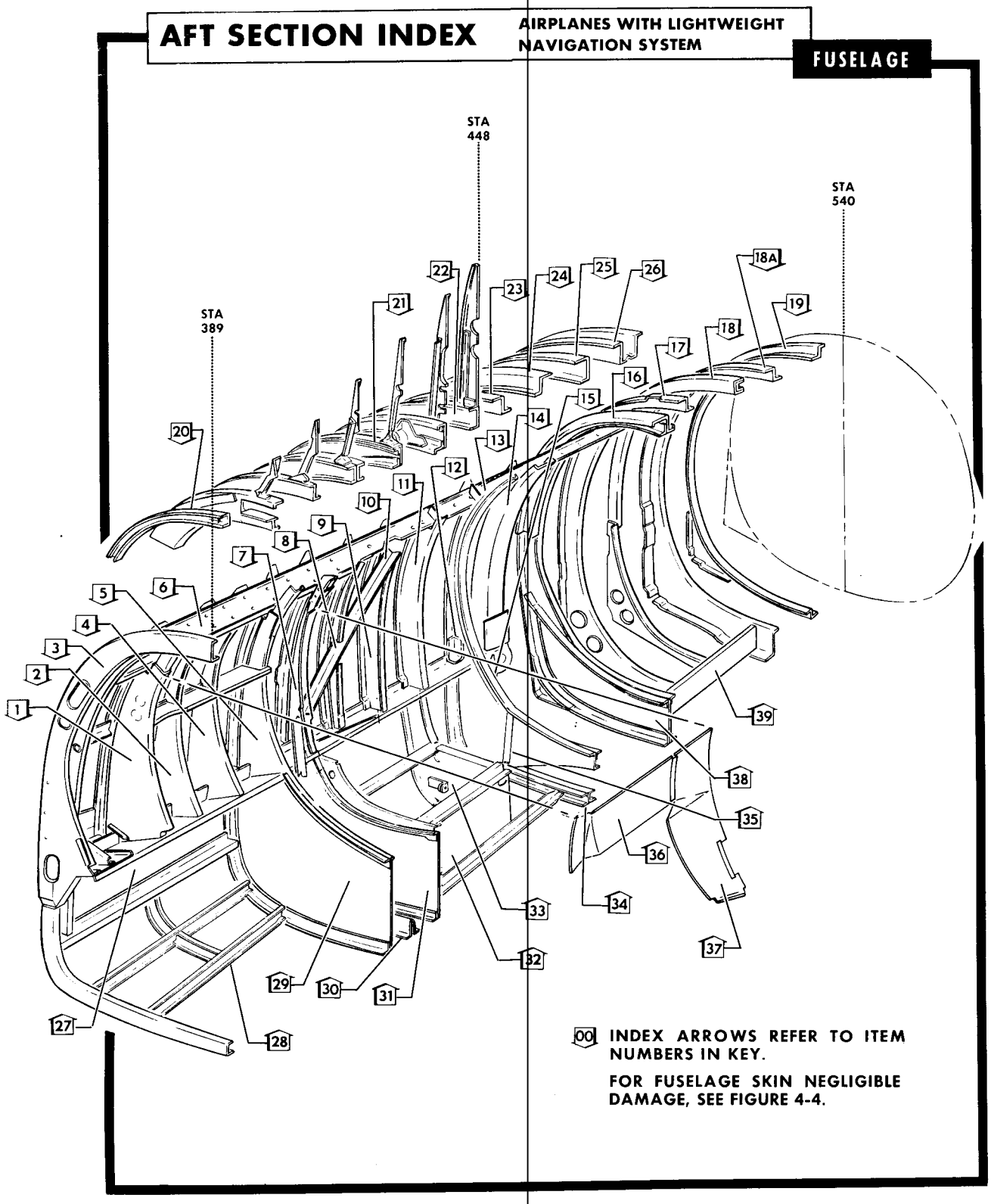


Figure 4-18A. Aft Section Repair Index—Airplanes With Lightweight Navigation System (Sheet 1 of 2)

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	192-314175	Skin	0.040	Titanium Aly Sh	4-31, 4-32, 4-33
2	192-314149	Doubler	0.071	Titanium Aly Sh	4-31, 4-32, 4-33
3	192-31504	Skin—Top	0.032	¼ Hard CRES	4-31, 4-32, 4-33
4	192-314175	Skin—Top	0.040	Titanium Aly Sh	4-31, 4-32, 4-33
5	235-314143	Skin—Side	0.025	½ Hard 301 CRES	
6	192-31509-3	Skin—Side	0.032	Titanium Aly Sh	4-31, 4-32, 4-33
7	235-314116	Door	0.040	2024-T42 Clad Sh	
8	192-31509-9	Skin—Side	0.032	Titanium Aly Sh	4-31, 4-32, 4-33
9	192-315173-3	Skin—Lower	0.020	Titanium Aly Sh	4-31, 4-32, 4-33
10	192-315173-5	Skin—Side	0.020	Titanium Aly Sh	4-31, 4-32, 4-33
11	192-315184	Skin—Side	0.020	Titanium Aly Sh	4-31, 4-32, 4-33
12	192-315184-5	Skin—Aspirator	0.020	Titanium Aly Sh	4-28, 4-28A, 4-31, 4-32, 4-33
13	192-315184-15	Skin—Aspirator	0.020	Titanium	4-28, 4-28A, 4-31, 4-32, 4-33
14	192-315184-11	Skin—Aspirator	0.020	Titanium	4-28, 4-28A, 4-31, 4-32, 4-33
15	192-315184-63	Skin—Aspirator	0.020	Titanium	4-28B
16	192-310287	Skin—Side	0.032	7075-T6 Clad	4-31, 4-32, 4-33
17	223-31391	Skin—Lower	0.040	7075-T6 Clad	4-31, 4-32, 4-33
18	223-31391	Skin—Bottom	0.032	7075-T6 Clad	4-31, 4-32, 4-33
19	223-31026	Skin—Bottom	0.032	7075-T6 Clad	4-31, 4-32, 4-33
20	235-314148	Skin—Lower	0.025	½ Hard 301 CRES	
21	235-314149	Skin—Bottom	0.025	½ Hard 301 CRES	
22	235-314115	Door	0.160	7075-T6 Bare Sh	
23	192-310307	Door	0.081	2024-T4 Clad	4-31, 4-32, 4-33
24	192-310280	Skin	0.040	2024-T4 Clad	
25	192-31414	Web	0.040	Titanium Aly Sh	
26	192-31596	Seal	0.040	Titanium Aly Sh	
27	255-314072	Shroud	0.016	Titanium Aly Sh	
28	192-31397	Shroud	0.016	Titanium Aly Sh	
29	255-314190	Shroud	0.025	Titanium Sh	
30	235-314184	Shroud	0.025 to 0.032	Titanium Sh	
31	255-314-85	Shroud	0.025 0.063	Titanium Sh Metalized Silicone Laminate	

***Drawing numbers are for reference only and are not to be used for ordering parts.**
†For additional repairs, refer to Section X.

Key to Figure 4-18A (Sheet 1 of 2)



F-100D-3-31-145A

Figure 4-18A. Aft Section Repair Index—Airplanes With Lightweight Navigation System (Sheet 2 of 2)

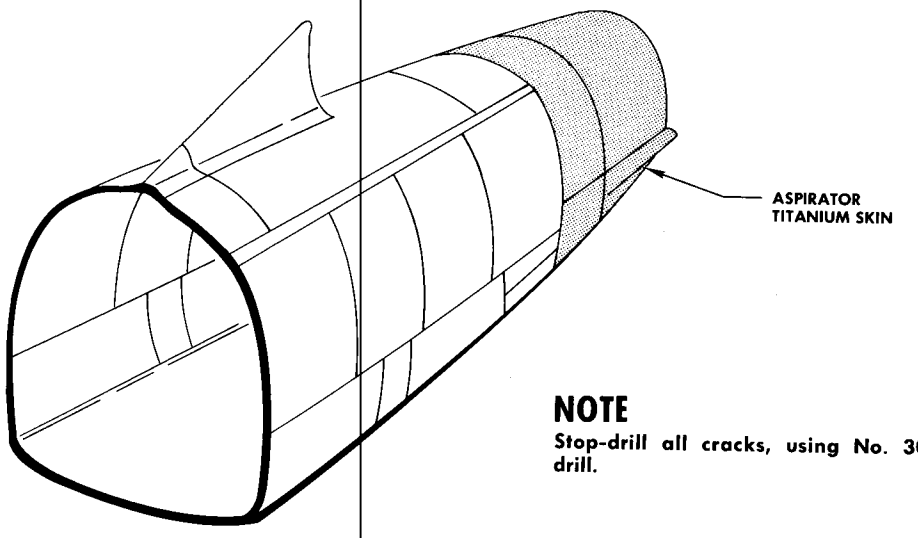
ITEM NO.	DRAWING NO.*	DESCRIPTION		GAGE	MATERIAL	REPAIR FIGURE NO.†
1	217-31400	Frame Assy	Sta 392	0.040	Titanium Aly Sh	
2	217-31411	Frame Assy	Sta 401	0.032	Titanium Aly Sh	4-21, 4-22, 4-23
3	192-31390	Bulkhead		0.050	Titanium Alloy	
4	217-31425	Frame Assy	Sta 410.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
5	192-31429	Bulkhead Assy	Sta 429.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
6	192-31420	Longeron Assy—Upper		0.071	Titanium Aly Sh	
7	192-31440	Frame Assy	Sta 440	0.063	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
8	192-31448	Frame Assy	Sta 448.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
9	192-31457	Frame Assy	Sta 456.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
10	192-31520	Angle	Sta 440 to 456.5	0.063	Titanium Aly Sh	
11	192-31466	Frame Assy	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
12	192-31475	Frame Assy	Sta 475	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
13	192-315132	Angle	Sta 475	0.090	Titanium Aly Sh	4-21, 4-22, 4-23
14	192-31485	Frame Assy	Sta 485	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
15	192-31512	Seal—Cont Well	Sta 485	0.020	Titanium	
16	192-310227	Angle	Sta 495	0.063	Titanium Aly Sh	
17	192-315171	Frame Assy	Sta 503.25	0.025	Titanium	4-21, 4-22, 4-23
18	192-315172	Frame Assy	Sta 511.5	0.025	Titanium	4-21, 4-22, 4-23
18A	223-31522	Frame Assy	Sta 519.875	0.090	301 CRES, ½ Hard	4-26B
19	192-31530	Frame Assy	Sta 532	0.032	Titanium	
20	192-31409	Upper Frame	Sta 410.5	0.050	Titanium Aly Sh	4-21, 4-22, 4-23
21	192-31787	Upper Frame Assy	Sta 420	0.025	Titanium	4-21, 4-22, 4-23
22	192-31439	Upper Frame Assy	Sta 440	0.032	Titanium Aly Sh	4-21, 4-22, 4-23
23	192-31451	Upper Frame Assy	Sta 456.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
24	192-31465	Upper Frame Assy	Sta 466.25	0.025	Titanium Aly Sh	4-21, 4-22, 4-23
25	192-31474	Upper Frame Assy	Sta 475	0.080	Titanium Aly Sh	4-21, 4-22, 4-23
26	192-31484	Upper Frame Assy	Sta 485	0.056	Titanium Aly Sh	4-21, 4-22, 4-23
27	192-31421	Longeron Assy—Lower		0.080	Titanium Aly Sh	4-19, 4-20
28	235-31460	Stringer	Sta 389	0.063	7075-T6	4-21, 4-23
29	223-31430	Bulkhead Assy	Sta 429.5	0.025	Titanium Aly Sh	4-21, 4-22, 4-23, 4-30
30	192-31434	Frame Assy	Sta 434.75	0.040	Titanium Aly Sh	
31	255-314041	Frame		0.032	CRES Sh	
32	235-314103	Web		0.040	2024-T4 Clad Sh	
33	255-314079	Frame		0.040	2024-T4 Clad Sh	
34	235-314104	Beam Cap	Sta 475	0.063	CRES Sh	
35	255-314088	Web		0.040	2024-T4 Clad Sh	
36	192-31588	Cover—Shroud	Sta 401 to 429.5	0.016	Titanium	
37	192-31397	Shroud—Cont Bay		0.016	Titanium	
38	192-315195	Frame Assy	Sta 495	0.040	Titanium Alloy	4-22
39	192-315170	Beam Assy—Tail Skid		0.050	Titanium Alloy	

*Drawing numbers are for reference only and are not to be used for ordering parts.
†For additional repairs, refer to Section X.

Key to Figure 4-18A (Sheet 2 of 2)

ASPIRATOR SKIN—NEGLIGIBLE DAMAGE

FUSELAGE



NOTE

Stop-drill all cracks, using No. 30 drill.

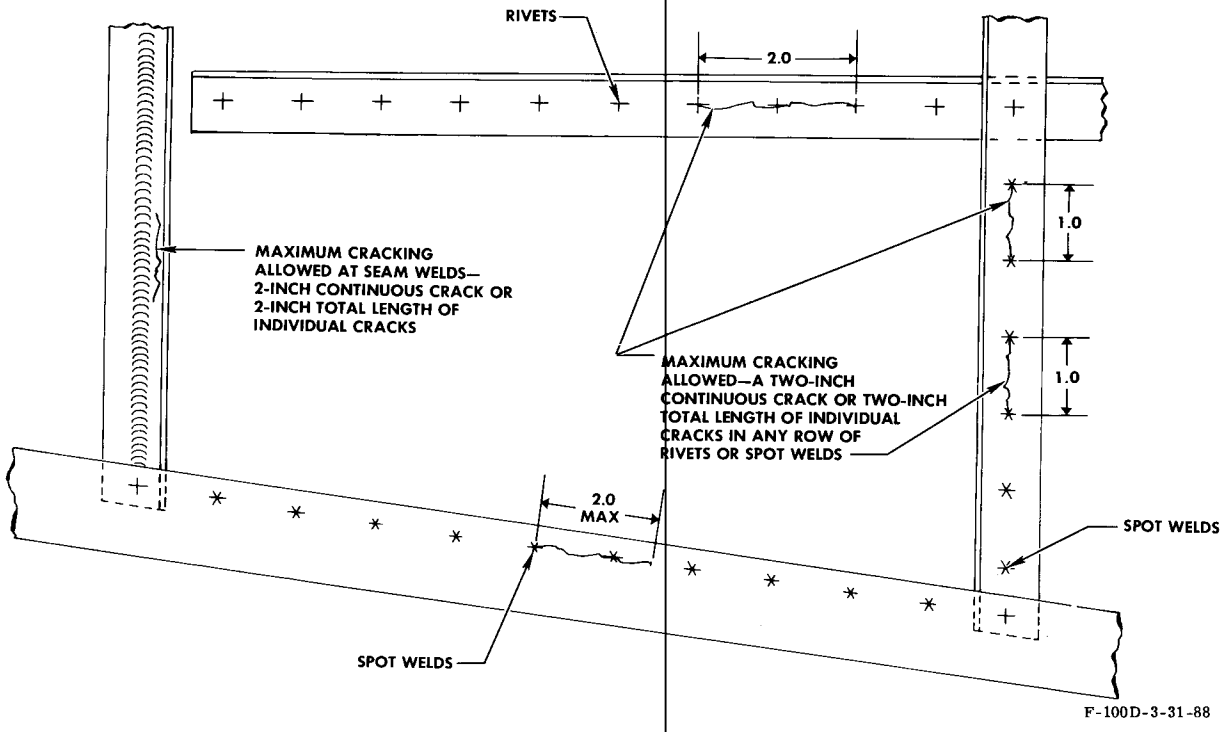
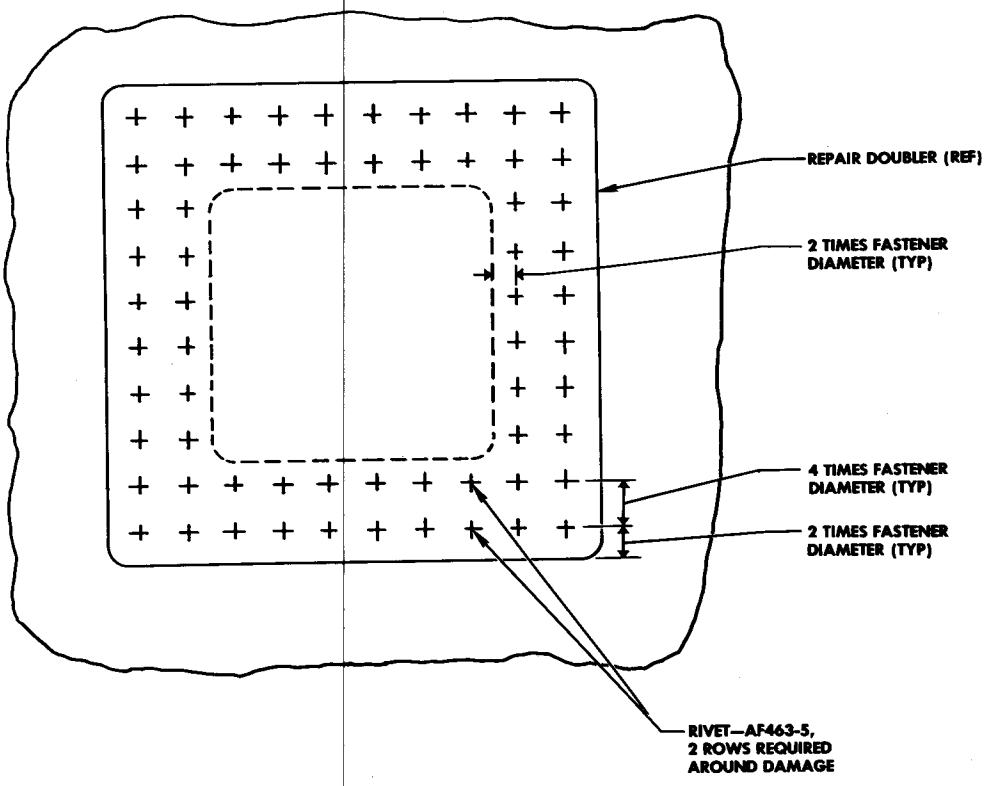
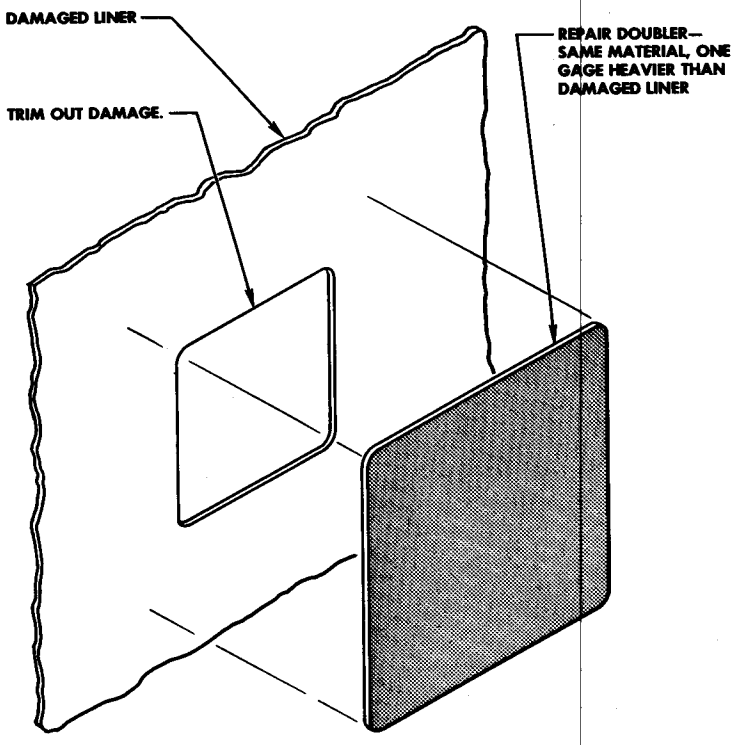


Figure 4-18A-1. Aspirator Skin Negligible Damage

PERMANENT CATEGORY A — FUSELAGE





1. Trim out damaged area, or stop-drill all cracks.

2. Make repair doubler from same material, one gage heavier than damaged liner.

3. Fasten repair doubler in place, using two rows of fasteners. Use type and size of fasteners and spacing called out.

4. If it is necessary to remove fasteners and they are larger than fasteners called out, replace with same size fasteners which were removed.

NOTE There is no limit to the size of the repair doubler.

- Replace all fasteners removed with existing sizes or larger.

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Figure 4-18B. Fuel Bay Liner Repair

PERMANENT—CATEGORY A

FUSELAGE

NOTE When the aft fuselage is being removed from the airplane, if the weight of the aft fuselage is not correctly supported, the longeron union stud may tear the access hole in the frame at station 389.0625. When this damage occurs, the repair shown may be used.

- Clean up damaged area and stop-drill all cracks with a No. 40 (0.098-inch) drill.

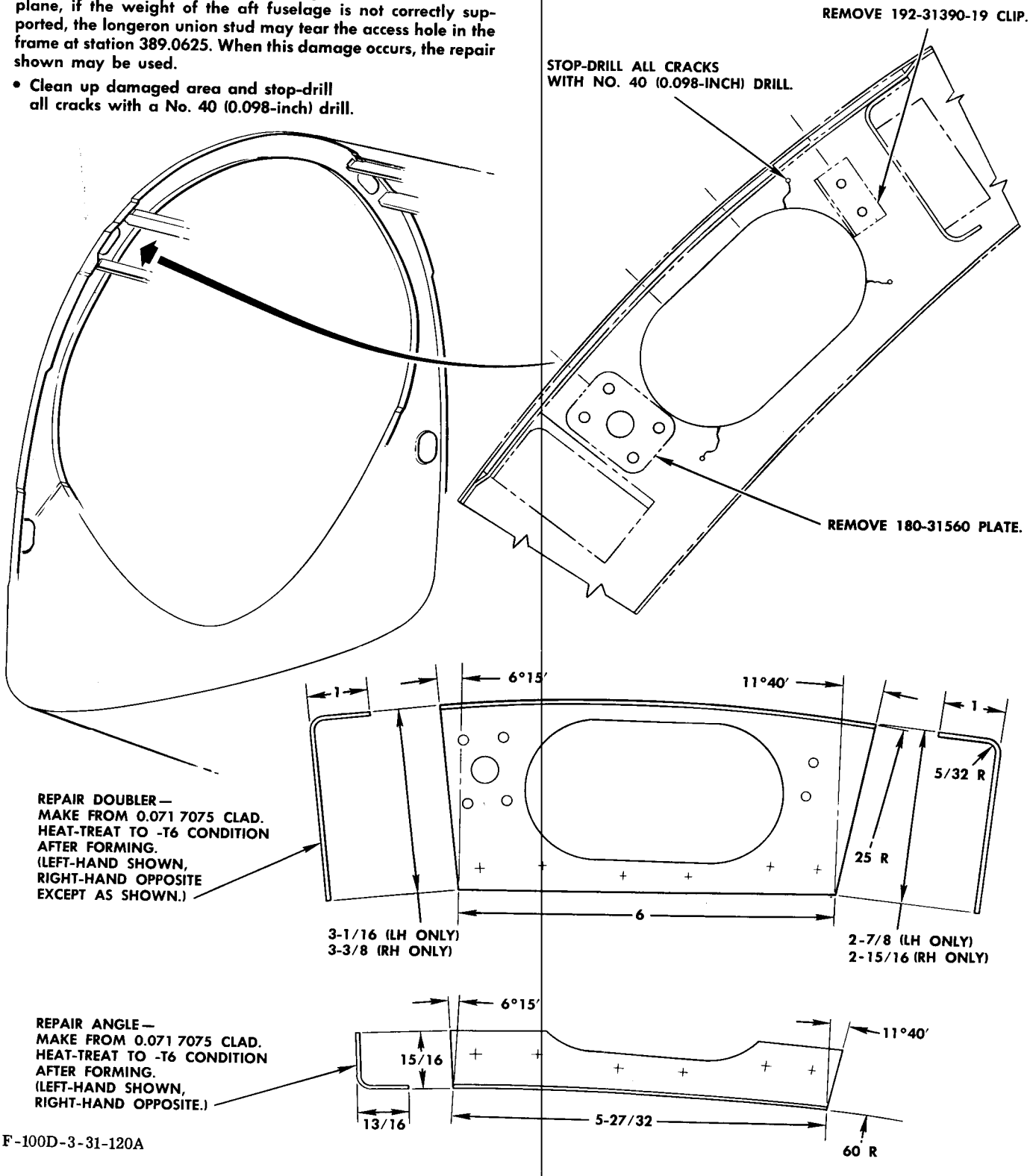


Figure 4-18C. Repair of Bulkhead at Station 389.062 (Sheet 1 of 2)

PERMANENT—CATEGORY A

FUSELAGE

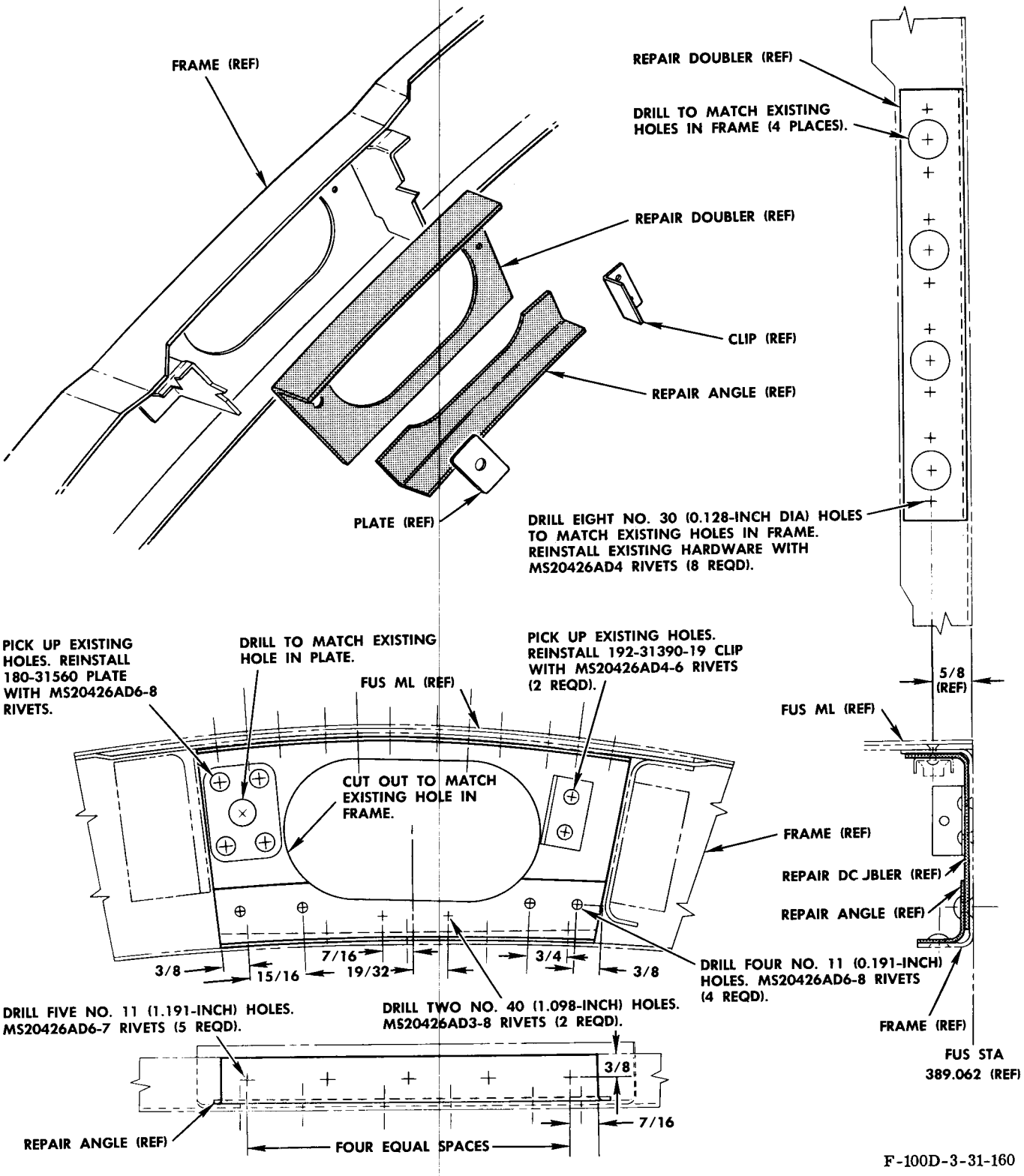
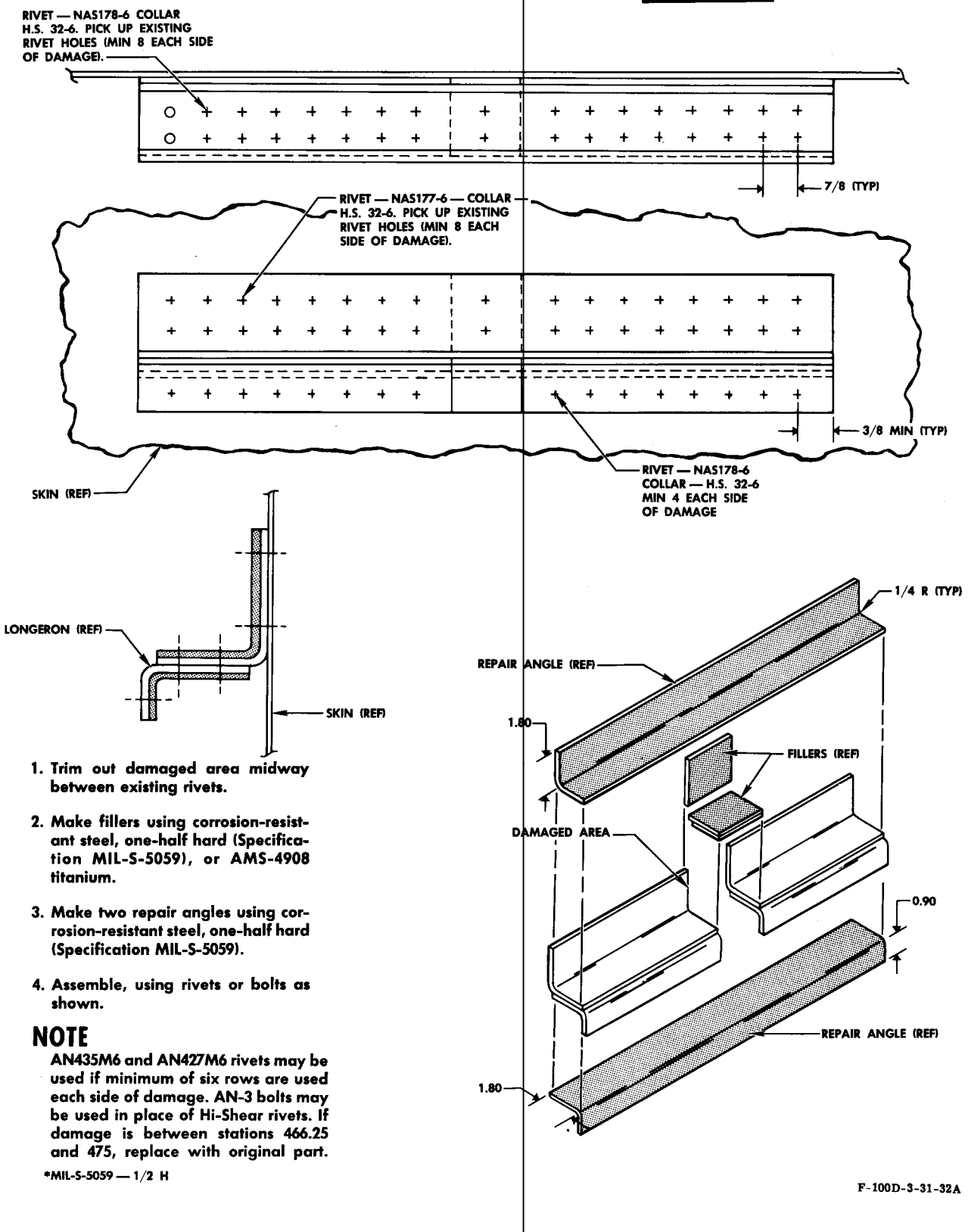


Figure 4-18C. Repair of Bulkhead at Station 389.062 (Sheet 2 of 2)

PERMANENT CATEGORY A

FUSELAGE



1. Trim out damaged area midway between existing rivets.
2. Make fillers using corrosion-resistant steel, one-half hard (Specification MIL-S-5059), or AMS-4908 titanium.
3. Make two repair angles using corrosion-resistant steel, one-half hard (Specification MIL-S-5059).
4. Assemble, using rivets or bolts as shown.

NOTE

AN435M6 and AN427M6 rivets may be used if minimum of six rows are used each side of damage. AN-3 bolts may be used in place of Hi-Shear rivets. If damage is between stations 466.25 and 475, replace with original part.

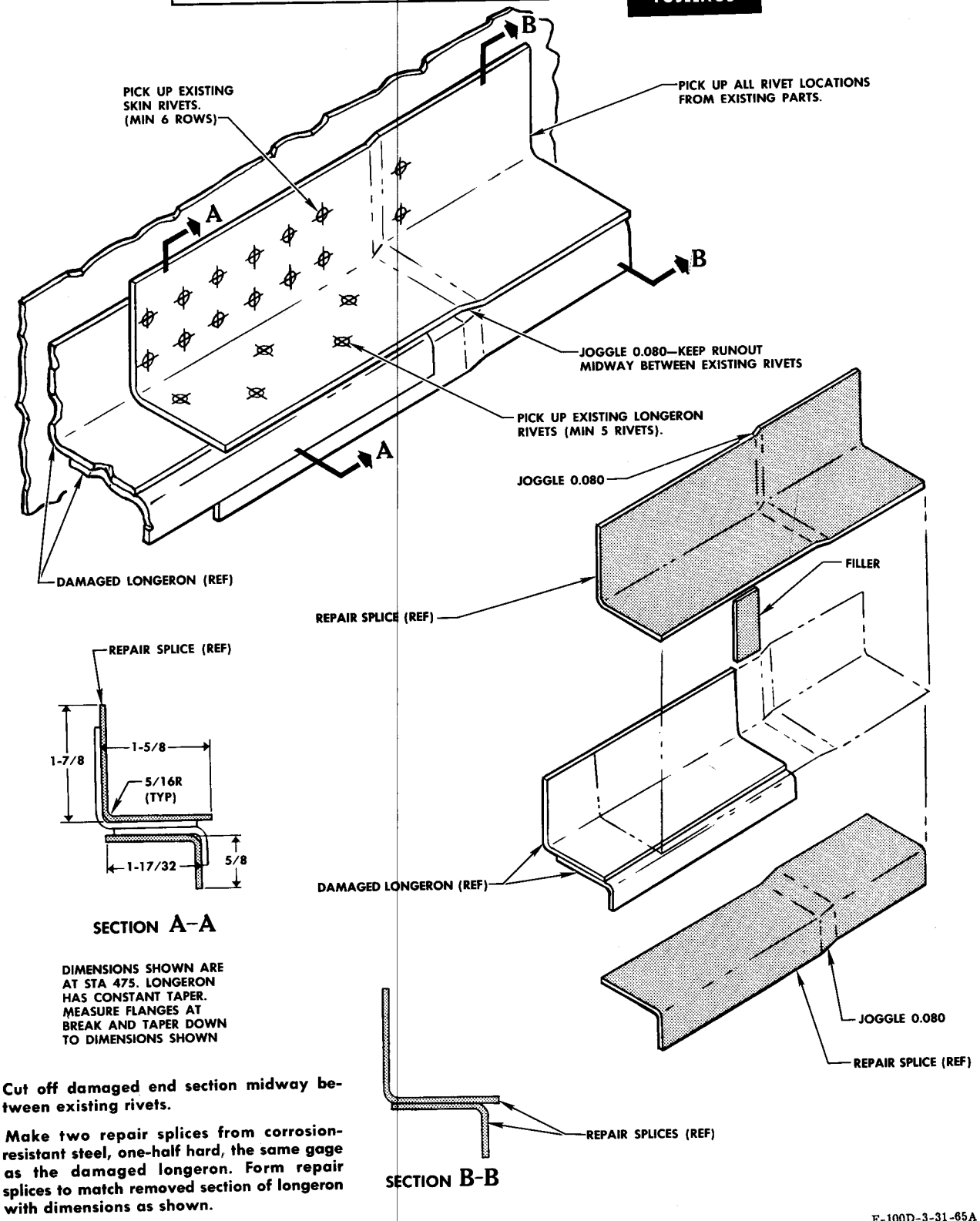
*MIL-S-5059 — 1/2 H

F-100D-3-31-32A

Figure 4-19. Lower Longeron—Station 456.5 to 475

PERMANENT—CATEGORY A

FUSELAGE



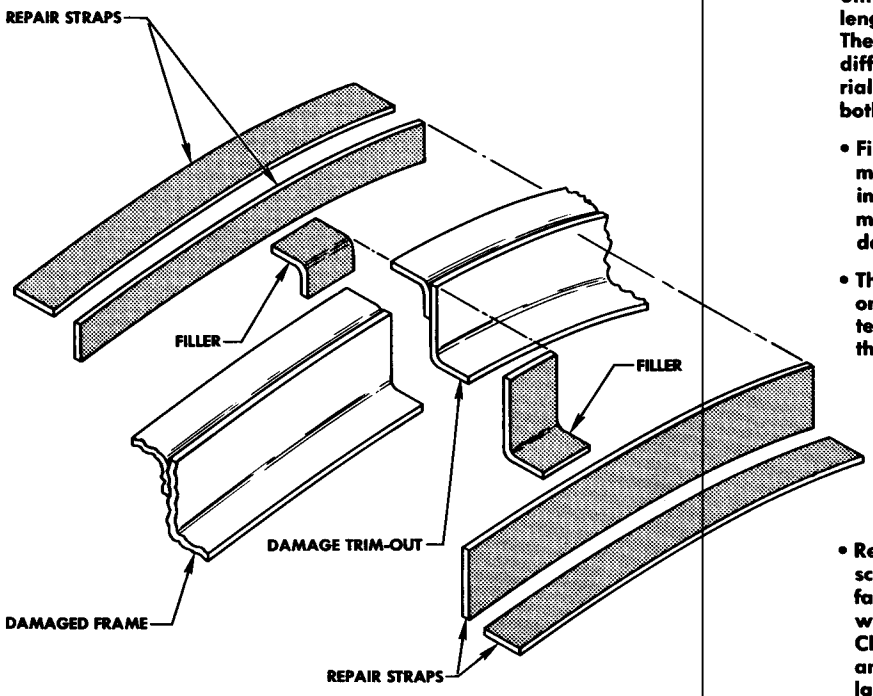
1. Cut off damaged end section midway between existing rivets.
2. Make two repair splices from corrosion-resistant steel, one-half hard, the same gage as the damaged longeron. Form repair splices to match removed section of longeron with dimensions as shown.
3. Assemble and rivet as shown.

F-100D-3-31-65A

Figure 4-20. Lower Longeron—Station 466 to 475

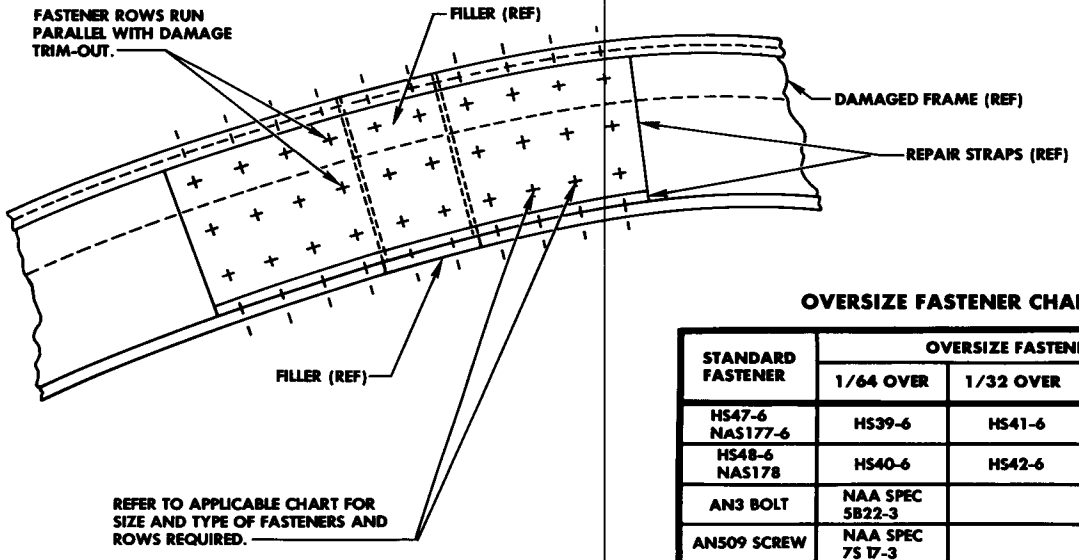
PERMANENT CATEGORY A

FUSELAGE



NOTE When titanium frames are being repaired with corrosion-resistant steel, and only one frame cap is damaged, the total length of the repair may not exceed 10 inches. These restrictions are necessary because of the different rates of expansion of the two materials. There are no restrictions to length when both frame caps are being repaired.

- Fillers for a damage length up to 9/16 inch may be made from flat stock. Fillers over 9/16 inch must match the original part. Fillers are made from the same material and gage as the damaged part.
- The length of the repair straps is dependent on the number of fastener rows required. Fastener rows are considered to run parallel with the damage.
- Replace monel rivets with AN. bolts, AN. screws, Hi-shear rivets, or Jo-bolts. Oversize fasteners will be necessary where monel rivets were removed. (Refer to "Oversize Fastener Chart" on this page.) Where edge distance and radius clearance permit, the next size larger fastener may be used.



OVERSIZE FASTENER CHART

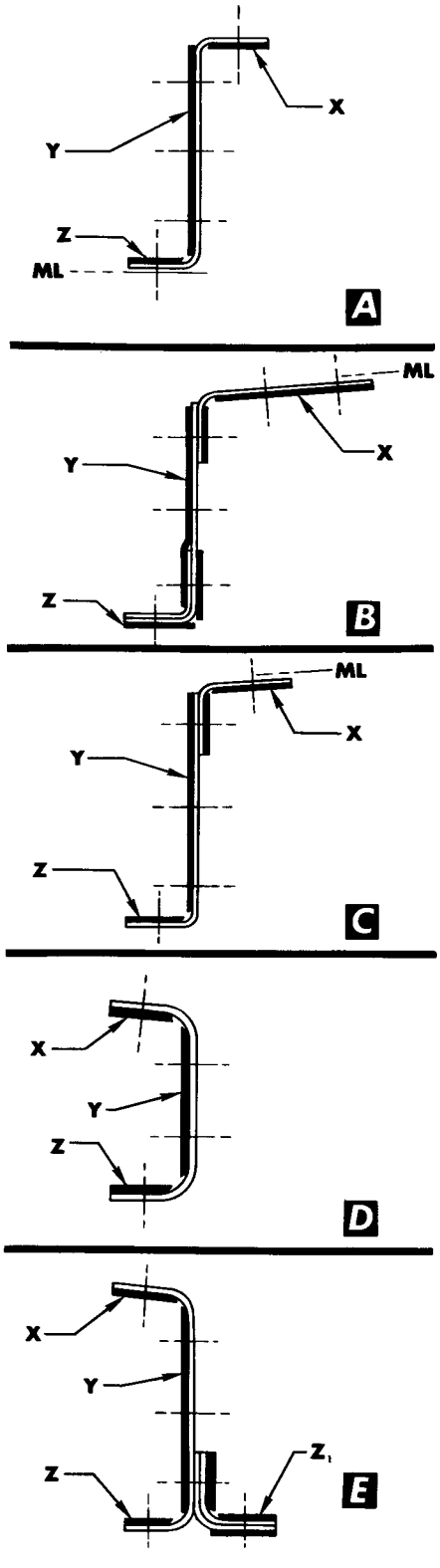
STANDARD FASTENER	OVERSIZE FASTENERS		
	1/64 OVER	1/32 OVER	1/64 OVER
HS47-6 NAS177-6	HS39-6	HS41-6	
HS48-6 NAS178	HS40-6	HS42-6	
AN3 BOLT	NAA SPEC 5B22-3		JO-BOLT P200 or P260
AN509 SCREW	NAA SPEC 7S 17-3		
CR-562 RV291C	CR-578		
CR-563 RV290C	CR-579		
Refer to index for sources of commercial products.			

F-100D-3-31-102

Figure 4-21. Frame Cap Repair Instructions

PERMANENT CATEGORY A

FUSELAGE



STA	VIEW	FASTENER	ROWS REQD EA SIDE OF DAMAGE X	Y	Z	Z_1	
456.5 LOWER	A	AN427M4			4		
		NAS508M5	4	4			
460.75 RH LOWER	A	AN427M4			4		
		NAS508M5	4	4			
466.25 LOWER	A	AN427M4			4		
		NAS508M5	4	4			
503.25 SIDE	A	NAS508M4			4		Z—RIVETS BETWEEN SPOT WELDS
		NAS508M5	4	4			
466.25 UPPER	B	AN427M5	4				
		AN435M5		4	4		
466.25 SIDE	C	2R14-6	4				X—2 ROWS THRU ML
		AN435M5		4	4		
456.5 UPPER	B	AN427M6	4				X—1 ROW THRU ML
		AN435M5		4	4		
456.5 SIDE	C	AN427M5	4				
		AN435M5		4	4		
448.5 UPPER	D	AN427M6	5				
		NAS508M5		5			
		NAS508M6			5		
475 UPPER	D	2R14-6	5				
		NAS508M5		5			
		NAS508M6			5		
511.5 SIDE	D	2R14-6	4				
		NAS508M5		4	5		
420 UPPER	E	AN427M5	5*				
		NAS508M5		4	5	5	

NOTE The splice straps are of one-half hard corrosion-resistant steel sheet, the next higher gage than the cap being repaired.

- Splice straps may be on either side of the webs.
- Fillers for a damage length up to 9/16 inch may be made from flat stock. Fillers over 9/16 inch must match the original part.
- Chamfer edges to fit radius.
- The fastener spacing is four times the fastener diameter. Use existing spacing wherever possible.
- If a removed fastener is larger than the fasteners called for on chart, replace with the same type of fastener as was removed.

* File fastener head flush with skin.

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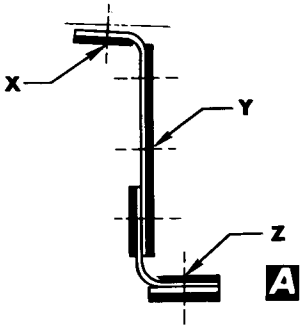
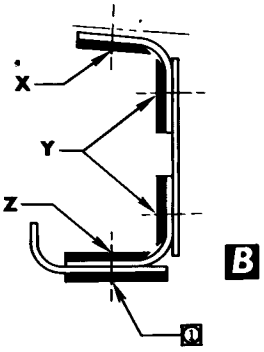
Figure 4-22. Frame Cap Repairs (Sheet 1 of 5)

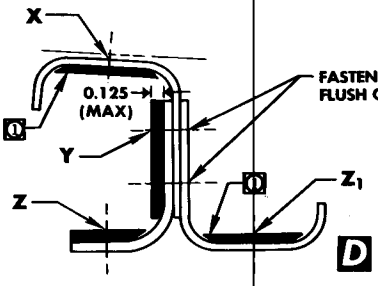
PERMANENT—CATEGORY A

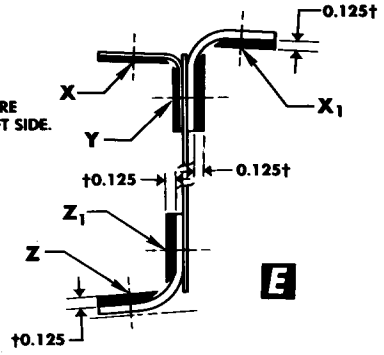
FUSELAGE

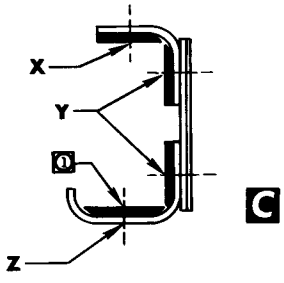
STA	VIEW	FASTENERS	ROWS REQUIRED EACH SIDE OF DAMAGE				
			X	X_1	Y	Z	Z_1
410.5 UPPER	A	AN427M4	6				
		NAS508M6			4	4	
475 SIDE	B	2R14-6	6				
		HS68-6, HS60-6			4		
		AN508M6				7	
475 LOWER	C	HS68-6, HS60-6	4		4	5	
495 UPPER AND SIDE	D	HS67-6, HS60-6	7		5		
		AN508M6				8	
		HS68-6, HS60-6					7
495 LOWER	E	2R14-6				7*	6*
		AN508M6		7	6		
		AN508M5	4				











NOTE The splice straps are of one-half hard corrosion-resistant steel sheet, the next higher gage than the frame caps being repaired, except as noted.

- The splice straps may be on either side of the web or cap.
- Fillers for a damage length up to 9/16 inch may be made from flat stock. Fillers over 9/16 inch must match the original part.
- Chamfer edges to fit radius.
- The fastener spacing is four times the fastener diameter.
- Use existing spacing whenever possible.
- If fastener removed is larger than called for on chart, replace with the same type of fastener as was removed.
- A maximum of two sheets of equal thickness may be used to build up the splice strap thickness if the required thickness is over 0.125 inch.

* For 235-31477, add three additional rivets each side of damage at X, Y, Z, and Z_1 or change to HS67-6 and HS68-6 rivets with HS60-6 collars.

† On 235-31477 only.

Ⓠ Splice straps are 1-1/2 times cap gage.

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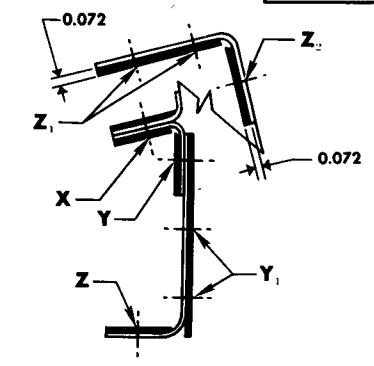
Figure 4-22. Frame Cap Repairs (Sheet 2 of 5)

PERMANENT—CATEGORY A

FUSELAGE



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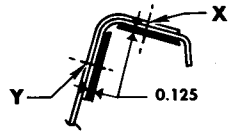

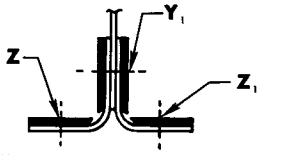

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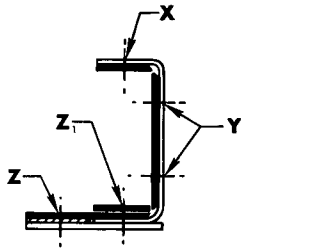

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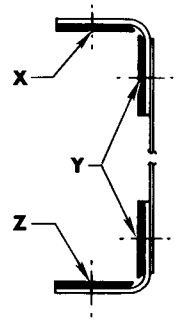
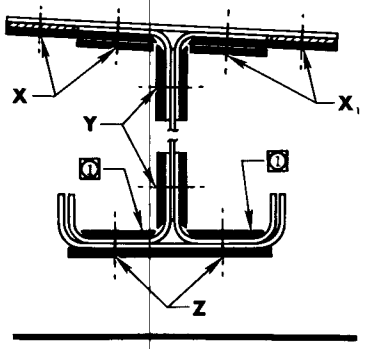
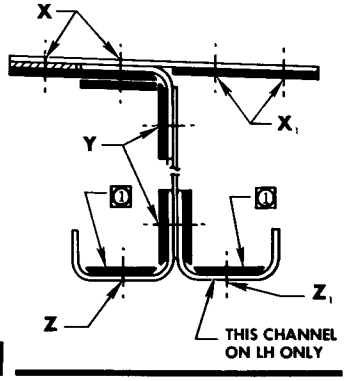


D

STATION	VIEW	FASTENER	ROWS OF FASTENERS REQD EACH SIDE OF DAMAGE						
			X	X_1	Y	Y_1	Z	Z_1	Z_2
401 UPPER	A	2R14-6						6 OR	
		AN427M6						6 OR	
		HS47-6, HS32-6						4	
		NAS508M5			4 OR	4 OR	4 OR		7 OR
		NAS508M6			3	3	3		4
		CR563M5	5 OR						
		CR563M6	3						
401 SIDE	B	2R14-6					4 OR	4 OR	
		AN427M6					4 OR	4 OR	
		AN427M5					6 OR	6 OR	
		HS47-6, HS32-6					3	3	
		NAS508M5	9 OR		9 OR	6 OR			
		NAS508M6	6		6	4			
448.5 LOWER	C	AN427M5					4	4	
		NAS508M5	4		4				
410.5 SIDE	D	AN427M5	7						
		NAS508M6			5		5		
420 SIDE LEFT-HAND ABOVE ℄ THRUST	E	2R14-6	4	4					
		NAS508M6			5		5		
420 SIDE LEFT-HAND BELOW C_L THRUST AND RIGHT-HAND	F	2R14-6	4	4					
		NAS508M6			4		5	5	



E



F

① STRAP IS 1-1/2 TIMES CAP GAGE.

NOTE The doublers are of one-half hard corrosion-resistant steel, the next higher gage than the cap being repaired, except as noted. The splice straps may be on either side of the cap or web.

- The fastener spacing is four times the fastener diameter. Use existing spacing whenever possible.
- The fillers for a damage length up to 9/16 inch may be made from flat stock. The fillers over 9/16 inch must match the original part.
- Chamfer edges to fit radius.
- If removed fastener is larger than fastener called for in chart, replace with same type of fastener as was removed.
- If more than one type of fastener is called out for the same strap, the selection of the fastener type is optional.

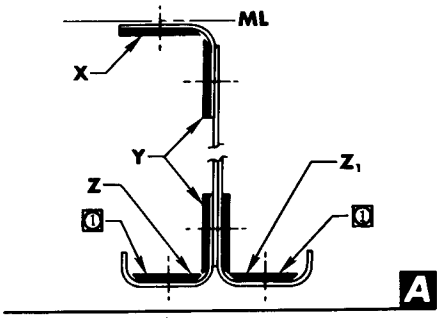
F-100D-3-31-107

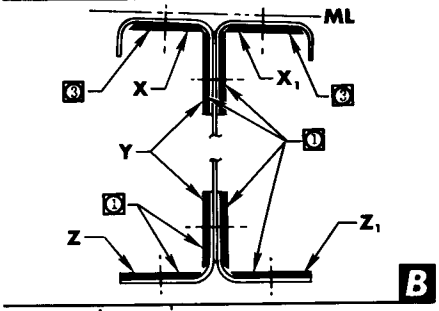
Figure 4-22. Frame Cap Repairs (Sheet 3 of 5)

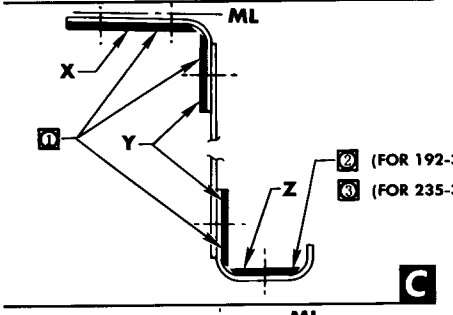
PERMANENT CATEGORY A

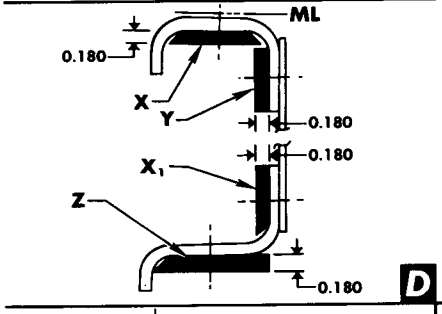
FUSELAGE

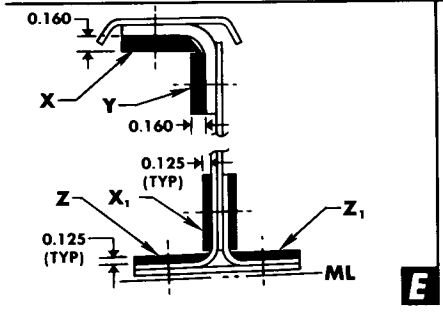
FRAME CAP REPAIR

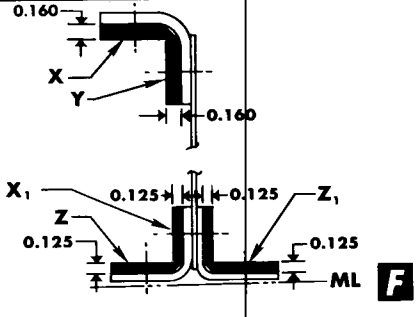












			ROWS REQD EACH SIDE OF DAMAGE					
STA	VIEW	FASTENER	X	X_1	Y	Z	Z_1	
429.5 UPPER (TI AL SH)	A	AN427M5	6					X-FILE HEADS FLUSH
		NAS508M5			6			
		NAS508M6				5	5	
429.5 UPPER (17-7 PH)	B	HS67-6, HS60-6	6	6				Y-SPACE RIVETS BETWEEN SPOT WELDS
		NAS508M6			6	7	7	
429.5 SIDE (TI AL SH)	C	HS67-6, HS60-6	4					X-RH HAS 1 ROW RIVETS THRU ML
		NAS508M6			5	5		
429.5 SIDE (17-7 PH)	C	HS67-6, HS60-6	5					X-RH HAS 1 ROW RIVETS THRU ML
		HS68-6, HS60-6			5	7		Y-SPACE RIVETS BETWEEN SPOT WELDS
440 UPPER	D	HS47-6, HS32-6	9 OR					
		HS47-8, HS32-8	5					
		HS68-6, HS60-6		6 OR	6 OR	9 OR		
		HS68-8, HS60-8		4	4	6		
440 SIDE	E	2R14-6				8 OR	8 OR	
		AN508M6	8 OR	8 OR	8 OR			
		HS47-6, HS32-6				5	5	
		HS68-6, HS60-6	5	5	5			
440 LOWER	F	2R14-6				7 OR	7 OR	
		AN508M6	10 OR	10 OR	7 OR			
		HS47-6, HS32-6				5	5	
		HS68-6, HS60-6		6	5			

NOTE The splice straps are of one-half hard corrosion-resistant steel sheet, the next higher gage than the cap being repaired, except as noted.

- The splice straps may be on either side of the web being repaired.
- Fillers for a damage length up to 9/16 inch may be made from flat stock. Fillers over 9/16 inch must match the original part.
- Chamfer splice strap edges to fit radius.
- The fastener spacing is four times the fastener diameter.
- Use existing spacing wherever possible.
- If a removed fastener is larger than called for on the chart, replace with the same type of fastener as was removed.
- A maximum of two sheets of equal thickness may be used to build up the splice strap thickness if the required thickness is over 0.125 inch.
- The fasteners should be spaced between the spot welds.
- If more than one type of fastener is called out for the same strap, the selection of the type of fastener is optional.

1 STRAP IS 1-1/2 TIMES CAP GAGE.

2 STRAP IS 2 TIMES CAP GAGE.

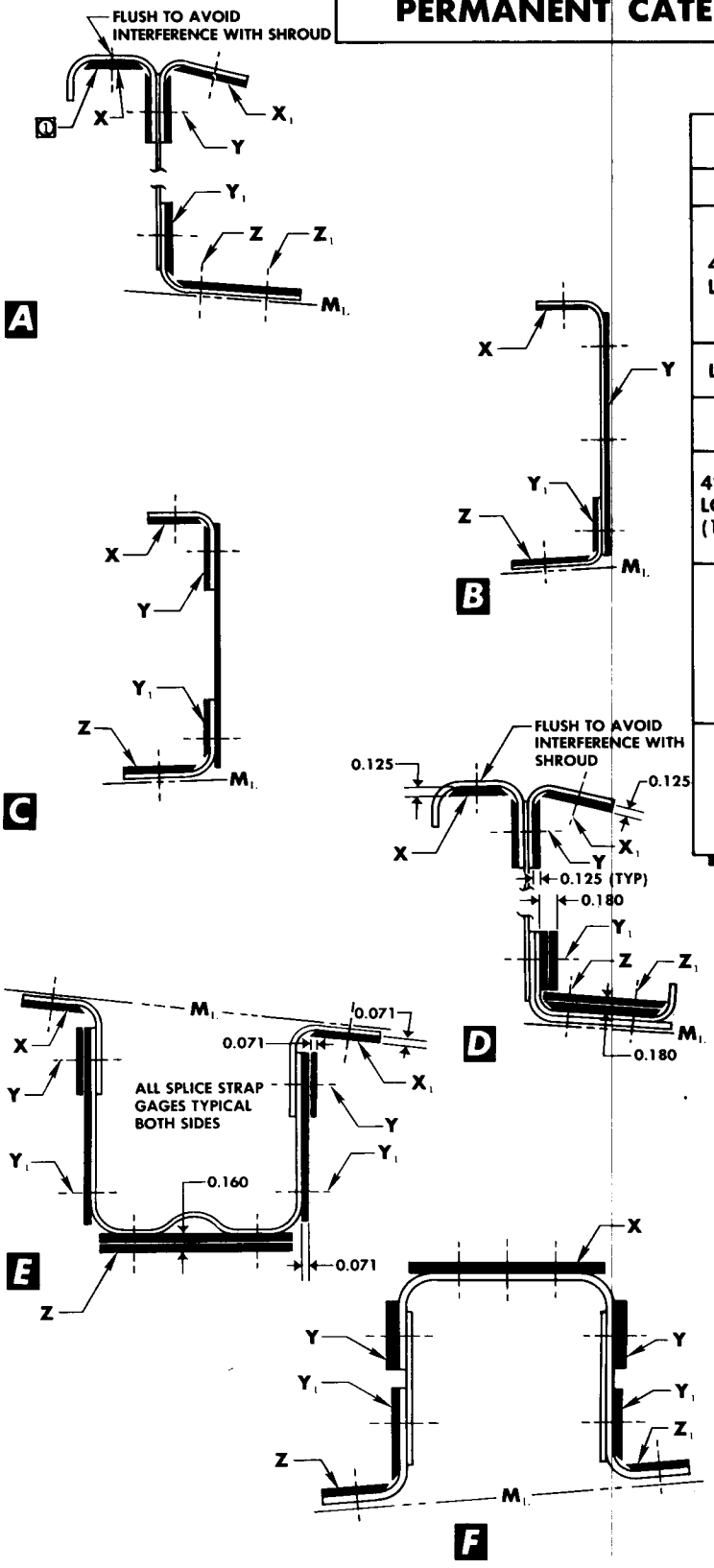
3 STRAP IS 2-1/2 TIMES CAP GAGE (MAXIMUM STRAP THICKNESS REQUIRED IS 0.125 INCH).

F-100D-3-31-106

Figure 4-22. Frame Cap Repairs (Sheet 4 of 5)

PERMANENT CATEGORY A

FUSELAGE



STA	VIEW	FASTENER	ROWS REQD EA SIDE OF DAMAGE X	X1	Y	Y1	Z	Z1
429.5 LOWER	A	2R14-6	5					
		CR563-5		7				
		NAS508M5			6	6		
		AN427M5						6
		AN427M4					6	
448.5 LOWER SIDE	B	AN427M5					4	
		NAS508M5	4		4	4		
448.5 UPPER SIDE	C	AN427M5					4	
		NAS508M5	4		4	4		
429.5 LOWER (17-7PH)	D	2R14-6	10 OR					
		P200 JO-BOLTS		7				
		HS47-6 HS15-6	7				7	7
		HS48-6 HS32-6			7	7		
485 UPPER	E	2R14-6	5 OR	5 OR				
		AN427M5	7 OR	7 OR				
		HS67-6 HS60-6	4	4				
		CR563-6			11 OR	10 OR		
		P200A			6	5	9 OR	9 OR
		P260A					6	6
485 SIDE	F	P200A	5 OR		5 OR	4		
		P260A	4		4	OR		
		CR563-6				8		
		2R14-6					6 OR	6 OR
		HS67-6 HS60-6					4	4

NOTE The splice straps are of one-half hard corrosion-resistant steel, the next higher gage than the cap being repaired, except as noted.

- The splice straps may be on either side of the webs.
- Fillers for a damage length up to 9/16 inch may be made from flat stock. Fillers over 9/16 inch must match the original part.
- Chamfer edges to fit radius.
- The fastener spacing is four times the fastener diameter. Use existing spacing whenever possible.
- If the fastener removed is larger than the fastener called for in the chart, replace with the same type of fastener as was removed.
- A maximum of two sheets of equal thickness may be used to build up the splice strap thickness if the required thickness is over 0.125 inch.
- If more than one type of fastener is called out for the same strap, the selection of the fastener type is optional.

Q Splice strap is 1-1/2 times cap gage.

F-100D-3-31-105

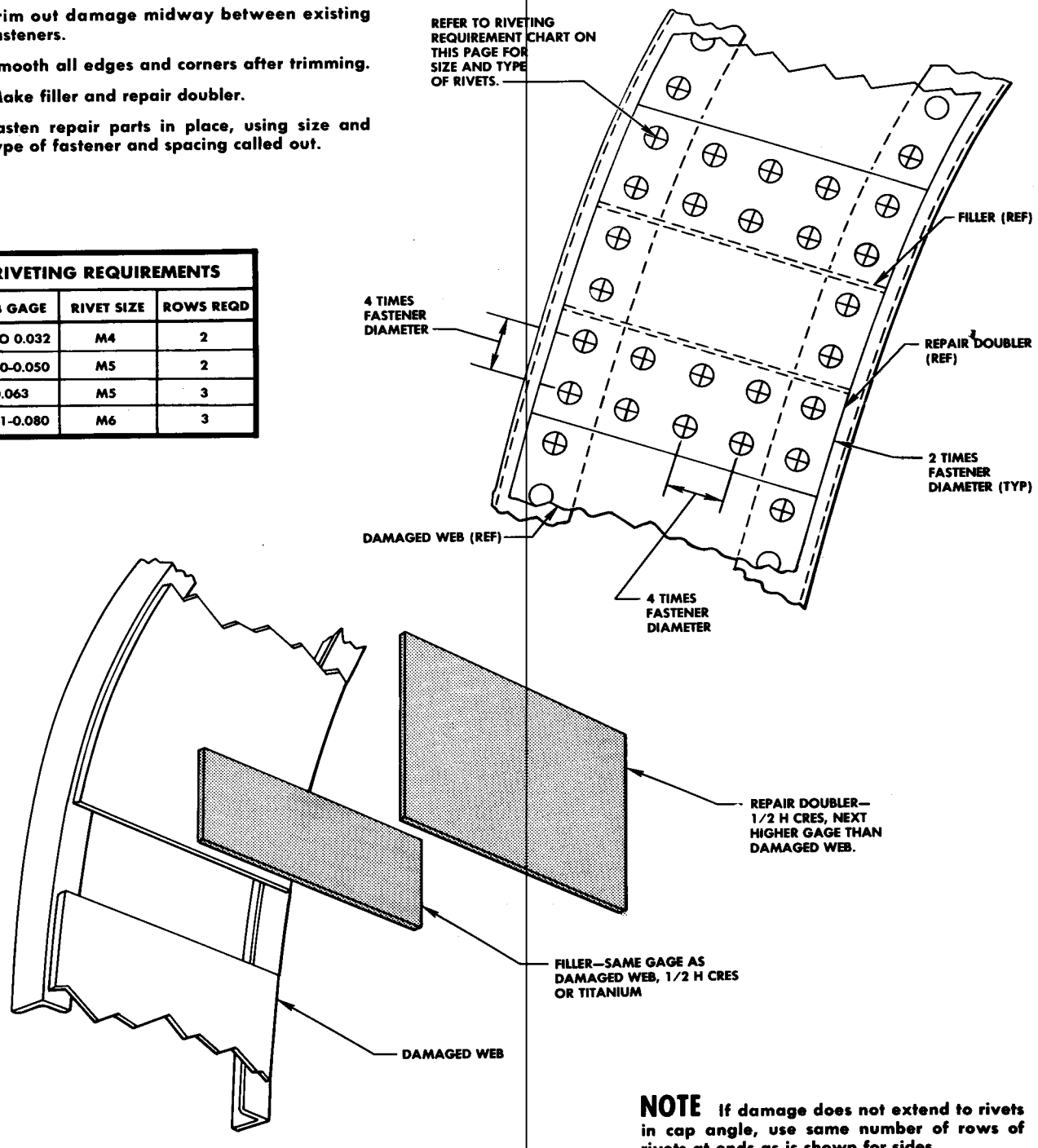
Figure 4-22. Frame Cap Repairs (Sheet 5 of 5)

PERMANENT-CATEGORY A

FUSELAGE

1. Trim out damage midway between existing fasteners.
2. Smooth all edges and corners after trimming.
3. Make filler and repair doubler.
4. Fasten repair parts in place, using size and type of fastener and spacing called out.

RIVETING REQUIREMENTS		
WEB GAGE	RIVET SIZE	ROWS REQD
UP TO 0.032	M4	2
0.040-0.050	M5	2
0.063	M5	3
0.071-0.080	M6	3



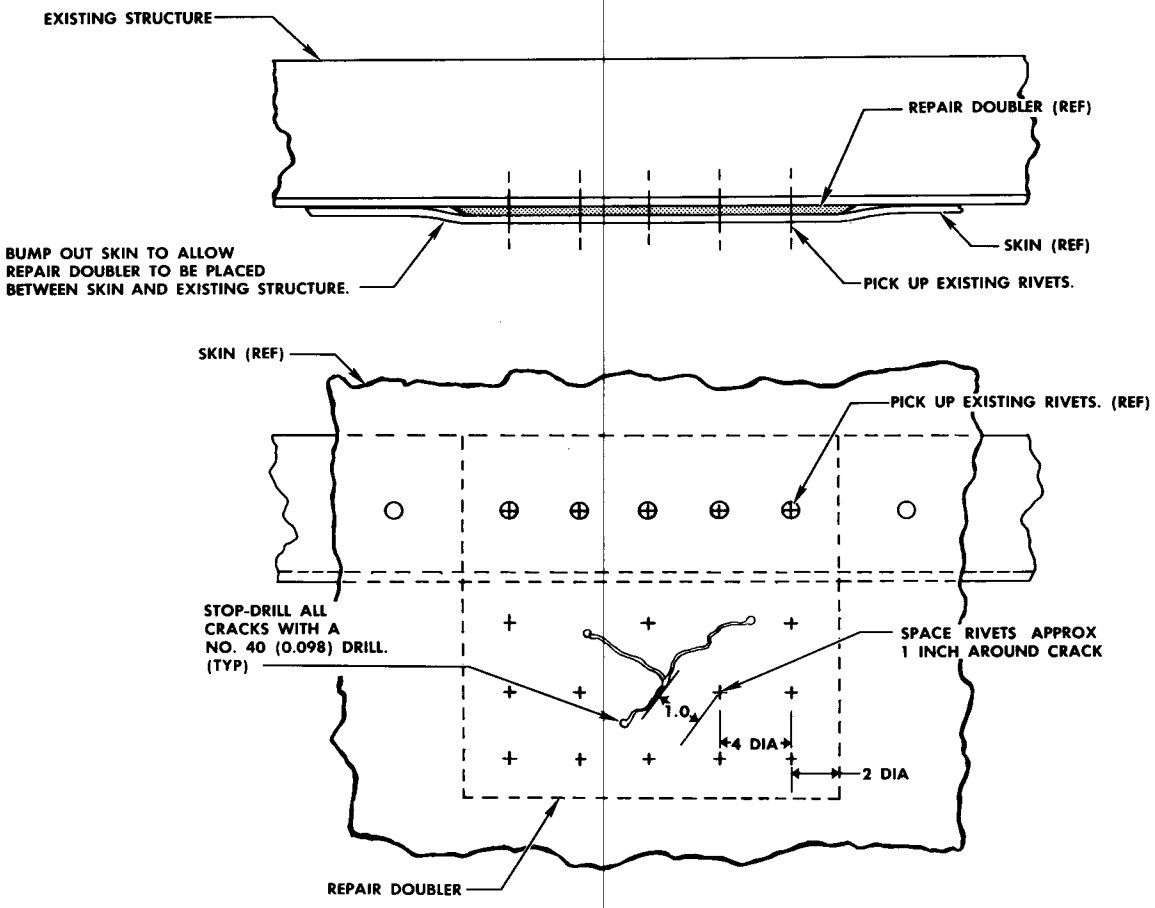
NOTE If damage does not extend to rivets in cap angle, use same number of rows of rivets at ends as is shown for sides.

F-100D-3-31-104

Figure 4-23. Titanium Web Repairs

PERMANENT CATEGORY A

FUSELAGE



1 Stop-drill all cracks with No. 40 (0.098-inch) drill.

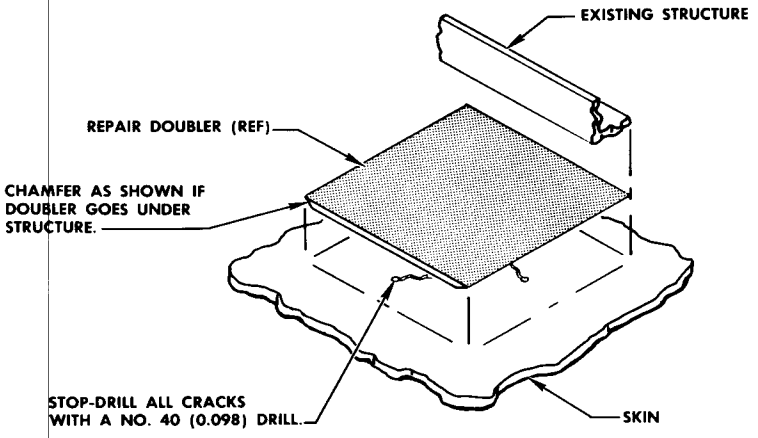
2 Fasten doubler in place with two rows of fasteners. The inner row should be spaced about one inch around the crack. The outer row should be spaced four times the rivet diameter with the edge distance two times the rivet diameter.

3 If doubler goes under substructure, chamfer all edges as shown.

4 Use AN427M rivets if titanium dimpling equipment is available. Do not countersink skin or doublers.

NOTE

- Temporary repair may be made using AN435M rivets when titanium dimpling equipment is not available.
- Doubler may be installed externally for temporary repair.

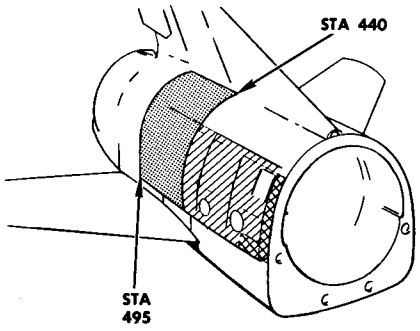


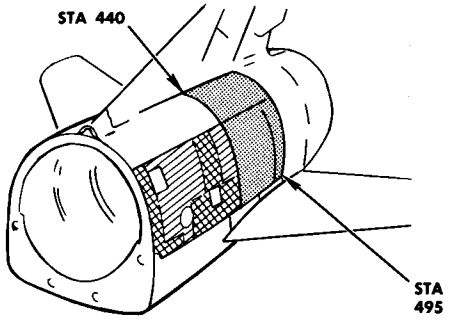
F-100D-3-31-47C

Figure 4-25. Aft Fuselage Skins—0.020- to 0.032-inch Titanium

PERMANENT CATEGORY A

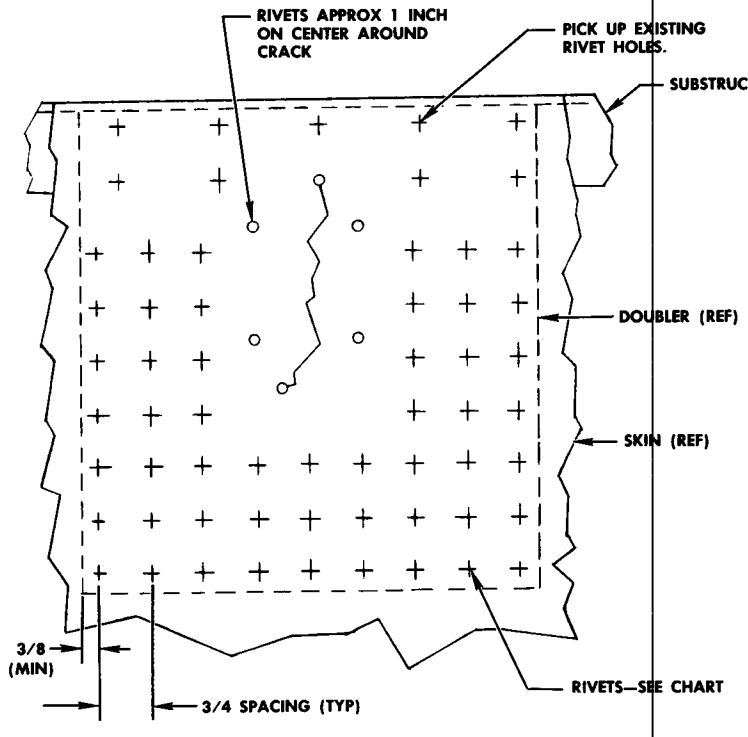
FUSELAGE

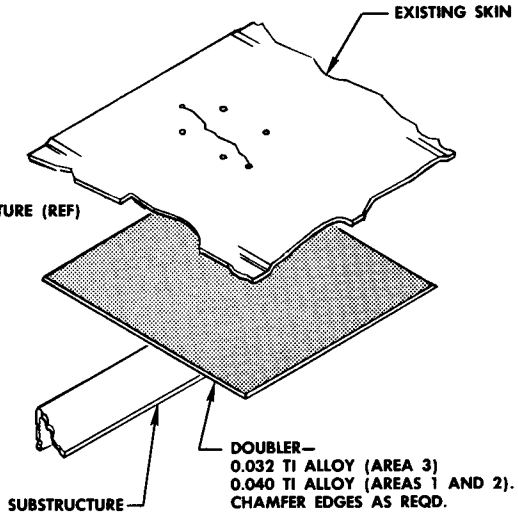




AREA	RIVETS	ALTERNATE RIVETS
1	2 ROWS AN435M6	1 ROW NAS178-6 HS32-6 COLLAR
2	3 ROWS AN435M6	2 ROWS NAS178-6 HS32-6 COLLAR
3	1 ROW AN435M6	1 ROW NAS178-6 HS32-6 COLLAR

AREA 1
AREA 2
AREA 3





1 Stop-drill all cracks with a No. 40 (0.098-inch) drill.

2 Determine doubler size from size of crack and rivet spacing shown on this page.

3 Determine size and type of fasteners to be used from fastener chart. Hi-Shear rivets may not be used with a flush-type dimpled repair.

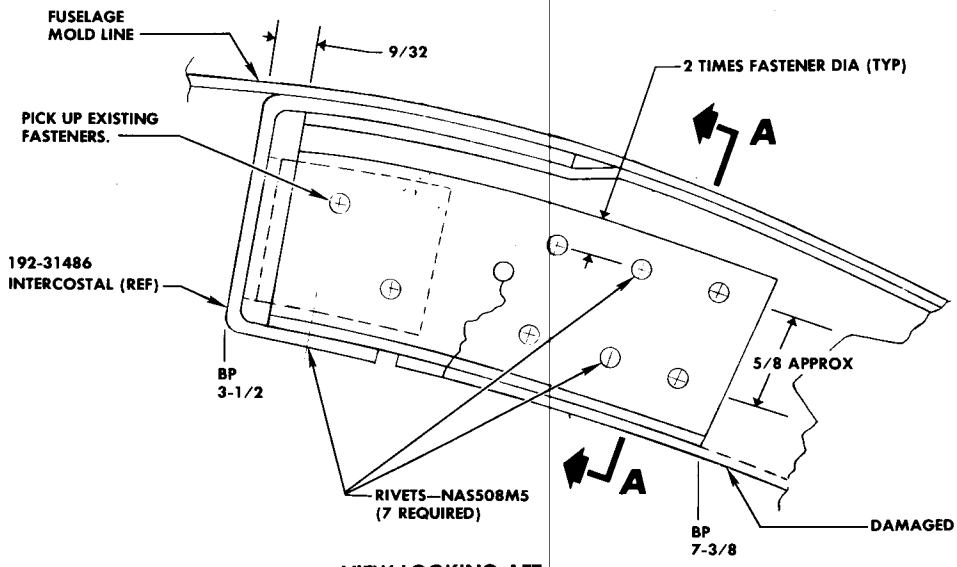
4 Do not countersink the skin or doubler.

5 This repair is also applicable to the center area of the skin with listed rows of rivets on all four sides.

F-100D-3-31-48A

Figure 4-26. Aft Fuselage Skins—0.040-inch Titanium Alloy

PERMANENT CATEGORY A — FUSELAGE



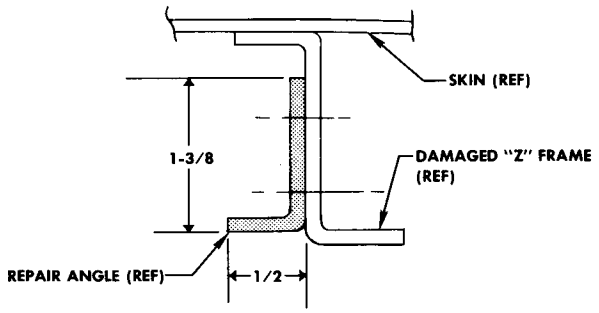
VIEW LOOKING AFT

1 Stop-drill all cracks with a 1/4-inch drill.

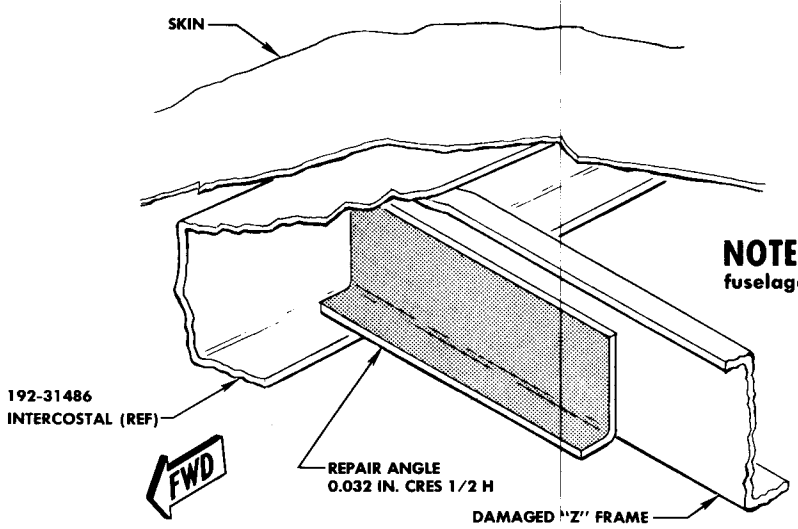
2 Make repair angle from 0.032-inch one-half hard corrosion-resistant steel.

3 Contour repair angle to match damaged frame.

4 Fasten repair angle in place, using seven NAS508M5 rivets and picking up the two existing fasteners.



SECTION A-A



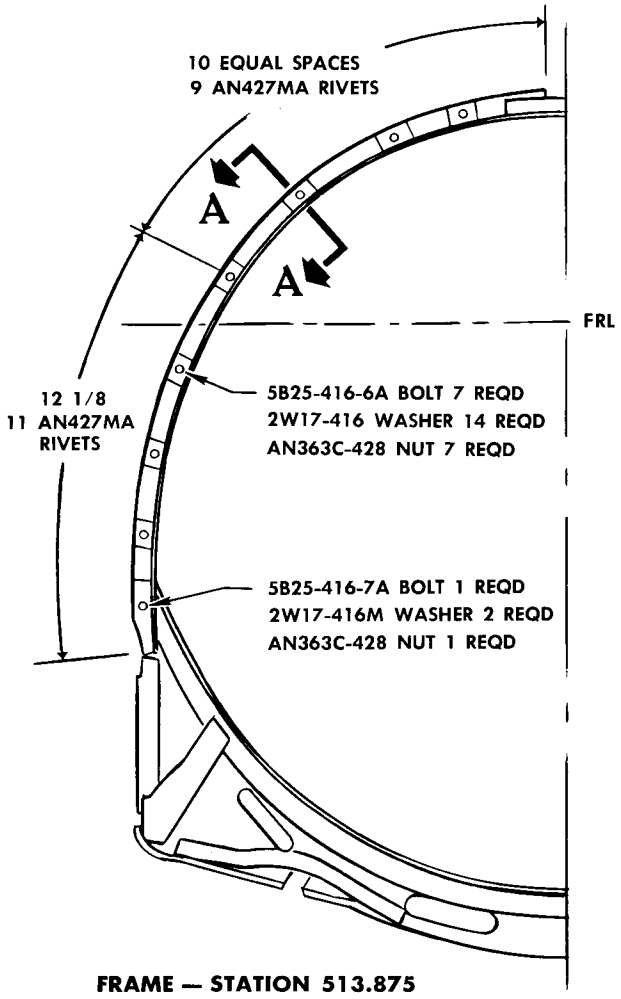
NOTE This repair is a specific repair for the upper fuselage frame at station 503.25.

F-100D-3-31-121

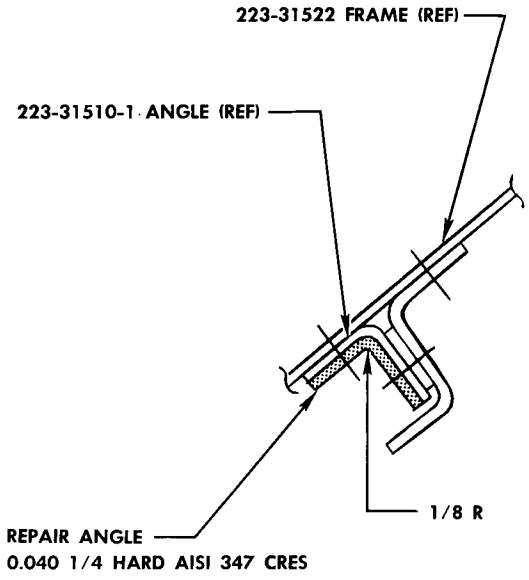
Figure 4-26A. Frame End Repair — Station 503.25

PERMANENT—CATEGORY A

FUSELAGE



FRAME — STATION 513.875
(LOOKING FORWARD LH SHOWN, RH OPPOSITE)



DETAIL A-A

NOTE Any cracks in area of bolt holes in 223-31510 angle will require an addition of a repair angle.

1 Remove bolts attaching 223-31510 angle to frame.

2 Fabricate repair angle of material indicated and form to shape of original angle.

3 Drill F(0.257) holes in repair angle to match 223-31522 frame.

4 Rivet repair angle to original angle and skin, using rivets and spacing indicated.

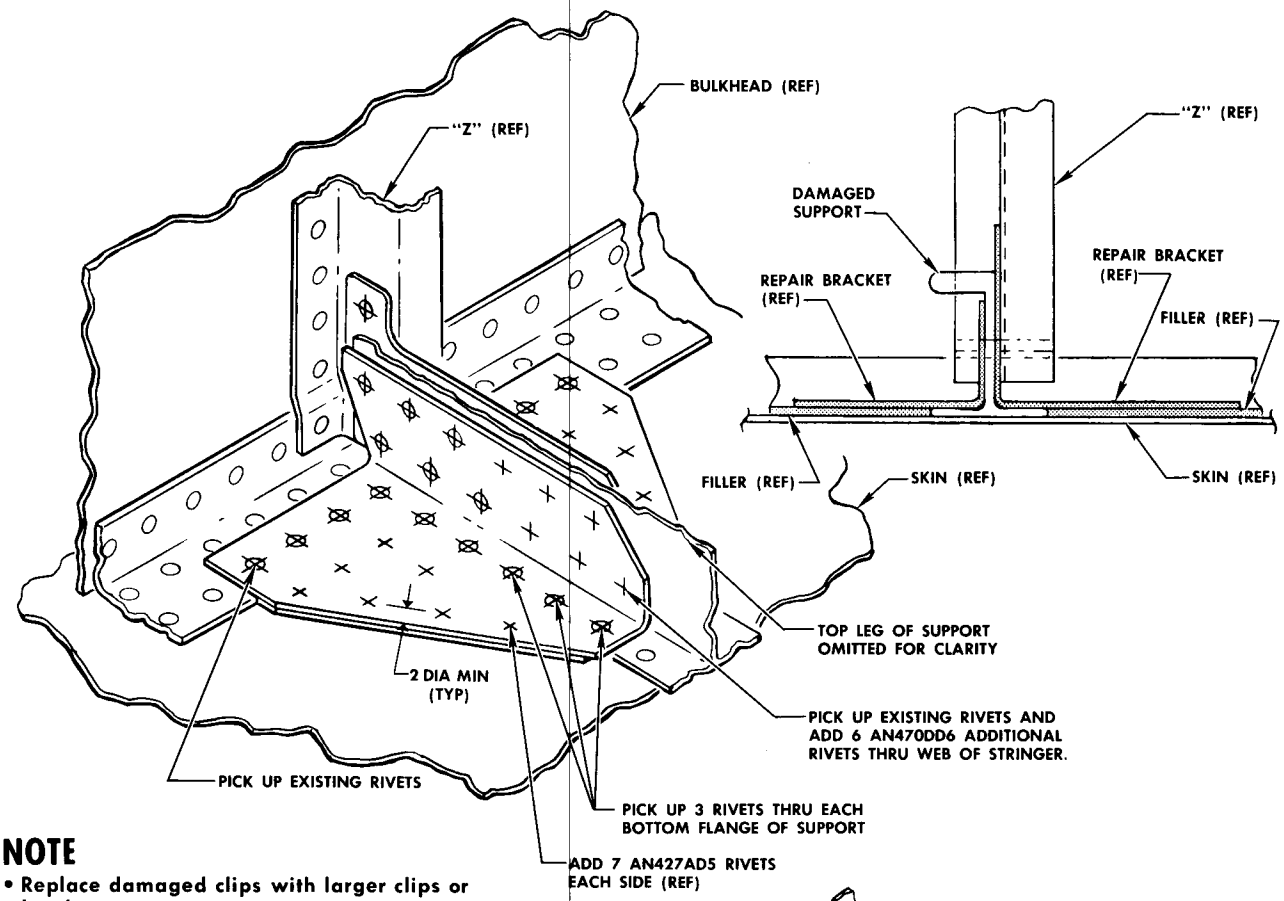
5 Reinstall nuts, bolts, and washers as indicated. Torque bolts to 30-40 inch-pounds.

F-100D-3-31-155

Figure 4-26B. Bolting Angle Crack Repair—Station 519.875

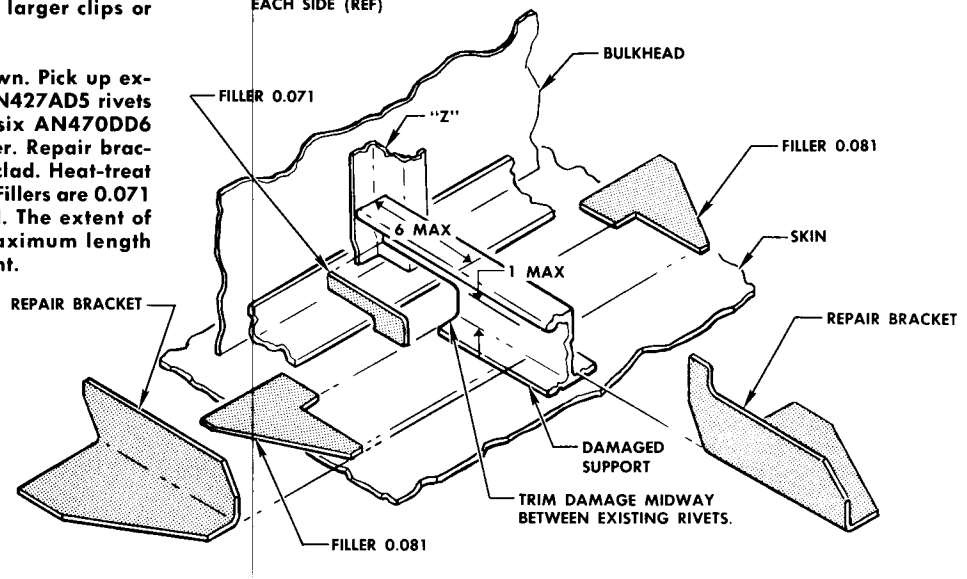
PERMANENT—CATEGORY A

FUSELAGE



NOTE

- Replace damaged clips with larger clips or brackets.
- Shape approximately as shown. Pick up existing rivets and add seven AN427AD5 rivets on each bottom flange and six AN470DD6 rivets through web of stringer. Repair brackets are 0.081-inch 7075-O clad. Heat-treat to T6 condition after forming. Fillers are 0.071 and 0.081 inch 7075-T6 clad. The extent of the damage is six inches maximum length and one inch maximum height.

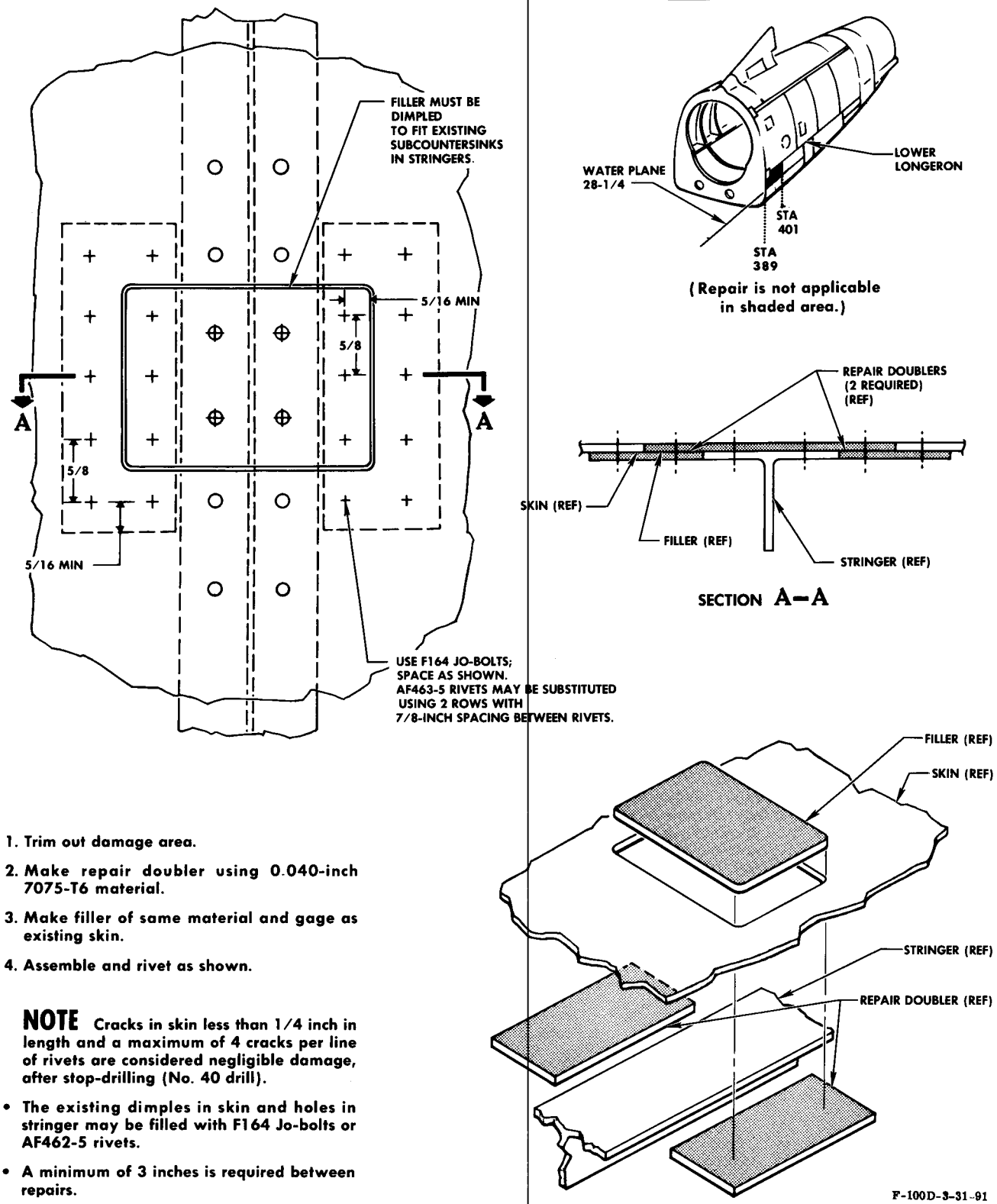


F-100D-3-31-64

Figure 4-27. Aft Fuel Cell Supports—Station 429.5

PERMANENT CATEGORY A

FUSELAGE



1. Trim out damage area.
2. Make repair doubler using 0.040-inch 7075-T6 material.
3. Make filler of same material and gage as existing skin.
4. Assemble and rivet as shown.

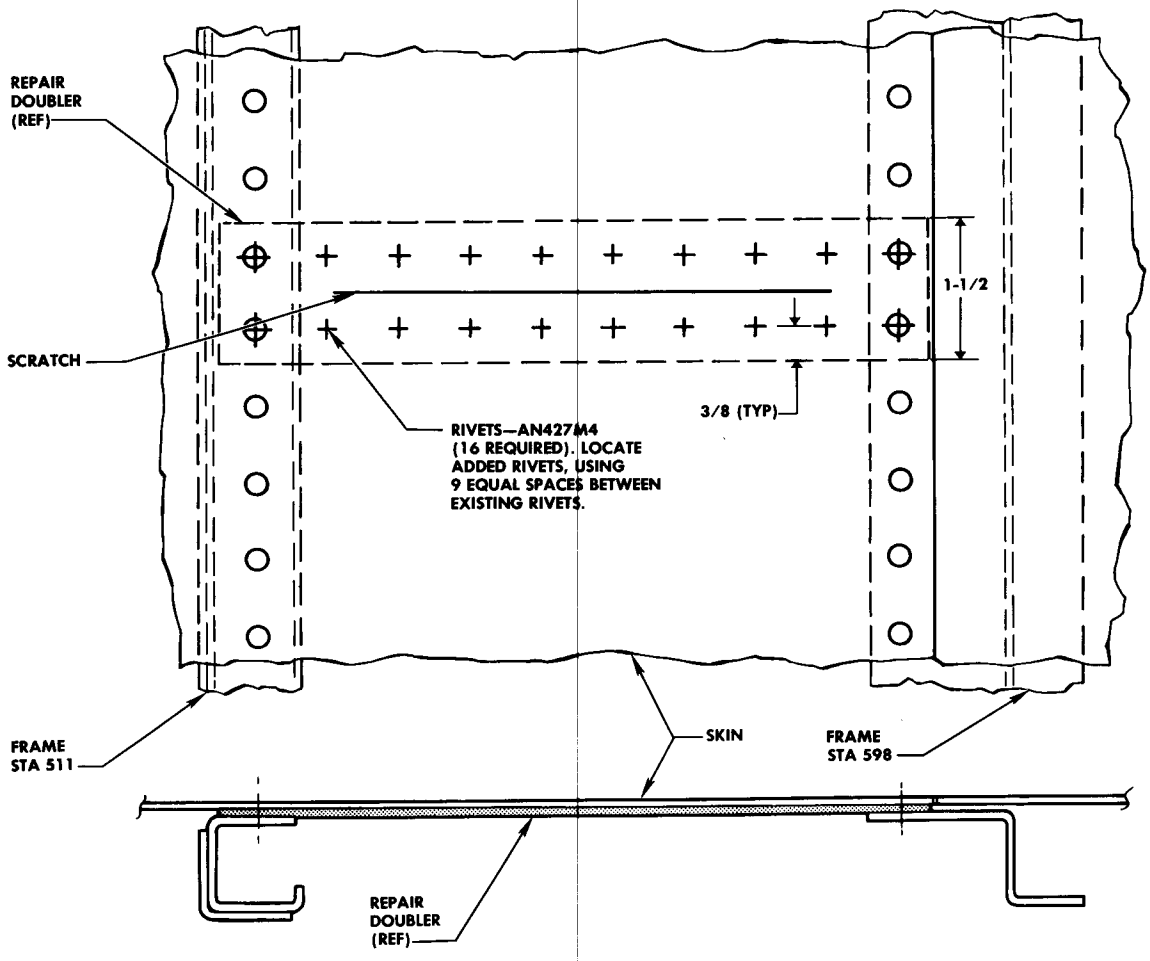
NOTE Cracks in skin less than 1/4 inch in length and a maximum of 4 cracks per line of rivets are considered negligible damage, after stop-drilling (No. 40 drill).

- The existing dimples in skin and holes in stringer may be filled with F164 Jo-bolts or AF462-5 rivets.
- A minimum of 3 inches is required between repairs.

Figure 4-27A. Cracks in Lower Fuselage Skin—Station 389 to 429

PERMANENT CATEGORY A

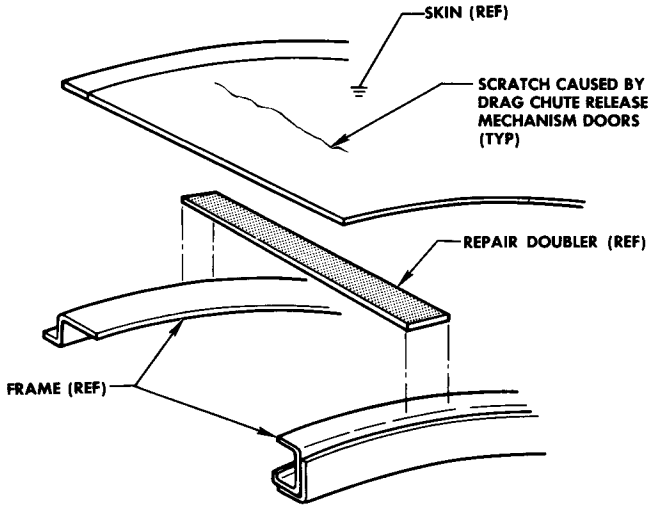
FUSELAGE



1. Make doubler, using 0.020-inch corrosion-resistant steel (Specification MIL-S-6721).
2. Remove existing rivets to install doubler.
3. Install doubler and rivet as shown.

NOTE For temporary repair, the doubler may be installed externally.

- When doubler is installed externally, use CR-563-4-2 rivets for added rivets and CR-562-4-2 rivets for replacing existing rivets.
- For internal doubler installation, NAS508M4 rivets may be used as an alternate for AN427-M4 rivets.



F-100D-3-31-90

Figure 4-27B. Skin Scratches at Drag Chute Release Mechanism Doors

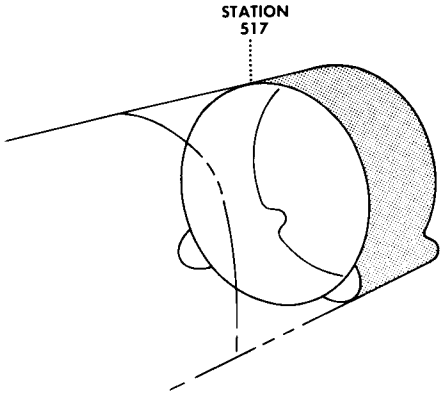
PERMANENT CATEGORY A

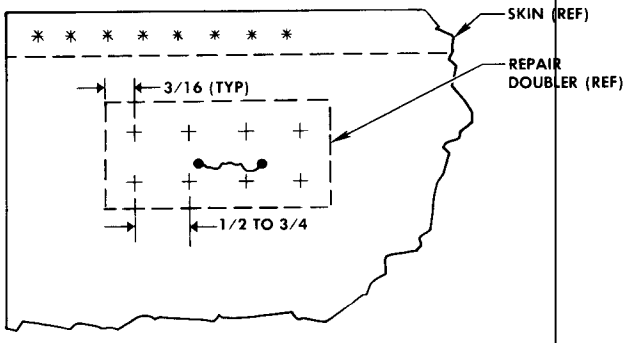
FUSELAGE

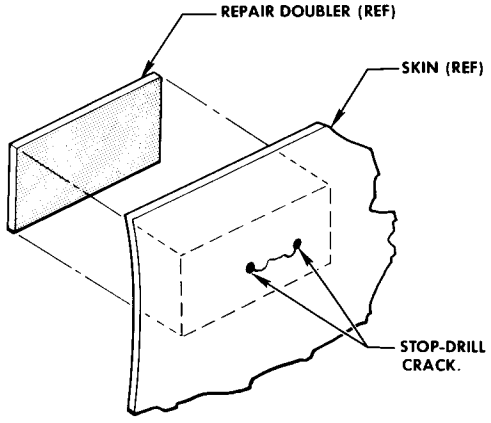
1. Stop-drill crack using a No. 40 drill.
2. Make repair doubler using 0.025-inch AMS-4900CR or AMS-4901 titanium.
3. Install repair doubler using AN435M-3 rivets.
4. Add stiffener in crack area as shown.

NOTE

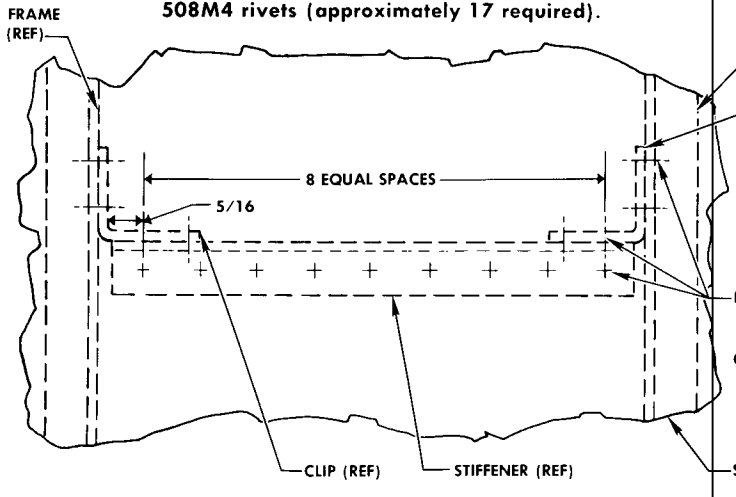
- CR563-4 Cherry rivets may be substituted for AN435M-3 rivets.
- Stiffener should be installed near, but not touching, doubler.

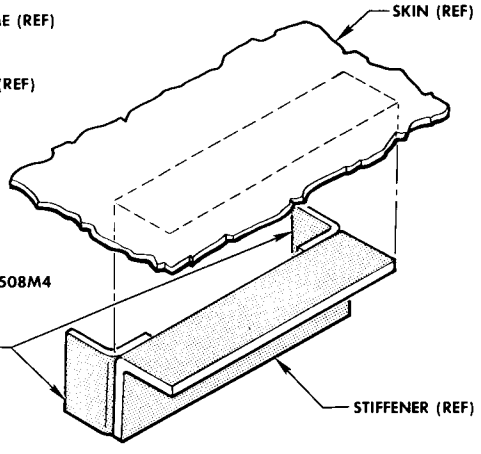






1. Make stiffener, using 0.032-inch corrosion-resistant steel (Specification MIL-S-6721) that is 3/4 x 3/4 inch and long enough to fit between frames. Make two clips, using 0.032-inch corrosion-resistant steel (Specification MIL-S-6721) which is 3/4 x 2-5/8 inches.
2. Install stiffener in crack area using NAS-508M4 rivets (approximately 17 required).





F-100D-3-31-49B

Figure 4-28. Aspirator Skin Cracks

PERMANENT CATEGORY A

FUSELAGE

NOTE Stop-drill cracks using 1/8-inch drill.

- To add a typical skin stiffener, use 0.032-inch corrosion-resistant steel (Specification MIL-S-6721) that is 3/4 x 3/4 inch and as long as required. Two clips are required. Make clips of 0.032-inch corrosion-resistant steel, 3/4 x 2-5/8 inch. Attach clips and stiffener, using NAS508M4, NAS508M5, or NAS508M6 rivets (17 required).
- CR-563-4, -5, or -6 rivets may be used in place of NAS508M4, NAS508M5, or NAS-508M6 rivets.
- CR-562-4 rivets may be used in place of NAS508M4 rivets.

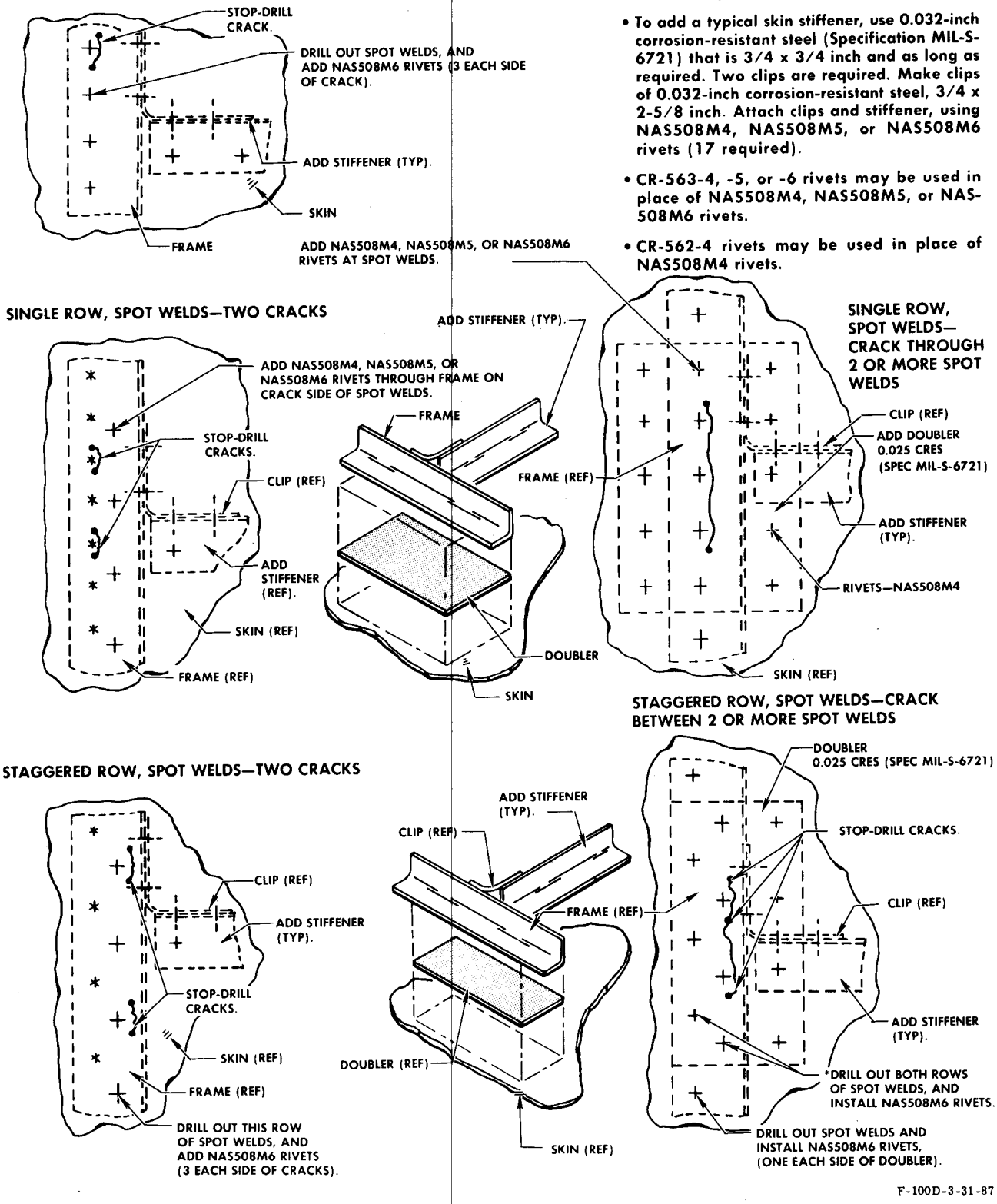
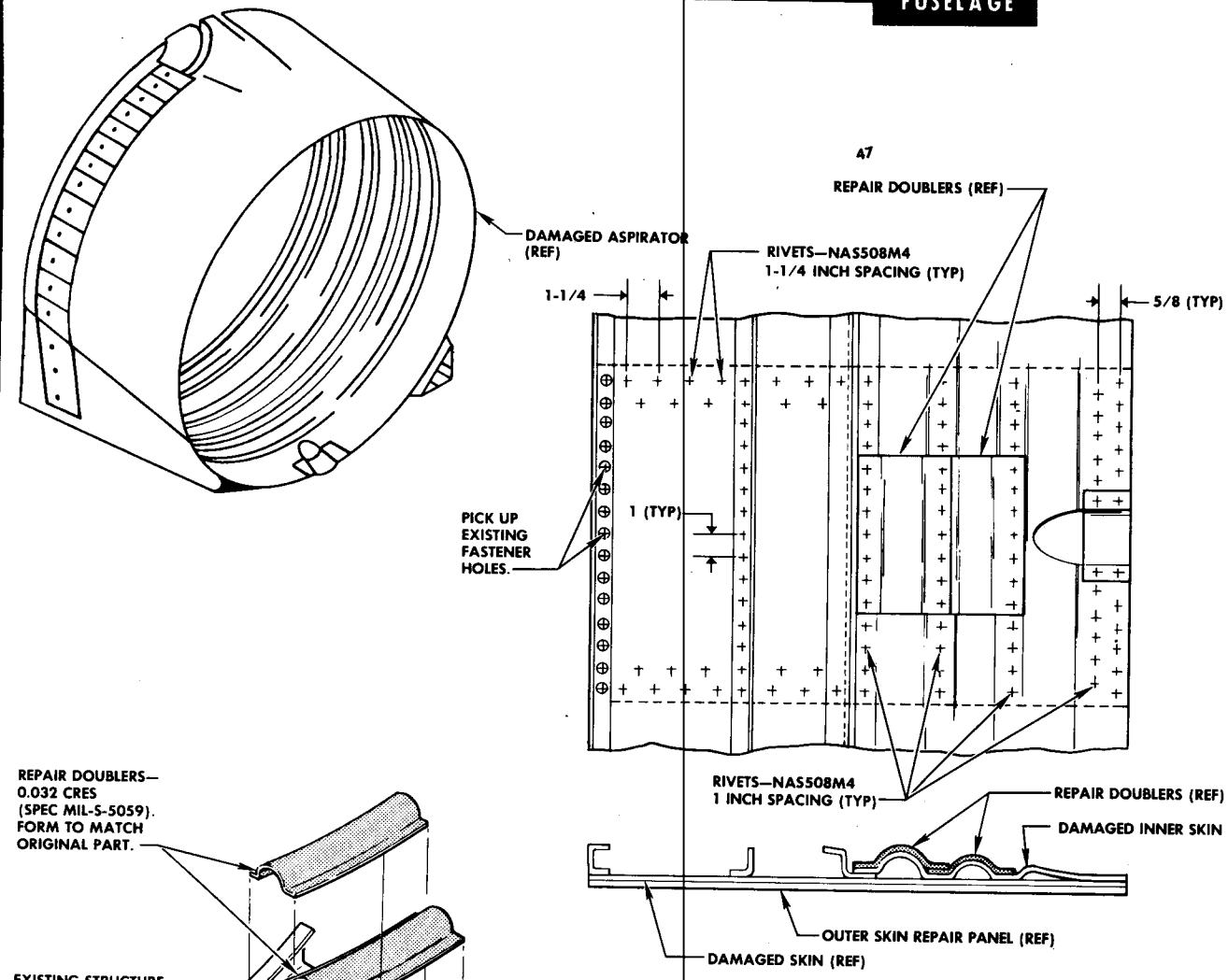


Figure 4-28A. Aspirator Crack Repair at Rivets or Welds

PERMANENT —CATEGORY A

FUSELAGE



1. Form damaged skins and stiffening beads back to original shape.
2. Stop-drill all cracks, and smooth up damaged area. (The damage need not be completely removed.)
3. If stiffening beads are collapsed or cracked, repair doublers must be used. Form repair doublers to match original part.
4. Make repair skin, and fasten in position, using the size and type of fasteners and spacing called out.

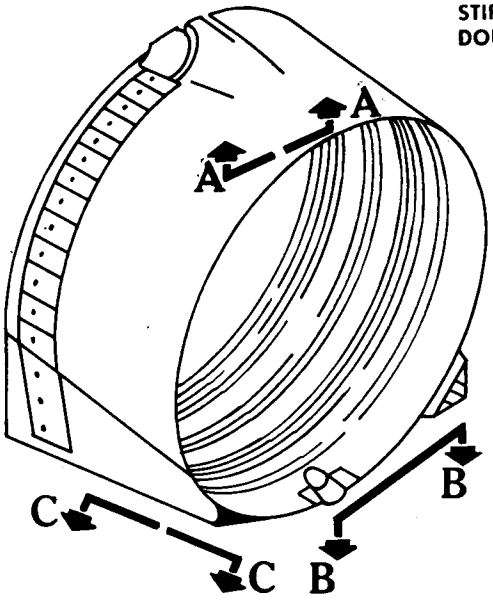
NOTE Refer to "Aspirator Bottom Skin Repair."

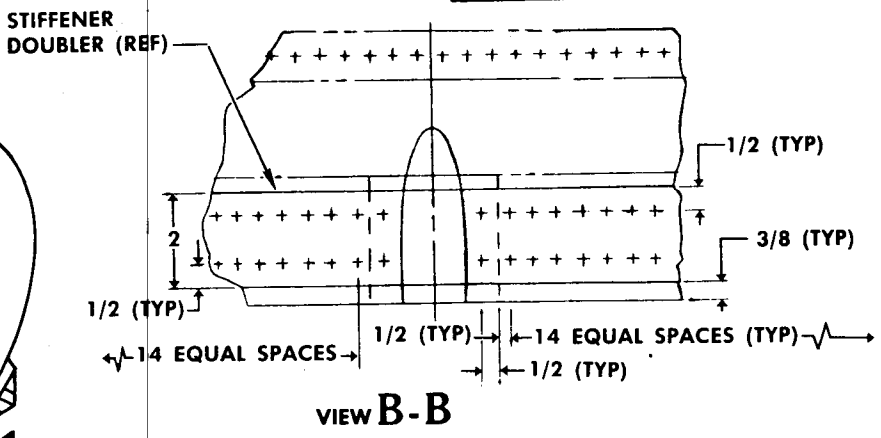
F-100D-3-31-96A

Figure 4-28B. Aspirator Bottom Skin Repair

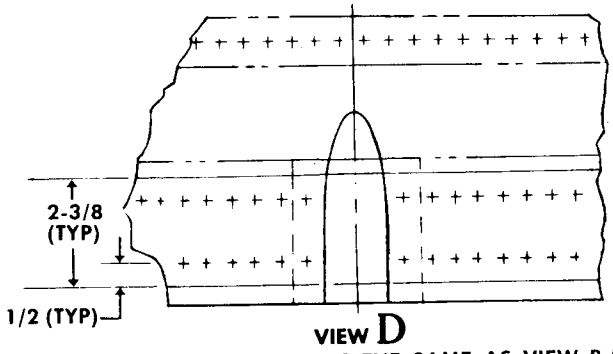
PERMANENT—CATEGORY A

FUSELAGE





VIEW B-B



VIEW D

ALL OTHER DIMENSIONS THE SAME AS VIEW B-B

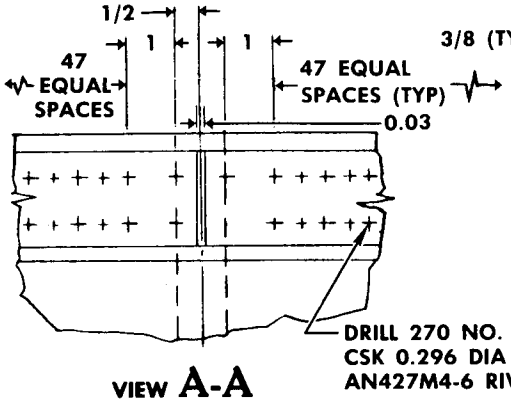
1 Fabricate and install aspirator repair as shown on page 4-62G, figure 4-28E.

2 Shape 0.032-inch CRES skin doubler to inside contour of aspirator skin and attach as shown.

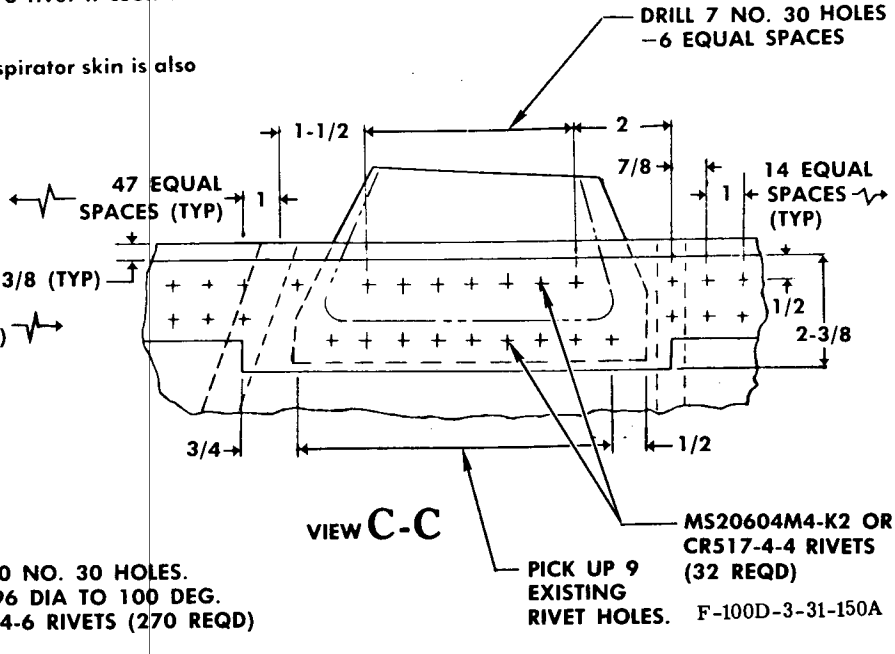
3 Rivets must be flush with outside skin.

NOTE Stiffening is required to maintain 1/4-inch clearance between afterburner fairing segments and aspirator skin.

- Substitute NAS508M4-6 for AN427F4-6 rivet if countersinking is not possible.
- See view D for revised dimensions if aspirator skin is also cracking.



VIEW A-A



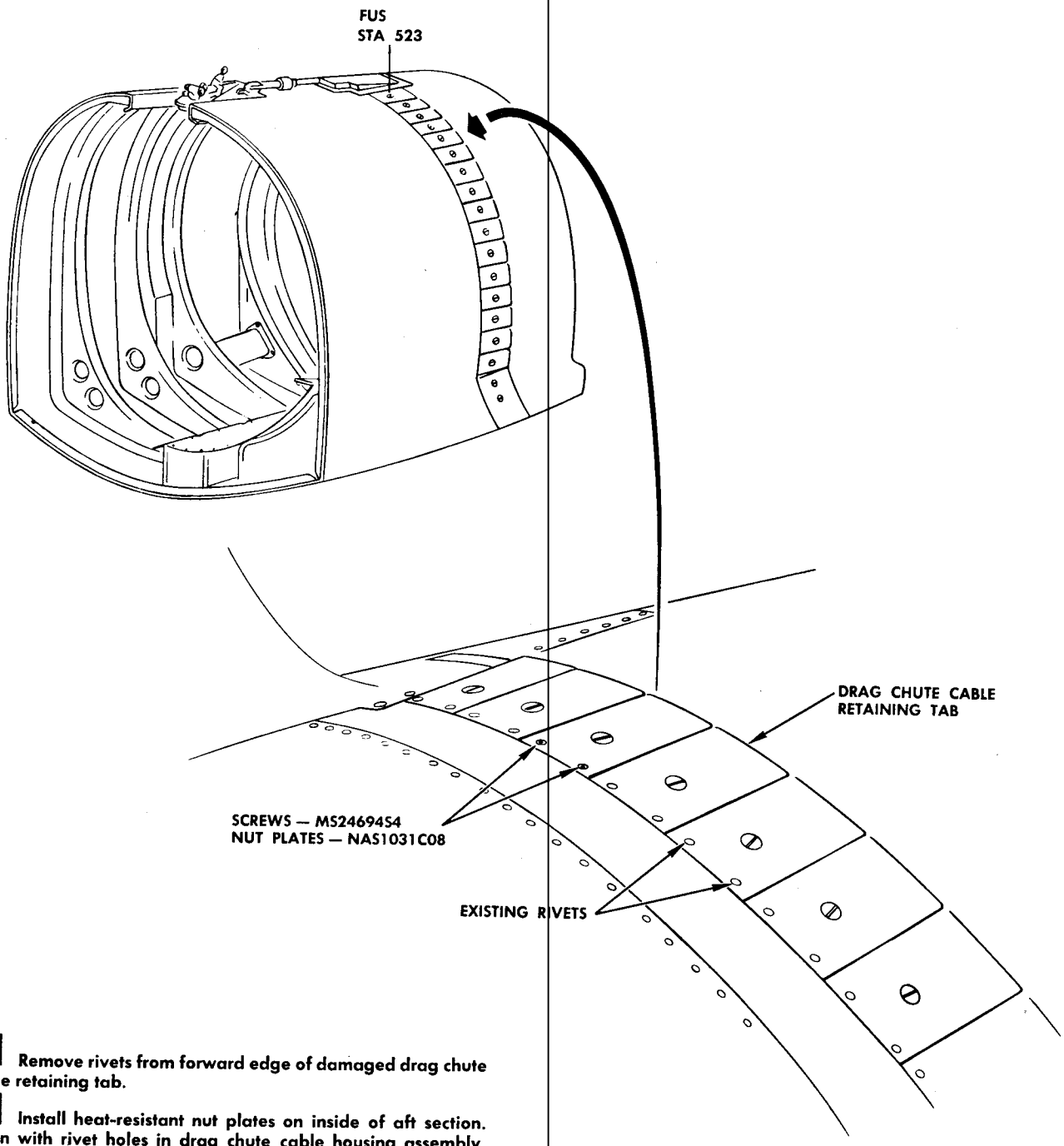
VIEW C-C

F-100D-3-31-150A

Figure 4-28C. Aspirator Skin Stiffening

PERMANENT—CATEGORY A

FUSELAGE



1. Remove rivets from forward edge of damaged drag chute cable retaining tab.
2. Install heat-resistant nut plates on inside of aft section. Align with rivet holes in drag chute cable housing assembly.
3. Dimple retaining tab to fit countersunk screws.
4. Install retaining tabs and torque attaching screws.

Figure 4-28D. Repair and Replacement of Drag Chute Retainer Tabs

MATERIAL	1 IN. X 3/8 IN. BAR STOCK	1-1/4 IN. HEX STOCK	1/2 IN. ROUND STOCK
2 PIECES	61 IN. BAR STOCK		
2 PIECES	37-5/8 IN. BAR STOCK		
4 PIECES	15-1/4 IN. BAR STOCK		
4 PIECES	11 IN. BAR STOCK		
6 PIECES	6-1/2 IN. BAR STOCK		
2 PIECES	8-1/2 IN. HEX STOCK		
4 PIECES	4 IN. RD STOCK		
2 EACH (RH)	3/4 IN. X 3 IN. BOLT		
2 EACH (LH)	3/4 IN. X 3 IN. BOLT		

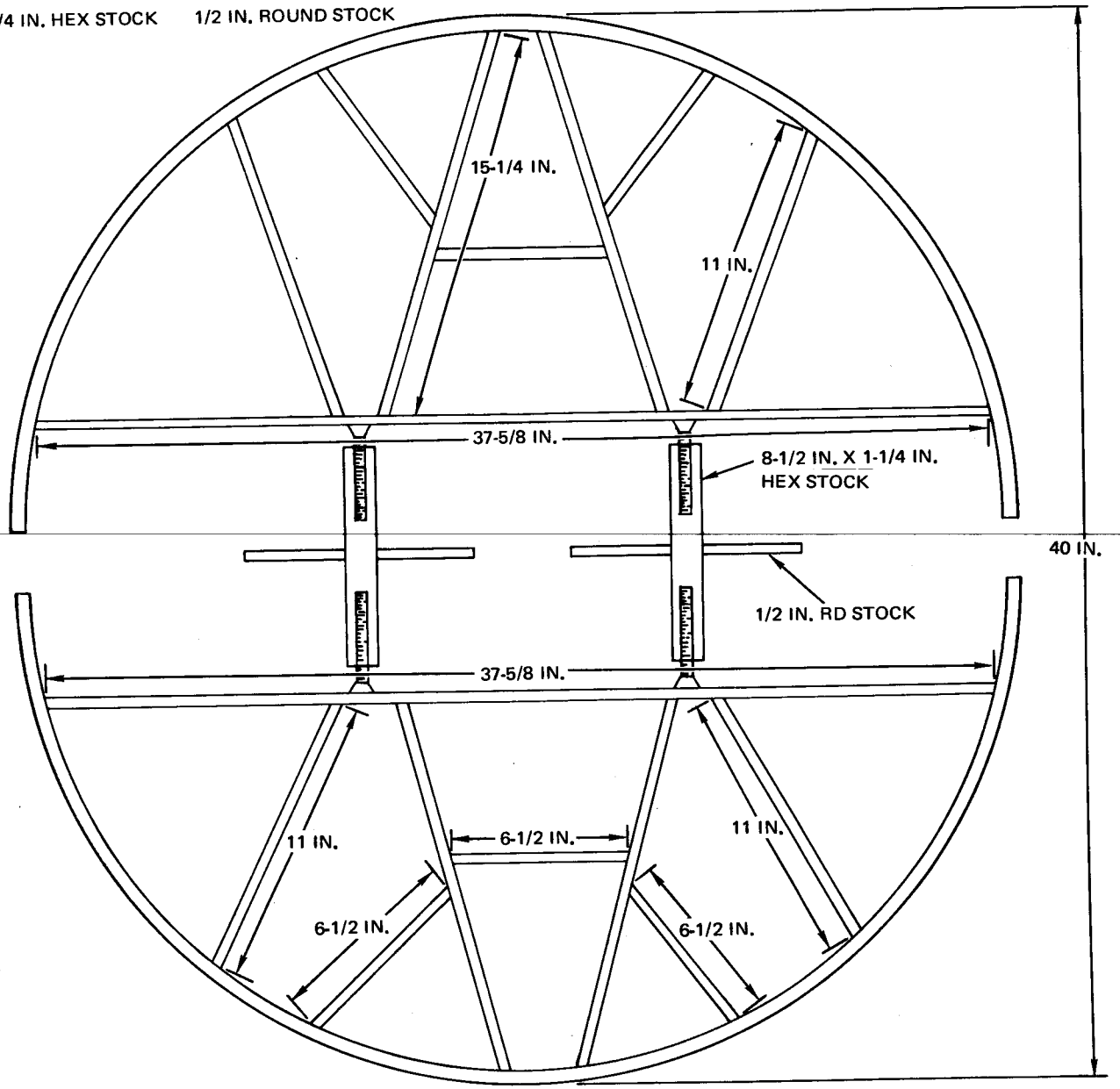
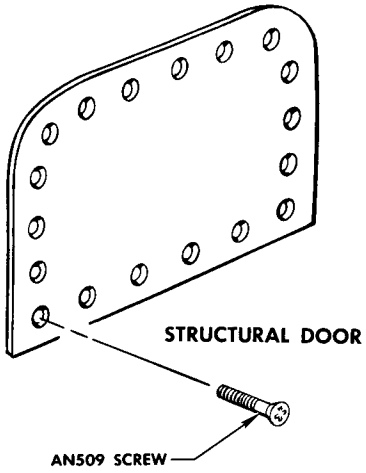
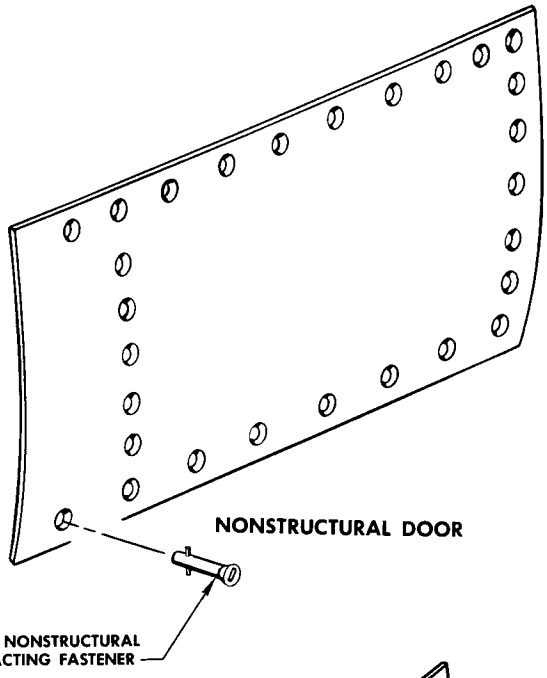


Figure 4-28E. Aspirator Repair Ring

DOORS AND COVERS

FUSELAGE

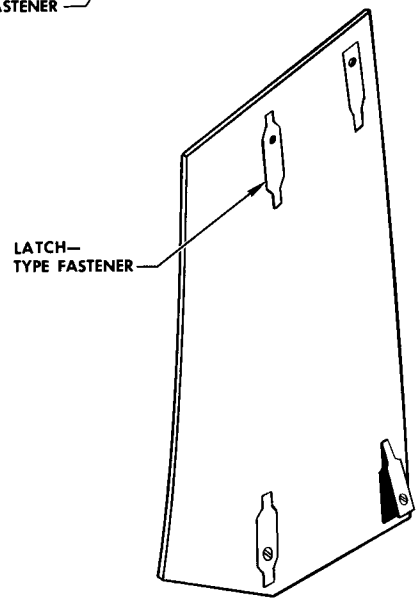




NOTE The doors on the F-100 Airplane consist of two types—structural and nonstructural. The structural doors differ from the nonstructural doors in that the structural doors are part of the primary structure of the airplane and the loads are carried through the door itself. Structural doors are fastened in place with AN. screws or bolts through close-tolerance holes. Shear repairs must be made on all doors in shear areas. Tension repairs are used in all tension areas. The amount of fasteners used is calculated from the rivet tables.

Nonstructural doors do not carry a structural load. The loads are carried by the structure around them. The doors are usually fastened in place with quick-acting fasteners—either the screw-type nonstructural fastener or the latch type.

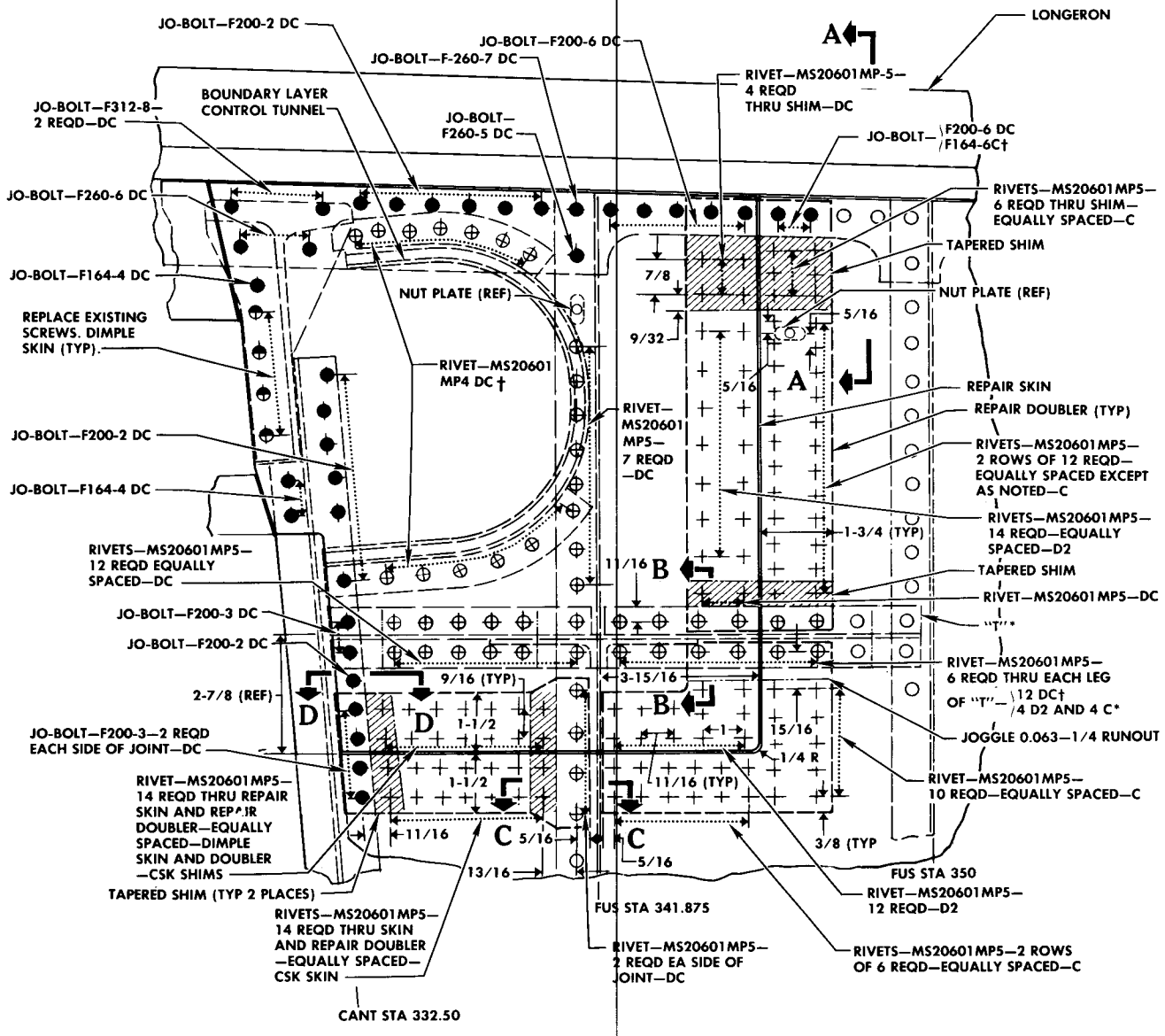
Repairs for both structural and nonstructural doors can be found in the alphabetical index.



F-100D-3-31-82A

Figure 4-29. Fuselage Doors and Covers

PERMANENT—CATEGORY A **FUSELAGE**



FASTENER CODING
◆ JO-BOLT IN EXISTING HOLE
⊕ SCREW IN EXISTING HOLE
⊕ RIVET IN EXISTING HOLE
+ RIVET ADDED
FASTENER INSTALLATION CODING
DC — DIMPLE SKIN—COUNTERSINK SUBSTRUCTURE
C — COUNTERSINK SKIN
D2 — DIMPLE SKIN AND SUBSTRUCTURE

F-100D-3-31-135A

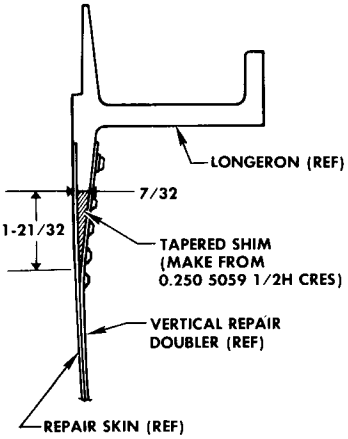
*Use a one-piece repair doubler on airplanes that do not have the "T" as shown (code 1 airplanes). Use same number of rivets shown

†Required on code 2 airplanes only

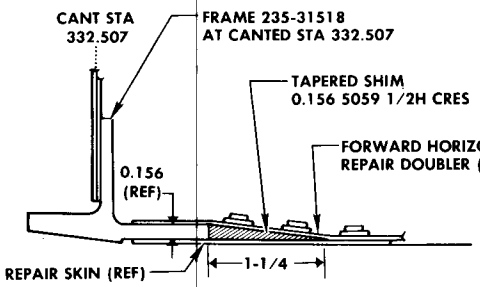
Figure 4-30. Skin Repair—Inboard of Flap at Fuselage Station 332.5 (Sheet 1 of 2)

PERMANENT—CATEGORY A

FUSELAGE

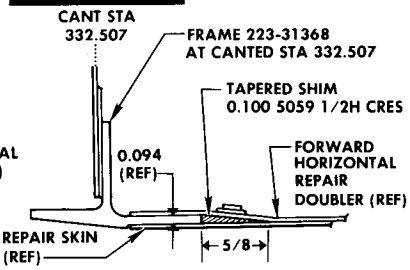


SECTION A - A



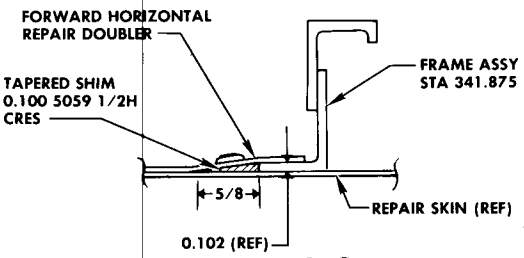
[1] Airplanes on which 235-31485 panel is installed

SECTION D-D

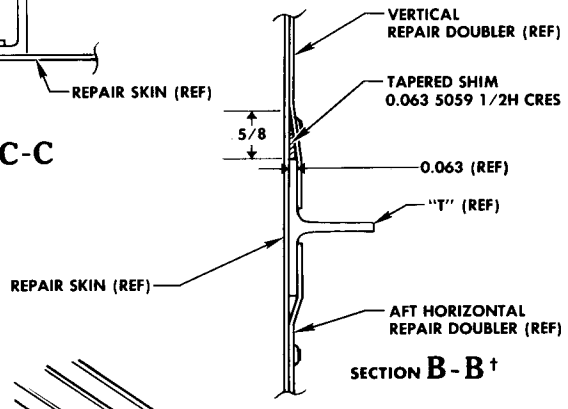


[2] Airplanes on which 223-31387 panel is installed

SECTION D-D



SECTION C-C



SECTION B - B†

[2] F-100D-1 through F-100D-85 airplanes

[1] F-100D-90 and later blocks of airplanes

F-100F-20 airplanes

1 Trim out damage midway between existing fasteners.

2 Smooth all edges and corners after trimming.

3 Make repair skin from 0.040-inch one-half hard corrosion-resistant steel (Specification MIL-S-5059) to match the damage cutout.

4 Make repair doublers from 0.040-inch one-half hard corrosion-resistant steel (Specification MIL-S-5059). Joggle repair doublers as required.

5 Make fillers from one-half hard corrosion-resistant steel (Specification MIL-S-5059) the thickness required.

6 For repair doublers, lay out fastener pattern on existing skin. Pick up existing fastener locations where required.

7 Drill fastener pattern through existing skin, repair doublers, and shims. Countersink existing skin for size and type of fasteners required. Pick up hole for nut plate.

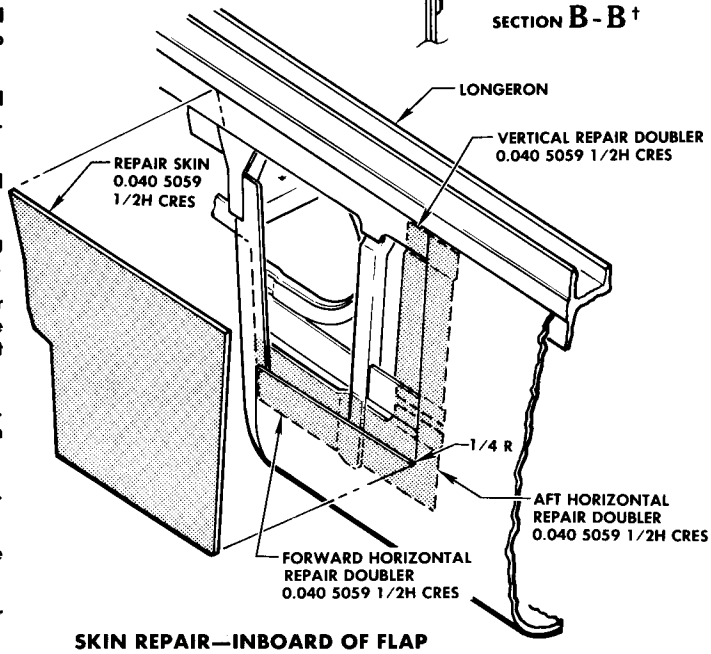
8 Temporarily fasten repair doublers and shims in place. Lay out and drill fastener pattern through repair skin and repair doublers.

9 Remove repair skin and repair doublers and shims. Dimple and countersink as necessary.

10 Replace nut plate on vertical repair doubler before installation.

11 With size and type of fasteners required, fasten repair doublers and shims to existing skin.

12 With size and type of fasteners required, fasten repair skin in place.



SKIN REPAIR—INBOARD OF FLAP

FUSELAGE STATION 332.5 TO 350

F-100D-3-31-140

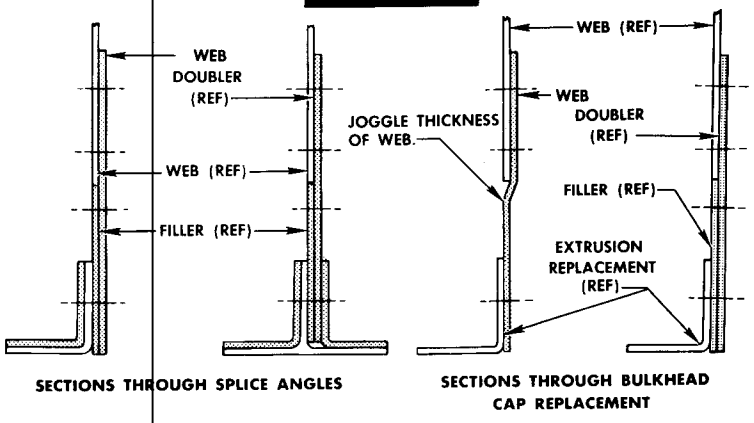
Figure 4-30. Skin Repair—Inboard of Flap at Fuselage Station 332.5 (Sheet 2 of 2)

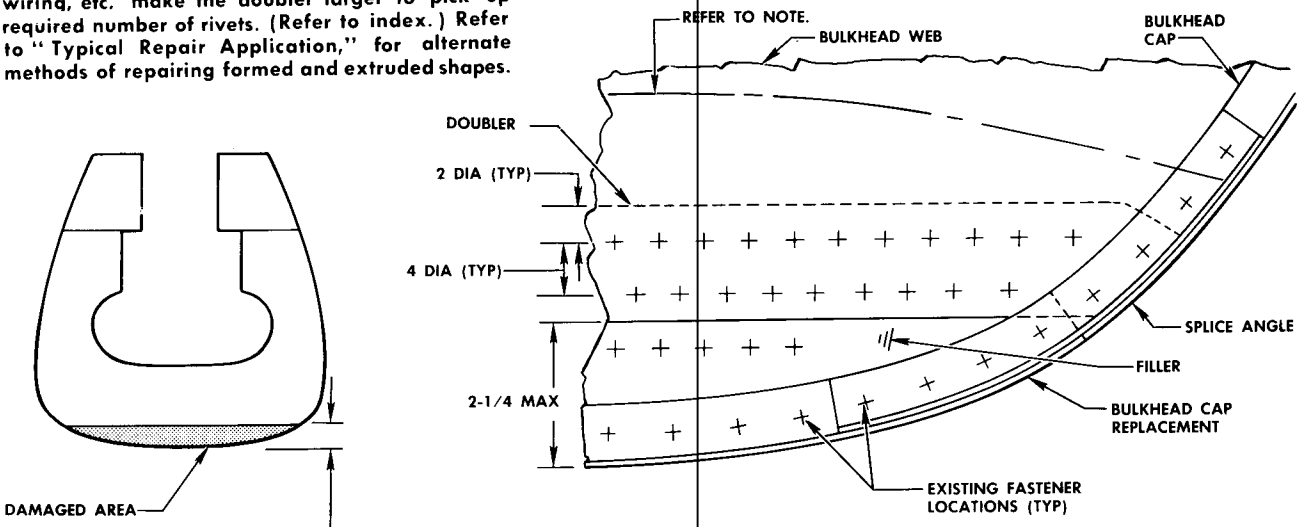
PERMANENT - CATEGORY A

FUSELAGE

1. Trim out damaged areas.
2. Make web doubler of same material, one gage heavier than existing web.
3. Make filler of same material and gage as existing web.
4. Make four splice angles of same material and gage as existing member.
5. Make cap replacement "T" of same material and section as existing members.
6. Assemble and rivet as shown.

NOTE In case of lightening holes or holes for wiring, etc. make the doubler larger to pick up required number of rivets. (Refer to index.) Refer to "Typical Repair Application," for alternate methods of repairing formed and extruded shapes.





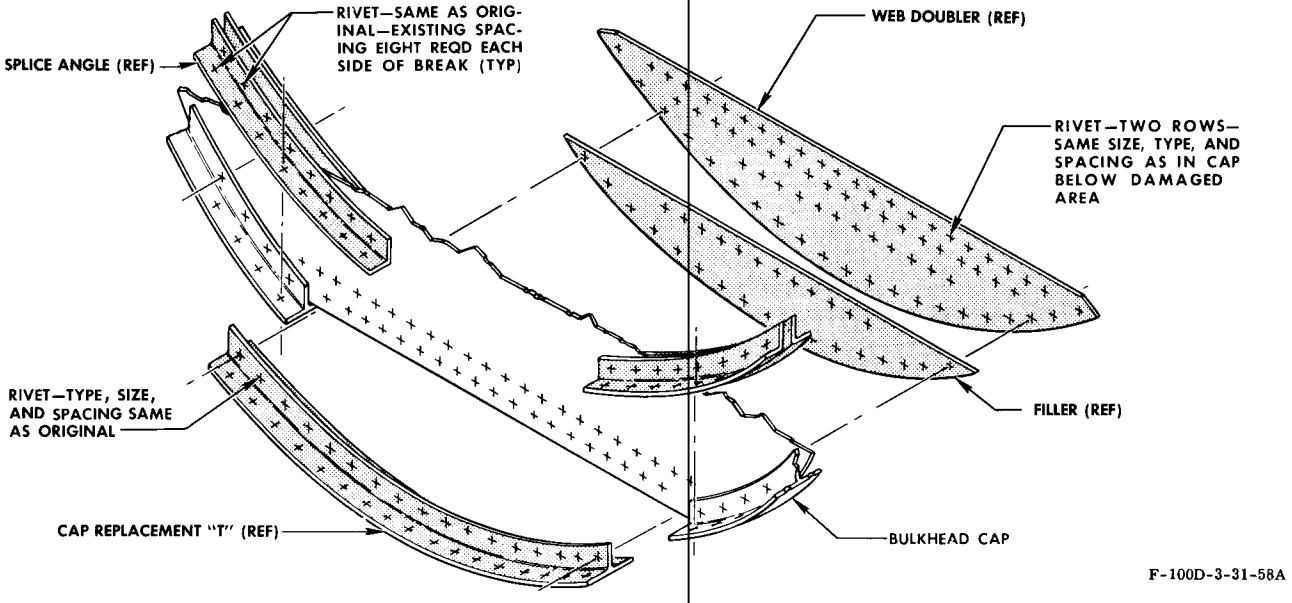
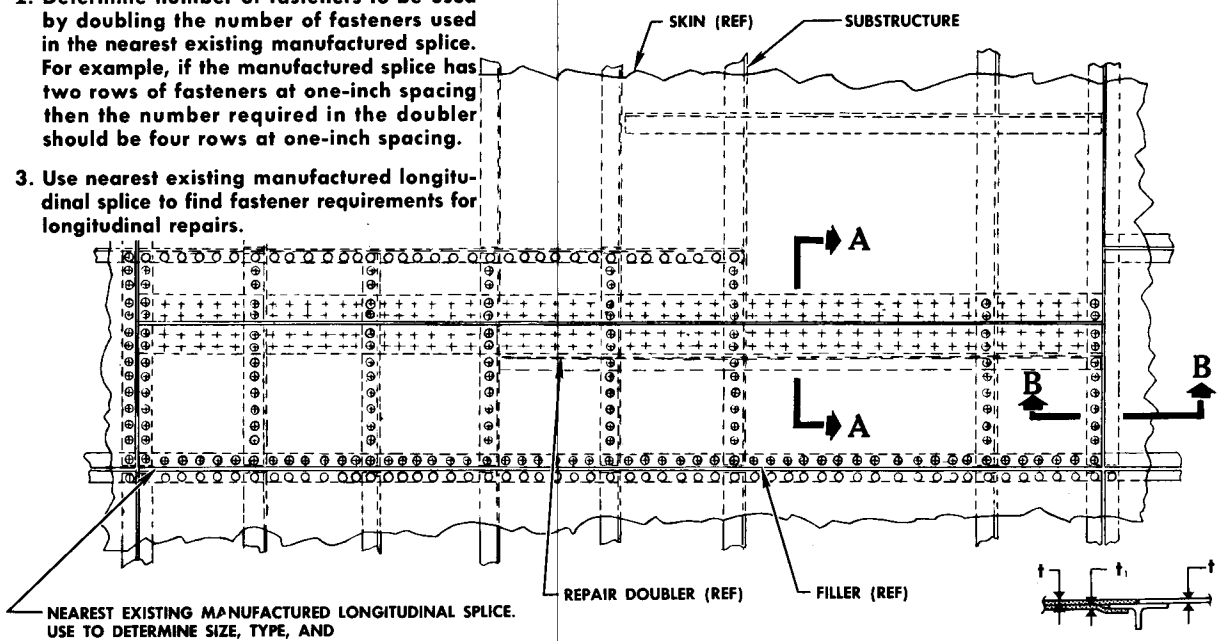


Figure 4-30A. Fuselage Web and Cap

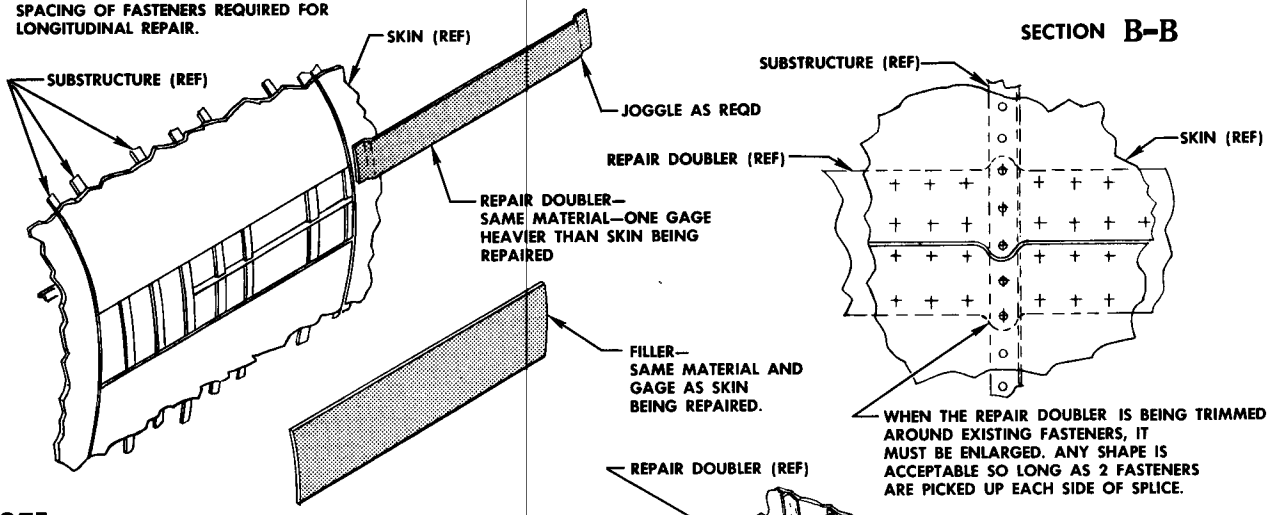
PERMANENT—CATEGORY A

FUSELAGE

1. Trim out damage midway between existing fasteners.
2. Determine number of fasteners to be used by doubling the number of fasteners used in the nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing then the number required in the doubler should be four rows at one-inch spacing.
3. Use nearest existing manufactured longitudinal splice to find fastener requirements for longitudinal repairs.

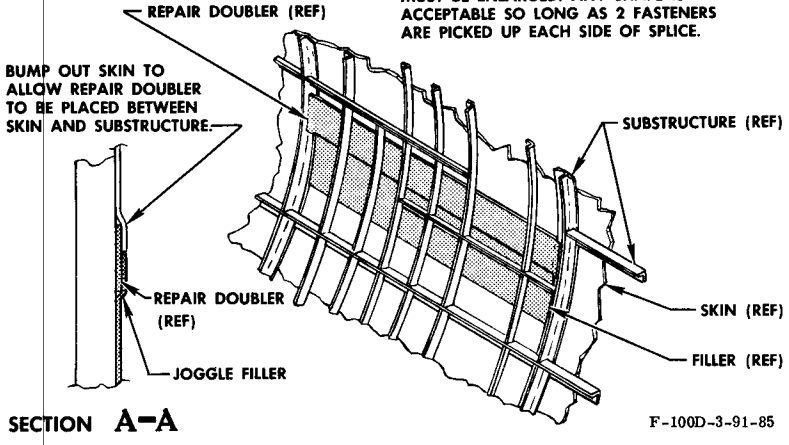


SECTION B-B



NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.



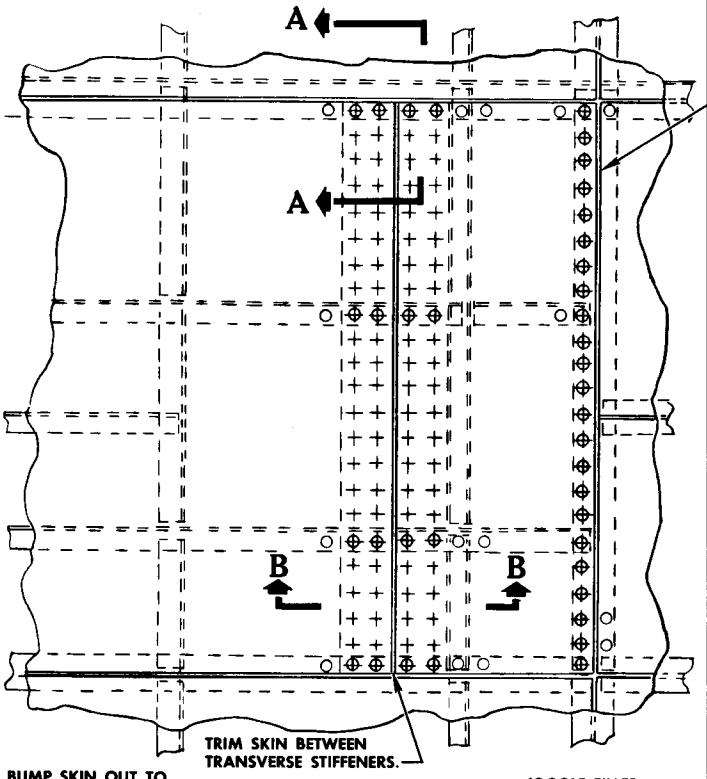
SECTION A-A

F-100D-3-91-85

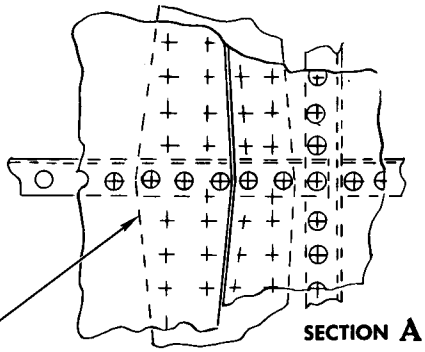
Figure 4-31. Skin Splice—Shear (Sheet 1 of 4)

PERMANENT—CATEGORY A

FUSELAGE



NEAREST EXISTING MANUFACTURED TRANSVERSE SPLICE USE TO DETERMINE SIZE, TYPE, AND SPACING OF FASTENERS TO BE USED IN REPAIR



SECTION A A

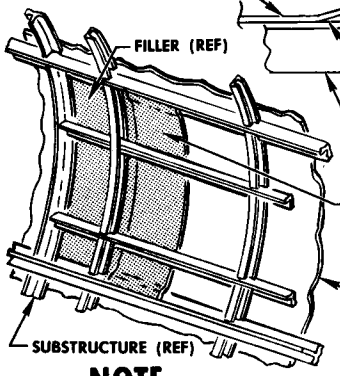
WHEN THE REPAIR DOUBLER IS BEING TRIMMED AROUND EXISTING FASTENERS, IT MUST BE ENLARGED AS SHOWN. ANY SHADE IS ACCEPTABLE AS LONG AS THE DOUBLER IS ENLARGED AND TWO FASTENERS ARE PICKED UP EACH SIDE OF SPLICE (MIN).

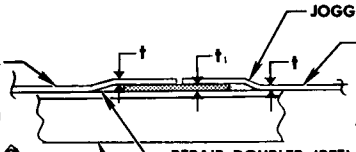
1 Trim out damage midway between existing fasteners.

2 Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the number required for the doubler would be four rows of fasteners at one-inch spacing.

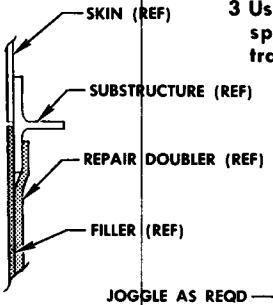
3 Use nearest existing manufactured transverse splice to find fastener requirements for transverse repairs.

BUMP SKIN OUT TO ALLOW DOUBLER TO BE PLACED BETWEEN SKIN AND SUBSTRUCTURE.

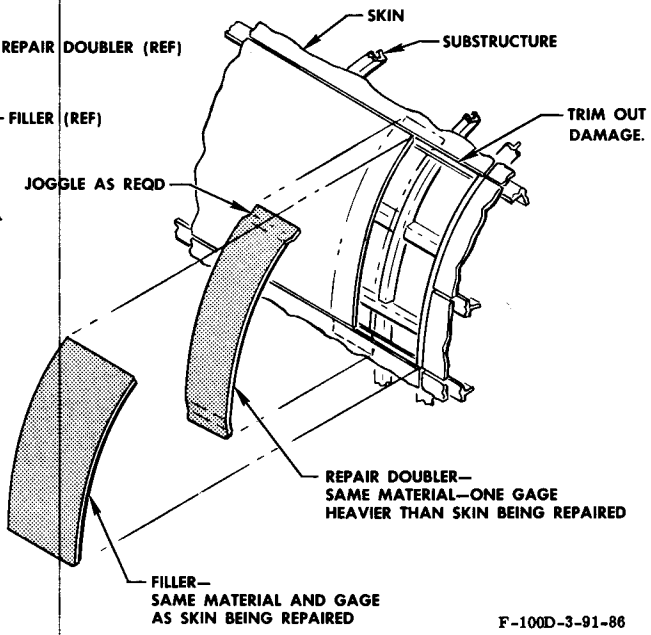




SECTION B—B



SECTION A—A



NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

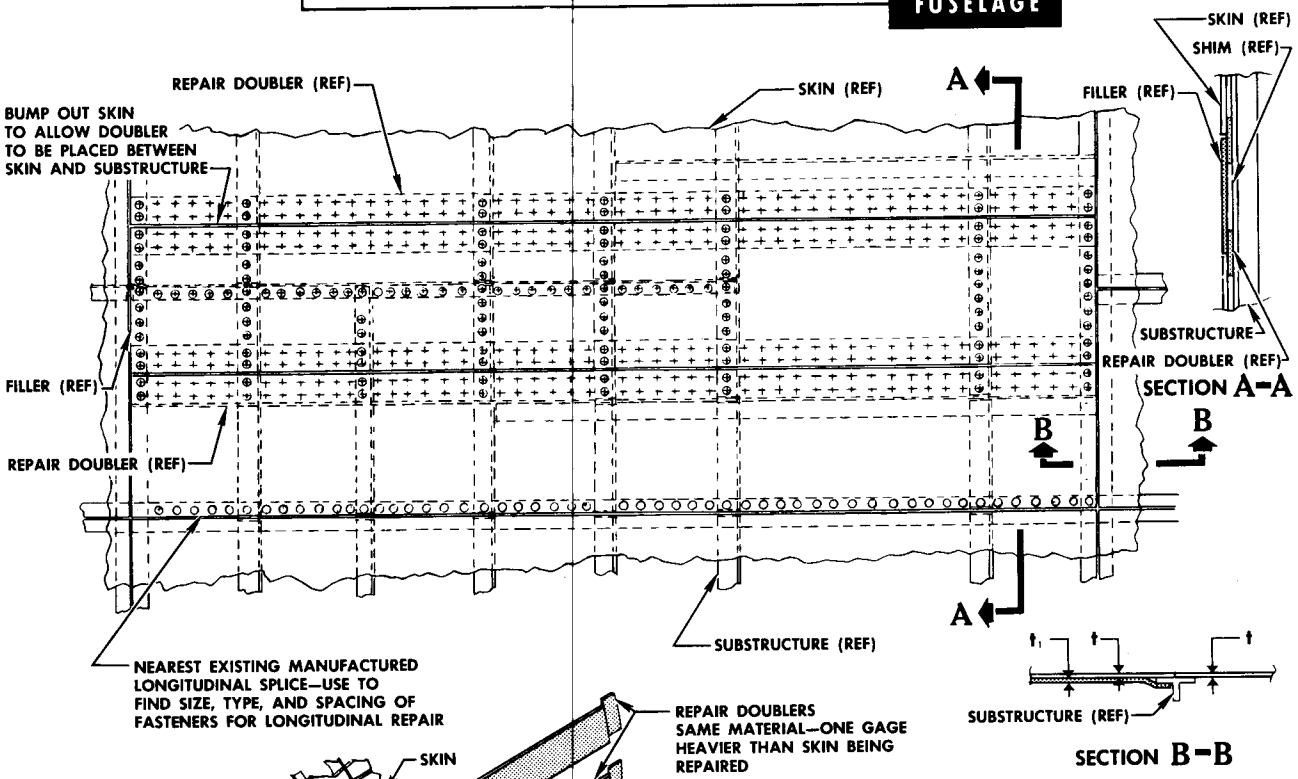
- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

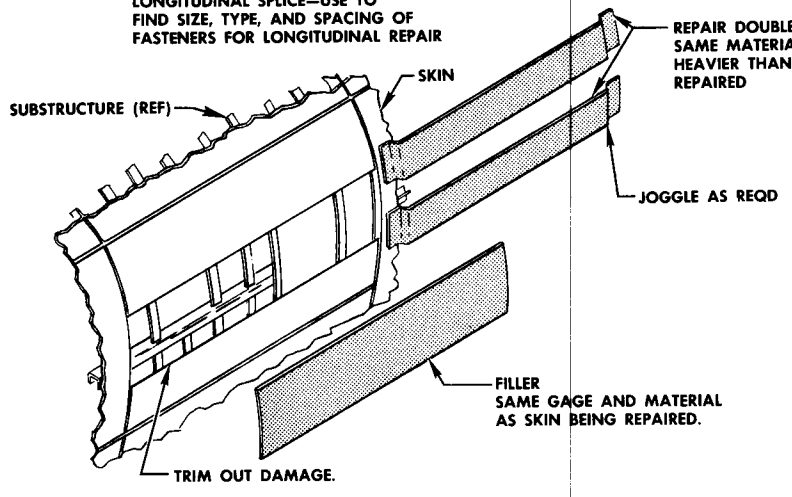
F-100D-3-91-86

Figure 4-31. Skin Splice—Shear (Sheet 2 of 4)

PERMANENT CATEGORY A

FUSELAGE

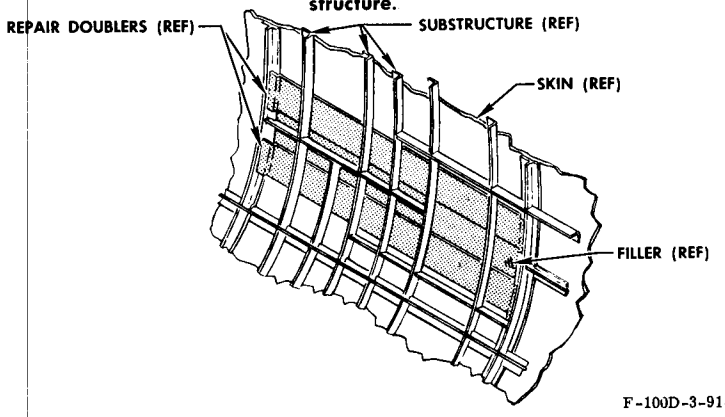




1. Trim out damage midway between existing fasteners.
2. Determine number of fasteners to be used by doubling the number of fasteners used in the nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, then the doubler would have four rows at one-inch spacing.
3. Use nearest existing manufactured splice to find fastener requirements for longitudinal repairs.
4. Place a shim on the substructure between doublers, where filler is away from substructure.

NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

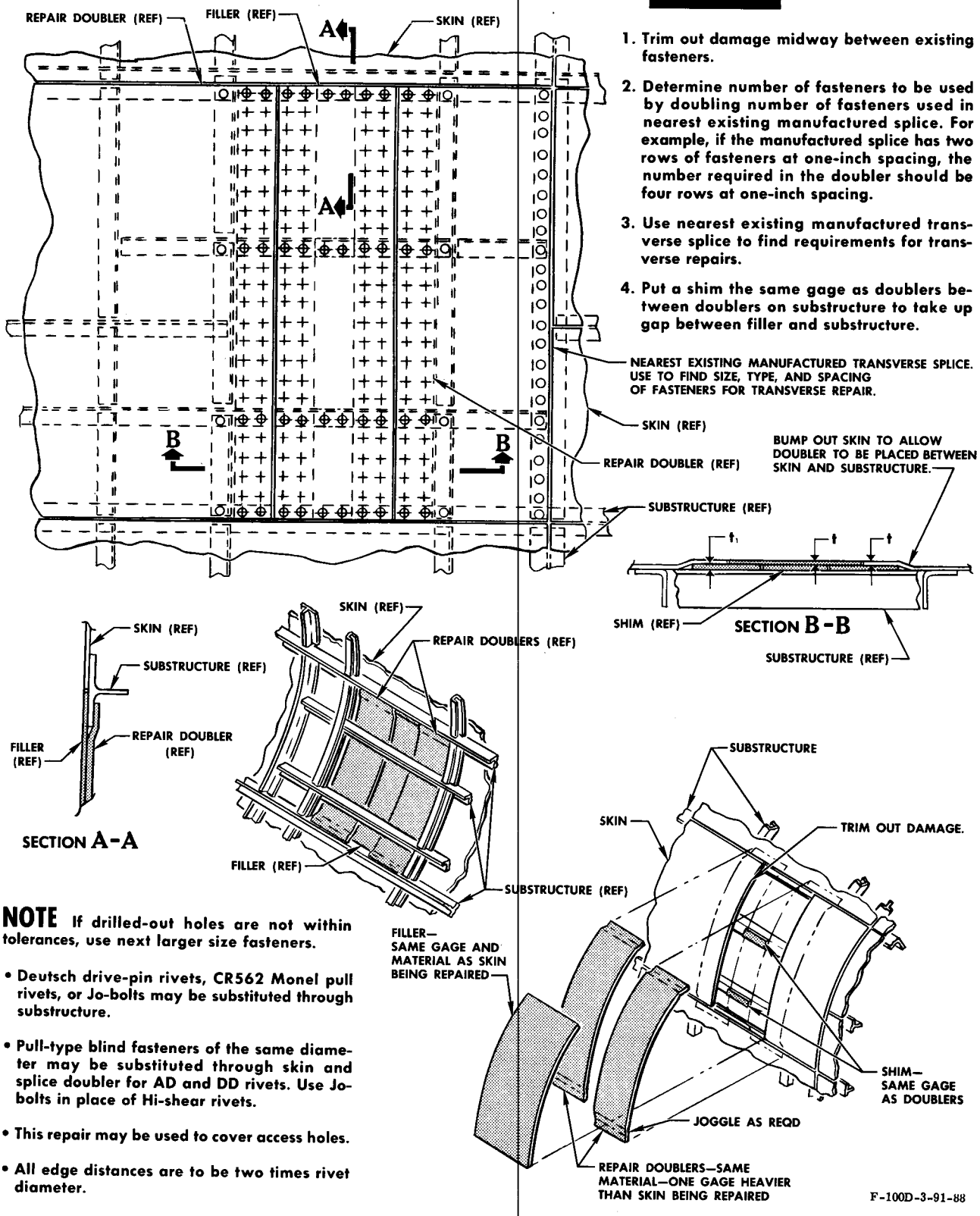


F-100D-3-91-87

Figure 4-31. Skin Splice—Shear (Sheet 3 of 4)

PERMANENT—CATEGORY A

FUSELAGE



1. Trim out damage midway between existing fasteners.
2. Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the number required in the doubler should be four rows at one-inch spacing.
3. Use nearest existing manufactured transverse splice to find requirements for transverse repairs.
4. Put a shim the same gage as doublers between doublers on substructure to take up gap between filler and substructure.

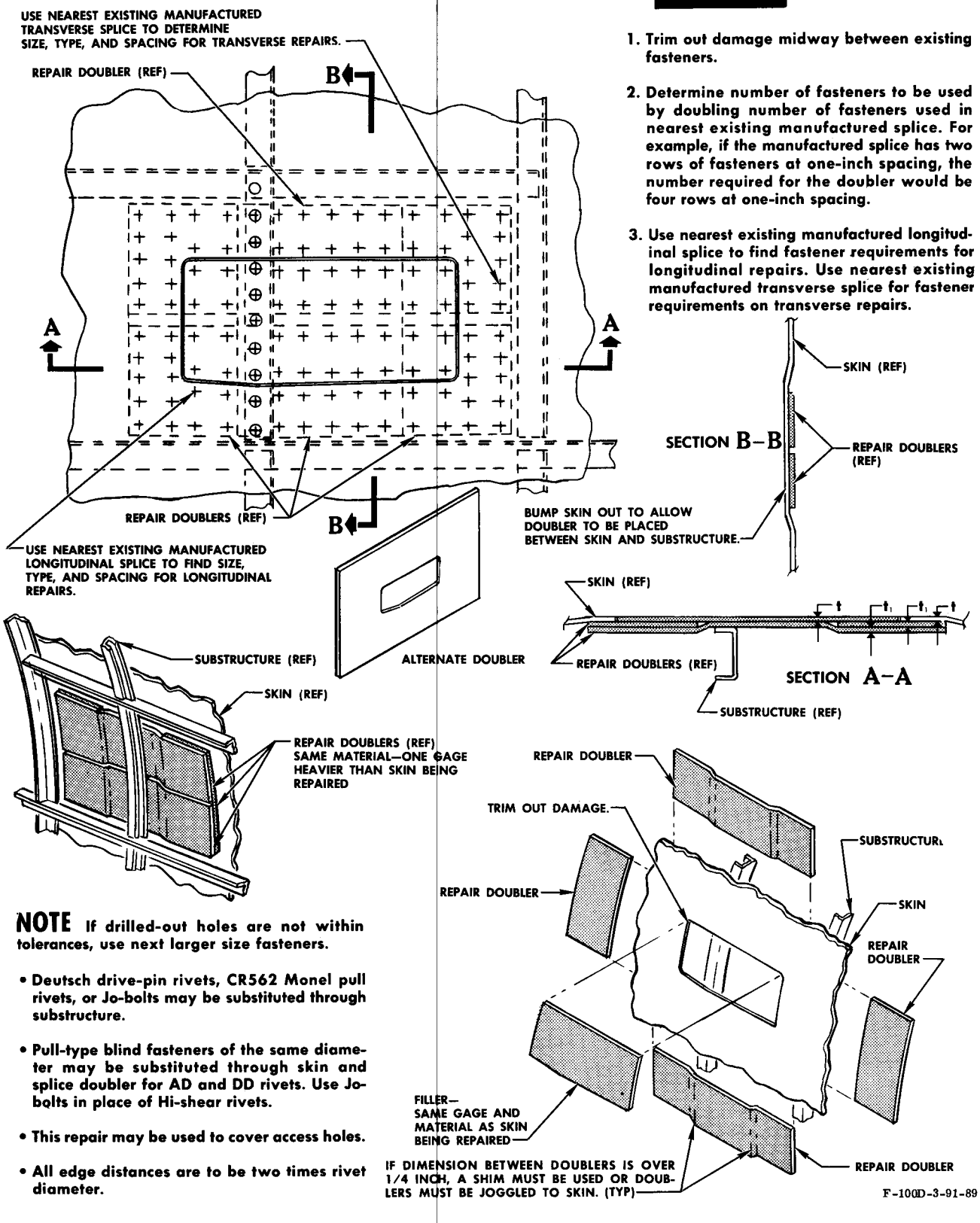
NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

Figure 4-31. Skin Splice—Shear (Sheet 4 of 4)

PERMANENT—CATEGORY A

FUSELAGE



1. Trim out damage midway between existing fasteners.
2. Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the number required for the doubler would be four rows at one-inch spacing.
3. Use nearest existing manufactured longitudinal splice to find fastener requirements for longitudinal repairs. Use nearest existing manufactured transverse splice for fastener requirements on transverse repairs.

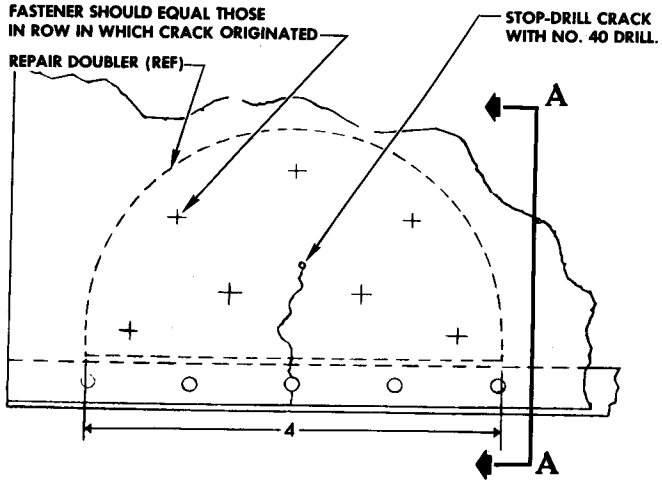
NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

Figure 4-32. Skin Center Repair—Shear

PERMANENT—CATEGORY A

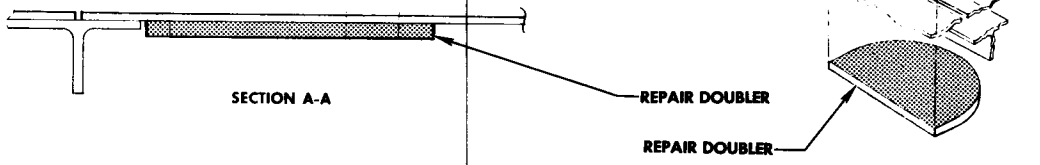
FUSELAGE

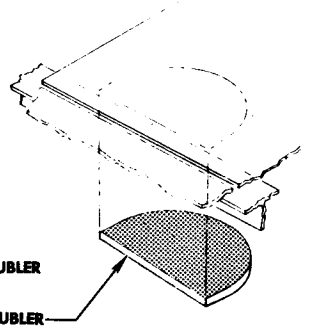


PANEL-EDGE CRACK—3/4-INCH MAXIMUM

NOTE

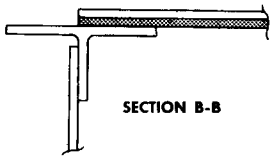
- Repair doubler material (titanium) must be 2 times the thickness of damaged skin. Half-hard CRES material may be used, provided the repair doubler is 1-1/2 times the thickness of the damaged titanium skin.
- For aluminum alloy repairs, use the same alloy repair doubler 2 times heavier or 1/2-hard CRES of equal gage.
- Minimum of 8 inches between cracks.

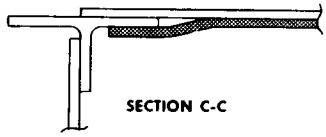


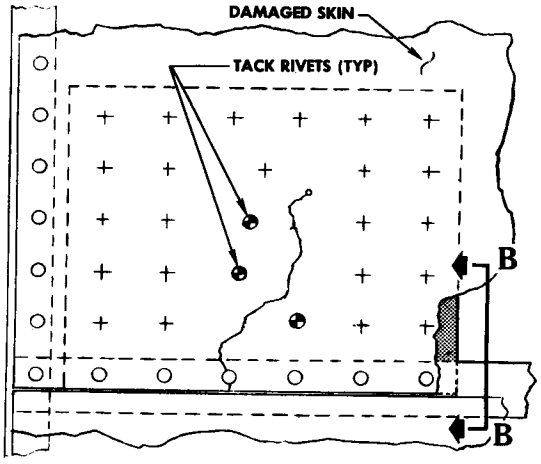


NOTE

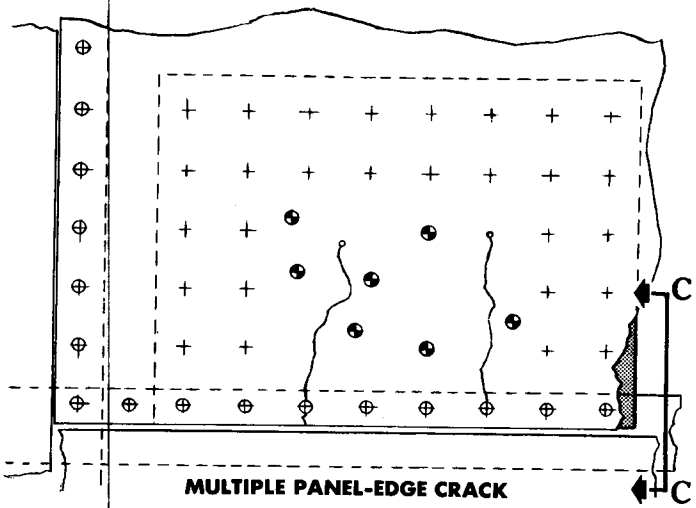
Refer to notes on sheet 2







PANEL-EDGE CRACK OVER 3/4 INCH.



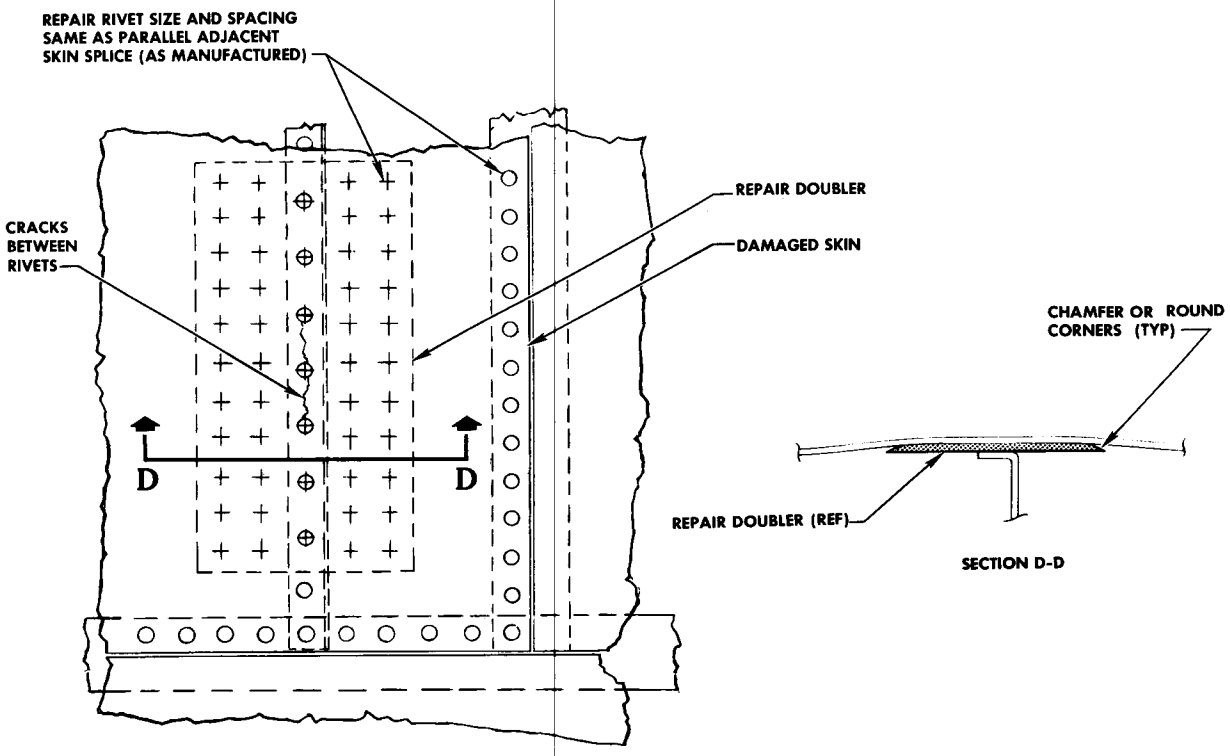
MULTIPLE PANEL-EDGE CRACK

F-100D-3-91-104

Figure 4-32A. Skin Crack Repairs (Sheet 1 of 2)

PERMANENT—CATEGORY A

FUSELAGE



CRACK IN RIVET PATTERN

NOTE

These illustrations are examples of cracked skin panel conditions. When no repair is given in this manual for a specific cracked panel condition, one of these repairs may be selected for any constant-thickness (nonmilled) skin of the fuselage.

The following rules apply for any of these repairs, except where noted otherwise.

A. Determine number of fasteners to be used in repair by doubling the number of fasteners used in adjacent parallel skin splice (as manufactured).

B. Pull-type Monel CR562 or CR563 blind fasteners of the same diameter may be substituted through skin and repair doubler for AD and DD rivets.

C. Size of fasteners in repair will correspond to those in adjacent parallel fastener pattern.

D. Use one gage heavier material for repair doubler when doubler and damaged skin materials are the same.

E. One-half hard CRES may be used for the repair doubler on 2024-T4 and 7075-T6 aluminum alloys. CRES gage for this repair may be half that of the aluminum alloy doubler, but not thinner than 0.030 inch when aluminum alloy fasteners are used.

F. One-half hard CRES may be used as a repair doubler for titanium and titanium alloy. CRES gage must equal that of the damaged titanium.

G. Stop-drill all cracks at termination point with a No. 40 drill.

H. Use tack rivets one size smaller than doubler attaching rivets around point of crack emanation, as illustrated, to eliminate canning of immediate crack area as required.

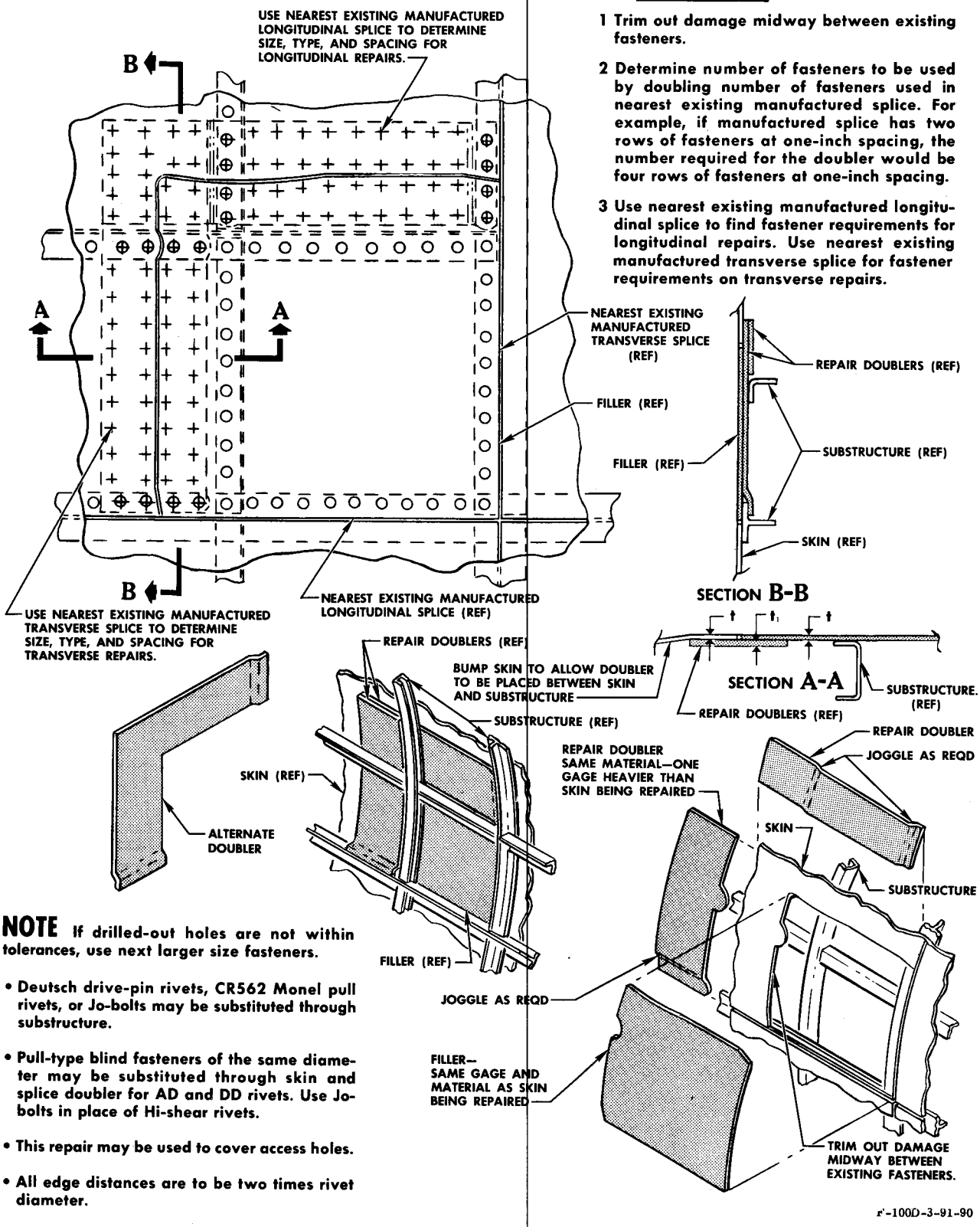
I. Main fastener edge distance 2 times fastener diameter measured from center of hole.

F-100D-3-91-103

Figure 4-32A. Skin Crack Repairs (Sheet 2 of 2)

PERMANENT—CATEGORY A

FUSELAGE



1 Trim out damage midway between existing fasteners.

2 Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if manufactured splice has two rows of fasteners at one-inch spacing, the number required for the doubler would be four rows of fasteners at one-inch spacing.

3 Use nearest existing manufactured longitudinal splice to find fastener requirements for longitudinal repairs. Use nearest existing manufactured transverse splice for fastener requirements on transverse repairs.

NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

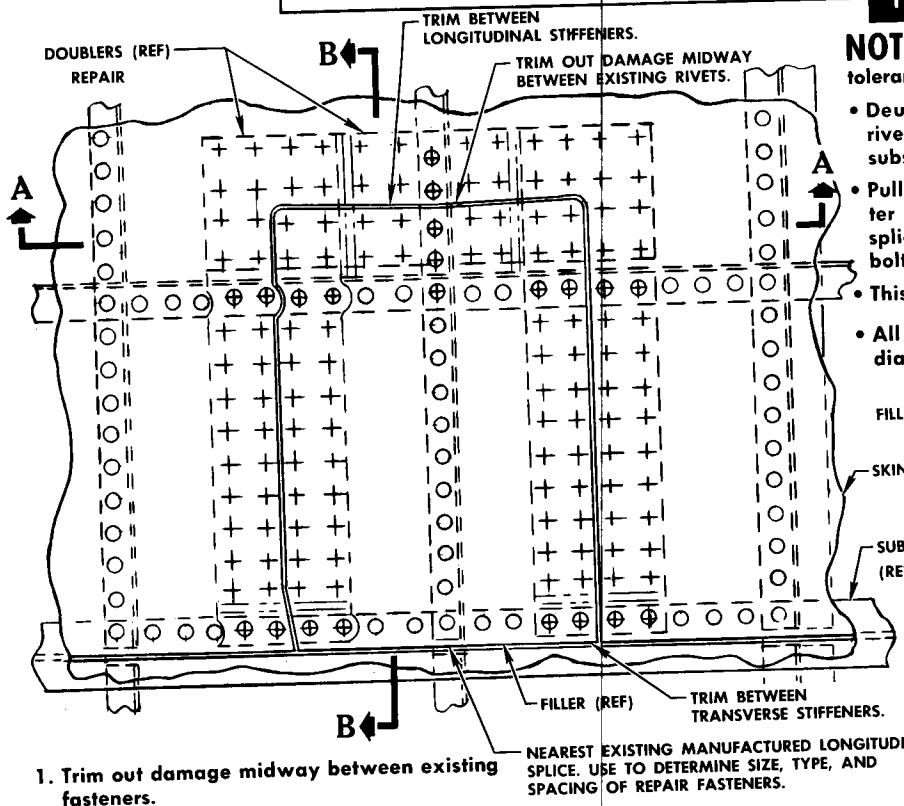
- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

F-100D-3-91-90

Figure 4-33. Skin End Repair—Shear (Sheet 1 of 3)

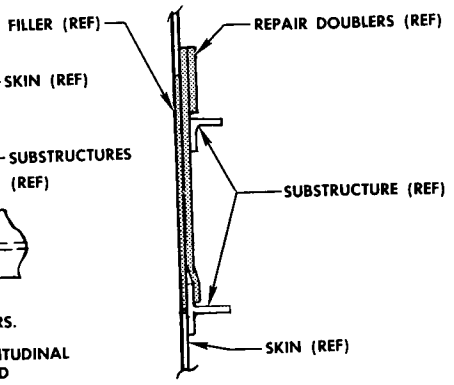
PERMANENT—CATEGORY A

FUSELAGE



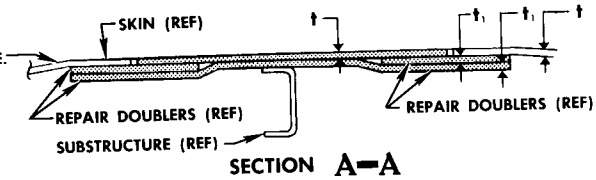
NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

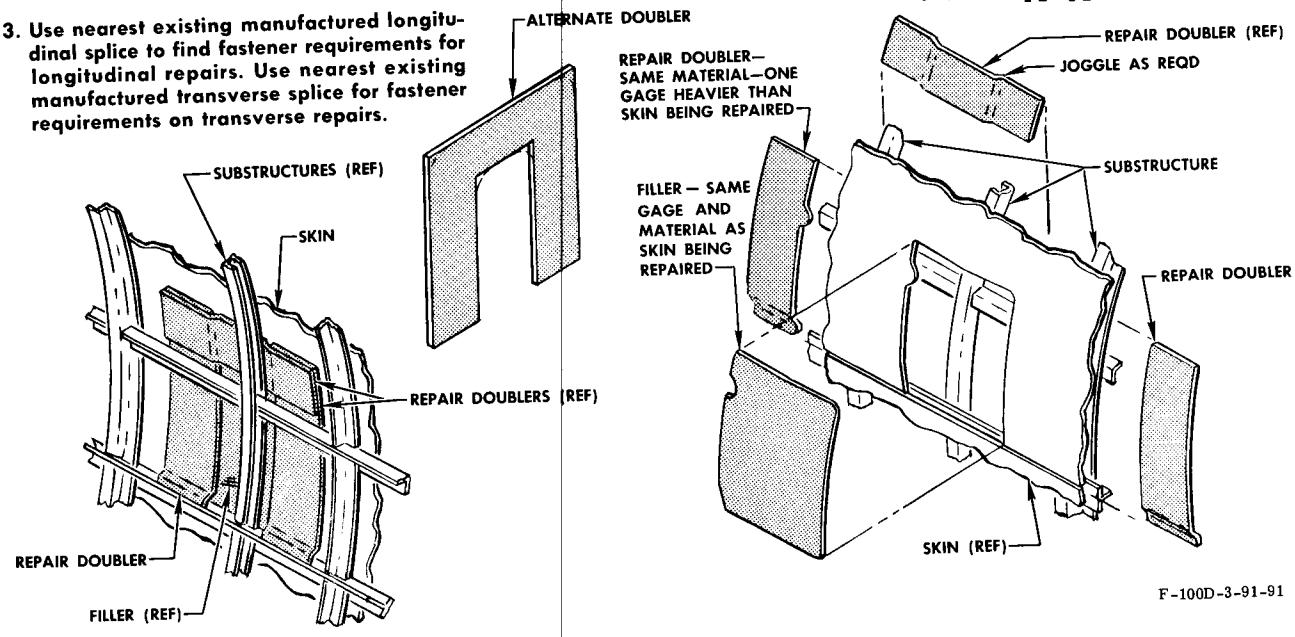


SECTION B—B

1. Trim out damage midway between existing fasteners.
2. Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the number required for the doubler should be four rows of fasteners at one-inch spacing.
3. Use nearest existing manufactured longitudinal splice to find fastener requirements for longitudinal repairs. Use nearest existing manufactured transverse splice for fastener requirements on transverse repairs.

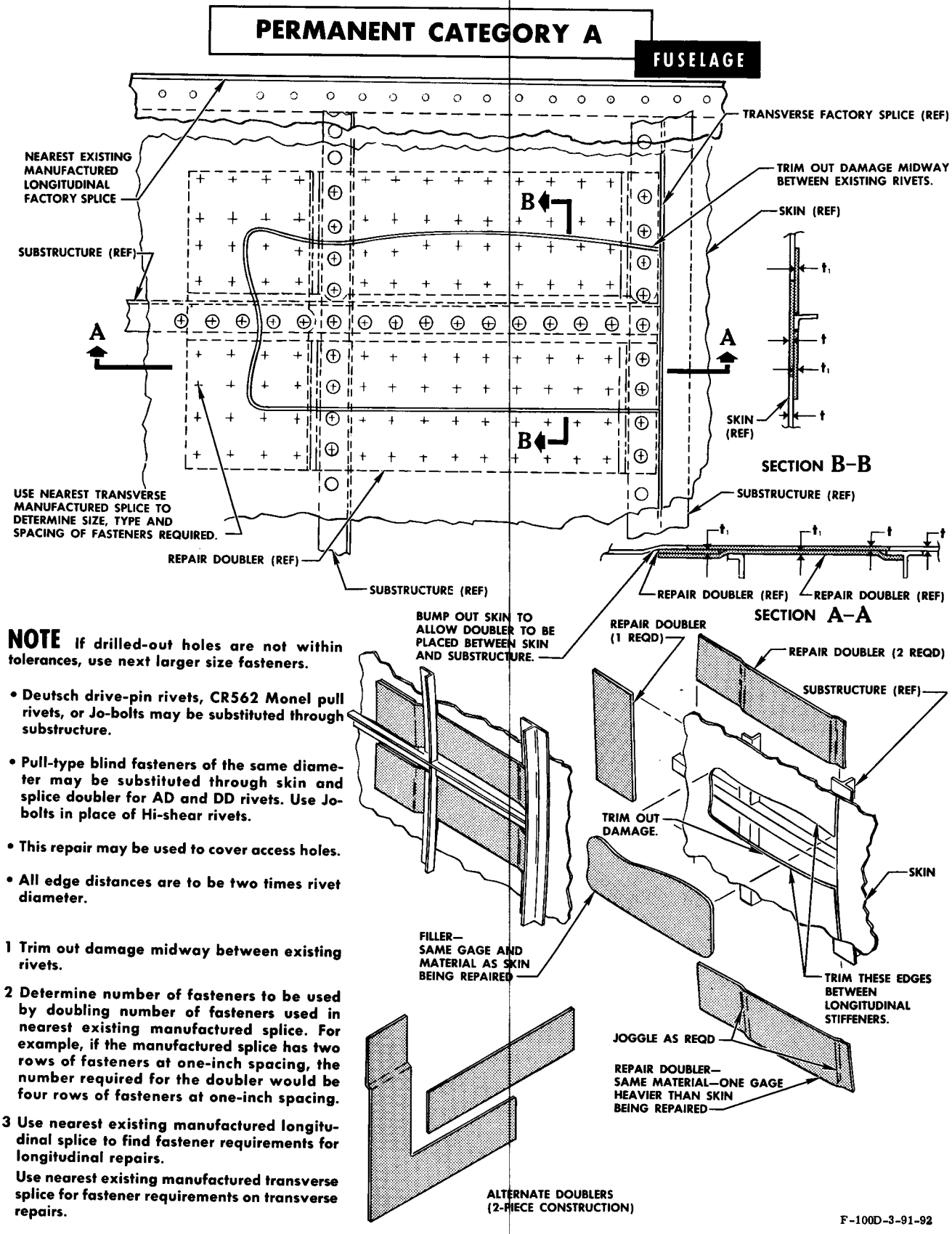


SECTION A—A



F-100D-3-91-91

Figure 4-33. Skin End Repair—Shear (Sheet 2 of 3)



NOTE If drilled-out holes are not within tolerances, use next larger size fasteners.

- Deutsch drive-pin rivets, CR562 Monel pull rivets, or Jo-bolts may be substituted through substructure.
- Pull-type blind fasteners of the same diameter may be substituted through skin and splice doubler for AD and DD rivets. Use Jo-bolts in place of Hi-shear rivets.
- This repair may be used to cover access holes.
- All edge distances are to be two times rivet diameter.

1 Trim out damage midway between existing rivets.

2 Determine number of fasteners to be used by doubling number of fasteners used in nearest existing manufactured splice. For example, if the manufactured splice has two rows of fasteners at one-inch spacing, the number required for the doubler would be four rows of fasteners at one-inch spacing.

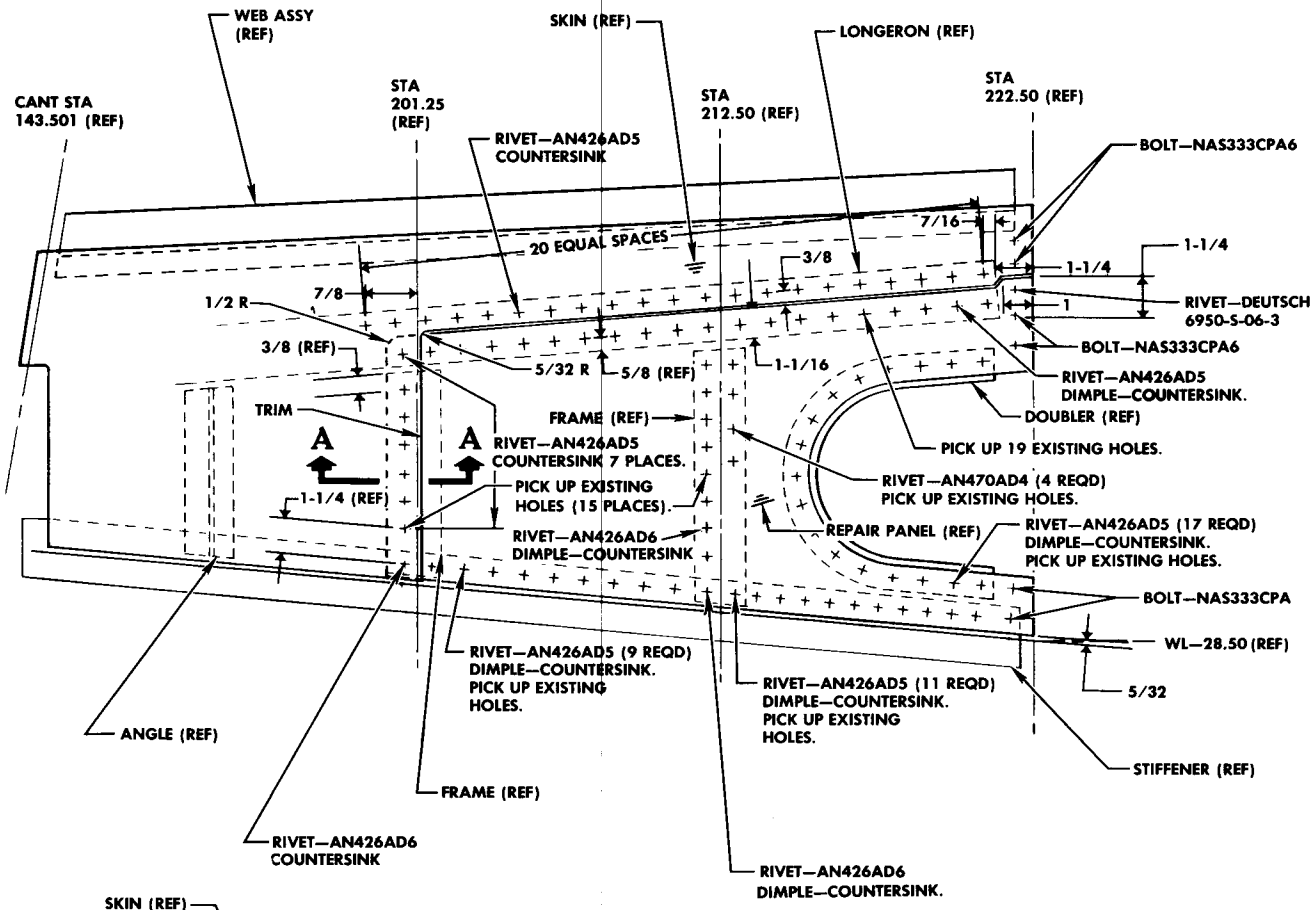
3 Use nearest existing manufactured longitudinal splice to find fastener requirements for longitudinal repairs.

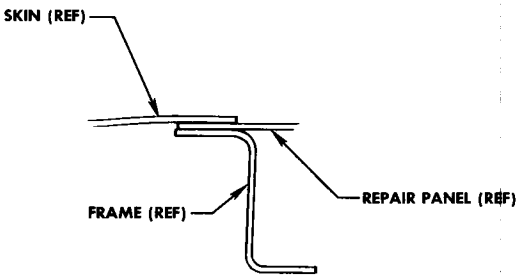
Use nearest existing manufactured transverse splice for fastener requirements on transverse repairs.

Figure 4-33. Skin End Repair—Shear (Sheet 3 of 3)

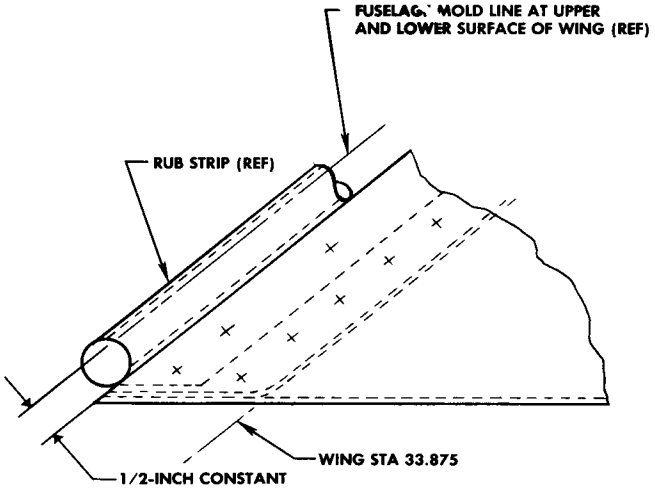
PERMANENT – CATEGORY A

FUSELAGE





VIEW A-A



1. Trim out damaged fuselage panel as shown, and smooth all edges.
2. Make a repair panel 0.032 x 12 x 21 inches from corrosion-resistant steel sheet (Specification MIL-S-5059, composition 301, 1/2 hard).
3. Fasten repair panel in place, using type and size of fasteners and spacing called out.
4. Install wing leading edge.

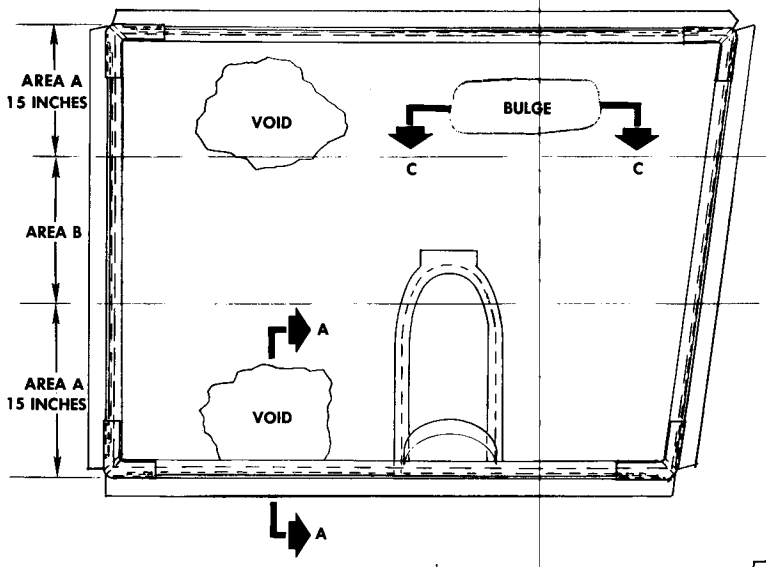
Check clearance between wing leading edge and fuselage. Remove leading edge, and trim it as indicated.

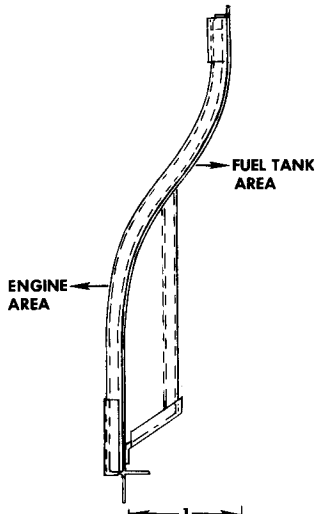
F-100F-3-31-10

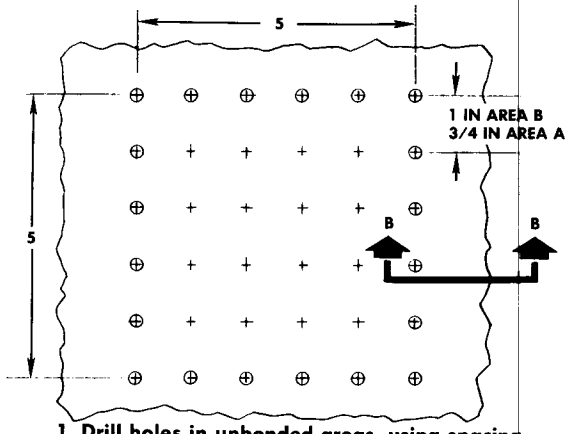
Figure 4-33A. Fuselage Skin Repair—Station 189.89 to 222.5—F-100F-1 Through F-100F-10 Airplanes, and F-100F-15 Airplanes AF56-3920 Through -3967

PERMANENT – CATEGORY A

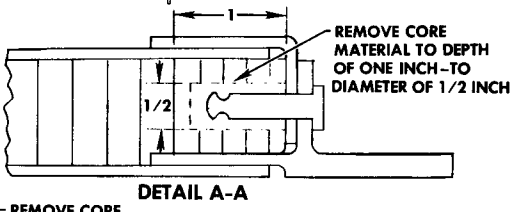
FUSELAGE



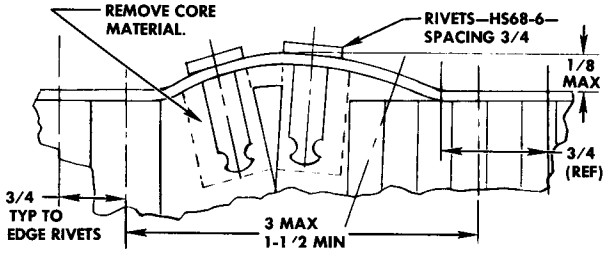




1. Drill holes in unbonded areas, using spacing as shown.
2. Remove core material, using typical tool shown.
3. See detail B-B.

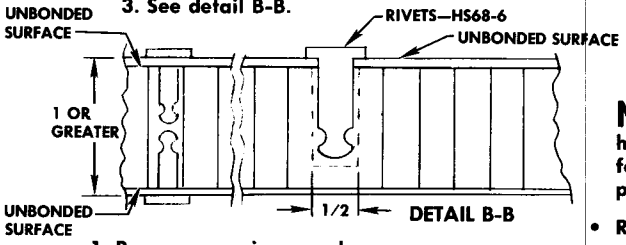


DETAIL A-A



DETAIL C-C

1. For bulges (maximum length 18 inches), remove core material as shown.
2. Inject bonding mixture. (Refer to note.)
3. Install rivets.

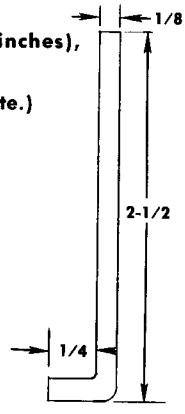


DETAIL B-B

1. Remove core in area shown.
2. Inject bonding material in core area. (Refer to note.)
3. Install HS68-6 Hi-Shear rivet.
4. When both surfaces are unbonded and the thickness of the honeycomb section exceeds one inch, a second rivet may be inserted from the opposite side. For thicknesses less than one inch, the rivets must be staggered.

NOTE For additional information on honeycomb structural repair, refer to index for honeycomb sandwich panel repair procedure.

- Rivets 1/2 inch or more in length must extend to three fourths of core depth; rivets less than 1/2 inch in length must extend to within one sixteenth of opposite face sheet.
- Bulge repair does not apply to sharp creases or buckles.
- Rivets must be cleaned of all foreign matter.
- Rivets must be taped on tank side.



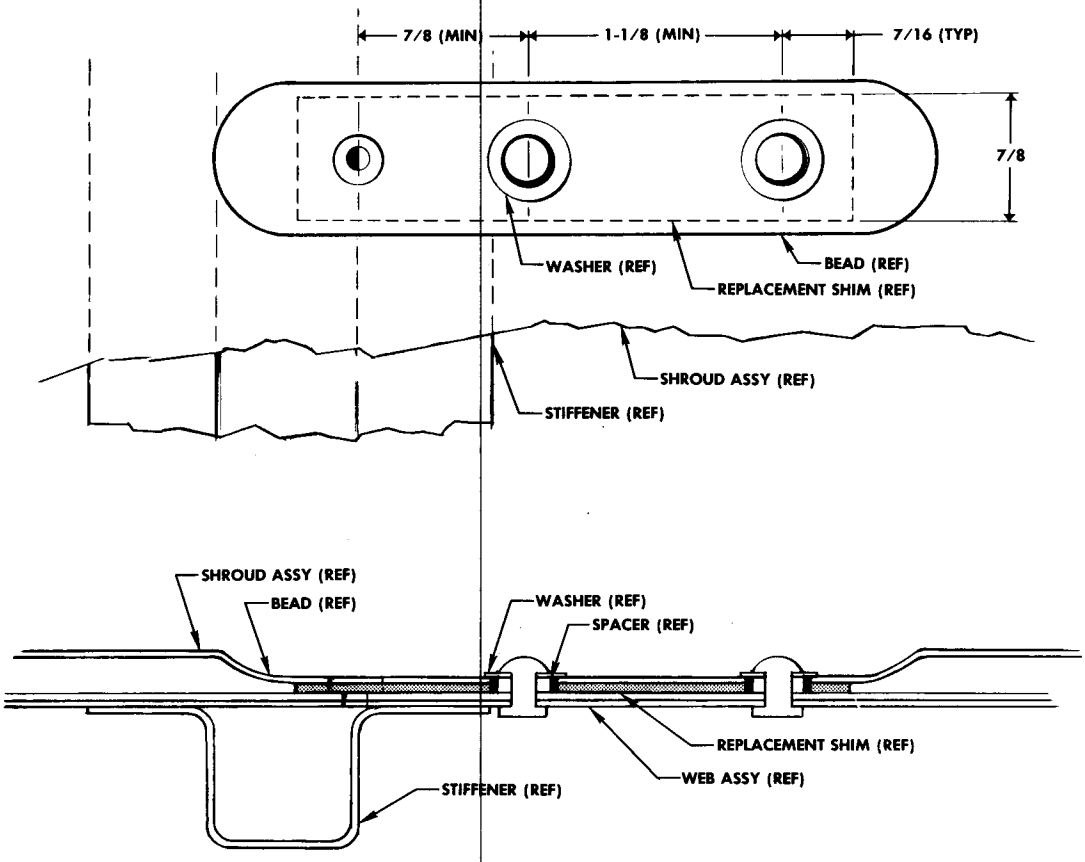
TYPICAL TOOL USED IN A DRILL MOTOR TO CLEAR OUT AREA IN SANDWICH MATERIAL FOR BONDING

F-100D-3-31-86

Figure 4-34. Fuel Bay Close-out Panels—Honeycomb

PERMANENT CATEGORY A

FUSELAGE



When replacing the control well shroud, some of the rivets may not be driven because of the hat sections and channels on the underside of the web assembly. If this condition occurs, the following procedure should be used:

1. Remove existing shims from the affected areas.
2. Make a new shim from 0.063-inch 6061-O material which is long enough to maintain edge distance from the original fastener holes, as shown in the illustration above.
3. Fasten new shim to shroud with either spot welds or tack rivets
4. In new locations, drill 7/16-inch diameter holes through shroud assembly and new shims.
5. Fasten shroud in place using riveting method shown above. Making sure that the rivet and spacer are centered in the oversized hole through shroud assembly and shims.

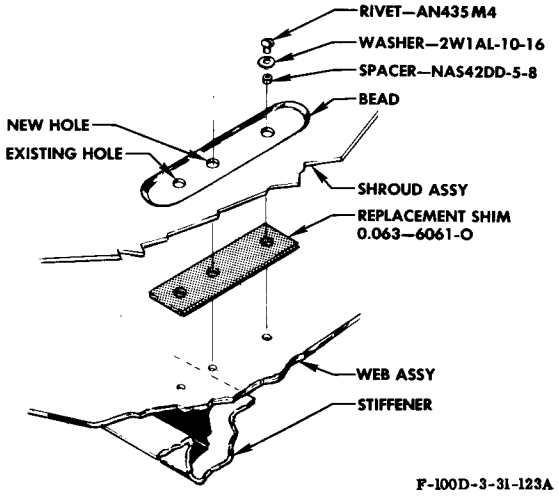


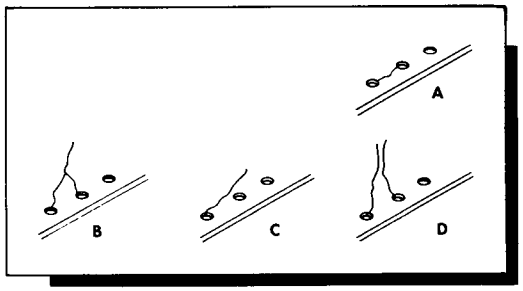
Figure 4-34A. Shroud Repair—Stabilizer Control Well

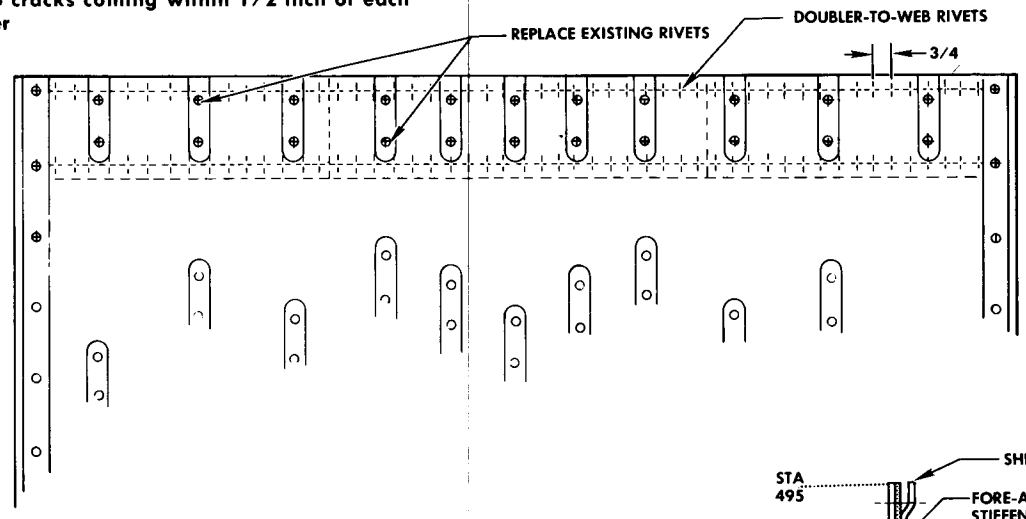
PERMANENT CATEGORY A

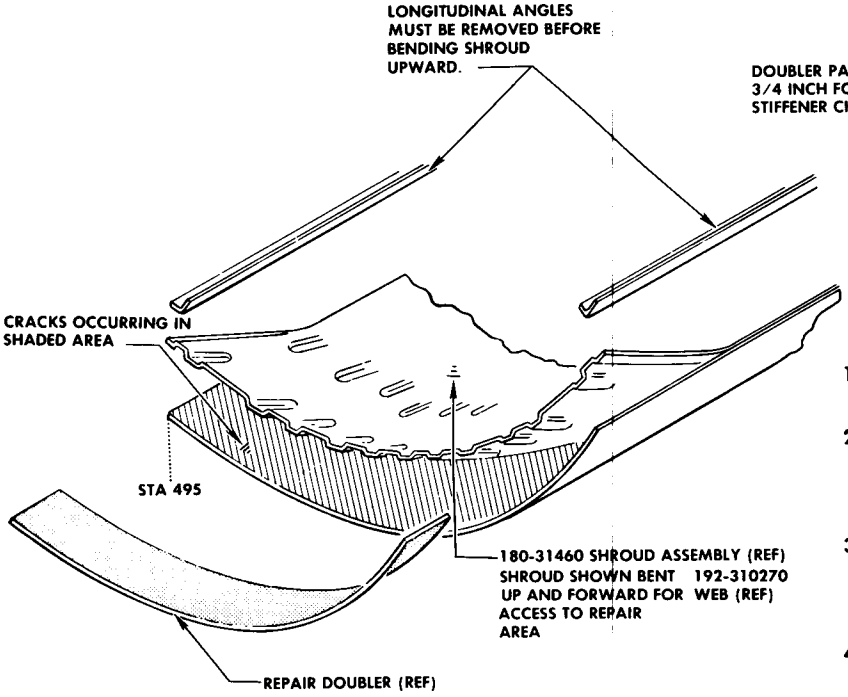
FUSELAGE

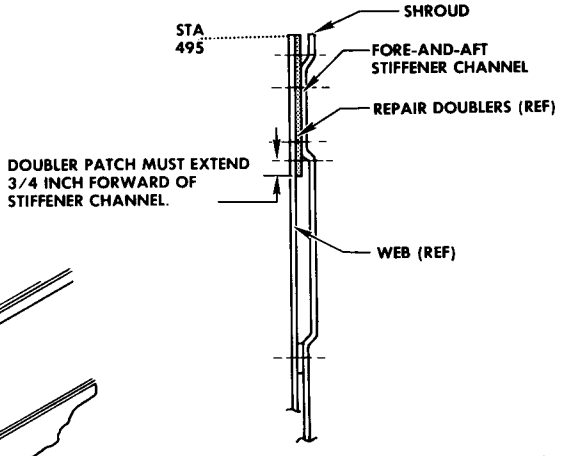
NOTE Repairs are required if any one of the following conditions occur. (Stop-drill all cracks.)

- Cracks running between rivets
- Cracks originating at two rivets which intersect
- Cracks originating at one rivet and passing within 1/2 inch of another rivet
- Two cracks coming within 1/2 inch of each other









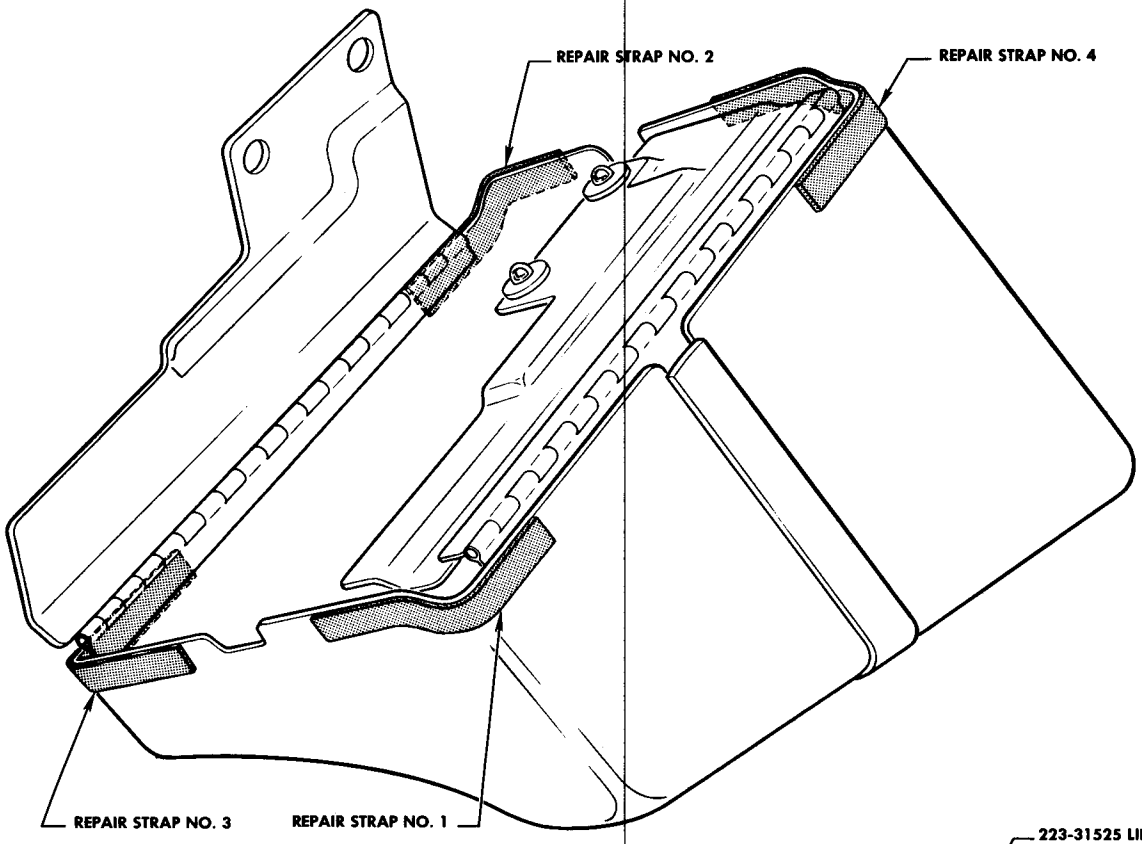
1. Remove existing rivets to allow shroud to be raised as shown.
2. Make repair doublers in one, two, or three sections. Use 0.020- to 0.032-inch corrosion-resistant steel (Specification MIL-5-5059), one-half hard.
3. Apply doubler to upper surface of web, and rivet, using AN435M4 or AN435M5 rivets. The rivets are to be installed at 3/4-inch spacing.
4. Replace shroud and rivet with AN435M4 or AN435M5 rivets, using existing holes. NAS508M rivets may be substituted for AN435 rivets.

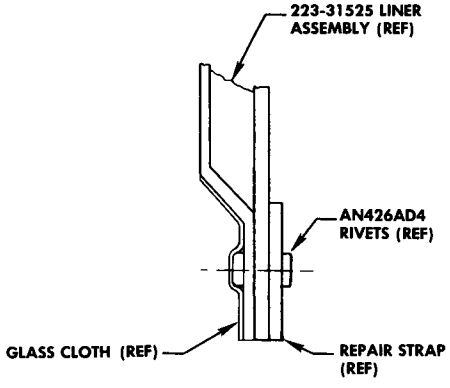
F-100D-3-31-92

Figure 4-35. Titanium Shroud Crack Repair

PERMANENT CATEGORY A

FUSELAGE





NOTE

Make repair straps from 0.020 inch 2024S-T3 clad sheet.

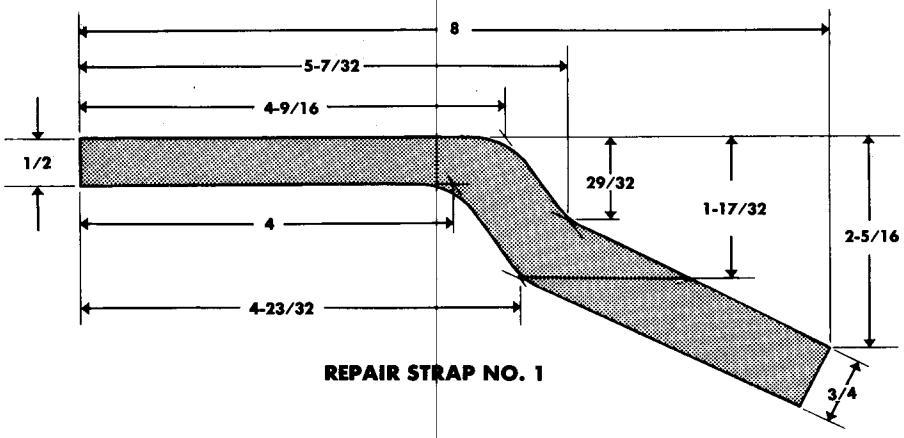
When drag chute liners are cracked and/or breaking out at corners, and when end rivets in hinge assembly are pulled through liners, proceed as follows:

1. Make repair straps as shown on sheet 2.
2. Remove existing rivets around corners of liner.
3. Attach repair straps with AN470AD4 and AN426AD3 rivets. Use flush rivets on sides of liner. Dimple repair straps.
4. Cover reinstalled rivets with glass cloth (forward end only).

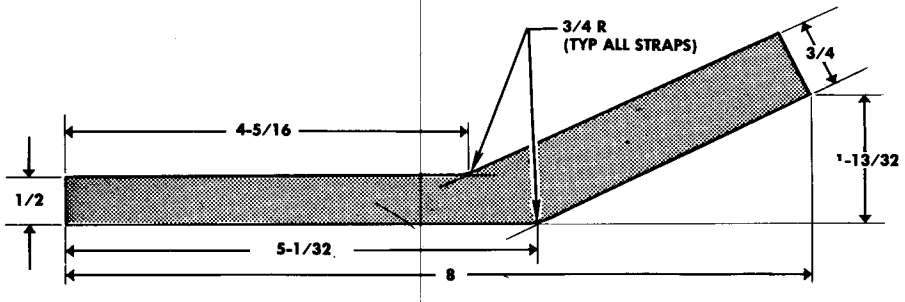
F-100D-3-31-130

Figure 4-36. Drag Chute Liner Repair (Sheet 1 of 2)

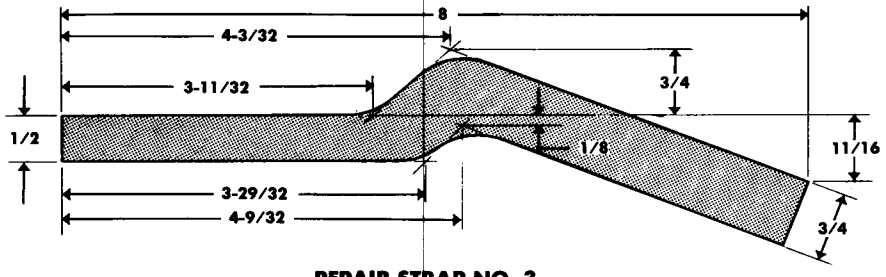
PERMANENT CATEGORY A — FUSELAGE



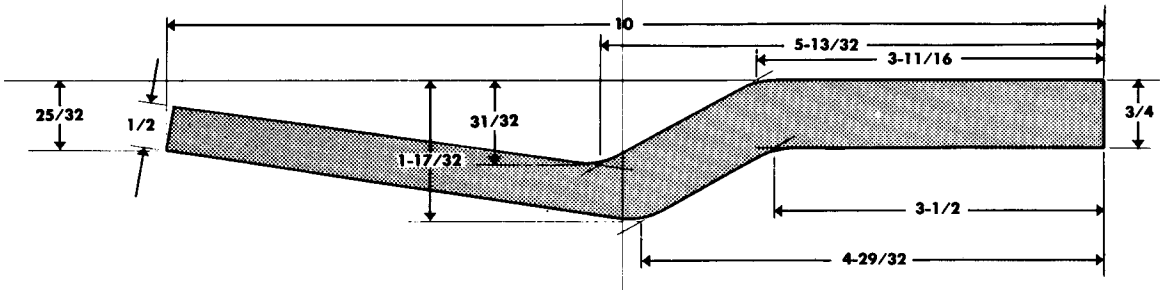
REPAIR STRAP NO. 1



REPAIR STRAP NO. 2



REPAIR STRAP NO. 3



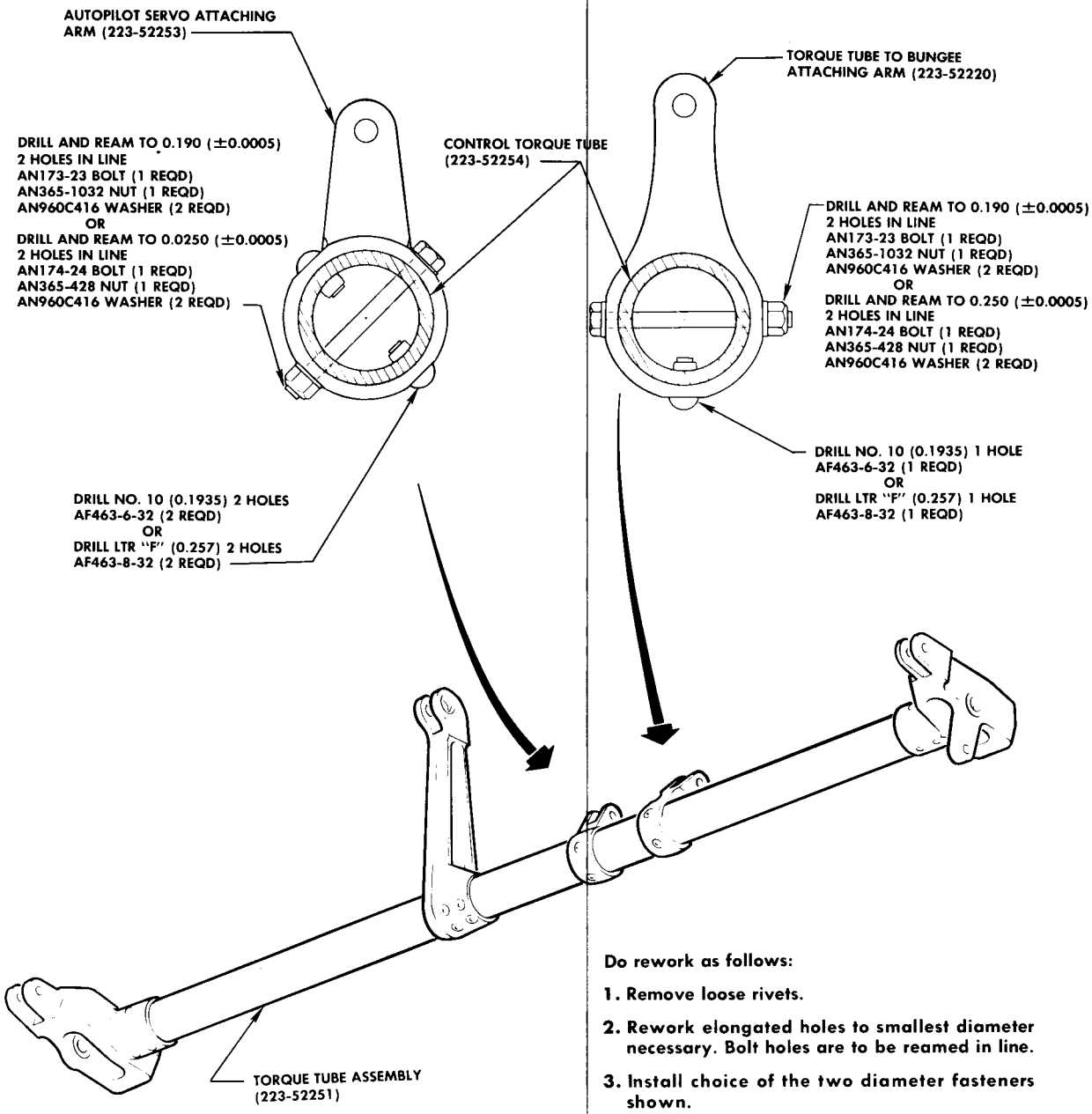
REPAIR STRAP NO. 4

F-100D-3-31-131

Figure 4-36. Drag Chute Liner Repair (Sheet 2 of 2)

PERMANENT CATEGORY A

FUSELAGE



Do rework as follows:

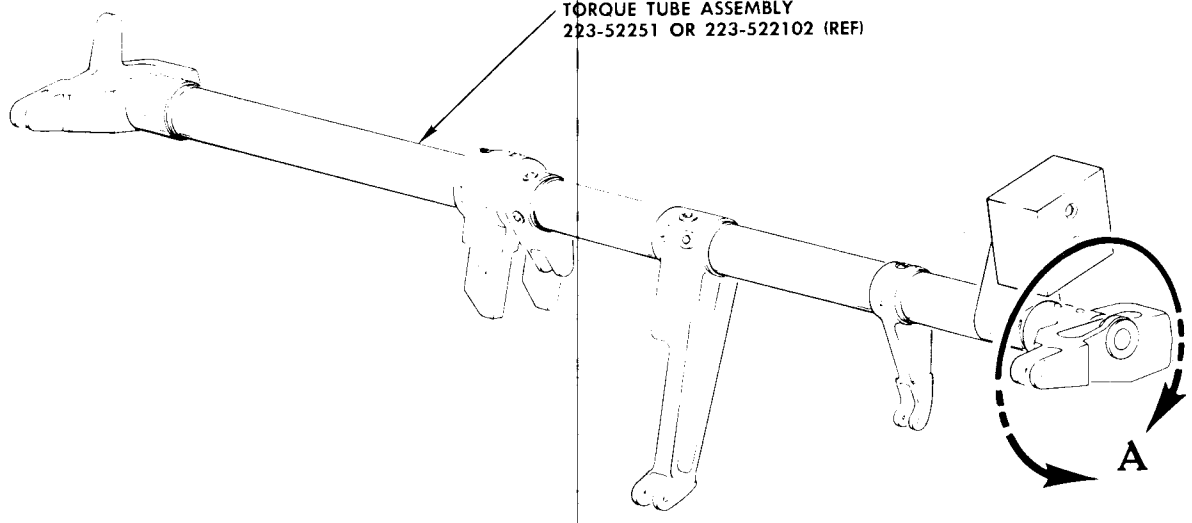
1. Remove loose rivets.
2. Rework elongated holes to smallest diameter necessary. Bolt holes are to be reamed in line.
3. Install choice of the two diameter fasteners shown.

   Use zinc chromate in the holes before installing fasteners.

F-100D-3-31-132

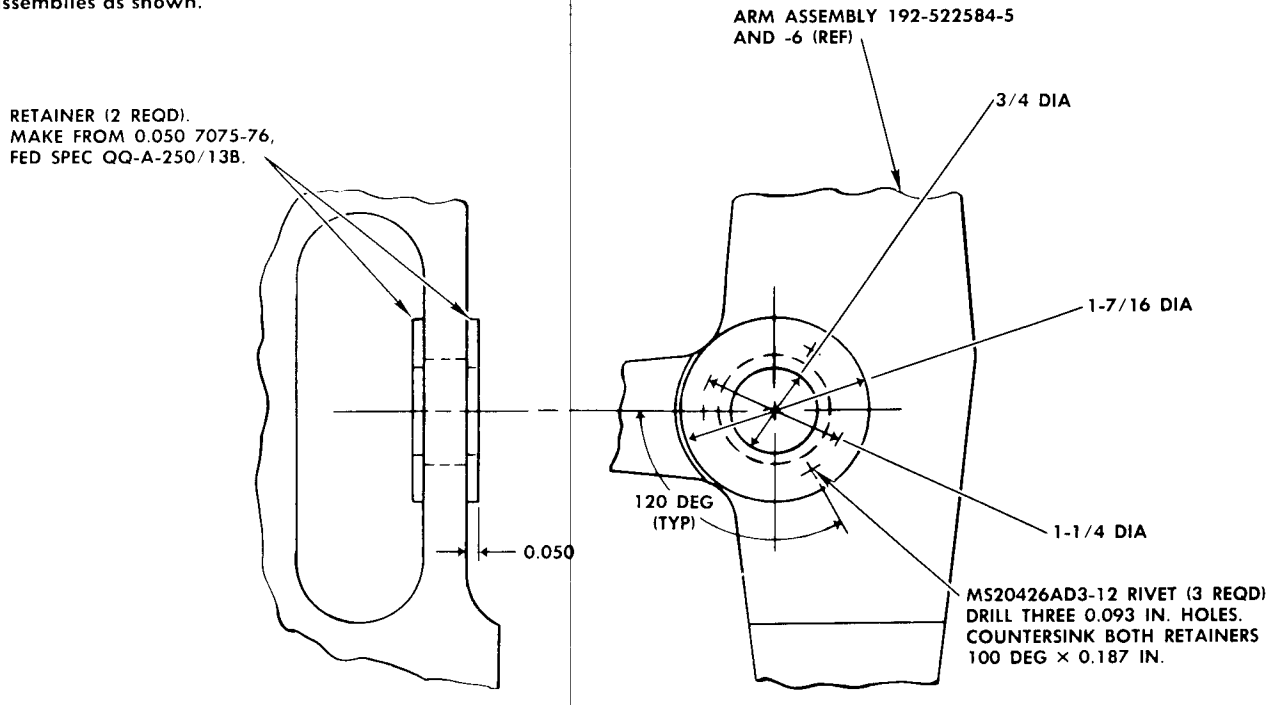
Figure 4-37. Torque Tube Arm Rework (Sheet 1 of 2)

PERMANENT—CATEGORY A **FUSELAGE**



Do rework as follows:

1. Remove damaged bearing and replace with new AN200KS4 bearing.
2. Secure bearings by riveting retainers to arm assemblies as shown.



DETAIL A

F-100D-3-31-161

Figure 4-37. Torque Tube Arm Rework (Sheet 2 of 2)

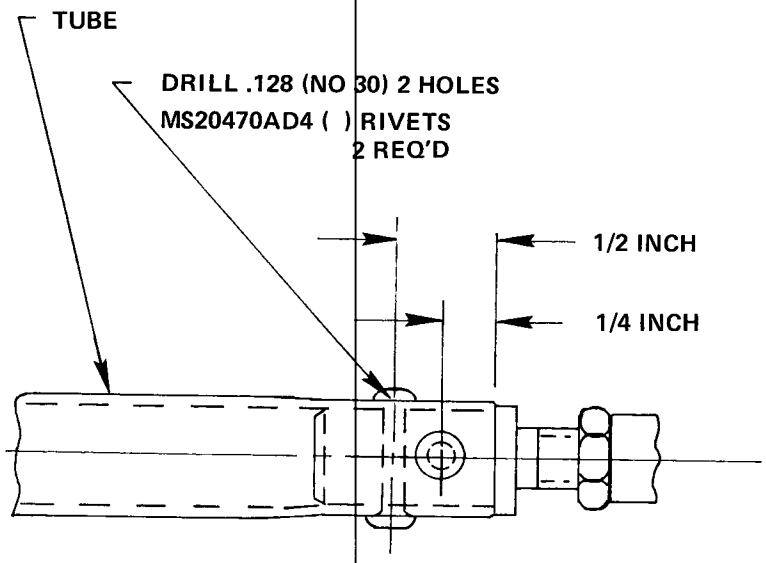


Figure 4-37A. Flight Control Rod Assembly-Typical Repair
(Permanent-Category A)

HOLDING FIXTURE DIMENSIONS

FUSELAGE

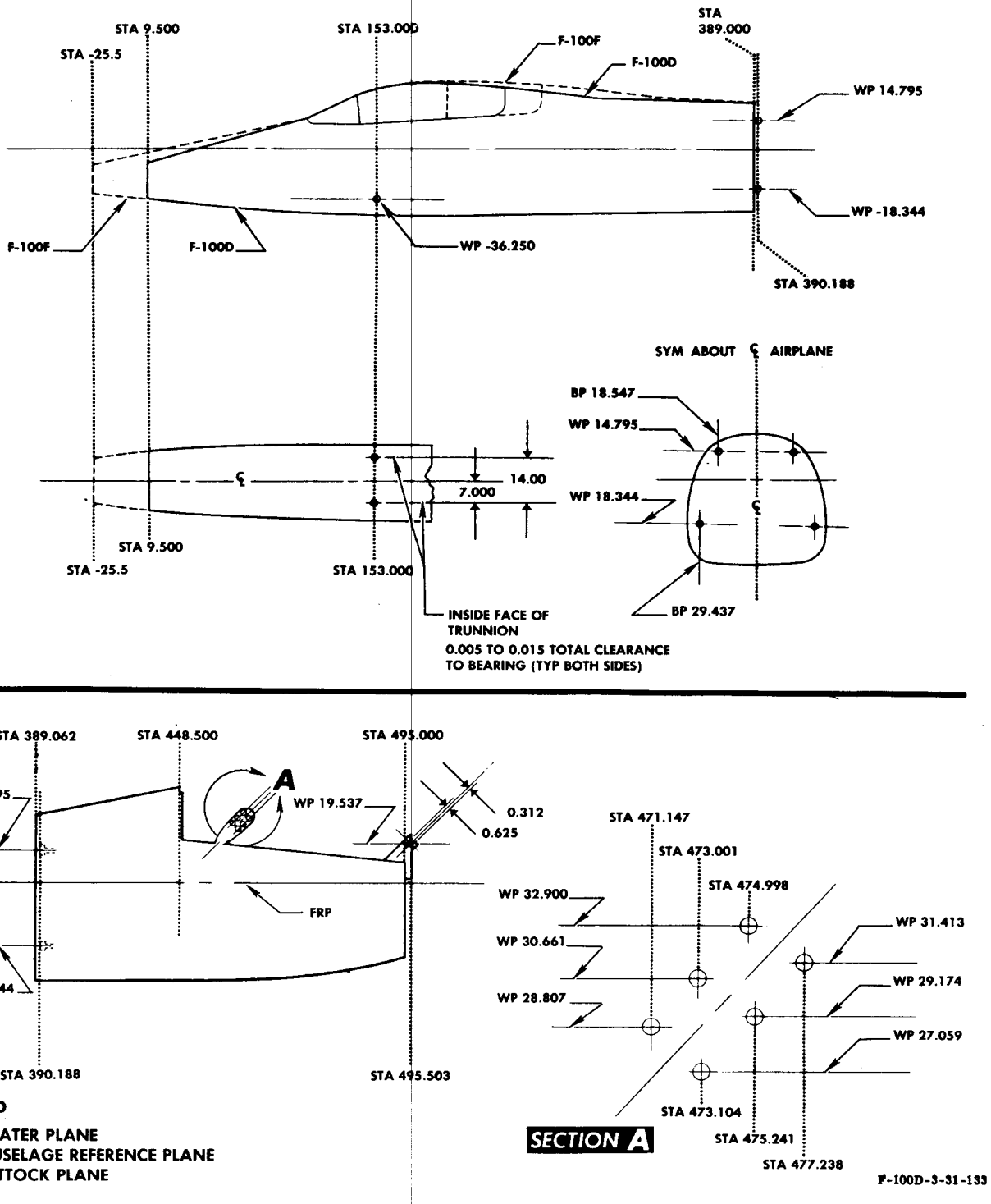


Figure 4-38. Holding Fixture Dimensions—Fuselage

Section V

LANDING GEAR

Contents

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CORRECTION OF MAIN LANDING GEAR SHIMMY	5-2A
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EMERGENCY LANDING GEAR RELEASE HANDLE SUPPORT REPAIR	5-3
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GENERAL DESCRIPTION OF LANDING GEAR.

The airplane has a tricycle-type landing gear with air-oil shock struts and extra-high-pressure tires. There is also a fully retractable tail skid. The tail skid is retracted flush with the lower aft surface of the fuselage. The landing gear is fully retracted inside the fuselage and is faired in by wheel and shock strut fairings. The landing gear is operated by an electrically controlled hydraulic system. The tail skid is operated electrically.

MAIN LANDING GEAR.

MAIN GEAR SHOCK STRUTS.

The main gear shock struts are of the air-oil, shock absorber type, containing a specific amount of compressed air and hydraulic fluid. The strut has a large cylinder assembly at the upper end, and a piston assembly (commonly referred to as the "oleo") at the lower end. The cylinder and the oleo assembly are connected together by torque links at the lower end of the complete strut assembly. The torque links transmit torque loads from the oleo assembly to the cylinder, and keep the cylinder and oleo assemblies in alignment. The outside of the cylinder is machined to provide trunnion support fittings and attach fittings for the actuating cylinder, side brace, and upper torque links. There is a filler valve at the top of the cylinder assembly for servicing purposes.

MAIN GEAR WHEELS.

A single split-type, cast-magnesium alloy, 30 x 8.8-inch, Type VII wheel is used on each main gear. The wheels are held together with bolts. A rotor-disk brake is splined directly to the axle on the outboard section, and side of the wheel. Mounted to each wheel is a 30 x 8.8-inch extra high-pressure 22-ply rating tire. The two halves of the wheel must be unbolted and separated for installation or removal of tires.

MAIN GEAR SHOCK STRUT FAIRING DOORS.

The main gear shock strut fairing doors close off and fair the main gear shock strut and gear side brace wells when the gear is retracted. There are three separate door fairings: the outboard side brace hinged fairing, the inboard side brace fixed fairing, and the main gear shock strut fairing. All the fairings are outboard of, and operated by, linkage to the gear side brace. The outboard side brace hinged fairing is attached to the lower structure of the airplane by a hinge fitting on one end, and to the side brace by a link on the opposite end. The inboard side brace fixed fairing is attached directly to the side brace. The main gear strut fairing is attached by a hinge fitting to the lower wing structure, just outboard of the gear trunnion mount. The fairing actuating link is attached to the door at the forward edge about the center of the unit, and to the inboard end of the side brace at the strut. All of the fairings are adjusted to close flush with the wing surface.

MAIN GEAR WHEEL FAIRING DOORS.

The main wheel fairing doors close off and fair the main landing gear wheel wells. The doors are hinged at the bottom centerline of the fuselage to a structural member and are hydraulically actuated. A spring-loaded tab attached to the aft outboard edge of each door fairs in the fuselage between the wheel and strut doors. The hinges and uplock rollers are adjustable for fairing the door to the skin line.

The main wheel fairing door is divided; it has an inner and an outer door. The outer door is closed by an actuating link connected to the structure near the door actuating cylinder.

NOSE LANDING GEAR.

NOSE GEAR SHOCK STRUT.

The nose shock strut assembly is on the forward lower centerline of the fuselage, directly under the cockpit. The shock strut is of the air-oil, shock absorber type, containing a specific amount of compressed air and hydraulic fluid. The strut has a large cylinder assembly at the upper end, and a piston assembly (commonly referred to as an "oleo" assembly) at the lower end. The cylinder assembly and the oleo assembly are connected together by torque links at the lower end of the strut assembly. The torque links transmit torque loads from the oleo assembly to the cylinder through the steer-damp unit to provide for steering and shimmy damping. The outside of the cylinder is machined to provide trunnion support fittings and attach fittings for the nose wheel steering mechanism, actuating cylinder, drag brace, and upper torque links. There is a filler valve at the top of the strut for servicing purposes. A centering cam and an oleo support assembly are at the bottom of the cylinder. The outside of the piston, or oleo assembly, consists of the lower torque links, an axle, and provisions for towing the airplane.

NOSE GEAR WHEEL ASSEMBLY.

Dual nose wheels are mounted on the lower end of the nose gear shock strut. When nose gear tires are replaced, they should be closely matched to prevent excessive nose wheel shimmy or vibration.

NOSE GEAR SHOCK STRUT FAIRING DOOR.

The nose gear shock strut fairing door closes off and fairs the forward end of the nose gear shock strut well. The door is hinged to structure on the forward end of the nose gear shock strut well and is operated by a connecting link from the nose gear shock strut. The forward edge of the door is adjusted to the skin line of the wheel well by washers between the hinges and the door. The aft end of the door is adjusted to the skin line by the connecting link from door to strut.

NOSE GEAR WHEEL FAIRING DOOR.

The nose gear wheel fairing door closes off and fairs the nose gear shock strut and wheel well. The door is hinged to the right side of the nose gear wheel well structure by three hinges. The hydraulic cylinder used for opening and closing the door is attached to the center hinge at the door. The door is adjusted to the skin line on the right side of the well by washers between the hinges and inside door frame. The lock rollers are on the left side of the inner door frame. The lock rollers are adjusted to fair the left door frame to the skin line.

TAIL SKID.

The tail skid is on the lower side of the fuselage rear is hinged at the forward end of the tail skid well and to the lower end of the shock strut. The force used to extend and retract the tail skid is applied to the tail skid through the lower end of the shock strut. The tail skid shock strut absorbs the impact loads imposed on the tail skid in tail-down take-offs and landings.

REPAIR OF LANDING GEAR.

Because of the close tolerances and highly stressed condition of the landing gear structural components, the repair of any damage more serious than small nicks (caused by flying rocks, contact with ground handling equipment, etc) is impractical.

Small nicks may be polished out with a 6-inch, No. 6 cut, Swiss pattern file or equivalent, and No. 400 or finer grit emery paper, or a burnishing tool. After damage has been removed, apply at least one coat of zinc-chromate primer (Specification MIL-P-6889) to the smoothed area.

Small cracks may be detected by application of a liberal coat of light oil to the part or area suspected. Thoroughly wipe away the oil and apply a coat of whiting (calcium carbonate, Specification JAN-C-293). Cracks will show by the appearance of oil on the whiting and are cause for replacement of the unit.

MAIN GEAR AXLE NEGLIGIBLE DAMAGE.

See figure 5-5.

TAIL SKID SHOE WEAR DIMENSIONS.

Wear on the tail skid shoe up to 50-percent "wear off" is permitted. When wear reaches the dimension shown in figure 5-3, the shoe is considered to be worn about 50 percent. A tail skid shoe worn beyond these dimensions must be replaced.

CORRECTION OF MAIN LANDING GEAR SHIMMY.

Note

When measurements of components are out of allowable tolerance, repair listed parts in accordance with USAF Drawings:

62C40062—Clevis Repair—MLG Side Brace

62B40053—Brace Repair—MLG Complete Side

62D40057—Bushing—Tabulated

1. Jack the airplane as outlined in "F-100D General Airplane," T.O. 1F-100D-2-1. The gear must be down and locked (hydraulic pressure off) and clear of the floor.

2. Check for excessive play in main landing gear side brace installation.

3. Use an inboard-outboard force of 160 pounds applied at centerline of axle and check for a side movement of 0.040 inch at centerline of strut pin (side brace attach). If total side movement exceeds 0.040 inch, check component parts as follows and replace parts when necessary. (See figure 5-10.)

4. CATEGORY A REPAIR. When need for strut replacement is indicated, see figure 5-12 for allowable tolerances.

Hole—strut pin, side brace	2.0005 in. max	MLG strut assy
Pin—strut, side brace	1.9982 in. min	MLG strut assy
Max allowable clearance	0.0023 in.	
Hole—clevis, side brace	2.0002 in. max	180-33104
Pin—strut, side brace	1.9982 in. min	MLG strut assy
Max allowable clearance	0.0020 in.	
Hole—pivot pin, clevis, side brace	1.0005 in. max	180-33104
Pin—pivot, side brace attach	0.9980 in. min	180-33113
Max allowable clearance	0.0025 in.	
Pin—clevis to side brace end play	0.002 in. min 0.010 in. max	180-33113
Hole—pivot pin, side brace	1.0005 in. max	180-33116
Pin—pivot, side brace attach	0.9980 in. min	180-33113
Max allowable clearance	0.0025 in.	
Adjustment—side brace to stop	0.010 in. max 0.004 in. min	180-33116 180-33510

Note

The clevis-to-side-brace attach pin (180-33113) has been known to corrode and stick in the side brace linkage. Lubricate well before reinstallation.

STRUT FREE-PLAY CHECK.

1. Check fore-and-aft movement of strut journal by applying a 50-pound force parallel to centerline of trunnion. Movement along the centerline of the trunnion should be no less than 0.005 inch and should not exceed 0.062 inch. If movement exceeds this dimension, check for:

- Improper torque of trunnion nut.
- Worn thrust bearing. (See figure 5-10.)

Note

Before installation of main landing gear strut in airplane, take micrometer reading of strut collar and of bronze bearing shoulder to ensure that required clearance is available. If clearance is insufficient, replace forward thrust bearing.

- Missing forward thrust bearing.

2. Check fore and aft play at uplock roller by applying a 160-pound force at centerline of axle.

3. Subtract play at trunnion from play found at uplock roller. If this measurement exceeds 1/16 inch, it is a good indication of a worn thrust bearing, strut journal, or journal bearing.

4. Examine strut journals and bearings thoroughly for galling, lubrication, wear, and fit.

JOURNAL	BEARING INSTALLED ID	MAX ALLOWABLE CLEARANCE	BEARING PART NO.
Fwd			
2.9987 in. max	3.0030 in.	0.0048 in.	180-33121
2.9982 in. min	3.0010 in.		
Aft			
4.9987 in. max	5.0040 in.	0.0058 in.	192-33122
4.9982 in. min	5.0020 in.		

5. Use temperature/diameter correction chart (figure 5-9) when measuring diameters. When strut journals exceed maximum or minimum measurements, replace strut.

6. When in-service trunnion bearing ID exceeds 3.003 inches for forward bearing and 5.004 inches for aft bearing, replace bearing.

CHECKING TRUNNION BEARINGS.

1. Remove bearing, using pulling and installing trunnion support bearing tool assembly (60SCJ6265).

Note

Kit 62B40582, containing the necessary main gear trunnion bearing and bearing cavity rework tools, can be obtained on loan from SMAMA (SMNBSP).

2. Measure inside diameter of trunnion bearing spar cavity in at least three dimensions—vertical, horizontal, and in-between diameter—using temperature/diameter measurement correction chart (figure 5-9) to ensure correct measurements. Bearing cavities must not exceed the following measurement for installation of a standard outside diameter bearing.

- Forward bearing cavity, 3.2500 (±0.0005) inches
- Aft bearing cavity, 5.2500 (±0.0005) inches

If forward and/or aft bearing cavity measurements are within this dimension, omit reaming operation and proceed with installing bearings.

REAMING FORWARD AND AFT BEARING CAVITIES.

Any bearing cavity measurement that varies more than the dimensions outlined in the preceding paragraph will require rework by reaming the bearing support cavity to accept a 0.010-inch oversize bearing.

1. Ream forward cavity to 3.2600 (±0.0005) inches, using ream tool (60SCD6279), as follows:

a. Position bar assembly (60SCD6285) in spar cavity. Lubricate bar as required.

b. Install guide assembly (60SCD6280) over bar assembly and slide guide assembly forward until shoulder of guide assembly contacts bearing (192-33122).

c. Turn AN365-624 nuts to expand fingers. Tighten opposite nuts, alternately, until guide assembly is held securely. Do not tighten nuts excessively.

d. Turn nut assembly (60SCC6283) onto guide assembly (60SCD6280) until bar assembly (60SCD-6285) is contacted.

e. Install driver (60SCB6281) on end of bar assembly.

f. Turn driver with suitable hand wrench, or power-drive at approximately 70 rpm or slower.

g. Feed bar assembly into work by turning nut assembly (60SCC6283) as far as possible. After cut is complete, retract bar by turning nut assembly back to starting position.

h. Back off AN365-624 nuts until fingers retract and remove entire boring tool.

i. Completely clean chips and other material from spar cavity.

2. Ream aft cavity to 5.2600 (±0.0005) inches, using reamer tool (60SCH6266) as follows:

a. Install stud (180-33140).

b. Screw -101 guide assembly in place. Torque to approximately 1000 inch-pounds. Lubricate with a light oil.

c. Install -1 head on the -101 guide assembly and slide forward as far as possible.

d. Screw -4 screw into -101 guide assembly and turn until contact is made with -1 head. Secure screw to head with two MS35455-35 screws.

e. Install MS16253-14 socket, MS16252-3 ratchet, and turn with MS16245-3 flex handle.

f. Hand-rotate reamer with a slow, steady movement to avoid tool shatter and an oversized cavity.

g. Turn until rotating parts bottom (flange of -4 screw contacts -101 guide assembly).

h. Remove the two MS35455-35 screws attaching -4 screw to -1 head. Remove -4 screw.

i. Attach -102 handle to -1 head with the two MS35455-35 screws. Pull -1 head back, and remove.

j. Remove the -101 guide assembly.

k. Completely clean out all chips from trunnion cavity.

l. The reamer is set to cut 0.010 inch oversize. To change size, expand blades, and finish-grind to diameter desired.

INSTALLING BEARINGS.

1. Clean bearing cavities and lubrication groove and treat with Alodine 1200 (Specification MIL-C-5541) to prevent corrosion, according to instructions in Section I.

2. Select applicable 0.010-inch oversize or standard bearings for installation as indicated in the following:

- Oversize forward trunnion bearing 180-33121-10.
- Oversize aft trunnion bearing 192-33122-10.

Note

A modified aft trunnion bearing (61040955-10) may be substituted for the 192-33122-10 bearing. This bearing can easily be identified by the four additional grease holes, making a total of eight grease holes.

3. Select bearings for installation in trunnion cavity for an interference fit of 0.0025 to 0.0040 inch for forward trunnion bearing, and 0.0045 to 0.0060 inch for aft bearing. The inside diameter of the bearing before installation must be 3.0032 (±0.0005) inches for the forward bearing and 5.0056 (±0.0005) inches for the aft bearing. When standard outside diameter bearings are required, the following bearings must be used:

- Forward trunnion bearing 180-33121.
- Aft trunnion bearing 192-33122.

Note

A modified aft trunnion bearing (61C40955) may be substituted for the 192-33122 bearing. This bearing can easily be identified by the four additional grease holes, making a total of eight grease holes.

4. Install selected bearings in trunnion cavity, using tool assembly 60SCJ6265, as follows:

a. Chill bearings with dry ice to assist installation.

b. Install bearings so that bearing flange contacts spar.

5. After bearing returns to normal temperature, use temperature correction chart (figure 5-9) to ascertain desired inside diameter of 3.001 to 3.002 inches for forward bearing and 5.002 to 5.003 inches for aft bearing.

6. An undersized bearing inside diameter may be honed to size, using a cylinder hone (FSC 5130-222-3905 or 5130-222-3906) powered by a suitable 90-degree air drill. Rotate hone at low rpm (80 rpm desired). Not more than 0.0005 inch may be removed in this manner. If honing is necessary, restone edges of oil groove and edges of bearing to 0.010 to 0.020 inch radius.

7. Clean bearing.

8. Apply grease gun to zerk fittings to ensure that lubrication grooves and holes are not obstructed.

9. Apply grease to strut bearing surfaces and install struts in airplane. (Refer to "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4.) Adjust main landing gear downlock clearance to obtain a minimum clearance of 0.004 to 0.006 inch.

10. Perform normal and emergency landing gear operational check. (Refer to "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4.)

11. Perform bleed and operational check. (Refer to "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4.)

12. Remove airplane from jacks and return to service.

CORRECTION OF NOSE WHEEL SHIMMY.

To correct nose wheel shimmy conditions, use the following procedures. (For jacking and removal procedures, refer to "F-100D General Airplane," T.O. 1F-100D-2-1, and "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4, respectively.)

Note

When measurements of components are out of allowable tolerance, repair listed parts in accordance with USAF Drawings:

62D40105—Link Repair—NLG Drag Brace Strut Upper
62C40059—Link Repair—NLG Drag Brace Strut Lower
62C40061—Brace Repair—NLG Drag Brace Support
62D40057—Bushing—Tabulated

1. Jack airplane and remove nose gear shock strut.

2. Remove drag brace assembly.

3. Remove nose landing gear hydraulic retract cylinder attaching pin (180-34114-7).

4. Remove nose landing gear beam assembly (180-34116-5).

5. Remove the two nose landing gear strut trunnion side brace attaching inner races and the two right-hand beam side brace attaching inner races from bearing assembly (142-33139).

6. Measure inside diameter of inner race; if it is over 1.125 inches, remove from airplane and condition-tag inner race. Install new bearing inner race (142-33139-4).

Note

The press fit of the outer race in the airplane structure is 0.001 to 0.003 inch. The race is installed with the bearing removal slots at a 45-degree angle, with the highest slot on the forward side.

7. Measure outside diameter of inner race; if it is less than 1.6275 inches, tag inner race as condition-condemned. Install new bearing race (142-33139-4.)

8. Measure inside diameter of outer race installed in airplane structure; if it exceeds 1.6305 inches, remove it and replace it with a new bearing outer race (142-33139-3). Condemn outer race originally installed.

9. If all measurements are within specified tolerances, reinstall inner races.

10. Remove inner race from left beam assembly side brace support bearing (120-33144).

11. Measure inside diameter of inner race; if it exceeds 1.3755 inches, tag inner bearing race as condition-condemned. Replace with a new bearing inner race (120-33144-3).

Note

The press fit of the outer bearing is 0.0000 to 0.0015 inch. The race is installed with the bear-

ing removal slots at a 45-degree angle, with the highest slot on the forward side.

12. Measure outside diameter of inner race; if it is less than 1.8145 inches, replace with new bearing inner race (120-33144-3). Condition-condemn removed race.

13. Measure inside diameter of outer race installed in airplane structure; if it exceeds 1.8175 inches, remove outer race. Tag as condition-condemned. Install new bearing outer race (120-33144-2).

14. If all measurements are within specified tolerances, reinstall inner race.

15. Measure attachment holes in beam assembly (180-34116-5). Replace beam assembly if dimensions exceed the following:

- Right-hand beam assembly pinhole, 1.1255 inches
- Left-hand beam assembly pinhole, 1.3755 inches
- Beam assembly drag brace attaching hole, 1.0005 inches

16. Measure beam-to-drag-brace attaching bolt (180-34110). Replace bolt if it is less than 0.9975 inch.

17. Measure drag brace beam support pins (left, 180-34127; right, 180-34128). Replace if dimensions are less than the following:

- Left, 1.3725 inches
- Right, 1.1225 inches

18. Install beam assembly (180-34116-5), with side play of 0.005 to 0.020 inch obtained by adding or removing washers as shown in figure 5-10. (The right-hand beam is shown; the left beam is installed in the same manner.)

19. Remove center bolt (180-34111) from drag brace assembly (180-34103). Replace bolt if diameter is less than 0.873 inch.

20. Measure drag brace center attaching bolthole in both upper and lower drag brace link assemblies.

Replace link assembly if dimension of hole exceeds 0.8755 inch.

21. Remove inner race from bearing (120-33144) in lower drag brace link assembly (180-34104-5).

22. Measure bearing in accordance with steps 6 through 9.

23. Check dimensions of beam attaching hole in upper drag brace link (180-34105-5). If diameter of hole exceeds 1.0005 inches, condition-tag removed link assembly and install new link assembly.

24. Replace GB-118 bearings in upper drag brace link when inside diameter exceeds 0.6901 inch.

25. Check critical areas for wear in strut trunnion and strut drag brace pinholes.

	ALLOWABLE TOLERANCE
Strut trunnion pinhole	1.1265 in. max
Strut trunnion pin	1.1230 in. min
Strut drag brace pinhole	1.3765 in. max
Strut drag brace pin	1.3730 in. min

26. Remove torque fitting (steering collar) and its bushings from strut outer cylinder and check as follows:

a. Torque fitting assembly bushing inside diameter, 4.0680 inches maximum; strut assembly outer cylinder, 4.0640 inches maximum; maximum allowable clearance, 0.0040 inch.

b. Torque fitting and its bushing, 0.002-inch minimum to 0.007-inch maximum clearance. This may require trying several nuts (try them reversed) or grinding to fit.

27. Check nose gear steering installation for clearance.

Hole—strut, lower torque arm pin	0.626 in. max
Pin—lower torque arm root	0.6243 in. min
Maximum allowable clearance	0.0017 in.
Hole—bearing, lower torque arm pin	0.6255 in. max
Pin—lower torque arm root	0.6243 in. min
Maximum allowable clearance	0.0012 in.
Hole—apex lower torque arm	0.3755 in. max
Pin—knee, torque arm apex	0.3725 in. min
Maximum allowable clearance	0.0030 in.
Hole—apex upper torque arm	0.3755 in. max
Pin—knee, torque arm apex	0.3725 in. min
Maximum allowable clearance	0.0030 in.
Hole—bearing upper torque arm pin	0.6255 in. max
Pin—upper torque arm root	0.6243 in. min
Maximum allowable clearance	0.0012 in.
Hole—strut, upper torque arm	0.6260 in. max
Pin—upper torque arm root	0.6243 in. min
Maximum allowable clearance	0.0017 in.

28. Install torque arms, and adjust upper and lower torque arm root pins to point of drag; then back off nut to first castellation to align castellation with cotter pin hole. Install cotter pin.

29. Shim lower torque arm bushing for zero clearance with upper torque arm and install knee pin.

Note

CATEGORY A REPAIR. When need for strut replacement is indicated, see figure 5-13 for allowable tolerance.

30. Install serviceable nose gear strut.

Note

When an overhauled strut is installed, the new pins supplied with the strut should be used.

31. Assemble drag brace and install it according to "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4. Maintain downlock-to-stop clearance of 0.002 to 0.005 inch.

32. When replacing wheels and tires, be sure that tires have tread, inflation, and balance comparable to those of their intended mates, i.e., use matched tires only.

33. Perform normal and emergency landing gear operational check according to instructions in "F-100D Hydraulically Operated Systems," T.O. 1F-100D-2-4.

34. Install landing gear safety locks, remove airplane from jacks, and return airplane to service.

EMERGENCY LANDING GEAR RELEASE HANDLE SUPPORT REPAIR.

See figure 5-15.

LANDING GEAR REPAIR INDEXES.

MAIN LANDING GEAR REPAIR INDEX.

See figure 5-1.

NOSE LANDING GEAR REPAIR INDEX.

See figure 5-2.

TAIL SKID REPAIR INDEX.

See figure 5-3.

TYPICAL REPAIRS.

Additional sheet-metal repairs are in Section X.

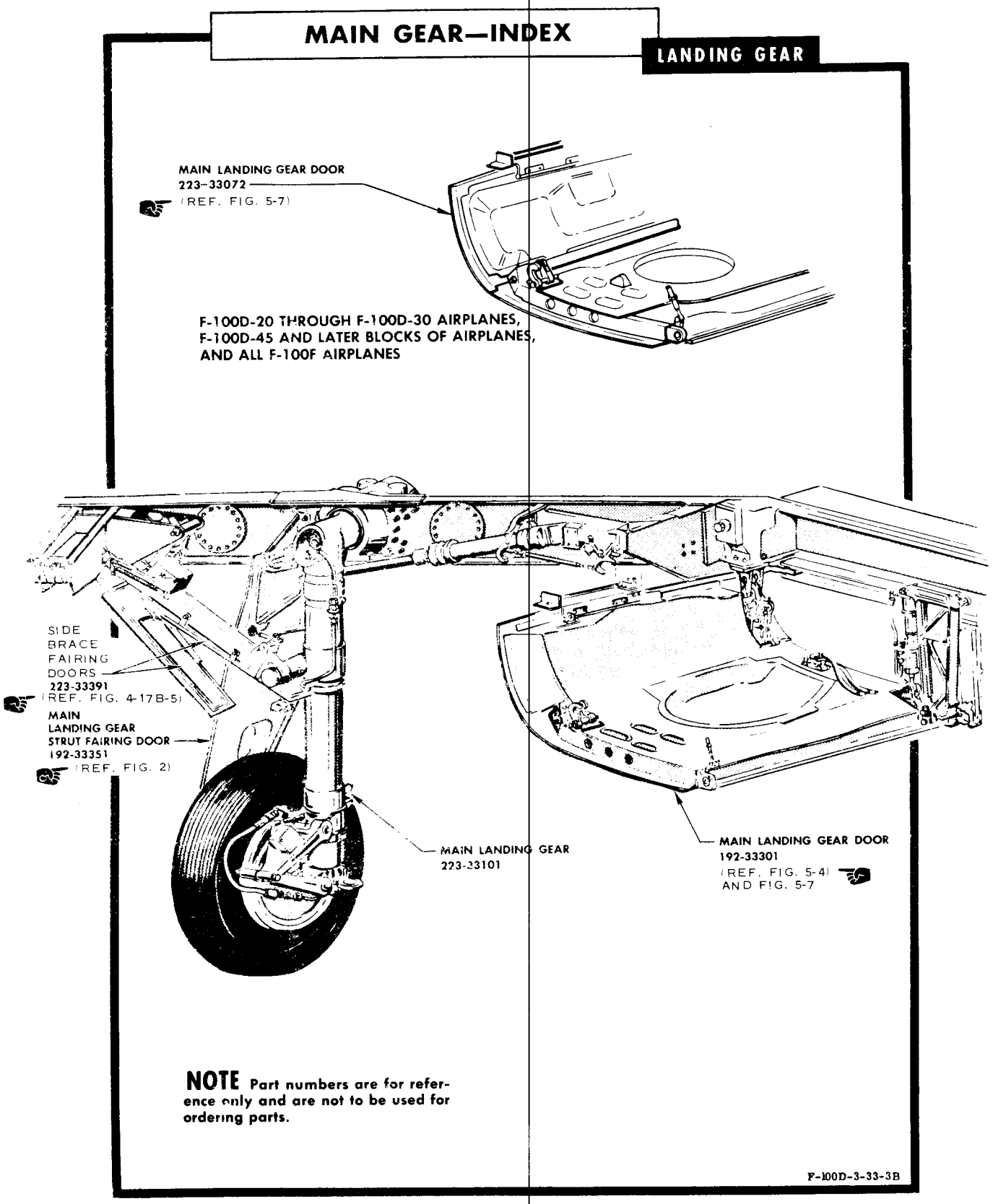
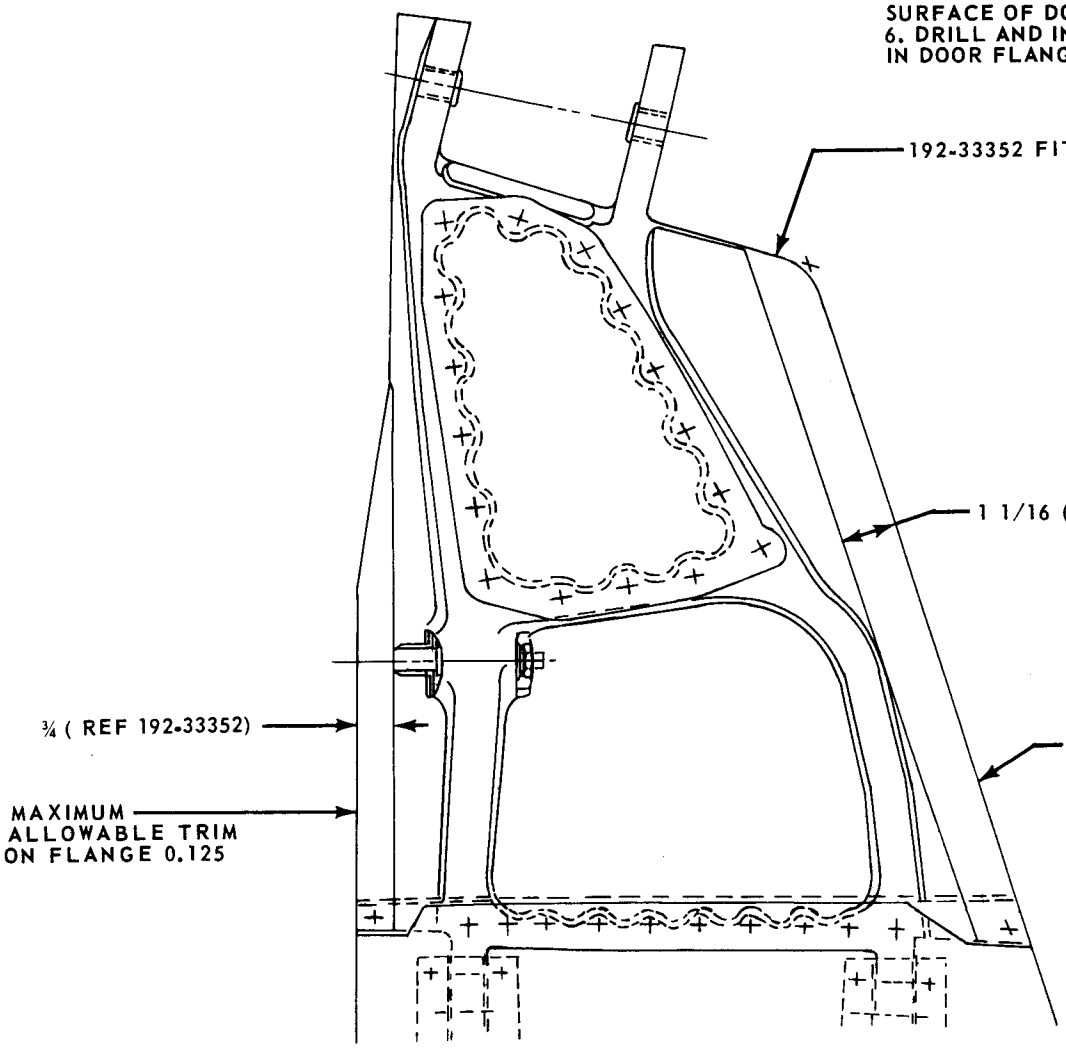
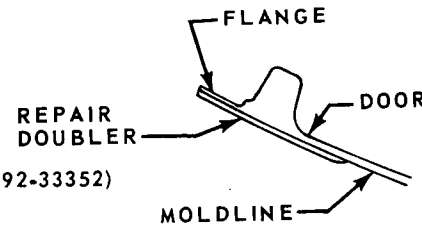


Figure 5-1. Main Landing Gear Repair Index

1. THIS REPAIR IS APPLICABLE TO F-100 DOOR ASSEMBLY
2. STOP DRILL EACH END OF CRACK WITH NO.40 (0.098) DR
3. REPAIR DOUBLER TO BE 0.080 2024-T4 CLAD ALUMINUM
4. REPAIR DOUBLER LENGTH TO BE A MINIMUM OF 2 INCH BEYOND EACH END OF CRACK,WIDTH PER SKETCH. CORNERS MAY BE ONE-FOURTH INCH RADIUS (OPTIONAL). CHAMFER EDGES OF DOUBLER 30 DEGREES AS NOTED.
5. INSTALL DOUBLER OVER CRACK ON MOLDLINE SURFACE OF DOOR.
6. DRILL AND INSTALL RIVETS AS SHOWN. RIVETS INSTALL IN DOOR FLANGE TO BE FLUSH ON BOTH SIDES.





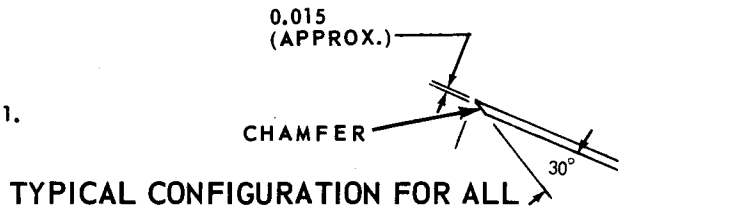
SECTION A-A

2 (MIN FROM OF C TYPIC

MAXIMUM FLANGE TRIM
VIEW LOOKING DOWN

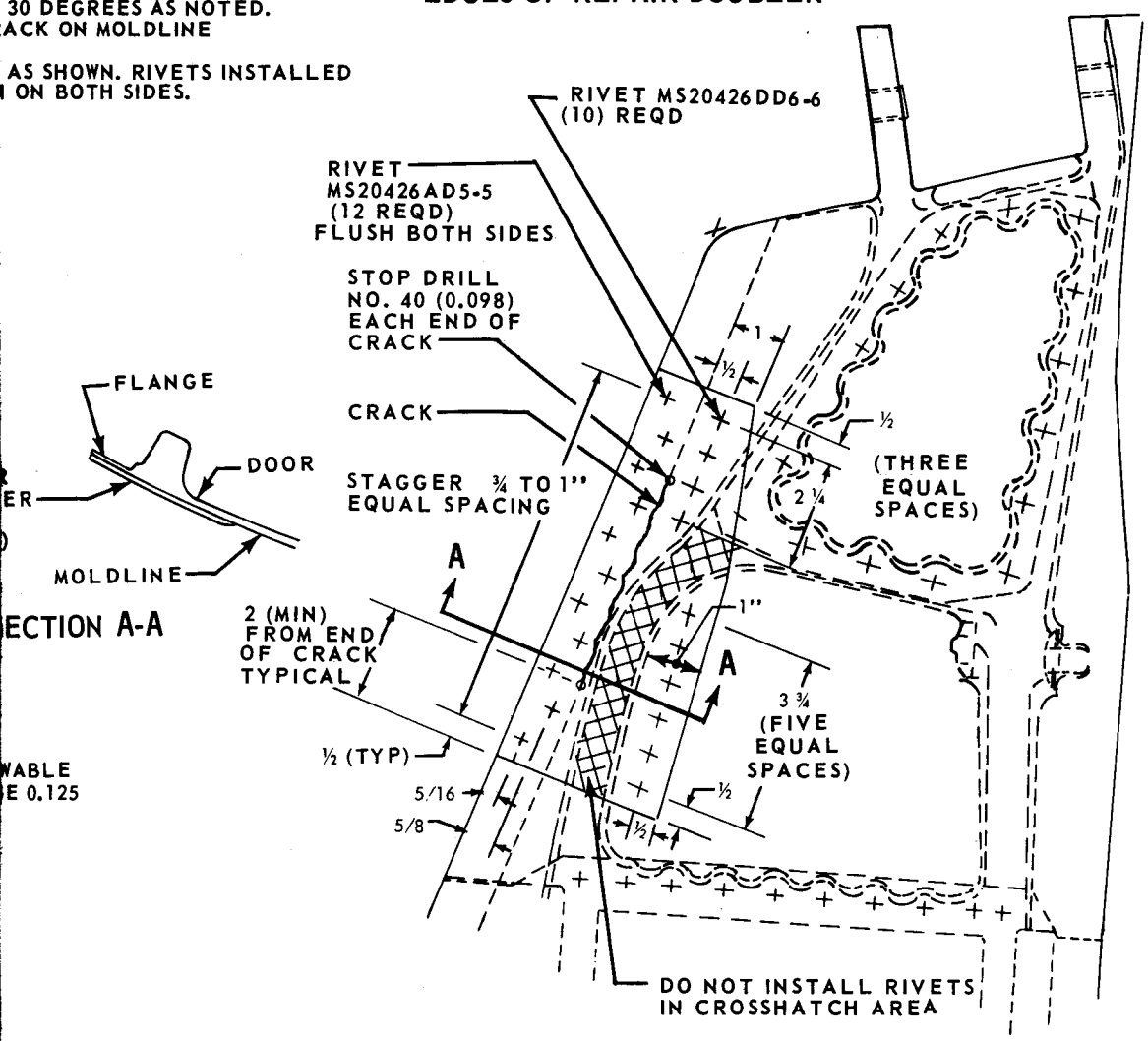
E TO F-100 DOOR ASSEMBLY 192-33351.
RACK WITH NO.40 (0.098) DRILL.
80 2024-T4 CLAD ALUMINUM SHEET.
TO BE A MINIMUM OF 2 INCHES
WIDTH PER SKETCH.
I INCH RADIUS (OPTIONAL).
30 DEGREES AS NOTED.
ACK ON MOLDLINE

AS SHOWN. RIVETS INSTALLED
I ON BOTH SIDES.



TYPICAL CONFIGURATION FOR ALL EDGES OF REPAIR DOUBLER

1. THIS REPAIR IS APPLICABLE
2. STOP DRILL END OF CRACK
3. -3 REPAIR DOUBLER TO BE 0.8
TO BE 0.125 INCH SHEET. MATERIA
STEEL SHEET SPECIFICATION MIL-
4. CORNERS OF -3 AND -5 REPAI
INCH RADIUS (OPTIONAL) EXCE
5. CHAMFER EDGES OF -5 DOUB
DRILL HOLES AND COUNTERSIN
RIVETS AT LOCATIONS NOTED.
6. REMOVE THIRTEEN EXISTING
7. ASSEMBLE DOUBLERS ON DO
8. INSTALL FASTENERS AS SHO
9. NATIONAL SCREW AND MANU
(1) CALIFORNIA.



ECTION A-A

WABLE
E 0.125

MOLDLINE OF DOOR
-5 DOUBLER (REF)

SECT

S REPAIR IS APPLICABLE TO F-100 DOOR ASSEMBLY 192-33351
P DRILL END OF CRACK WITH NO. 40(0.098) DRILL
REPAIR DOUBLER TO BE 0.80 INCH SHEET AND -5 REPAIR DOUBLER
0.125 INCH SHEET. MATERIAL TO BE 321 CORROSION RESISTANT
SHEET SPECIFICATION MIL-S-6721 OR EQUIVALENT.
RNERS OF -3 AND -5 REPAIR DOUBLERS MAY BE ONE-FOURTH
RADIUS (OPTIONAL) EXCEPT AS NOTED.
AMFER EDGES OF -5 DOUBLER 30 DEGREES PER SKETCH.
HOLES AND COUNTERSINK FOR INSTALLATION OF
S AT LOCATIONS NOTED.
MOVE THIRTEEN EXISTING FASTENERS AS NOTED
EMBLE DOUBLERS ON DOOR
TALL FASTENERS AS SHOWN
TIONAL SCREW AND MANUFACTURING CO., LOS ANGELES,
FORNIA.

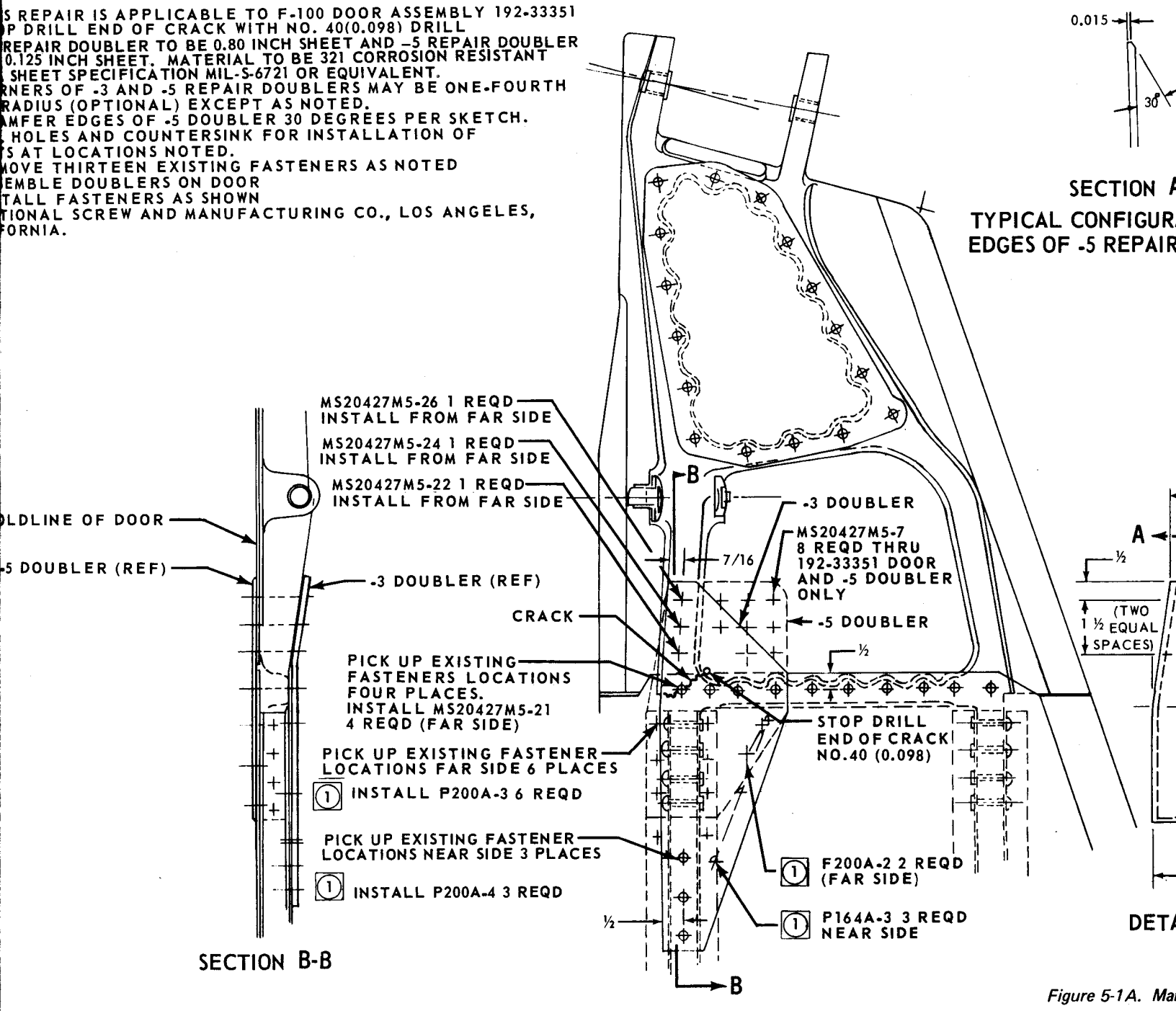


Figure 5-1A. Ma

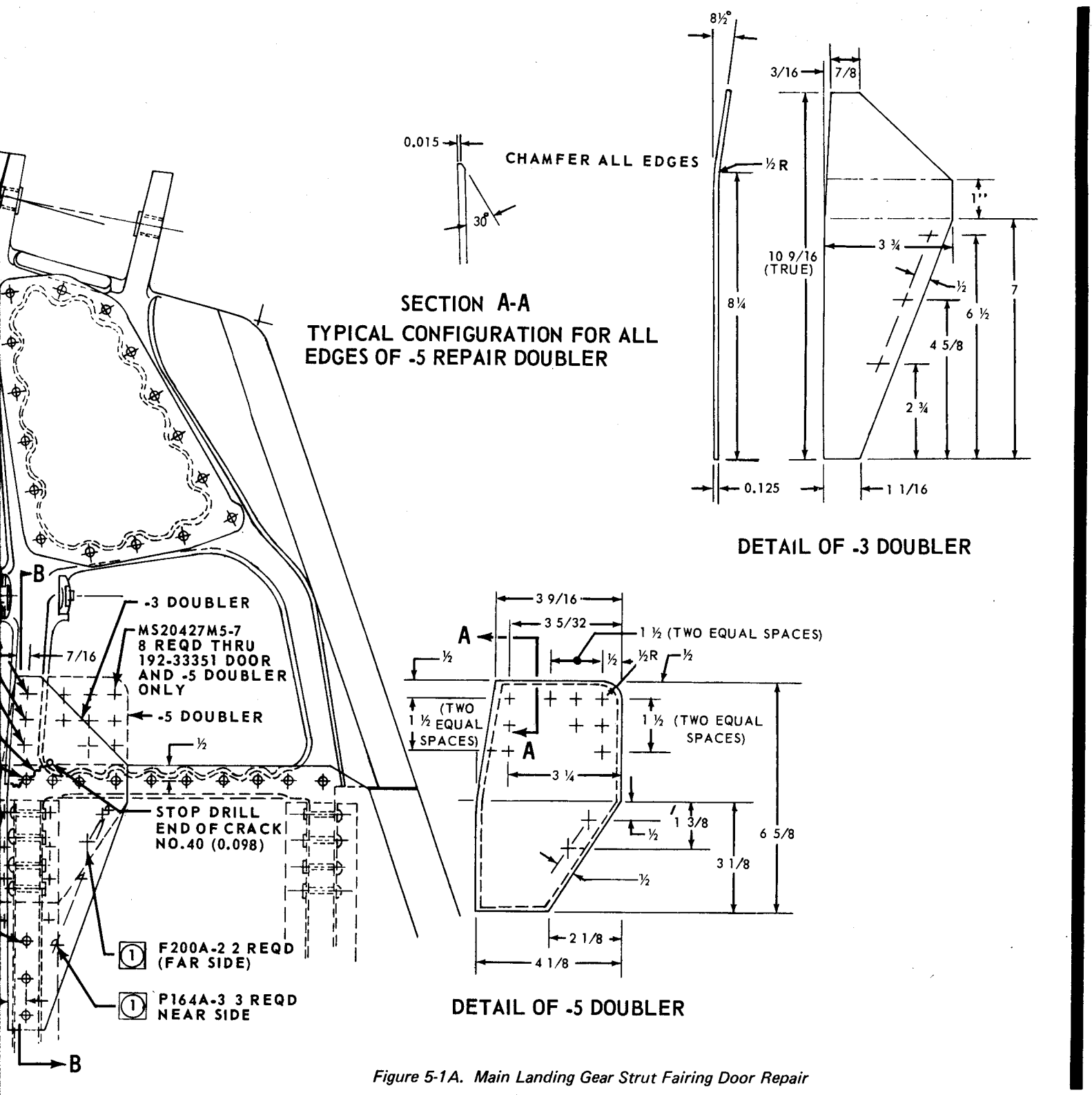
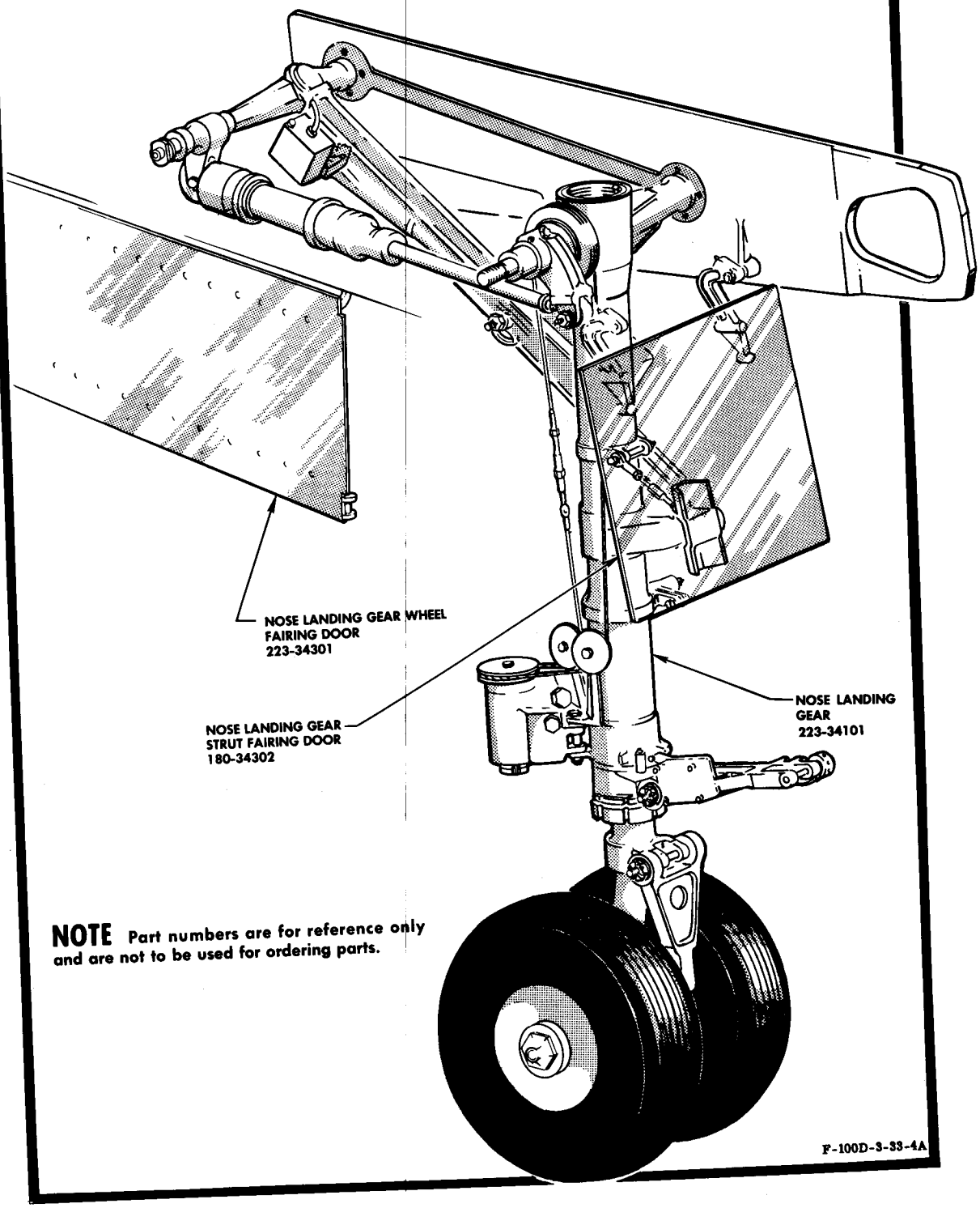


Figure 5-1A. Main Landing Gear Strut Fairing Door Repair

NOSE LANDING GEAR—INDEX

LANDING GEAR

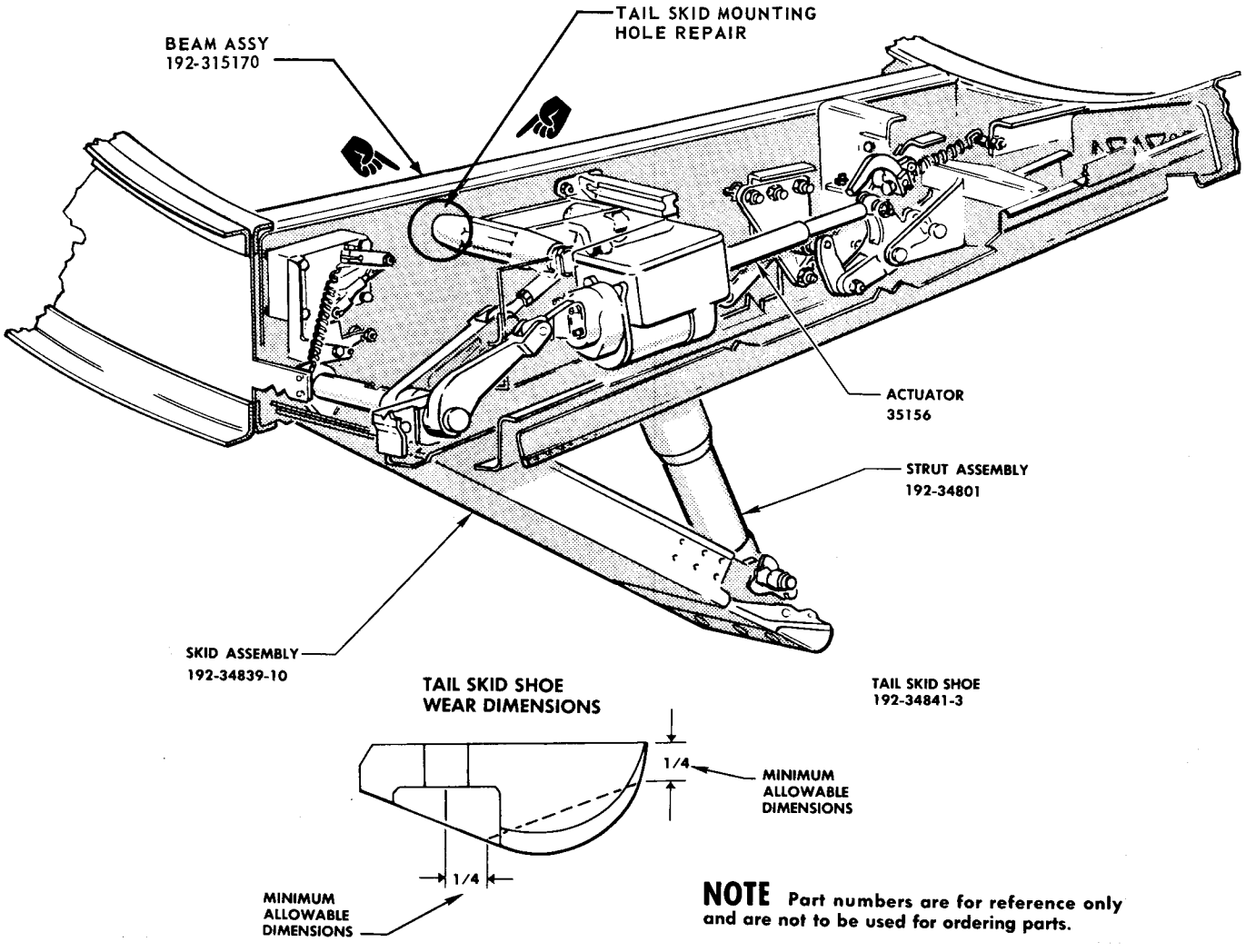


NOTE Part numbers are for reference only and are not to be used for ordering parts.

Figure 5-2. Nose Landing Gear Repair Index

TAIL SKID—INDEX

LANDING GEAR



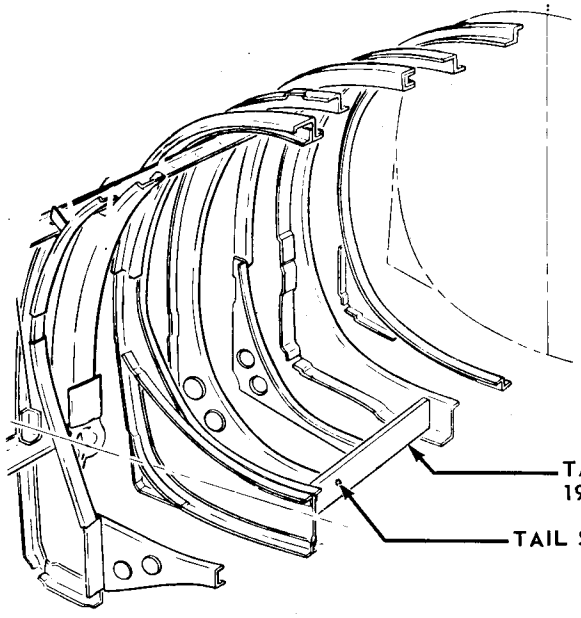
NOTE Part numbers are for reference only and are not to be used for ordering parts.

F-100D-3-33-1A

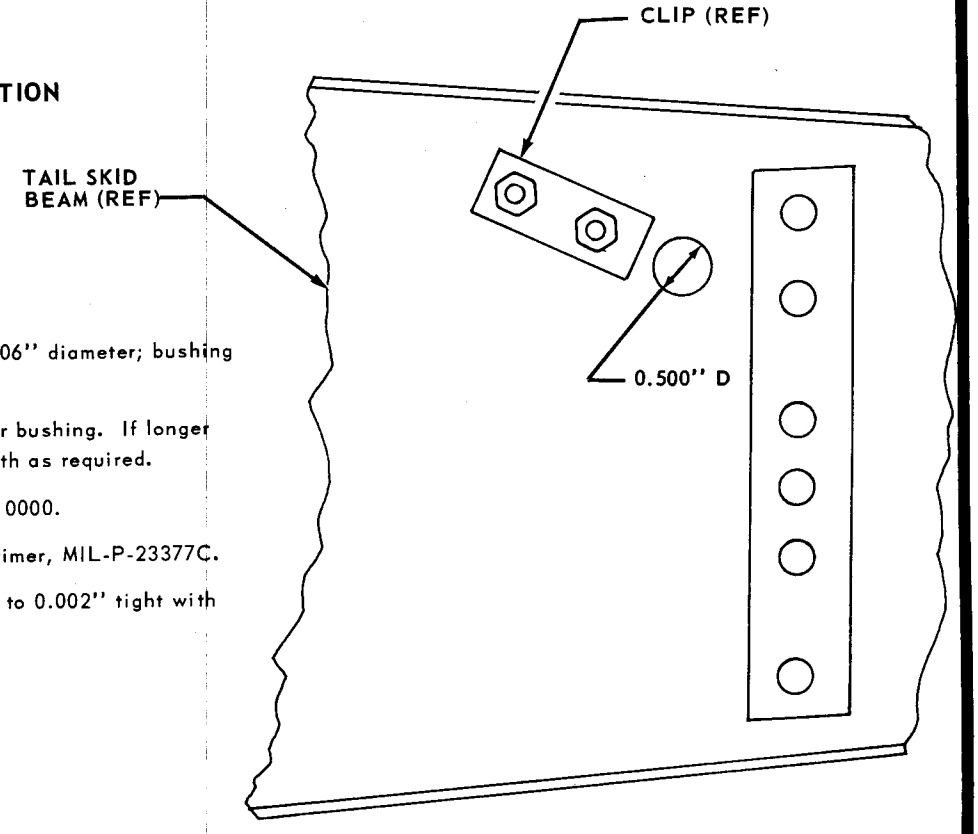
Figure 5-3. Tail Skid Repair Index

PERMANENT CATEGORY A

FUSELAGE



AFT FUSELAGE SECTION



TAIL SKID MOUNTING HOLE
VIEW LOOKING INB'D

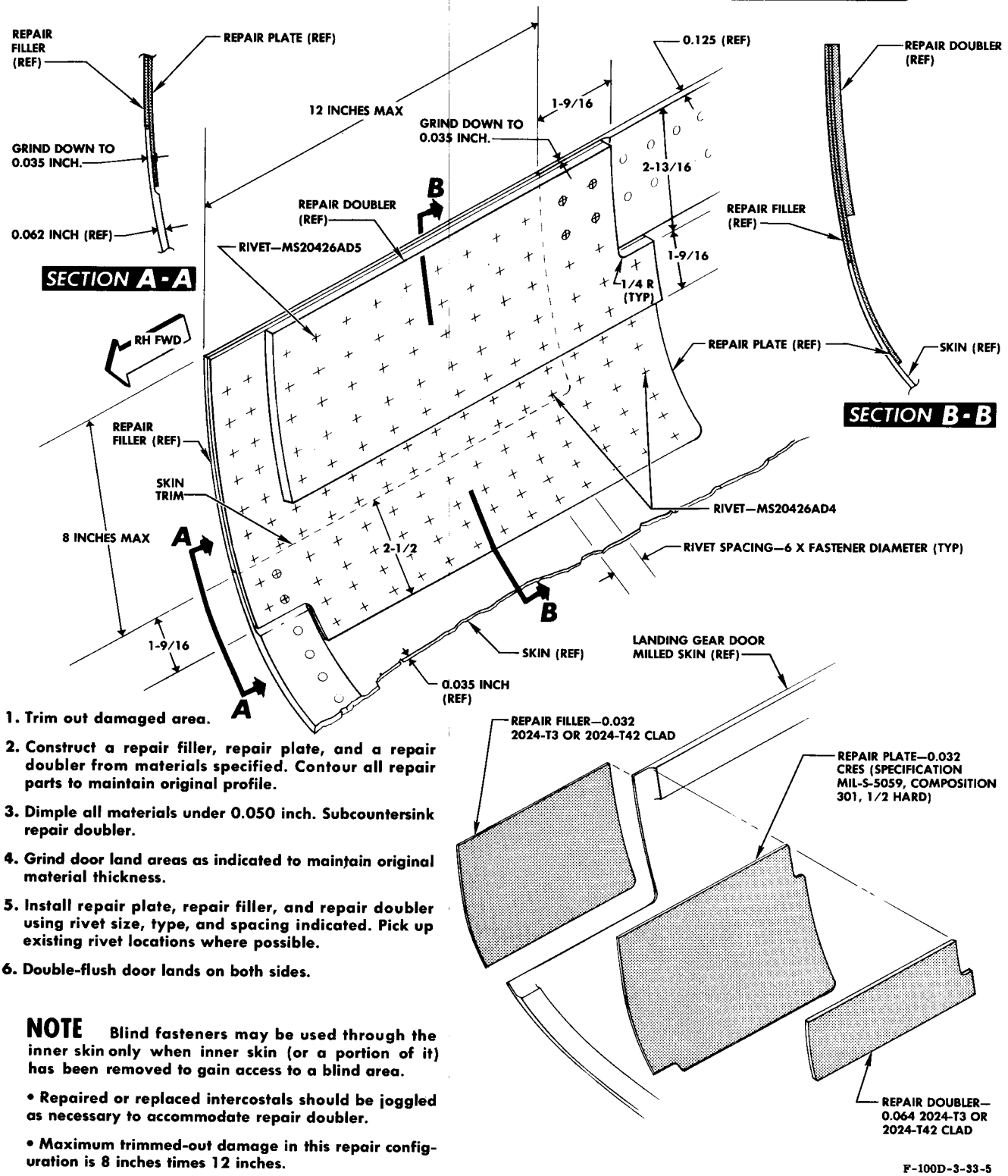
NOTES:

1. If hole diameter exceeds 0.506" diameter; bushing is required.
2. Use a 4B20MS8-006 or longer bushing. If longer bushings are used, trim length as required.
3. Ream worn hole to 0.625" ± 0000.
4. Prime bushing with epoxy primer, MIL-P-23377C.
5. Bushing pressed in 0.0015" to 0.002" tight with collar facing inb'd.

Figure 5-3A. Tail Skid Bushing Repair

PERMANENT-CATEGORY A

LANDING GEAR



1. Trim out damaged area.
2. Construct a repair filler, repair plate, and a repair doubler from materials specified. Contour all repair parts to maintain original profile.
3. Dimple all materials under 0.050 inch. Subcountersink repair doubler.
4. Grind door land areas as indicated to maintain original material thickness.
5. Install repair plate, repair filler, and repair doubler using rivet size, type, and spacing indicated. Pick up existing rivet locations where possible.
6. Double-flush door lands on both sides.

NOTE Blind fasteners may be used through the inner skin only when inner skin (or a portion of it) has been removed to gain access to a blind area.

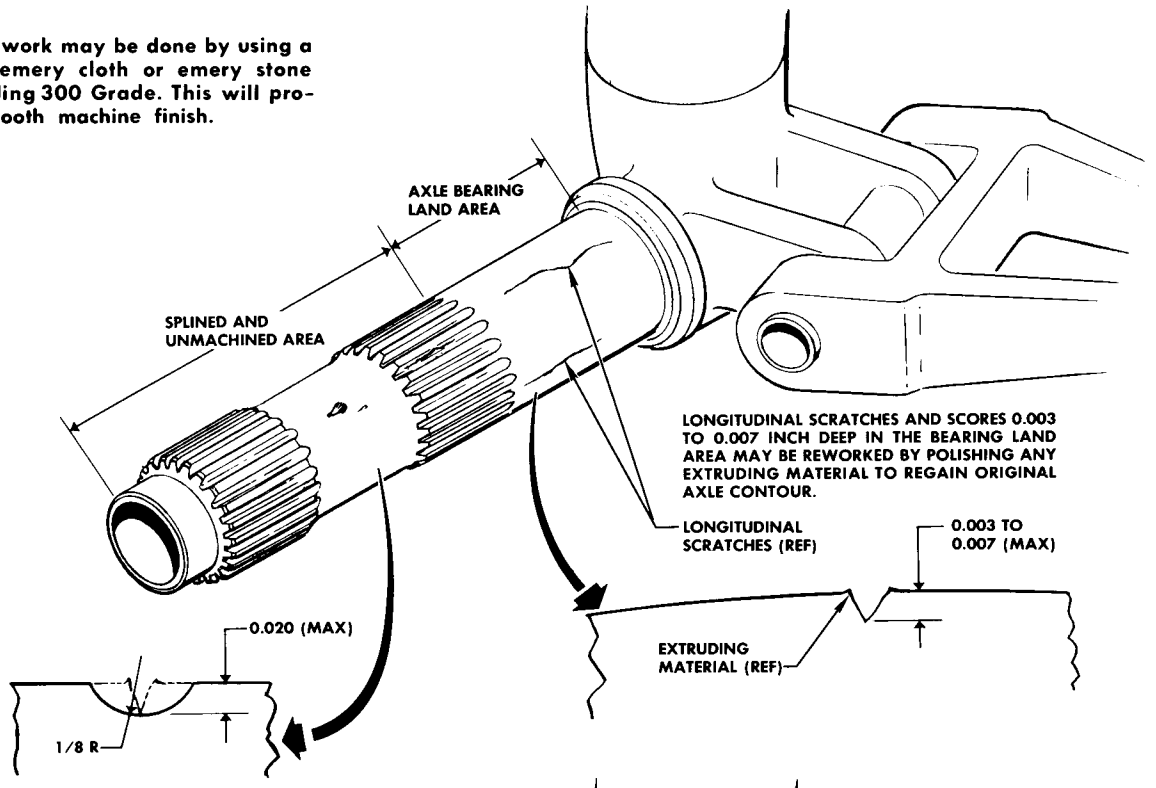
- Repaired or replaced intercostals should be joggled as necessary to accommodate repair doubler.
- Maximum trimmed-out damage in this repair configuration is 8 inches times 12 inches.

Figure 5-4. Main Landing Gear Door Repair—F-100D-1 Through F-100D-15, F-100D-35, and F-100D-40 Airplanes

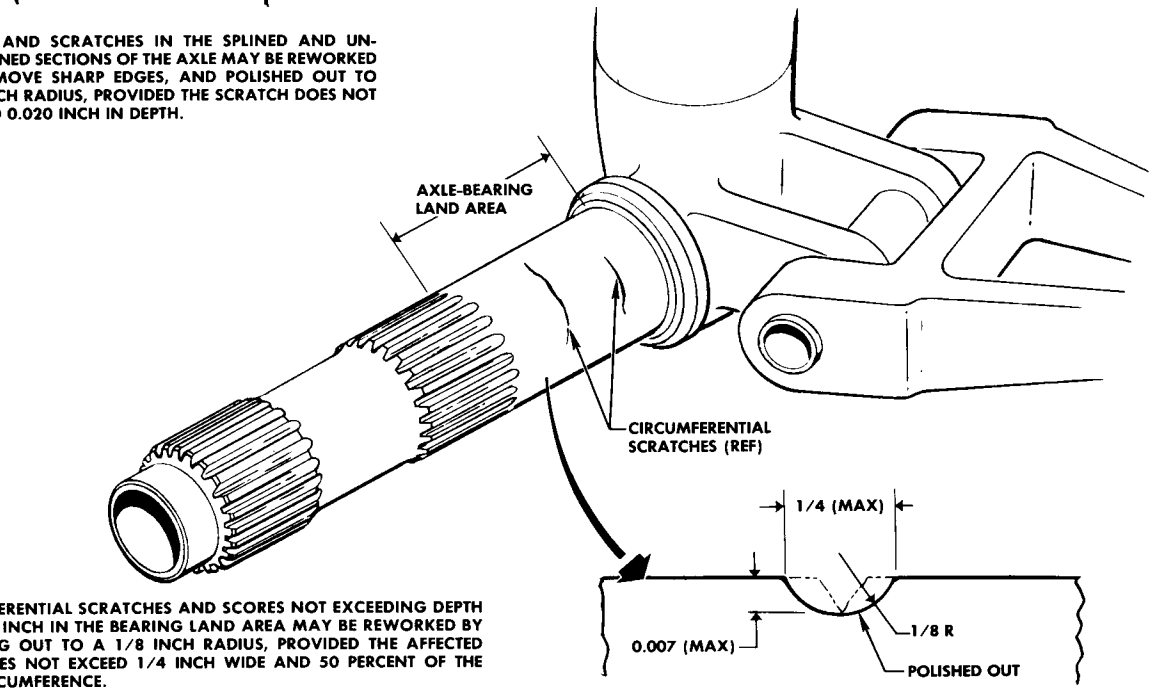
MAIN GEAR AXLE NEGLIGIBLE DAMAGE

LANDING GEAR

NOTE Rework may be done by using a standard emery cloth or emery stone not exceeding 300 Grade. This will provide a smooth machine finish.



NICKS AND SCRATCHES IN THE SPLINED AND UNMACHINED SECTIONS OF THE AXLE MAY BE REWORKED TO REMOVE SHARP EDGES, AND POLISHED OUT TO 1/8 INCH RADIUS, PROVIDED THE SCRATCH DOES NOT EXCEED 0.020 INCH IN DEPTH.

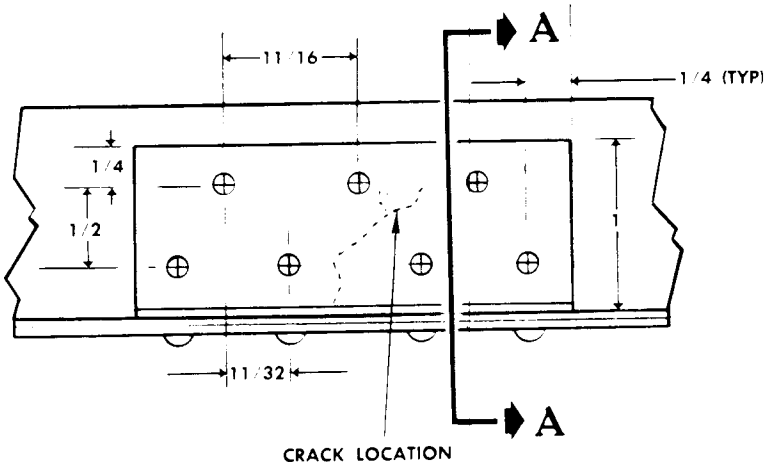


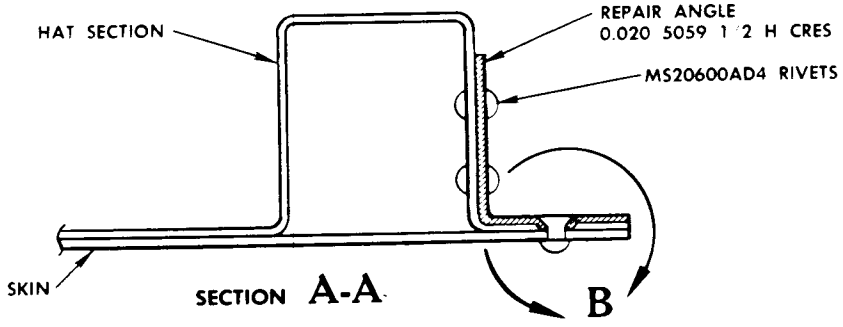
CIRCUMFERENTIAL SCRATCHES AND SCORES NOT EXCEEDING DEPTH OF 0.007 INCH IN THE BEARING LAND AREA MAY BE REWORKED BY POLISHING OUT TO A 1/8 INCH RADIUS, PROVIDED THE AFFECTED AREA DOES NOT EXCEED 1/4 INCH WIDE AND 50 PERCENT OF THE AXLE CIRCUMFERENCE.

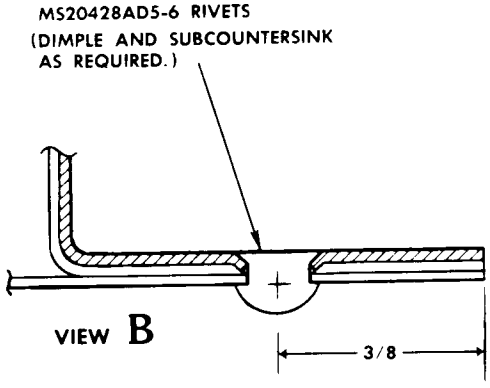
F-100D-3-33-7

Figure 5-5. Main Gear Axle Negligible Damage

PERMANENT—CATEGORY A — FUSELAGE







Repair Procedures

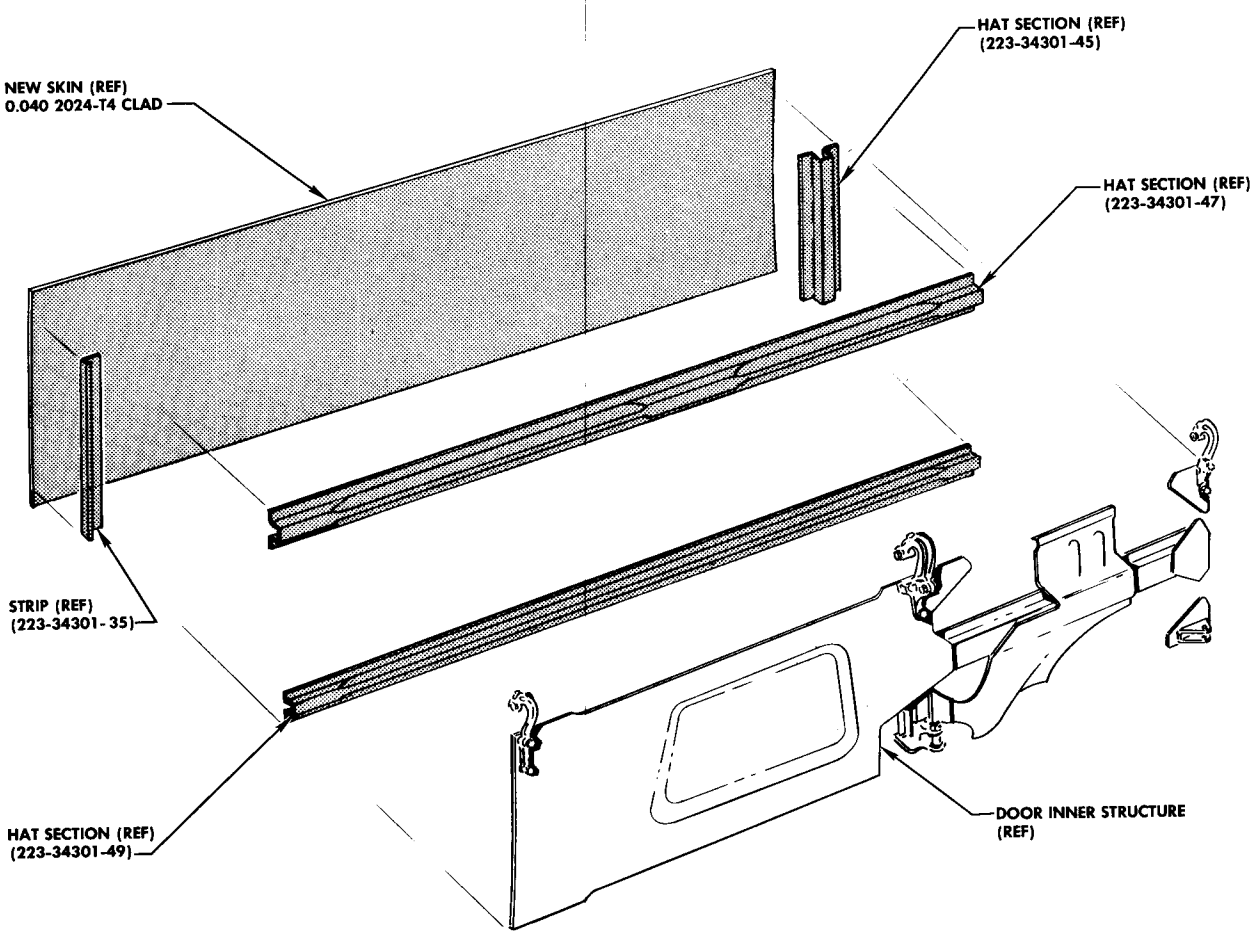
1. Stop-drill all cracks.
2. Make repair angle from 5059 1/2 H CRES. Repair angle must be of sufficient length to pick up three rivets at each end of damage.
3. Lay out and drill rivet pattern on repair angle. Dimple repair angle and subcountersink hat section as required.
4. Apply two coats of zinc-chromate primer (Specification MIL-P-8585) to all faying surfaces of repair angle and damaged hat section.
5. Fasten repair angle in place, using size and type of rivets called out.

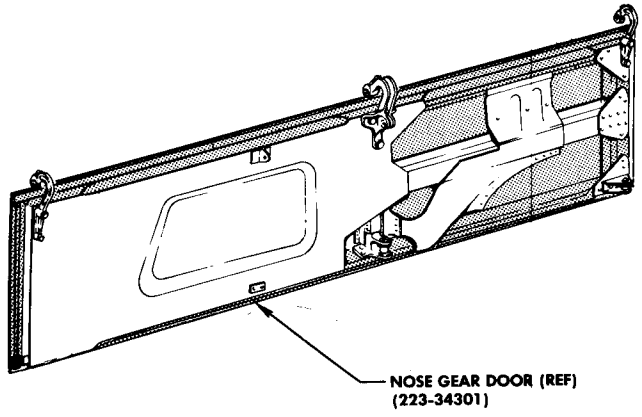
NOTE Square hat section shown, tapered hat repair similar.

- Total length of the repair splice is limited to 50 percent of the length of the damaged hat section.

Figure 5-5A. Nose Gear Door Stiffener Repair

PERMANENT CATEGORY A — **LANDING GEAR**





- When installing a new outer skin on 223-34301 door, replace -45, -47, and -49 hat sections and -35 strip with new parts.
- The skin must be seam-welded and riveted to the inner structure according to the applicable manufacturing drawings.
- Structure details are omitted for clarity.

1. Remove spot-welded skin as follows:
   a. Drill only through skin being removed.
   b. Separate parts with a thin chisel.
   c. Grind remaining nugget from parts which are to be re-used.

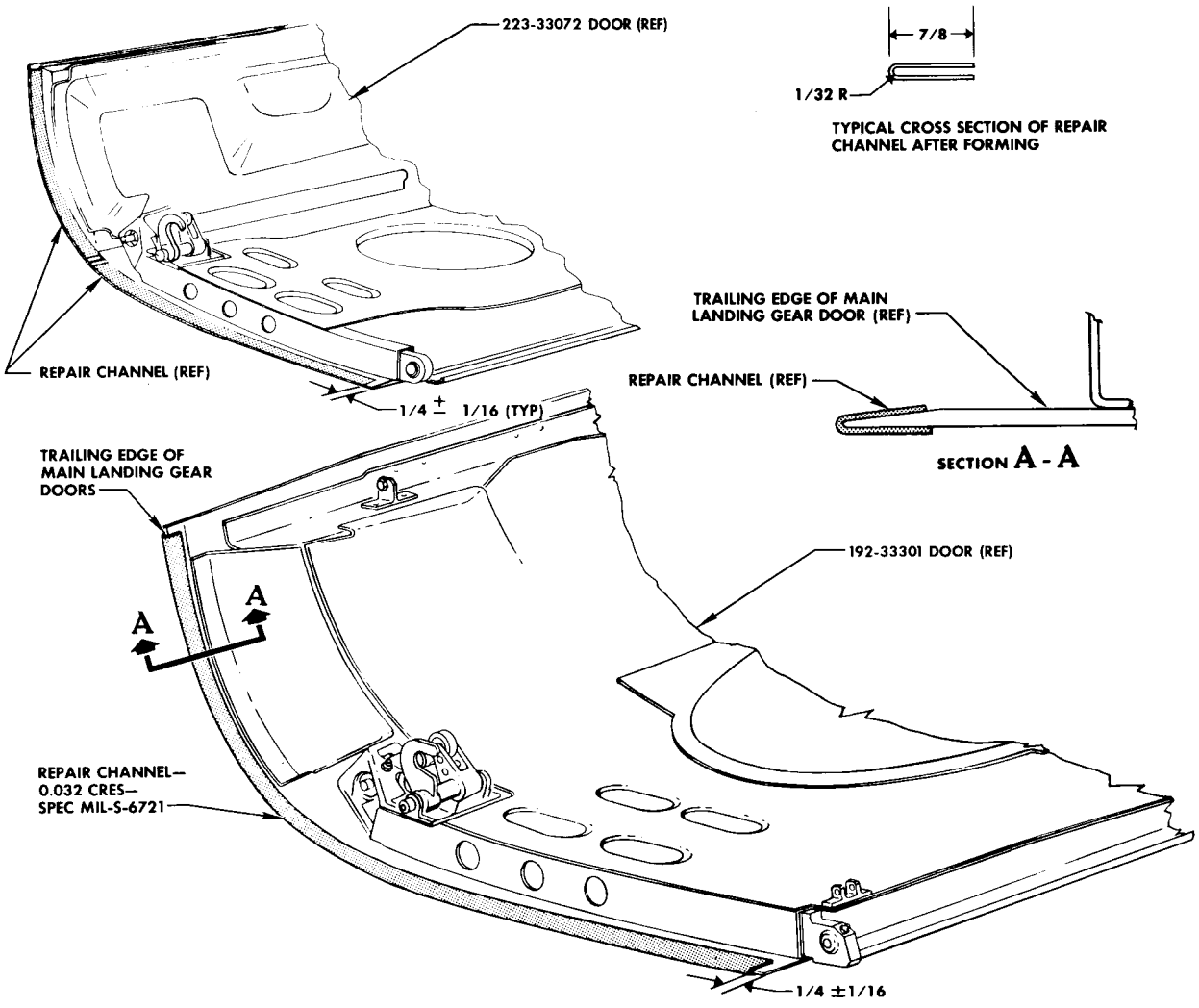
2. Remove seam-welded skin as follows:
   a. Grind weld from skin being removed.
   b. Separate parts with a thin chisel.
   c. Grind remaining weld material from parts which are to be re-used.

F-100D-3-33-6

Figure 5-6. Replacing Nose Gear Door Skin

PERMANENT—CATEGORY A

LANDING GEAR



Damage to the trailing edges of the main landing gear doors may be repaired as follows:

1 Make repair channel from Type 321 corrosion-resistant steel sheet, 0.032 × 1-13/16 × 40 inches (Specification MIL-S-6721.)

2 Form repair channel to match trailing edge of main landing gear doors by inserting strip of 0.063-inch aluminum alloy sheet between formed legs of repair channel; contour to desired shape.

3 Remove aluminum strip and start working repair channel into position by placing inboard end of repair channel 1/4 inch from edge of landing gear door skin. Gradually work complete length of repair channel into position by tapping it into place with a rawhide mallet.

4 Secure repair channel to landing gear door skins with AN426-3 rivets. Countersink both sides of repair channel, and double-flush the AN426-3 rivets. Rivet spacing should be approximately 1-1/2 inches, with an edge distance of 1/4 inch.

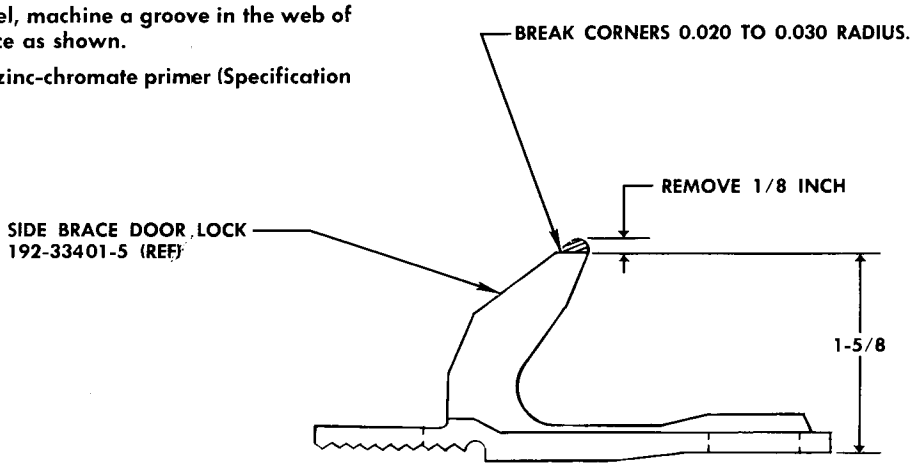
F-100D-3-33-8

Figure 5-7. Main Landing Gear Door Trailing Edge Repair

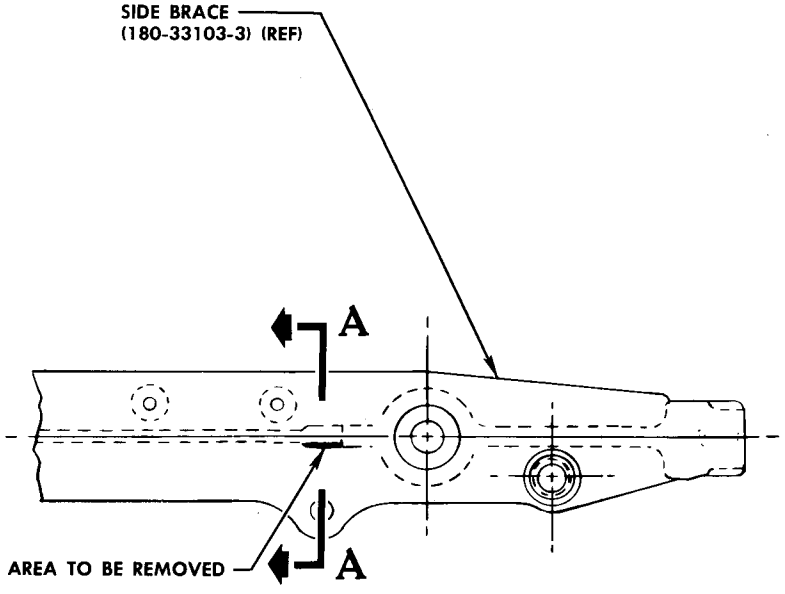
MAIN GEAR SIDE-BRACE DOOR LOCK

NOTE To correct interference between the 192-33401-5 main landing gear side-brace door lock and the 180-33103-3 side brace, rework the lock or side brace as shown.

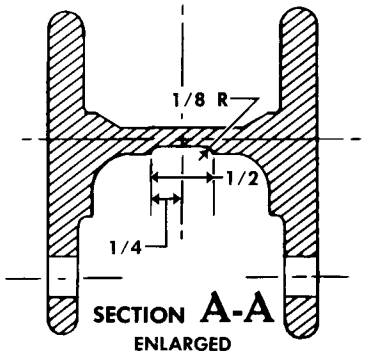
- On assembled airplanes, remove the 192-33401-5 lock from the side-brace door and rework as shown.
- At depot maintenance level, machine a groove in the web of the 180-33103-3 side brace as shown.
- Paint reworked area with zinc-chromate primer (Specification MIL-P-8585).



SIDE-BRACE DOOR LOCK



SIDE BRACE



SECTION A-A
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F-100D-3-33-11

Figure 5-8. Providing Clearance for Main Gear Side Brace Door Lock

BEARING MEASUREMENT CORRECTION CHART

LANDING GEAR

F-100 AIRPLANE MAIN LANDING GEAR
TRUNNION SUPPORT AND BEARINGS

5 AND 5-1/4 INCH				3 AND 3-1/4 INCH	
BEARING OD	SUPPORT AND INSTALLED BEARING ID		TEMP (°F)	BEARING OD	SUPPORT AND INSTALLED BEARING ID
0.0015	0.0031	SUBTRACT FROM MICROMETER READING	120	0.0010	0.0031
0.0013	0.0028		115	0.0009	0.0028
0.0012	0.0025		110	0.0008	0.0025
0.0010	0.0022		105	0.0007	0.0022
0.0009	0.0019		100	0.0006	0.0019
0.0007	0.0015		95	0.0005	0.0015
0.0006	0.0012		90	0.0004	0.0012
0.0004	0.0009		85	0.0003	0.0009
0.0003	0.0006		80	0.0002	0.0006
0.0001	0.0003		75	0.0001	0.0003
0.000	0.000		70	0.000	0.000
0.0002	0.0003	ADD TO MICROMETER READING	65	0.0001	0.0003
0.0003	0.0006		60	0.0002	0.0006
0.0005	0.0009		55	0.0003	0.0009
0.0006	0.0012		50	0.0004	0.0012
0.0007	0.0016		45	0.0005	0.0016
0.0009	0.0019		40	0.0006	0.0019
0.0010	0.0022		35	0.0007	0.0020
0.0012	0.0025		30	0.0008	0.0025
0.0013	0.0028		25	0.0009	0.0028
0.0015	0.0031		20	0.0010	0.0031

Measuring tools are adjusted to standards at 70°F.

F-100D-3-33-9

Figure 5-9. Measurement Correction Chart—Temperature/Diameter

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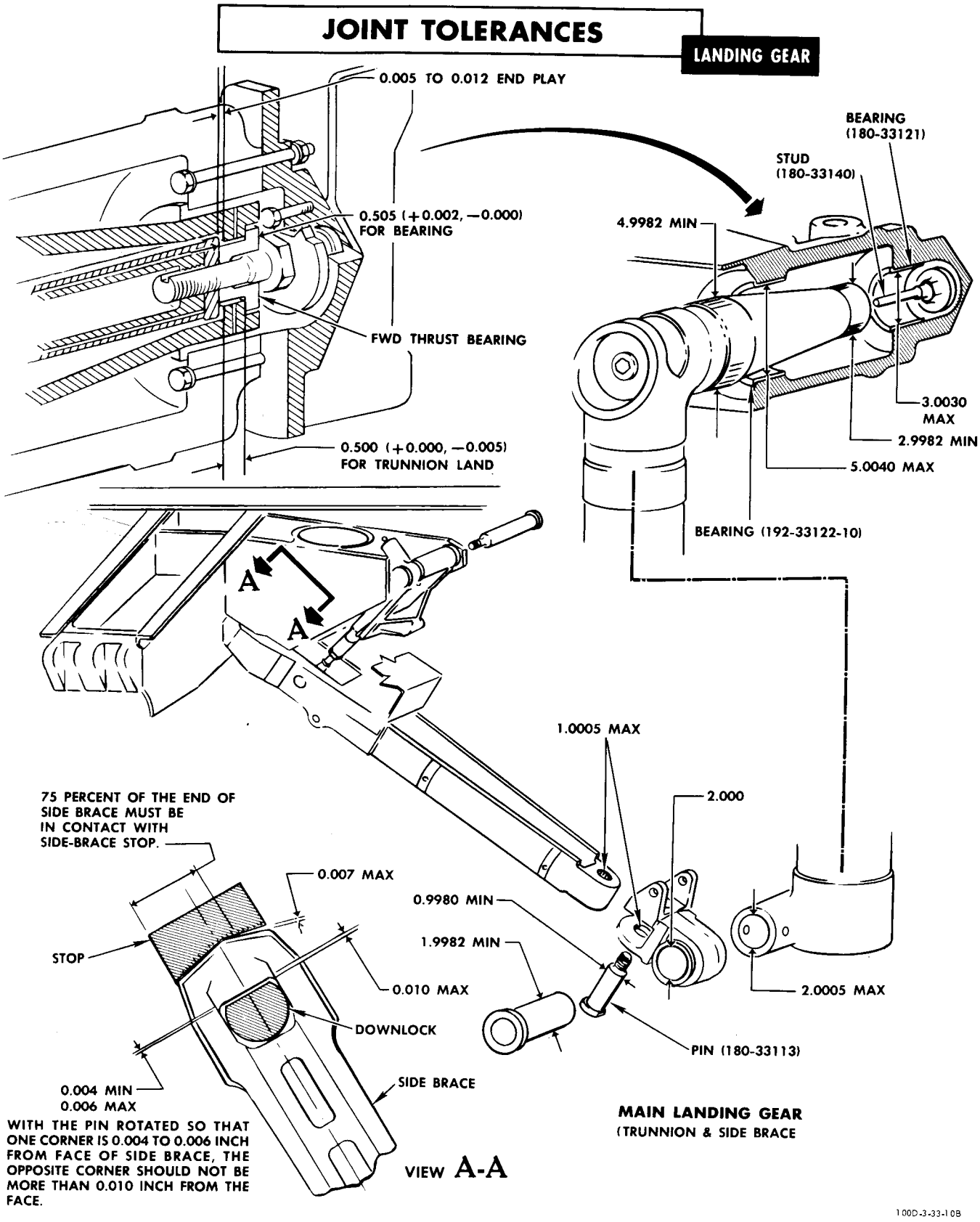


Figure 5-10. Main and Nose Landing Gear Tolerances (Sheet 1 of 2)

JOINT TOLERANCES

LANDING GEAR

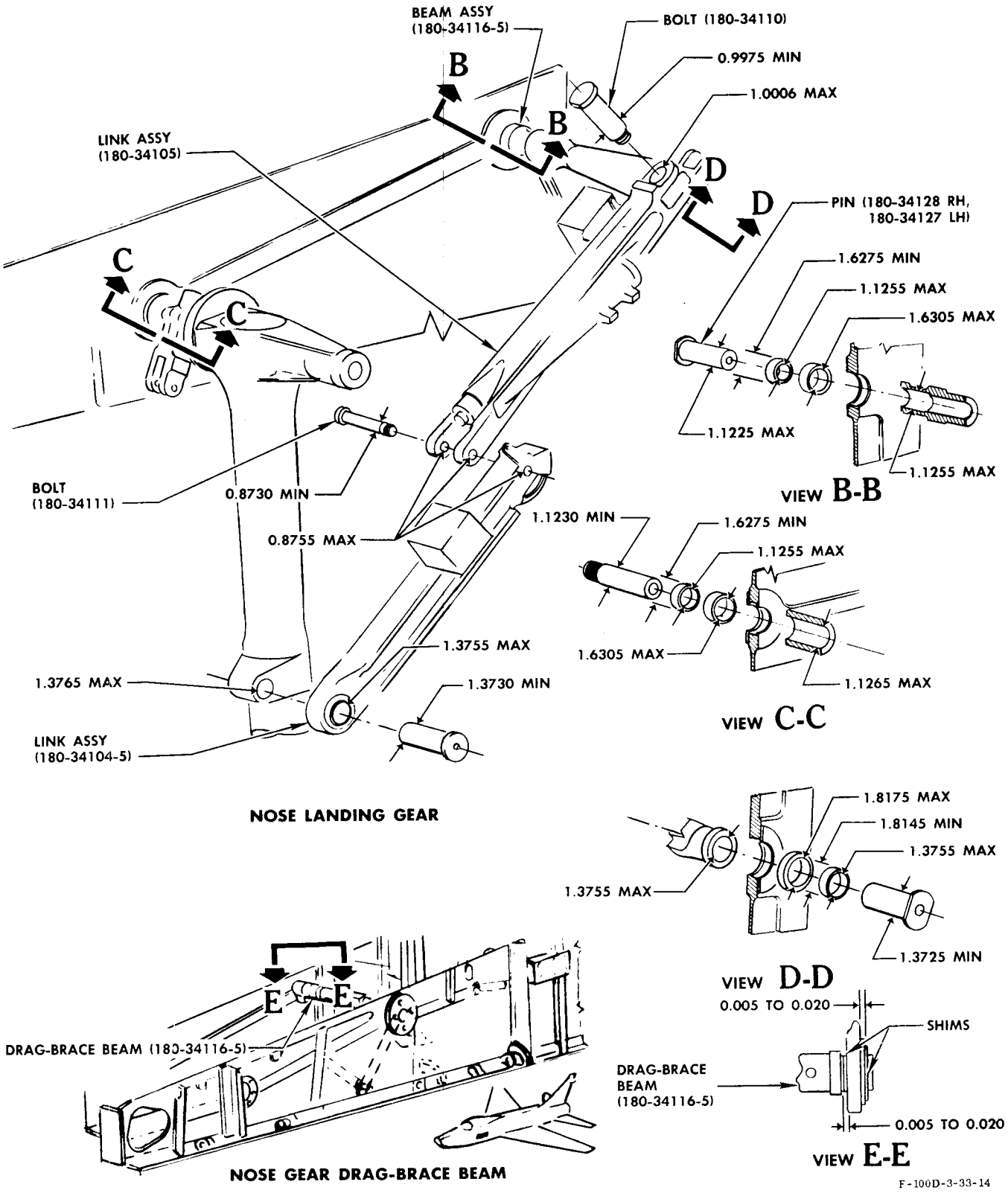
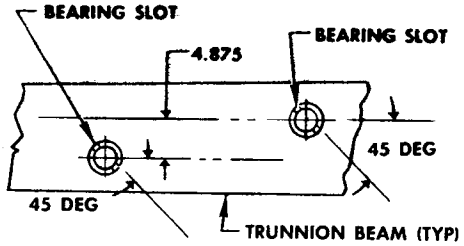
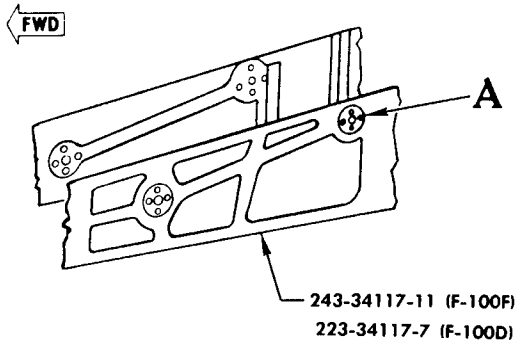


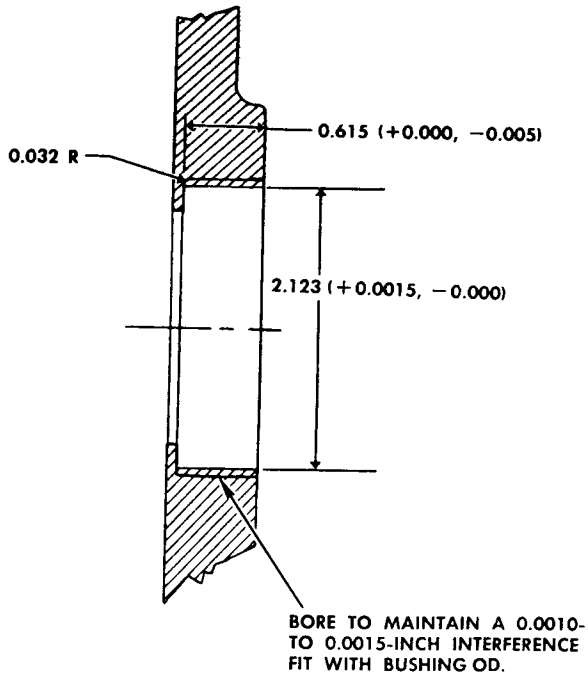
Figure 5-10. Main and Nose Landing Gear Tolerances (Sheet 2 of 2)

BEARING HOLES

LANDING GEAR







SECTION A

NOTE On F-100F Airplanes, remove 243-61125-1 ammunition box support to provide clearance for machining operations that follow.

1 Bore worn bearing seat to obtain a 0.0010- to 0.0015-inch interference fit when bearing is installed. Use cutter with a 0.020-inch radius.

2 Fabricate bushing from 303SE, Cond A, cres bar (MIL-S-7720). Bushing dimensions are:

OD — 2.190 (+0.0020, −0.0000) inches
ID — 2.123 (+0.0015, −0.0000) inches
Length — 0.610 (+0.002, −0.000) inch

Chamfer outside end of bushing 0.020 inch by 54 degrees.

3 Coat reworked bushing seat with zinc-chromate primer (MIL-P-8585), and install bushing while primer is wet. Installed bushing must be flush with, or below, surface of the seat.

4 Install bearing 120-33144 and retainer 180-34125.

NOTE Position bearing slots as shown in sketch. (Chilling the bearing outer race in dry ice will facilitate bearing installation.)

- An oversize bushing, Part No. 63C40336-5, is available if cavity bore exceeds the listed dimensions. Maintain same interference fit when installing oversize bushing.

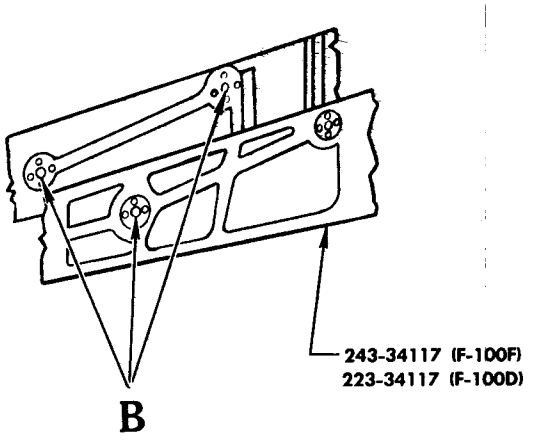
5 On F-100F Airplanes, reinstall ammunition box support 243-61125-1.

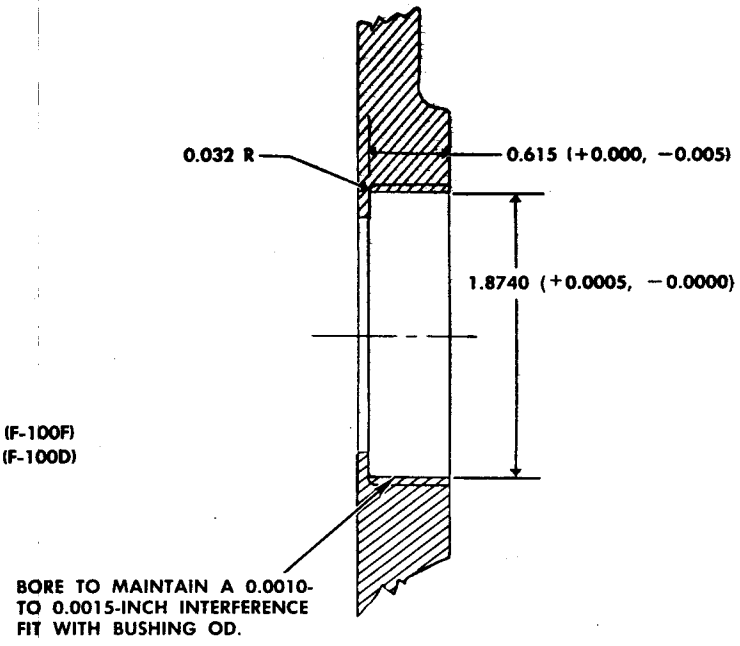
100D-3-33-12A

Figure 5-11. Reworking Bearing Holes in Nose Gear Trunnion (Sheet 1 of 2)

BEARING HOLES

LANDING GEAR





1 Bore worn bearing seats to obtain 0.0010- to 0.0015-inch interference fit when bushing is installed. Remove sharp edges (0.032-inch radius).

NOTE New holes must not exceed depth of original holes (0.615 inch). Treat bare aluminum alloy with Alodine (MIL-C-5541).

2 Fabricate bushings from 303SE, Cond A, CRES bar (MIL-S-7720). Bushing dimensions are:

OD — 1.938 (+0.002, −0.000) inches
ID — 1.874 (+0.0005, −0.000) inches
Length — 0.610 (+0.002, −0.000) inch

Chamfer both edges of bushings.

3 Check bushing and seat for a 0.001- to 0.0015-inch interference fit.

4 Coat reworked bushing seat with zinc-chromate primer (MIL-P-8585) and install bushing while primer is wet. Installed bushing must be pressed tight to the shoulder of the hole and flush, or slightly below outboard surface of trunnion plate.

5 Coat inside of installed bushing (after boring) with zinc-chromate primer and install bearings 142-33139 and retainers 180-34123 while chromate is wet.

NOTE Position bearing slots as shown in sketch (sheet 1).

- Chilling bearing outer race in dry ice will facilitate bearing installation.
- An oversize bushing, Part No. 63C40336-3, is available if cavity bore exceeds the listed dimensions. Maintain same interference fit when installing oversize bushing.

100D-3-33-13A

Figure 5-11. Reworking Bearing Holes in Nose Gear Trunnion (Sheet 2 of 2)

PERMANENT—CATEGORY A LANDING GEAR

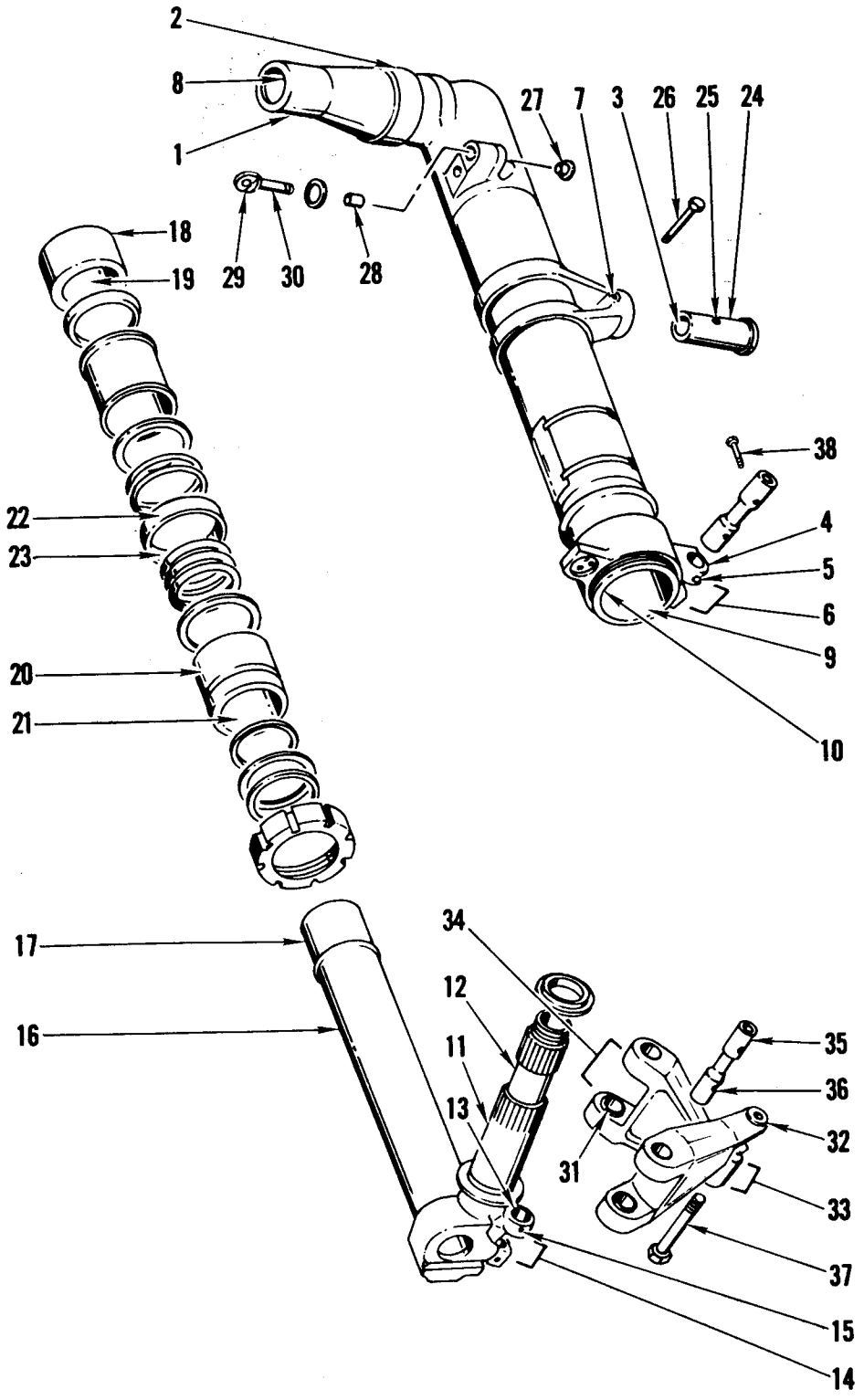


Figure 5-12. Wear Tolerances—Main Landing Gear Strut—Part No. 62J31600-1 and -2 (Sheet 1 of 2)

Index No.	Original Dimensions	Wear Limit (O.D.)	Wear Limit (I.D.)	Nomenclature	Part No.	Req'd RMS Finish
1 –	2.9987/2.9982	2.9977		Fwd Trunnion Bearing	62K31602	40
2 –	4.9987/4.9982	4.9966		Aft Trunnion Bearing		40
3 –	2.0005/1.9990		2.0012	Side Brace Socket		100
4 –	1.1255/1.1240		1.1260	Torque Arm Lugs		
5 –	3.6845/3.6815			T.A. Pin Cross Bolt		
6 –	0.261/0.255	0.2550		Across T.A. Lugs		
7 –	0.376/0.375	0.3745		S.B. Pin Cross Bolt		
8 –	0.500/0.495	0.4940		Fwd Trun Attach Land		
9 –	4.939/4.937		4.9390	Lower Bearing Bore		
10 –	4.751/4.749		4.754	Upper Bearing Bore		20
11 –	2.8748/2.8740	2.8735		Inner Axle Bear Land	62J31621	63
12 –	2.8123/2.8115	2.8110		Outer Axle Bear Land		63
13 –	1.1255/1.1240		1.1260	T.A. Pin Lug Hole		
14 –	3.6845/3.6815			Across T.A. Boss		
15 –	0.261/0.255			T.A. Bolt Cross Pin Hole		
16 –	4.251/4.249	4.2490		Piston Diameter		16
17 –	4.062/4.061	4.0610		Upper Bearing Land		
* 18 –	4.745/4.743	4.7420		Piston Head (O.D.)	62C31638	100
19 –	4.065/4.063		4.0650	Piston Head (I.D.)		
20 –	4.9340/4.9314	4.9314		L. Bearing (O.D.)	62C31632	
21 –	4.257/4.254		4.2580	L. Bearing (I.D.)		125
22 –	4.935/4.933	4.9300		Packing Adapter (O.D.)	62D31634	
23 –	4.264/4.259		4.2660	Packing Adapter (I.D.)		
24 –	1.9987/1.9982	1.9982		Side Brace Pin	62C31607	40
25 –	0.376/0.375		0.3765	S.B. Pin Cross Pin Hole		
26 –	0.373/0.370	0.3700		S.B. Pin Cross Bolt	62C31608	
27 –	0.8755/0.8740		0.8765	Retract Pin Bushing	62C31605	40
28 –	1.0005/0.9990		0.9980	Retract Pin Bushing	62C31606	40
29 –	0.9985/0.9970	0.9970		Retract Pin Bear Land	62C31644	32
30 –	0.8735/0.8720	0.8720		Retract Pin Bear Land		
31 –	1.1255/1.1240		1.1260	Torque Arm Bushing	62C31661	40
32 –	0.7505/0.7490		0.7510	T.A. Apex Bushing	62C31662	40
33 –	2.252/2.248			Across Apex Lug	62D31658	
34 –	3.6895/3.6865			Between Bushing Flanges		
35 –	1.1235/1.1230	1.1230		T.A. Pin Bearing Lands	62C31666(Pin)	40
36 –	0.261/0.255		0.261	Cross Pin Hole		
37 –	0.748/0.747	0.747		T.A. Apex Bolt	62C31663	40
38 –	0.249/0.246	0.246		T.A. Pin Cross Bolt	AN4-14A	

* Allowable wear limit providing applicable cylinder is within original manufacturer's technical order tolerances. Dimensions in the above chart are before cadmium plating where applicable, therefore, larger or smaller measurements, depending whether an O.D. or an I.D. is measured respectively, are allowable when surfaces being measured have been plated.

100D-3-33-18

Figure 5-12. Wear Tolerances—Main Landing Gear Strut—Part No. 62J31600-1 and -2 (Sheet 2 of 2)

PERMANENT—CATEGORY A **LANDING GEAR**

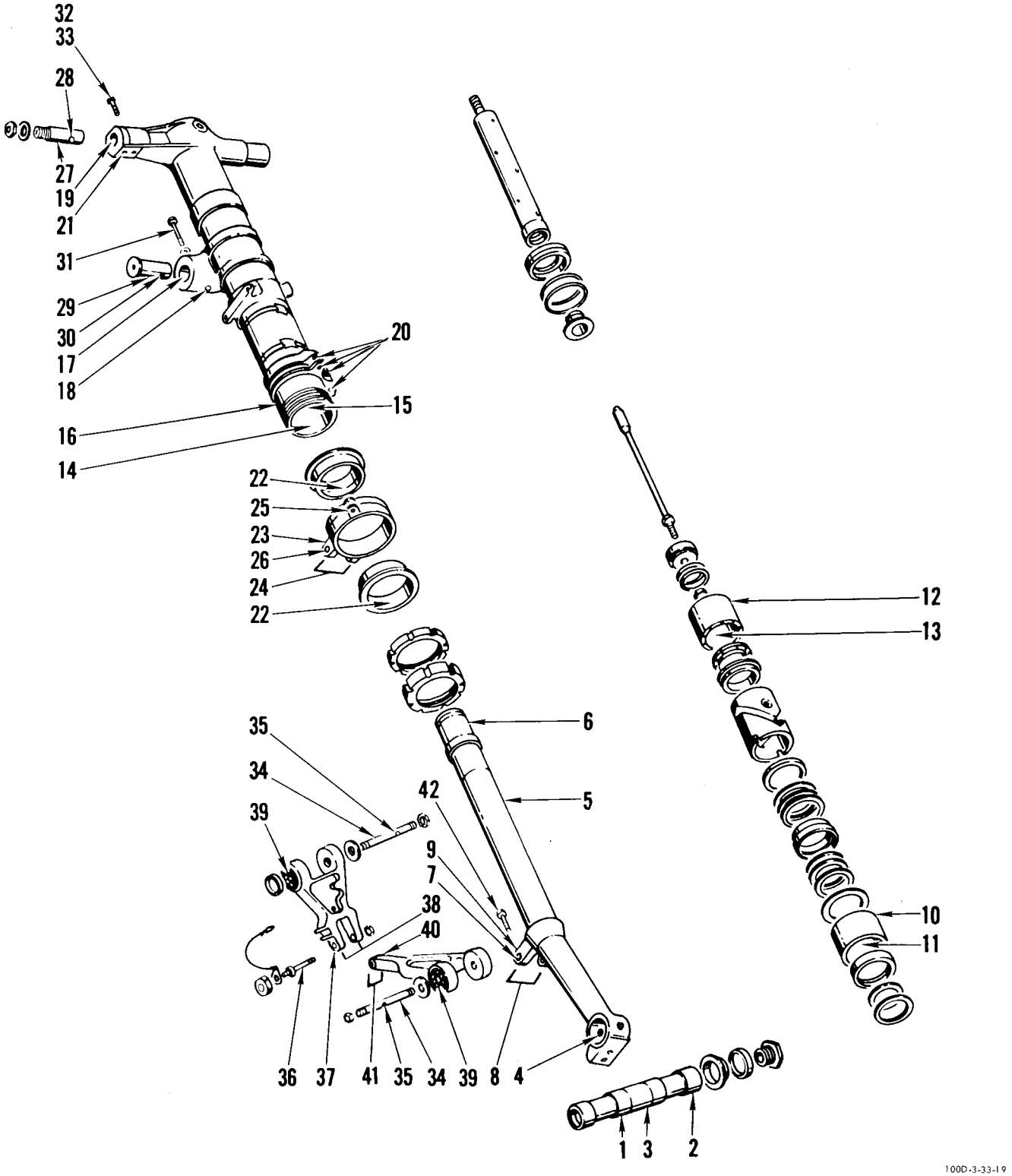


Figure 5-13. Wear Tolerances—Nose Gear Strut—Part No. 62J31440 (Sheet 1 of 2)

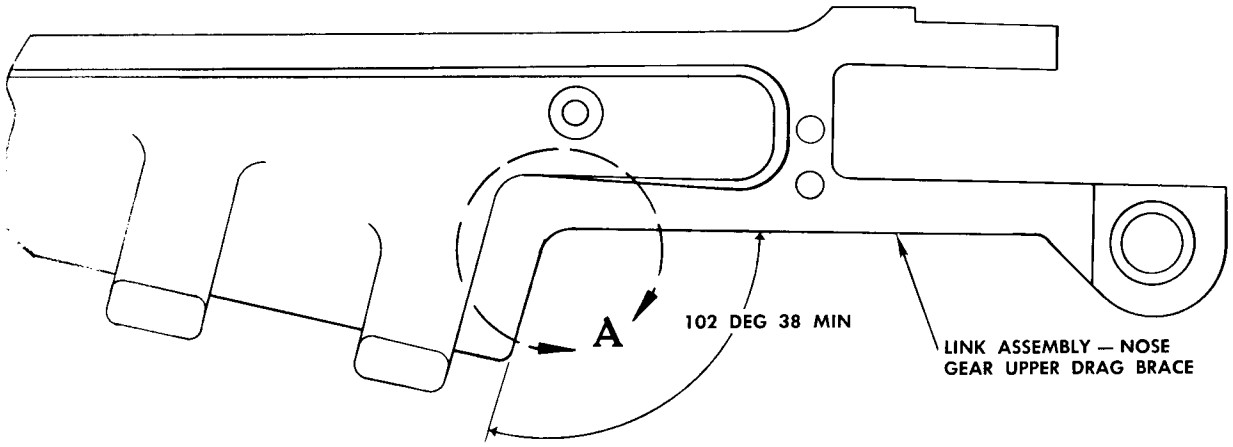
Index No.	Original Dimensions	Wear Limit (O.D.)	Wear Limit (I.D.)	Nomenclature	Part Number
1	1.9998/1.9993	1.9993		Inner Bearing Land	62C31505
2	1.8123/1.8118	1.8118		Outer Bearing Land	
3	2.0651/2.0644	2.0644		Socket Land	
4	2.0636/2.0625		2.0636	Axle Socket	62J31403
5	3.001 2.999	2.999		Piston Diameter	
6	2.765 2.7635	2.7635		Upper Bearing Land	
7	0.625/0.624		0.6255	Torque Arm Pin Boss	
8	2.760 2.740			Across Boss	
9	0.2530/0.2515		0.2530	Torque Pin Cross Boss	
10	3.6220/3.6204	3.6204		Lower Bearing O.D.	62C31488
* 11	3.005 3.003		3.006	Lower Bearing I.D.	
* 12	3.434 3.432	3.430		Upper Bearing O.D.	62C31500
13	2.770 2.768		2.770	Upper Bearing I.D.	
14	3.627 3.625		3.627	Lower Bearing Bore	62J31442
15	3.439 3.437		3.442	Upper Bearing Bore	
16	4.065 4.064	4.064		Steer Collar B.A.	
17	1.3760/1.3750		1.3765	Drag Brace Socket	
18	0.3130/0.3115		0.3130	Drag Brace C.P. Boss	
19	1.1255/1.1245		1.1260	Trunnion Pin Boss	
20	0.438 0.437		0.438	Steering Attach Lug	
21	0.2515 0.2500		0.2515	Trunnion Cross Pin	
22	4.0665 4.0660		4.0675	S.C. Bearings I.D.	62C31476
23	0.625/0.624		0.6255	T.A. Pin Boss I.D.	62D31472
24	2.760/2.740			T.A. Boss (Across)	
25	0.313/0.312		0.314	Steering Unit Attach	
26	0.2530 0.2515		0.2530	T.A. Pin Cross Bolt Hole	
27	1.1240/1.1235	1.1230		Trunnion Pin (O.D.)	62C31453
28	0.265/0.259		0.265	T.P. Cross Bolt Hole	
29	1.3745/1.3740	1.3755		D.B. Pin (O.D.)	62C31463
30	0.320/0.314		0.320	D.B. Cross Pin Hole	
31	0.3117/0.3102	0.3102		D.B. Cross Bolt	MS20005-52
32	0.2492/0.2477	0.2477		L. Trunnion Cross Bolt	MS20004-40
33	0.2492/0.2477	0.2477		R. Trunnion Cross Bolt	MS20004-34
34	0.6239/0.6233	0.6233		Torque Arm Pin	62C31535
35	0.2530/0.2515		0.2530	Cross Pin Hole	
36	0.3730/0.3725	0.3725		T.A. Apex Bolt	62C31530
37	0.3755/0.3740		0.3760	T.A. Apex Bush (I.D.)	62C31519/62C31520
38	1.126/1.124			T.A. Apex Lugs (Across)	62D31517
39	0.6255/0.6250		0.6255	T.A. Bearing Cone (I.D.)	03062
40	0.3755/0.3740		0.3760	T.A. Bushing Apex (I.D.)	62C31527
41	1.122/1.120			Apex Bushings (Across)	62C31525
42	0.249/0.246			T.A. Pin Cross Bolts	AN4-15A & AN4-25A

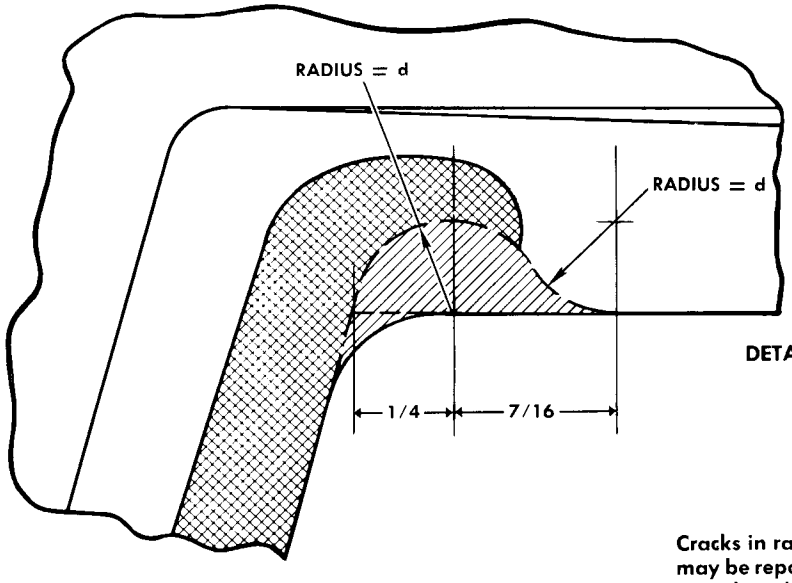
* Allowable wear limit providing applicable cylinder is within original manufacturer's technical order tolerances. Dimensions in the above chart are before cadmium plating where applicable, therefore, larger or smaller measurements, depending whether on O.D. or I.D. is measured respectively, are allowable when surfaces being measured have been plated.

100D-3-33-20

Figure 5-13. Wear Tolerances—Nose Gear Strut—Part No. 62J31440 (Sheet 2 of 2)

PERMANENT—CATEGORY A **FUSELAGE**





DETAIL A

MAXIMUM MATERIAL TO BE REMOVED

UNDER NO CIRCUMSTANCES IS MATERIAL TO BE REMOVED FROM THIS AREA.

NOTE

Cracks in radii area of nose gear upper drag brace may be repaired, if the maximum material removal (d) is less than 0.250 inch in depth.

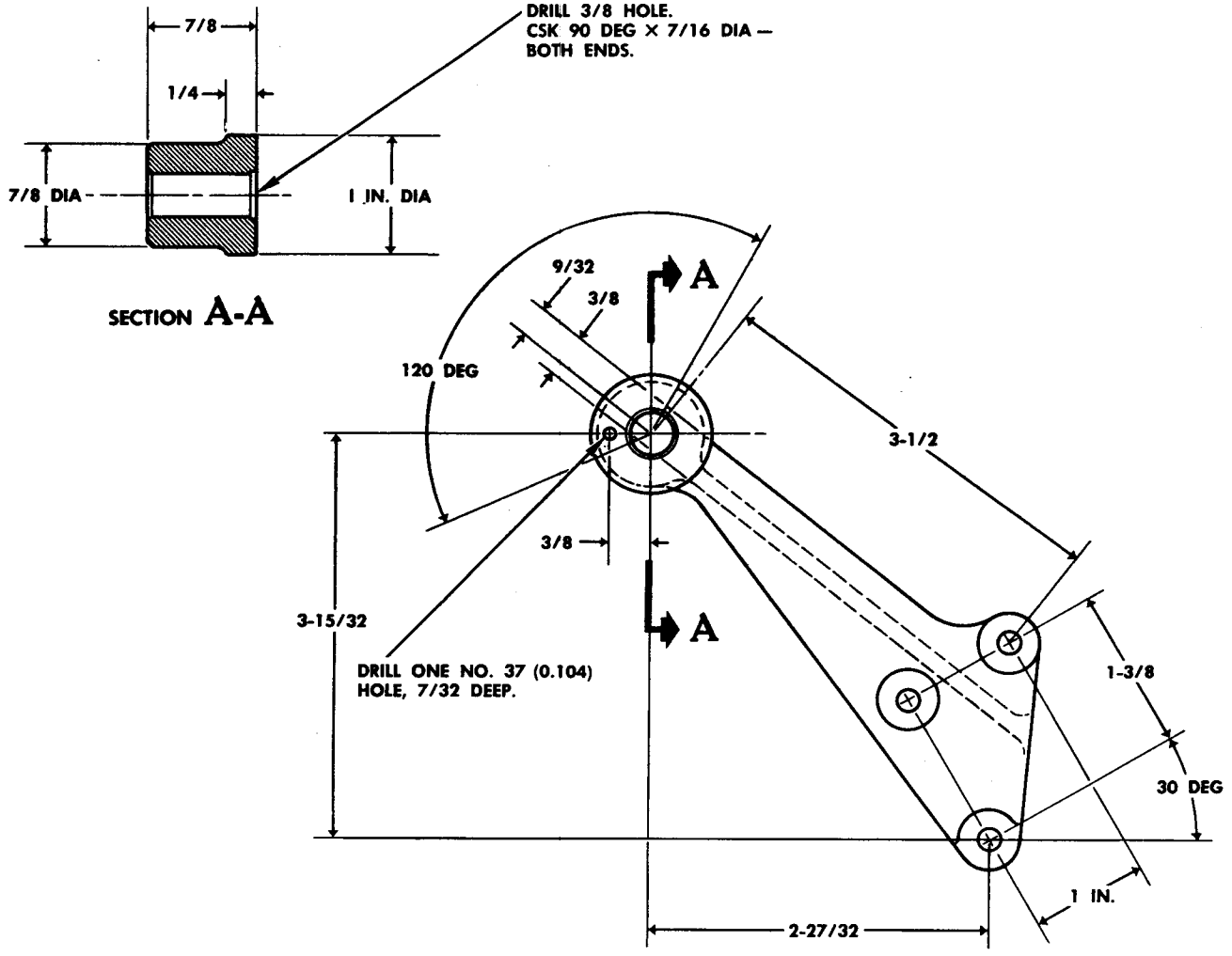
1. Smooth all edges after material removal with 600-grit emery cloth.
2. Apply two coats of primer (Specification MIL-P-7962) and two coats of gloss lacquer (Specification MIL-L-19537) to all exposed surfaces.

100D-3-33-15

Figure 5-14. Nose Gear Upper Drag Brace Repair

PERMANENT—CATEGORY B

LANDING GEAR



NOTE To repair cable guide hole when it has been worn beyond tolerance, weld and machine casting as shown.

- Material — magnesium alloy sand casting, Type AZ63-HTA (Federal Specification QQ-M-56)
- Process used — tungsten inert gas
- Filler rod — AZ63 or AZ92 (Specification MIL-W-6944)

1 Cut out wall section sector at boss hole to include worn area and small positioning hole (0.104 inch). Cutout area should be about 120 degrees arc.

2 Clean area to be welded of paint and any other contaminant.

3 Preheat casting to 400°F to 450°F before welding.

4 Weld hole and cutout area solid. Deposit sufficient bead material for machining to original configuration.

5 Stress-relieve casting at 400°F to 450°F, using acetylene torch.

6 Machine the welded boss area to original configuration.

7 Apply corrosion-preventive coating per Specification MIL-M-3171, Type 7 or equivalent.

8 Repaint to original finish.

100A-3-33-5

Figure 5-15. Emergency Landing Gear Release Handle Support Repair

Section VI

ENGINE

Contents Page

GENERAL....6-1
REPAIRS....6-1
ENGINE REPAIR INDEX....6-1

GENERAL.

The F-100D and F-100F Airplanes are powered by a J57-P-21 or J57-F-21 turbojet engine. The J57-P-21 turbojet engine is built by the Pratt & Whitney Aircraft Corporation; the J57-F-21 engine is built by the Ford Motor Company.

The engine compartment is contained within the fuselage, starting at about fuselage station 305. The engine air intake duct is part of the fuselage. For description and repair, refer to Sections IV and X.

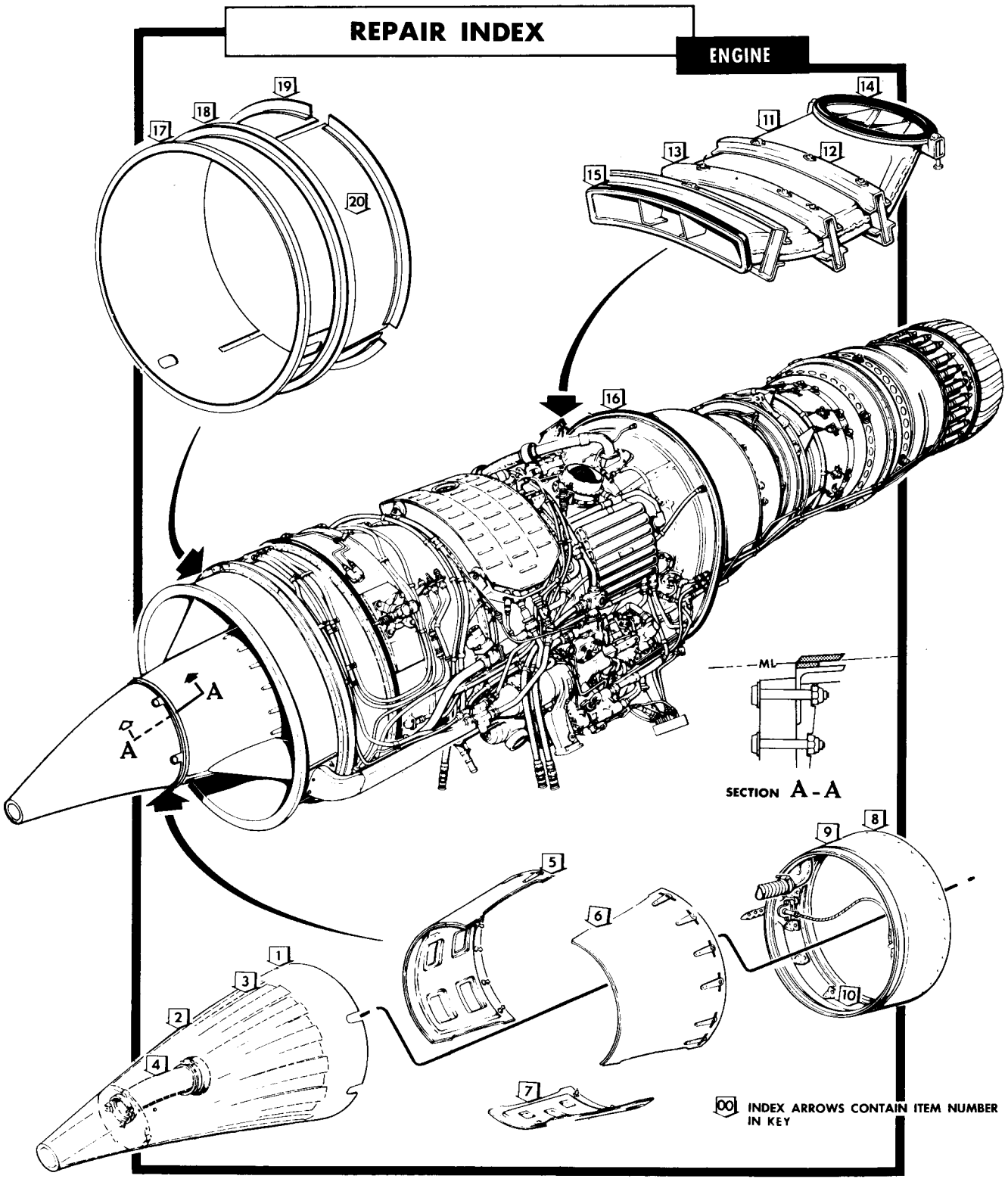
Care should be taken to keep dust, dirt, rivets, nuts, washers, and other foreign objects out of the engine areas. If any such objects are dropped into the engine, the assembly process must stop until they are found and removed, regardless of the time this may require. Suitable plugs, caps, and other coverings must be used to protect all engine openings as soon as they are exposed. The use of blind rivets should be avoided; however, if they are used, the blind rivet should be installed so that a loose rivet stem will not fall into the engine duct and be drawn into the engine. A clean repair job is of the utmost importance. The repair man must make sure that all tools, equipment, metal filings, trimmings, hardware, and any material used to make the repair are cleaned up.

REPAIRS.

Most of the engine repair information is found in the engine manufacturer's overhaul manual. Because of the special problems and materials involved in engine repair, the manufacturer's instructions should be followed. Sheet-metal parts, such as cooling-air ducts, accessory section covers, air-guide extension, etc, may be repaired using repairs found in this section or in Section X.

ENGINE REPAIR INDEX.

See figure 6-1.



F-100D-3-40-2C

Figure 6-1. Engine Repair Index

ITEM NO.	DRAWING NO.*	DESCRIPTION	GAGE	MATERIAL	REPAIR FIGURE NO.†
1	223-42003	Eng Fwd Cover Assy			
2	223-42003-5	Cover	0.040	6061-O	
3	223-42003-7	Inner Skin	0.040	6061-O	
4	223-42175	Duct		6061-O Tube	
5	223-42013	Eng Door Cover Assy—Inter Sect RH	0.040	2024-T4 Clad	6-7
6	223-42012	Eng Door Cover Assy—Inter Sect LH	0.040	2024-T4 Clad	6-7
7	223-42014	Eng Door Cover Assy—Inter Sect Lower	0.040	2024-T4 Clad	6-7
8	223-42005	Engine Aft Cover Assy			
9	223-42005-3	Skin—Outer	0.040	6061-O	
10	223-42005-5	Skin—Inner	0.040	6061-O	
11	192-42066	Duct Assy—Afterburner Cooling-air Removable			
12	192-42066-100	Skin	0.025	Corr & Heat-resistant Steel Sheet	‡
13	192-42066-7, -9, -39, & -41	Hat	0.025	Corr & Heat-resistant Steel Sheet	‡
14	192-42101	Seal		Gasket Matl	
15	192-42102	Seal		Gasket Matl	
16	XKJ-10075-BB	Fire Seal—Eng		Cal Fiber	
17	223-42006-5, -31	Channel	0.063	2024-T4 Clad	6-5
18	223-42006-9, -7	Channel	0.063	7075-T6 Clad	6-5
19	223-42006-11, -13	Angle	1E514	2024-T4 Extr	
20	223-42006-3	Skin	0.040	7075-T6 Clad	6-4

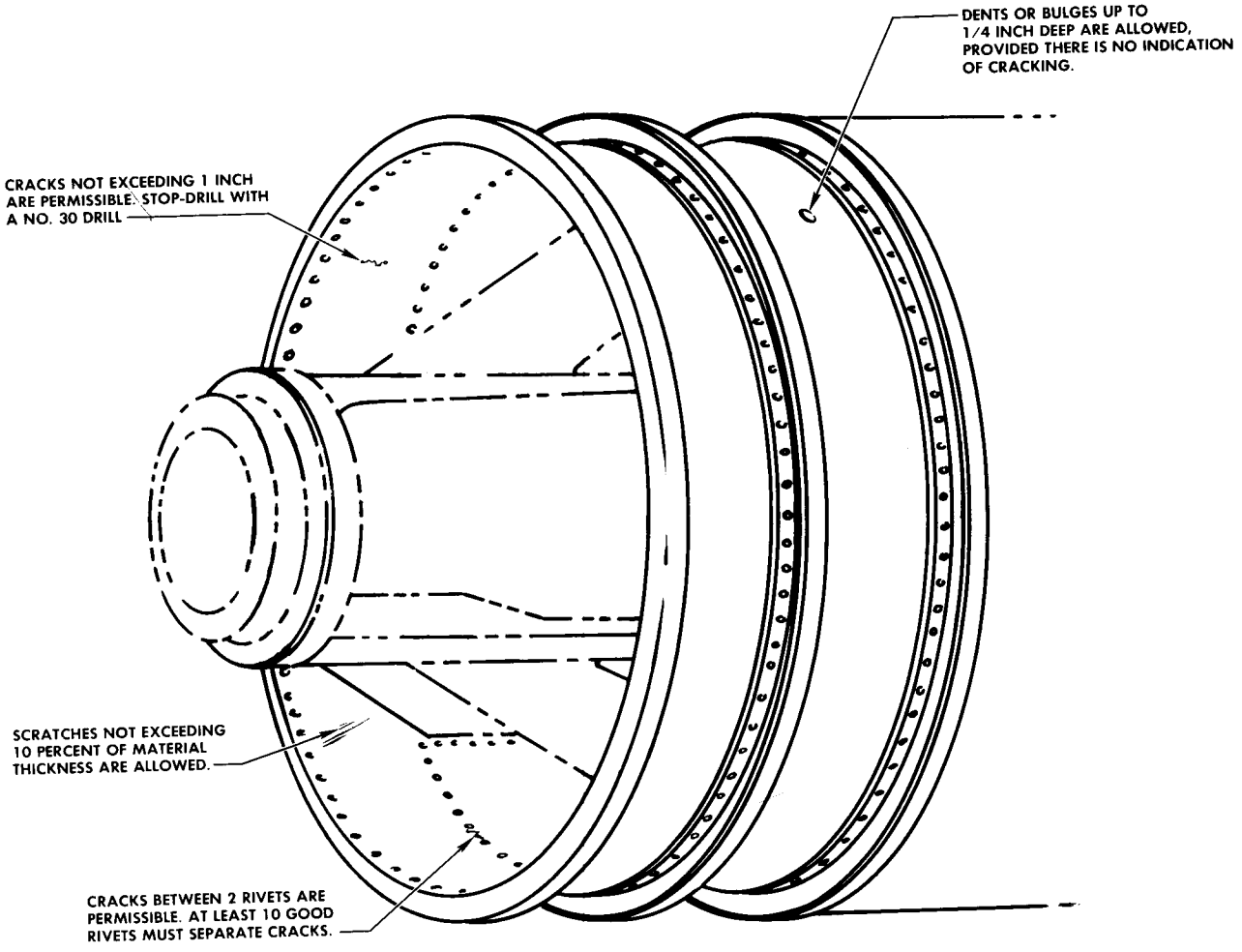
*** Drawing numbers are for reference only and are not to be used for ordering parts.**
† For additional repairs, refer to Section X.
‡ Refer to Section XII for repairs.

Key to Figure 6-1

(Figure 6-2 deleted)

NEGLIGIBLE DAMAGE

ENGINE



ENGINE AIR GUIDE

NOTE All cracks should be checked frequently for progression.

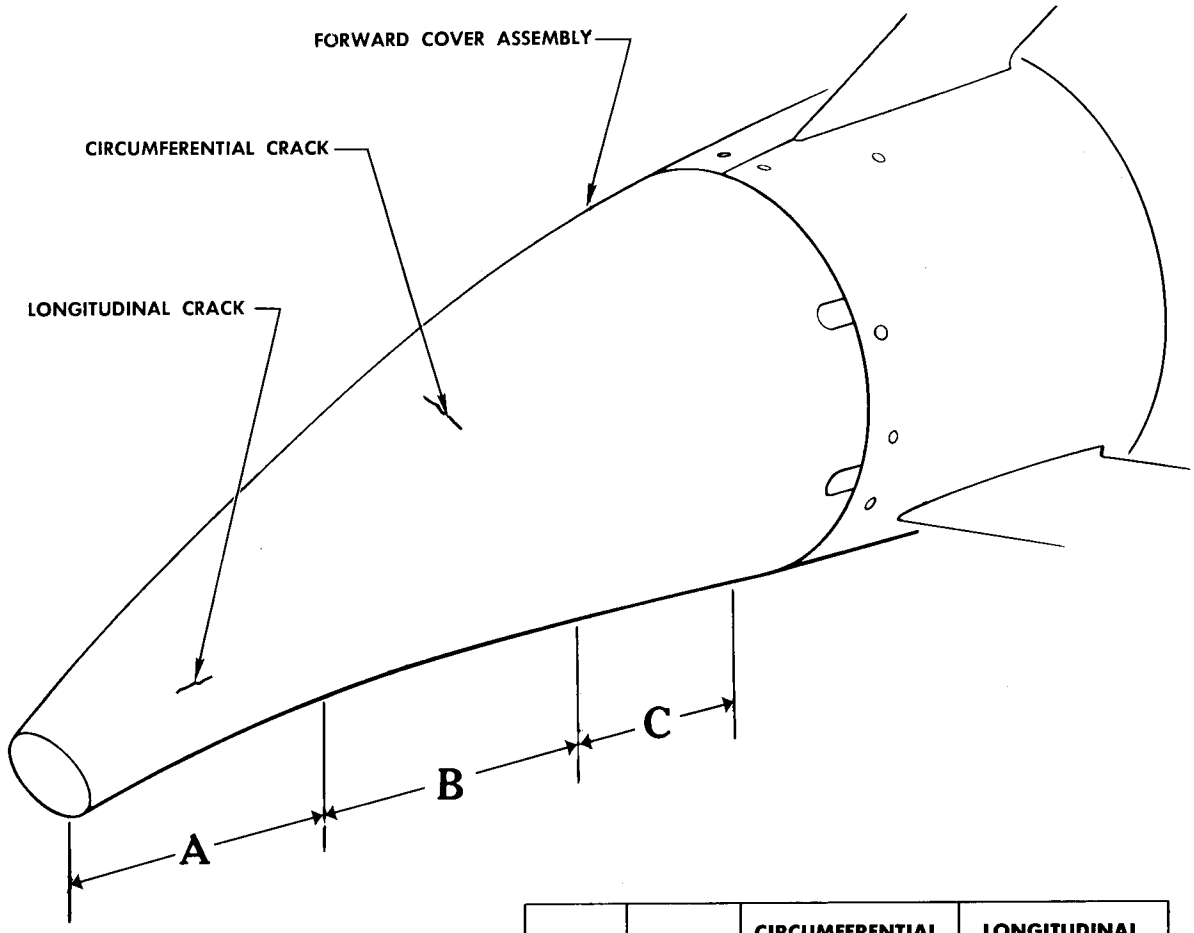
- If damage exceeds limits shown, a repair must be made.
- Refer to "Negligible Damage" in Section I.

F-100D-3-40-6

Figure 6-3. Engine Negligible Damage

NEGLIGIBLE DAMAGE

ENGINE



AREA	LENGTH	CIRCUMFERENTIAL CRACK LENGTH	LONGITUDINAL CRACK LENGTH
A	10	1	1/2
B	10	3/4	3/8
C	6	5/8	1/4

NOTE Stop-drill cracks with No. 40 drill.

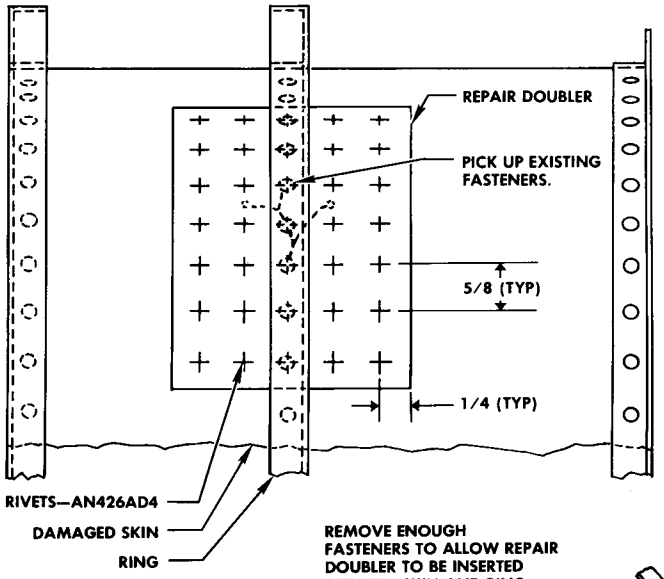
- All cracks should be checked frequently for progression.
- No two cracks are permissible within 10 inches of each other.
- Cracks exceeding limits indicated in table will require a flush skin repair (2024-T4 may be substituted for 6061-SO).

F-100D-3-40-8

Figure 6-3A. Forward Engine Cover Negligible Damage

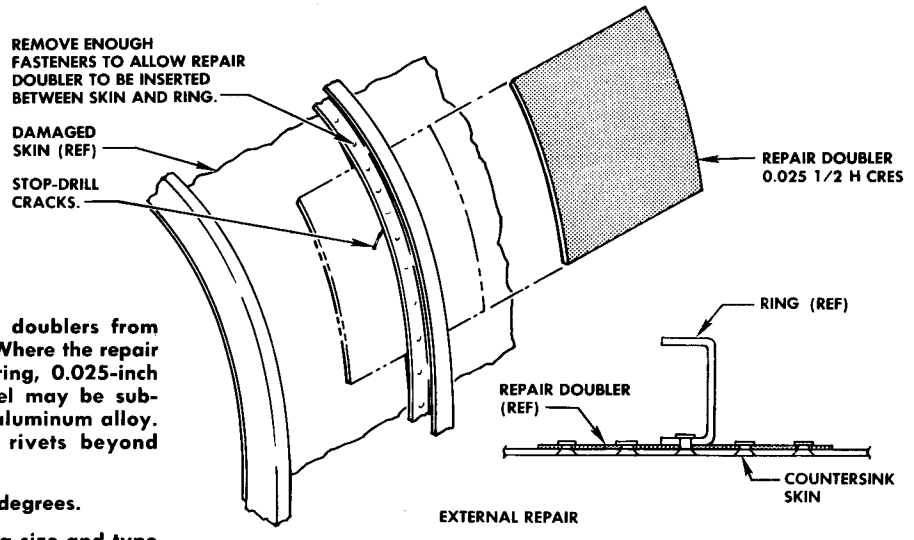
PERMANENT CATEGORY A

ENGINE

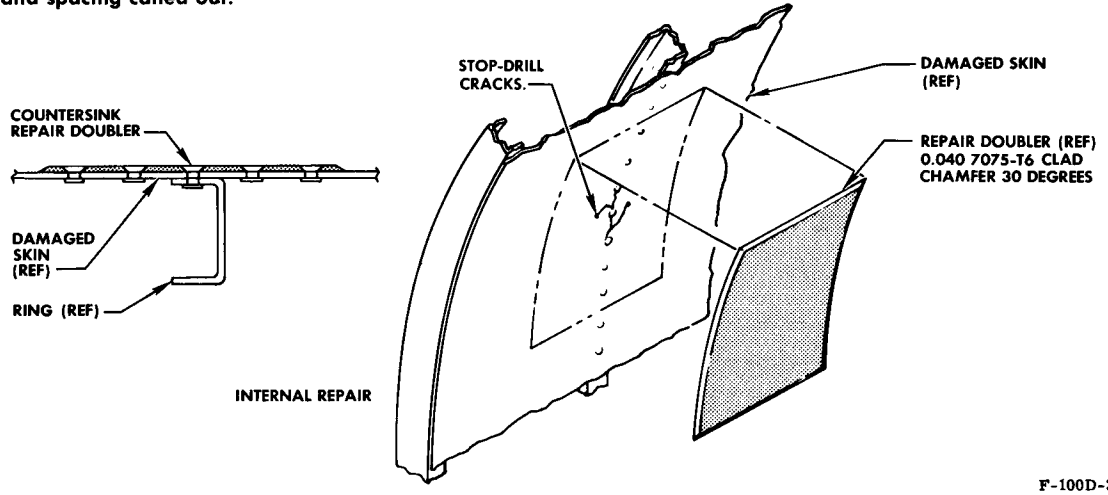


NOTE The length of the repair must not exced 18 inches.

The repairs may be made either on the inside or the outside of the air guide.



1. Stop-drill all cracks.
2. Make internal and external repair doublers from 0.040-inch 7075-T6 aluminum alloy. Where the repair doubler must be installed under a ring, 0.025-inch one-half hard corrosion-resistant steel may be substituted for the 0.040-inch 7075-T6 aluminum alloy. Extend repair doublers two good rivets beyond damage.
3. Chamfer internal repair doublers 30 degrees.
4. Fasten repair doublers in place, using size and type of fasteners and spacing called out.

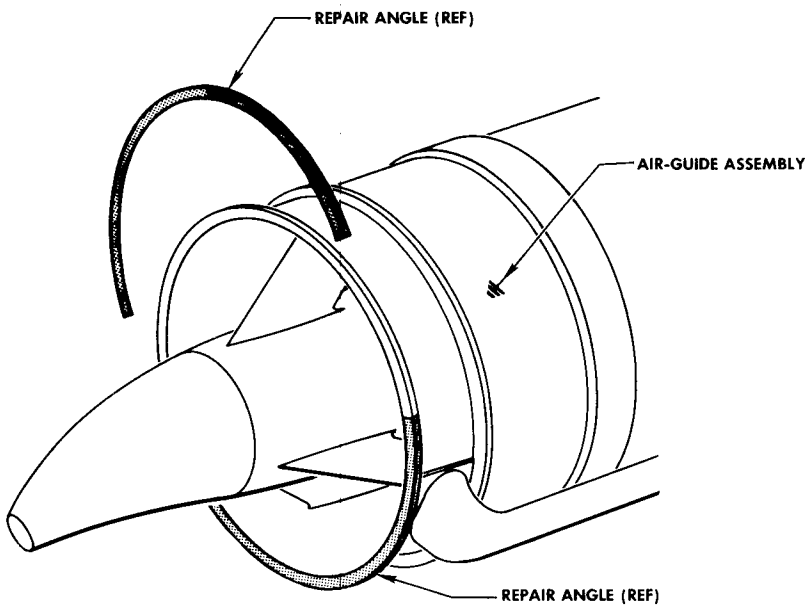


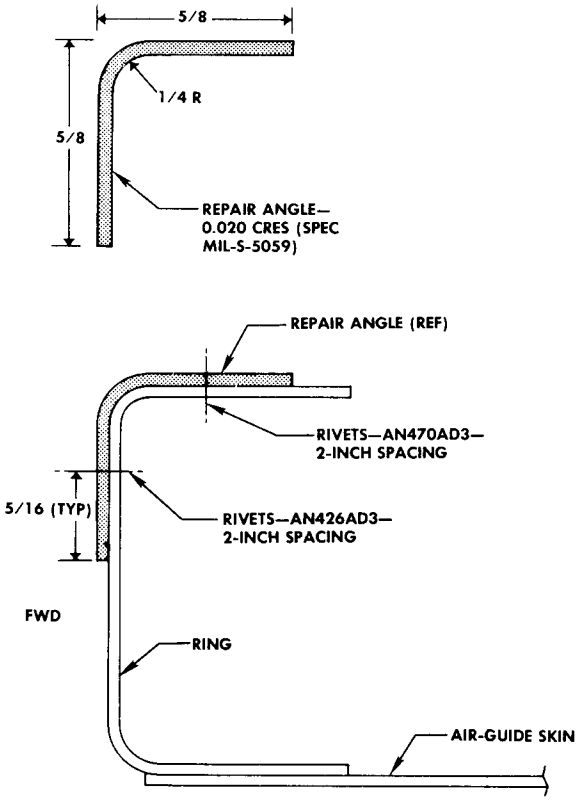
F-100D-3-40-5

Figure 6-4. Air Guide—Skin Repair

PERMANENT CATEGORY A

ENGINE





Movement between the engine air guide and the bumper assembly of the engine air inlet duct will cause the forward ring of the air guide to be worn and chafed.

Repairs may be made in the following manner:

1. Form repair angles from 0.020-inch corrosion-resistant steel to match the air-guide ring.
2. Lay out and drill rivet holes through both legs of repair angles.
3. Remove repair angles from air guide, and dimple the fastener holes in the forward face of the repair angles. Subcountersink the fastener holes in the forward face of the ring.
4. Fasten repair angles in place, using size and type of fastener and spacing called out.

100D-3-40-4A

Figure 6-5. Air-guide—Channel Repair

(Figure 6-6 deleted)

PERMANENT CATEGORY A

ENGINE

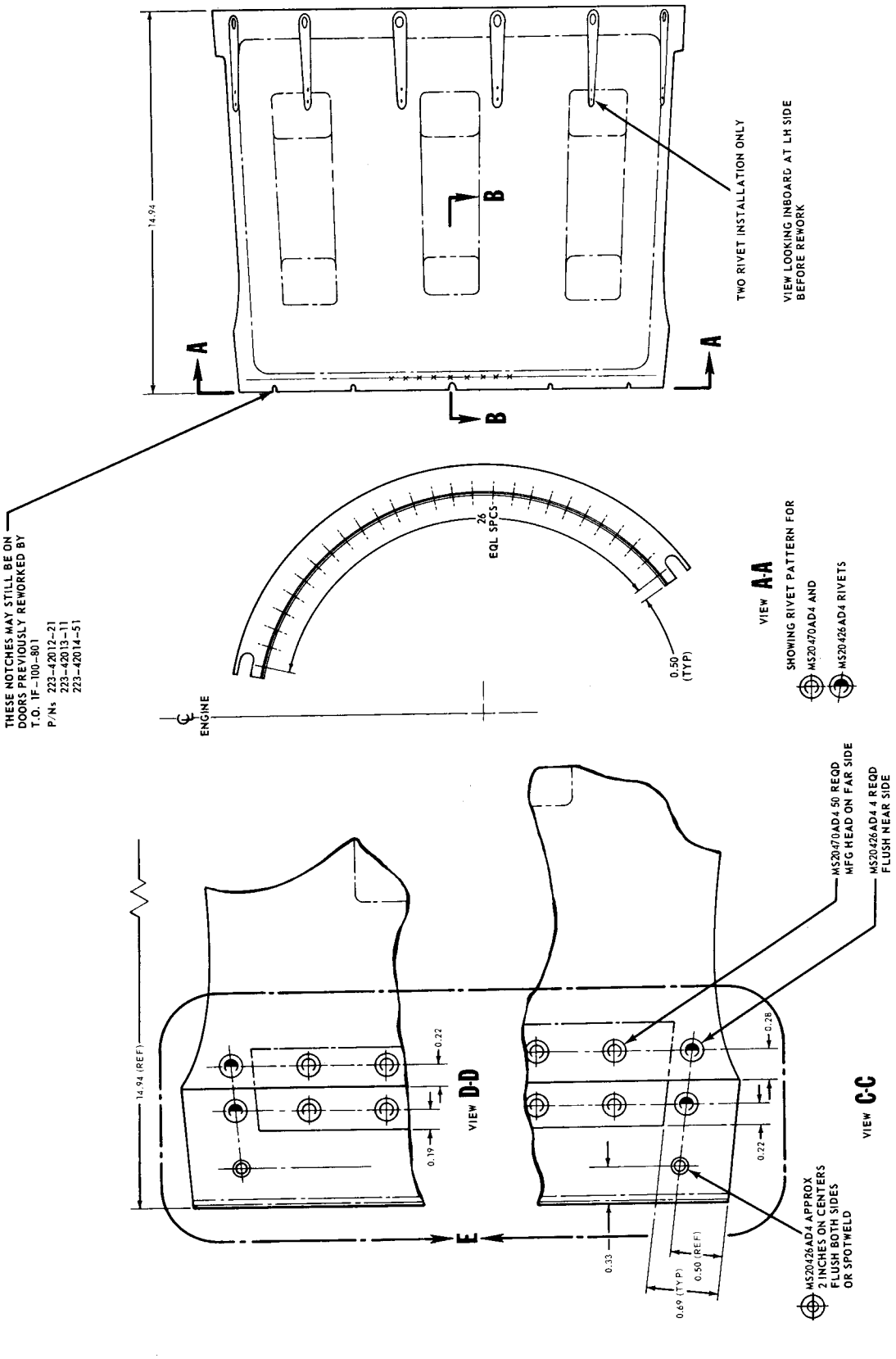


Figure 6-7. Door Assembly – Engine Access Cover, Repair. (Sheet 1 of 2.)

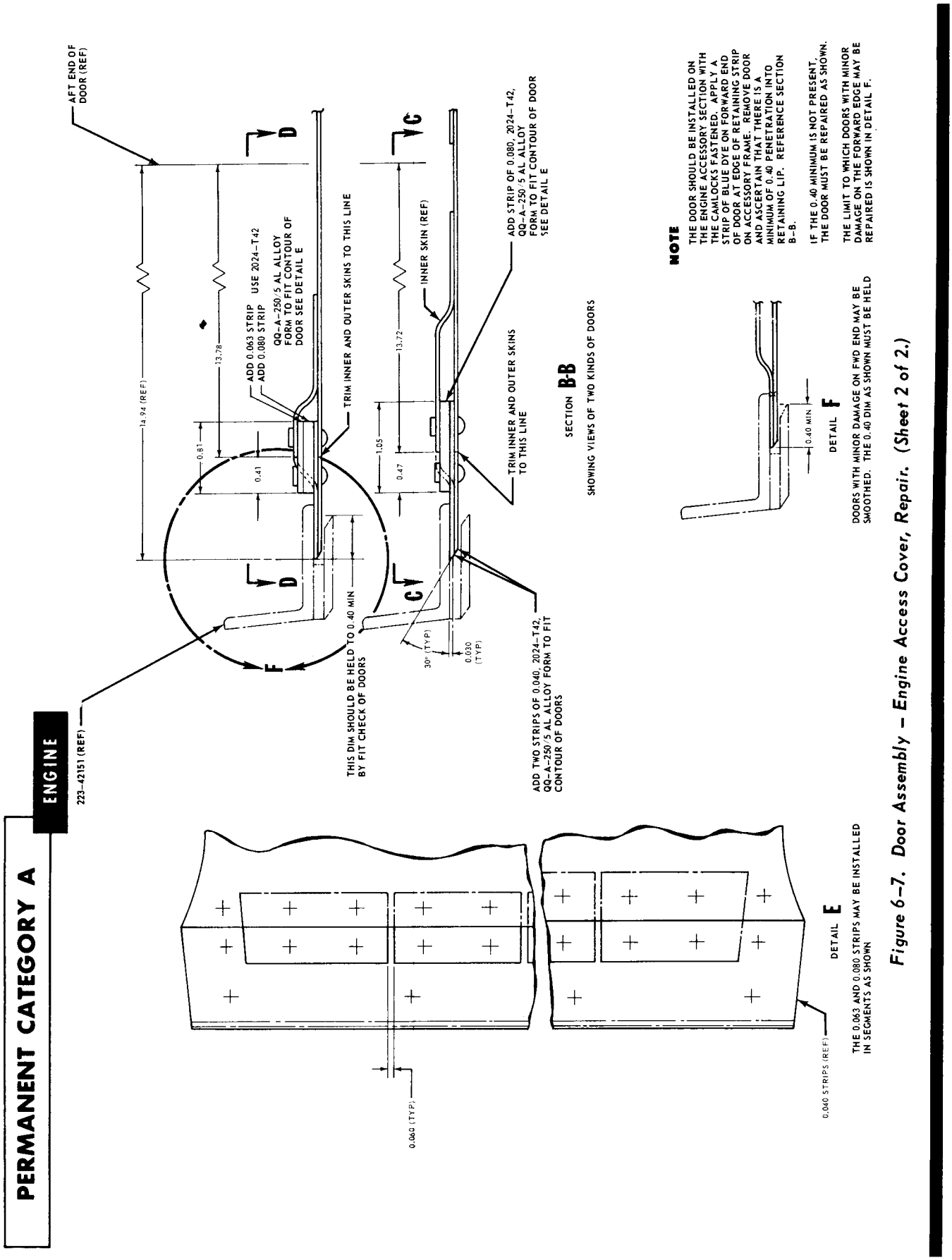


Figure 6-7. Door Assembly – Engine Access Cover, Repair. (Sheet 2 of 2.)

Section VII

FABRICS AND PLASTICS

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GENERAL.

Glass-fabric reinforced plastic parts are used for radio and radar antenna covers and various shields and covers for electrical equipment. Clear plastic is used for the cockpit canopy and navigation light covers on the wing, empennage, and fuselage. Many of these parts are small and should be replaced if damaged. The larger parts may be repaired, if necessary, provided the damage is not too great and is not in a critical area. The canopy is made from plastic material conforming to Specification MIL-P-8184. Information on repair of canopies may be found in "Fabrication, Maintenance, and Repair of Transparent Plastics," T.O. 1-1A-12.

GLASS-FABRIC-REINFORCED PLASTIC.

Glass-fabric-reinforced plastic parts are made of layers of glass fabric which have been impregnated with chemical resins, molded to a particular contour, and cured to a hard consistency. Polyester resin, which is used in most cases, is a thermosetting (hardens when exposed to heat), low-pressure, laminating resin. When the resin is mixed with a catalyst according to the manufacturer's specification, and cured by heat or at room temperature, it gradually changes from thick sirupy liquid to a hard solid. The curing action is controlled by the type of catalyst used. The function of the glass fabric in glass-fabric-reinforced plastic is just as important as the function of the resin in providing good strength properties. When the glass fabric and resin are combined, a very strong, lightweight material is obtained. This material is resistant to water, weather, aging, and a variety of chemicals. This material also has desirable electrical properties so that it may be used for antenna covers and electrical shields.

REPAIRING GLASS-FABRIC-REINFORCED PLASTIC.

Laminated glass-fabric-reinforced parts are strong and long wearing under normal conditions, but like any aircraft material they are subject to damage and abrasive wear. Glass-fabric-reinforced plastic may be damaged by sharp objects, flying stones, or sharp blows much the same as metal aircraft parts. Damage usually consists of tears, bruised or delaminated areas, or a combination of these damages. The most common type of damage is skin fracture or tears. When cuts or tears extend through several layers of skin in locations where appearance is not important, the damage may be repaired by bonding a patch to the exterior surface of the part. This type of repair may be used on some of the glass-fabric-reinforced plastic electrical covers, etc, but should not be used on antenna covers. Damage to antenna covers and exterior surfaces should be repaired using a "stepped" method. When this method is used, glass fabric layers are cut from the damaged area in rectangular strips which gradually decrease in size until the limit of damage is reached. Layers that are removed are saved to serve as templates. New strips of glass fabric are saturated with the resin mixture, accurately cut to match the size of the removed material, and then carefully fitted into the damaged area. When damage occurs to the radome or antenna covers in the area of radio or radar transmission, repair personnel should take extra care to maintain the exact design thickness of the original installation. A natural tendency to "beef-up" the repair area should be avoided. The antenna cover admits a particular frequency range of the electromagnetic spectrum at a particular angle or ray incidence; therefore, any changes in the thickness on the cover, in this area, may cause distortion of electrical waves. Laminated plastic parts may become bruised or delaminated beneath the surface if subjected to a sharp blow. Damage of this type, which is not over a large area, can often be repaired by injecting a small quantity of resin through the skin with a hypodermic syringe. If the damaged area is too large to be filled with a hypodermic syringe, a series of No. 60 holes, evenly spaced through the area of damage, may be drilled. Enough resin to ensure bonding of all the separated surfaces is then injected through the holes. The entire area must be placed under enough pressure to remove excess resin. After curing, the area should be sanded and refinished. Delamination may be visible on the surface of the part as a bubble or blister. In some cases, however, subsurface delamination may not be visible. If delamination is suspected, but cannot be seen, the

extent of damage may be determined by tapping the surface with a coin or small tool. Undamaged areas will respond with a sharp metallic ring, whereas delaminated areas will produce a hollow sound. The tapping procedure is not satisfactory for surfaces coated with rain erosion protective material because the coating dulls the sound of the coin or tool.

MATERIALS USED.

MATERIALS	SOURCE
Low-pressure laminating resin	Spec MIL-R-7575
Liquid catalyst (curing agent) methyl ethyl ketone peroxide (DDM)	Lucidol Div Novadel-Agene Corp Buffalo, N.Y.
Glass fabric, Type VIII	Spec MIL-F-9084
Rain-erosion-resistant coating material—primer (Bostik 1007)	B B Chemical Co Cambridge, Mass.
Top-coat base cement and accelerator	Spec MIL-C-7439
Cobalt naphthenate (6-percent)	Nuodex Corp Elizabeth F, N.J.
Toluol	Fed Spec TT-T-548
Cellophane	Commercial
Polyvinyl alcohol sheet (0.003- to 0.005-inch)	Commercial
Felt sheet (0.250-inch)	Commercial
Sandpaper (120-grit)	Commercial
Masking tape	Commercial
Scotch tape	Commercial

The following equipment is suggested to aid in making this repair. However, substitute equipment may be used as needed.

MATERIALS	SOURCE
Razor blades (single-edge)	Commercial
Infrared heat lamp (250-watt, 105- to 120-volt or equivalent)	Commercial
Small spatula	Commercial
Paint brush (½- or one-inch width)	Commercial
Rolling pin or equivalent	Commercial

REMOVING RAIN-EROSION-RESISTANT COATING.

If the area to be repaired is covered with a rain-erosion-resistant coating, the coating must be removed to a minimum of 3 inches around the damaged area. To remove coating, place a piece of ¼-inch thick commercial felt over the area and fasten felt in place securely with masking tape. Saturate felt with toluol and keep it saturated for 30 to 45 minutes. At the end of this time, the coating should come loose from the glass fabric easily. Fair in edges of coating with a cloth soaked in toluol, or with commercial sandpaper, 120-grit.

PREPARATION OF DAMAGED AREA.

Clean out damage to a rectangular shape large enough to remove any areas that have delaminated. The rectangular shape makes it easier to fit the repair layers of glass cloth. A sharp knife or razor blade may be used to cut the material. After the hole has been sized, the correct amount of overlap for the top layer may be determined. (See figure 7-1.) Carefully trim and peel top layer of cloth from overlap area. Be careful not to cut into next layer of cloth, and do not let layers separate outside damaged area. This would result in an area which is structurally weaker than the surrounding area. Cut and peel each layer in stair-step fashion as shown in figure 7-1. A spatula is often helpful in lifting the layers and for separating the layers for removal.

MIXING LAMINATING RESIN.

The laminating resin must be mixed with the curing agent or catalyst according to the manufacturer's instructions before use. After adding curing agent to resin, mix thoroughly for several minutes. Laminating resin should be mixed in small amounts, to minimize waste, because the mixed resin has a pot life of only about 6 hours.

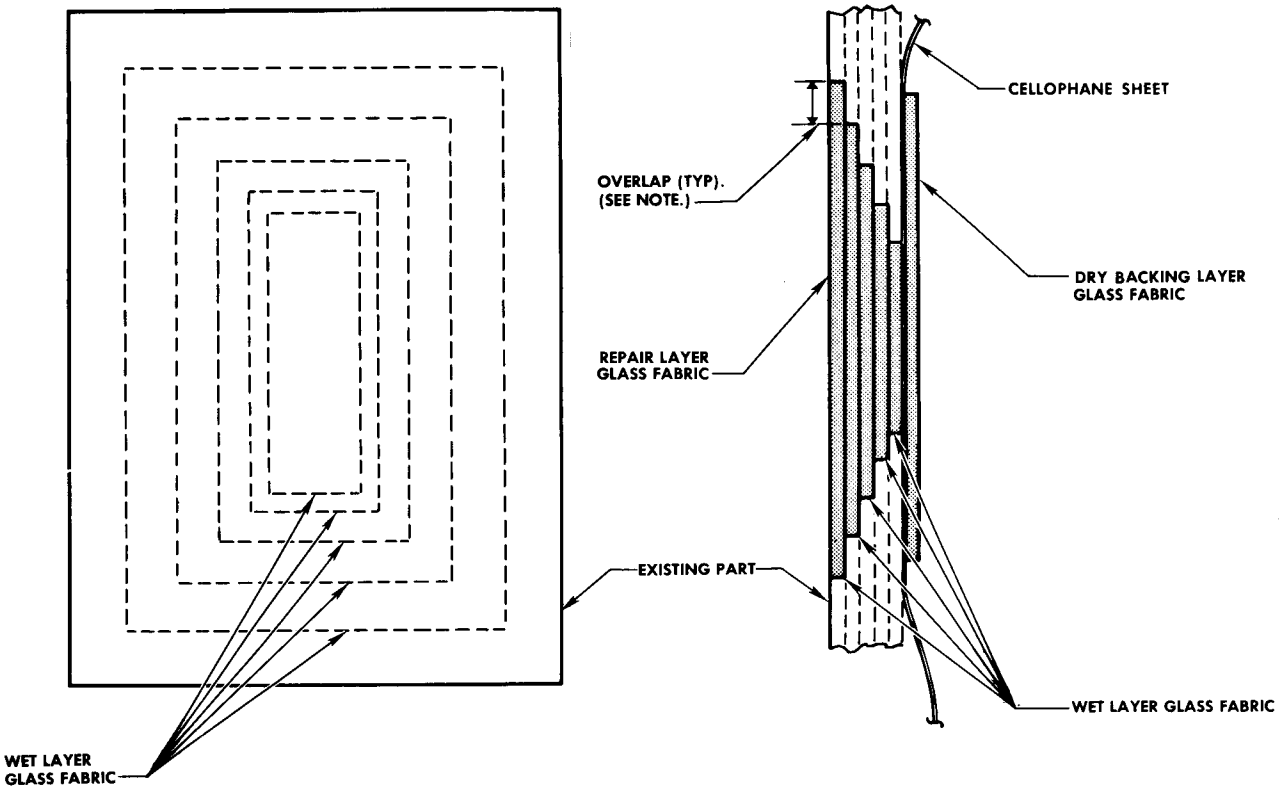
PREPARING "WET" LAYER.

Spread glass fabric on a sheet of cellophane on a flat surface. Saturate fabric with mixed resin, spreading it uniformly over surface with a spatula or other flat tool. Cover saturated fabric with another sheet of cellophane, and roll gently but firmly to force resin into fabric. This rolling action forces the excess resin out to the edge of the fabric. This impregnated fabric between sheets of cellophane may then be cut to the required size. To avoid contamination, the cellophane should remain in contact with the "wet" fabric until the layer is to be used.

PREPARING "DRY" LAYER.

The "dry" or precured layer is used as a backing layer. This layer should be contoured to match the part being repaired as near as possible. The preparation is the same as for a "wet" layer, except that it is not cut to size until after it is cured. To cure "dry" layer, lay fabric smoothly over a contouring block and expose it to heat from an infrared heat lamp, or equivalent, for one hour. The lamp should be held about 12 to 15

PERMANENT CATEGORY A — FABRICS AND PLASTICS



NOTE For a hole up to 2-1/2 by 4 inches, use 1/2-inch overlap; for a hole up to 5 by 8 inches, use 3/4-inch overlap; for a hole over 5 by 8 inches, use one-inch overlap.

1. Trim out damaged area in "stair-step" fashion, as shown. Be careful not to cut into layer below the one being removed.
2. Mix resin and prepare layers. (Refer to text.)
3. Install dry layer as shown.
4. Install wet layers.
5. Cover repair with polyvinyl alcohol sheet on both sides. Stretch tight and tape in place.
6. Cure repair and refinish area as prescribed in text.

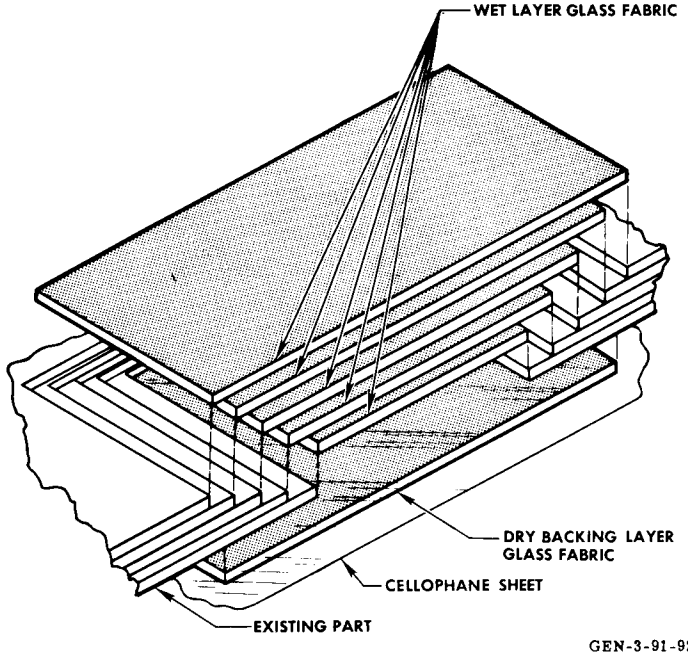
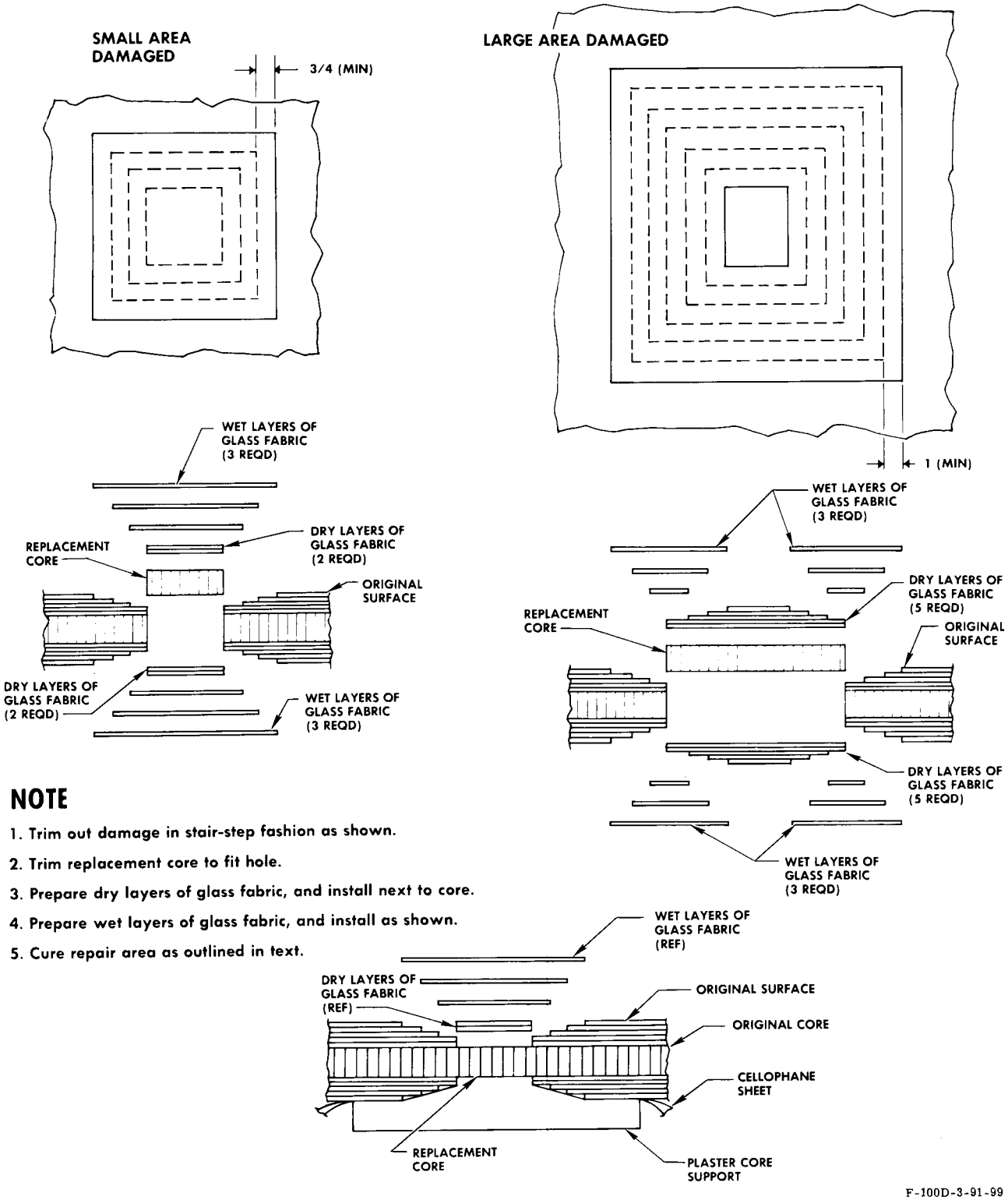


Figure 7-1. Glass Fabric Laminate Repair

HONEYCOMB REPAIR

FABRICS AND PLASTICS



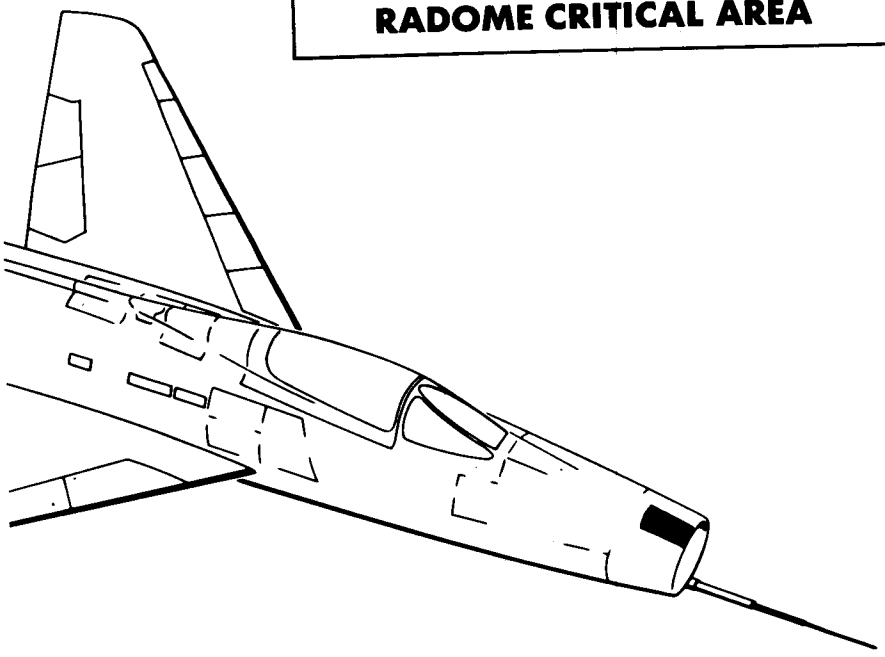
NOTE

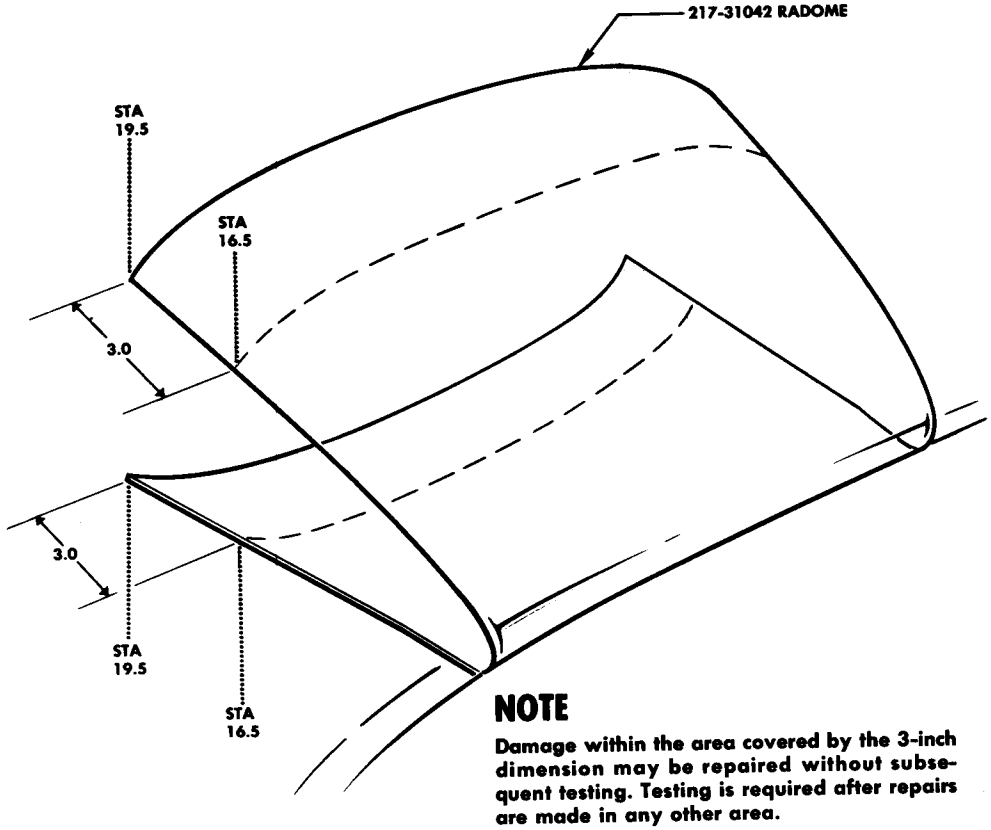
1. Trim out damage in stair-step fashion as shown.
2. Trim replacement core to fit hole.
3. Prepare dry layers of glass fabric, and install next to core.
4. Prepare wet layers of glass fabric, and install as shown.
5. Cure repair area as outlined in text.

Figure 7-2. Glass Fabric Honeycomb Repair

RADOME CRITICAL AREA

FABRICS AND PLASTIC





NOTE

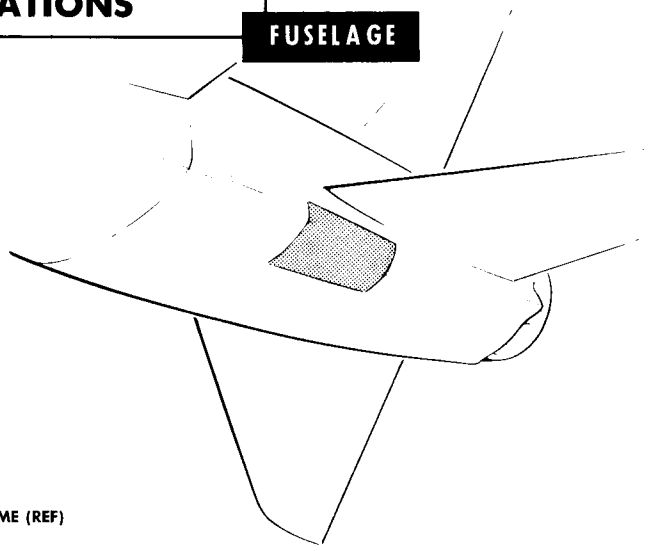
Damage within the area covered by the 3-inch dimension may be repaired without subsequent testing. Testing is required after repairs are made in any other area.

F-100D-3-81-101

Figure 7-3. Radome Critical Area

REPAIR LIMITATIONS

FUSELAGE

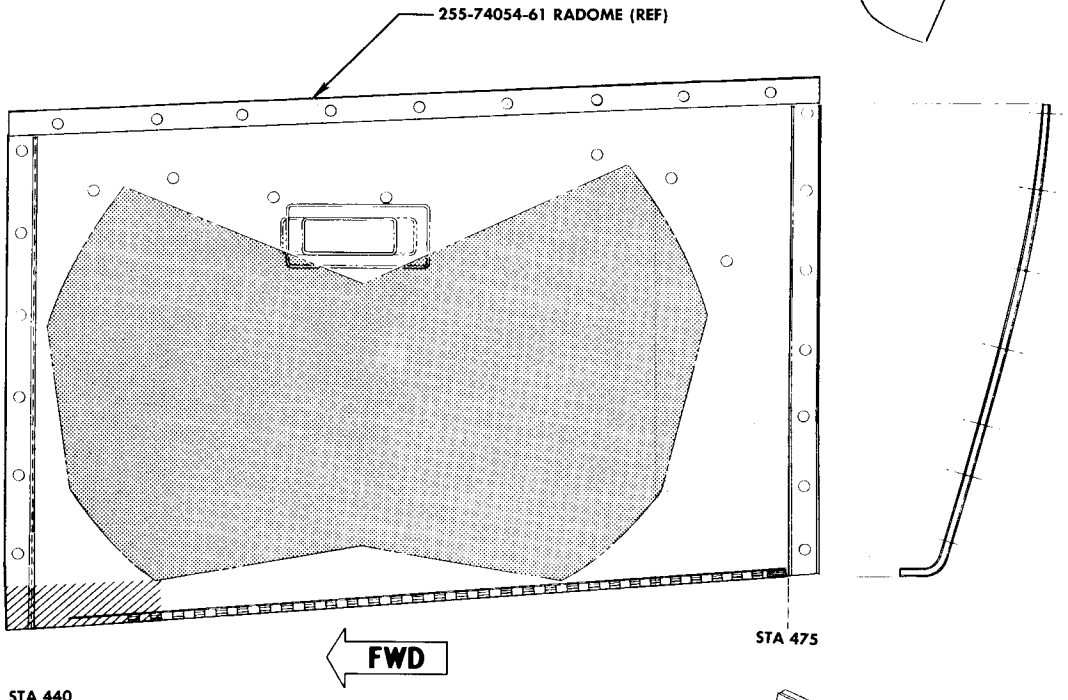


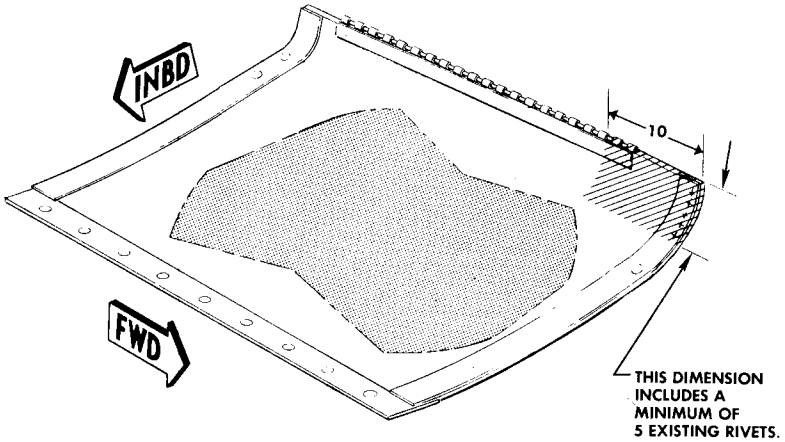
CRITICAL AREA (OPERATIONAL)

Repairs must not be made in this area if the damage exceeds six laminations. The total repaired area must not exceed 15 percent of this area.

CRITICAL AREA (STRUCTURAL)

No repairs are permitted in this area.





NOTE The 255-74054-51 radome has a slightly different configuration; however, the critical areas are the same as shown in the illustration.

- Refer to "Repairing Glass-fabric-reinforced Plastic" for repair procedure.

F-100D-3-91-105

Figure 7-4. Repair Limitations—255-74054 Radome

inches from the surface of the layer. The surface temperature should be 150°F to 200°F for at least one hour. After the layer has cooled, it should be relatively hard, if it is completely cured. The cellophane sheets should not be removed.

FITTING REPLACEMENT CORE.

When repairing honeycomb sandwich type construction, it may be necessary to replace the core in the damaged area. Cut out the core replacement to provide a tight fit. Be sure that core surfaces are flush with adjacent core, and that replacement is same thickness as original core. If the repair is in an area of sharp contour, it may be necessary to use several pieces of core to reproduce the same contour as the original surface. Do not remove fuzz from sawed edges of replacement core. These frayed surfaces are important aids in obtaining a strong secondary bond between the core and the inner layers of the faces.

CORE SUPPORTS.

When repairing honeycomb sandwich type construction, it is necessary to support the core while making the repair lay-up on one skin. When one skin is cured, it will support the core while the other side is repaired. There are many ways the core may be supported for this first phase of repair. One of the simplest and most accurate is the plaster "splash" method. To do this, stretch a piece of cellophane or polyvinyl alcohol sheet over an area about four times as large as the repair and as close to the repair as possible. As much as possible, choose an area with the same contour as the repair area. Mix a large enough batch of plaster of Paris to coat an area of cellophane the size of the repair plus about one-inch overlap on all sides, and to a depth of ¼ to ⅜ inch. When plaster is completely dry, remove it from cellophane, and trim core support to leave a male-type part which matches core surface. This plaster mold will support the replacement core after it is taped securely in place with a piece of cellophane between the plaster and the core plug. Trimming of the plaster may be done with a knife or a piece of coarse sandpaper (garnet No. 1/0 or No. 1/2).

REBUILDING DAMAGED AREA.

Where both faces and the core of a honeycomb-type panel are damaged, a support for the core on the far side is required during the lay-up of the repair. The far face of the panel will provide this support normally, unless it has had to be removed. Brush a coat of the mixed resin onto the four sides of replacement core and core cavity. Fit replacement core carefully into place with a layer of cellophane next to replacement core on far side. Put core support in place, and tape down securely. The cellophane sheet should be large enough to prevent leakage of the resin onto the adjacent layers of the panel and the core support. The cutout area to be rebuilt should be masked off with cellophane or masking tape. This masking should cover an area at least 2 inches wide around the entire cutout. Masking saves considerable time and trouble in the cleanup process after the repair has cured. Peel cellophane from one side of precut "dry" layer, and brush a thin coat of mixed resin onto exposed surface. Position layer on core, with resin-coated surface against core. Rub layer gently with a smooth tool to remove excess resin and to secure layer in place. Peel cellophane from one side of first precut "wet" layer and from "dry" layer already in place. Position layer as before and rub gently to remove excess resin and any trapped air bubbles. Repeat this procedure for all layers until repaired area has the same number of layers as original part.

When last layer is in place, remove outer cellophane from it and brush a thin coat of resin over outside layer. Cover repair area with a sheet of polyvinyl alcohol sheet. Stretch this sheet as tightly as possible, without wrinkles, and secure it with cellophane or masking tape. Rub repair area gently with a smooth tool to remove excess resin and to work any air bubbles out of layers and onto masking. Too much rubbing should be avoided because it may reduce the resin content of the fabric enough so that a poor bond results.

Laminated glass fabric without the honeycomb core is repaired about the same as the honeycomb type except a precured backing layer is used to support the repair layers. This backing layer should be formed to the same contour as the repair area. The repair consists of replacing the skin layers as required.

CURING LAY-UP.

Cure repair area with infrared heat lamp or equivalent. The lamp should be placed 12 to 15 inches from the surface to maintain a surface temperature of 150°F to 200°F. The curing time should be about one hour. If the repair is properly cured, the area should have a metallic ring when tapped with a coin. If a dull sound is produced, or if repair is soft to the touch, continue curing until repair is hard to the touch and produces a metallic ring. After repair is cured, remove polyvinyl sheet and masking around repair. The complete repair may be sanded lightly to obtain good fairing and a smooth surface.

If heating equipment is not available, the repair may be cold-cured. A special resin mix is required for a cold or

room-temperature cure. This special mix is made by adding 6-percent cobalt naphthenate to unmixed resin according to the manufacturer's instructions. Mix thoroughly, add curing agent, and then mix again for several minutes.

CAUTION

The cobalt naphthenate must be mixed into the resin before the curing agent is added. Contact of the undiluted cobalt naphthenate with the curing agent produces an explosive mixture.

The pot life of the mixed resin for a cold cure is about 10 to 15 minutes at about 70°F. This time varies with the temperature of the air. The higher the temperature, the less time it takes the resin to harden. At temperatures below about 60°F, the resin will not harden. For this reason, the room-temperature cure cannot be used below about 60°F. Curing time at about 70°F for this material is at least 2 hours or until the repair is hard to the touch and produces a metallic ring when tapped with a coin.

REPAIR OF DRAG CHUTE LINERS.

The aluminum foil initially used on the exterior surfaces of the drag chute liners is easily damaged and has a tendency to peel after continued use. New liners received from the factory are sprayed with a coating of gold reflectorizing-type resinous paint, in accordance with NAA Specification FA6-272. The following method of reworking both kinds of liners is recommended:

1. Remove all existing foil from old-type liners by peeling or scraping. With the new-type liners, this step is not applicable.
2. Smooth out all exterior lamination surfaces by sanding. Remove all dust particles.
3. Clean exterior surfaces with methyl ethyl ketone.
4. Prepare a coating mixture of the following composition, by weight:

   8 parts No. 828 epoxy, Shell Chemical Co

   1 part aluminum pigment (Federal Specification TT-A-468)

   1 part "T" hardener (Diethylene Triamine), Shell Chemical Co
5. Mix epoxy and aluminum pigment thoroughly.
6. Strain mixture through cheesecloth.
7. Add catalyst hardening agent just before using. Mix thoroughly.

WARNING

Do not mix more than one quart at a time. This mixture becomes *very hot*, and larger amounts could cause fire. Hand protection should be worn; touching the container could cause severe burns.

8. Apply a coating of this epoxy-aluminum compound to liner, Part No. 223-31525-31, by spraying or brushing. Do not coat straps.

Note

Spraying is preferred, because it imparts a smoother reflectorized coating to the outside surface of the liner. Spraying equipment must be cleaned immediately after use of the gun.

9. Bake coated liner at 160°F to 180°F for one hour.

REFINISHING RAIN-EROSION COATING.

Surfaces coated with rain-erosion-resistant material are coated basically to protect them from rain impingement during high-speed flight. To satisfactorily protect the surface, the coating must be well-bonded and free of all blemishes. Blemishes erode very rapidly, while poorly bonded areas blister under rain impact. Once the coating has become penetrated, water droplets, along with the air stream, tend to peel the coating away and expose the parent surface to extremely rapid erosion because of direct impact. Parts in which the coating is damaged more than 5 percent must be completely recoated. Parts with less than 5 percent of the coating damaged may have the coating patched without completely replacing the coating.

RECOATING—TOTAL AREA.

The coating must be applied to scuff-sanded surfaces which are free of pinholes and any surface irregularities. Following the initial scuff sanding, any pinholes or defects must be filled with Tuf-On P-24A Filiplast filler or its equivalent. Filler material can be knifed into the defect or it can be sprayed after the filler has been thinned with toluene or xylene. When the filler has dried, the surface must be smoothed by sanding with either sand or garnet paper. Use a grit no coarser than 1/0. Clean surface of all dust by using an air hose with clean, dry air or by wiping with clean cloth dampened with toluene or similar solvent. Spray surface with Bostik 1007 primer to thickness of 0.001 to 0.002 inch. Two coats are normally enough. Allow about 10 minutes between coats. If the surface is not perfectly smooth after applying the primer, it should be sanded lightly with sand or garnet paper with a grit no coarser than 4/0. After primer has been allowed to dry for at least 30 minutes, spray surface with MIL-C-7439 top-coat material which has been catalyzed to the manufacturer's specifications. All materials must be used within 12 hours after catalization. The initial coats should be very light to prevent the primer from being dissolved into the top-coat material. Spray from wet to dry surfaces, in long, even passes. Two or three fast passes per coat are preferable to one slow pass because it gives greater coat thickness with less chance of running or ragging. Under

normal conditions, 10 to 15 minutes drying time between coats should be allowed. The finished thickness of the top-coat material should be from 0.007 to 0.009 inch thick. Any entrapped air bubbles can be replaced by spraying the surface gently with a mist coat of thinner, or a fifty-fifty mixture of toluene and methyl ethyl ketone. The finished coating must be dried for a minimum of 72 hours under interior conditions before installation. The finished coating must be free of pinholes, blisters, lumps, runs, sags, and any other surface irregularities. The cured coat can be cleaned with a mild detergent and water. No other solvents should be used. When desired, the coating may be polished with a carnuba base *paste* wax. Liquid oils, polishes, and cleaners must not be used.

PATCHING—LESS THAN 5 PERCENT OF TOTAL AREA.

Mask off a rectangle which includes the damaged area plus a perimeter of about ½ inch of the nondamaged area. The masking should be at least 2 inches wide. With a sharp knife, carefully cut and remove rain-erosion coating inside of marked area, being careful not to cut glass laminate. Sand exposed area with sand or garnet paper no coarser than 1/0. Extreme caution should be used so as not to loosen the edges of the adjacent coating. Fill all pinholes with Tuf-On P-24A Filiplast or equivalent, and sand smooth. Remove all dust and foreign matter by wiping area with clean cloth dampened with toluene. Brush a coat of Bostik 1007 primer on prepared surface. Care should be used so that the primer barely contacts the edge of the adjacent surface. The presence of more than a slight amount of the primer against the coating will cause blistering and loss of adhesion. This will necessitate repeating the entire procedure. Air-dry the Bostik 1007 primer for 2 hours before application of top coating. Brush on as many coats of catalyzed top coating as required to bring repaired surface slightly higher than masking tape around edge of repair. Allow at least 30 minutes drying time between each coat. After last coat has dried for 24 hours, remove masking tape and carefully feather edges of repair, using clean cheese cloth *slightly* dampened with toluene. The excessive use of toluene will cause the repair to blister. Air-dry completed repair for at least 72 hours before installation.

Section VIII
EXTRUSION CHARTS

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EXTRUSIONS AND ROLL-FORMED SECTIONS.

Extrusions are die-formed parts that can be made to any specific cross-sectional shape. The material, in a plastic or soft state, is forced through the die by very high pressure. The material takes the form of the die and becomes solid in that shape. The extrusion process makes possible the production in quantity of many sections of unusual shape. There is also the advantage of a considerable reduction in weight.

Extrusions may be made of many different types of material, both metal and plastic. The F-100D Airplane makes extensive use of extrusions made of aluminum alloy (2024 and 7075) and magnesium. When extrusions are replaced in an airplane, the material of the replacement part must be the same as that of the original.

Roll-formed sections differ from the extrusions in that the roll-formed sections have a uniform thickness and the number of shapes that can be made is limited.

Like extrusions, roll-formed sections are made of many different types of materials. Sections used for replacement in the airplane must be of the same material as the original part.

IDENTIFICATION OF EXTRUSIONS AND ROLL-FORMED SECTIONS.

Extrusions are identified by a die or shape number. Identification numbers may be either those of the contractor or an Alcoa die number. Numbers that have identification letter "E" are contractor numbers. Roll-formed sections are identified by contractor number.

EXTRUSION AND ROLL-FORMED SECTION SUBSTITUTES.

Sheet-metal substitutes may be made for some of the less complicated extruded or roll-formed shapes. When substitutes are made for either extrusion or roll-formed sections, it is important that the material used in the substitute is the same as the original part, unless otherwise noted on the extrusion chart. Bend radii must be made according to standard sheet-metal practices. (Refer to index for bend radii table.) Either annealed or heat-treated material may be used. Annealed material is recommended if the material is over 0.064 inch thick. The 2024-O material should be heat-treated to a T4 condition and the 7075-O material should be heat-treated to a T6 condition after forming. If a sheet-metal substitution cannot be made because of the shape of the extrusion, the shape may be machined from bar stock. The bar stock material and heat-treatment must be the same as the original extrusion.

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EXTRUSIONS

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1E419	8-7	1
1E428	8-7	2
1E429	8-7	1
1E431	8-2	
1E433	8-1	
1E436	8-7	1

NAA DIE NO.	FIG. NO.	SHEET NO.
1E439		
1E440	8-7	2
1E443	8-2	
1E446	8-2	
1E448	8-2	
1E450	8-2	
1E451	8-2	
1E452	8-7	
1E453	8-7	
1E454	8-7	
1E455	8-2	
1E456	8-7	2
1E459	8-7	
1E460	8-7	
1E462	8-7	
1E472	8-7	
1E473	8-7	
1E474	8-7	
1E476	8-7	
1E477	8-7	
1E478	8-7	
1E480	8-2	
1E482	8-2	
1E485	8-7	
1E498	8-7	
1E499	8-2	
1E500	8-2	
1E510	8-7	
2E23	8-3	
2E44	8-4	
2E172	8-4	
2E177	8-7	2
2E185	8-3	
2E200	8-4	
2E202	8-7	6
2E221	8-3	
4E5	8-5	
4E88	8-5	
4E117	8-5	
4E162	8-7	6
4E188	8-5	
4E189	8-6	
4E222	8-5	
4E225	8-5	
4E239	8-5	
4E240	8-6	
4E242	8-5	
4E255	8-5	
4E280	8-6	
4E294	8-7	6
4E297	8-7	3
4E301	8-7	3
4E304	8-7	3

NAA DIE NO.	FIG. NO.	SHEET NO.
4E307	8-7	3
4E312	8-6	
4E314	8-7	4
4E322	8-7	4
4E323	8-7	4
4E324	8-7	5
4E327	8-7	5
4E328	8-7	5
4E329	8-7	5
4E332	8-7	6
4E334	8-6	
4E335	8-5	
4E344	8-6	
4E350	8-5	
4E351	8-5	
4E352	8-6	
4E356	8-7	7
4E358	8-5	
4E359	8-6	
4E360	8-6	

NAA DIE NO.	FIG. NO.	SHEET NO.
4E361	8-5	
4E362	8-5	
4E363	8-6	
4E373	8-6	
4E374	8-6	
4E378	8-6	
4E379	8-6	
4E380	8-6	
4E386	8-6	
4E397	8-6	
4E400	8-5	
5E191	8-7	7
6E171	8-7	7
6E514	8-7	7
6E524	8-7	8
6E548	8-7	8
6E558	8-7	9
6E564	8-7	8
6E565	8-7	8
6E567	8-7	9
6E568	8-7	9
6E576	8-7	9
6E579	8-7	10
6E580	8-7	10
6E585	8-7	11
6E587	8-7	11
6E600	8-7	11
6E613	8-7	10

ALCOA DIE NO.

AND 10133-0701	8-1	
AND 10133-0702	8-1	
AND 10133-1203	8-1	
AND 10133-1403	8-1	
AND 10133-1601	8-1	
AND 10134-0702	8-1	
AND 10134-1006	8-1	
AND 10134-1204	8-1	
AND 10134-1205	8-1	
AND 10134-1206	8-1	
AND 10134-1404	8-1	
AND 10134-1601	8-1	
AND 10134-2003	8-1	
AND 10134-2009	8-1	
AND 10136-1703	8-5	
AND 10136-2005	8-5	
AND 10136-2006	8-5	
AND 10136-2007	8-5	
AND 10137-2002	8-3	
77B	8-1	
77U	8-1	

ALCOA DIE NO.	FIG. NO.	SHEET NO.
78A	8-1	
78C	8-1	
78F	8-1	
78J	8-1	
78K	8-1	
78M	8-1	
78U	8-1	
78Y	8-1	
734-2	8-1	
734-6	8-1	
734N	8-1	
734GG	8-1	
734HH	8-1	
734JJ	8-1	
3660	8-5	
5688	8-3	
6494	8-1	
12327	8-1	
12871	8-1	
12883	8-1	
13630	8-5	
13816	8-1	
13870	8-5	
14044	8-1	
14263	8-1	
22487	8-5	
22885	8-1	
22897	8-5	
22969	8-1	
24826	8-5	
24877	8-5	
27918	8-1	
28554	8-1	
29140	8-5	
30663	8-1	

ROLL-FORMED SECTIONS

NAA DIE NO.

1S155	8-8	1
1S205-064	8-8	1
1S3	8-8	1
1S4-2	8-8	1
1S4-3	8-8	1
1S166	8-8	2
1S151	8-8	2
1S203-5	8-8	2
1S202-7	8-8	2
1S77	8-8	2
1S91	8-8	2

EXTRUSIONS

EXTRUSION CHARTS

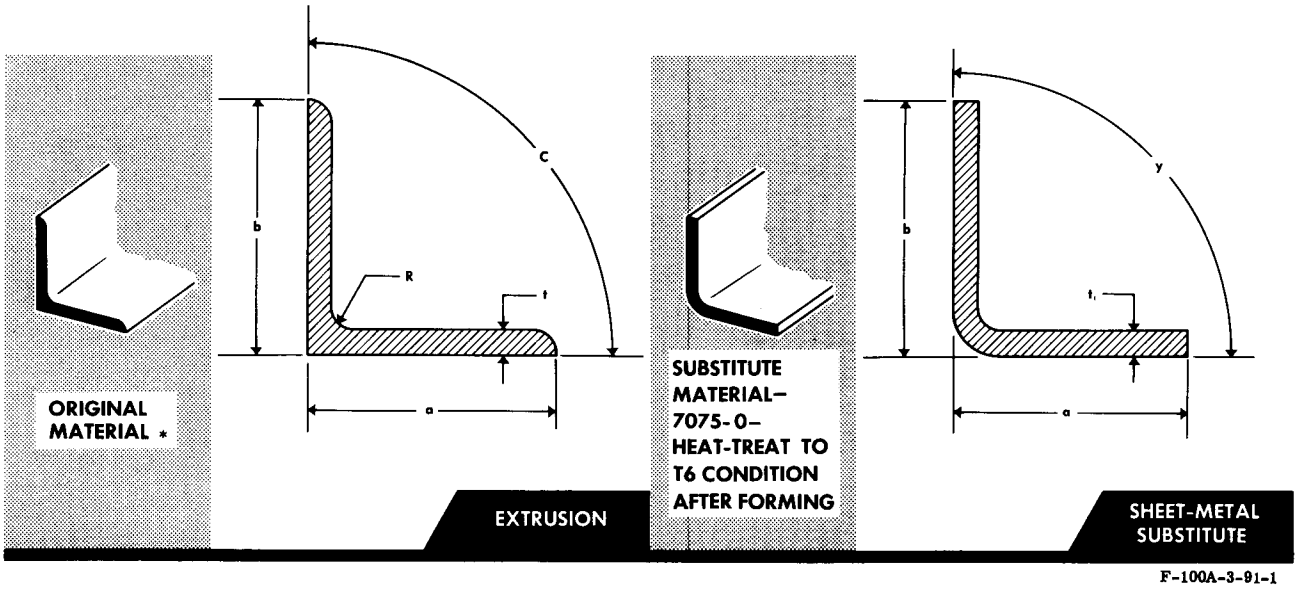

Figure 8-1. Angle—90-degree, Equal Thickness

EXTRUSION NO.		DIMENSIONS							
				y (ANGLE)					
ALCOA NO.	NAA NO.	a	b	DEG	MIN	R	t	t_1	*MATL
28002	1E14	1.125	0.750	90		0.125	0.125	0.156	7075-T6
27890	1E24	1.750	1.000	84	30	0.094	0.094	0.102	2024-T4
30006	1E38	2.125	1.000	90		0.094	0.125	0.156	2024-T4
29147	1E44	2.000	1.750	90		0.094	0.156	0.188	7075-T6
29146	1E45	2.000	1.625	90		0.094	0.125	0.156	7075-T6
29033	1E49	1.000	1.000	84		0.094	0.125		2024-T4
24512	1E77	1.000	1.000	106		0.125	0.094	0.102	2024-T4
24511	1E78	1.125	1.125	125		0.188	0.125	0.156	7075-T6
16869	1E87	0.875	0.750	90		0.063	0.125		7075-T6
14870	1E95	1.250	0.750	83		0.125	0.094	0.102	2024-T4
14654	1E100	0.813	0.625	90		0.094	0.078	0.091	7075-T6
734JJ	1E110	1.000	0.750	90		0.063	0.125	0.156	7075-T6 2024-T42
30725	1E121	2.000	1.125	85		0.156	0.093	0.102	2024-T4
30796	1E122	1.125	1.000	90		0.125	0.125	0.156	7075-T6
31154	1E129	1.000	0.875	90		0.063	0.125		7075-T6
44259	1E214	2.000	1.000	90		0.188	0.062	0.072	7075-T6
44451	1E216	1.375	1.250	90		0.062	0.062	0.072	2024-T42 7075-T6
44334	1E219	1.000	0.750	90		0.094	0.040	0.051	2024-T42
51325	1E233	3.000	1.375	90		0.156	0.125	0.156	2024-T42 7075-T6
44767	1E237	1.000	1.000	82	30	0.125	0.094	0.102	2024-T42
	1E241	1.000	1.000	105		0.125	0.125		2024-T4
44842	1E267	2.000	1.156	90		0.062	0.078	0.091	7075-T6
44843	1E268	2.000	1.156	90		0.062	0.094	0.102	7075-T6 2024-T42

Key to Figure 8-1

EXTRUSION NO.		DIMENSIONS							
				γ (ANGLE)					
ALCOA NO.	NAA NO.	a	b	DEG	MIN	R	t	t_1	*MATL
59566	1E296	1.125	1.125	90		0.125	0.156		7075-T6
R-8140	1E321	2.188	1.125	93	10	0.125	0.156		7075-T6
60080	1E322	1.000	0.875	85	45	0.094	0.094	0.102	7075-T6
									2024-T42
57847	1E324	1.188	1.125	94	15	0.125	0.125	0.156	7075-T6
	1E328	1.000	1.000	105		0.094	0.078	0.091	7075-T6
59858	1E345	1.812	1.188	99	45	0.188	0.188		7075-T6
56776	1E371	1.125	1.125	92	45	0.125	0.094	0.102	7075-T6
	1E378	2.125	1.875	93		0.125	0.125	0.156	7075-T6
R-7933	1E403	2.000	0.875	93	30	0.125	0.078	0.091	2024-T4
60250	1E408	1.312	1.312	90		0.125	0.188		7075-T6
R-8160	1E409	1.063	1.000	90		0.063	0.125		7075-T6
R-8157	1E410	1.063	1.000	90		0.063	0.125		7075-T6
R-8156	1E411	1.000	0.875	98	50	0.063	0.125		7075-T6
REY8378	1E412	1.125	1.125	90		0.125	0.188		7075-T6
	1E413	3.500	2.312	90		0.125	0.125	0.156	7075-T6
R-8126	1E415	1.000	1.000	112		0.125	0.125		7075-T6
	1E416	2.000	1.500	90		0.062	0.094	0.102	2024-T42
R-8159	1E417	1.750	1.250	90		0.125	0.125	0.156	7075-T6
	1E433	2.325	2.312	90		0.125	0.125	0.156	7075-T6
	1E452	1.375	1.688	97		0.188	0.188		7075-T6
	1E453	1.375	1.000	87		0.062	0.062	0.072	7075-T6
	1E454	1.375	1.000	93		0.062	0.062	0.072	7075-T6
	1E459	1.875	0.875	90		0.094	0.094	0.102	7075-T6
	1E460	1.000	0.938	95	50	0.125	0.062	0.072	7075-T6
	1E462	1.625	1.000	72	30	0.125	0.125	0.156	7075-T6
									2024-T42
R-8471	1E472	2.000	1.750	85	45	0.094	0.125	0.156	7075-T6
									2024-T42
	1E473	0.875	0.625	90		0.062	0.125		7075-T6
	1E474	1.250	1.000	86	56	0.125	0.125		7075-T6
									2024-T42
	1E476	1.812	1.625	86	5	0.094	0.125	0.156	7075-T6
	1E477	2.750	1.062	79		0.188	0.188		7075-T6
	1E478	2.750	1.125	101		0.188	0.188		7075-T6
									2024-T42
	1E485	1.000	0.875	98	50	0.094	0.094	0.102	7075-T6
	1E498	1.000	1.250	81		0.156	0.125	0.156	7075-T6
	1E510	1.125	1.000	91	50	0.125	0.125	0.156	7075-T6
77B		1.000	1.000	90		0.125	0.125		7075-T6
									2024-T42
77U		1.750	1.750	90		0.188	0.188		7075-T6
78A		1.250	1.250	90		0.188	0.188		7075-T6
78C		0.750	0.750	90		0.125	0.094	0.102	2024-T42
78F		1.000	1.000	90		0.125	0.094	0.102	7075-T6
									2024-T42
78J		1.000	1.000	90		0.062	0.062	0.072	7075-T6
									2024-T42
78K		0.750	0.750	90		0.125	0.063	0.072	7075-T6
									2024-T42
78M		1.000	1.000	90		0.125	0.188		2024-T42
78U		1.125	1.125	90		0.188	0.125	0.156	7075-T6
78Y		1.250	1.250	90		0.094	0.094	0.102	7075-T6
AND 10133-0701		0.875	0.875	90		0.094	0.063		
AND 10133-0702		0.875	0.875	90		0.094	0.094		
AND 10133-1203		1.250	1.250	90		0.188	0.125	0.156	7075-T6
AND 10133-1403		1.500	1.500	90		0.188	0.125	0.156	7075-T6

Key to Figure 8-1 (Continued)

EXTRUSION NO.		DIMENSIONS							
ALCOA NO.	NAA NO.	a	b	y (ANGLE) DEG	MIN	R	t	t_1	*MATL
AND 10133-1601		1.750	1.750	90		0.250	0.094		7075-T6
									2024-T42
AND 10134-0702		0.875	0.750	90		0.125	0.063	0.072	7075-T6
AND 10134-1006		1.000	0.875	90		0.125	0.094	0.102	2024-T42
AND 10134-1204		1.250	1.000	90		0.125	0.063	0.072	7075-T6
AND 10134-1205		1.250	1.000	90		0.125	0.094	0.102	7075-T6
AND 10134-1206		1.250	1.000	90		0.125	0.125	0.156	7075-T6
AND 10134-1404		1.500	1.000	90		0.156	0.125	0.156	7075-T6
AND 10134-1601		1.750	1.000	90		0.156	0.125	0.156	7075-T6
									2024-T42
AND 10134-2003		2.000	1.250	90		0.188	0.125	0.156	7075-T6
									2024-T42
AND 10134-2009		2.000	1.750	90		0.188	0.125	0.156	7075-T6
734-2		1.625	1.250	90		0.125	0.125	0.156	7075-T6
734-6		2.500	1.250	90		0.188	0.125	0.156	7075-T6
734GG		1.500	1.000	90		0.156	0.156		7075-T6
									2024-T42
734HH		1.250	1.000	90		0.125	0.125	0.156	7075-T6
734N		2.000	1.250	90		0.188	0.188		7075-T6
6494		1.750	1.125	90		0.125	0.064	0.072	7075-T6
12327		2.255	1.500	93	58	0.156	0.156	0.188	7075-T6
									2024-T42
12871		0.813	0.813	90		0.063	0.062	0.072	7075-T6
12883		0.750	0.625	90		0.063	0.051	0.064	7075-T6
									2024-T42
13816		3.500	1.750	90		0.156	0.093	0.102	7075-T6
14044		1.000	0.750	90		0.125	0.094	0.102	7075-T6
14263		1.750	1.000	90		0.094	0.094	0.102	7075-T6
22885		1.000	0.750	90		0.063	0.063	0.072	7075-T6
22969		1.125	0.756	90		0.125	0.062	0.072	7075-T6
27918		1.500	1.250	90		0.125	0.094	0.102	2024-T42
									7075-T6
28554		1.000	1.000	97		0.125	0.093	0.102	7075-T6
									2024-T42
30663		2.250	1.000	90		0.094	0.093	0.102	7075-T6

Key to Figure 8-1 (Continued)

EXTRUSIONS

EXTRUSION CHARTS

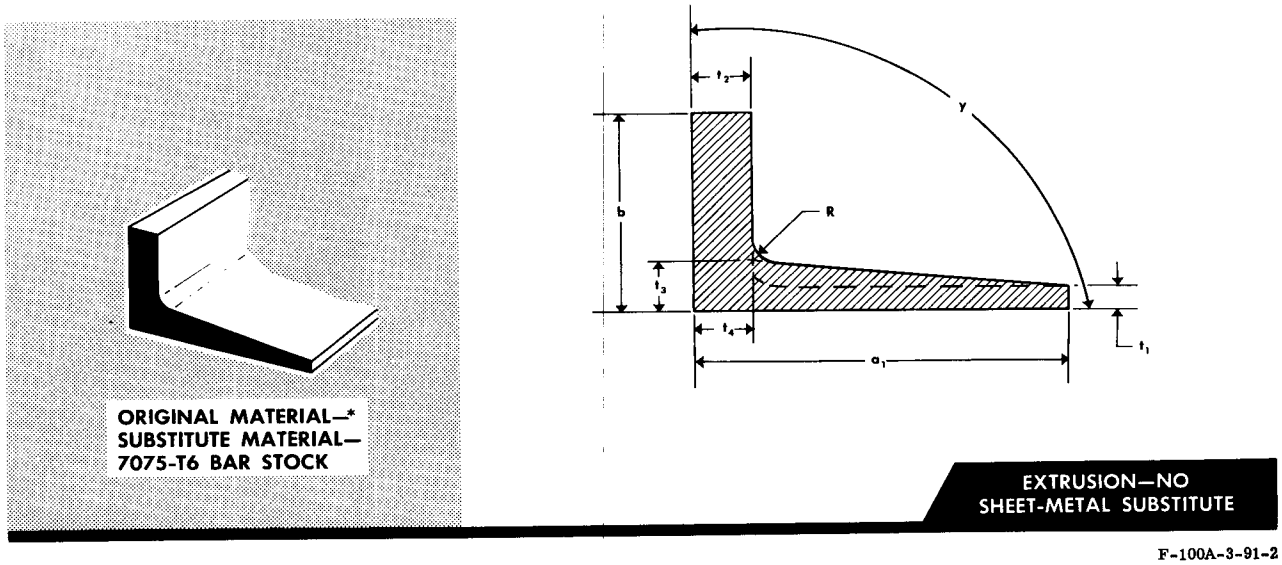

Figure 8-2. Angle—90-degree, Tapered Leg, Unequal Thickness

EXTRUSION NO.		DIMENSIONS									
									y (ANGLE)		
ALCOA NO.	NAA NO.	a	b	t_1	t_2	t_3	t_4	R	DEG	MIN	*MATL
24766	1E67	2.000	1.063	0.125	0.313	0.250		0.188			7075-T6
24578	1E75	5.500	1.375	0.094	0.438	0.344	0.618	0.500			7075-T6
15626	1E93	1.125	1.000	0.188	0.344			0.125			7075-T6
62098	1E155	3.875	1.125	0.188	0.250			0.188			7075-T6
31928	1E157	6.500	1.625	0.094	0.500	0.375		0.500			7075-T6
44848	1E270	1.375	1.281	0.125	0.250			0.250			2024-T42
59912	1E356	1.125	1.000	0.156	0.125			0.125			7075-T6
											2024-T42
	1E383	3.312	0.938	0.125	0.188			0.250			7075-T6
60169	1E385	2.000	1.125	0.062	0.125	0.125		0.125			7075-T6
R-7809	1E391	1.625	1.250	0.188	0.250			0.188	150		7075-T6
R-7825	1E392	1.250	1.125	0.125	0.188			0.156			7075-T6
	1E404	2.781	1.562	0.064	0.188	0.328	0.300	0.750	86	30	7075-T6
	1E405	2.781	1.562	0.064	0.188	0.312	0.263	0.750	93	30	2024-T42
	1E431	4.250	1.375	0.094	0.438	0.250		0.375			7075-T6
R-8375	1E439	4.844	1.312	0.062	0.453	0.281	0.625	0.500			7075-T6
	1E443	1.406	0.906	0.188	0.094			0.125			7075-T6
	1E446	3.000	1.062	0.094	0.250	0.250		0.188			2024-T42
	1E448	3.250	1.125	0.188	0.250			0.188	87	15	2024-T42
	1E450	4.688	1.375	0.094	0.375	0.281		0.500			7075-T6
	1E451	3.438	1.375	0.094	0.375	0.281		0.500			7075-T6
											2024-T42
	1E455	0.844	0.781	0.094	0.250			0.062			7075-T6
											2024-T42
	1E480	1.500	1.375	0.188	0.125			0.125			7075-T6
	1E482	3.688	1.625	0.219	0.438	0.375		0.500			7075-T6
	1E499	2.062	1.375	0.125	0.312			0.500	109	42	7075-T6
	1E500	2.000	1.312	0.125	0.312			0.218			7075-T6

Key to Figure 8-2

EXTRUSIONS

EXTRUSION CHARTS

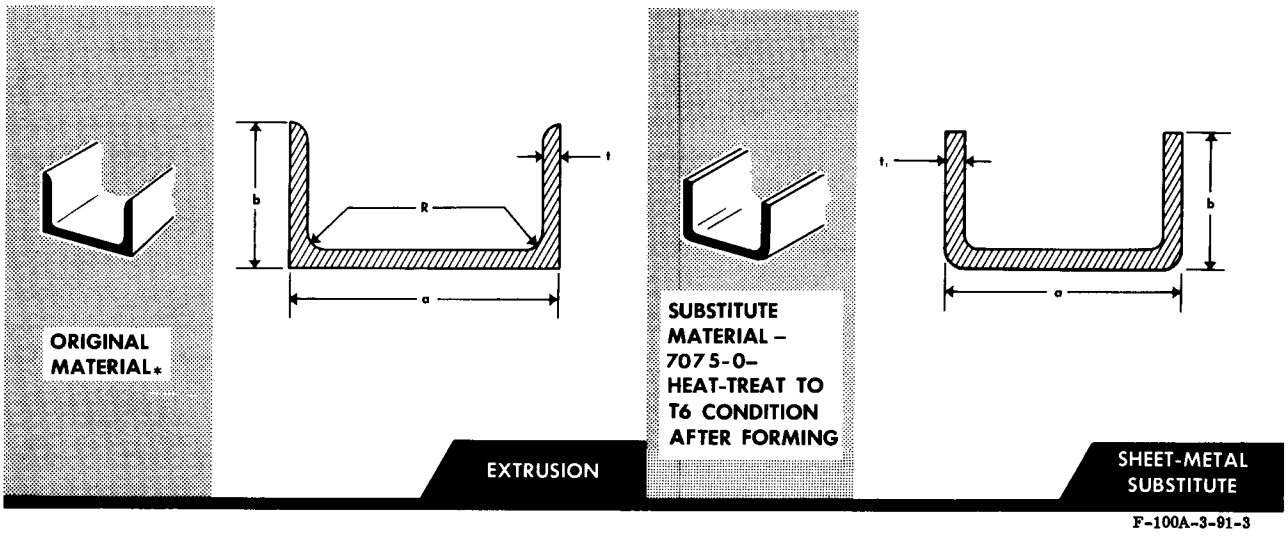

Figure 8-3. Channel—90-degree, Equal Thickness

EXTRUSION NO.		DIMENSIONS					
ALCOA NO.	NAA NO.	a	b	t	t_1	R	*MATL
24886	2E23	1.500	0.875	0.080	0.091	0.063	7075-T6
R-7826	2E185	1.875	1.000	0.125	0.156	0.125	7075-T6
	2E221	2.000	1.000	0.188		0.188	7075-T6
AND 10137-2002		2.000	0.750	0.125	0.156	0.125	7075-T6
5688		1.500	1.000	0.188		0.016	7075-T6

Key to Figure 8-3

EXTRUSIONS

EXTRUSION CHARTS

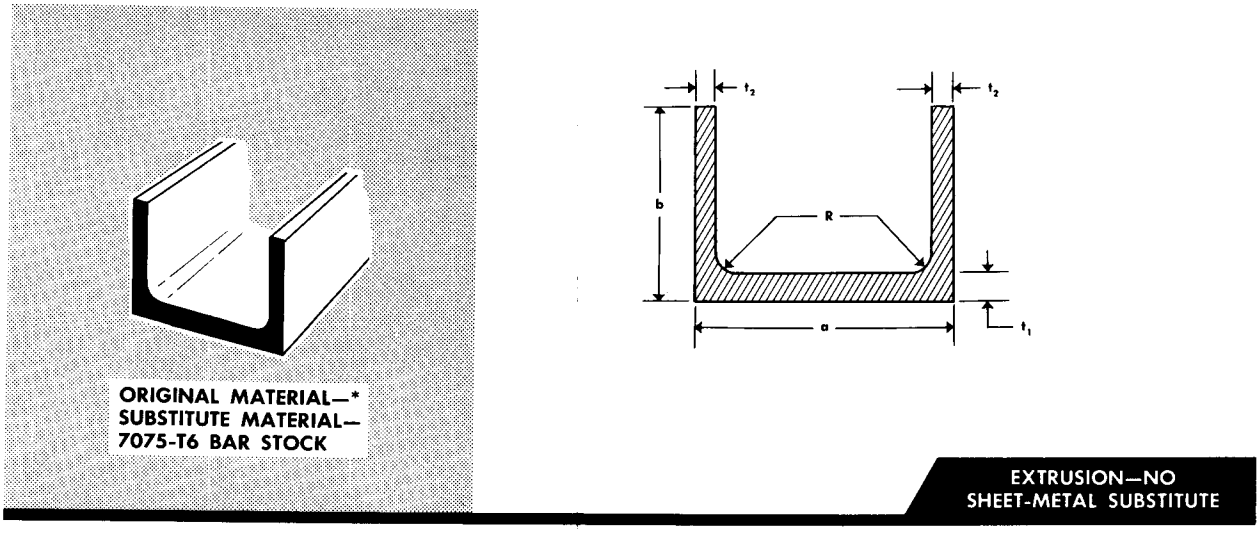

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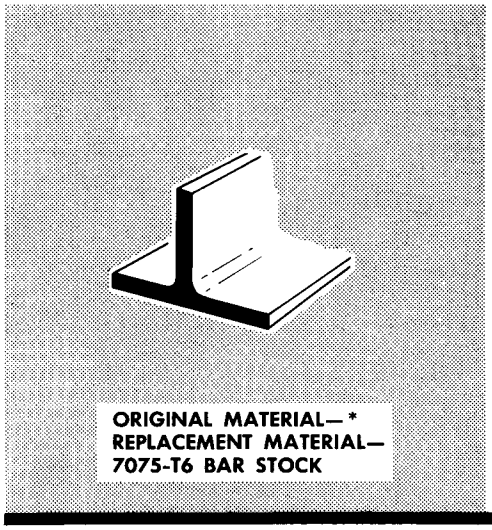
Figure 8-4. Channel—90-degree, Unequal Thickness

EXTRUSION NO.		DIMENSIONS					
ALCOA NO.	NAA NO.	a	b	t_1	t_2	R	*MATL
	2E44	0.875	1.500	0.156	0.093	0.156	7075-T6
	2E172	3.375	3.000	0.156	0.312	0.125	2024-T42
	2E200	1.062	2.188	0.250	0.156	0.094	2024-T42

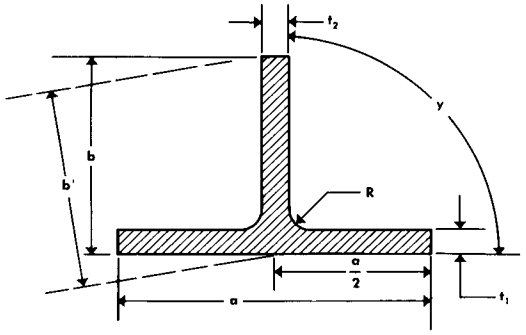
Key to Figure 8-4

EXTRUSIONS

EXTRUSION CHARTS



ORIGINAL MATERIAL—*
REPLACEMENT MATERIAL—
7075-T6 BAR STOCK



EXTRUSION—NO
SHEET-METAL SUBSTITUTE

F-100A-3-91-5

Figure 8-5. "T"—90-degree, Unequal Thickness

EXTRUSION NO.		DIMENSIONS								
							y (ANGLE)			
ALCOA NO.	NAA NO.	a	b	R	t_1	t_2	DEG	MIN	b^1	*MATL
23965	4E5	1.250	1.250	0.094	0.125	0.125	90			2024-T42
44281	4E117	1.750	1.000	0.094	0.062	0.062	90			7075-T6
	4E188	1.625	2.000	0.156	0.188	0.125	90			7075-T6
62399	4E222	2.000	1.000	0.125	0.062	0.062	90			7075-T6
										2024-T42
60038	4E225	1.750	1.750	0.125	0.125	0.125	90			7075-T6
	4E239	1.500	1.500	0.031	0.062	00.62	90			2024-T4
	4E242	1.875	2.125	0.125	0.125	0.125	90			7075-T6
R-8381	4E350	2.000	1.562	0.125	0.078	0.078	90			7075-T6
R-8379	4E351	1.750	1.562	0.125	0.078	0.078	90			7075-T6
R-8477	4E358	2.500	1.250	0.125	0.156	0.156	90			7075-T6
	4E361	1.500	0.933	0.063	0.063	0.063	90			2024-T42
	4E400	1.875	1.062	0.125	0.062	0.062	90			7075-T6
										2024-T4
43373	4E88	2.000		0.125	0.125	0.125	100	34	1.000	2024-T42
	4E255	1.750		0.063	0.063	0.063	108	35	1.000	7075-T6
	4E335	1.500		0.062	0.109	0.078	106		1.812	2024-T4
	4E362	3.000		0.156	0.156	0.156	91	26	1.375	7075-T6
AND 10136-1703		1.875	1.000	0.125	0.125	0.125	90			7075-T6
AND 10136-2005		2.000	1.250	0.125	0.094	0.094	90			7075-T6
										2024-T42
AND 10136-2006		2.000	1.250	0.125	0.125	0.125	90			2024-T42
AND 10136-2007		2.000	1.750	0.125	0.094	0.094	90			7075-T6
3660		1.500	1.000	0.063	0.094	0.094	90			2024-T42
13630		1.937	3.250	0.156	0.125	0.125	90			7075-T6
13870		1.500	1.000	0.062	0.125	0.187	90			2024-T42

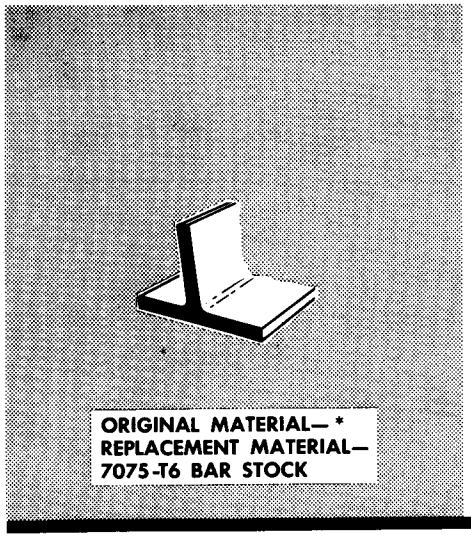
Key to Figure 8-5

EXTRUSION NO.		DIMENSIONS								
ALCOA NO.	NAA NO.	a	b	R	t_1	t_2	y (ANGLE) DEG	MIN	b^1	*MATL
22487		3.000	1.375	0.187	0.156	0.250	90			7075-T6
22897		1.500	0.750	0.062	0.062	0.062	90			7075-T6
24826		1.500	1.250	0.063	0.063	0.063	90			7075-T6 2024-T42
24877		1.500	0.750	0.125	0.050	0.050	90			2024-T42
29140		1.750	1.000	0.125	0.090	0.090	90			7075-T6

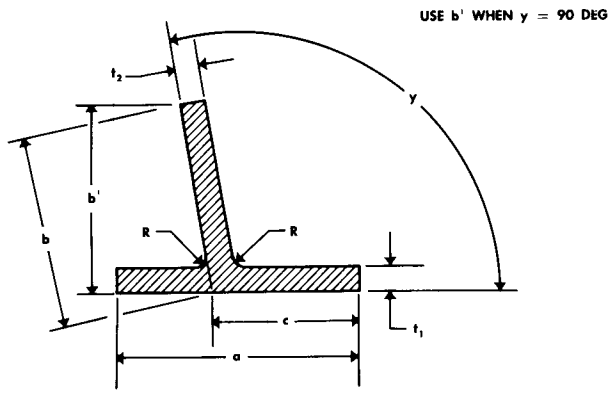
Key to Figure 8-5 (Continued)

EXTRUSIONS

EXTRUSION CHARTS



ORIGINAL MATERIAL— *
REPLACEMENT MATERIAL—
7075-T6 BAR STOCK



EXTRUSION—NO SHEET-METAL SUBSTITUTE

F-100A-3-91-6

Figure 8-6. "T"—Angular Center Web, Unequal Thickness

EXTRUSION NO.		DIMENSIONS									
					y (ANGLE)						
ALCOA NO.	NAA NO.	a	b	c	DEG	MIN	R	t_1	t_2	b^1	*MATL
52800	4E189	3.000	2.563	1.438	106		0.125	0.125	0.125		7075-T6
	4E240	3.875		2.219	90		0.156	0.156	0.188	4.625	7075-T6
	4E280	3.000		1.578	90		0.125	0.078	0.078	1.562	2024-T4
	4E312	1.875		1.125	90		0.064	0.375	0.250	1.188	7075-T6
	4E334	2.688		1.750	90		0.094	0.094	0.094	0.875	7075-T6
60464	4E344	2.500	2.500	1.500	78		0.250	0.312	0.250		7075-T6
	4E352	1.938		1.312	90		0.344	0.125	0.188	1.625	2024-T42
	4E359	2.500		1.281	90		0.156	0.156	0.188	3.500	2024-T42
	4E360	2.500		1.188	90		0.156	0.156	0.188	4.625	7075-T6
	4E363	2.500	2.438	1.625	92	30	0.156	0.156	0.125		7075-T6
	4E373	0.920		0.618	90		0.150	0.812	0.188	3.312	7075-T6
	4E374	2.375	2.688	1.062	101		0.125	0.188	0.188		2024-T42
	4E378	1.562		0.562	109		0.125	0.188	0.188	1.000	2024-T42
	4E379	2.938		1.540	90		0.125	0.102	0.102	0.750	7075-T6 2024-T42
	4E380	2.094		1.078	90		0.156	0.156	0.156	1.250	7075-T6 2024-T42
	4E386	2.938	2.531	1.844	87	18	0.250	0.125	0.156		7075-T6
	4E397	1.688	1.823	0.977	74		0.062	0.109	0.078		2024-T42

Key to Figure 8-6

EXTRUSIONS

EXTRUSION CHARTS

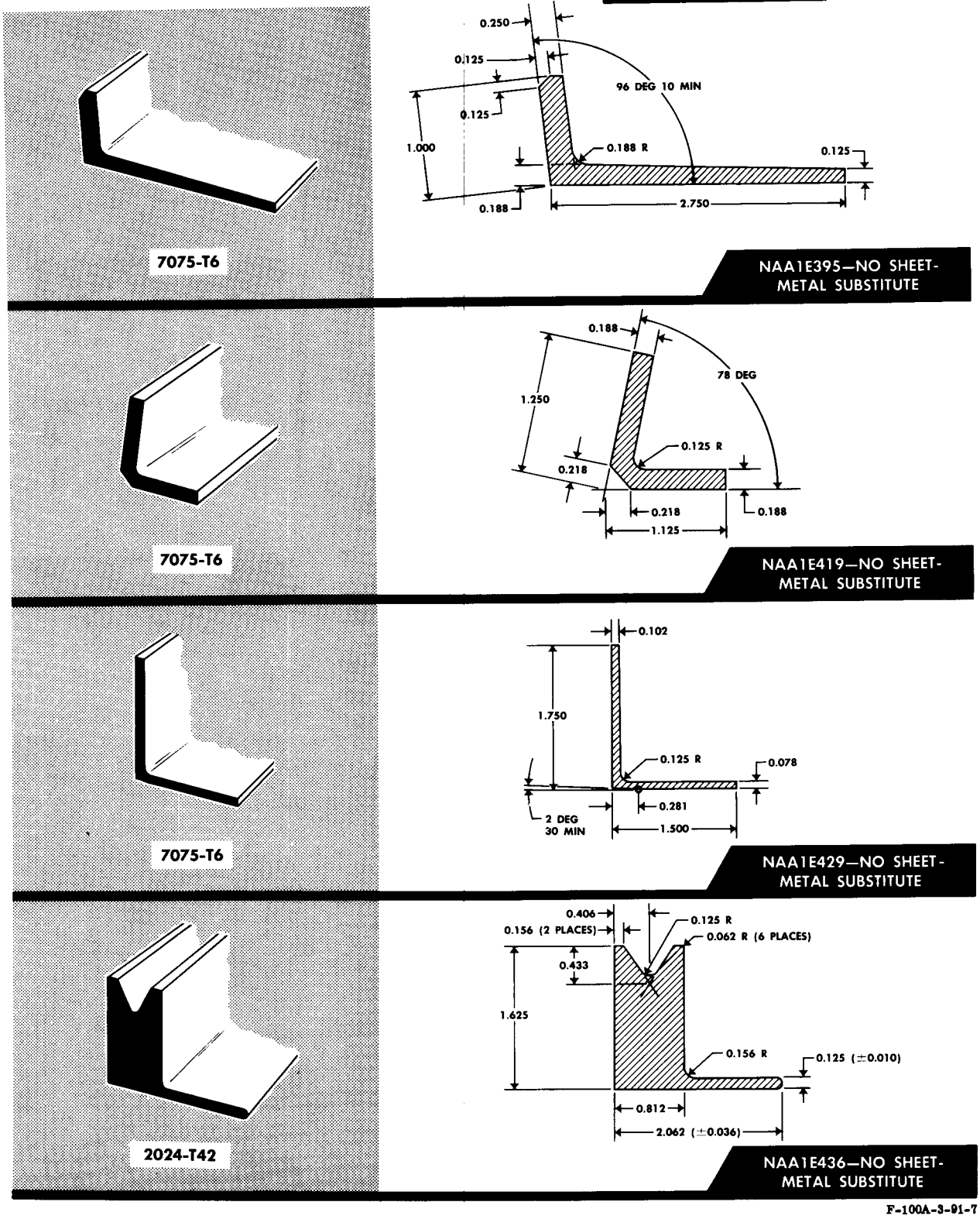


Figure 8-7. Extrusions (Sheet 1 of 11)

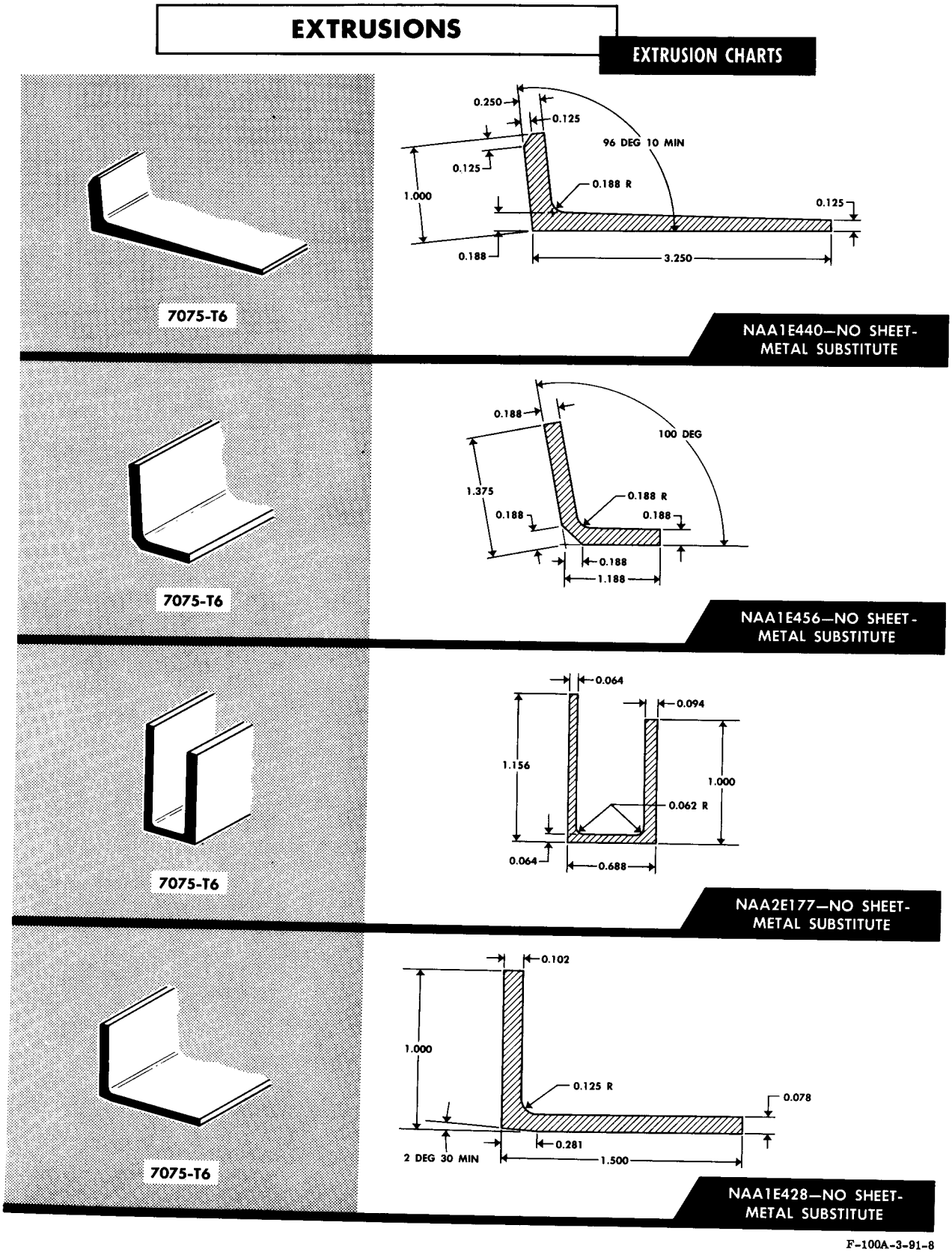
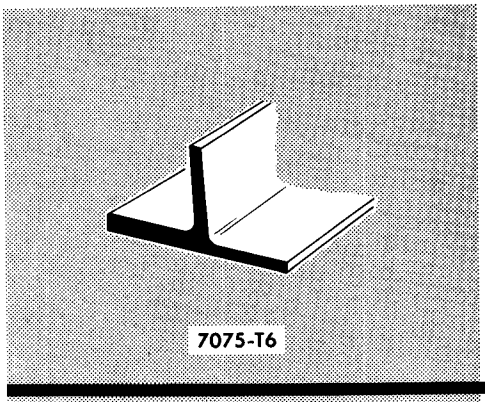


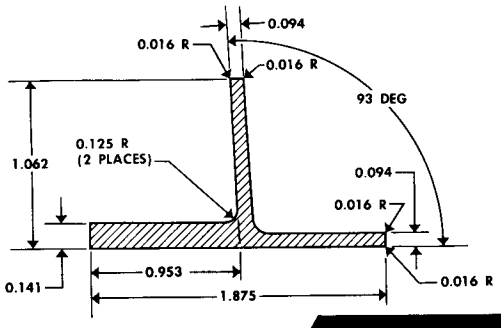
Figure 8-7. Extrusions (Sheet 2 of 11)

EXTRUSIONS

EXTRUSION CHARTS



7075-T6



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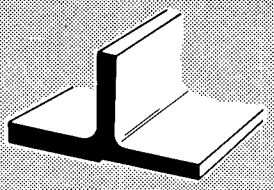
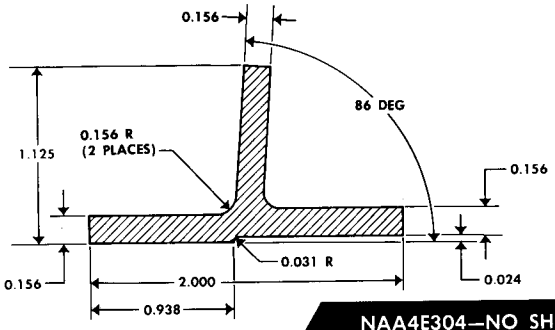


7075-T6



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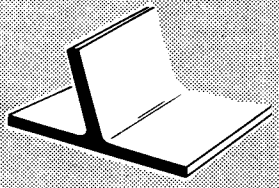
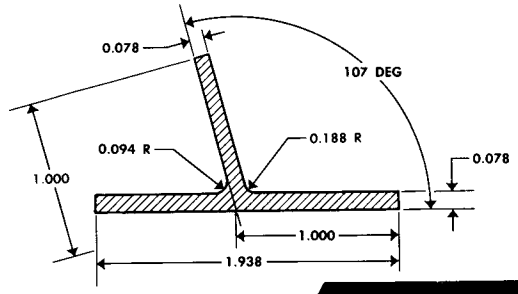


7075-T6



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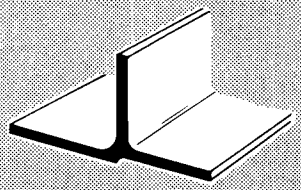
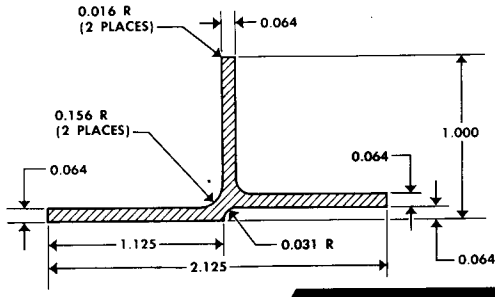


7075-T6



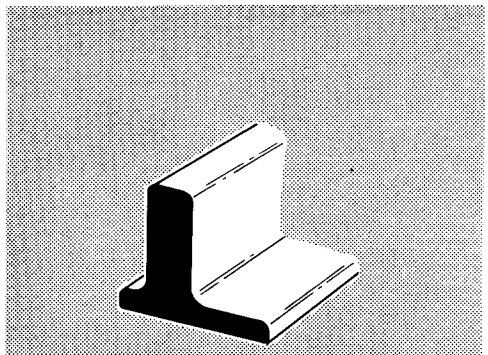
NAA4E297—NO SHEET-METAL SUBSTITUTE

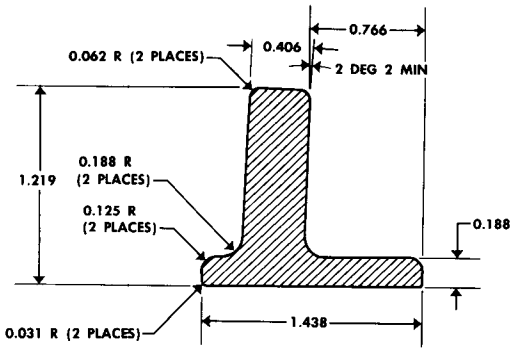
F-100A-3-91-9

Figure 8-7. Extrusions (Sheet 3 of 11)

EXTRUSIONS

EXTRUSION CHARTS





7075-T6

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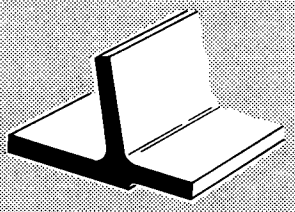

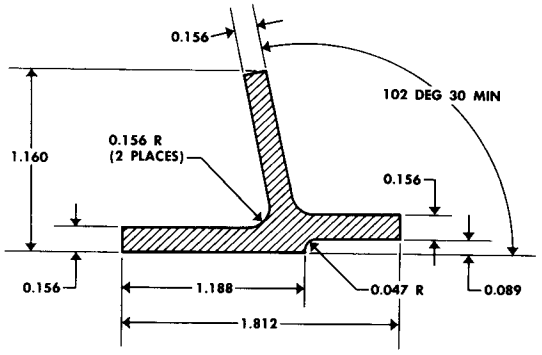

7075-T6

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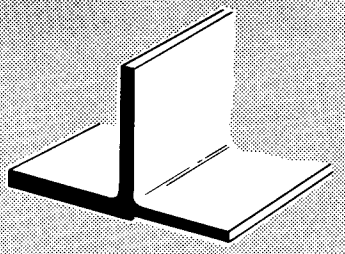
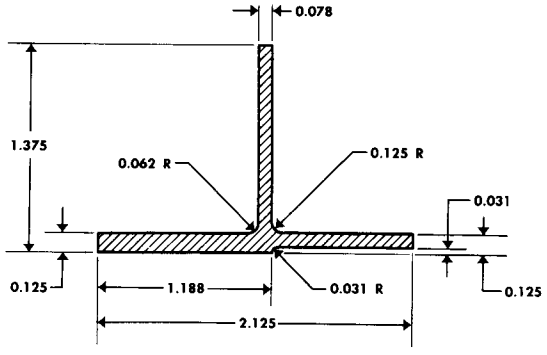




7075-T6

NAA4E323—NO SHEET-METAL SUBSTITUTE

F-100A-3-91-10

Figure 8-7. Extrusions (Sheet 4 of 11)

EXTRUSIONS

EXTRUSION CHARTS

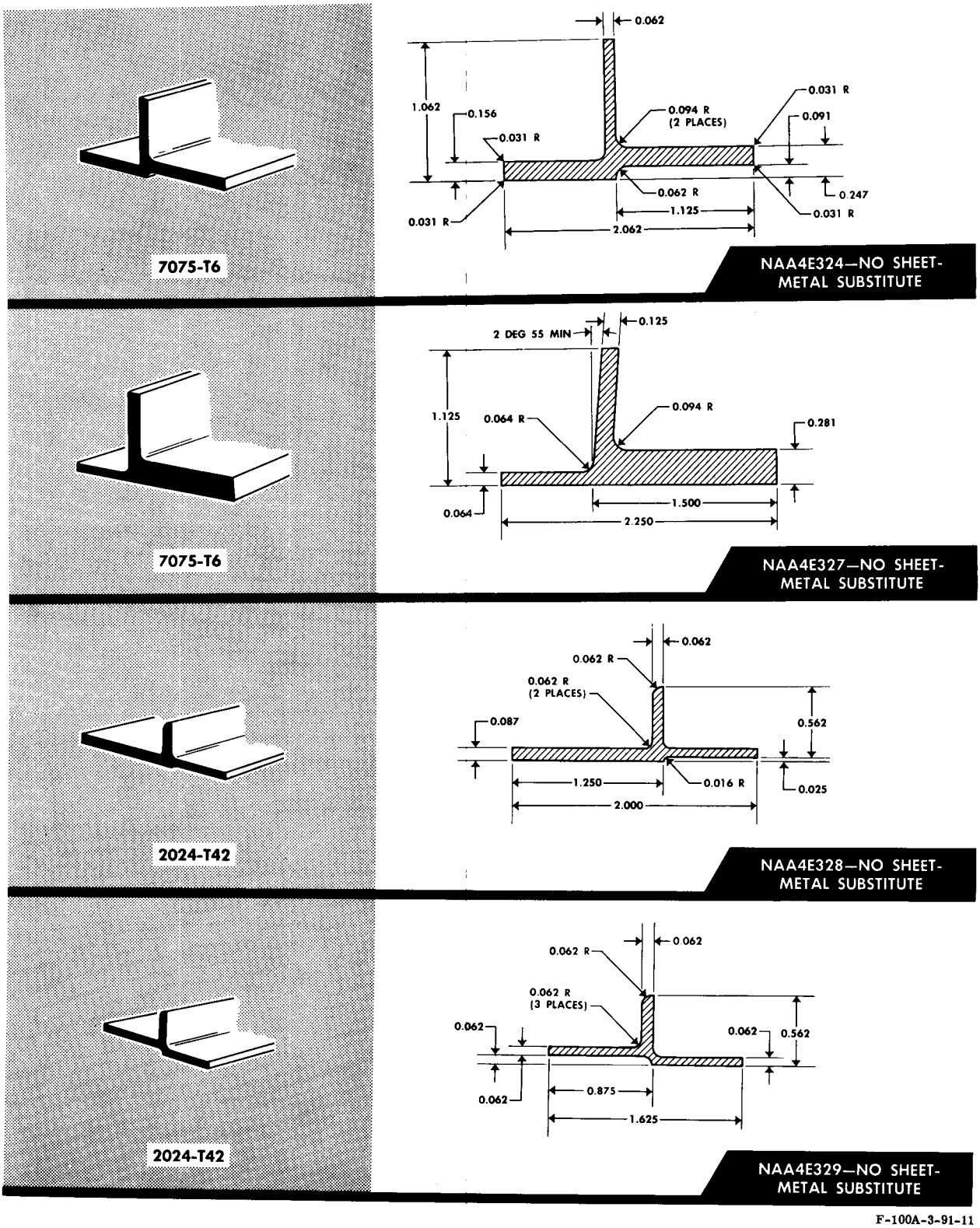


Figure 8-7. Extrusions (Sheet 5 of 11)

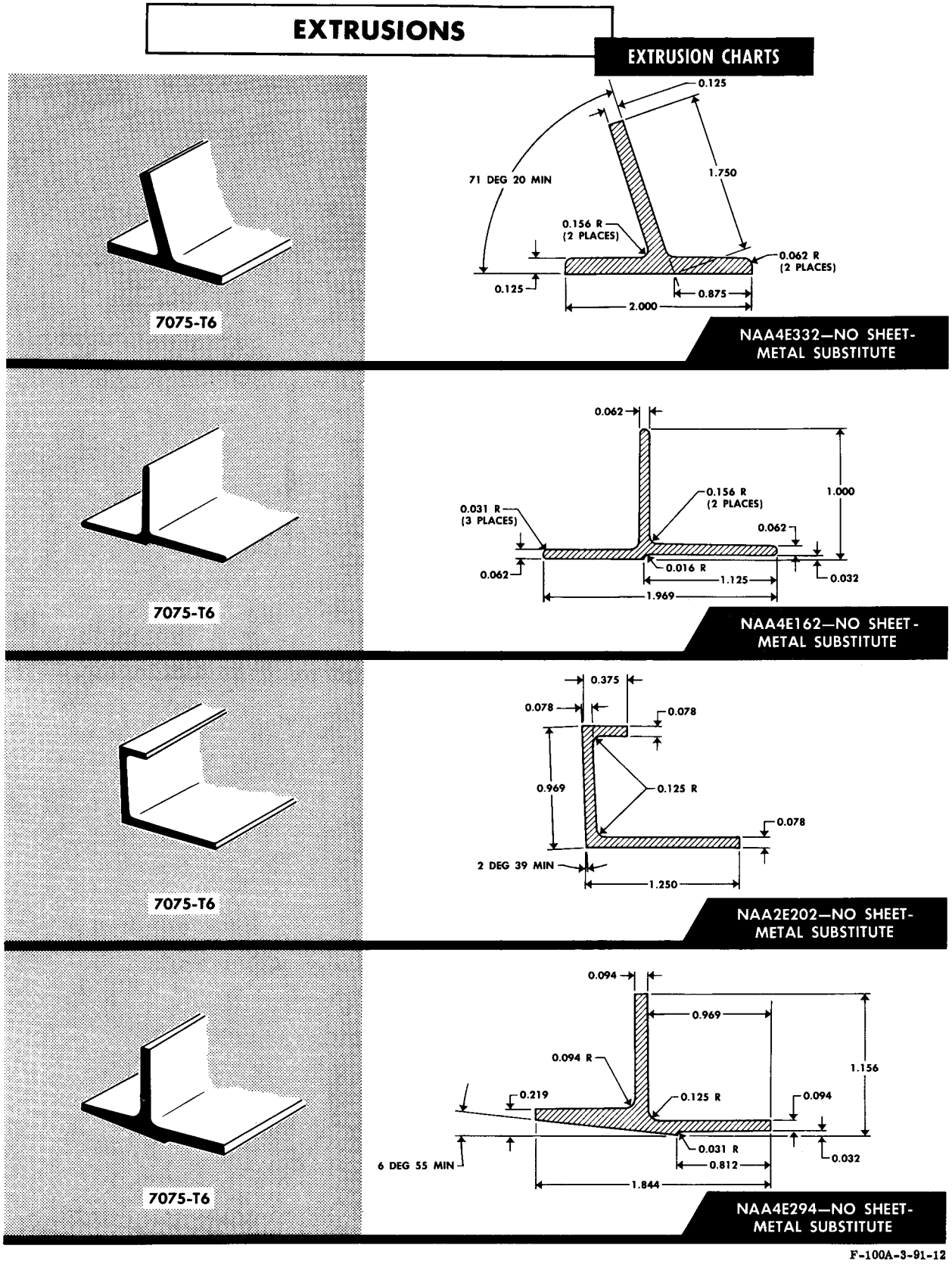
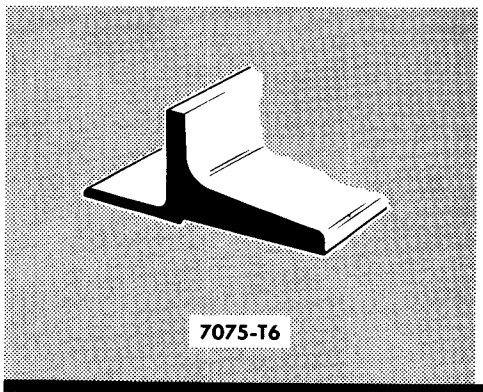


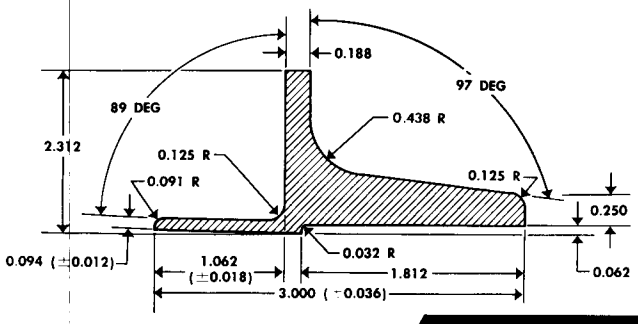
Figure 8-7. Extrusions (Sheet 6 of 11)

EXTRUSIONS

EXTRUSION CHARTS



7075-T6



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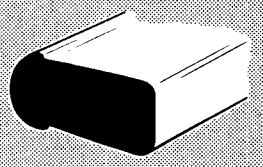
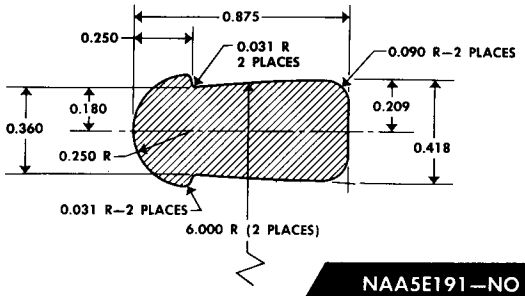


7075-T6



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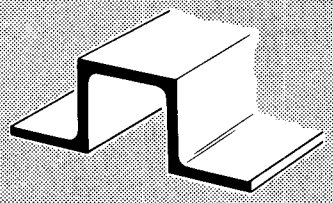
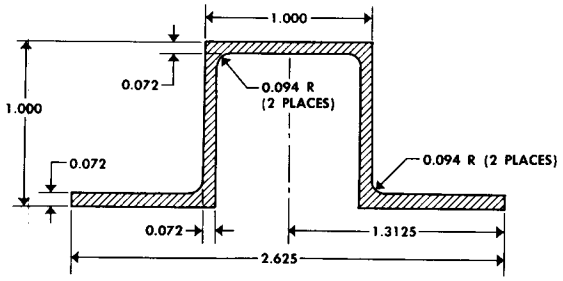


7075-T6



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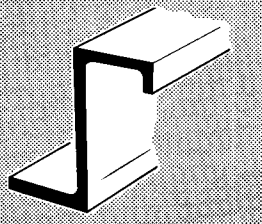
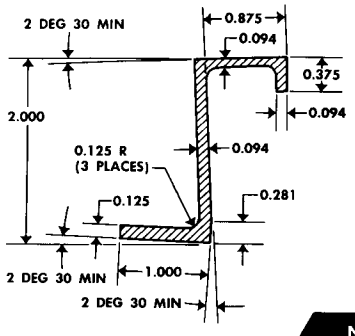


7075-T6



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F-100A-3-91-13

Figure 8-7. Extrusions (Sheet 7 of 11)

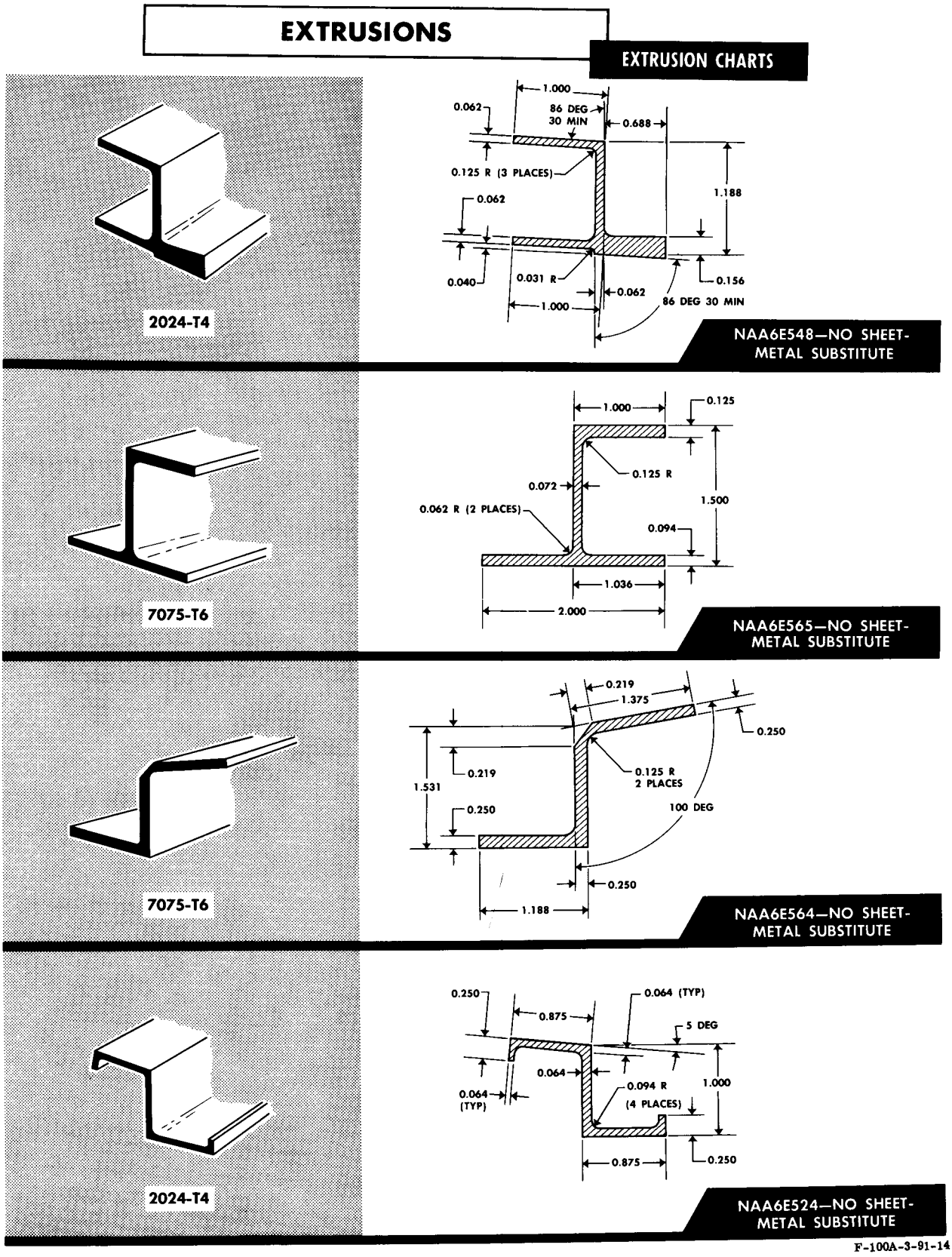
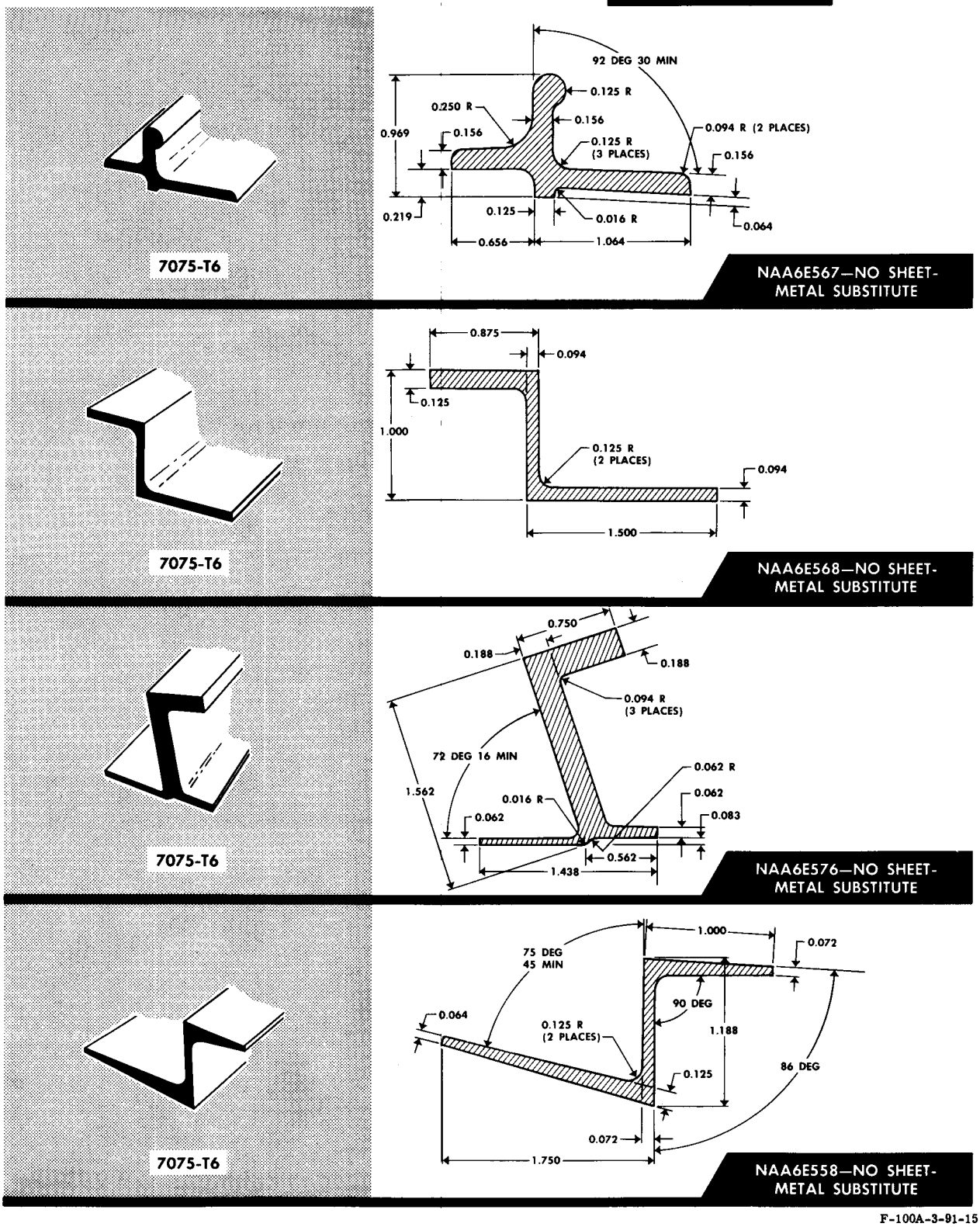


Figure 8-7. Extrusions (Sheet 8 of 11)

EXTRUSIONS

EXTRUSION CHARTS



F-100A-3-91-15

Figure 8-7. Extrusions (Sheet 9 of 11)

EXTRUSIONS

EXTRUSION CHARTS

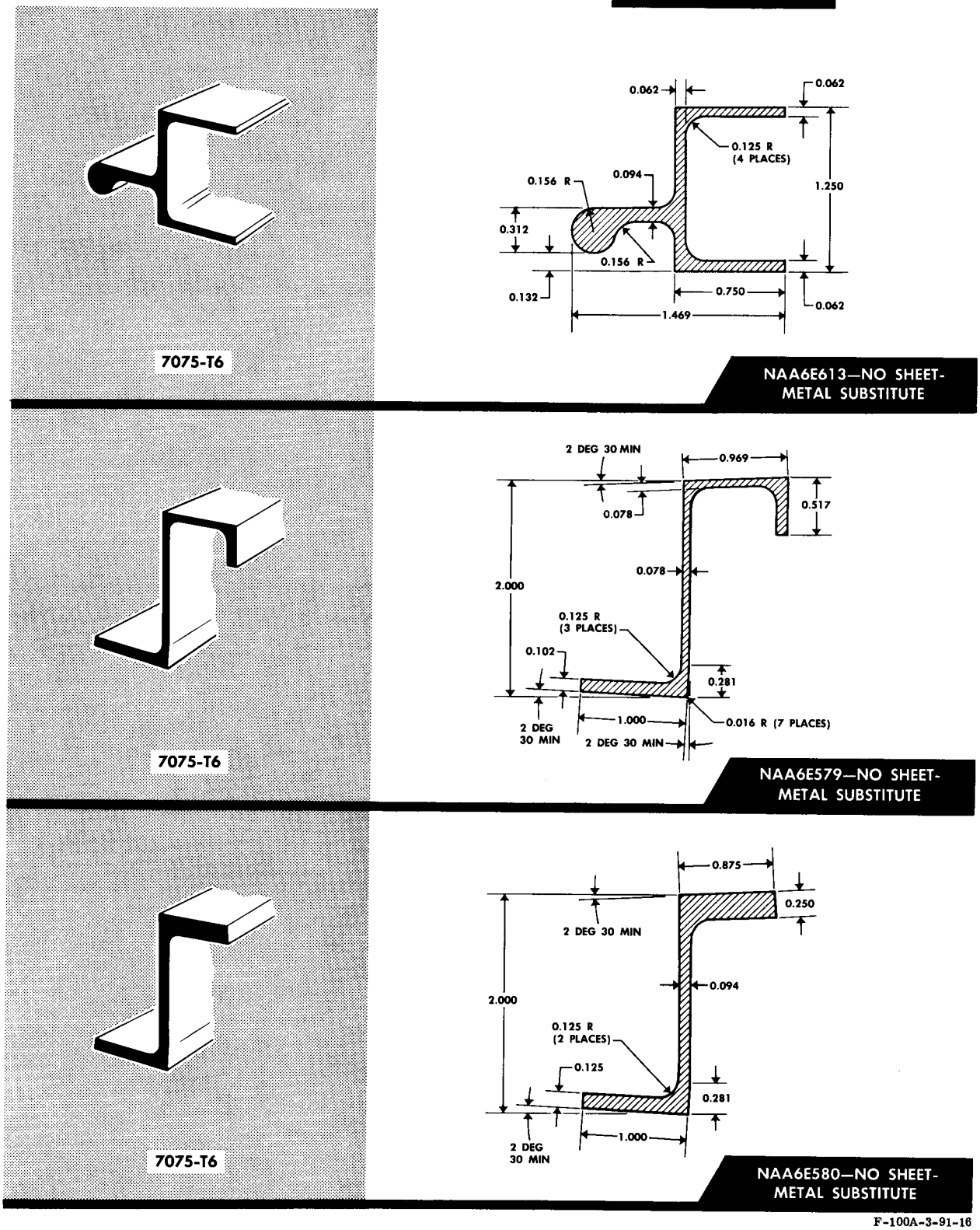


Figure 8-7. Extrusions (Sheet 10 of 11)

EXTRUSIONS

EXTRUSION CHARTS

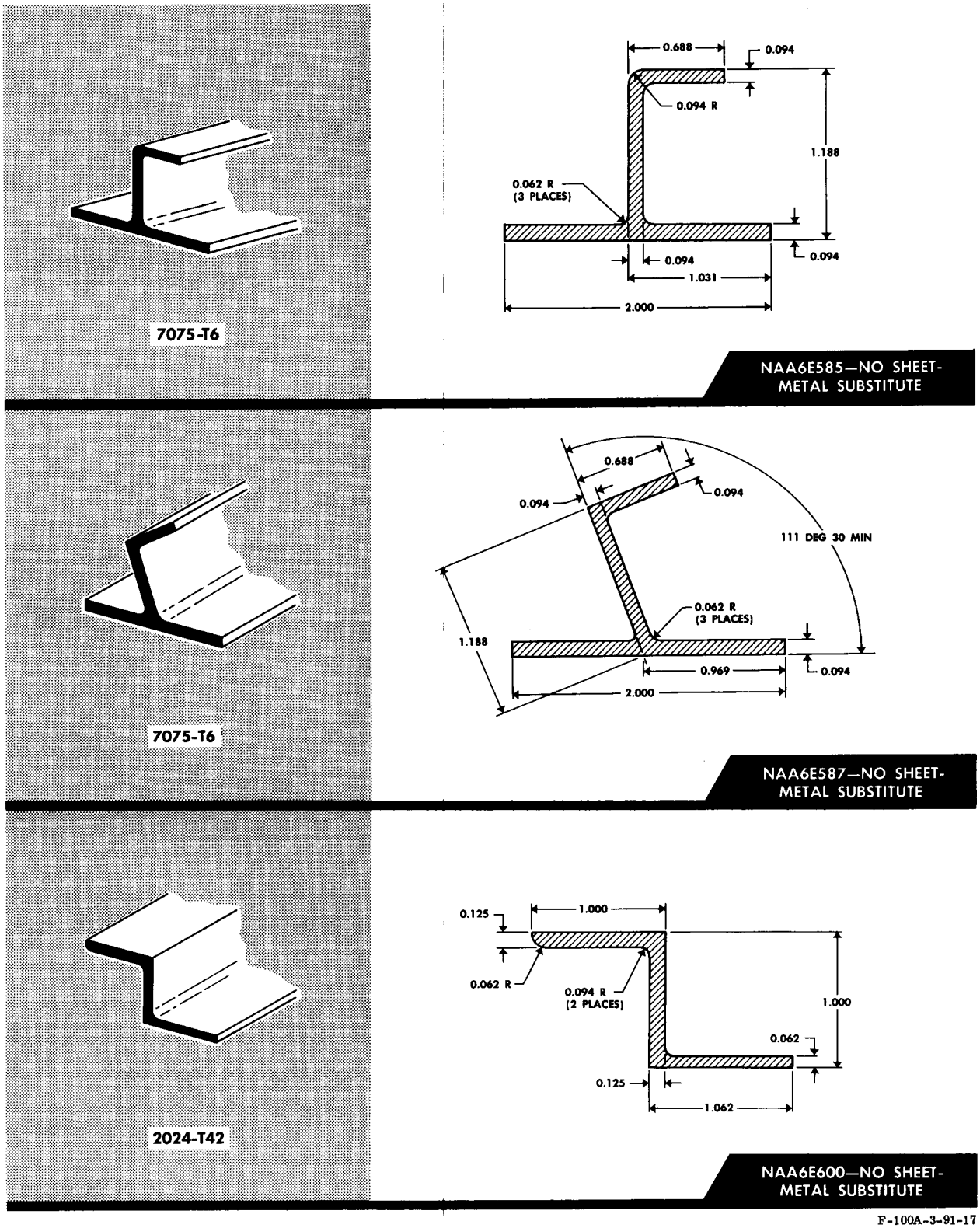


Figure 8-7. Extrusions (Sheet 11 of 11)

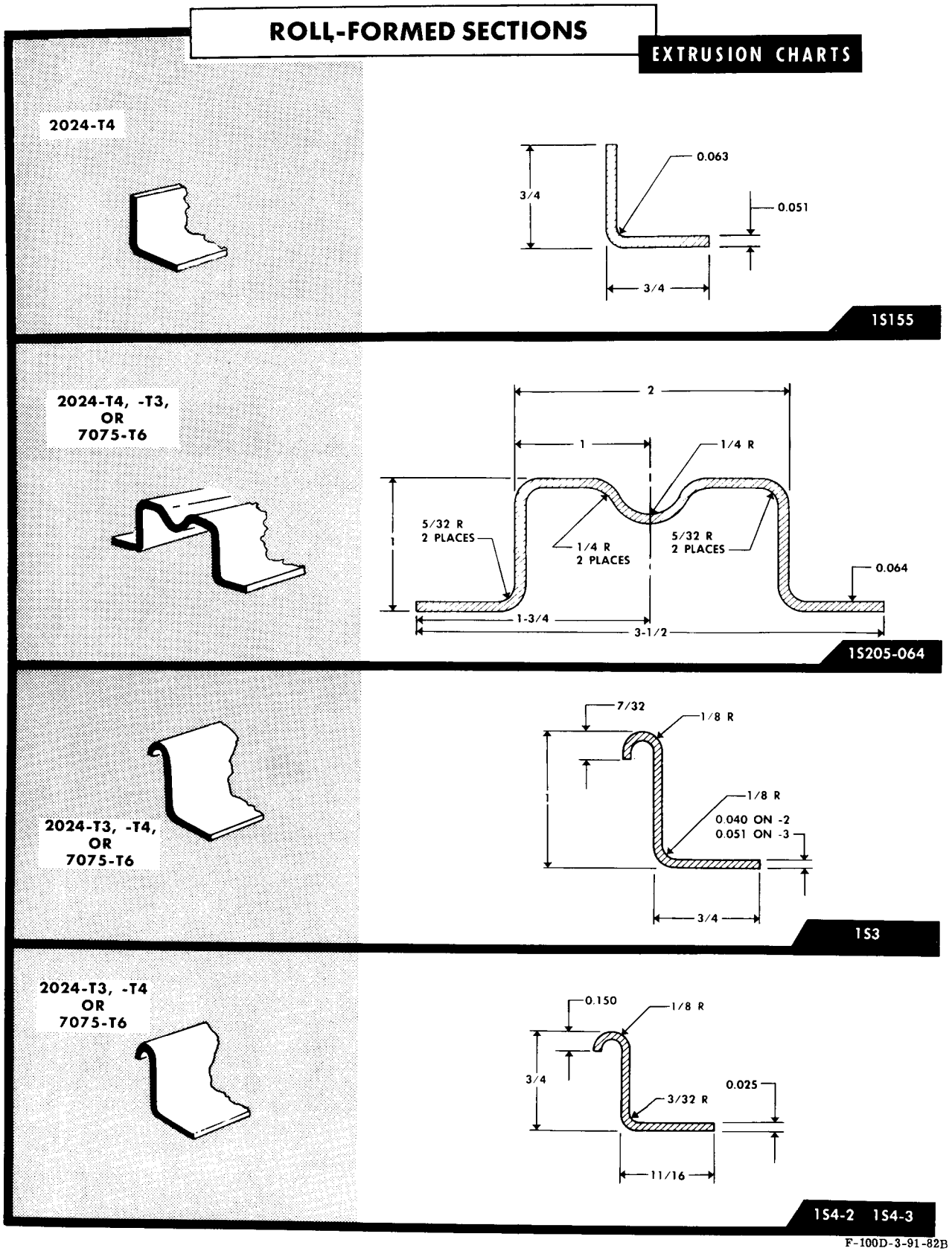
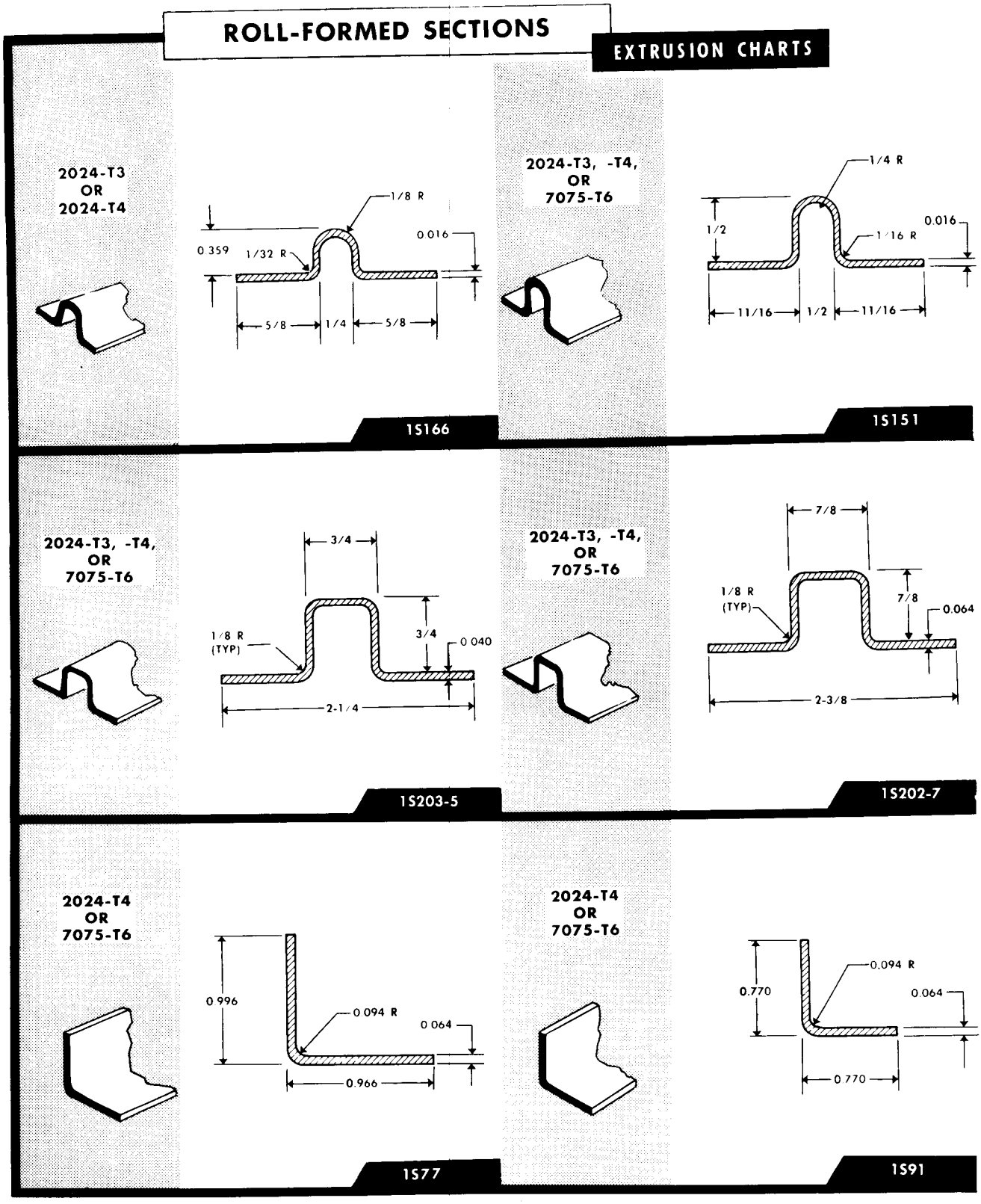


Figure 8-8. Roll-formed Sections (Sheet 1 of 2)



F-100D-3-91-83B

Figure 8-8. Roll-formed Sections (Sheet 2 of 2)

Section IX

GEAR-UP LANDING DAMAGE

Contents

	Page
GEAR-UP LANDING DAMAGE	9-1
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WING ASSEMBLY DAMAGE	9-2B
HORIZONTAL STABILIZER DAMAGE	9-2B
SELECTING REPAIRS	9-2C
NOSE-GEAR-UP LANDING REPAIR PROCEDURES	9-2C

Note

The information and illustrations in this section are for the F-100D Airplane. However, they apply also to the F-100F Airplane.

GEAR-UP LANDING DAMAGE.

Damage to the airplane resulting from gear-up landing will vary with each landing condition such as airplane attitude at touchdown, landing surface, etc. This section covers a general or average condition. Replacement parts, repair sections, and splice points shown are those which would normally be used. It may be necessary to deviate from, or alter, conditions given to meet the individual situation.

THE ILLUSTRATIONS.

The three sheets of figure 9-1 are indexes. Each shows the areas of the airplane that are assumed to be damaged from a gear-up landing condition. Figure 9-2 comprises breakdown views of the airframe showing damage to structural components, manufacturing splice location, recommended splice locations other than manufacturing splices, and assembly drawing numbers.

In these views is shown the general collective damage which might result from the three types of wheels-up landings discussed. Notes within the illustration, plus the gear-up landing damage indexes, clarify landing condition, or conditions, to which the damage is applicable.

Small lower frames, beams, and other relatively small items are shown exploded (disconnected) from the main structure. They are broken at their original splice point. These items, when available, would be replaced rather than repaired. Damaged items which are shown connected to the main structure will normally be repaired in place.

Recommended splice points are shown when the manufacturing splice points are too far removed from the damaged area. These are merely recommended splice points, however, based on average damage. The splice points may be moved closer to the bottom centerline or farther out from it as required for practical and adequate repair. Selecting the splice points is discussed further under "Fuselage Damage."

Drawing numbers of assemblies are shown on the illustrations. These numbers are presented for reference only and should not be used for ordering parts. When ordering parts from stock, use the F-100D Illustrated Parts Breakdown, T.O. 1F-100D-4, or US Air Force Supply Catalog, S-01M-F100. When ordering parts which are to be manufactured (not listed in the aforementioned catalogs) select the part numbers from blueprints or microfilm.

Figures 9-3 through 9-12 provide repair instructions, tool fabrication, and other data necessary to accomplish straightening and repair of fuselage forward section damage resulting from a nose-gear-up landing. These figures and supporting text provide a procedure for repairing, instead of replacing, damaged fuselage nose sections.

AIRFRAME DAMAGE INSPECTION—GENERAL.

After an airplane has been landed in a gear-up condition, certain areas of the bottom skins and covers will be worn off. The wear-off will usually progress through the skins and into the substructure where loads are the heaviest. Above the "wear-off" area, there is usually a bending or buckling condition in the bulkhead, frame, and beam webs. This damage is obvious as soon as the damaged underskins and covers are removed. There is likely to be a secondary damage, however, which often is uncovered only by a thorough and systematic inspection. The vertical forces transferred through the frames and bulkheads from the initial contact with the runway can loosen or shear rivets at any point within the members. All skin and web rivets should be closely inspected over the entire length of these members.

The engine air intake duct should be carefully inspected for cracks and buckling. Bending forces which deflect the fuselage may cause a fracture on one side of the duct and buckling on the opposite side. The two types of damage may be close together or widely spaced. (For further discussion of damage inspection, refer to "Investigating Damage" and "Exceeding Design Limitations in Flight or Landing" in Section I.)

AIRFRAME DAMAGE INSPECTION—NOSE-GEAR-UP LANDING.

Typical damage areas resulting from nose-gear-up landing are shown in figure 9-1. Breakdown views of the airframe showing typical damage locations and damage inspection areas are provided in figure 9-3. In addition to the general airframe damage inspection requirements, the following inspections must be complied with to determine extent of damage and feasibility of repair following a nose-gear-up landing.

The following tools and equipment are required to accomplish this inspection:

- An air motor and common twist drills for rivet removal.
- A set of jacks and jack pads. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)
- A magnifying glass and a mirror.
- Common hand tools for removal of access doors.

1. Jack airplane to convenient height with fuselage about level laterally and longitudinally. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

Note

- Airplane jacking may be postponed until after inspection, but must be done before straightening and repair operations are started.
- The nose section must be off the ground for a thorough inspection.

2. Remove access doors as required for inspection and subsequent repair. These doors should include, but not be limited to, the following:

F-100D	F-100F
F1*	F1*
F1A	F1A
F3	F3
F4	F3A
F5	F4
F6	F5
	F6
	F7
	F15
	F16

*Open hood only for inspection. Reinstall hood before straightening of fuselage is started.

3. Remove pitot boom assembly. (Refer to "F-100D Instrument Systems," T.O. 1F-100D-2-7.)

4. Remove fasteners from side panel skins in areas adjacent to buckles.

Note

Remove only the number of fasteners necessary to accomplish inspection and later repair operations. It should never be necessary to remove more fasteners than are shown in figure 9-9.

5. Remove lower forward nose skin. (See figure 9-3.)

Note

- All damaged areas are inspected at this time to determine parts to be replaced and to determine sheet-metal parts that require straightening or structural repair. All bent or buckled members must be checked after straightening to determine if cracks were introduced during the straightening process.
- Record all discrepancies found during the initial inspection.
- Perform all visual inspections with the aid of a mirror and/or magnifying glass as required. If discrepancies are suspected on aluminum structure, use dye penetrant or X-ray to determine extent of damage. If discrepancies are suspected in steel fasteners, the fasteners should be removed and inspected.

6. Thoroughly clean all areas before inspecting so that surface can be clearly seen. Refer to Section I for cleaning, paint removal, and repainting procedures.

7. Check all buckled fuselage side and duct skins between fuselage stations 89 and 116 (F-100D) or 54 and 81 (F-100F). Note the severity of the skin buckling and and deformation. Buckling which terminates in the form of sharp creases should be carefully inspected for cracks. Since sharp creases can impose high loading on adjacent fasteners, any fasteners not removed previously should be examined for head failure, tipping, or looseness.

8. Check all bulkheads, frames, and beams in the areas of buckled fuselage and duct skins. Inspect these members for cracks, buckling, or bending conditions. Since initial contact of nose section with runway can transfer high shear loads through substructure, inspect for loose or sheared fasteners at any point within these members.

9. Check fuselage and duct skins and frames at pitot boom attach point. Give particular attention to the pitot-boom-forward attach-fitting-to-frame joint.

10. Check lower duct frames forward of pitot boom attach fitting. Inspect for cracks and buckling.

11. Check fuselage-and-duct leading edge extrusion. Inspect for cracks, buckling, and/or wear-off conditions due to damage caused by pitot boom on contact with runway. Also inspect fasteners in adjacent area for signs of damage.

12. Check upper and lower longerons between fuselage stations 89 and 116 (F-100D) or 54 to 81 (F-100F). Note any crippling or buckling conditions. No waviness or bowing of any nature is acceptable. Inspect longeron splice fittings on each side of pressure bulkhead at fuselage station 106 (F-100D) or 71 (F-100F) for cracks or other damage. Examine fitting and joint fasteners for signs of failure or deformation.

13. Check nose landing gear trunnion support beams. Examine fitting for cracks.

14. Check radar bay hood skin and frames. Inspect for cracks and buckling.

15. Check aircraft records. Particular attention should be given to previous damage and repairs noted in area from fuselage station 106 (F-100D) or 71 (F-100F) forward.

FUSELAGE DAMAGE.

When the fuselage forward section is severely damaged from wear-off and buckling, it is sometimes advisable to replace the entire fuselage section from station 106 forward including frame assembly 223-31103 at station 106. The electronic compartment hood assembly may also be damaged to an extent requiring replacement.

If it is determined by procedures in "Airframe Damage Inspection—Nose-gear-up Landing" that the fuselage forward section is reparable, it is then necessary to decide whether a satisfactory repair will cause less airplane down time than procurement and replacement of parts. In most cases, repair of the fuselage forward section will result in less airplane down time and in a reduction in man-hours expenditure.

Fuselage station 106 is a production splice point where all members of the forward section terminate. The skins are riveted to the frame assembly and the duct frames are bolted together at this point. The longerons and intercostals are also bolted and stringers are joined with Hi-Shear rivets. For replacement of the forward section of the fuselage, the damaged section must be separated from the fuselage at station 106. The new fuselage forward section is secured in place by the four longeron fittings. The fuselage skins are not trimmed or fastened in place until the duct skins have been aligned and mated together. The only special tool required to replace the fuselage forward section is an engine air intake expansion plug (NAA tool T437989-2). The expansion plug is installed in the duct with one half of the plug on each duct section. The tool is then expanded until all of its outer surfaces are firmly pressed against the duct skins on both the forward and aft sides of the tool. When perfect mating of the duct skins is obtained, bolt holes in the forward duct mating frame are picked up from existing holes in the aft mating frame. The duct sections are then bolted together and the expansion plug removed.

Refer to "Nose-gear-up Landing Repair Procedures" for complete instructions on straightening and repair of fuselage forward section.

Fuselage bottom damage aft of station 106 is usually confined to worn-off skins, doors, frame caps, and beam caps. In the areas of heaviest load, and therefore heaviest wear, the wear-off may progress up into the frame and bulkhead webs. Also, above the wear-off, webs will often be bent or buckled, requiring replacement of the entire lower frames or the splicing-in of sections of the frames.

When damage is such that the splice locations shown in the damage illustrations are not satisfactory, the repairman will have to select the best possible location. The most important consideration in selecting a splice point is to have enough room on each side of the joint for a repair doubler containing an adequate number of rivets. Rivet requirements should be calculated before the cut is made. When splicing both caps and webs, the splices should be staggered enough to permit the repair doublers for each item to be installed without overlapping each other. This method of splicing simplifies the repair, making it easier to fabricate and install the repair, and the result is a stronger, less bulky repair.

When a bottom frame is pushed up slightly but has no definite buckle or cracks, it may be shimmed on the outside, up to 0.064 inch, to tighten the skins and maintain the original contour.

When an airplane has been landed in a nose-gear-up condition, the gear will usually stay in place. The nose gear, nose gear door, and nose gear trunnion beams may require replacement. When the airplane has been landed with the nose gear down and nose gear has later broken loose from its mountings, the gear will usually be forced up through the engine air intake duct, necessitating replacement of duct skins and frames in this area. Typically, a nose-gear-up landing will cause the nose of the airplane to be bent upward from station 106 (F-100D) or 71 (F-100F) forward. This bending will result in buckles on both left and right sides of the fuselage and at the left and right rear corners of the radar bay hood. (See figure 9-3.) The pitot boom will contact the runway, causing damage to the nose section lower frames forward of station 36.6 (F-100D) or 1.6 (F-100F), the intake duct leading edge extrusion, the intake duct skins, and the lower forward nose skin. Figures 9-4 through 9-12 and supporting text provide the necessary information to repair this typical nose-gear-up landing damage, instead of replacing the entire nose section of the airplane.

NOSE-GEAR-UP LANDING DAMAGE REQUIRING REPAIR.

The following items can tolerate no damage and must be repaired or straightened if they are cracked or deformed in any way.

1. Engine air intake duct skins
2. Duct inner and outer frame flanges
3. Longerons
4. Fuselage frame flanges

Note

Longeron and frame flanges that attach to the skins are particularly critical.

Skins are less critical than the items listed, but must be straightened and/or repaired when:

1. Buckles terminate in the corner of a bay as a sharp crease.
2. Buckles cross a row of rivets.
3. Cracked beyond negligible damage limits given in figure 4-4.
4. Rivets are missing or loose, or have tipped heads.

Nose section misalignment exceeding limits in figure 9-4 must be brought into tolerance.

NOSE-GEAR-UP LANDING DAMAGE NOT REQUIRING REPAIR.

The following items may remain unrepaired without affecting airplane airworthiness or reliability.

1. Buckles that are entirely contained within a structural bay (does not cross a rivet row) and does not terminate in the corner of a bay as a sharp crease.
2. Cracks, scratches, and dents within negligible damage limits in figure 4-4.
3. Nose section misalignment which is within limits shown in figure 9-4.

JIGS.

A jig must be provided for installation of the nose gear trunnion support fittings. Shear pin and shear pin hole locating jigs will also be necessary in rebuilding the lower frames at the field break (station 389). These are the only jigs required in rebuilding the lower side of the fuselage. Necessary locating points for other replacement members can be found by measurement.

FUSELAGE SKINS.

Fuselage skin sections may be either replaced or spliced. The decision should be based on practicability. It is usually advisable to replace small panels and larger panels when extensively damaged. It is often more convenient to splice larger panels when the damage is slight, such as along one edge only. At other times, a splice may be indicated when complete removal of the panel necessitates disassembly of structural or system components not otherwise necessary.

WING ASSEMBLY DAMAGE.

The wing is normally undamaged in the wheels-up landing conditions when auxiliary fuel tanks are not installed. When the tanks are installed, damage is likely to occur at the tank mounting points. While the main wing panel is usually undamaged in a wheels-up landing, the wing tip will be dragged if the plane rolls over at too great an angle. Drag damage may be slight, requiring but minor repair; or it may be extensive, requiring replacement of the tip assembly from leading edge station inboard. The tip can be replaced without jigs if an aileron rig fixture is used to rig the aileron to neutral or zero degrees. The wing tip trailing edge is then aligned with the aileron trailing edge.

The landing flaps are worn off and bent in the all-gear-up and main-gear-up landing conditions and should be replaced.

HORIZONTAL STABILIZER DAMAGE.

The lower side of the forward inboard horizontal stabilizer tips contacts the runway when the airplane is landed in the all-gear-up and main-gear-up conditions. Damage varies from a slight wear-off and bending to a

wear-off of 12 to 15 inches of the forward inboard tip, depending on the angle of the stabilizer. In the main-gear-up condition, the aft outboard corners of the stabilizer are also likely to be damaged. The extent of this damage will also vary greatly, depending on the stabilizer angle.

SELECTING REPAIRS.

Repairs will be selected from the applicable repair index and from the typical repair section, except for repairs to fuselage forward section following nose-gear-up landing, which are provided in this section. (Refer to "Selection of Typical Repairs" in Section X.)

NOSE-GEAR-UP LANDING REPAIR PROCEDURES.

PREPARATION.

The following tools and equipment are required to accomplish repair:

- Common sheet-metal tools, including pneumatic drill and riveting hammer.
- Shot or sand bags.
- Weight support sling. (Fabricate as shown in figure 9-5.)
- Longeron straightening tool. (Fabricate as shown in figure 9-6.)
- Duct frame forming blocks. (Fabricate as shown in figure 9-7.)
- Minijack, Blackhawk Model S45 (or equivalent).

Before straightening operations are started, repair any damage to the frame at station 36.6 (F-100D) or 1.6 (F-100F) and to duct skins in the same area.

1. Jack airplane, using wing and fuselage forward section jack points, to a convenient height with airplane level. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

2. Place canvas weight sling over fuselage forward section at station 36.625 (F-100D) or 1.625 (F-100F), so that sling hangs vertically.

3. Add weight to sling until structure starts to move. Movement will be indicated by a squeaking sound in the structure.

CAUTION

Do not add more than 2000 pounds of weight at this time.

FUSELAGE STRAIGHTENING.

Note

- Use nonmetallic mallets, sand or shot bags, and techniques that will minimize skin stretching.
- After rework and straightening, check all affected areas for cracks or other damage such as failure or deformation of adjacent fasteners.

1. Add weight as necessary up to a maximum of 2000 pounds, while smoothing buckled fuselage side skins and duct skins and straightening longerons. Use straightening tool for longerons as shown in figure 9-6.

2. Complete smoothing of engine air intake duct skins.

3. Using forming blocks and minijack, press engine air intake duct frame, at forward pitot boom attach

point, back to its original contour. (See figure 9-7.) Straighten duct frame webs and reshape lower (outer) flanges as necessary.

4. Remove weight from sling and check alignment of fastener holes in side panel skins and substructure. If holes do not line up, replace weights previously removed and add weight as necessary up to 2000 pounds, until fastener holes remain aligned with weight removed.

5. If weight in excess of 2000 pounds is required to properly straighten fuselage, the following procedures must be used.

a. Install access doors and skin riveting aft of fuselage station 106 (F-100D) or 71 (F-100F).

b. Install lower longeron riveting between fuselage stations 89 and 106 (F-100D) or 54 and 71 (F-100F).

c. Increase weight as necessary in small amounts up to 3000 pounds maximum. Inspect fuselage and duct skin, during loading, for cracks, particularly between stations 89 and 106 (F-100D) or 54 and 71 (F-100F).

6. Straighten and recontour intake duct leading edge extrusion.

Note

Before continuing with repair, check fuselage forward section alignment as shown in figure 9-4 and realign as necessary.

7. Repair cracked duct leading edge extrusion as shown in figure 9-8. If more extensive repair is required, refer to Section IV.

8. Replace fasteners in side skin panels as shown in figure 9-9. Repair any cracked skins in accordance with instructions in Section IV.

9. Straighten and align lower duct frames. Repair any cracked frames forward of station 36.6 (F-100D) or 1.6 (F-100F) as shown in figure 9-10. Repair any other damaged frames as shown in Section IV.

10. Install a new lower forward skin. The new skin is to be made of 7075-T6 clad aluminum sheet, 0.063 x 27.00 x 29.00 inches, finished on the inside with medium green lacquer and on the outside with aluminized lacquer. (Refer to "Final Protective Paint Finish for Aluminum or Steel" in Section I.) Fasten skin with MS20603D5 rivets, picking up existing locations, except at intake duct leading edge, use MS20426AD5 rivets.

11. Install pitot boom attach fittings.

Note

If repair to frames in the pitot boom support fitting area has resulted in loss of fitting alignment or removal of fitting mounting holes, realign and install fittings as shown in figure 9-11.

12. Repair damaged engine inlet duct skins in accordance with Section IV.

13. Repair or replace other damaged parts or structure in accordance with instructions in applicable sections.

14. Replace pitot boom. (Refer to "F-100D Instruments," T.O. 1F-100D-2-7.)

RADAR BAY HOOD (RADAR EQUIPMENT ACCESS DOOR) STRUCTURE REPAIR.

The following procedures are to be used if it is determined by inspection that the hood assembly is reparable. The following special tools and equipment are required for repair of the hood assembly:

- Common sheet metal tools, including pneumatic drill and riveting hammer.
- Shot or sand bags.
- Radar bay hood rework tool. (Fabricate as shown in figure 9-12.)

Note

Use nonmetallic mallets, shot or sand bags, and techniques which will minimize skin stretching.

1. Remove hood assembly (radar equipment access door). (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

2. Install buckled hood assembly in tool fixture as shown in figure 9-12.

3. Apply load, by tightening turnbuckle at forward end of tool, until structure starts to move.

4. Smooth buckled skins, frames, and longerons, while increasing load as required.

5. Remove load and check alignment of hood by installing on repaired fuselage. Adjust as required to fit fuselage.

6. Repair or replace all cracked members in accordance with Section IV.

Note

After completing straightening and rework procedures, check for cracks or other damage including failure or deformation of fasteners adjacent to the reworked areas.

7. Install hood (radar equipment access door) on fuselage. (Refer to "F-100D General Airplane," T.O. 1F-100D-2-1.)

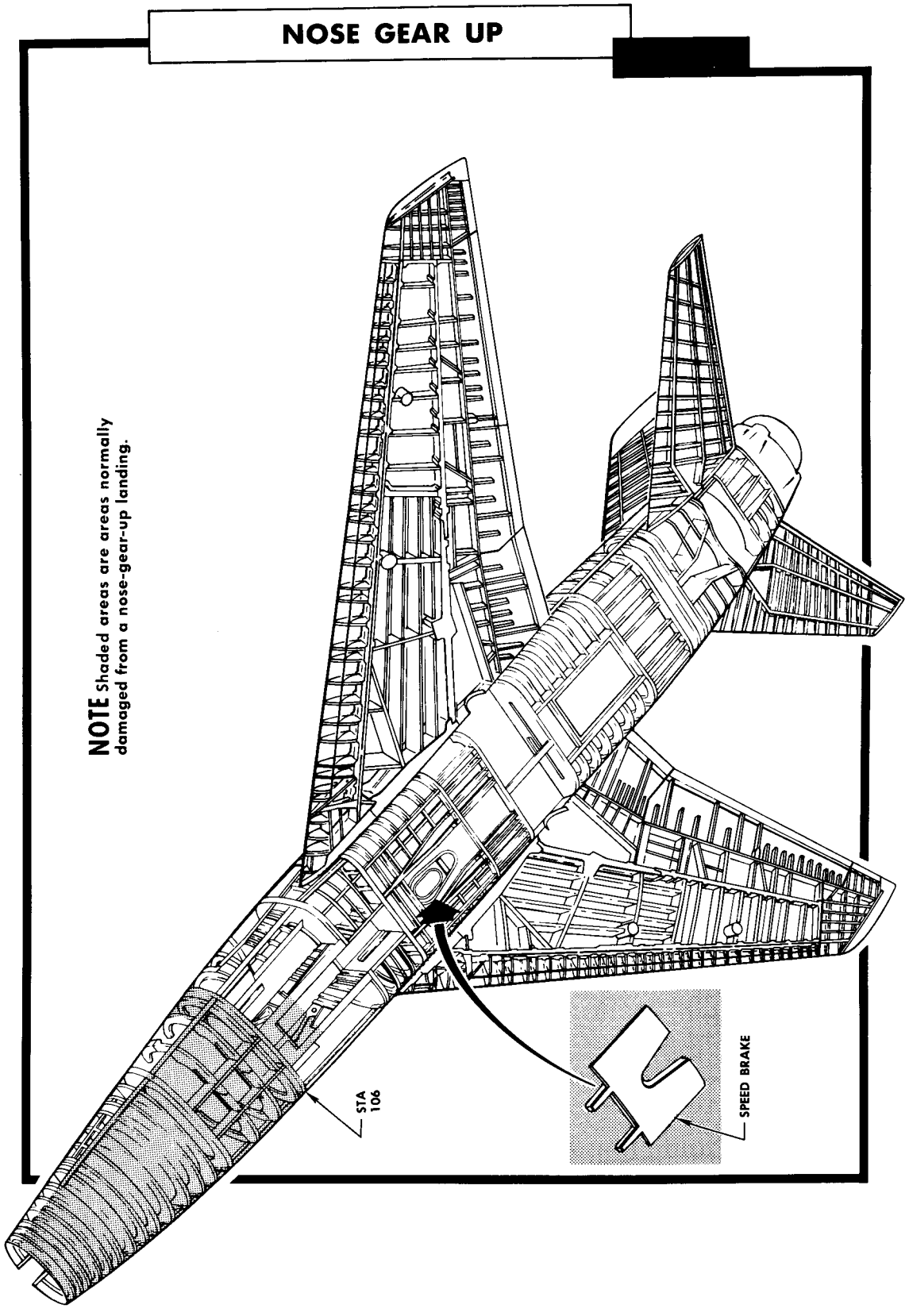

Figure 9-1. Gear-up Landing Damage Index (Sheet 1 of 4)

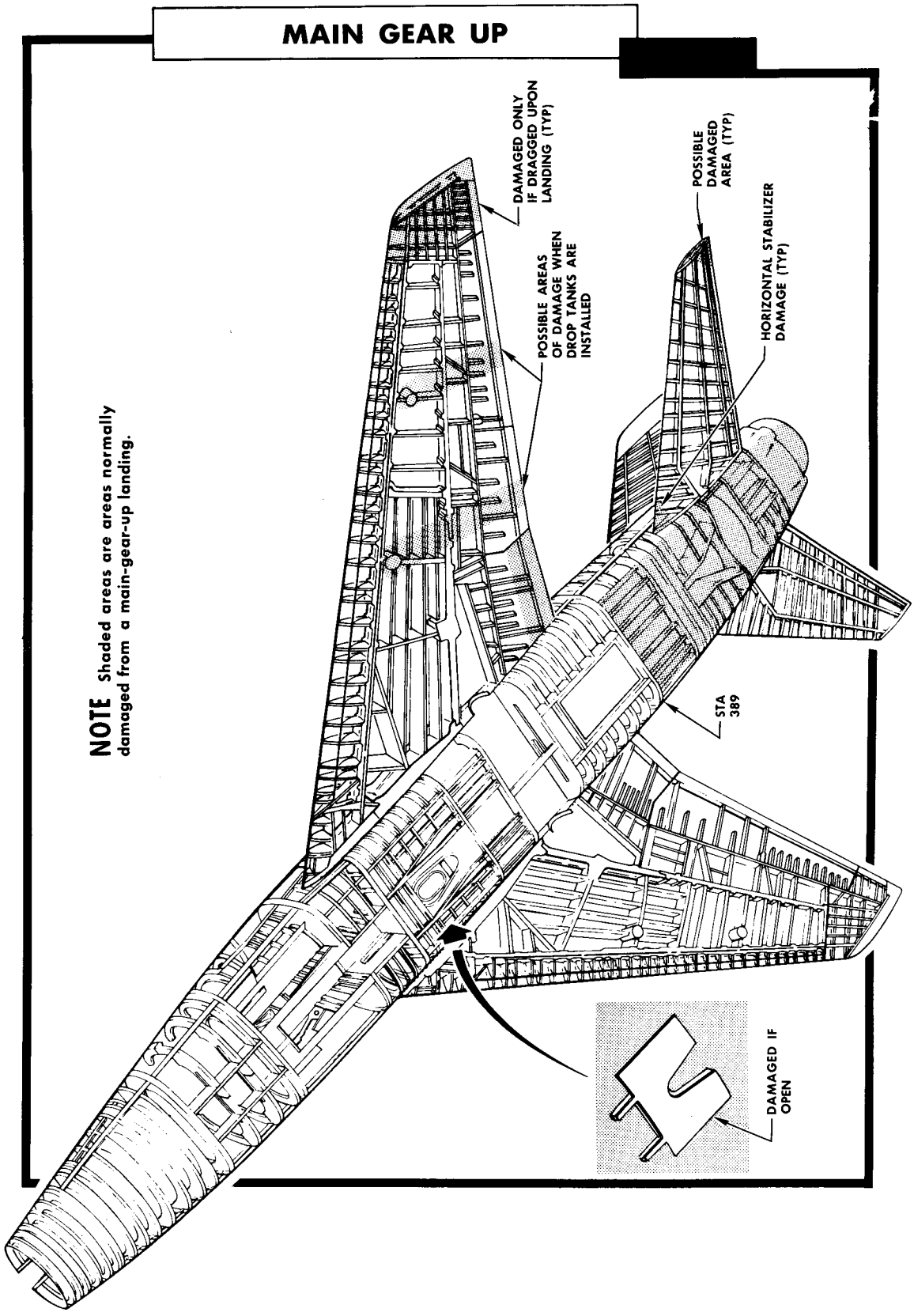

Figure 9-1. Gear-up Landing Damage Index (Sheet 2 of 4)

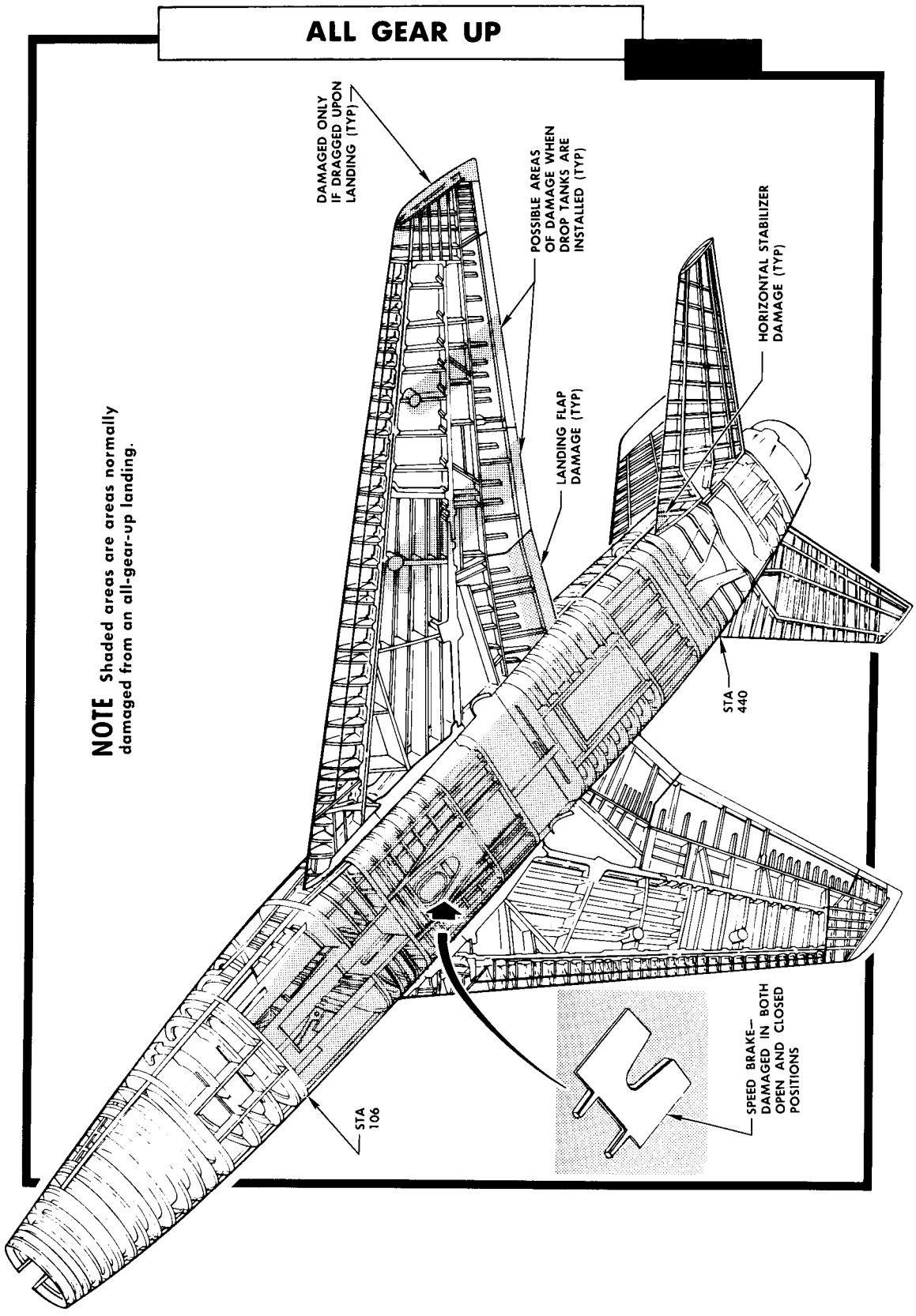

Figure 9-1. Gear-up Landing Damage Index (Sheet 3 of 4)

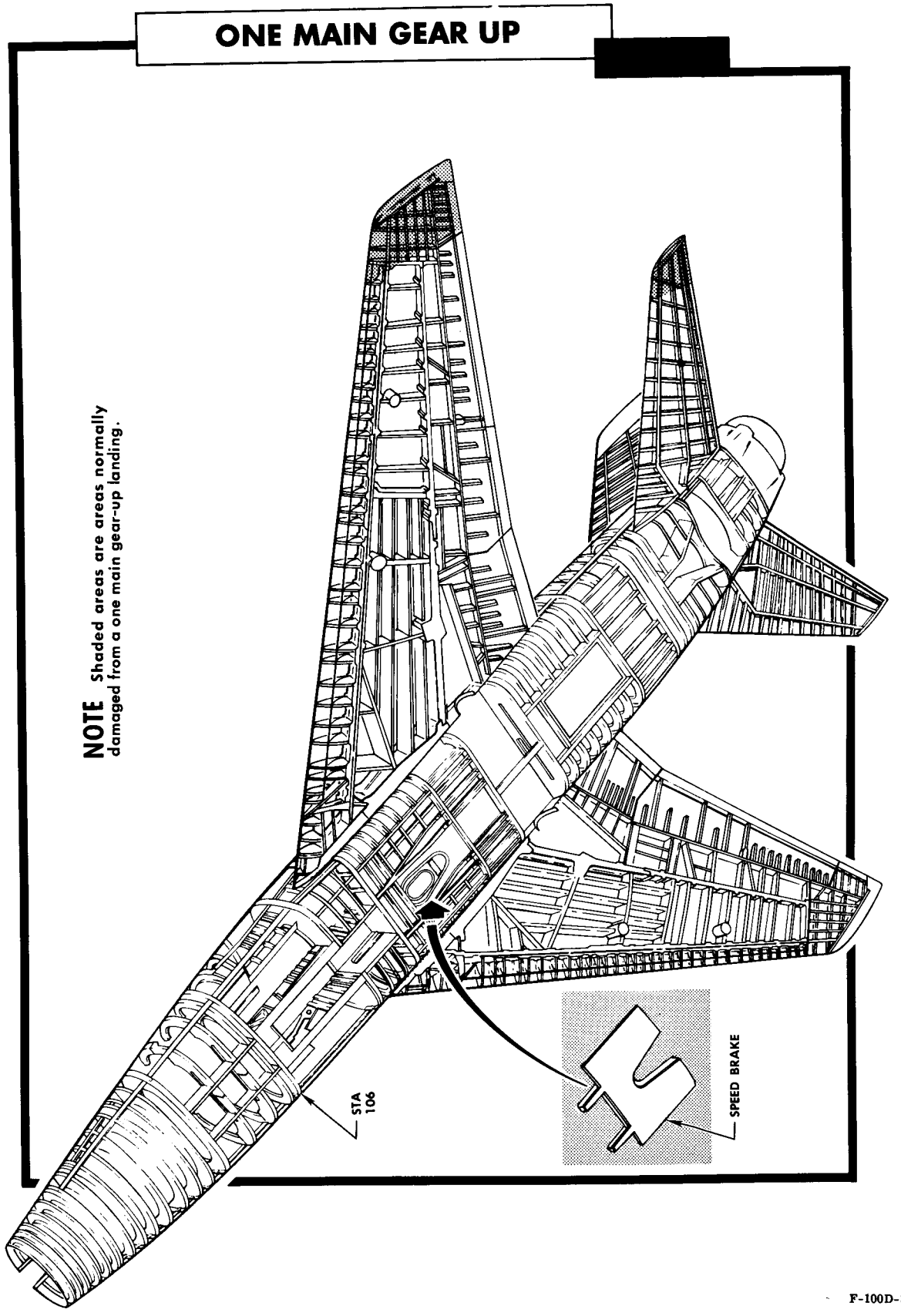

Figure 9-1. Gear-up Landing Damage Index (Sheet 4 of 4)

GEAR-UP LANDING DAMAGE

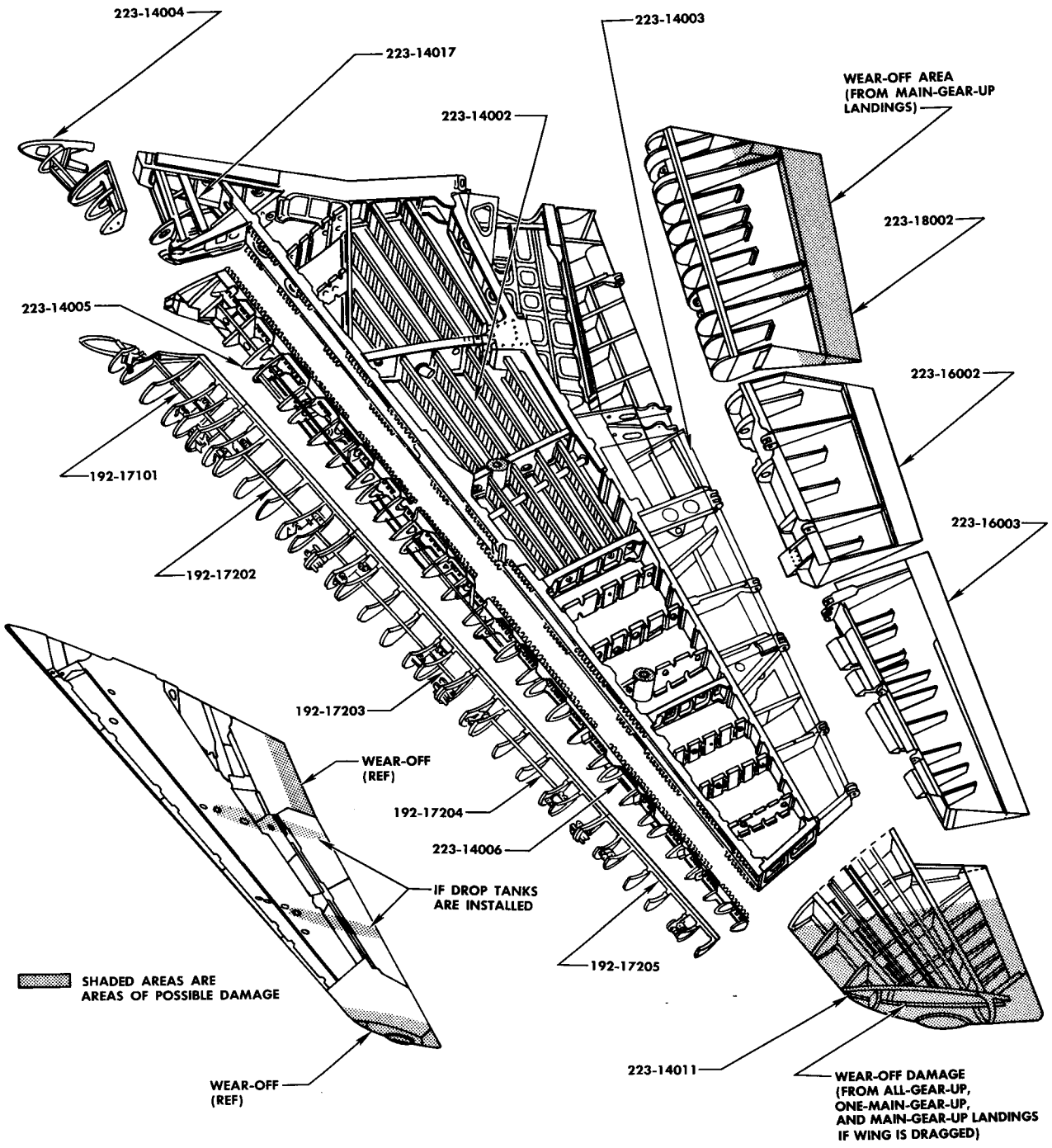

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Figure 9-2. Gear-up Landing Damage (Sheet 1 of 6)

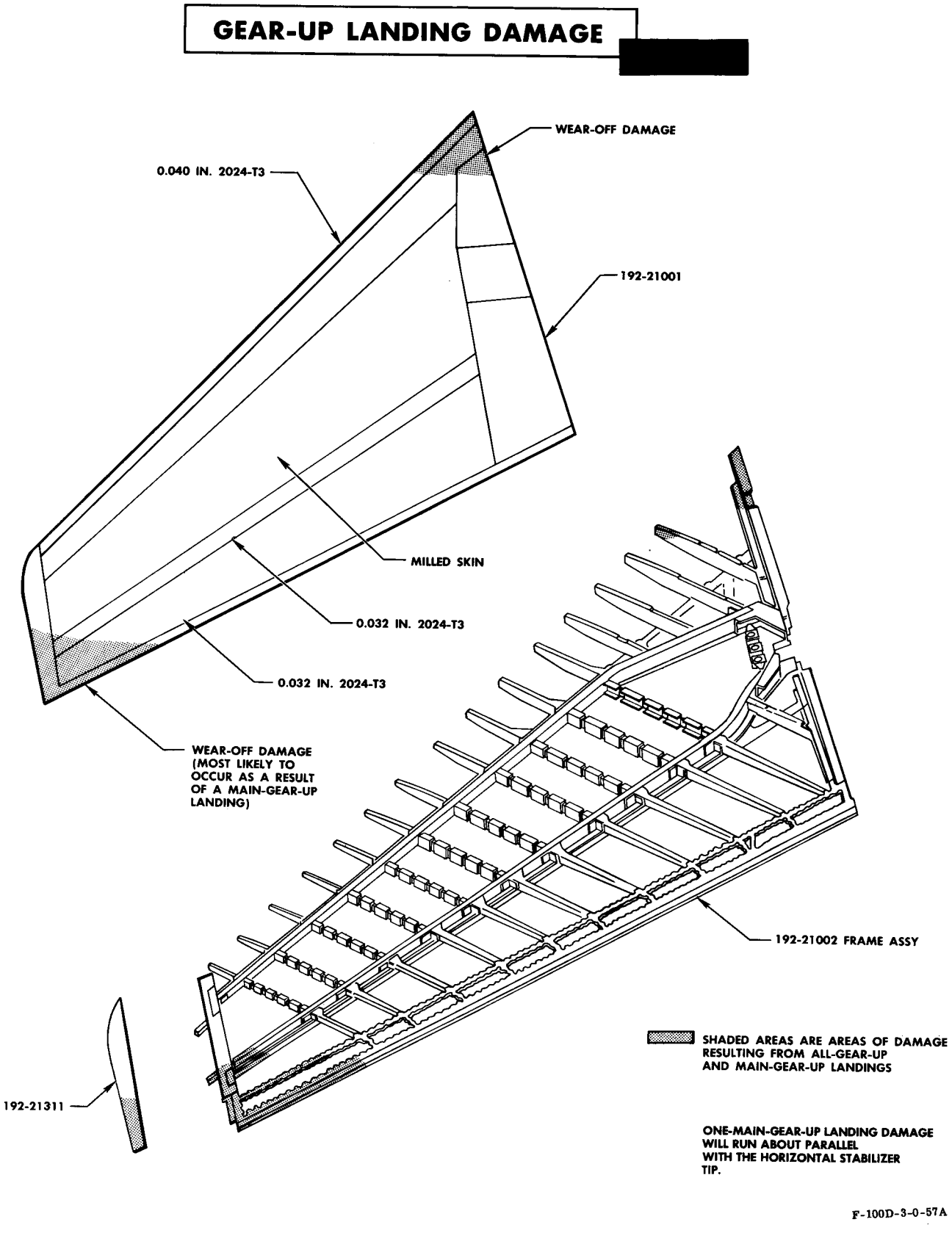


Figure 9-2. Gear-up Landing Damage (Sheet 2 of 6)

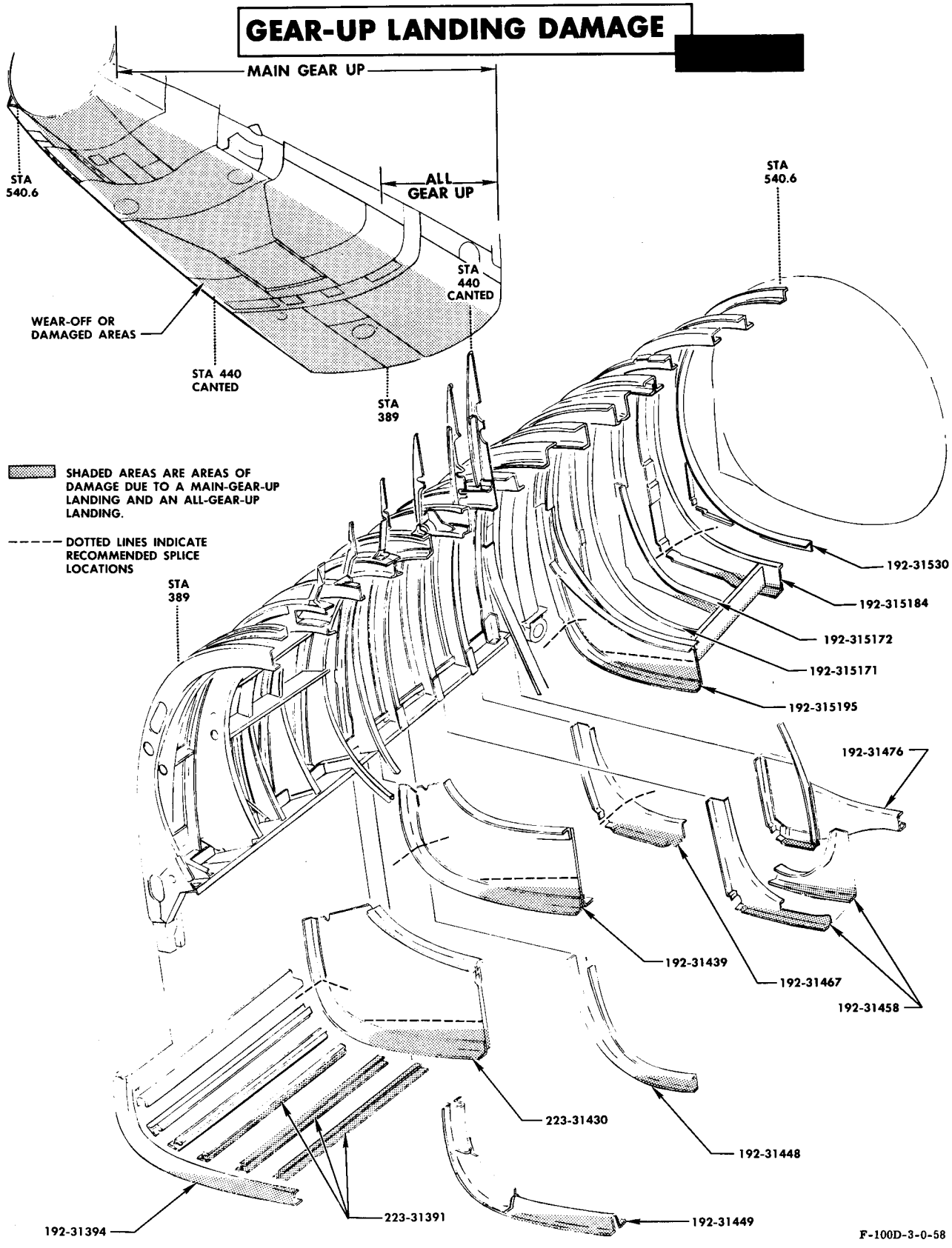
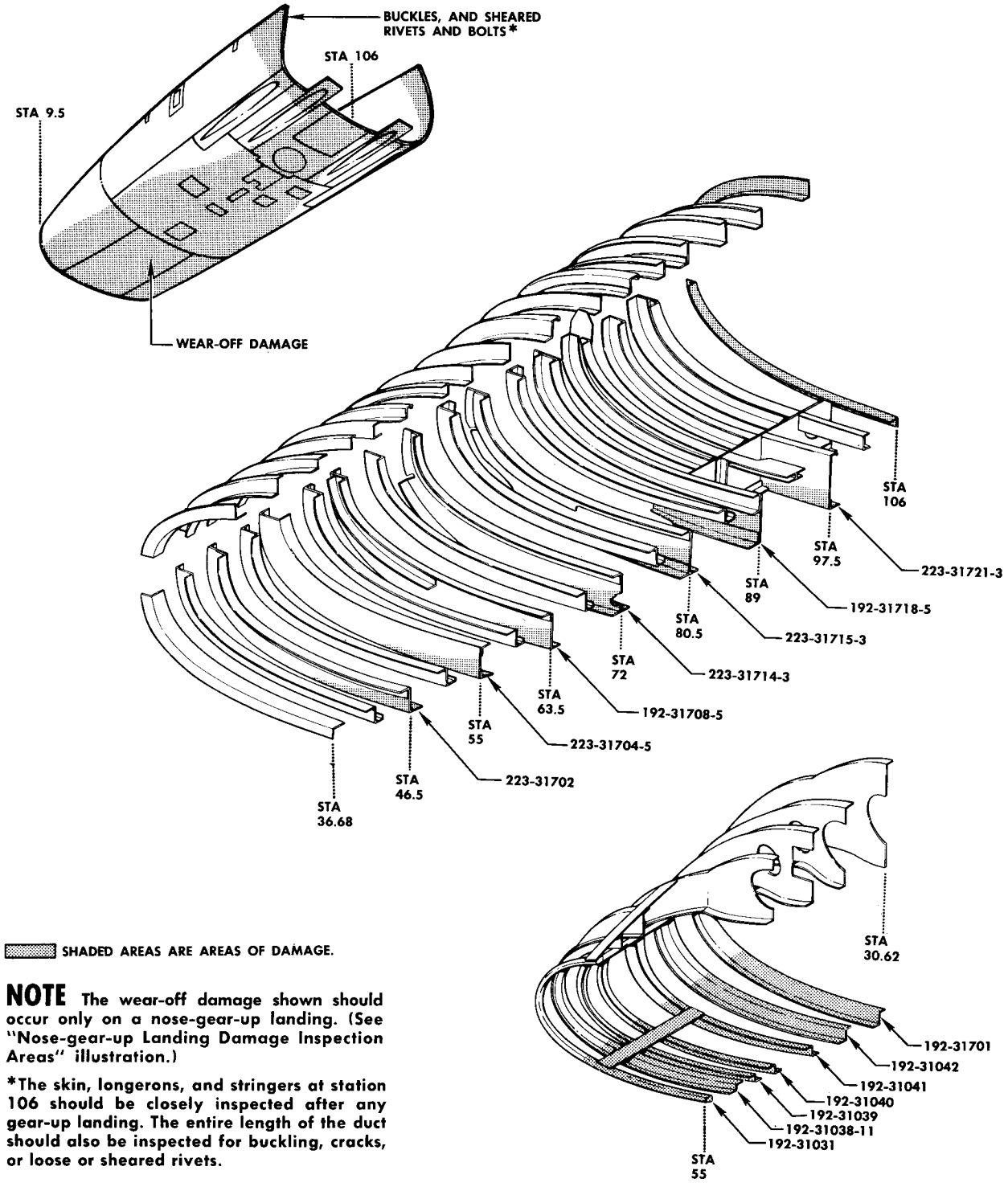


Figure 9-2. Gear-up Landing Damage (Sheet 3 of 6)

GEAR-UP LANDING DAMAGE



NOTE The wear-off damage shown should occur only on a nose-gear-up landing. (See "Nose-gear-up Landing Damage Inspection Areas" illustration.)

*The skin, longerons, and stringers at station 106 should be closely inspected after any gear-up landing. The entire length of the duct should also be inspected for buckling, cracks, or loose or sheared rivets.

100D-3-0-59A

Figure 9-2. Gear-up Landing Damage (Sheet 4 of 6)

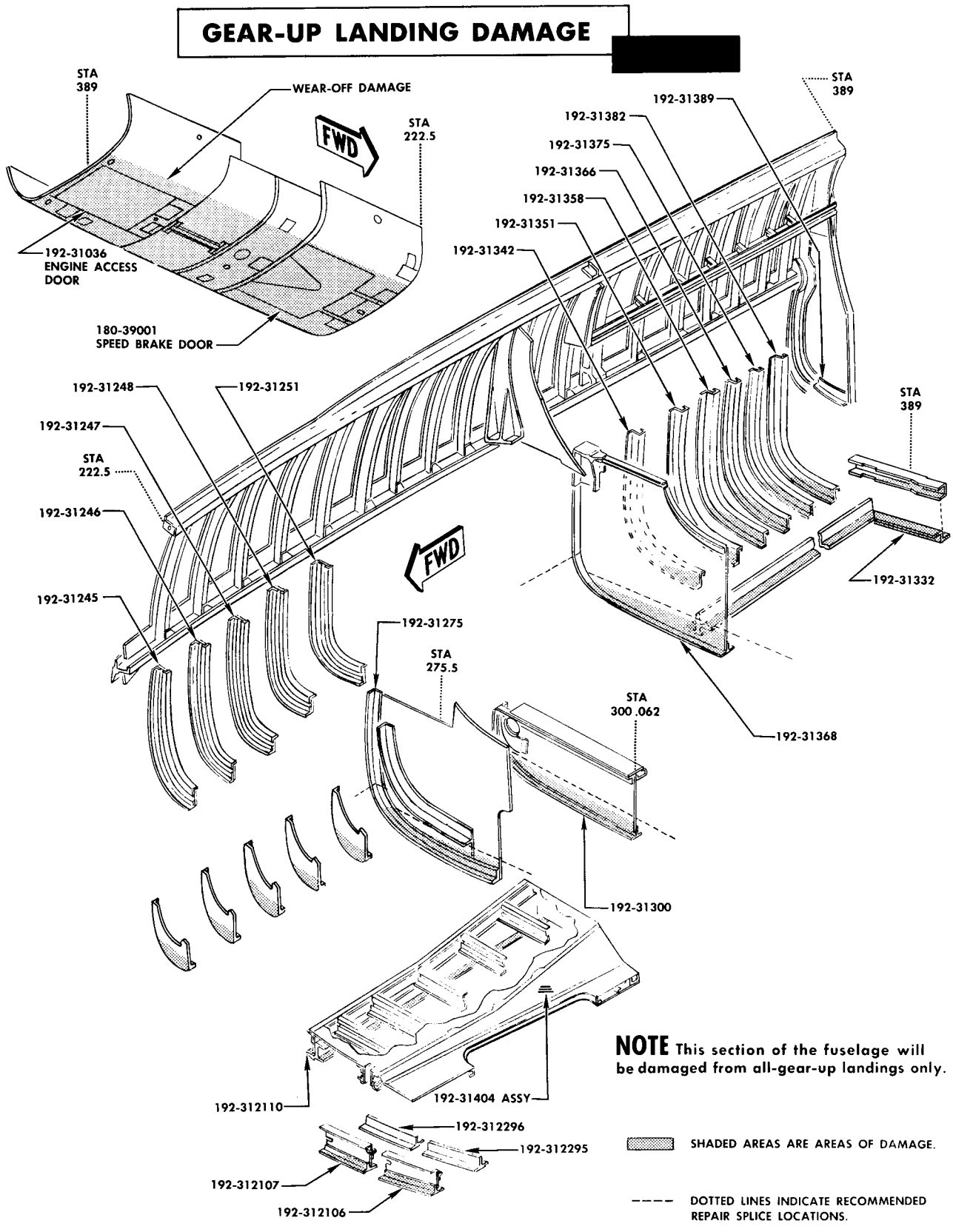


Figure 9-2. Gear-up Landing Damage (Sheet 5 of 6)

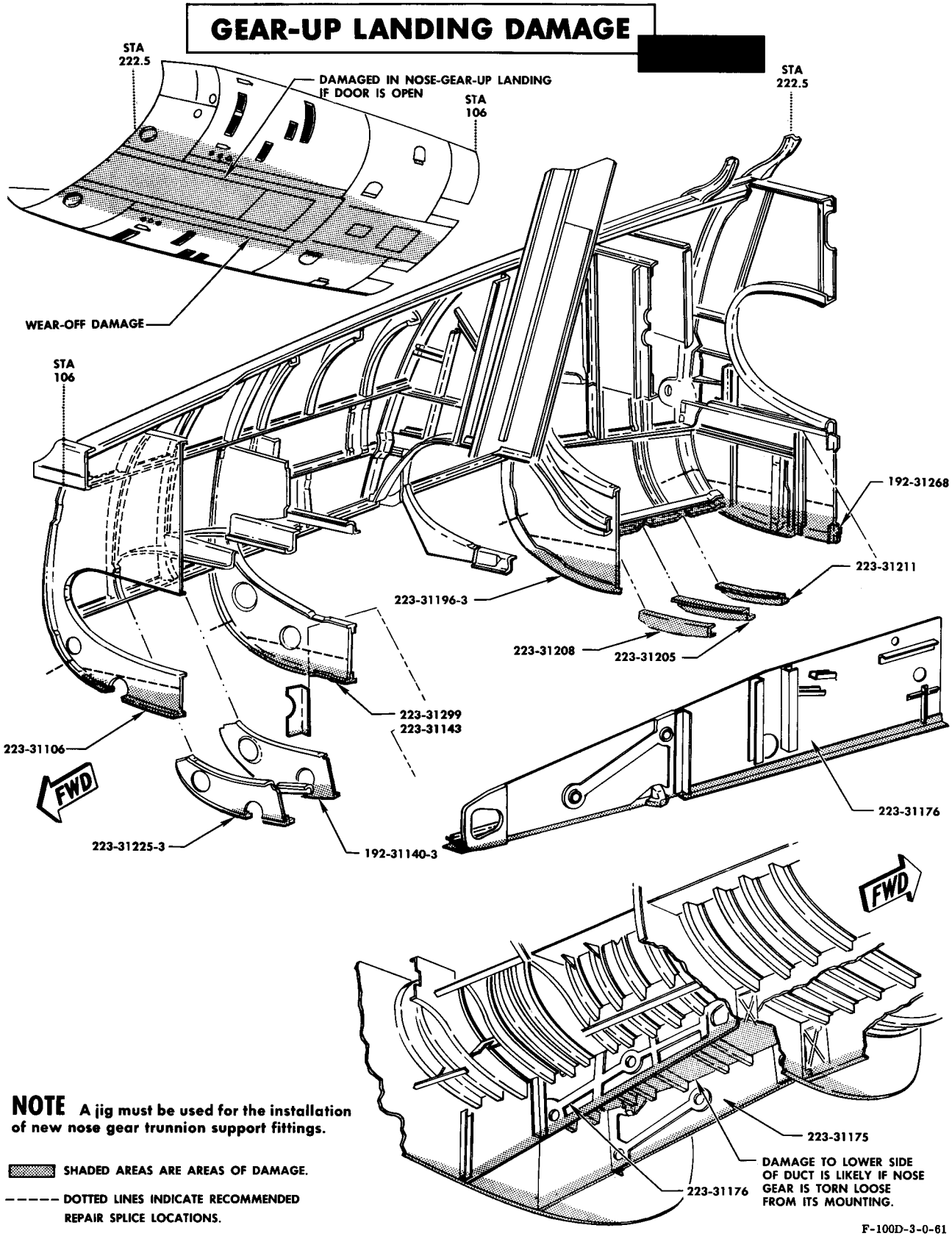


Figure 9-2. Gear-up Landing Damage (Sheet 6 of 6)

NOSE-GEAR-UP LANDING DAMAGE

INSPECTION

NOTE Inspect areas as indicated in text.

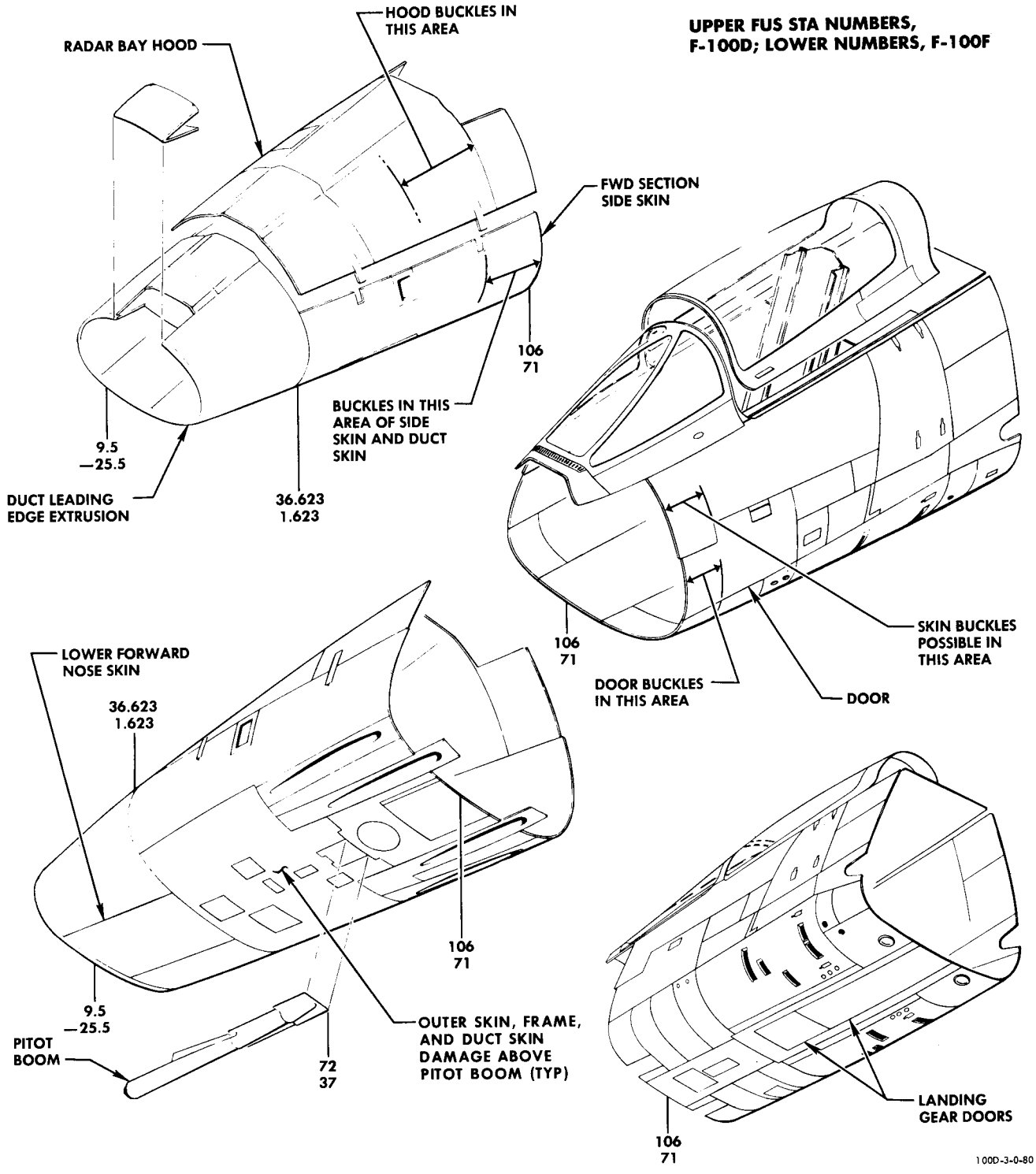


Figure 9-3. Inspection Areas (Sheet 1 of 2)

NOSE-GEAR-UP LANDING DAMAGE

INSPECTION

NOTE Inspect areas as indicated in text.

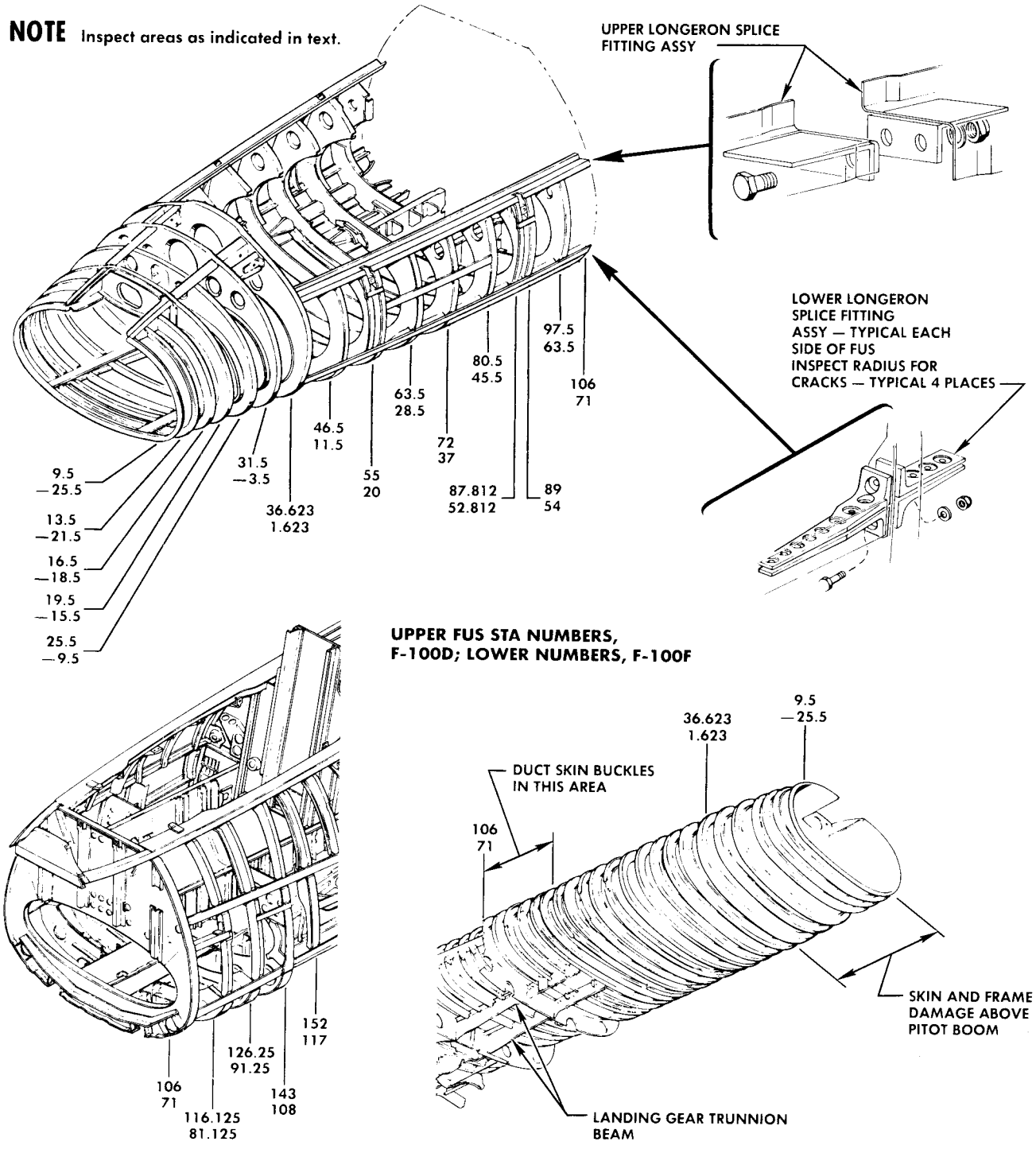


Figure 9-3. Inspection Areas (Sheet 2 of 2)

NOSE-GEAR-UP LANDING DAMAGE

REPAIR

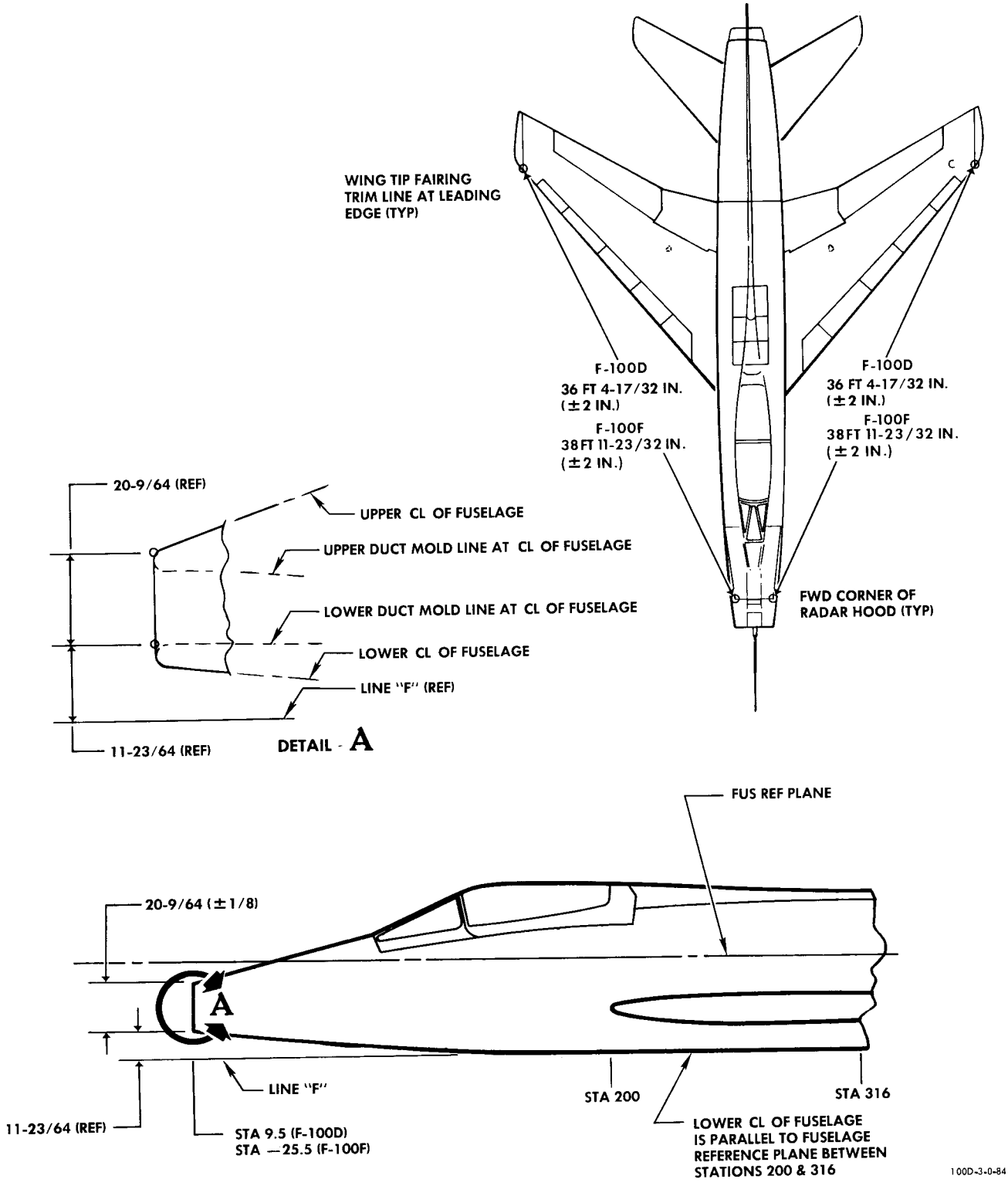


Figure 9-4. Fuselage Forward Section Alignment

NOSE-GEAR-UP LANDING DAMAGE

REPAIR

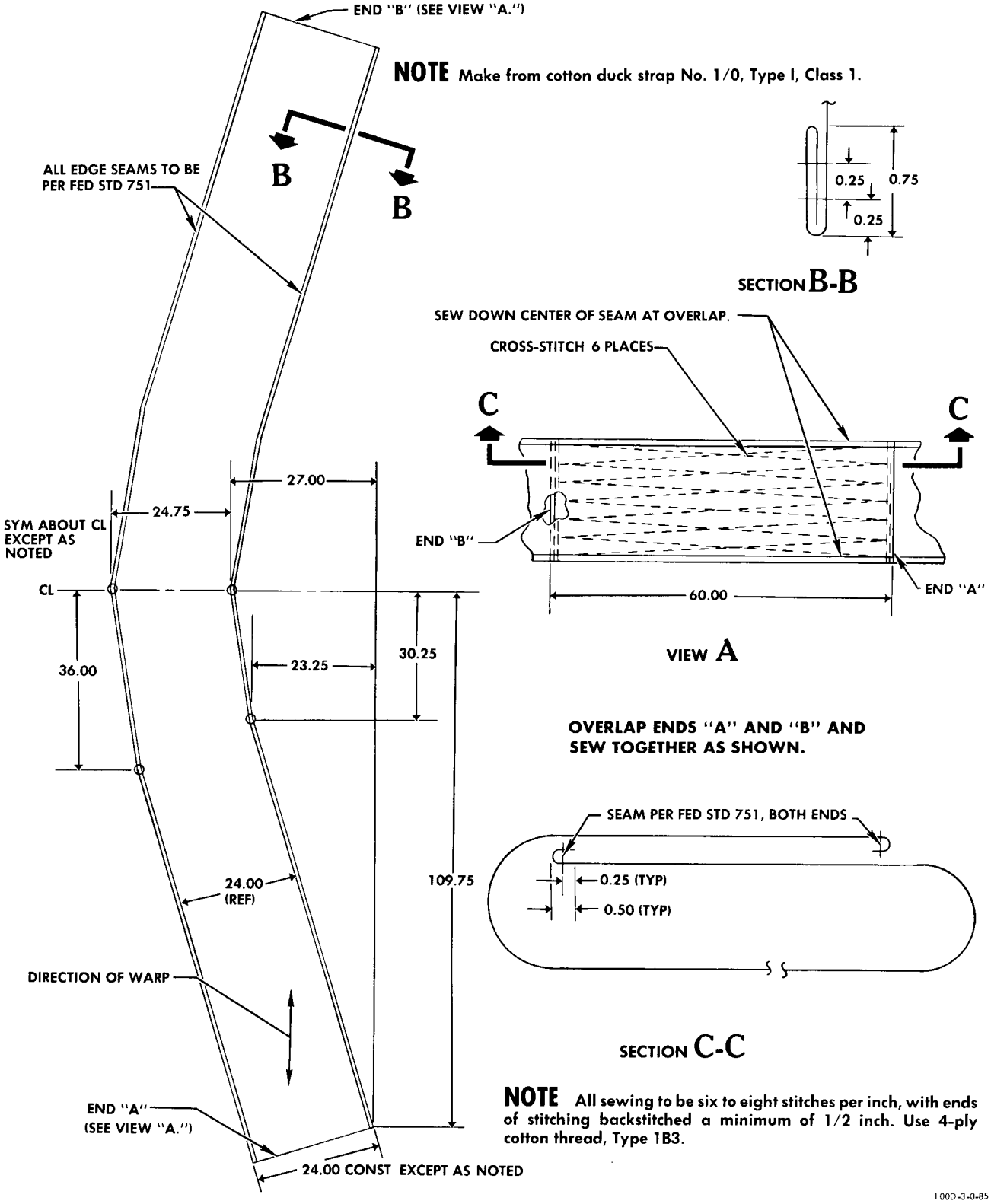
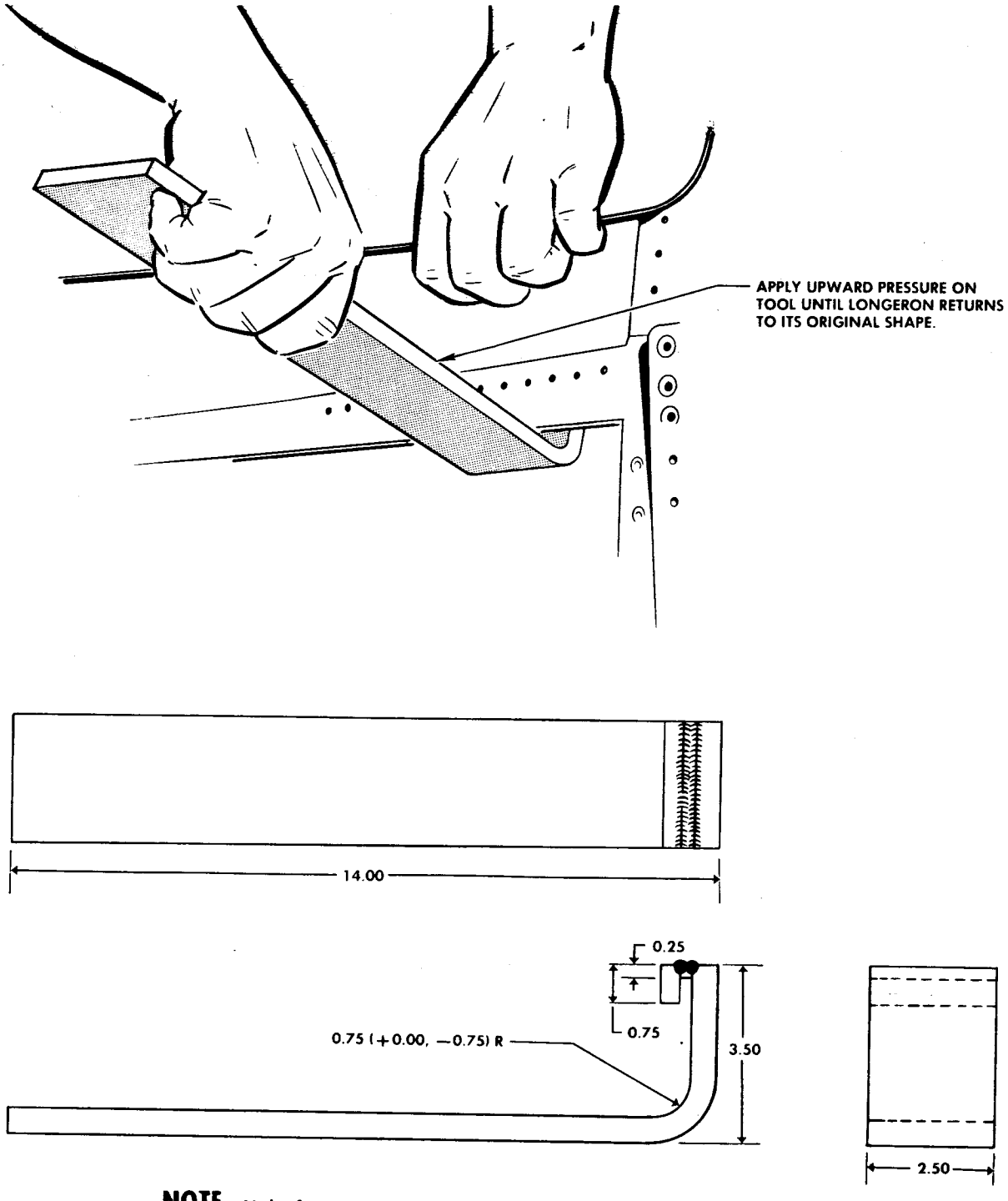


Figure 9-5. Weight Support Sling

NOSE-GEAR-UP LANDING DAMAGE

REPAIR



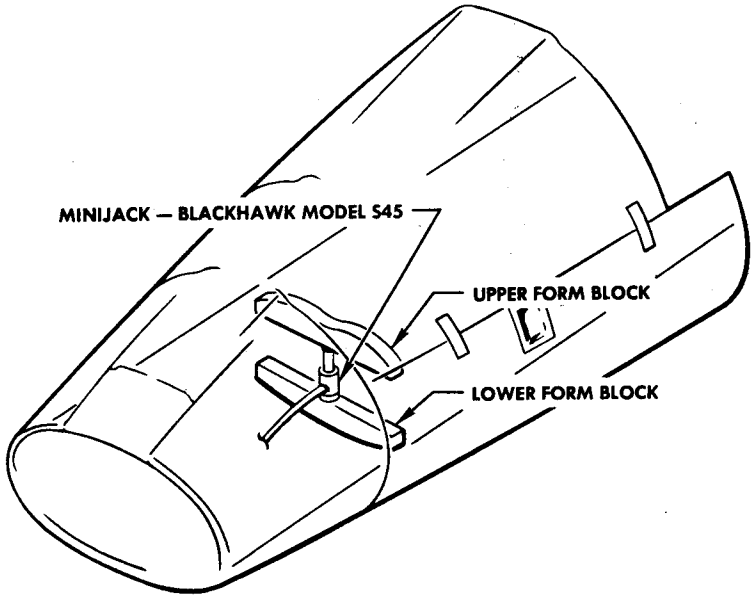
NOTE Make from 4130 steel sheet.

100D-3-0-82

Figure 9-6. Straightening Longerons

NOSE-GEAR-UP LANDING DAMAGE

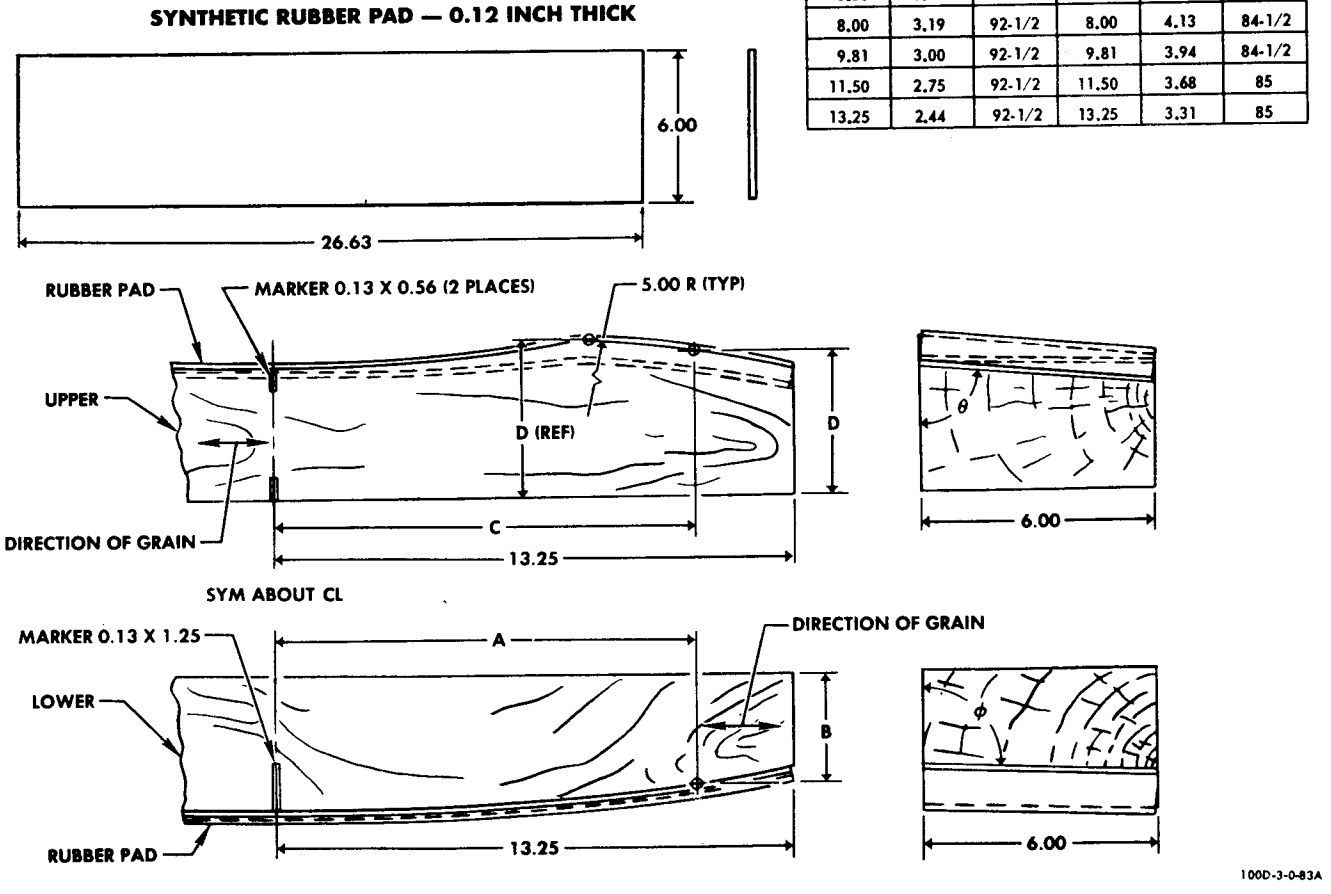
REPAIR



1 Make form blocks from white or red oak, Grade No. 1.

2 Cement rubber pads to blocks as shown, using general-purpose synthetic-base cement (Specification MIL-A-5092, Type II).

LOWER			UPPER		
A DIM. (IN.)	B DIM. (IN.)	BEVEL θ (DEG)	C DIM. (IN.)	D DIM. (IN.)	BEVEL ϕ (DEG)
0	3.50	92	0	3.44	88
1.62	3.50	92	1.62	3.47	87-1/2
3.25	3.47	92	3.25	3.53	87
4.88	3.41	92	4.88	3.66	86-1/2
6.50	3.31	92	6.50	3.88	86
8.00	3.19	92-1/2	8.00	4.13	84-1/2
9.81	3.00	92-1/2	9.81	3.94	84-1/2
11.50	2.75	92-1/2	11.50	3.68	85
13.25	2.44	92-1/2	13.25	3.31	85

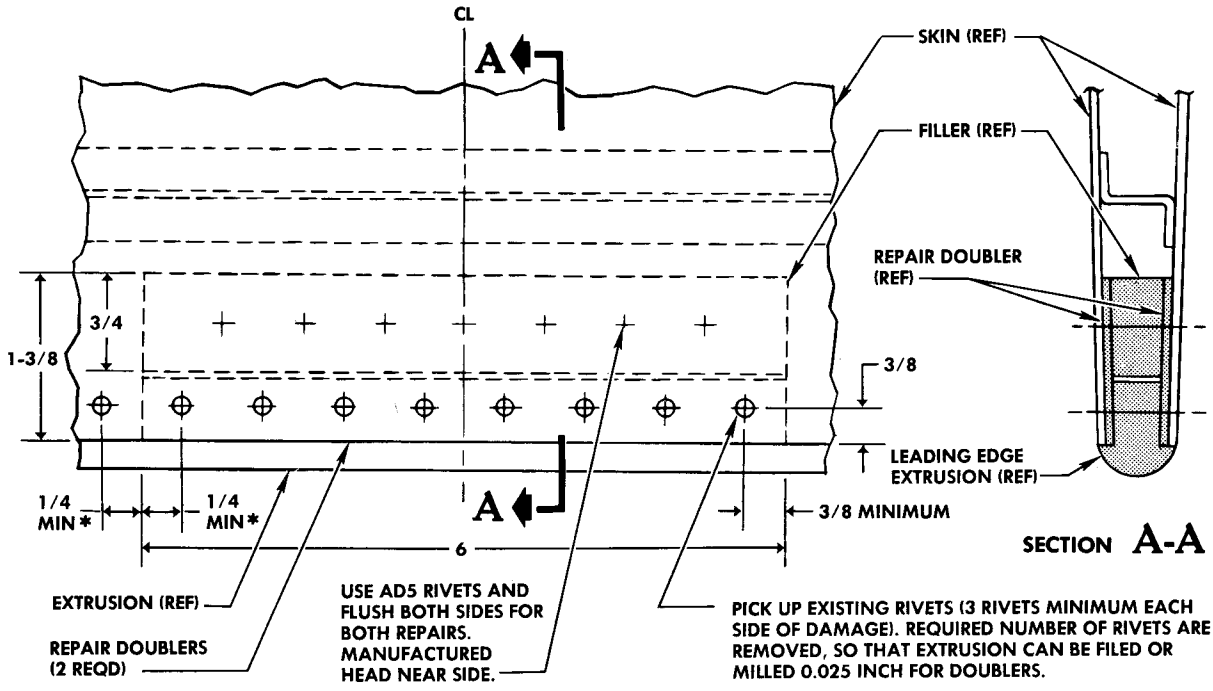


100D-3-0-83A

Figure 9-7. Straightening Intake Duct Frames

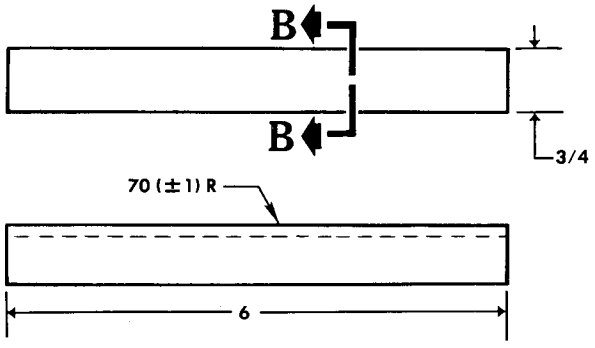
NOSE-GEAR-UP LANDING DAMAGE

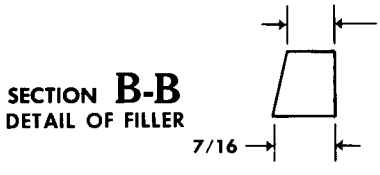
REPAIR



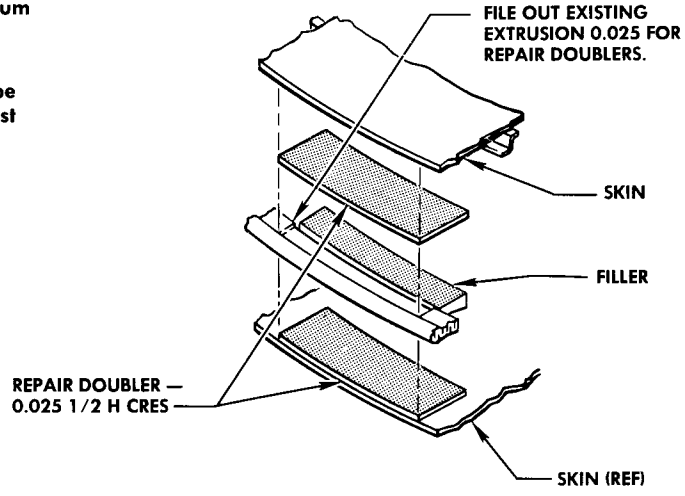
* Trim fillers and doublers as necessary to maintain minimum dimension.

NOTE Repair shown is for centerline damage. Repair may be moved up to 7 inches either side of centerline, provided at least three rivets on each side of crack are picked up.





SECTION B-B
DETAIL OF FILLER



1 Mill out existing extrusion 0.025 inch to provide for repair doublers as shown.

2 Make filler using 2024-T4 material to fit contour.

3 Make two repair doublers as shown, using 0.025-inch corrosion-resistant steel, one-half hard.

4 Assemble and rivet as shown.

100D-3-0-75

Figure 9-8. Intake Duct Leading Edge Repair

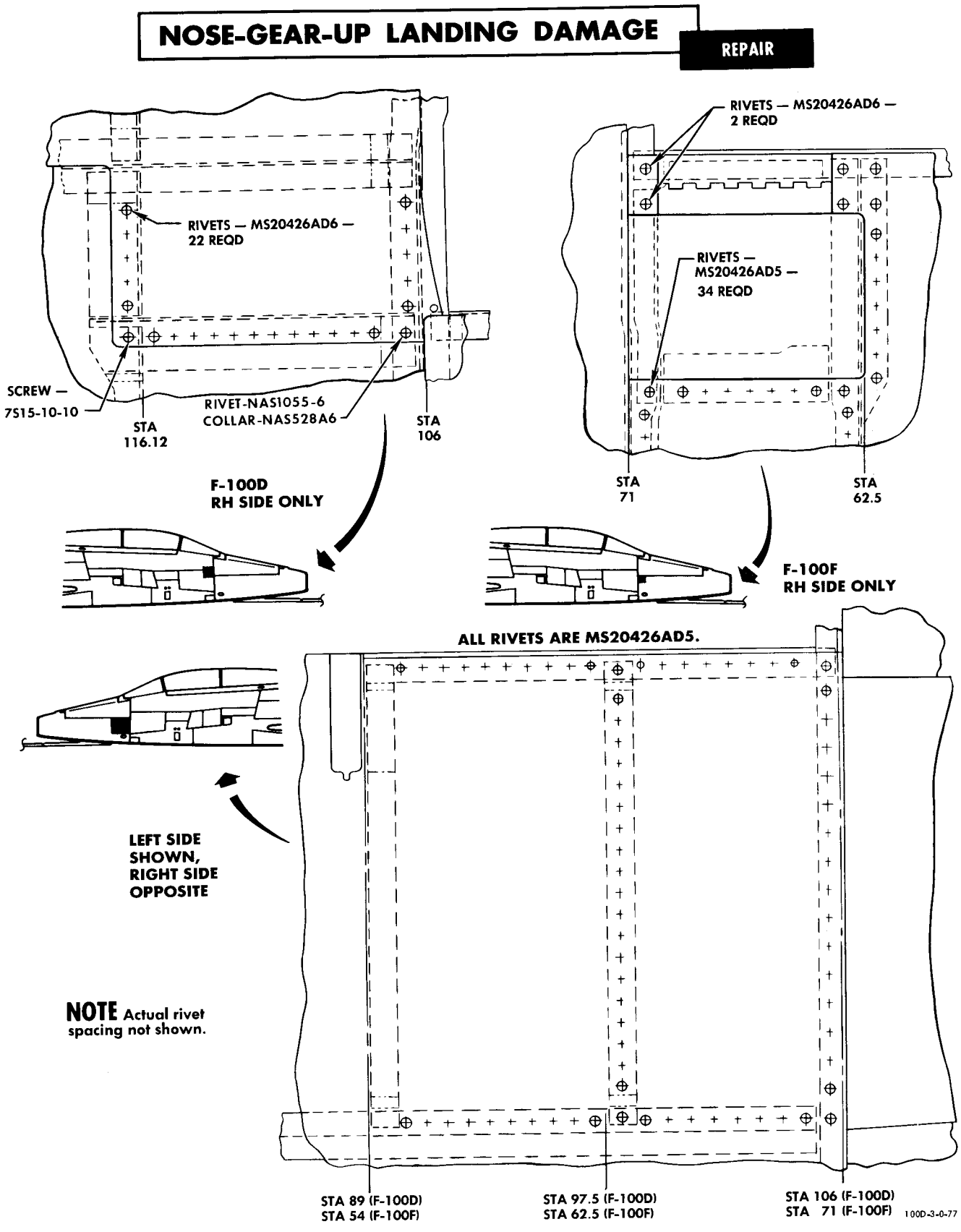
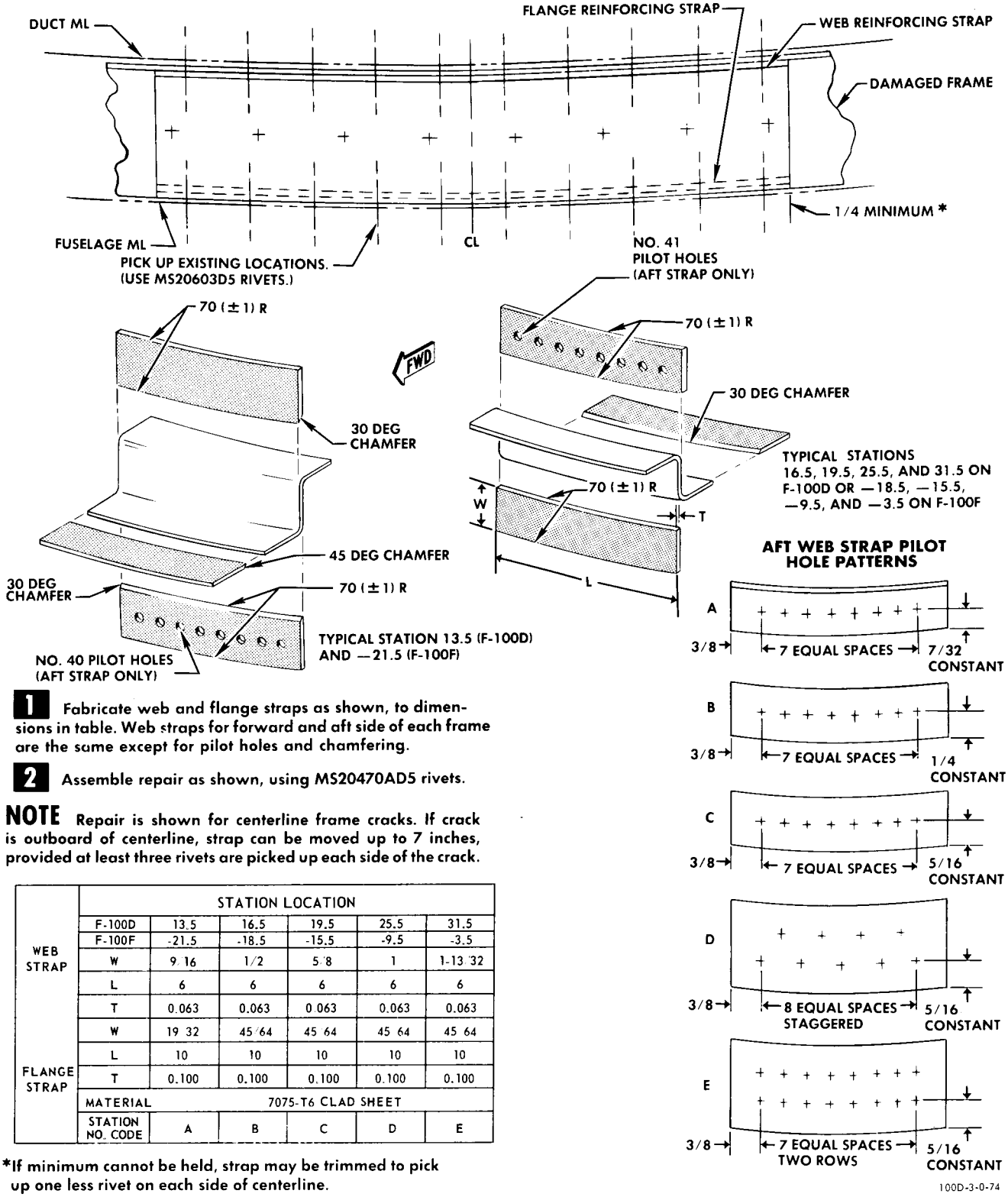


Figure 9-9. Skin Fastener Replacement

NOSE-GEAR-UP LANDING DAMAGE

REPAIR



1 Fabricate web and flange straps as shown, to dimensions in table. Web straps for forward and aft side of each frame are the same except for pilot holes and chamfering.

2 Assemble repair as shown, using MS20470AD5 rivets.

NOTE Repair is shown for centerline frame cracks. If crack is outboard of centerline, strap can be moved up to 7 inches, provided at least three rivets are picked up each side of the crack.

	STATION LOCATION					
WEB STRAP	F-100D	13.5	16.5	19.5	25.5	31.5
	F-100F	-21.5	-18.5	-15.5	-9.5	-3.5
	W	9 16	1/2	5 8	1	1-13 32
	L	6	6	6	6	6
	T	0.063	0.063	0 063	0.063	0.063
FLANGE STRAP	W	19 32	45 64	45 64	45 64	45 64
	L	10	10	10	10	10
	T	0.100	0.100	0.100	0.100	0.100
	MATERIAL	7075-T6 CLAD SHEET				
	STATION NO. CODE	A	B	C	D	E

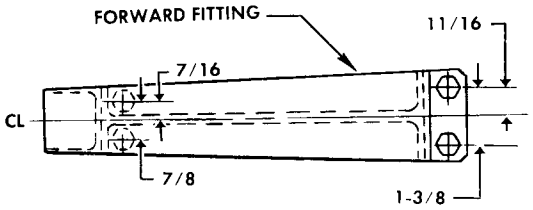
*If minimum cannot be held, strap may be trimmed to pick up one less rivet on each side of centerline.

Figure 9-10. Intake Duct Frame Repair

NOSE-GEAR-UP LANDING DAMAGE

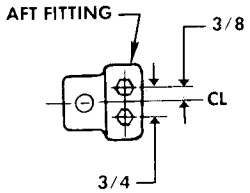
REPAIR

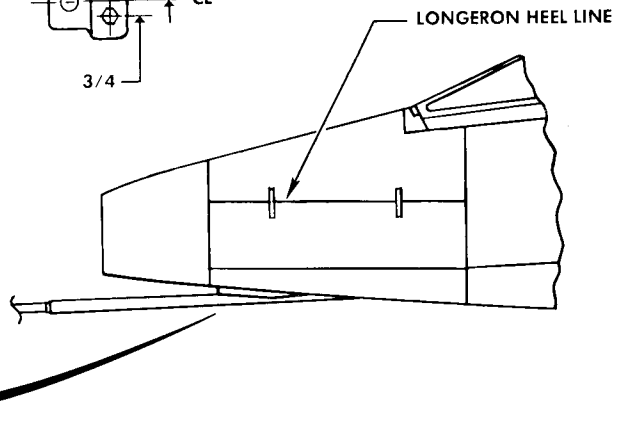
1 Locate lower centerline of airplane by measuring midpoint between heel lines of longeron under radar bay hood at three places minimum. Mark established centerline on lower frames at fuselage stations 46.5, 55, and 72 on F-100D or 11.5, 20, and 37 on F-100F.

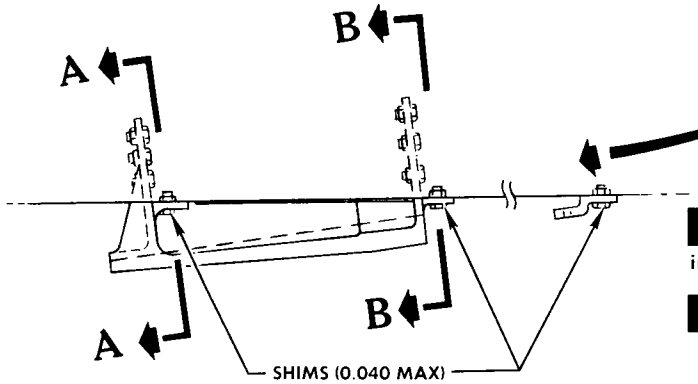


2 Temporarily install forward and aft boom support fittings on pitot boom and place entire assembly on bottom of fuselage at centerline.

3 Shim forward or aft fitting as necessary to match fuselage mold line. Any gap exceeding 0.010 inch, but less than 0.050 inch, must be shimmed. Gap exceeding 0.050 inch requires further straightening of fuselage structure.

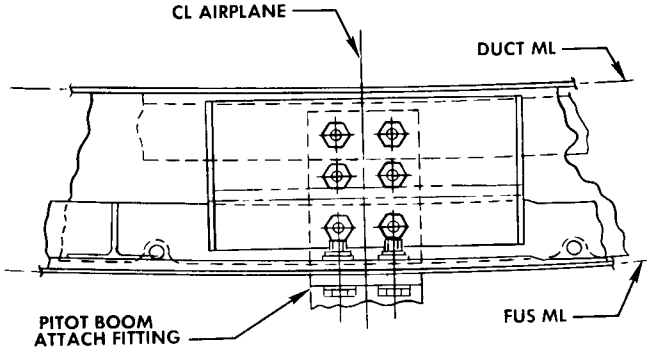




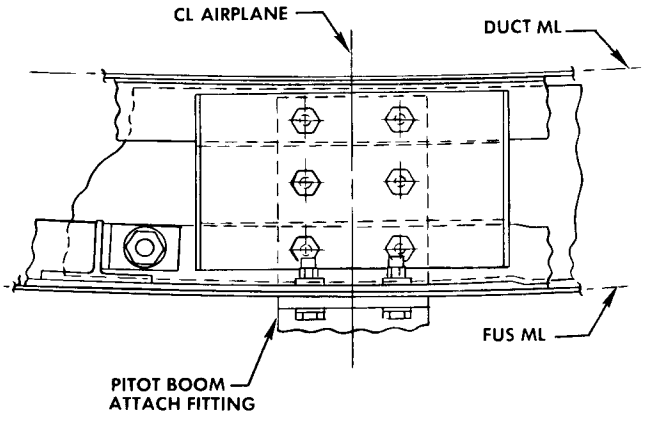


4 With support fittings in place against frames, locate holes in frames and drill boltholes in frame webs and flanges.

5 Bolt pitot fixed boom, with supports, to frames as shown.



SECTION A-A



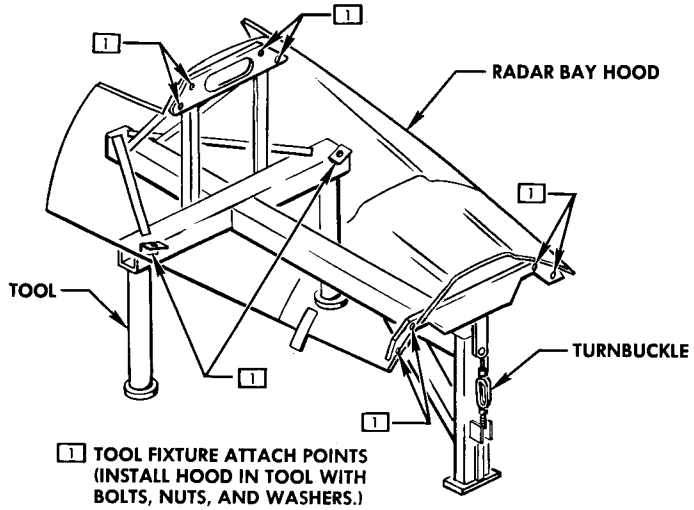
SECTION B-B

100D-3-0-76

Figure 9-11. Installing and Aligning Pitot Boom Support Fittings

NOSE-GEAR-UP LANDING DAMAGE

REPAIR



MATERIAL LIST *	
-1	WELDED STEEL TUBE, 2-1/2 INCHES SQUARE, 3/16-INCH WALL THICKNESS
-2	2024-T42 ALUMINUM PLATE, 1/2-INCH THICK
-3	STEEL ANGLE, 1-1/2 x 1-1/2 x 1/4-INCH
-4	STEEL ANGLE, 2 x 2 x 1/4-INCH
-5	STEEL ANGLE, 1-1/4 x 1-1/4 x 1/4-INCH
-6	STEEL PLATE, 1/4-INCH THICK
-7	STEEL PLATE, 1/2-INCH THICK
-8	STEEL PIPE, 1-1/2 INCH OD, 9/64-INCH WALL THICKNESS,

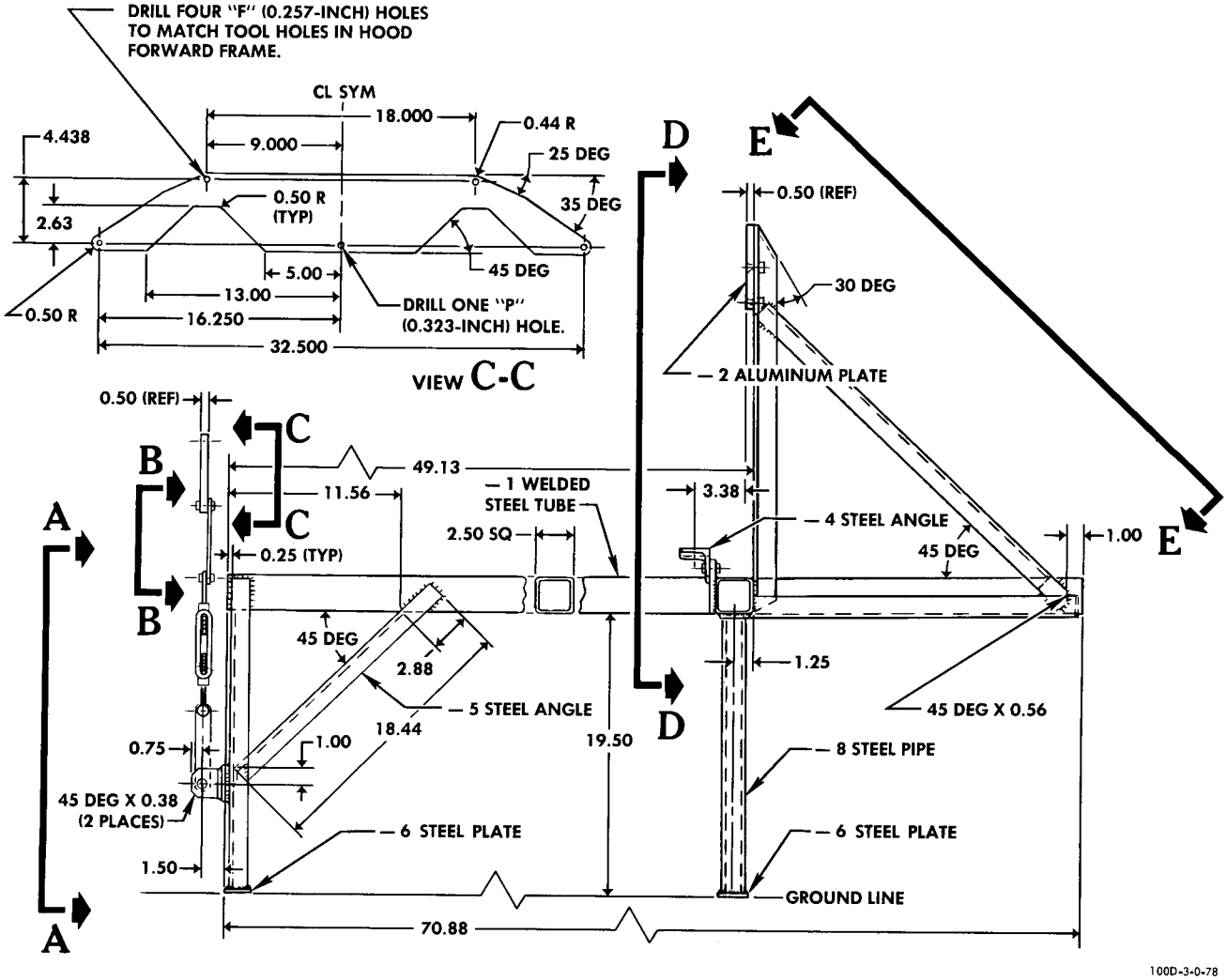


Figure 9-12. Straightening Hood Assembly (Sheet 1 of 2)

NOSE-GEAR-UP LANDING DAMAGE

REPAIR

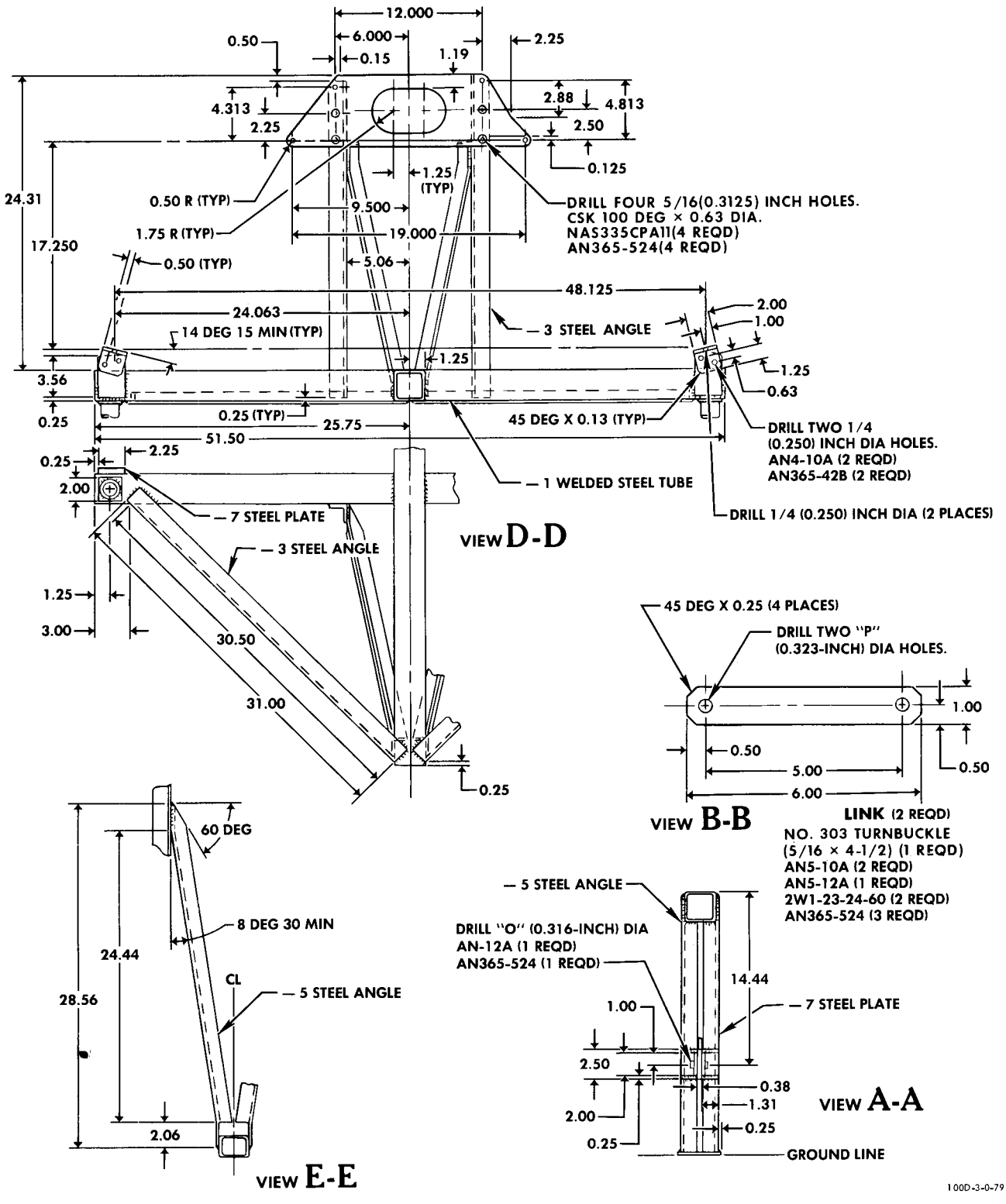


Figure 9-12. Straightening Hood Assembly (Sheet 2 of 2)

SECTION IXA

BARRIER ENGAGEMENT DAMAGE (GEAR DOWN)

Contents

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BARRIER ENGAGEMENT DAMAGE (GEAR DOWN) 9A-1
PARTS LIST 9A-1

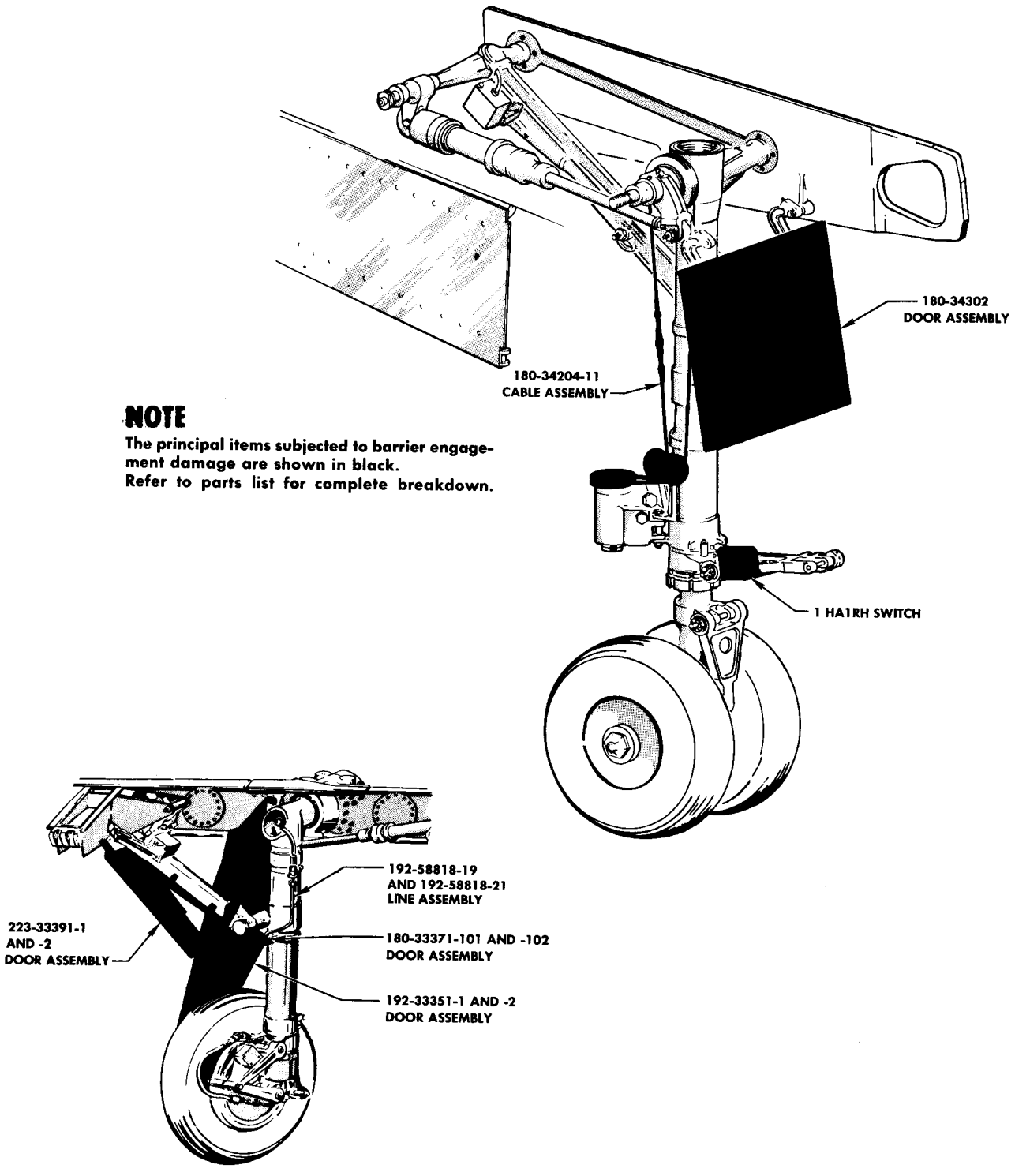
BARRIER ENGAGEMENT DAMAGE (GEAR DOWN).

Damage to the airplane resulting from gear-down barrier engagement varies with conditions. This section covers a general or average condition. From experience gained as a result of these conditions, figure 9A-1 shows the area of damage to the airplane which is most generally encountered. Refer to "Parts List" for those parts most generally required to make repairs. Drawing numbers as shown on the illustration and parts list are presented for reference only and should not be used for ordering parts. When ordering parts from stock, use applicable Illustrated Parts Breakdown (-4 handbook) or US Air Force Supply Catalog.

PARTS LIST.

DRAWING NO.	DESCRIPTION
180-33211	Bracket
180-33353	Pin
180-33360-7	Clevis
180-33366-1	Fitting Assembly
180-33366-2	Fitting Assembly
180-33371-101	Door Assembly
180-33371-102	Door Assembly
180-34204-11	Cable Assembly
180-34205	Guard
180-34221	Bracket
180-34302	Door Assembly
180-34303	Link
192-12157-1	Plate
192-12157-2	Plate
192-33218	Drive Assembly
192-33351-1	Door Assembly
192-33351-2	Door Assembly
192-33356	Washer
192-33363	Cable Assembly
192-33365-3	Fitting Assembly
192-33365-4	Fitting Assembly
192-53818-19	Line Assembly
192-53818-21	Line Assembly
223-18006-11	Trailing Edge, Flap
223-18006-12	Trailing Edge, Flap
223-33391-1	Door Assembly
223-33391-2	Door Assembly
223-33395	Pin
223-33396	Shim
223-34206	Bracket Assembly
223-58807-69	Line Assembly
223-58807-71	Line Assembly
223-58847-69	Line Assembly
1HA1RH	Switch
AN219-4	Pulley

BARRIER ENGAGEMENT DAMAGE



NOTE

The principal items subjected to barrier engagement damage are shown in black.
Refer to parts list for complete breakdown.

F-100D-3-0-73A

Figure 9A-1. Barrier Engagement Damage (Gear Down)

Section X
TYPICAL REPAIRS

Contents

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SELECTION OF TYPICAL REPAIRS	10-1
SELECTING FASTENER TABLE	10-1
FASTENER DISTRIBUTION PATTERNS	10-2
DEVELOPING A TYPICAL REPAIR	10-2

GENERAL.

Typical repairs are designed for the widest possible applications. The aim in designing repairs for areas considered to be primary structure is to rebuild the material. This is necessary because the typical repair has to be strong enough for the most heavily loaded part on which it may be used. Repairs for areas considered to be secondary structure are somewhat lighter, since the known loading in those areas is lighter. Repairs for secondary structure are skin repairs only. All substructure is to be considered as *primary structure* for repair purposes. Secondary structure repairs are noted as such in their titles. Nonstructural door repairs are also secondary structure repairs. All other repairs in this section are applicable to primary structures. Typical repair indexes show certain covers and doors as secondary structure. These covers and doors are nonstructural. However, the load-carrying ability of a structure must be continuous, so that, when a nonstructural door is installed, the load paths are transferred to the framework around the door and/or to the compartment which it covers.

TYPICAL REPAIR INDEX.

The typical repair index contains the necessary information for selection of a typical repair. All outer skins are designated as primary or secondary structure. Primary structure skins are coded as to how they are loaded; that is, whether they are loaded in shear or tension, or in combined shear and tension. This repair index gives the *type* of repair required for each area only. It permits selection of the most suitable repair within the type.

The index applies to the skins only, not to the structures underneath them. For repair purposes, all formed and extruded shapes are considered to be tension-loaded. Use tension riveting tables when calculating rivet requirements. All internal webs are considered to be loaded in shear. Use shear riveting tables when calculating requirements for web repairs. Tension repairs used in this section are actually extra-strong shear repairs. The number of rivets required for "l" (cutout dimensions) is the same, regardless of the direction of tension. A 4-inch opening would have an equal number of rivets along each of its sides. This eliminates the need for determining the directions of the tension loads.

Typical repairs may be used whenever they meet the shape and material requirements, as long as they are applied to areas as previously specified. There are no repairs in this section for tapered or sculptured skins. The following paragraphs discuss selecting the repair and fastener table, and also fastener distribution. Refer to "Developing a Typical Repair" for an itemized list of the various factors which must be considered when planning and working up a typical repair. It provides a convenient reference to the problems involved, and supplies the answers or refers to data which contains the answers.

SELECTION OF TYPICAL REPAIRS.

Typical repairs may be selected in two ways. The repair index keys in each section list the figure numbers of certain typical repairs that apply to the listed part. All possible applications, however, are too numerous to index. The second method of selecting a typical repair is by reference to the alphabetical index.

The repair titles are actually a brief description of the repair. For example: To repair a hole near a frame in a fuselage skin panel, find the damaged area on the fuselage view of the typical repair index. This diagram indicates that the damaged area is primary structure and that a shear-type repair is required. In the index, under "Repairs—Typical," select "Skin Repair Adjacent to Substructure, Shear and Tension." Each of the three sheets of this figure shows a different method of dealing with the problem. Select the method which best fits your

needs. A brief study of the repair titles, listed in the index under "Typical Repairs," familiarizes the repairman with the many types of repairs he can make by using the information in this section.

SELECTING FASTENER TABLE.

Fastener tables, listed in the alphabetical index, are titled according to the type of fastener and the shape of its head; the material or materials for which calculations were made; and how the repair is loaded (shear or tension).

The type of fastener chosen for a repair should be the one that best fits the need. In skin and web repairs, it is best to match the type and size of fastener that is in general use in the area. When it is necessary to use blind fasteners, they should be of the same size and head type as the original fasteners in the surrounding area.

Protruding head fasteners must be used to repair extruded or formed shapes, except where external skins are picked up in the fastener pattern.

Because of the space available for overlap of repair parts, several calculations may be necessary to determine the suitable fastener. Minimum (or greater) fastener edge distance spacing must be observed to obtain the required strength in the repair.

FASTENER DISTRIBUTION PATTERNS.

The fastener distribution patterns (figure 10-2) show how fastener patterns are developed for the more common types of repairs. Fastener layout determines the size and shape of the repair doubler.

The first three sheets of figure 10-2 deal with problems in skin and web repairs; the fourth sheet deals with problems in repairing extruded and roll-formed shapes. In both cases, the problems are about the same. Whenever there is a break or gap in a piece of metal, it must be joined together with another piece of metal (a repair doubler). The repair doubler must be fastened to the broken material with a given number of fasteners. The required number, as found in the fastener tables is the number that is required for *each side* of the break.

In skin and web repairs, the trimmed-out damage is usually rectangular or circular in shape. The rectangle has two directions of break, at 90 degrees to each other. Each direction is treated separately and the required fasteners are distributed on opposite sides of that particular direction of break.

The circle is considered to have but one break, its circumference. Fasteners are distributed evenly around the cutout.

A crack in sheet metal may have one or two directions, depending on the path the crack has taken. When it is a generally straight line, it is measured as a straight line and the required fasteners are distributed on each side of the break. When the path of the crack deviates from a generally straight line, it is considered to have two directions of break and is treated as a rectangular repair. The area of skin inside the rivet pattern actually becomes the "filler" material and may be tacked to the repair doubler with small structural rivets as a filler would be.

The only skin and web repair which does *not* require the specified number of fasteners on each side of the break is the overlap splice. In this repair, the fasteners required for the break are distributed within the overlap area.

In extruded and roll-formed shape repairs, the length of the break may vary from a partial damage to one flange to a complete separation of the part. In all cases, fastener calculations are made for each surface of the shape affected by the break. Fasteners required for each surface are distributed in that surface on each side of the break. Fasteners may be added in any surface to obtain a uniform fastener pattern.

DEVELOPING A TYPICAL REPAIR.

The following items, listed in the left column as problems, are factors which must be considered when making a typical repair. The right column contains either the solution or reference to where the solution will be found.

PROBLEMS	SOLUTIONS
1. Identifying damaged material.	Refer to: Material column in repair index keys. "Identification of Materials by Chemical Testing," Section I.
2. Identifying material thickness ("t" on repair figure).	Refer to: Material "Gage" column in repair index keys.
3. Determining type of material to use in the repair.	Match the damaged materiel (2024-T3 may be used when 2024-T4 was used in the original part). The following material substitutions are allowed: One-half hard CRES may be substituted for aluminum alloys in skin repairs. Use half the thickness required for the aluminum alloy. One-half hard CRES may be substituted for titanium and titanium alloy. Use same gage as required for titanium.

(Page 10-2A deleted)

PROBLEMS	SOLUTIONS
4. Determining whether primary or secondary structure repairs are required.	Refer to "Typical Repair Index."
5. Determining whether a tension-type or shear-type repair is required.	Refer to "Typical Repair Index."
6. Selecting appropriate repair.	Select the repair that best fits the need and satisfies conditions in problems 3 and 4.
7. Selecting appropriate fastener type.	Select fastener type on basis of material being repaired, ease of installation, blind area problems, and tools available. Protruding-head fasteners may be used *temporarily* if proper dimpling tools are not available. See figure 10-21 for rivet hole sizes. **Note** Hole sizes for blind fasteners are critical. Tolerances given must be maintained.
8. Selecting appropriate fastener table.	Select table which satisfies conditions in problems 1 and 4.
9. Determining (from fastener tables) number of fasteners required.	The required number of fasteners for a given "l" dimension is found where the material thickness line crosses the selected rivet column, under the given "l" dimension. To compute for a dimension not shown, add requirements for smaller dimensions which *are* shown. Example: For an "l" of 7 inches, add requirements for 4-, 2-, and 1-inch dimensions.
10. Determining minimum fastener edge ("a" dimension on repair figures).	Minimum fastener edge distance = two times the fastener diameter, measured from center of fastener (unless otherwise noted).
11. Determining minimum and maximum fastener spacing ("b" dimension on repair figures).	Minimum fastener spacing = four times the fastener diameter. Maximum spacing = 1½ inches. Spacing is measured from centers of fasteners.
12. Determining fastener distribution in skin and web repairs.	See figure 10-2. (The referenced figure shows fastener distribution patterns in one-piece doubler repairs. Fastener distribution for split doublers and special situations are noted on the repair figure.)
13. Determining fastener distribution in extruded- and formed-shape repairs.	See figure 10-2. (A roll-formed "Z" section is used in the fastener distribution examples. These examples, however, apply to all roll-formed and extruded shapes, for which repairs have been designed. Special fastener problems are noted on the repair figure.)

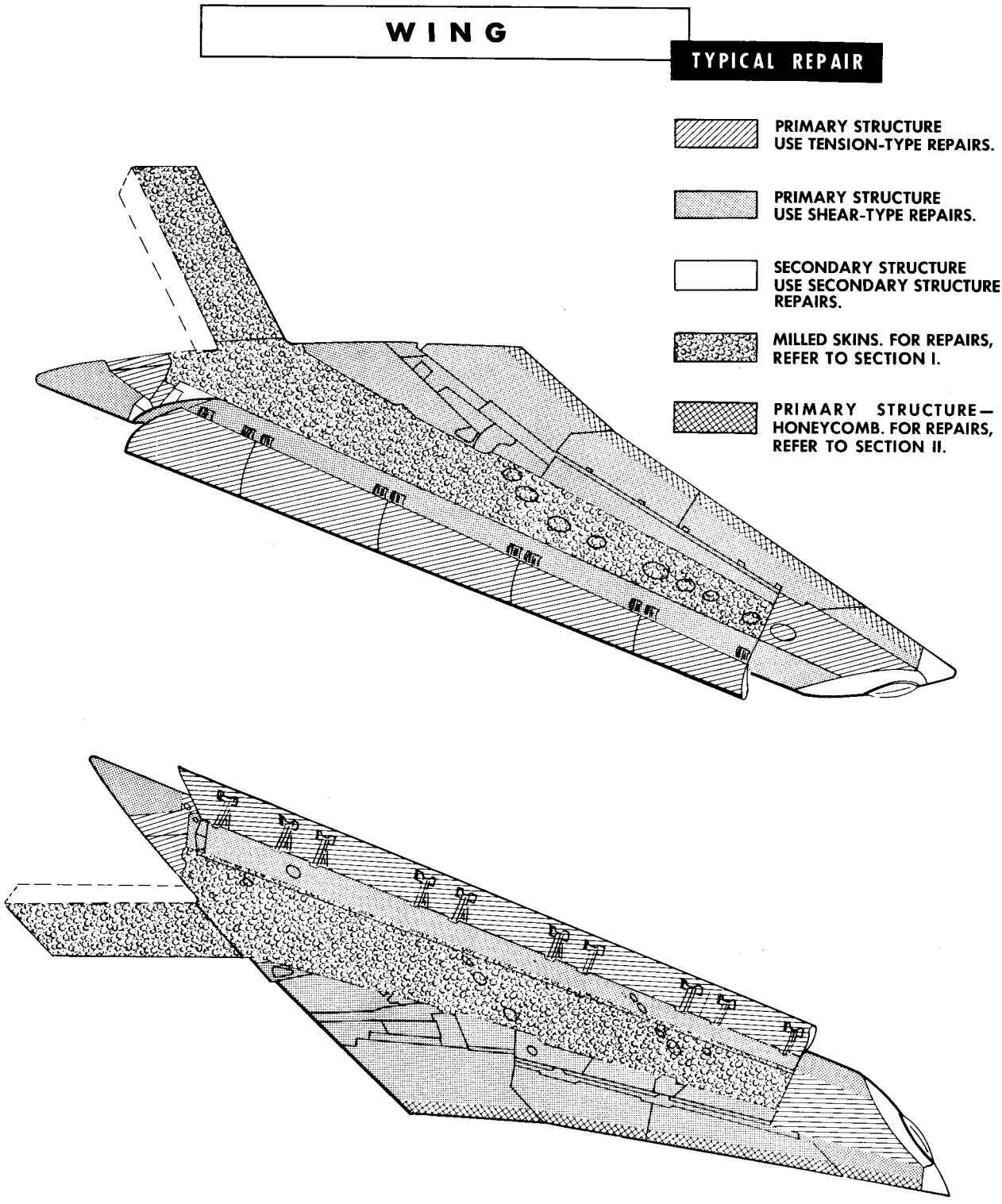


Figure 10-1. Typical Repair Index (Sheet 1 of 4)

(Pages 10-4A thru 10-4B, figures 10-1A and 10-2A deleted)

EMPENNAGE

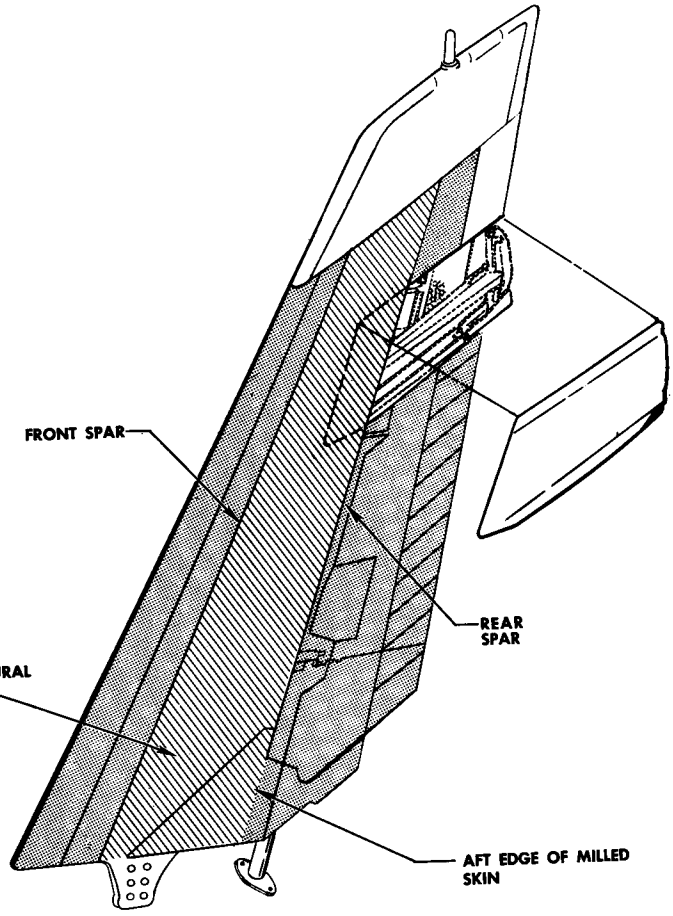
TYPICAL REPAIR

PRIMARY STRUCTURE
(Use tension-type repairs.)

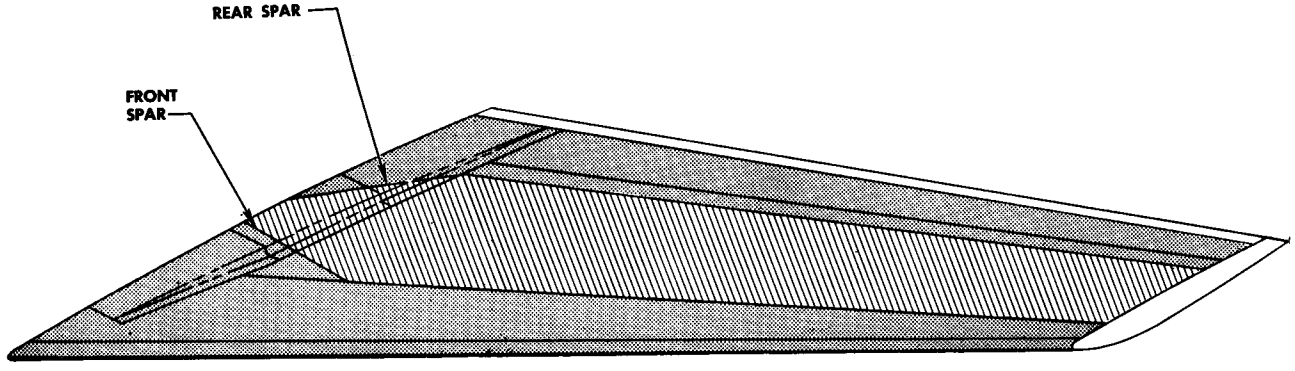
PRIMARY STRUCTURE
(Use shear-type repairs.)

SECONDARY STRUCTURE
(Use secondary structure repairs.)

NOTE Repairs in this section are designed for skins of constant thickness and are not to be used on tapered or sculptured skins.



VERTICAL STABILIZER ASSY



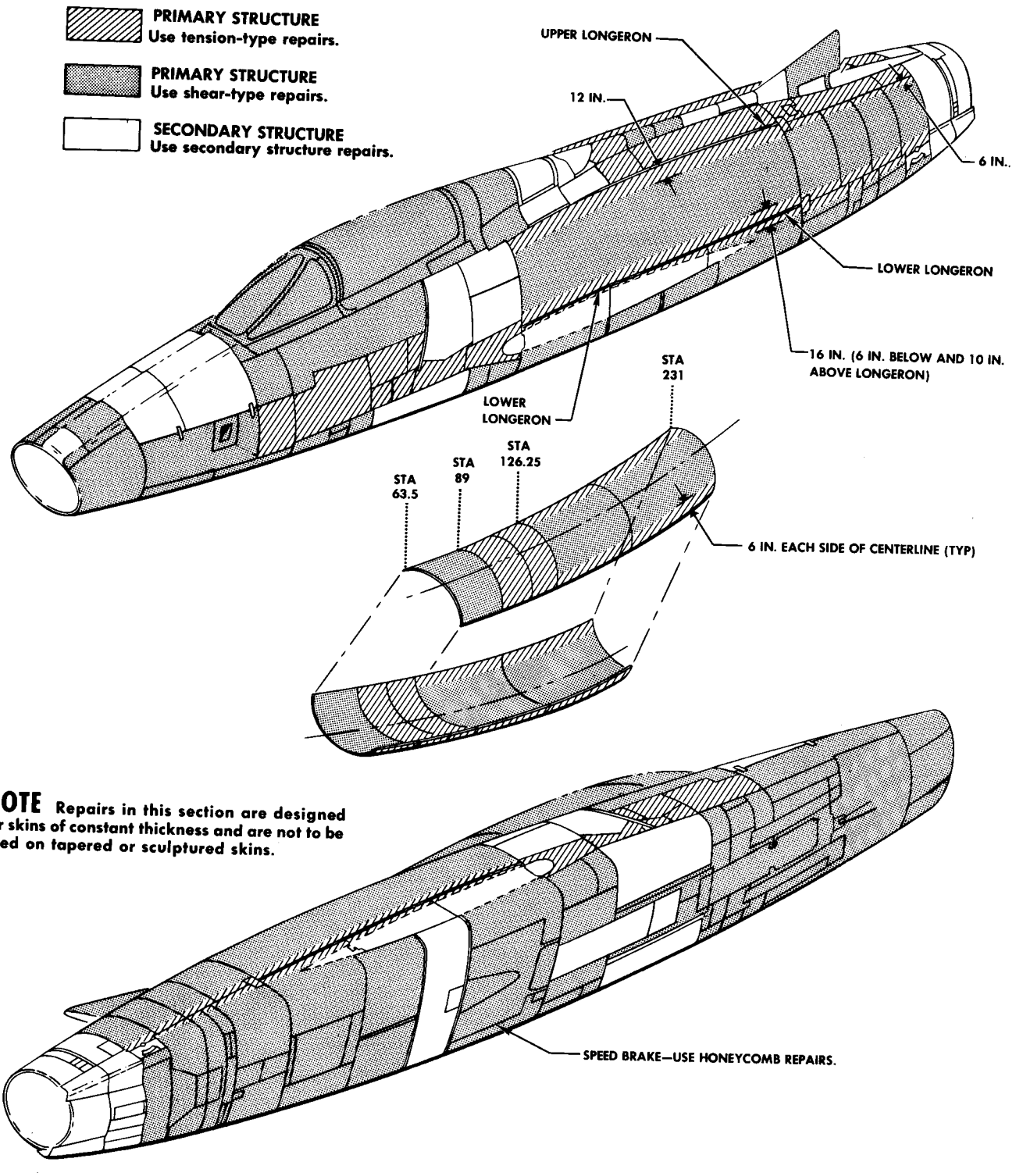
HORIZONTAL STABILIZER

F-100D-3-01-97

Figure 10-1. Typical Repair Index (Sheet 2 of 4)

FUSELAGE - F-100D AIRPLANE

TYPICAL REPAIR



NOTE Repairs in this section are designed for skins of constant thickness and are not to be used on tapered or sculptured skins.

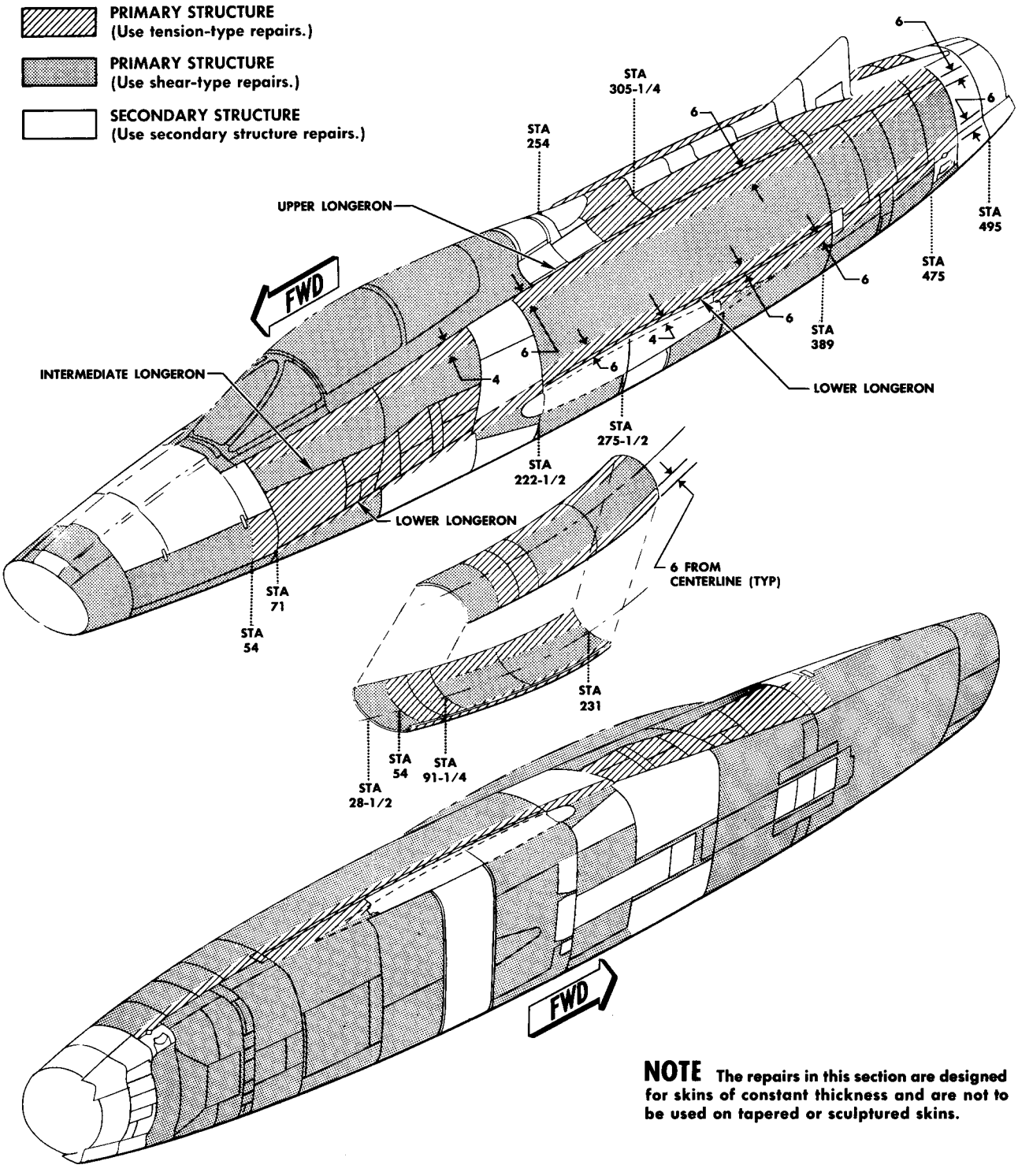
F-100D-3-91-96

Figure 10-1. Typical Repair Index (Sheet 3 of 4)

(Pages 10-6A thru 10-6B, figures 10-4A thru 10-4B deleted) **Revised 29 November 1957**

FUSELAGE – F-100 F AIRPLANE

TYPICAL REPAIR



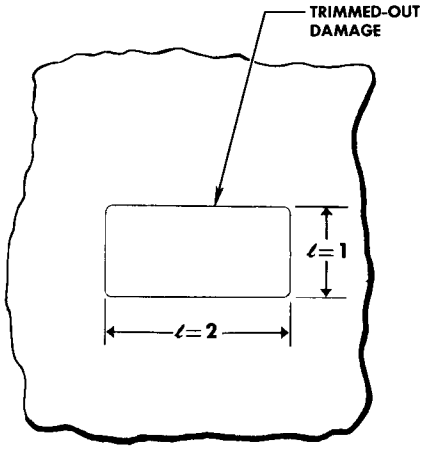
NOTE The repairs in this section are designed for skins of constant thickness and are not to be used on tapered or sculptured skins.

F-100F-3-91-4

Figure 10-1. Typical Repair Index (Sheet 4 of 4)

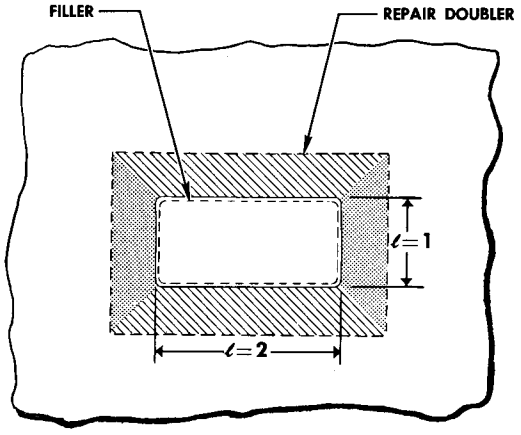
SKIN AND WEB

TYPICAL REPAIR



RECTANGULAR HOLE

A rectangular hole has two "ℓ" dimensions. If one-inch side shown requires five fasteners, use five fasteners on *each* side of one-inch opening. If two-inch side requires ten fasteners, use ten fasteners on *each* side of two-inch opening.



NORMAL FASTENER DISTRIBUTION

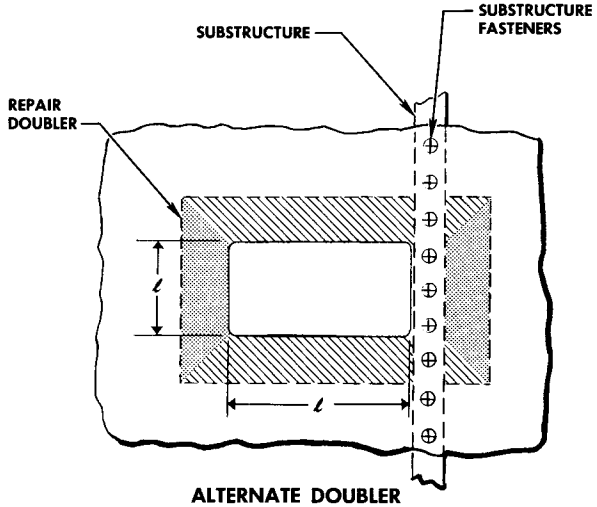
FASTENERS FOR 1 IN. SIDES.

FASTENERS FOR 2 IN. SIDES.

Divide area around hole as shown. Lay out fasteners in an approximately even pattern around hole. Watch minimum edge distance and minimum and maximum spacing.

The fastener pattern will determine size of repair doubler. Fasteners in filler may be the smallest structural-type rivet available.

See alternate fastener and doubler pattern.



ALTERNATE DOUBLER AND FASTENER PATTERN

This pattern is used when existing substructure fasteners are *not* counted in the number required for the repair.

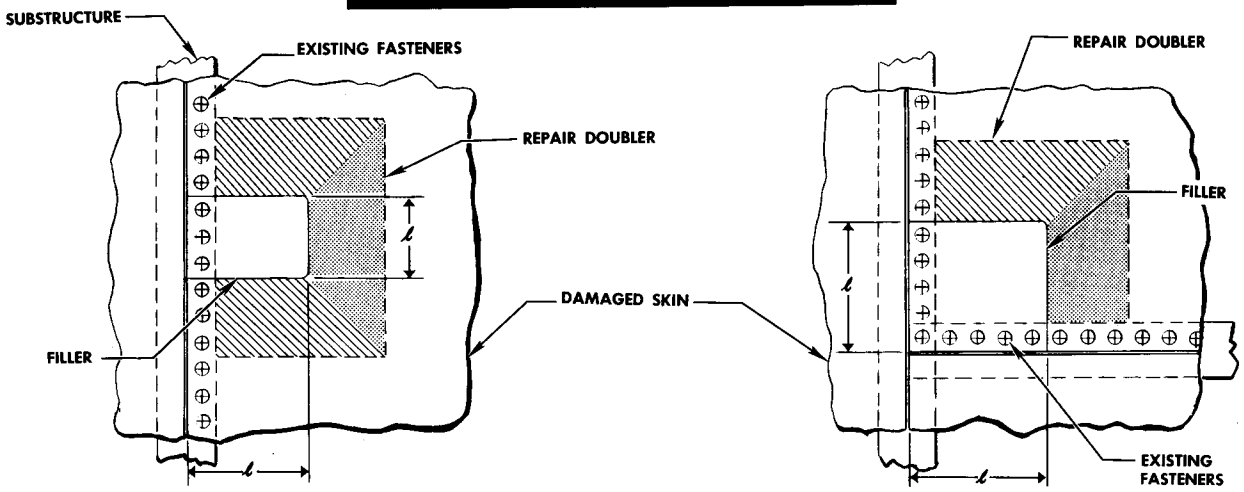
GEN-3-91-85A

Figure 10-2. Fastener Distribution Patterns (Sheet 1 of 4)

SKIN AND WEB

TYPICAL REPAIR

VARIATIONS OF THE RECTANGULAR REPAIR

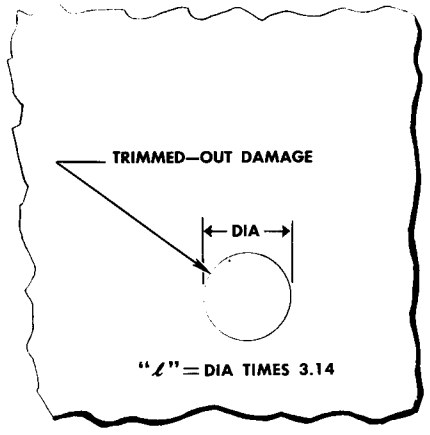


SKIN END REPAIR

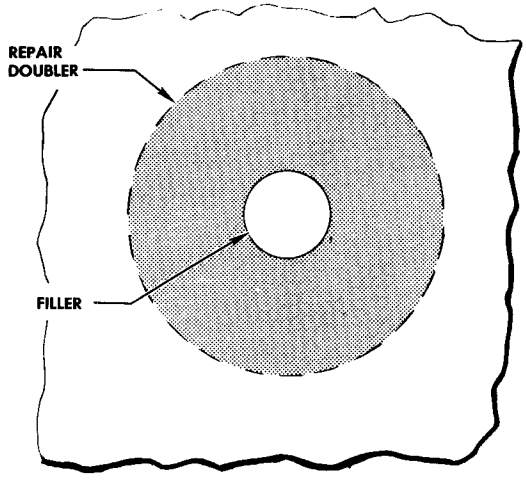
SKIN END REPAIR—CORNER

Repairs to skin end are treated the same as rectangular repairs shown on sheet No. 1, except that skin attaching fasteners through the substructure are not counted in the fasteners required for the repair.

CIRCULAR REPAIR



A circular cutout has one "ℓ" dimension, which is the circumference of the hole. To find "ℓ," multiply the diameter by pi. (3.14)



Distribute fasteners required evenly around hole in shaded area (repair doubler).

Watch minimum edge distance and spacing. Fasteners in filler may be the smallest structural-type rivet available.

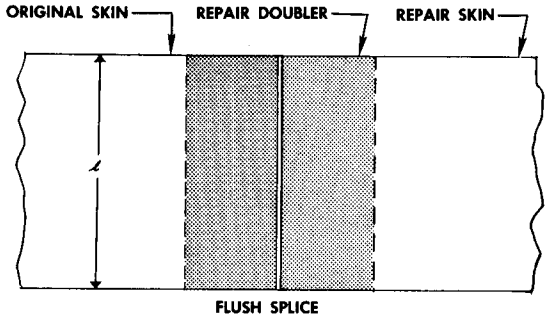
Gen-3-91-86

Figure 10-2. Fastener Distribution Patterns (Sheet 2 of 4)

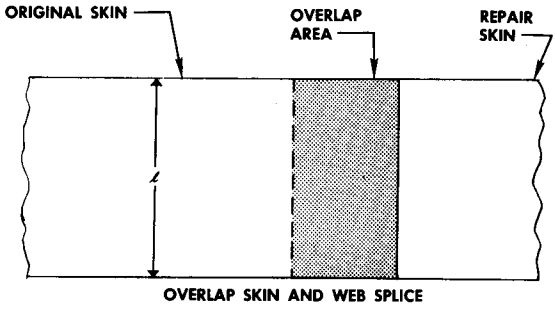
SKIN AND WEB

TYPICAL REPAIR

SPLICES

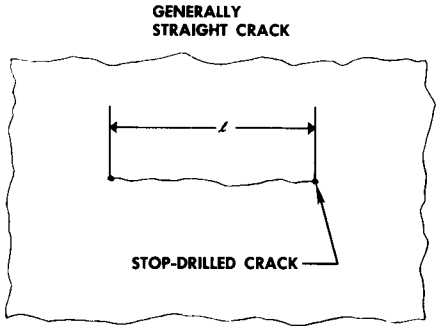


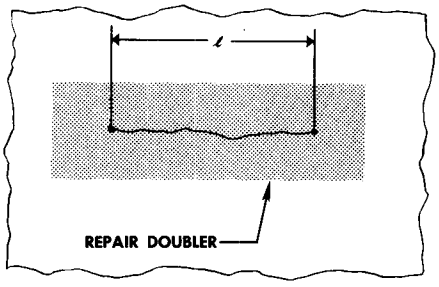
Distribute the number of fasteners required for "ℓ" evenly on each side of the splice. (If "ℓ" requires 24 fasteners, use 24 fasteners on each side of the splice.)



Required fasteners for "ℓ" are distributed evenly in the overlap area.

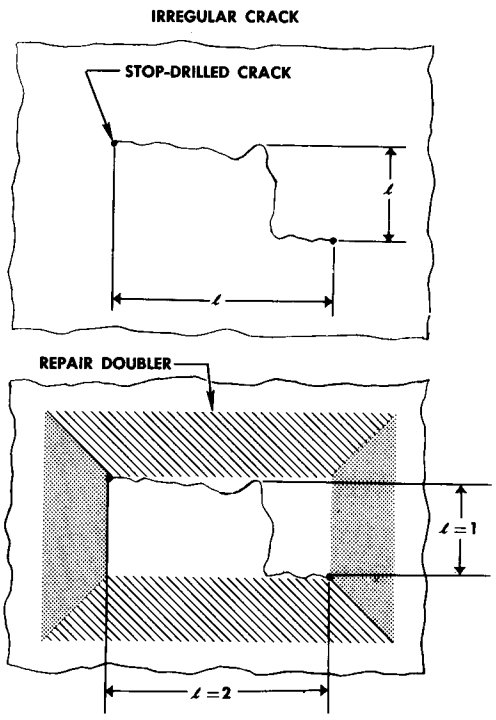
CRACKS





A crack which runs in a generally straight line can be considered to have one "ℓ" dimension.

If six rivets are required for "ℓ," distribute six rivets evenly through the doubler on each side of the crack.



A crack which deviates from a generally straight line should be treated as a rectangle with two "ℓ" dimensions.

Distribute rivets as shown and described for a rectangular repair.

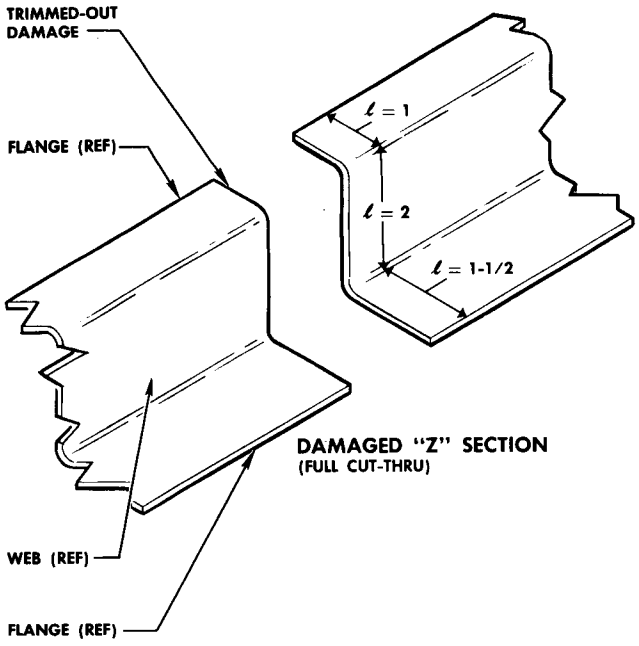
Use smallest structural-type rivets available to hold down irregular edges.

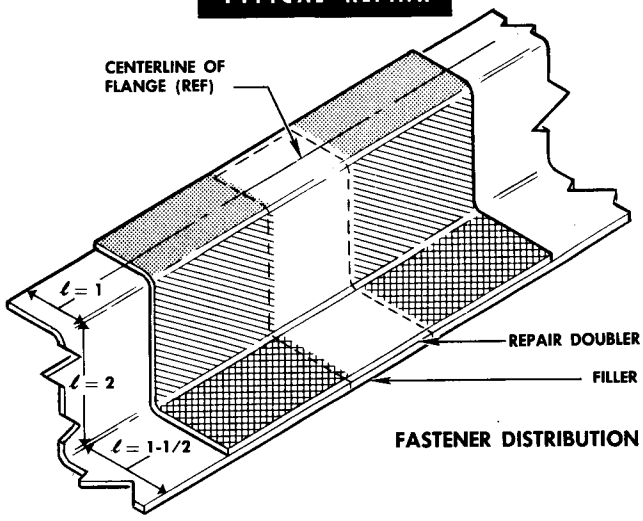
Gen-3-91-87

Figure 10-2. Fastener Distribution Patterns (Sheet 3 of 4)

EXTRUDED AND FORMED SHAPES

TYPICAL REPAIR



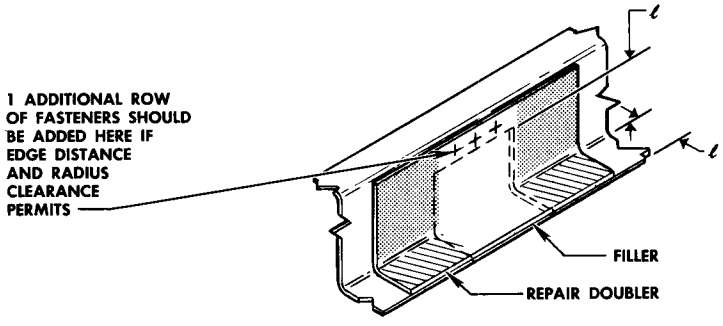


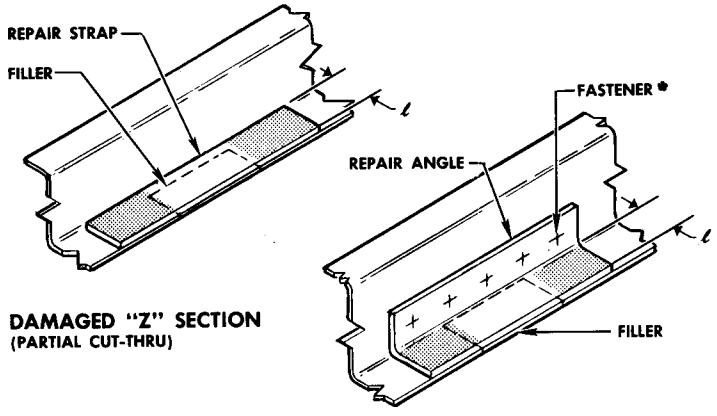
FASTENERS FOR 1-INCH FLANGE

FASTENER FOR 2-INCH WEB

FASTENERS FOR 1-1/2-INCH FLANGE

Each "ℓ" dimension shown is considered separately. If four fasteners are required for the one-inch flange, use four fasteners on each side of the one-inch opening. Repeat the calculations for the 2-inch web and the 1-1/2-inch flange. Distribute the fasteners as shown.





The fastener pattern should be similar on each side of the break.

When a flange has one row of fasteners, that row should be approximately on the centerline of the flange.

Observe minimum edge distance and minimum and maximum spacing.

Fasteners in the filler may be the smallest structural-type rivet available unless the filler is attached to a skin or web. In this case, fasteners should match those used in the repair.

NOTE

The examples of fastener distribution shown apply to all other shapes for which typical repairs have been designed. Special fastener problems are noted on the repair figure.

*When an angle is used in repairing a partial flange damage, the fasteners attaching the repair angle to the undamaged portion of the shape are the same as those used in the remainder of the repair. The number required is determined by the length of the repair angle. Fastener spacing may be a maximum of 1-1/2 inches.

Gen-3-91-88

Figure 10-2. Fastener Distribution Patterns (Sheet 4 of 4)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1 — COUNTERSUNK

MATL	t	DOUBLER MATL	ℓ =	t_1*	FASTENERS REQUIRED: 1/2 INCH										1 INCH										2 INCHES										4 INCHES									
					AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
2024-T4	0.032	2024-T4	0.040		5								10								21								41							
	0.040		0.050		6	4							12	8							24	15							47	29						
	0.050		0.063		7	5	3			3			14	9	6			5			28	17	12			9			56	34	24			18		
	0.063		0.071		9	6	4	3		3	2		18	11	8	5		5	4		35	21	15	9		9	8		70	42	29	17		18	15	
	0.071		0.080			6	4	3		3	2	2		12	8	5		5	4	4		23	16	9		9	7	7			46	32	18		18	14	13
	0.080		0.090			7	5	3		3	2	2		13	9	5		5	4	3		25	17	10		9	7	6			50	35	20		18	14	12
	0.090		0.100				5	3	2	3	2	2			10	6	4	5	4	3			19	11	7	9	7	6			38	21	14	18	14	11
	0.100		0.125				5	3	2	3	2	2			11	6	4	5	4	3			21	12	8	9	7	6			41	23	15	17	14	11
	0.125		0.160				4	2	2	2	2	2			7	4	4	4	4	3			13	8	8	7	6				26	16	16	14	11	
2024-T4	0.160	2024-T4	0.180				4	3	2	3	2				8	5	5	3	3				16	10	9	7	6				32	19	18	13	11	

TABLE NO. 2 — DIMPLED

MATL	t	DOUBLER MATL	ℓ =	t_1*	FASTENERS REQUIRED: 1/2 INCH										1 INCH										2 INCHES										4 INCHES									
					AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
2024-T4	0.020	2024-T4	0.025		3	2							6	4							11	8							22	15						
	0.025		0.032		3	2	2						6	4	3						12	8	6						24	16	12					
	0.032		0.040		4	3	2	2					7	5	4	3					14	9	7	5					28	18	13	10				
	0.040		0.050		5	3	2	2	2				9	5	4	3	3				17	10	7	5	6				33	20	15	10	11			
	0.050		0.063			3	2	2	2	2				6	5	3	3	3				12	9	6	5	6				24	17	11	9	11			
	0.063		0.071				3		2	2	2				6		3	3	3				11		5	6	5					21		9	11	9	
	0.071		0.080						2		1	1					3		3	2					5		5	4						9	9	8	
	0.080		0.090						2		1	1					3		2	2					5		4	4						10	8	7	
2024-T4	0.090	2024-T4	0.100						2								3								5									10								

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

† Dimple t and subcountersink t_1.

NOTE Refer to "Developing a Typical Repair" before using this table.

GEN-3-91-15A

Figure 10-3. Fastener Tables—Conventional and Hi-Shear, Flush, 2024-T4, Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

MATL	t	COUNTERSUNK DOUBLER MATL	∠=	t1*	FASTENERS REQUIRED 1/2 INCH										1 INCH										2 INCHES										4 INCHES									
					AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
2024-T4	0.032	2024-T4		0.040	4								7								13								25							
	0.040			0.050	4	3							7	5							15	9							29	18						
	0.050			0.063	5	3	2			2			9	6	4			3			17	11	8			6			34	21	15			11		
	0.063			0.071	6	4	3	2		2	2		11	7	5	3		3	3		22	13	9	6		6	5		43	26	18	11		11	9	
	0.071			0.080		4	3	2		2	2	1		7	5	3		3	3	2		14	10	6		6	5	4		28	20	11		11	9	8
	0.080			0.090		4	3	2		2	2	1		8	6	3		3	3	2		15	11	6		6	5	4		30	21	12		11	9	7
	0.090			0.100			3	2	2	2	1	1			6	4	3	3	2	2			12	7	5	6	4	4			23	13	9	11	8	7
	0.100			0.125			4	2	2	2	2	1			7	4	3	3	3	2			13	7	5	5	5	4			25	14	9	10	9	7
	0.125			0.160				2	2	2	1	1				4	3	3	2	2				8	5	5	4	4				16	10	10	8	7
2024-T4	0.160	2024-T4		0.190				3	2	2	1	1				5	3	3	2	2				10	6	6	4	4				19	12	11	8	7

TABLE NO. 2

MATL	t	DIMPLED DOUBLER MATL	∠=	t1*	FASTENERS REQUIRED 1/2 INCH										1 INCH										2 INCHES										4 INCHES									
					AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
2024-T4	0.020	2024-T4		0.025	2	2							4	3							7	5							13	10						
	0.025			0.032	2	2	1						4	3	2						8	5	4						15	10	8					
	0.032			0.040	3	2	1	1					5	3	2	2					9	6	4	3					17	11	8	6				
	0.040			0.050	3	2	2	1	1				5	3	3	2	2				10	6	5	3	4				20	12	9	6	7			
	0.050			0.063		2	2	1	1	1				4	3	2	2	2				8	5	4	3	4				15	10	7	6	7		
	0.063			0.071			2		1	1	1				4		2	2	2				7		3	4	3				13		6	7	6	
	0.071			0.080					1	1	1	1					2	2	2	2					3	4	3	3					6	7	6	5
	0.080			0.090					1	1	1	1					2	2	2	2					3	4	3	3					6	7	6	5
2024-T4	0.090	2024-T4		0.100					1	1	1						2	2	2						4	4	3						7	6	5	

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

†Dimple t and subcountersink t_1.

GEN-3-91-16A

Figure 10-4. Fastener Tables—Conventional and Hi-Shear, Flush, 2024-T4, Shear

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

MATL	COUNTERSUNK t	DOUBLER MATL	t1*	FASTENERS REQUIRED: 1/2 INCH AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	1 INCH AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	2 INCHES AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	4 INCHES AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
7075-T6	0.032	7075-T6	0.040	7								14								27								53							
	0.040		0.050	8	5							16	10							31	20							62	39						
	0.050		0.063	10	6	4			3			19	12	8			5			37	23	16			10			74	45	32			19		
	0.063		0.071	11	7	5	3		3	2		22	13	9	6		5	4		44	27	19	11		10	8		88	53	37	22		19	15	
	0.071		0.080		8	5	3		3	2	2		15	10	6		5	4	3		29	20	12		9	8	7		58	40	23		18	15	13
	0.080		0.090		8	6	3		3	2	2		16	11	6		5	4	3		31	22	13		9	7	6		62	43	25		17	14	12
	0.090		0.100			6	4	3	2	2	2			12	7	5	4	4	3			24	13	9	9	7	6			47	26	17	17	14	12
	0.100		0.125			7	4	3	2	2	2			13	7	5	4	4	3			26	15	9	8	7	6			51	29	18	15	14	12
	0.125		0.160				5	3	3	2	2				9	5	5	3	3				17	10	9	6	6				33	20	18	12	11
7075-T6	0.160	7075-T6	0.190				5	3	3	2	2				10	6	6	4	3				20	12	11	7	5				40	23	22	13	10

TABLE NO. 2

MATL	DIMPLED t	DOUBLER MATL	t1*	FASTENERS REQUIRED: 1/2 INCH † AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	1 INCH † AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	2 INCHES † AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10	4 INCHES † AD3	AD4	AD5	DD6	DD8	NAS177-6	NAS177-8	NAS177-10
7075-T6	0.020	7075-T6	0.025	4	3							7	5							14	10							28	20						
	0.025		0.032	4	3	2						8	5	4						16	11	8						32	21	16					
	0.032		0.040	5	3	2	2					10	6	4	3					19	12	8	6					37	23	16	12				
	0.040		0.050	6	4	3	2	2				11	7	5	3	3				22	14	9	6	5				44	27	17	12	10			
	0.050		0.063	7	4	3	2	2	2			14	8	5	4	3	4			27	16	10	7	5	7			53	32	20	14	9	13		
	0.063		0.071			3		2	2	2				6		3	3	3				12		5	6	5				23		10	12	10	
	0.071		0.080					2		2	1					3		3	2					5		5	4					10		10	9
7075-T6	0.080	7075-T6	0.090					2		2	1					3		3	2					6		5	4					11		10	8

† Dimple t and subcountersink t_1.

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-17A

Figure 10-5. Fastener Tables—Conventional and Hi-Shear, Flush, 7075-T6, Tension

FASTENER TABLES

TYPICAL REPAIR

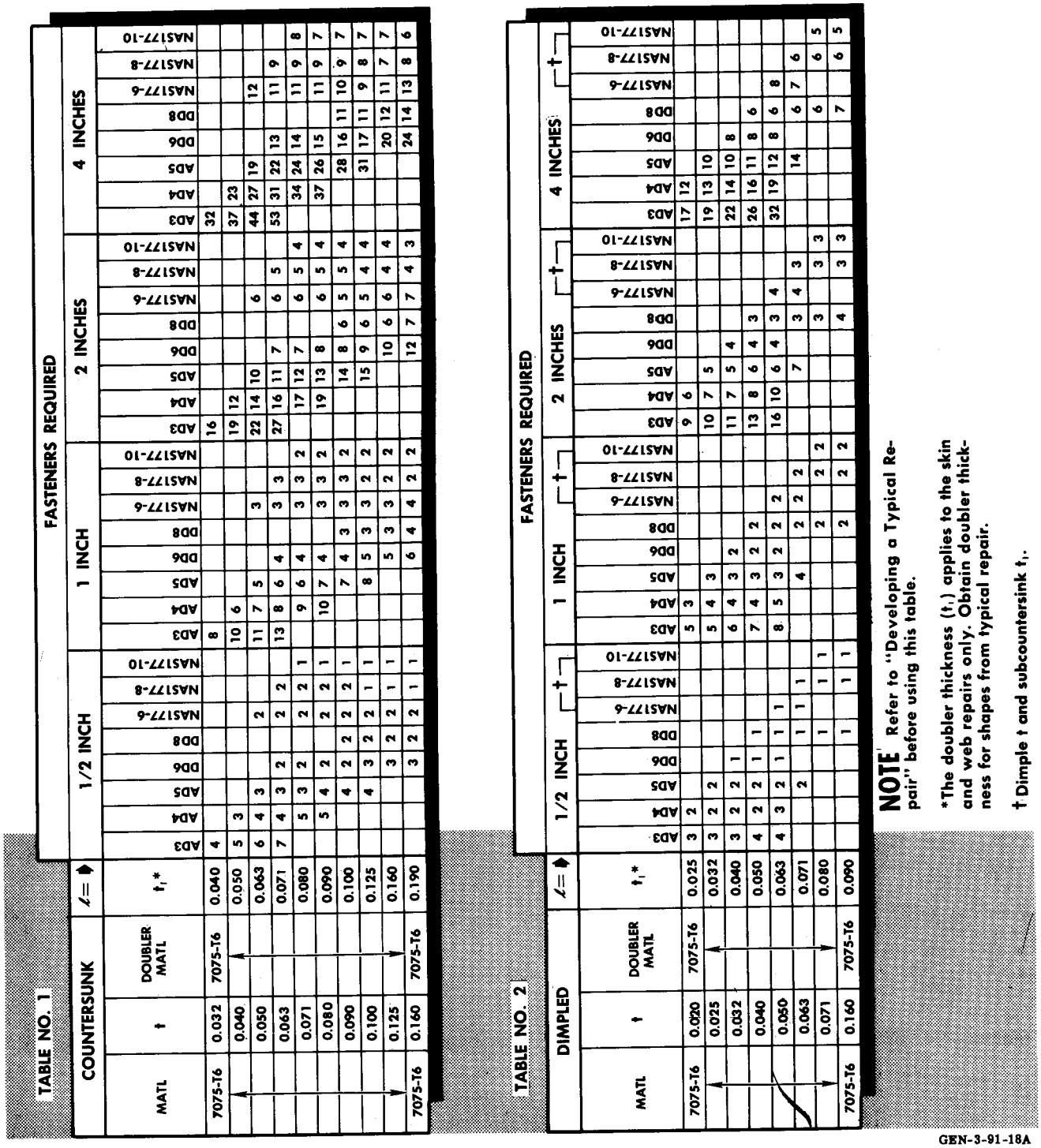


Figure 10-6. Fastener Tables—Conventional and Hi-Shear, Flush, 7075-T6, Shear

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

SINGLE SHEAR				FASTENERS REQUIRED																				
			ℓ=	1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	DOUBLER MATL	t_1*	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
2024-T4	0.020	2024-T4	0.025	4					7					14					27				
↑	0.025	↑	0.032	4	3				7	5				14	10				27	20			
	0.032		0.040	5	3	2			9	5	4			17	10	8			33	20	16		
	0.040		0.050	6	3	2	2		11	6	4	4		21	12	8	7		42	24	16	14	
	0.050		0.063	7	4	3	2	2	14	8	5	4	3	27	15	10	7	5	53	30	20	14	10
	0.063		0.071		5	4	2	2		10	7	4	3		20	13	7	6		39	26	14	10
	0.071		0.080		6	4	2	2		11	8	4	3		22	15	8	6		44	29	15	10
	0.080		0.090			5	3	2			9	5	3			16	9	6			32	17	10
↓	0.090	↓	0.100				3	2				5	3				10	6				19	11
2024-T4	0.100	2024-T4	0.125					2					3					6					12
DOUBLE SHEAR																							
2024-T4	0.032	2024-T4	0.025	4					7					14					27				
↑	0.040	↑	0.032	4	3				7	5				14	10				27	20			
	0.050		0.040	4	3	2			7	5	4			14	10	8			28	20	16		
	0.063		0.050	5	3	3			9	6	5			18	11	9			35	21	17		
	0.071		0.063		3	3	2			6	5	4			12	9	7			23	17	14	
	0.080		0.071		4	3	2			7	5	4			13	9	7			25	17	14	
	0.090		0.080			3	2	2			5	4	3			10	7	6			19	14	10
	0.100		0.090			3	2	2			6	4	3			11	7	6			21	14	10
↓	0.125	↓	0.100				2	2				4	3				7	6				14	10
2024-T4	0.160	2024-T4	0.125				2	2				4	3				8	6				16	10

TABLE NO. 2

SINGLE SHEAR				FASTENERS REQUIRED																				
			ℓ=	1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	DOUBLER MATL	t_1*	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
7075-T6	0.020	7075-T6	0.025	4					8					15					29				
↑	0.025	↑	0.032	5	3				9	6				18	11				35	21			
	0.032		0.040	6	4	3			11	7	5			22	13	9			43	25	17		
	0.040		0.050	7	4	3	2		14	8	6	3		28	16	11	6		55	31	21	11	
	0.050		0.063	9	5	4	2	2	18	10	7	4	3	35	20	13	7	5	70	39	26	14	9
	0.063		0.071		7	4	2	2		13	8	4	3		25	16	8	5		49	32	16	10
	0.071		0.080		7	5	3	2		14	9	5	3		28	18	9	6		55	36	19	11
	0.080		0.090			6	3	2			10	6	3			21	11	6			41	21	12
↓	0.090	↓	0.100					2					4					7					13
7075-T6	0.100	7075-T6	0.125					2					4					8					15
DOUBLE SHEAR																							
7075-T6	0.025	7075-T6	0.020	3	2				6	4				12	8				24	16			
↑	0.032	↑	0.025	4	3	2			7	5	4			14	9	7			27	18	13		
	0.040		0.032	4	3	2			8	5	4			16	10	8			31	20	15		
	0.050		0.040	5	3	2	2		10	6	4	3		19	12	8	6		37	23	16	11	
	0.063		0.050	6	4	3	2		11	7	5	3		22	14	9	6		44	27	18	11	
	0.071		0.063		4	3	2	2		8	5	3	3		15	10	6	5		29	20	11	9
	0.080		0.071		4	3	2	2		8	6	3	3		16	11	6	5		31	22	12	9
	0.090		0.080		5	3	2	2		9	6	4	3		18	12	7	5		35	24	13	9
	0.100		0.090			4	2	2			7	4	3			13	7	5			26	14	9
↓	0.125	↓	0.100			4	2	2			8	4	3			16	8	5			31	16	10
7075-T6	0.160	7075-T6	0.125					2					3					6					11

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-19A

Figure 10-7. Fastener Tables—Conventional, Protruding-head, 2024-T4 and 7075-T6, Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

SINGLE SHEAR				FASTENERS REQUIRED																				
			ℓ =	1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	DOUBLER MATL	t_1*	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
2024-T4	0.020	2024-T4	0.025	2					4					8					16				
	0.025		0.032	3	2				5	3				9	6				17	12			
	0.032		0.040	3	2	2			5	3	3			10	6	5			20	12	10		
	0.040		0.050	4	2	2	2		7	4	3	3		13	7	5	5		26	14	10	8	
	0.050		0.063	4	3	2	2	1	8	5	3	3	2	16	9	6	5	3	32	18	12	8	6
	0.063		0.071		3	2	2	1		6	4	3	2		12	8	5	4		24	16	9	7
	0.071		0.080		4	3	2	1		7	5	3	2		14	9	5	4		27	18	9	7
	0.080		0.090			3	2	1			5	3	2			10	5	4			20	10	7
	0.090		0.100				2	1				3	2				6	4				12	7
2024-T4	0.100	2024-T4	0.125					1					2					4					7
DOUBLE SHEAR																							
2024-T4	0.032	2024-T4	0.025	2					4					8					16				
	0.040		0.032	2	2				4	3				8	6				16	12			
	0.050		0.040	3	2	2			5	3	3			9	6	5			17	12	10		
	0.063		0.050	3	2	2			6	4	3			11	7	5			22	13	10		
	0.071		0.063		2	2	2			4	3	3			7	5	5			14	10	9	
	0.080		0.071		2	2	2			4	3	3			8	6	5			15	11	9	
	0.090		0.080			2	2	1			3	3	2			6	5	4			12	9	7
	0.100		0.090			2	2	1			4	3	2			7	5	4			13	9	7
	0.125		0.100				2	1				3	2				5	4				9	7
2024-T4	0.160	2024-T4	0.125				2	1				3	2				5	4				10	7

TABLE NO. 2

SINGLE SHEAR				FASTENERS REQUIRED																				
			ℓ =	1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	DOUBLER MATL	t_1*	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
7075-T6	0.020	7075-T6	0.025	3					5					9					18				
	0.025		0.032	3	2				6	4				11	7				21	13			
	0.032		0.040	4	2	2			7	4	3			13	8	6			26	15	11		
	0.040		0.050	5	3	2	1		9	5	4	2		17	10	7	4		33	19	13	7	
	0.050		0.063	6	3	2	1	1	11	6	4	2	2	21	12	8	4	3	42	24	16	8	5
	0.063		0.071		4	3	2	1		8	5	3	2		15	10	5	3		29	19	10	6
	0.071		0.080		5	3	2	1		9	6	3	2		17	11	6	4		33	22	11	7
	0.080		0.090			3	2	1			6	4	2			12	7	4			24	13	7
	0.090		0.100					1					2					4					8
7075-T6	0.100	7075-T6	0.125					2					3					5					9
DOUBLE SHEAR																							
7075-T6	0.025	7075-T6	0.020	2	2				4	3				8	5				15	10			
	0.032		0.025	2	2	1			4	3	2			8	6	4			16	11	8		
	0.040		0.032	3	2	2			5	3	3			10	6	5			18	12	9		
	0.050		0.040	3	2	2	1		6	4	3	2		11	7	5	4		22	14	10	7	
	0.063		0.050	4	2	2	1		7	4	3	2		13	8	6	4		26	16	11	7	
	0.071		0.063		3	2	1	1		5	3	2	2		9	6	4	3		17	12	7	5
	0.080		0.071		3	2	1	1		5	4	2	2		10	7	4	3		19	13	7	5
	0.090		0.080		3	2	1	1		6	4	2	2		11	7	4	3		21	14	8	5
	0.100		0.090			2	1	1			4	2	2			8	4	3			15	8	5
	0.125		0.100			3	2	1			5	3	2			10	5	3			19	10	6
7075-T6	0.160	7075-T6	0.125					1					2					4					7

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-20A

Figure 10-8. Fastener Tables—Conventional, Protruding-head, 2024-T4 and 7075-T6, Shear

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

SINGLE SHEAR	ℓ=	FASTENERS REQUIRED																			
		1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
2024-T4	0.040	6	4	3	2		11	7	5	3		23	13	9	6		45	25	17	12	
	0.051	7	4	3	2	2	15	8	6	3	3	29	16	11	6	5	57	32	21	12	9
	0.062	9	5	4	2	2	18	10	7	4	3	35	20	13	7	5	69	39	25	13	9
	0.064		5	4	2	2		10	7	4	3		20	13	7	5		40	26	14	9
	0.078			4		2			8		3			16		5			32		9
	0.094					2					3					6					11
	0.100					2					3					6					12
2024-T4	0.125					2					4					8					15
DOUBLE SHEAR																					
2024-T4	0.040	4	3	2			7	5	4			13	9	7			25	17	14		
	0.051	4	3	2			8	5	4			15	10	7			30	19	14		
	0.062	5	3	2	2		9	6	4	3		18	11	8	6		35	21	15	12	
	0.064	5	3	2	2		9	6	4	3		18	11	8	6		36	22	15	12	
	0.078	6	4	3	2	2	11	7	5	3	3	22	13	9	6	5	44	25	17	12	9
	0.094		4	3	2	2		8	5	3	3		15	10	6	5		30	20	12	9
	0.100		4	3	2	2		8	6	3	3		16	11	6	5		31	21	12	9
	0.125		5	4	2	2		10	7	4	3		20	13	7	5		39	26	13	9
2024-T4	0.156			4	2	2			8	4	3			16	8	5			32	16	9

TABLE NO. 2

SINGLE SHEAR	ℓ=	FASTENERS REQUIRED																			
		1/2 INCH					1 INCH					2 INCHES					4 INCHES				
MATL	t	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8	AD3	AD4	AD5	DD6	DD8
7075-T6	0.040		5	3	2			9	6	4			17	12	7			34	23	13	
	0.051		6	4	2	2		11	7	4	3		22	14	8	5		43	28	15	10
	0.062		7	5	3	2		13	9	5	3		26	17	9	5		52	34	17	10
	0.064		7	5	3	2		14	9	5	3		27	18	9	5		53	35	18	10
	0.078			6	3	2			11	6	3			21	11	6			42	22	12
	0.094					2					4					8					15
7075-T6	0.100					2					4					8					16
DOUBLE SHEAR																					
7075-T6	0.040	5	3	2	2		9	6	4	4		17	11	8	7		33	21	16	13	
	0.051	5	4	3	2	2	10	7	5	4	3	20	13	9	7	5	40	25	18	13	10
	0.062	6	4	3	2	2	12	7	5	4	3	23	14	10	7	5	46	28	20	13	10
	0.064	6	4	3	2	2	12	8	5	4	3	24	15	10	7	5	48	29	20	13	10
	0.078		5	3	2	2		9	6	4	3		17	12	7	5		33	23	13	10
	0.094		5	4	2	2		10	7	4	3		20	13	7	5		39	26	14	10
	0.100		6	4	2	2		11	7	4	3		21	14	8	5		42	28	15	10
	0.125			5	3	2			9	5	3			17	9	6			34	17	11
	0.156			6	3	2			11	6	3			21	11	6			42	22	12
7075-T6	0.188					2					4					8					15

NOTE Refer to "Developing a Typical Repair" before using this table.

GEN-3-91-21A

Figure 10-9. Fastener Tables—Conventional, Protruding-head, Extrusions, Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

SINGLE SHEAR—BRAZIER HEAD

1/2 HARD CRES				FASTENERS REQUIRED											
			ℓ=	1/2 IN.			1 IN.			2 IN.			4 IN.		
MATL	t	DOUBLER MATL	t_1*	AT4	AT5	AT6	AT4	AT5	AT6	AT4	AT5	AT6	AT4	AT5	AT6
CRES	0.010	CRES	0.016	4	3	3	7	6	5	14	11	9	27	22	18
	0.016		0.020	4	3	3	7	6	5	14	11	9	27	22	18
	0.020		0.025	4	3	3	7	6	5	14	11	9	27	22	18
	0.025		0.032	4	3	3	7	6	5	14	11	9	27	22	18
	0.032		0.036	4	3	3	8	6	5	16	11	9	31	22	18
	0.036		0.040	5	3	3	9	6	5	17	12	9	34	23	18
	0.040		0.045	5	4	3	10	7	5	19	13	9	38	26	18
	0.045		0.050	6	4	3	11	7	5	21	14	10	42	28	20
	0.050		0.063	6	4	3	12	8	6	24	16	11	47	31	22
	0.063		0.071	8	5	4	15	10	7	30	19	14	59	38	27
	0.071		0.080		6	4		11	8		22	15		43	30
CRES	0.080	CRES	0.090		6	5		13	9		25	17		49	34

TABLE NO. 3

SINGLE SHEAR—FLUSH HEAD (COUNTERSUNK)

1/2 HARD CRES				FASTENERS REQUIRED					
			ℓ=	1/2	1 IN.		2 IN.	4 IN.	
MATL	t	DOUBLER MATL	t_1*	BF6	BF6		BF6	BF6	
CRES	0.063	CRES	0.071	6	11		22	44	
	0.071		0.080	6	12		23	46	
	0.080		0.090	7	13		25	49	
	0.090		0.100	7	14		27	53	
	0.100		0.125	8	15		29	57	
	0.125		0.160	9	17		34	67	
CRES	0.160	CRES	0.190	10	20		40	80	

TABLE NO. 2

DOUBLE SHEAR—BRAZIER HEAD

1/2 HARD CRES				FASTENERS REQUIRED											
			ℓ=	1/2 IN.			1 IN.			2 IN.			4 IN.		
MATL	t	DOUBLER MATL	t_1*	AT4	AT5	AT6	AT4	AT5	AT6	AT4	AT5	AT6	AT4	AT5	AT6
CRES	0.020	CRES	0.025	4			7			13			25		
	0.025		0.032	3	3		6	5		12	10		24	20	
	0.032		0.036	3	3	2	6	5	4	12	10	8	24	19	16
	0.036		0.040	3	3	2	6	5	4	12	10	8	24	19	16
	0.040		0.045	3	3	2	6	5	4	12	10	8	24	19	16
	0.045		0.050	4	3	2	7	5	4	13	10	8	25	19	16
	0.050		0.063	4	3	2	7	5	4	14	10	8	27	20	16
	0.063		0.071	4	3	3	8	6	5	16	11	9	31	22	17
	0.071		0.080	5	3	3	9	6	5	17	12	9	34	24	18
	0.080		0.090	5	4	3	10	7	5	19	13	10	38	26	19
CRES	0.090	CRES	0.100	6	4	3	11	7	6	21	14	11	42	28	21

TABLE NO. 4

SINGLE SHEAR—BRAZIER HEAD (BLIND MONEL RIVETS)

1/2 HARD CRES				FASTENERS REQUIRED											
			ℓ=	1/2 IN.			1 IN.			2 IN.			4 IN.		
MATL	t	DOUBLER MATL	t_1*	1/8	5/32	3/16	1/8	5/32	3/16	1/8	5/32	3/16	1/8	5/32	3/16
CRES	0.020	CRES	0.025	4			8			16			32		
	0.025		0.032	5	4		9	7		17	14		33	27	
	0.032		0.040	5	4	3	9	7	6	18	14	11	36	27	22
	0.040		0.050	6	4	3	11	8	6	21	15	12	41	30	23
	0.050		0.063	6	5	4	13	9	7	25	17	13	49	34	26
CRES	0.063	CRES	0.071		5	4		10	8		20	15		39	29

NOTE Refer to "Developing a Typical Repair" before using this table.

- "AT" designates NAS508M rivets;
- "BF" designates AN427M rivets.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-22A

Figure 10-10. Fastener Tables—Flush, Protruding-head and Blind, 1/2 Hard CRES, Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

COUNTERSUNK				RIVETS REQUIRED											
			ℓ=	1/2 INCH			1 INCH			2 INCHES			4 INCHES		
MATL	t	DOUBLER MATL	t_1*	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6
2024-T4	0.032	2024-T4	0.040												
↑	0.040	↑	0.050	7			13			25			50		
	0.050		0.063	6	6		12	11		24	21		48	42	
↓	0.063	↓	0.071	6	5	11	12	10	21	24	19	42	48	38	
2024-T4	0.071	2024-T4	0.080	6	5	5	12	10	10	24	19	19	48	38	38
DIMPLED															
2024-T4	0.025	2024-T4	0.032	4	3		8	7		15	12		29	25	
↑	0.032	↑	0.040	4	3	3	8	6	5	16	12	10	31	23	20
	0.040		0.050	5	3	3	9	6	5	17	12	10	34	23	19
	0.050		0.063	5	4	3	10	7	5	20	13	10	40	26	20
	0.063		0.071		4	3		8	6		16	12		32	24
↓	0.071	↓	0.080			4			7			13			26
2024-T4	0.080	2024-T4	0.090			4			7			14			28

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

TABLE NO. 2

COUNTERSUNK				RIVETS REQUIRED											
			ℓ=	1/2 INCH			1 INCH			2 INCHES			4 INCHES		
MATL	t	DOUBLER MATL	t_1*	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6	AF462-4	AF462-5	AF462-6
7075-T6	0.032	7075-T6	0.040												
↑	0.040	↑	0.050	9			17			33			66		
	0.050		0.063	8	7		15	14		30	28		60	55	
↓	0.063	↓	0.071	8	6	13	15	12	26	30	24	52	59	48	
7075-T6	0.071	7075-T6	0.080	8	6	6	15	12	12	30	24	24	60	47	48
DIMPLED															
7075-T6	0.025	7075-T6	0.032	5	4		10	8		19	16		38	32	
↑	0.032	↑	0.040	5	4	4	10	8	7	20	15	13	40	30	25
	0.040		0.050	6	4	4	12	8	7	23	16	13	45	31	25
	0.050		0.063	7	5	4	13	9	7	26	18	14	52	35	27
	0.063		0.071		5	4		10	8		20	15		40	30
	0.071		0.080			4			9			17			33
7075-T6	0.080	7075-T6	0.090			5			9			18			35

TABLE NO. 3

CR-562 MONEL RIVETS COUNTERSUNK				RIVETS REQUIRED											
			ℓ=	1/2 INCH			1 INCH			2 INCHES			4 INCHES		
MATL	t	DOUBLER MATL	t_1*	CR562-4	CR562-5	CR562-6	CR562-4	CR562-5	CR562-6	CR562-4	CR562-5	CR562-6	CR562-4	CR562-5	CR562-6
7075-T6	0.040	7075-T6	0.050	5			10			19			38		
↑	0.050	↑	0.063	5	4		10	8		20	16		39	31	
	0.063		0.071	5	4	4	10	8	7	20	16	13	40	32	26
	0.071		0.080	6	4	4	11	8	7	21	16	13	42	32	26
	0.080		0.090	6	5	4	11	9	7	22	17	14	44	33	27
	0.090		0.100	6	5	4	12	9	7	24	17	14	47	34	27
↓	0.100	↓	0.125	7	5	4	13	9	7	25	19	14	50	37	28
7075-T6	0.125	7075-T6	0.160	7	5	4	14	10	8	28	20	16	56	40	31

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Figure 10-11. Fastener Tables—Cherry, Flush, 2024-T4 and 7075-T6, Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

COUNTERSUNK MATL	t	DOUBLER MATL	t_1*	RIVETS REQUIRED: 1/2 INCH AF462-4	1/2 INCH AF462-5	1/2 INCH AF462-6	1 INCH AF462-4	1 INCH AF462-5	1 INCH AF462-6	2 INCHES AF462-4	2 INCHES AF462-5	2 INCHES AF462-6	4 INCHES AF462-4	4 INCHES AF462-5	4 INCHES AF462-6
2024-T4	0.040	2024-T4	0.050	4			8			15			30		
	0.050		0.063	4	4		7	7		14	13		28	25	
	0.063		0.071	4	3	7	8	6	13	15	12	26	29	23	51
2024-T4	0.071	2024-T4	0.080	4	3	3	8	6	6	15	12	12	29	23	23
DIMPLED															
2024-T4	0.025	2024-T4	0.032	3	2		5	4		9	8		18	15	
	0.032		0.040	3	2	2	5	4	3	10	7	6	19	14	12
	0.040		0.050	3	2	2	6	4	3	11	7	6	21	14	12
	0.050		0.063	3	2	2	6	4	4	12	8	7	24	16	12
	0.063		0.071		3	2		5	4		10	8		20	15
	0.071		0.080			2			4			8			16
2024-T4	0.080	2024-T4	0.090			3			5			9			18

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

TABLE NO. 2

COUNTERSUNK MATL	t	DOUBLER MATL	t_1*	RIVETS REQUIRED: 1/2 INCH AF462-4	1/2 INCH AF462-5	1/2 INCH AF462-6	1 INCH AF462-4	1 INCH AF462-5	1 INCH AF462-6	2 INCHES AF462-4	2 INCHES AF462-5	2 INCHES AF462-6	4 INCHES AF462-4	4 INCHES AF462-5	4 INCHES AF462-6
7075-T6	0.032	7075-T6	0.040												
	0.040		0.050	5			10			20			39		
	0.050		0.063	5	5		9	9		18	17		36	33	
	0.063		0.071	5	4	8	9	8	16	18	15	31	35	29	62
7075-T6	0.071	7075-T6	0.080	5	4	4	9	7	8	18	14	15	36	28	29
DIMPLED															
7075-T6	0.025	7075-T6	0.032	3	3		6	5		12	10		23	19	
	0.032		0.040	3	3	2	6	5	4	12	9	8	24	18	15
	0.040		0.050	4	3	2	7	5	4	14	10	8	27	19	15
	0.050		0.063	4	3	2	8	6	4	16	11	8	31	21	16
	0.063		0.071		3	3		6	5		12	9		24	18
	0.071		0.080			3			5			10			20
7075-T6	0.080	7075-T6	0.090			3			6			11			21

TABLE NO. 3

COUNTERSUNK MATL	t	DOUBLER MATL	t_1*	RIVETS REQUIRED: 1/2 INCH CR562-4	1/2 INCH CR562-5	1/2 INCH CR562-6	1 INCH CR562-4	1 INCH CR562-5	1 INCH CR562-6	2 INCHES CR562-4	2 INCHES CR562-5	2 INCHES CR562-6	4 INCHES CR562-4	4 INCHES CR562-5	4 INCHES CR562-6
7075-T6	0.040	7075-T6	0.050	3			6			12			22		
	0.050		0.063	3	3		6	5		12	9		24	18	
	0.063		0.071	3	3	2	6	5	4	12	10	8	24	19	15
	0.071		0.080	4	3	2	7	5	4	13	10	8	25	19	16
	0.080		0.090	4	3	2	7	5	4	14	10	8	27	20	16
	0.090		0.100	4	3	3	7	6	5	14	11	9	28	21	16
	0.100		0.125	4	3	3	8	6	5	15	11	9	30	22	17
7075-T6	0.125	7075-T6	0.160	5	3	3	9	6	5	17	12	10	33	24	19

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Figure 10-12. Fastener Tables—Cherry, Flush, 2024-T4 and 7075-T6, Shear

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

AMS 4901 TITANIUM SINGLE SHEAR

FASTENERS REQUIRED

AN. AND NAS SERIES BOLTS† — 1/2 INCH

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	2	1	1	1
	0.040		0.045	2	1	1	1
	0.045		0.050	2	1	1	1
	0.050		0.056	2	1	1	1
	0.056		0.063	2	1	1	1
	0.063		0.071	2	1	1	1
	0.071		0.080	2	1	1	1
	0.080		0.090		1	1	1
	0.090		0.100		1	1	1
	0.100		0.112		1	1	1
	0.112		0.125			1	1
	0.125		0.140			1	1
	0.140	AMS4901	0.160				1
AMS4901	0.160						

AN. AND NAS SERIES BOLTS† — 1 INCH

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	3	2	2	2
	0.040		0.045	3	2	2	2
	0.045		0.050	3	2	2	2
	0.050		0.056	3	2	2	2
	0.056		0.063	3	2	2	2
	0.063		0.071	3	2	2	2
	0.071		0.080	3	2	2	2
	0.080		0.090		2	2	2
	0.090		0.100		2	2	2
	0.100		0.112		2	2	2
	0.112		0.125			2	2
	0.125		0.140			2	2
	0.140	AMS4901	0.160				2
AMS4901	0.160						

AN. AND NAS SERIES BOLTS† — 2 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	6	4	4	3
	0.040		0.045	6	4	4	3
	0.045		0.050	6	4	4	3
	0.050		0.056	6	4	4	3
	0.056		0.063	6	4	4	3
	0.063		0.071	6	4	4	3
	0.071		0.080	6	4	4	3
	0.080		0.090		4	4	3
	0.090		0.100		4	4	3
	0.100		0.112		4	4	3
	0.112		0.125			4	3
	0.125		0.140			4	3
	0.140	AMS4901	0.160				3
AMS4901	0.160						

AN. AND NAS SERIES BOLTS† — 4 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	11	8	7	6
	0.040		0.045	11	8	7	6
	0.045		0.050	11	8	7	6
	0.050		0.056	11	8	7	6
	0.056		0.063	11	8	7	6
	0.063		0.071	11	8	7	6
	0.071		0.080	11	8	7	6
	0.080		0.090		8	7	6
	0.090		0.100		8	7	6
	0.100		0.112		8	7	6
	0.112		0.125			7	6
	0.125		0.140			7	6
	0.140	AMS4901	0.160				6
AMS4901	0.160						

DOUBLE SHEAR

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.080	AMS4901	0.090	2			
	0.090		0.100	2			
	0.100		0.112	2			
	0.112		0.125	2	1		
	0.125		0.140	2	1		
	0.140		0.160	2	1	1	
	0.160		0.180		1	1	1
AMS4901	0.180	AMS4901	0.190		1	1	1

AN. AND NAS SERIES BOLTS† — 1 INCH

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.080	AMS4901	0.090	3			
	0.090		0.100	3			
	0.100		0.112	3			
	0.112		0.125	3	2		
	0.125		0.140	3	2		
	0.140		0.160	3	2	2	
	0.160		0.180		2	2	2
AMS4901	0.180	AMS4901	0.190		2	2	2

AN. AND NAS SERIES BOLTS† — 2 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.080	AMS4901	0.090	6			
	0.090		0.100	6			
	0.100		0.112	6			
	0.112		0.125	6	4		
	0.125		0.140	6	4		
	0.140		0.160	6	4	4	
	0.160		0.180		4	4	3
AMS4901	0.180	AMS4901	0.190		4	4	3

AN. AND NAS SERIES BOLTS† — 4 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.080	AMS4901	0.090	11			
	0.090		0.100	11			
	0.100		0.112	11			
	0.112		0.125	11	8		
	0.125		0.140	11	8		
	0.140		0.160	11	8	7	
	0.160		0.180		8	7	6
AMS4901	0.180	AMS4901	0.190		8	7	6

HI-SHEAR RIVETS‡ — 1/2 INCH

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	2	2	1	1
	0.040		0.045	2	2	1	1
	0.045		0.050	2	1	1	1
	0.050		0.056	2	1	1	1
	0.056		0.063	2	1	1	1
	0.063		0.071	2	1	1	1
	0.071		0.080	2	1	1	1
	0.080		0.090		1	1	1
	0.090		0.100		1	1	1
	0.100		0.112		1	1	1
	0.112		0.125			1	1
	0.125		0.140			1	1
	0.140	AMS4901	0.160				1
AMS4901	0.160						

HI-SHEAR RIVETS‡ — 1 INCH

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	3	3	2	2
	0.040		0.045	3	3	2	2
	0.045		0.050	3	2	2	2
	0.050		0.056	3	2	2	2
	0.056		0.063	3	2	2	2
	0.063		0.071	3	2	2	2
	0.071		0.080	3	2	2	2
	0.080		0.090		2	2	2
	0.090		0.100		2	2	2
	0.100		0.112		2	2	2
	0.112		0.125			2	2
	0.125		0.140			2	2
	0.140	AMS4901	0.160				2
AMS4901	0.160						

HI-SHEAR RIVETS‡ — 2 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	6	5	4	4
	0.040		0.045	6	5	4	4
	0.045		0.050	6	4	4	4
	0.050		0.056	6	4	4	4
	0.056		0.063	6	4	4	4
	0.063		0.071	6	4	4	4
	0.071		0.080	6	4	4	3
	0.080		0.090		4	4	3
	0.090		0.100		4	4	3
	0.100		0.112		4	4	3
	0.112		0.125			4	3
	0.125		0.140			4	3
	0.140	AMS4901	0.160				3
AMS4901	0.160						

HI-SHEAR RIVETS‡ — 4 INCHES

MATL	t	DOUBLER MATL	t_1*	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	11	10	8	7
	0.040		0.045	11	10	8	7
	0.045		0.050	11	8	8	7
	0.050		0.056	11	8	8	7
	0.056		0.063	11	8	8	7
	0.063		0.071	11	8	7	7
	0.071		0.080	11	8	7	7
	0.080		0.090	11	8	7	6
	0.090		0.100		9	7	6
	0.100		0.112		8	7	6
	0.112		0.125			7	6
	0.125		0.140			7	6
	0.140	AMS4901	0.160				6
AMS4901	0.160						

† 125,000 psi bolts (3/16 inch—AN3 and NAS221, 1/4 inch—AN3 and NAS222, 5/16 inch—AN5 and NAS223, 3/8 inch—AN6 and NAS224)

‡ 125,000 psi protruding head Hi-shear rivets (NAS177, NAS178, NAA2R3, NAA2R7, and NAA2R6)

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

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Figure 10-13. Fastener Tables—Hi-Shear Rivets and Bolts, Protruding-head, Titanium, Tension (Sheet 1 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 2

AMS4908 TITANIUM

FASTENERS REQUIRED

SINGLE SHEAR				NAS, MS & 5B—24 SERIES BOLTS †										HIGH-STRENGTH						HI-SHEAR RIVETS ‡					
				1/2 INCH			1 INCH			2 INCHES			4 INCHES	1/2 INCH			1 INCH			2 INCHES			4 INCHES	
MATL	t	DOUBLER	t, *	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8
AMS4908	0.036	AMS4908	0.040	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5	2	2	1	1	3	3	2	2	5	5	4	3	10	9	7	6
	0.040		0.045	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5	2	2	1	1	3	3	2	2	5	5	4	3	10	9	7	6
	0.045		0.050	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5	2	1	1	1	3	2	2	2	5	4	4	3	10	8	7	6
	0.050		0.063	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	6	2	1	1	1	3	2	2	2	5	4	4	3	10	8	7	6
	0.063		0.071	2	1	1	1	3	2	2	2	6	4	3	3	12	8	6	5	2	1	1	1	3	2	2	2	6	4	3	3	12	8	6	6
	0.071		0.080		1	1	1		2	2	2		4	3	3		8	6	5		1	1	1		2	2	2		4	3	3		8	6	5
	0.080		0.090			1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5
	0.090		0.100			1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5
	0.100		0.112				1				2				3				5				1			2				3				5
AMS4908	0.112	AMS4908	0.125				1				2				3				6				1			2				3				6
DOUBLE SHEAR																																			
AMS4908	0.071	AMS4908	0.080	2				3				5				10				2				3				5				10			
	0.080		0.090	2	1			3	2			5	4			10	8			2	1			3	2			5	4			10	8		
	0.090		0.100	2	1			3	2			5	4			10	8			2	1			3	2			5	4			10	8		
	0.100		0.112	2	1	1		3	2	2		5	4	3		10	8	6		2	1	1		3	2	2		5	4	3		10	8	6	
	0.112		0.125		1	1			2	2			4	3			8	6			1	1			2	2			4	3			8	6	
	0.125		0.140		1	1	1		2	2	2		4	3	3		8	6	5		1	1	1		2	2	2		4	3	3		8	6	5
	0.140		0.160		1	1	1		2	2	2		4	3	3		8	6	5		1	1	1		2	2	2		4	3	3		8	6	5
	0.160		0.180			1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5
AMS4908	0.180	AMS4908	0.190			1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5

NOTE Refer to "Developing a Typical Repair" before using this table.

- Use EWB18 Series bolts for 5B-24 (HT TR 180,000 psi).
- Use EB Series nuts for full tensile strength.
- Refer to index for source of commercial products.

* The doubler thickness (t,) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

† 160,000 psi bolts (1/4-inch—NAS144 and MS20004; 5/16-inch —NAS145 and MS20005; 3/8-inch—NAS146 and MS20006)

‡ 160,000 psi protruding-head Hi-Shear rivets (HS47, HS48, HS10, HS38P, HS39P, HS51P, and HS52P)

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Figure 10-13. Fastener Tables—Hi-Shear Rivets and Bolts, Protruding-head, Titanium, Tension (Sheet 2 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

AMS4901 TITANIUM SINGLE SHEAR			ℓ ⟹	AN. AND NAS SERIES BOLTS† 1/2 INCH				1 INCH				2 INCHES				4 INCHES				HI-SHEAR RIVETS‡ 1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER	t_1*	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8
AMS4901	0.036	AMS4901	0.040	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.040	↑	0.045	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.045	↑	0.050	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.050	↑	0.056	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.056	↑	0.063	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.063	↑	0.071	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	4
↑	0.071	↑	0.080	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3	1	1	1	1	2	2	1	1	3	3	2	2	6	5	4	3
↑	0.080	↑	0.090		1	1	1		2	1	1		3	2	2		5	4	3		1	1	1		2	1	1		3	2	2		5	4	3
↑	0.090	↑	0.100		1	1	1		2	1	1		3	2	2		5	4	3		1	1	1		2	1	1		3	2	2		5	4	3
↑	0.100	↑	0.112		1	1	1		2	1	1		3	2	2		5	4	3		1	1	1		2	1	1		3	2	2		5	4	3
↑	0.112	↑	0.125			1	1			1	1		2	2			4	3			1	1			1	1		2	2			4	3	
↑	0.125	↑	0.140			1	1			1	1		2	2			4	3			1	1			1	1		2	2			4	3	
AMS4901	0.140	AMS4901	0.160				1				1			2				3					1				1			2				3
DOUBLE SHEAR																																			
AMS4901	0.080	AMS4901	0.090	1				2				3				6				1				2				3				6			
↑	0.090	↑	0.100	1				2				3				6				1				2				3				6			
↑	0.100	↑	0.112	1				2				3				6				1				2				3				6			
↑	0.112	↑	0.125	1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5		
↑	0.125	↑	0.140	1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5		
↑	0.140	↑	0.160	1	1	1		2	2	1		3	3	2		6	5	4		1	1	1		2	2	1		3	3	2		6	5	4	
↑	0.160	↑	0.180		1	1	1		2	1	1		3	2	2		5	4	3		1	1	1		2	1	1		3	2	2		5	4	3
AMS4901	0.180	AMS4901	0.190		1	1	1		2	1	1		3	2	2		5	4	3		1	1	1		2	1	1		3	2	2		5	4	3

† 125,000 psi bolts (3/16 inch—AN3 and NAS221, 1/4 inch—AN3 and NAS222, 5/16 inch—AN5 and NAS223, 3/8 inch—AN6 and NAS224)

‡ 125,000 psi protruding head Hi-shear rivets (NAS177, NAS178, NAA2R3, NAA2R7, and NAA2R6)

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-28A

Figure 10-14. Fastener Tables—Hi-Shear Rivets and Bolts, Protruding-head, Titanium, Shear (Sheet 1 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 2

AMS4908 TITANIUM SINGLE SHEAR				NAS, MS, & 5B-24 SERIES BOLTS†												HIGH-STRENGTH HI-SHEAR RIVETS‡											
			ℓ=	1/2 INCH				1 INCH				2 INCHES				4 INCHES				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER	t_1*	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8
AMS4908	0.036	AMS4908	0.040	1	1	1	1	2	2	2	2	4	3	3	2	7	6	5	4	1	1	1	1	2	2	2	2	4	4	3	3	7	7	5	5
←	0.040	←	0.045	1	1	1	1	2	2	2	2	4	3	3	2	7	6	5	4	1	1	1	1	2	2	2	2	4	4	3	3	7	7	5	5
	0.045		0.050	1	1	1	1	2	2	2	2	4	3	3	2	7	6	5	4	1	1	1	1	2	2	2	2	4	4	3	3	7	6	5	5
	0.050		0.063	1	1	1	1	2	2	2	2	4	3	3	2	7	6	5	4	1	1	1	1	2	2	2	2	4	4	3	3	7	6	5	5
	0.063		0.070	1	1	1	1	2	2	2	2	4	3	3	2	8	6	5	4	1	1	1	1	2	2	2	2	4	3	3	3	8	6	5	5
	0.071		0.080		1	1	1		2	2	2		3	3	2		6	5	4		1	1	1		2	2	2		3	3	2		6	5	4
	0.080		0.090			1	1			2	2			3	2			5	4			1	1			2	2			3	2			5	4
	0.090		0.100				1				2				2				4				1				2				2				4
	0.100		0.112																																
AMS4908	0.112	AMS4908	0.125				1				2				2				4				1				2				2				4
DOUBLE SHEAR																																			
AMS4908	0.071	AMS4908	0.080	1				2				4				7								2				4				7			
←	0.080	←	0.090	1	1			2	2			4	3			7	6			1	1			2	2			4	3			7	6		
	0.090		0.100	1	1			2	2			4	3			7	6			1	1			2	2			4	3			7	6		
	0.100		0.112	1	1	1		2	2	2		4	3	3		7	6	5		1	1	1		2	2	2		4	3	3		7	6	5	
	0.112		0.125		1	1			2	2			3	3			6	5			1	1			2	2			3	3			6	5	
	0.125		0.140		1	1	1		2	2	2		3	3	2		6	5	4		1	1	1		2	2	2		3	3	2		6	5	4
	0.140		0.160		1	1	1		2	2	2		3	3	2		6	5	4		1	1	1		2	2	2		3	3	2		6	5	4
	0.160		0.180		1	1			2	2			3	2			5	4			1	1			2	2			3	2			5	4	
AMS4908	0.180	AMS4908	0.190		1	1			2	2			3	2			5	4			1	1			2	2			3	2			5	4	

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

†160,000 PSI bolts (1/4 inch—NAS144 and MS20004, 5/16 inch—NAS145 and MS20005, 3/8 inch—NAS146 and MS20006)

‡160,000 psi protruding head Hi-shear rivets (HS47, HS48, HS10, HS38P, HS39P, HS51P, and HS52)

NOTE Refer to "Developing a Typical Repair" before using this table.

- Use EB Series nuts for full tensile strength.
- Use EWB18 Series bolts for 5B-24. (Heat-treat to 180,000 psi.)
- Refer to index for source of commercial products.

GEN-3-91-29A

Figure 10-14. Fastener Tables—Hi-Shear Rivets and Bolts, Protruding-head, Titanium, Shear (Sheet 2 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

TENSION			ℓ=	FASTENERS REQUIRED - PROTRUDING HEAD															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6
2024-T4	0.040	2024-T4	0.050	2				4				7				14			
	0.050		0.063	2	2			4	3			7	6			14	11		
	0.063		0.071	2	2	2		4	3	3		7	6	5		14	11	9	
	0.071		0.080	2	2	2	1	4	3	3	2	7	6	5	4	14	11	9	7
	0.080		0.090	2	2	2	1	4	3	3	2	7	6	5	4	14	11	9	7
	0.090		0.100	2	2	2	1	4	3	3	2	7	6	5	4	14	11	9	7
	0.100		0.112	2	2	2	1	4	3	3	2	7	6	5	4	14	11	9	7
	0.112		0.125	2	2	2	1	4	3	3	2	7	6	5	4	14	11	9	7
	0.125		0.140		2	2	1		3	3	2		6	5	4		11	9	7
	0.140		0.160			2	1			3	2			5	4			9	7
	0.160		0.180			2	1			3	2			5	4			9	7
2024-T4	0.180	2024-T4	0.190				1				2				4				7

TABLE NO. 2

SHEAR			ℓ=	FASTENERS REQUIRED - PROTRUDING HEAD															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6
2024-T4	0.040	2024-T4	0.050	2				3				5				9			
	0.050		0.063	2	1			3	2			5	4			9	7		
	0.063		0.071	2	1	1		3	2	2		5	4	3		9	7	6	
	0.071		0.080	2	1	1	1	3	2	2	2	5	4	3	3	9	7	6	5
	0.080		0.090	2	1	1	1	3	2	2	2	5	4	3	3	9	7	6	5
	0.090		0.100	2	1	1	1	3	2	2	2	5	4	3	3	9	7	6	5
	0.100		0.112	2	1	1	1	3	2	2	2	5	4	3	3	9	7	6	5
	0.112		0.125	2	1	1	1	3	2	2	2	5	4	3	3	9	7	6	5
	0.125		0.140		1	1	1		2	2	2		4	3	3		7	6	5
	0.140		0.160			1	1			2	2			3	3			6	5
	0.160		0.180			1	1			2	2			3	3			6	5
2024-T4	0.180	2024-T4	0.190				1				2				3				5

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-30A

Figure 10-15. Fastener Tables—Hi-Shear and AN Bolts, Protruding-head, Shear and Tension (Sheet 1 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 3

TENSION			ℓ⇒	FASTENERS REQUIRED — PROTRUDING HEAD															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6
7075-T6	0.040	7075-T6	0.045	2				3				6				12			
↑	0.045	↑	0.050	2				3				6				12			
	0.050		0.056	2	2			3	3			6	5			12	9		
	0.056		0.063	2	2			3	3			6	5			12	9		
	0.063		0.071	2	2	1		3	3	2		6	5	4		12	9	7	
	0.071		0.080	2	2	1	1	3	3	2	2	6	5	4	3	12	9	7	6
	0.080		0.090		2	1	1		3	2	2		5	4	3		9	7	6
	0.090		0.100		2	1	1		3	2	2		5	4	3		9	7	6
	0.100		0.112		2	1	1		3	2	2		5	4	3		9	7	6
	0.112		0.125			1	1			2	2			4	3			7	6
↓	0.125	↓	0.140			1	1			2	2			4	3			7	6
7075-T6	0.140	7075-T6	0.160				1				2				3				6

TABLE NO. 4

SHEAR			ℓ⇒	FASTENERS REQUIRED — PROTRUDING HEAD															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6	NAS178-6 AN3	NAS178-8 AN4	NAS178-10 AN5	NAS178-12 AN6
7075-T6	0.040	7075-T6	0.045	1				2				4				7			
↑	0.045	↑	0.050	1				2				4				7			
	0.050		0.056	1	1			2	2			4	3			7	5		
	0.056		0.063	1	1			2	2			4	3			7	5		
	0.063		0.071	1	1	1		2	2	1		4	3	2		7	5	4	
	0.071		0.080	1	1	1	1	2	2	1	1	4	3	2	2	7	5	4	4
	0.080		0.090		1	1	1		2	1	1		3	2	2		5	4	4
	0.090		0.100		1	1	1		2	1	1		3	2	2		5	4	4
	0.100		0.112		1	1	1		2	1	1		3	2	2		5	4	4
	0.112		0.125			1	1			1	1			2	2			4	4
↓	0.125	↓	0.140			1	1			1	1			2	2			4	4
7075-T6	0.140	7075-T6	0.160				1				1				2				4

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-31A

Figure 10-15. Fastener Tables—Hi-Shear and AN Bolts, Protruding-head, Shear and Tension (Sheet 2 of 2)

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

TENSION			ℓ=	FASTENERS REQUIRED—HUCK LOCK BOLTS															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8
7075-T6	0.040	7075-T6	0.050	2				4				8				15			
	0.050		0.063	2	2		2	3	3		3	6	5		5	12	10		9
	0.063		0.071	2	2	1	2	4	3	2	3	7	5	4	5	13	10	8	9
	0.071		0.080	2	2	1	2	4	3	2	3	7	5	4	5	13	10	8	9
	0.080		0.090	2	2	1	2	4	3	2	3	7	5	4	5	14	10	8	10
	0.090		0.100	2	2	1	2	4	3	2	3	8	6	4	5	15	11	8	10
	0.100		0.125		2	2	2		3	3	3		6	5	5		11	9	10
	0.125		0.160			2	2			3	3			5	6			10	11
7075-T6	0.160	7075-T6	0.190				2				3				6				12

TABLE NO. 2

SHEAR			ℓ=	FASTENERS REQUIRED—HUCK LOCK BOLTS															
				1/2 INCH				1 INCH				2 INCHES				4 INCHES			
MATL	t	DOUBLER MATL	t_1*	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8	OS-8	OS-10	OS-12	BL-8
7075-T6	0.040	7075-T6	0.050	2				3				5				9			
	0.050		0.063	1	1		1	2	2		2	4	3		3	7	6		5
	0.063		0.071	1	1	1	1	2	2	2	2	4	3	3	3	8	6	5	6
	0.071		0.080	1	1	1	1	2	2	2	2	4	3	3	3	8	6	5	6
	0.080		0.090	2	1	1	1	3	2	2	2	5	3	3	3	9	6	5	6
	0.090		0.100	2	1	1	1	3	2	2	2	5	4	3	3	9	7	5	6
	0.100		0.125		1	1	1		2	2	2		4	3	3		7	5	6
	0.125		0.160			1	1			2	2			3	4			6	7
7075-T6	0.160	7075-T6	0.190				1				2				4				7

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-32A

Figure 10-16. Fastener Tables—Huck Lock Bolt, 7075-T6, Shear and Tension

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

| MATL | t | DOUBLER MATL | ∠= ▶ | FASTENERS REQUIRED | | |

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

MATL	t	DOUBLER MATL	t_1*	FASTENERS REQUIRED																											
				PROTRUDING				COUNTERSUNK																		DIMPLED						
				1/2	1	2	4	1/2 INCH				1 INCH				2 INCHES				4 INCHES				1/2	1	2	4				
			ℓ = ⬍	F164	F164	F164	F164	F164	F200	F260	F312	F164	F200	F260	F312	F164	F200	F260	F312	F164	F200	F260	F312	F164	F164	F164	F164				
2024-T4	0.025	2024-T4	0.032	2	3	6	11																	2	3	6	12				
	0.032		0.040	2	3	6	11																	2	3	6	12				
	0.040		0.050	2	3	6	11																	2	3	6	12				
	0.050		0.063	2	3	6	11	2				3				6				12				2	3	6	12				
	0.063		0.071	2	3	6	12	2	2			4	3			7	6			13	11			2	4	7	13				
	0.071		0.080	2	3	6	12	2	2			4	3			7	6			13	12			2	4	7	13				
	0.080		0.090	2	3	6	12	2	2			4	3			7	6			13	11										
	0.090		0.100	2	3	6	12	2	2	2		4	3	3		7	6	5		13	11	9									
	0.100		0.125	2	3	6	12	2	2	2	1	4	3	3	2	7	6	5	4	13	11	9	8								
	0.125		0.160	2	3	6	12	2	2	2	1	4	3	3	2	7	6	5	4	13	11	9	8								
	0.160		0.190	2	4	7	14	2	2	1	1	4	3	2	2	7	6	4	4	14	11	8	8								
	0.190		0.250	3	5	9	17	3	2	1	1	5	3	2	2	9	6	4	4	17	12	8	8								
	0.250		0.375						2	2	1		4	3	3		8	5	5		16	10	9								
2024-T4	0.375	2024-T4	0.500							2	2			4	3			7	6			14	11								

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

GEN-3-91-34A

Figure 10-18. Fastener Tables—Jo-bolt, Flush and Protruding-head, 2024-T4, Shear

FASTENER TABLES

TYPICAL REPAIR

TABLE NO. 1

FASTENERS REQUIRED — PROTRUDING HEAD

TENSION MATL	TENSION t	DOUBLER MATL	ℓ= t1*	1/2 INCH P164	1/2 INCH P200	1/2 INCH P260	1/2 INCH P312	1 INCH P164	1 INCH P200	1 INCH P260	1 INCH P312	2 INCHES P164	2 INCHES P200	2 INCHES P260	2 INCHES P312	4 INCHES P164	4 INCHES P200	4 INCHES P260	4 INCHES P312
7075-T6	0.025	7075-T6	0.032	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.032		0.040	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.040		0.050	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.050		0.063	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.063		0.071	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.071		0.080	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.080		0.090	2	2	2	2	4	4	3	3	8	7	5	5	16	13	10	9
	0.090		0.100	3	2	2	2	5	4	3	3	9	7	5	5	17	13	10	9
	0.100		0.125	3	2	2	2	5	4	3	2	9	7	5	4	18	13	10	8
	0.125		0.160	3	2	2	2	6	4	3	3	11	8	5	5	22	15	10	9
	0.160		0.190		3	2	2		5	3	3		10	6	5		19	12	9
	0.190		0.250		3	2	2		6	4	3		11	7	5		22	14	10
	0.250		0.375			3	2			5	3			9	6			17	12
7075-T6	0.375	7075-T6	0.500			3	3			6	5			12	9			24	17

TABLE NO. 2

FASTENERS REQUIRED — COUNTERSUNK

FASTENERS REQUIRED — DIMPLED

TENSION MATL	TENSION t	DOUBLER MATL	ℓ= t1*	1/2 INCH F164	1/2 INCH F200	1/2 INCH F260	1/2 INCH F312	1 INCH F164	1 INCH F200	1 INCH F260	1 INCH F312	2 INCHES F164	2 INCHES F200	2 INCHES F260	2 INCHES F312	4 INCHES F164	4 INCHES F200	4 INCHES F260	4 INCHES F312
7075-T6	0.025	7075-T6	0.032									3	2	2	1	5	4	3	2
	0.032		0.040									3	2	2	1	5	4	3	2
	0.040		0.050									3	3	2	2	6	5	3	3
	0.050		0.063	3				6				24							
	0.063		0.071	3	3			6	5			24	19						
	0.071		0.080	3	3			6	5			24	19						
	0.080		0.090	3	3			6	5			24	19						
	0.090		0.100	3	3	2		6	5	4		24	19	15					
	0.100		0.125	3	3	2	2	6	5	4	4	24	19	15	13				
	0.125		0.160	3	3	2	2	6	5	4	4	24	19	15	13				
	0.160		0.190	4	3	2	2	8	5	4	4	29	20	15	13				
	0.190		0.250	5	3	2	2	9	6	4	4	34	23	15	13				
	0.250		0.375	6	4	3	2	11	8	5	4	33	29	18	13				
7075-T6	0.375	7075-T6	0.500	9	6	3	3	17	11	6	5	33	22	12	9	66	43	24	18

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

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Figure 10-19. Fastener Tables—Jo-bolt, Flush and Protruding-head, 7075-T6, Tension

FASTENER TABLES

TYPICAL REPAIR

NOTE Refer to "Developing a Typical Repair" before using this table.

*The doubler thickness (t_1) applies to the skin and web repairs only. Obtain doubler thickness for shapes from typical repair.

TABLE NO. 1

SHEAR MATL	SHEAR t	DOUBLER MATL	ℓ =	FASTENERS REQUIRED — PROTRUDING HEAD: 1/2 INCH t_1*	P164	P200	P260	P312	1 INCH P164	P200	P260	P312	2 INCHES P164	P200	P260	P312	4 INCHES P164	P200	P260	P312
7075-T6	0.025	7075-T6		0.032	2	1	1		3	2	2		5	4	3		10	8	6	
	0.032			0.040	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.040			0.050	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.050			0.063	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.063			0.071	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.071			0.080	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.080			0.090	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.090			0.100	2	1	1	1	3	2	2	2	5	4	3	3	10	8	6	5
	0.100			0.125	2	1	1	1	3	2	2	2	6	4	3	3	11	8	6	5
	0.125			0.160	2	2	1	1	4	3	2	2	7	5	3	3	13	9	6	5
	0.160			0.190		2	1	1		3	2	2		6	4	3		12	7	5
	0.190			0.250		2	1	1		4	2	2		7	4	3		13	8	6
	0.250			0.375			2	1			3	2			5	4			10	8
7075-T6	0.375	7075-T6		0.500			2	2			4	3			8	5			15	10

TABLE NO. 2

SHEAR MATL	SHEAR t	DOUBLER MATL	ℓ =	FASTENERS REQUIRED — COUNTERSUNK: 1/2 INCH t_1*	F164	F200	F260	F312	1 INCH F164	F200	F260	F312	2 INCHES F164	F200	F260	F312	4 INCHES F164	F200	F260	F312
7075-T6	0.025	7075-T6		0.032																
	0.032			0.040																
	0.040			0.050																
	0.050			0.063	2				4				7				14			
	0.063			0.071	2	2			4	3			7	6			14	12		
	0.071			0.080	2	2			4	3			7	6			14	12		
	0.080			0.090	2	2			4	3			7	6			14	12		
	0.090			0.100	2	2	2		4	3	3		7	6	5		14	12	9	
	0.100			0.125	2	2	2	1	4	3	3	2	7	6	5	4	14	12	9	8
	0.125			0.160	2	2	2	1	4	3	3	2	7	6	5	4	14	12	9	8
	0.160			0.190	3	2	2	1	5	3	3	2	9	6	5	4	17	12	9	8
	0.190			0.250	3	2	2	1	5	4	3	2	10	7	5	4	20	14	9	8
	0.250			0.375		3	2	1		5	3	2		9	6	4		17	11	8
7075-T6	0.375	7075-T6		0.500			2	2			4	3			8	6			15	11

FASTENERS REQUIRED — DIMPLED	1/2 INCH F164	F200	F260	F312	1 INCH F164	F200	F260	F312	2 INCHES F164	F200	F260	F312	4 INCHES F164*	F200*	F260	F312
	2	1	1		3	2	1		6	4	3		11	8	6	5
	2	2	1	1	3	2	2	2	6	5	3	3	12	9	6	5
	2	2	1	1	4	3	2	2	7	5	4	3	13	10	7	6
	2	2	1	1	4	3	2	2	7	6	4	3	14	11	8	6
	2	2	1	1	4	3	2	2	8	6	4	4	15	11	8	7
	2		2	1	4		3	2	8		5	4	15		9	7
			2	1			3	2			5	4			9	7
			2	1			3	2			5	4			9	7

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Figure 10-20. Fastener Tables—Jo-bolt, Flush and Protruding-head, 7075-T6, Shear

RIVET HOLE SIZES

TYPICAL REPAIR

HOLE SIZE												
	BLIND FASTENERS									DRIVE PIN RIVET (DEUTSCH)		
		FINAL HOLE SIZE										
			PULL									
				CHERRY & OLYMPIC		HUCK						
FASTENER SIZE	CONVENTIONAL RIVET*		EXPLOSIVE (DUPONT)	STANDARD RIVET	OVERSIZE RIVET	BL SERIES BLIND LOCK BOLT	OS & OSR SERIES RIVET	JO-BOLT		PILOT-DRILL BEFORE DIMPLING	FINAL HOLE SIZE	
3/32	0.097-0.100 (NO. 40)											
1/8	0.1290-0.1305 (NO. 30)		0.135-0.139 (NO. 29)	0.128-0.132 (NO. 30)	0.137-0.141 (NO. 29)					0.097-0.100 (NO. 40)	0.1255-0.1265	
5/32	0.160-0.162 (NO. 21)		0.172-0.176 (NO. 17)	0.160-0.164 (NO. 20)	0.177-0.181 (NO. 16)			0.164-0.166		0.128-0.132 (NO. 30)	0.1565-0.1575	
3/16	0.192-0.194 (NO. 11)		0.203-0.207 (NO. 6)	0.192-0.196 (NO. 10)	0.206-0.210 (NO. 5)			0.199-0.202		0.172-0.176 (NO. 17)	0.191-0.193	
1/4	0.256-0.261 (F)			0.256-0.261 (F)		0.260-0.265 (G)	0.271-0.276 (I)	0.260-0.263		0.227-0.231 (NO. 1)	0.259-0.262	
3/8								0.312-0.315		0.2802-0.2852 (9/32)	0.315-0.318	
								0.375-0.379			0.385-0.388	

*Holes after dimpling must not exceed the nominal drill size by more than 15 percent.

Holes to be dimpled for explosive and pull fasteners must be pilot-drilled using nominal drill size given for conventional rivets, dimpled, and redrilled to size.

If the hole size exceeds the limits given for the standard rivet but is within the limits given for the oversize rivet, use the oversize rivet.

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Figure 10-21. Rivet Hole Sizes

CONVENTIONAL RIVET TYPES AND APPLICATIONS

TYPICAL REPAIR

	RIVET MATERIAL	AL ALLOY 2117-T4 (1)	AL ALLOY 2024-T4 (2)	AL ALLOY 5056-F (1)	AL 1100-F (1) (3)	MILD STEEL CAD-PLATED (1) (5)	MONEL (1)
IDENTIFICATION	HEAD MARKINGS	DIMPLE	RAISED DOUBLE DASH	RAISED CROSS	PLAIN	RECESSED TRIANGLE	DIMPLES / PLAIN
	UNIVERSAL HEAD	AN470AD	AN470DD	AN470B	AN470A		NAS508M
	ROUND HEAD					AN435	
	FLUSH HEAD	AN426AD	AN426DD	AN426B	AN426A	AN427	AN427M
	MAXIMUM TEMPERATURE	200°F	200°F	200°F		550°F	1000°F
	SIZES	3/32, 1/8, & 5/32 DIA	3/16 DIA & OVER			5/16 DIA MAX	3/16 DIA MAX
APPLICATION	AL ALLOY TO AL ALLOY	PREFERRED	PREFERRED	PERMISSIBLE BUT UNDESIRABLE	(3)	PROHIBITED	PROHIBITED
	AL ALLOY TO CAD PLATED STEEL AL ALLOY TO CRE STEEL AL ALLOY TO TITANIUM	(4)	(4)	PROHIBITED	PROHIBITED	PREFERRED (5)	PERMISSIBLE BUT UNDESIRABLE
	CAD PLATED STEEL TO CAD PLATED STEEL CAD PLATED STEEL TO CRE STEEL CAD PLATED STEEL TO TITANIUM	PERMISSIBLE BUT UNDESIRABLE	PERMISSIBLE BUT UNDESIRABLE				
	CRE STEEL TO CRE STEEL CRE STEEL TO TITANIUM TITANIUM TO TITANIUM	PROHIBITED	PROHIBITED	PROHIBITED	PROHIBITED	PROHIBITED	PREFERRED
	MAGNESIUM ALLOY TO AL ALLOY MAGNESIUM ALLOY TO CAD PLATED STEEL MAGNESIUM ALLOY TO MAGNESIUM ALLOY	PROHIBITED	PROHIBITED	PREFERRED	PROHIBITED	PROHIBITED	PROHIBITED
	SOFT MATERIAL	ACCEPTABLE	PROHIBITED	PERMISSIBLE BUT UNDESIRABLE	PREFERRED	PROHIBITED	PROHIBITED

(1) Use rivets in "as received" condition.

(2) Rivets should be reheat-treated and refrigerated before using, in accordance with Specification MIL-H-6088.

(3) May be used for welding, where extra softness is required, or for plugging holes.

(4) May be used for weight saving or strength.

(5) Not recommended for structural applications.

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Figure 10-22. Conventional-type Rivets and Applications

ALUMINUM HONEYCOMB REPAIRS

TYPICAL

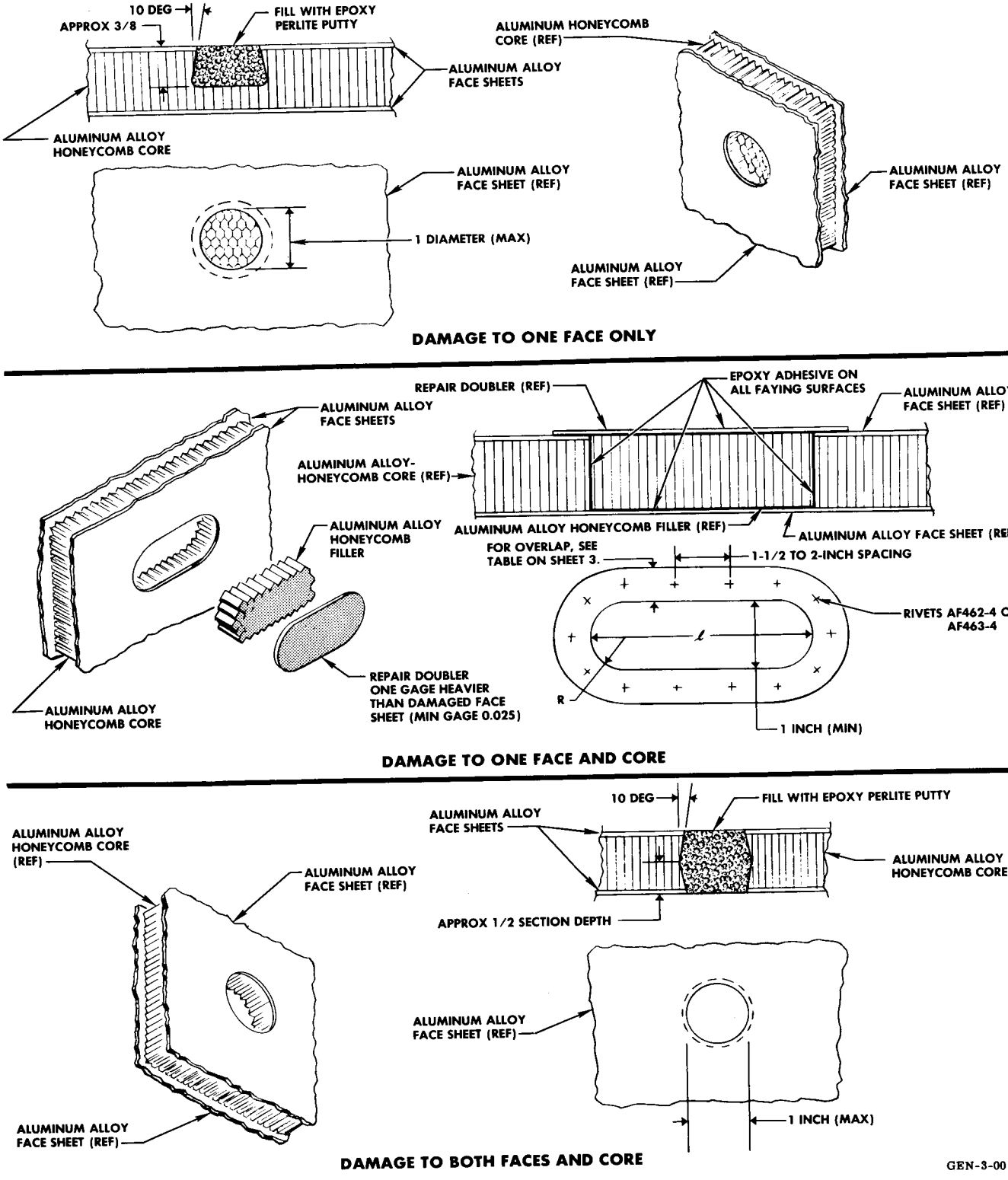


Figure 10-22A. Aluminum Honeycomb Repairs (Sheet 1 of 3)

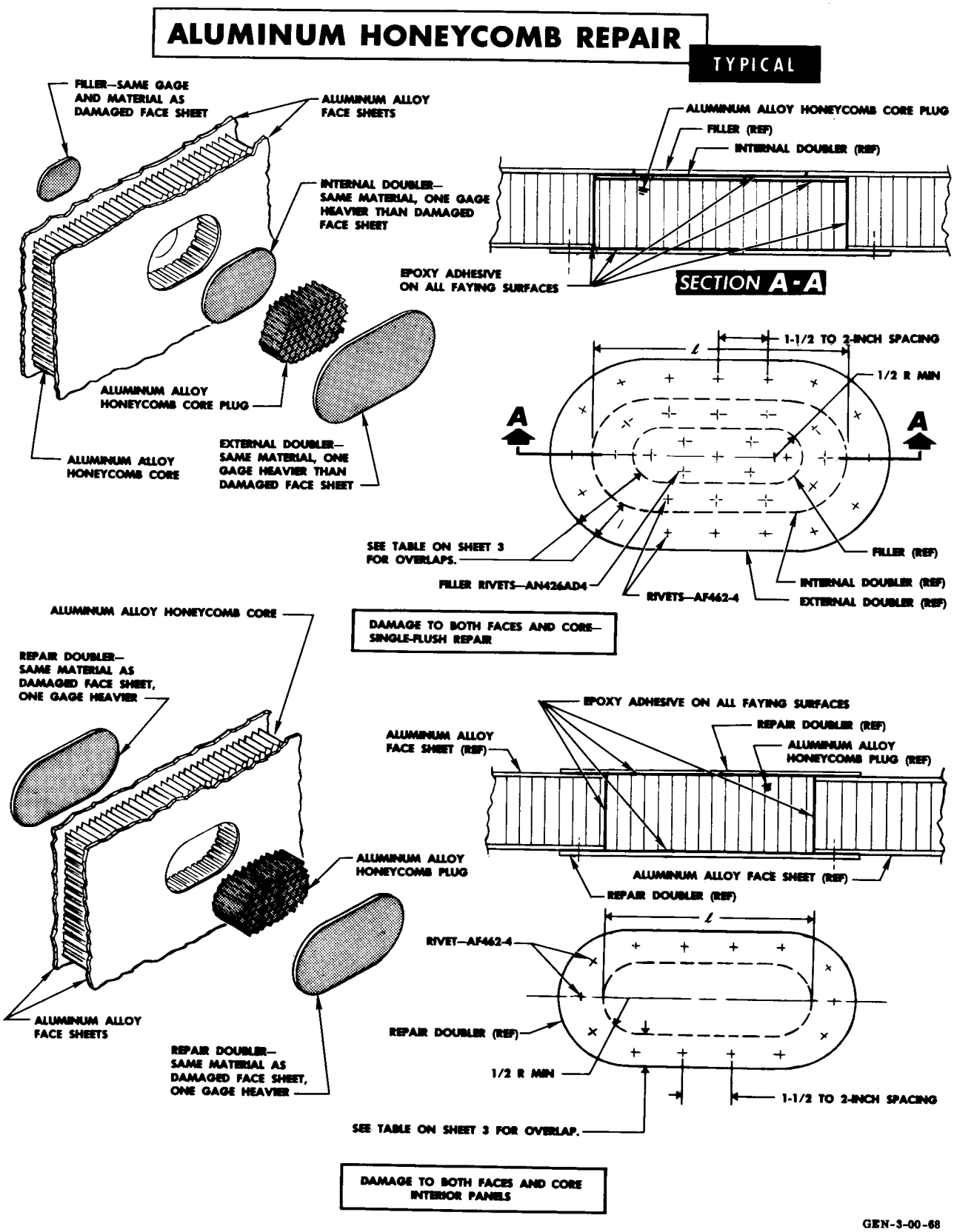
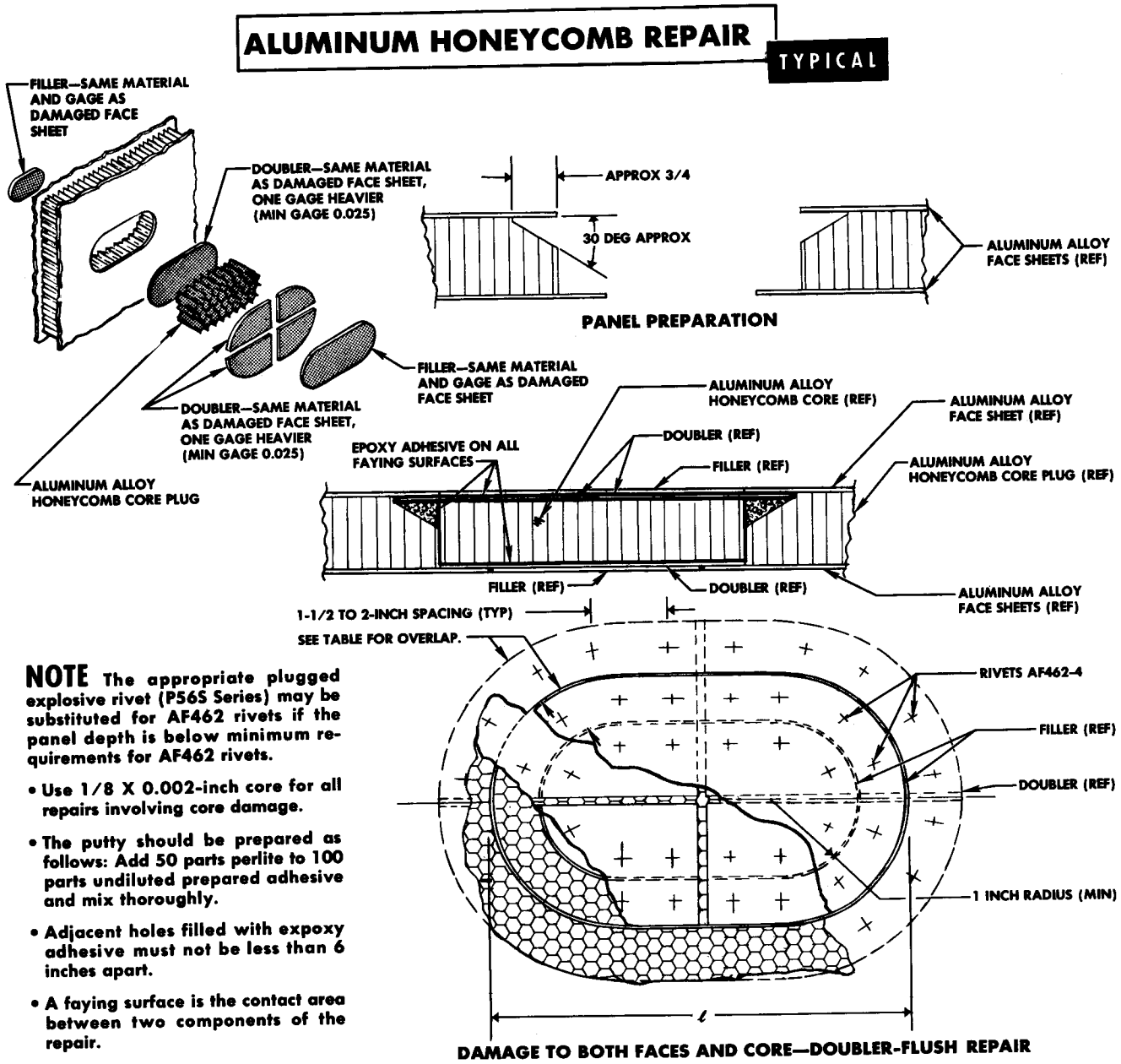


Figure 10-22A. Aluminum Honeycomb Repairs (Sheet 2 of 3)



NOTE The appropriate plugged explosive rivet (P56S Series) may be substituted for AF462 rivets if the panel depth is below minimum requirements for AF462 rivets.

- Use 1/8 X 0.002-inch core for all repairs involving core damage.
- The putty should be prepared as follows: Add 50 parts perlite to 100 parts undiluted prepared adhesive and mix thoroughly.
- Adjacent holes filled with expoxy adhesive must not be less than 6 inches apart.
- A faying surface is the contact area between two components of the repair.
- Maximum skin cutout must not exceed 8 inches.

MINIMUM DOUBLER OVERLAP FOR 7075-T6 CLAD FACE SHEETS

SKIN \ ℓ	2 IN.	4 IN.	6 IN.	8 IN.
0.010	1/2	19/32	5/8	21/32
0.016	23/32	27/32	29/32	31/32
0.020	13/16	1	1-1/8	1-5/32
0.025	31/32	1-7/32	1-5/16	1-3/8
0.032	1-1/8	1-13/32	1-9/16	1-11/32
0.040	1-5/16	1-21/32	1-7/8	2
0.050	2	2-1/2	2-3/4	3
0.063	2-1/4	2-3/4	3	3-1/4

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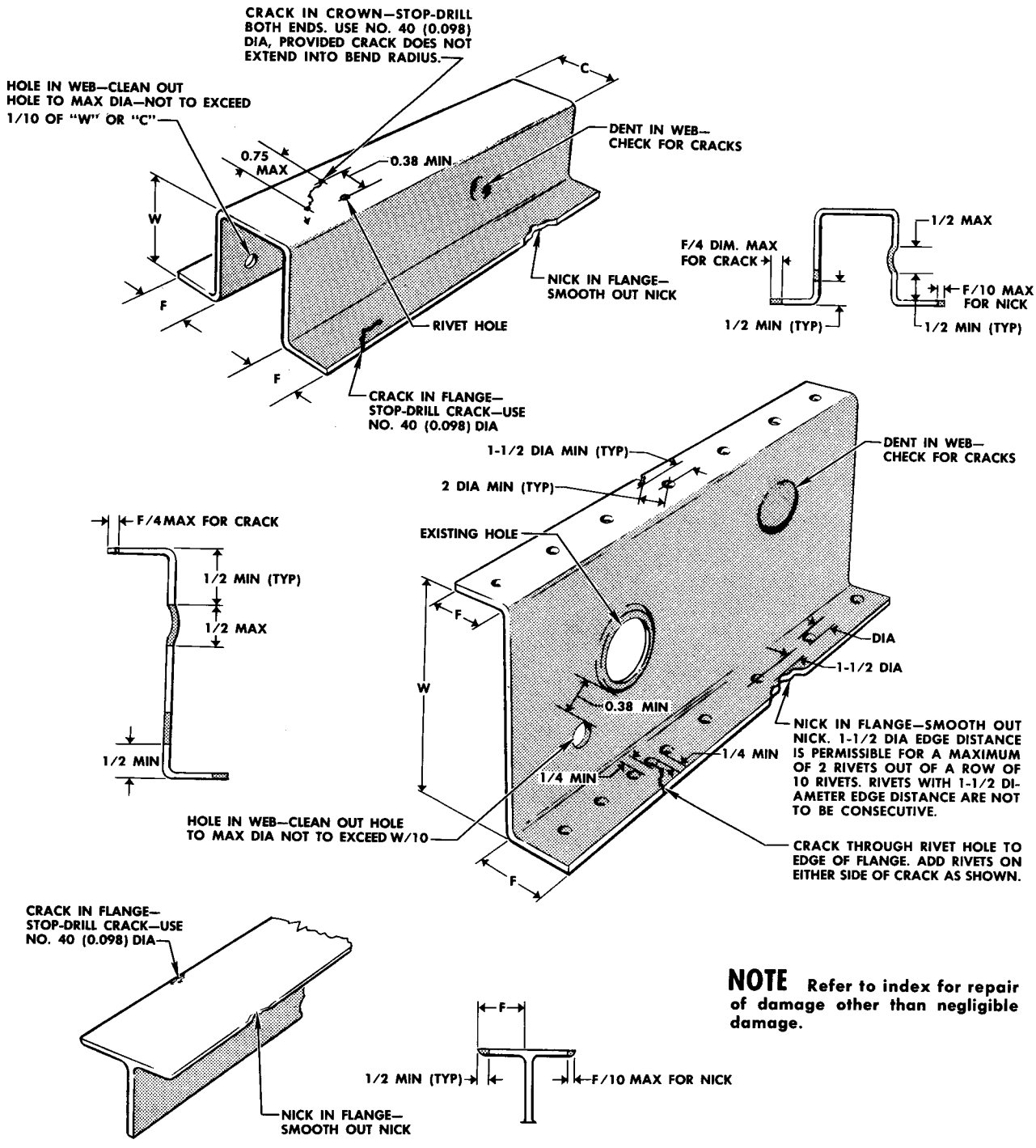
Figure 10-22A. Aluminum Honeycomb Repairs (Sheet 3 of 3)

NEGLIGIBLE DAMAGE SHAPES AND EXTRUSIONS

TYPICAL REPAIR



NOTE Refer to index for repair of damage other than negligible damage.

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Figure 10-23. Negligible Damage—Sheet-metal Shapes and Extrusions

SKIN REPAIR - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

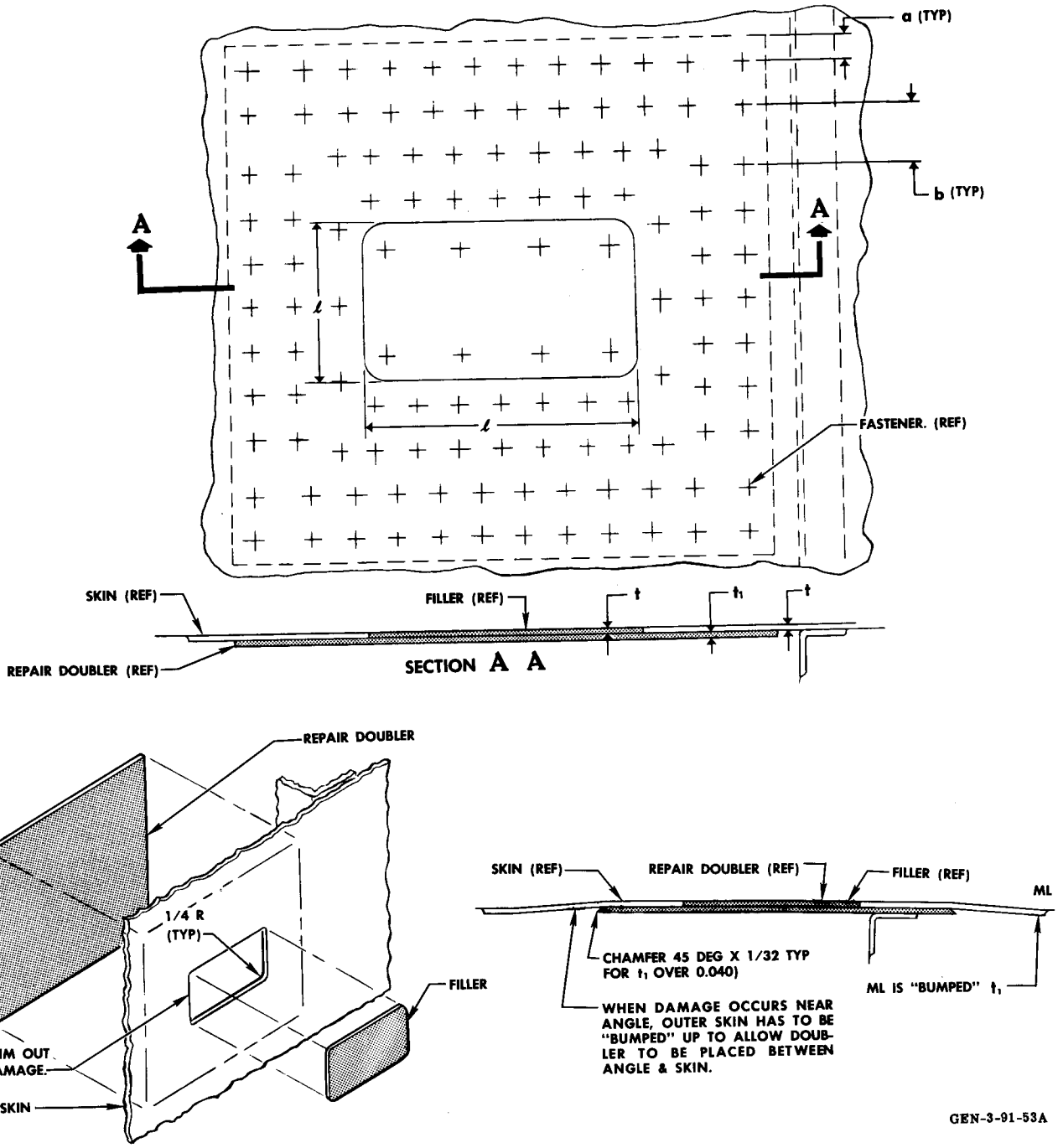
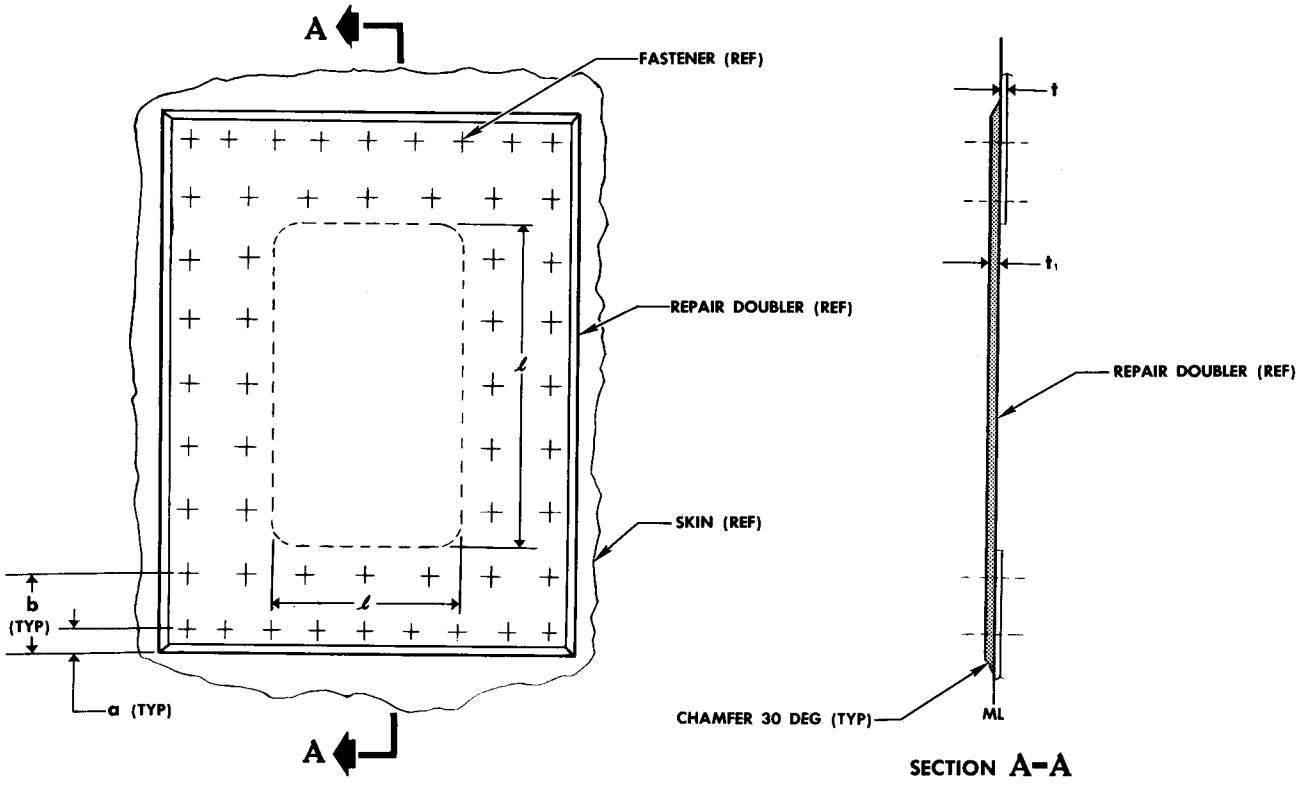


Figure 10-24. Skin Repair—Shear and Tension

SKIN REPAIR—SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



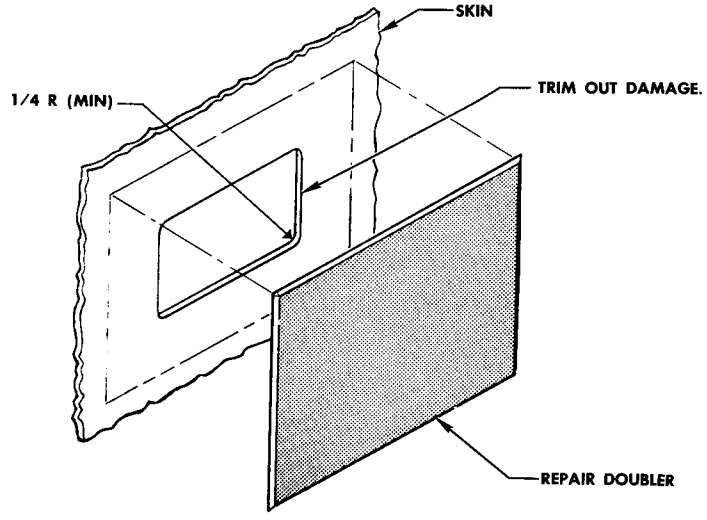


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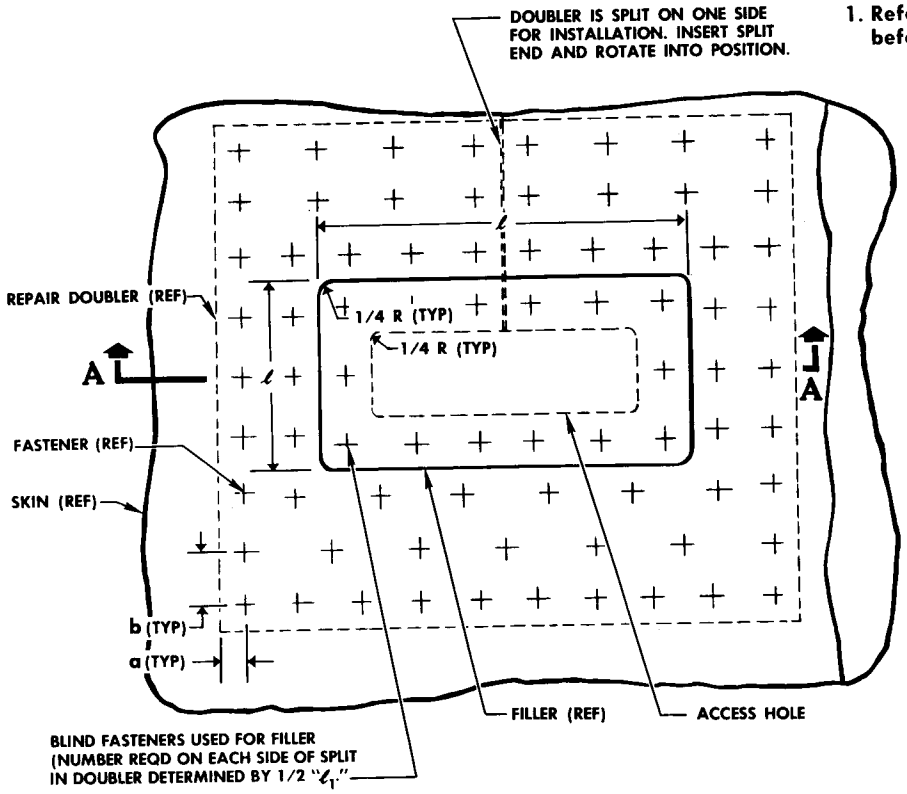
Figure 10-25. Skin Repair—Shear and Tension, Temporary

SKIN REPAIR - BLIND

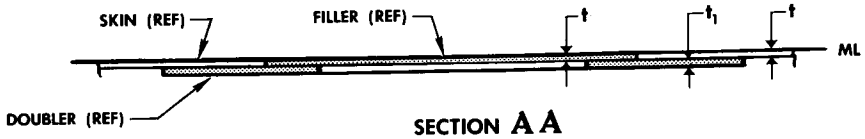
TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



NOTE Use "ℓ" line in the fastener tables for "ℓ1" values.

The fasteners required for "ℓ1" are distributed through filler and doubler approximately as shown. (If the "ℓ1" requirement is fourteen AF463-4 rivets, use seven AF463-4 rivets through filler and doubler on *each side* of the split in the doubler.)



SECTION AA

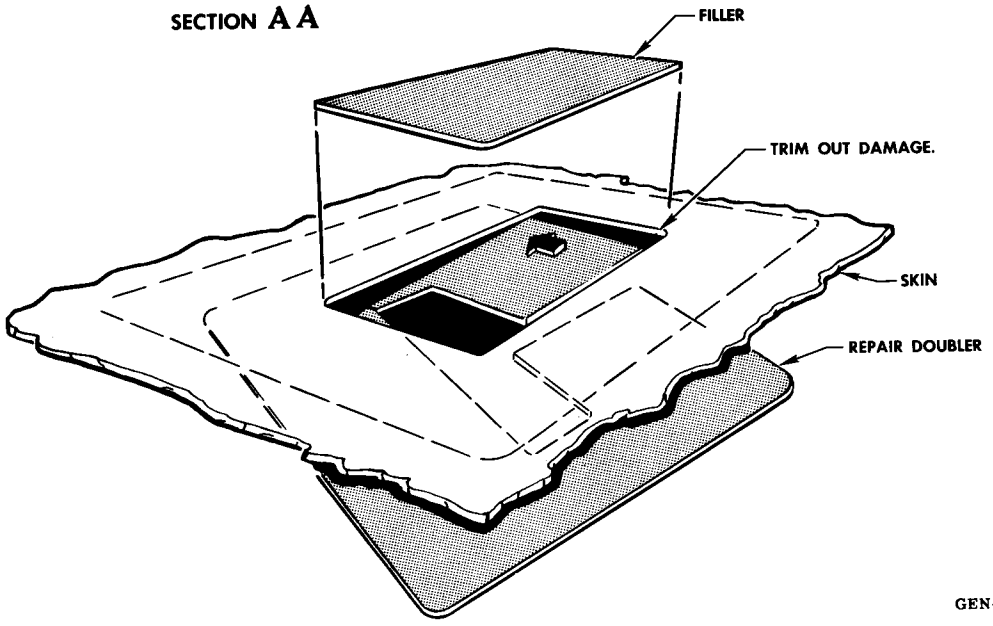


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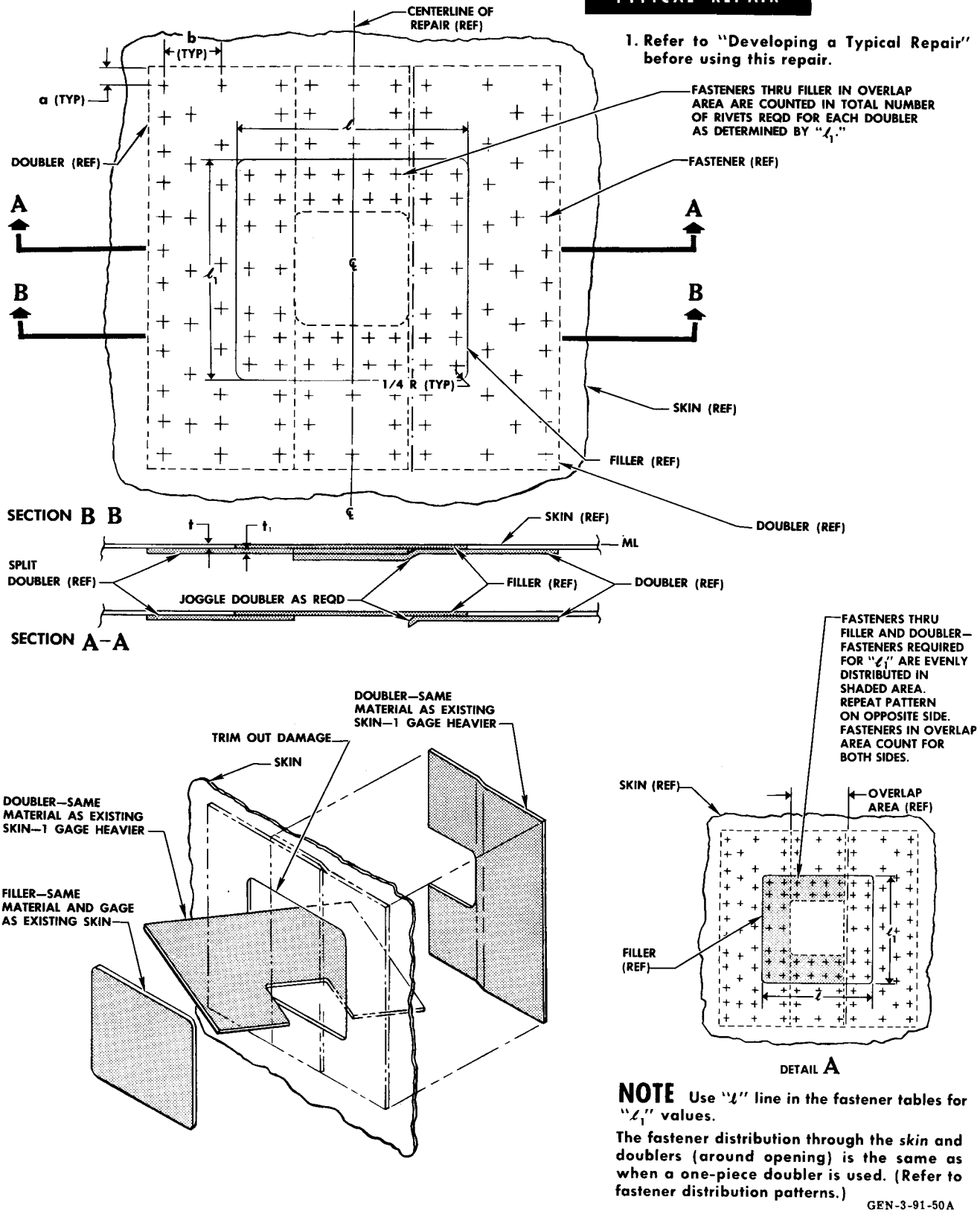
Figure 10-26. Skin Repair—Blind—Shear and Tension (Sheet 1 of 3)

SKIN REPAIR—BLIND

TYPICAL REPAIR



1. Refer to "Developing a Typical Repair" before using this repair.

NOTE Use "l" line in the fastener tables for "l_1" values.

The fastener distribution through the *skin* and doublers (around opening) is the same as when a one-piece doubler is used. (Refer to fastener distribution patterns.)

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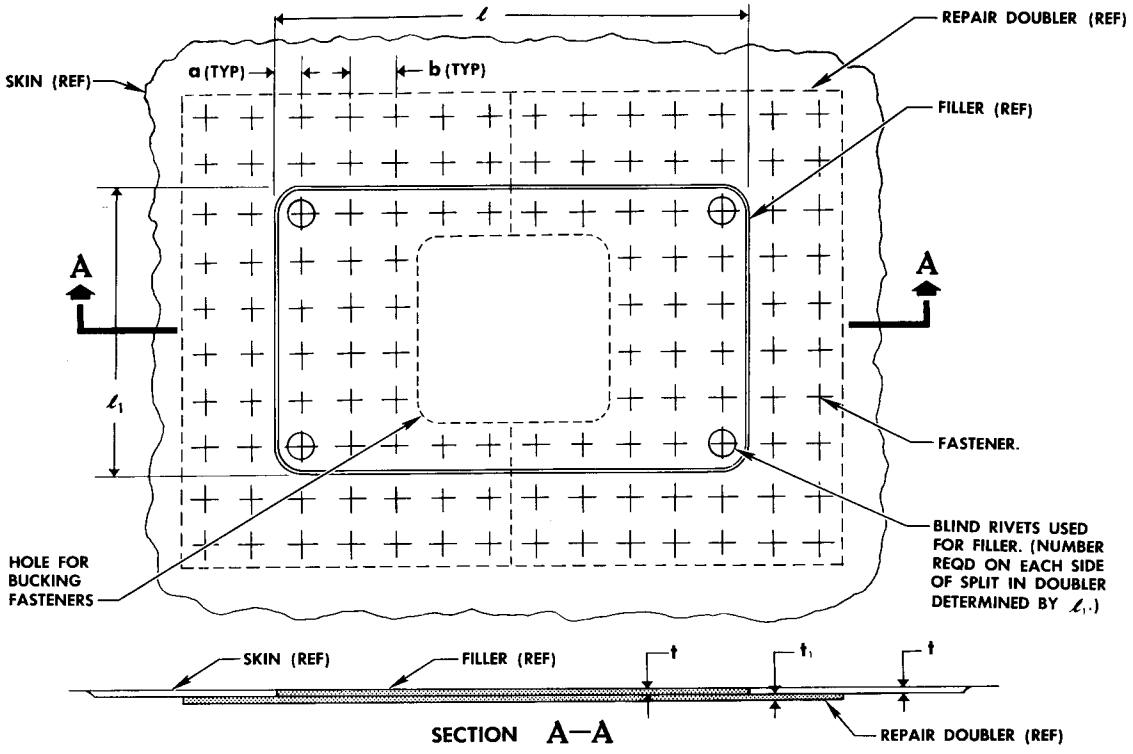
Figure 10-26. Skin Repair—Blind—Shear and Tension (Sheet 2 of 3)

SKIN REPAIR - BLIND

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



NOTE Use "ℓ" line in the fastener tables for "ℓ_1" values. The blind fasteners required for "ℓ_1" are distributed evenly through filler and doublers, approximately as shown. (If the "ℓ_1" requirement is twenty-two AF462-4 rivets, use twenty-two AF462-4 rivets through *each* doubler and filler.)

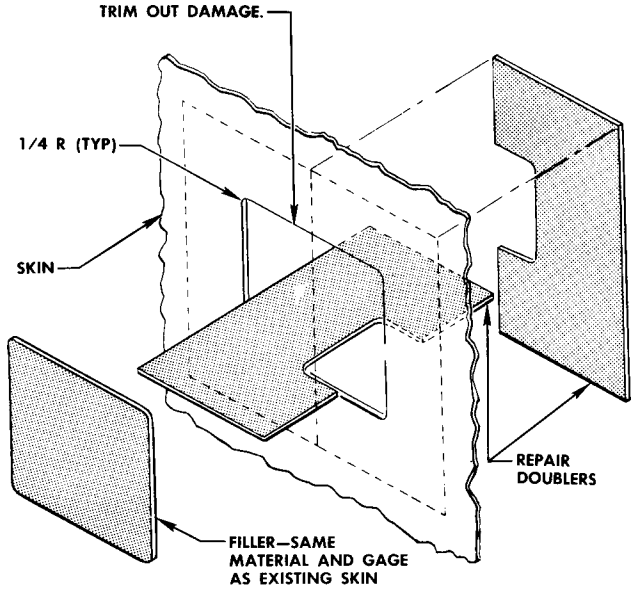


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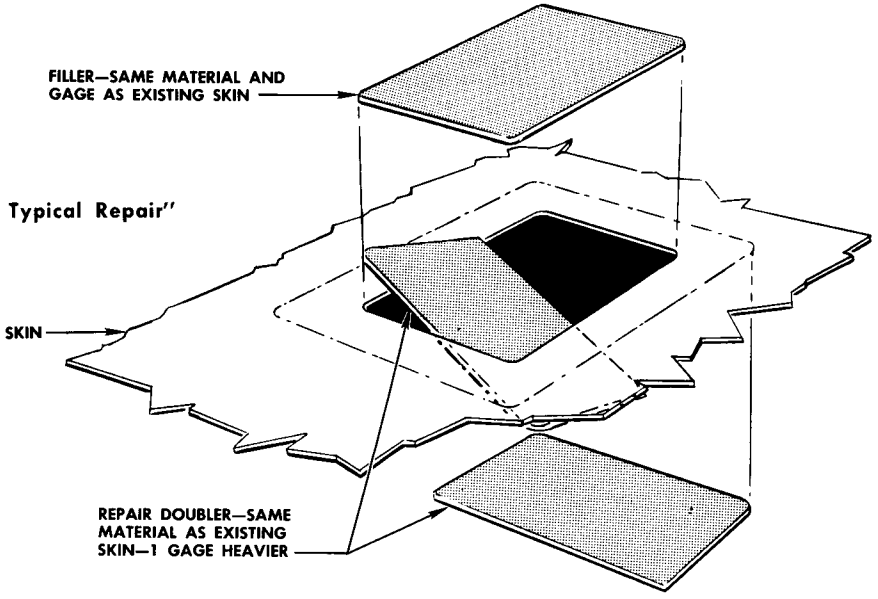
Figure 10-26. Skin Repair—Blind—Shear and Tension (Sheet 3 of 3)

SECONDARY STRUCTURE—BLIND

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



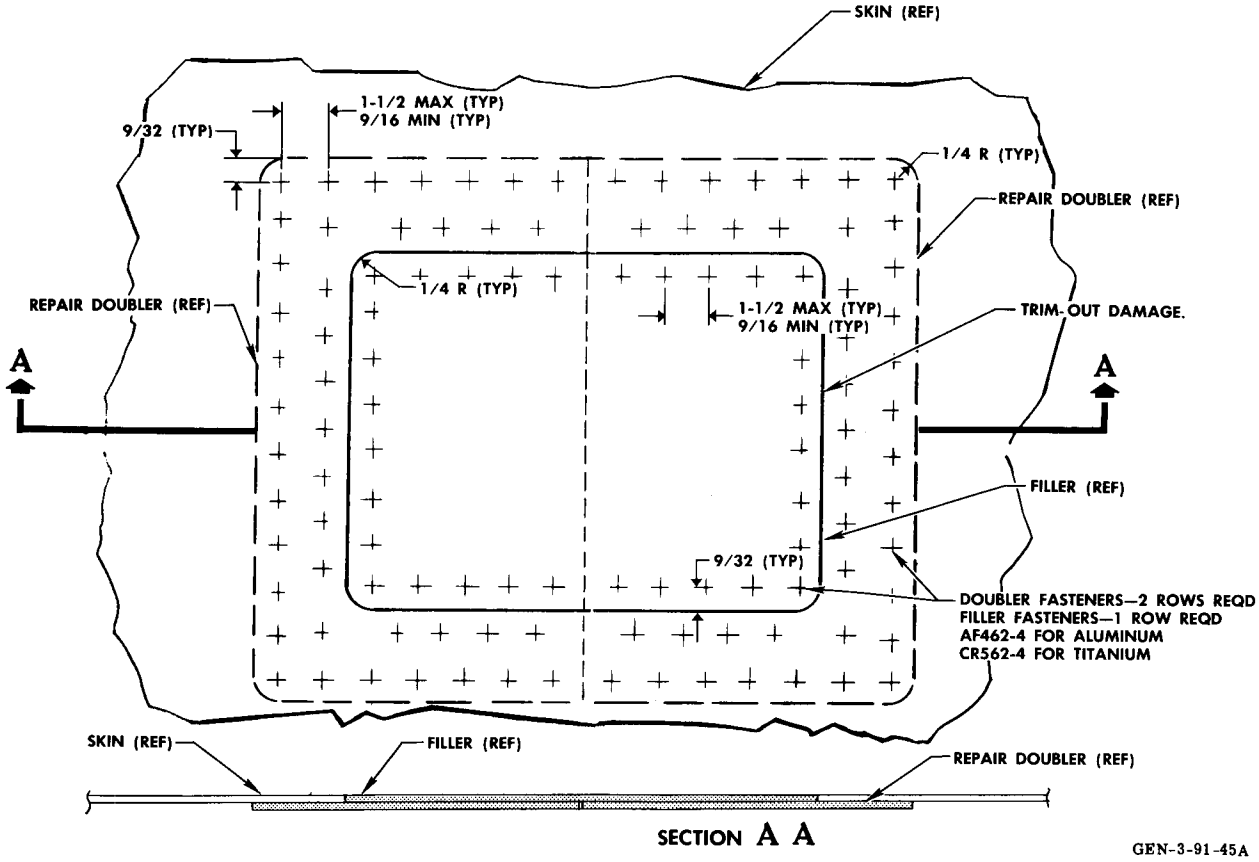
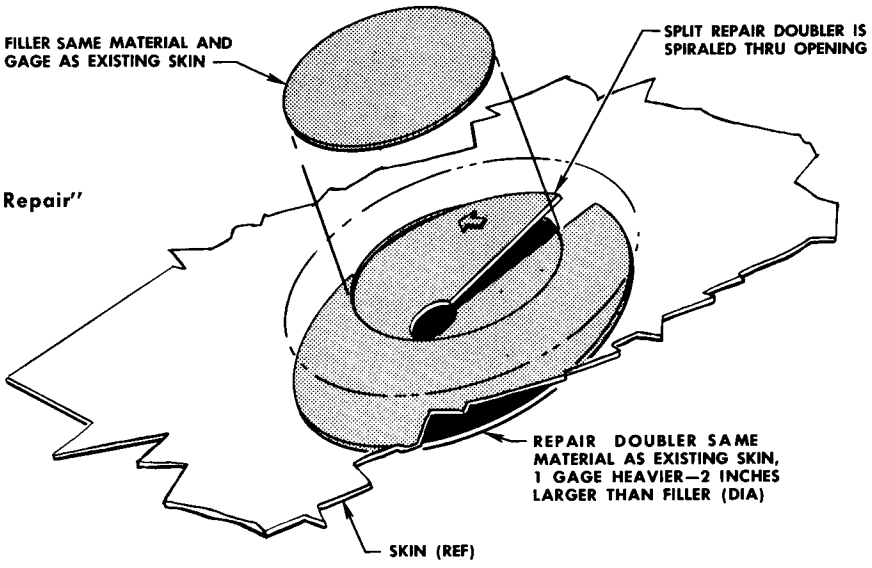


Figure 10-27. Skin Repair—Blind, Secondary Structure (Sheet 1 of 5)

SECONDARY STRUCTURE—BLIND

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



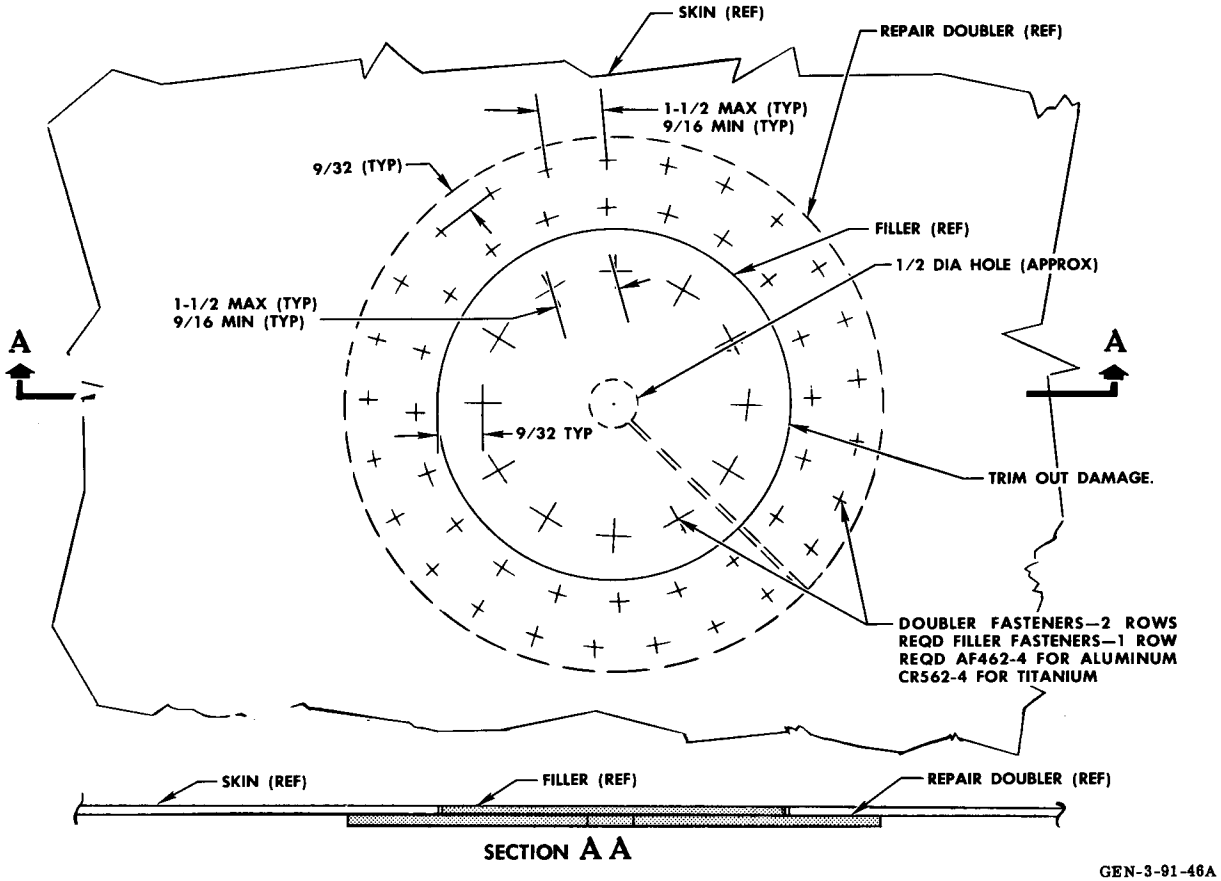
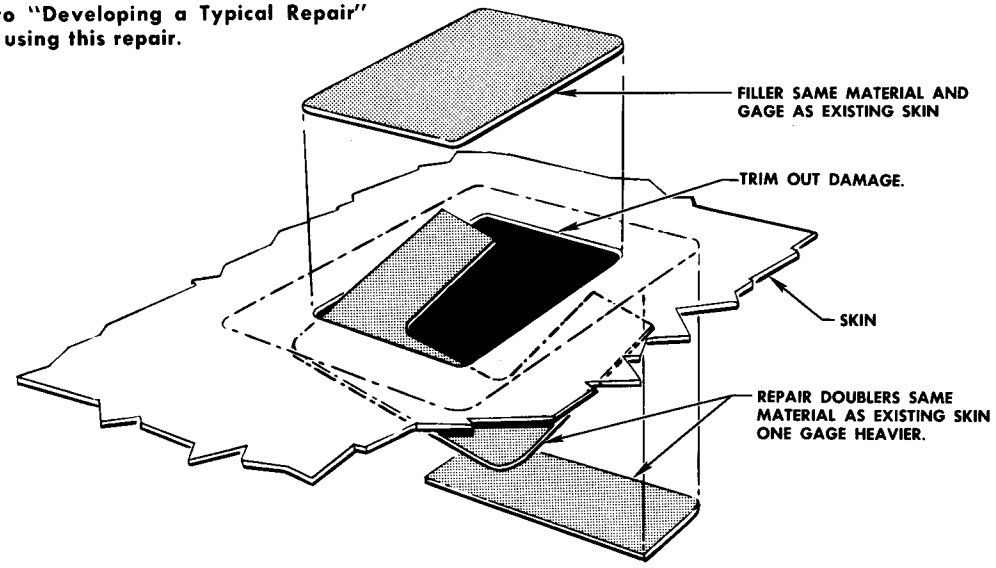


Figure 10-27. Skin Repair—Blind, Secondary Structure (Sheet 2 of 5)

SECONDARY STRUCTURE - BLIND

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



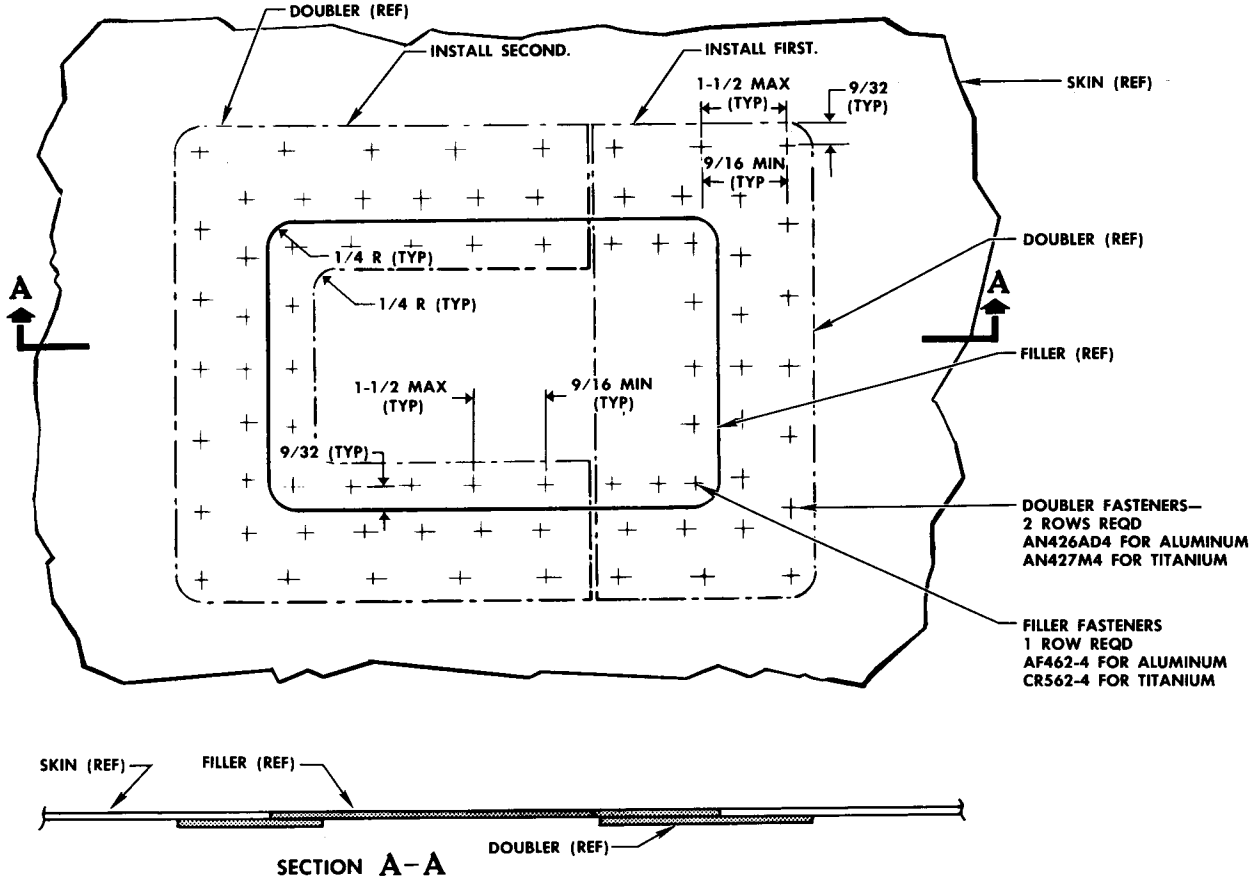


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Figure 10-27. Skin Repair—Blind, Secondary Structure (Sheet 3 of 5)

SECONDARY STRUCTURE—BLIND

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

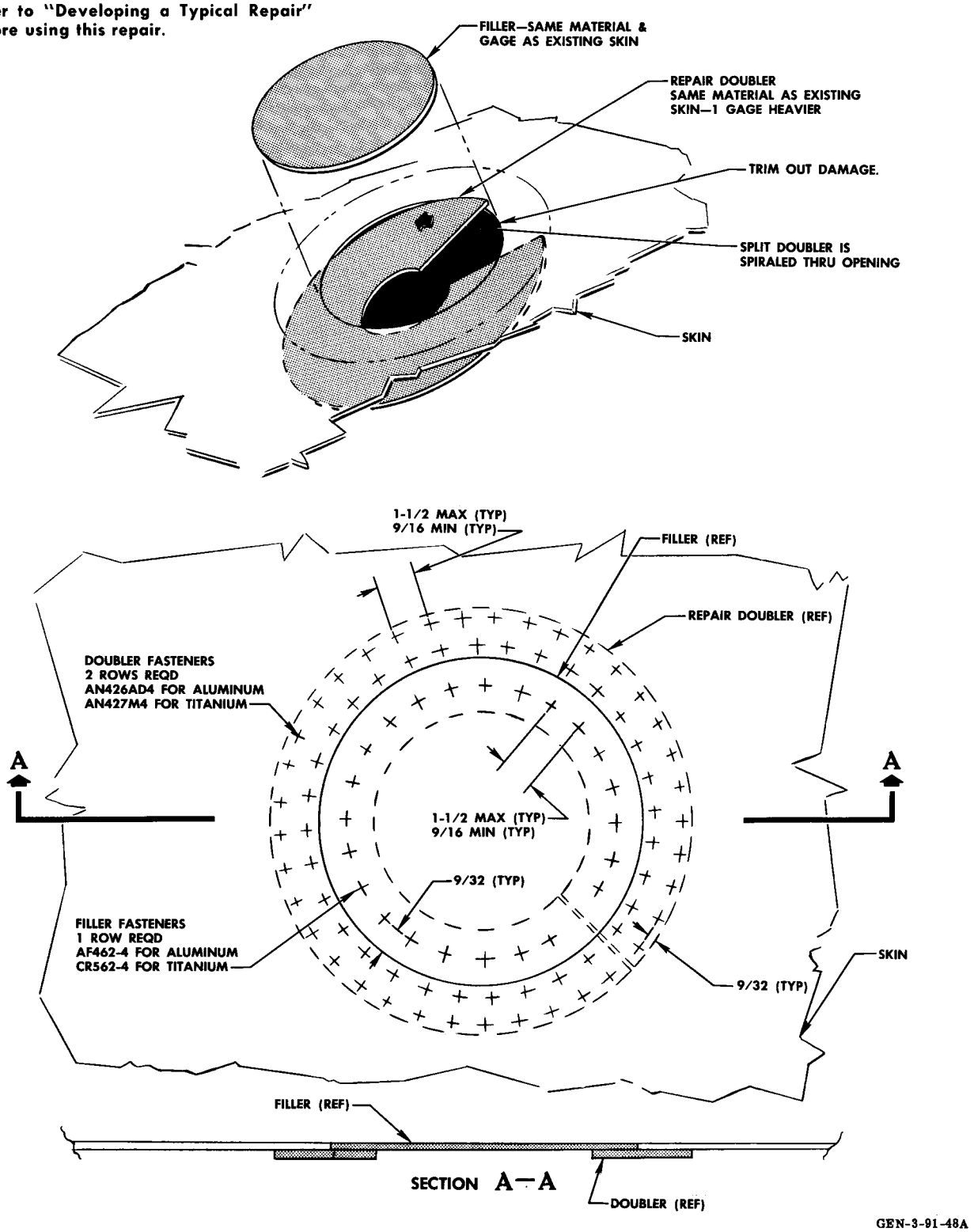
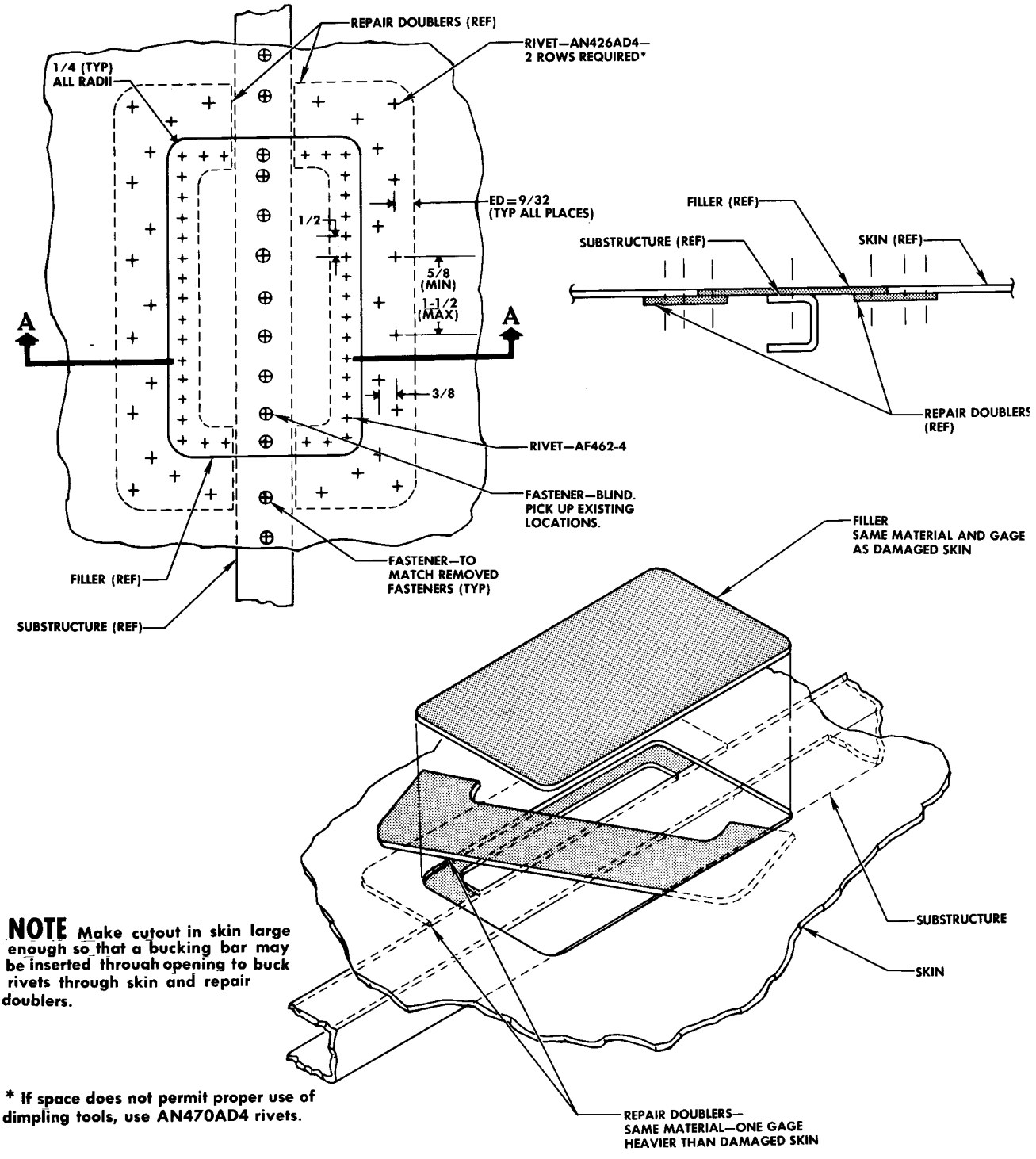


Figure 10-27. Skin Repair—Blind, Secondary Structure (Sheet 4 of 5)

SECONDARY STRUCTURE—BLIND TYPICAL REPAIR



NOTE Make cutout in skin large enough so that a bucking bar may be inserted through opening to buck rivets through skin and repair doublers.

* If space does not permit proper use of dimpling tools, use AN470AD4 rivets.

GEN-3-91-91

Figure 10-27. Skin Repair—Blind, Secondary Structure (Sheet 5 of 5)

SKIN END REPAIR—SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

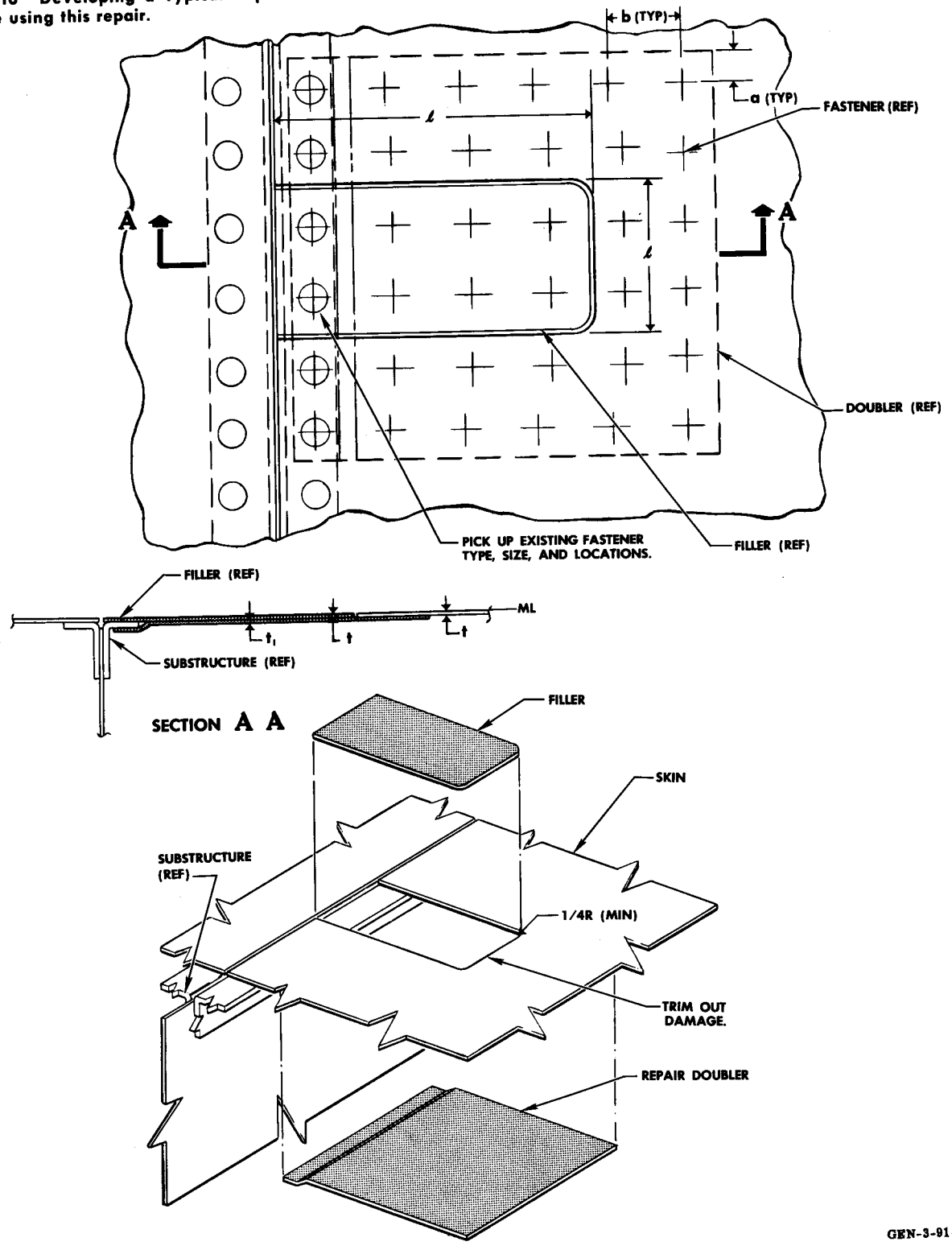
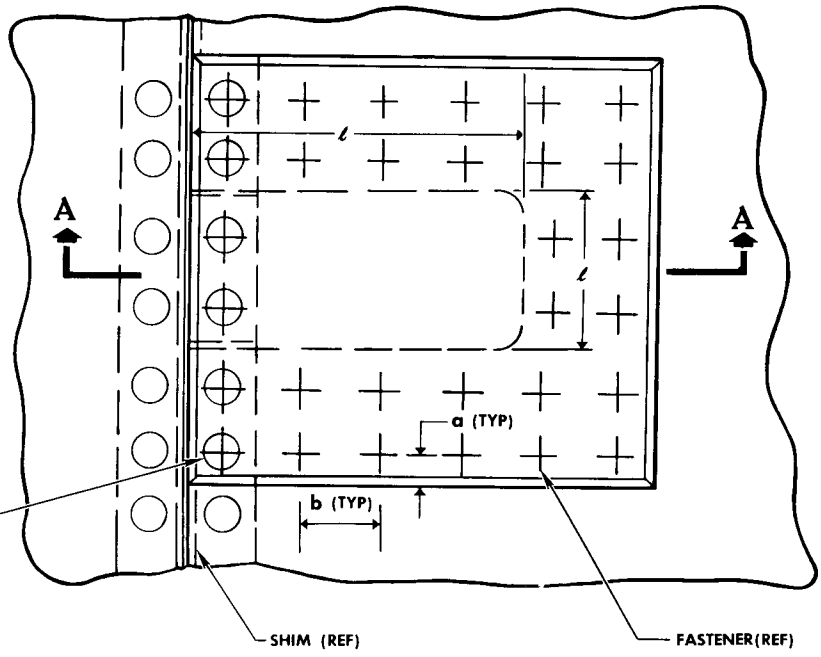


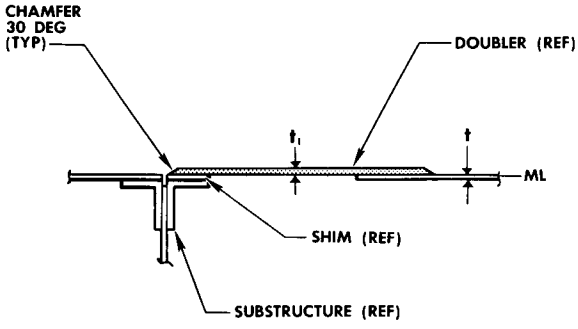
Figure 10-28. Skin End Repair—Shear and Tension

SKIN END REPAIR—SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.





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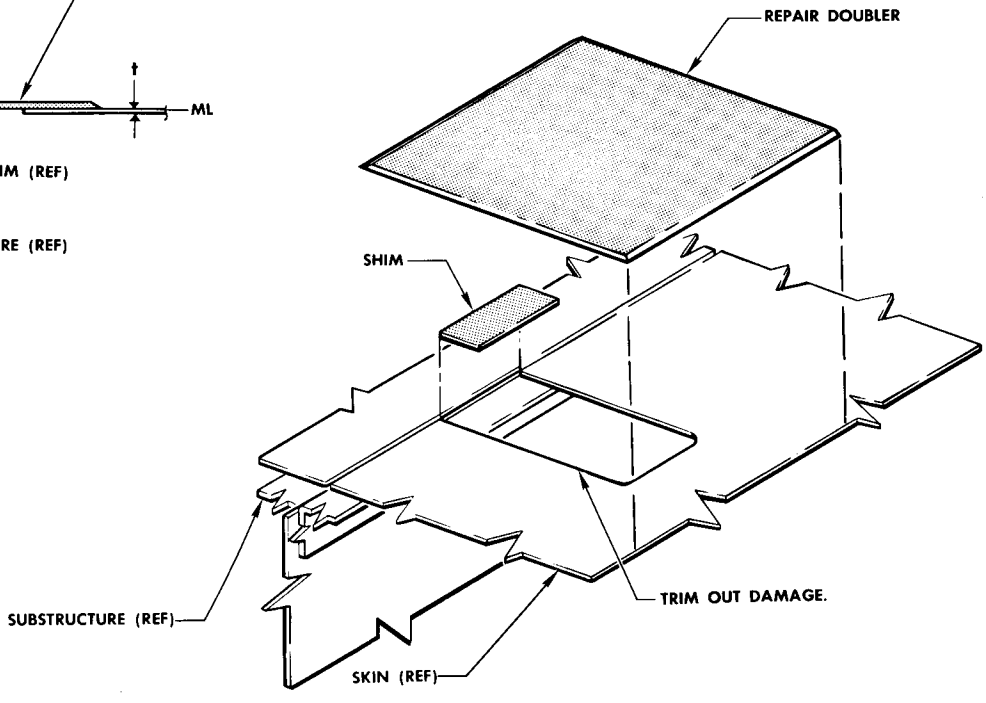


GEN-3-91-41A

Figure 10-29. Skin End Repair—Shear and Tension, Temporary

SKIN END REPAIR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

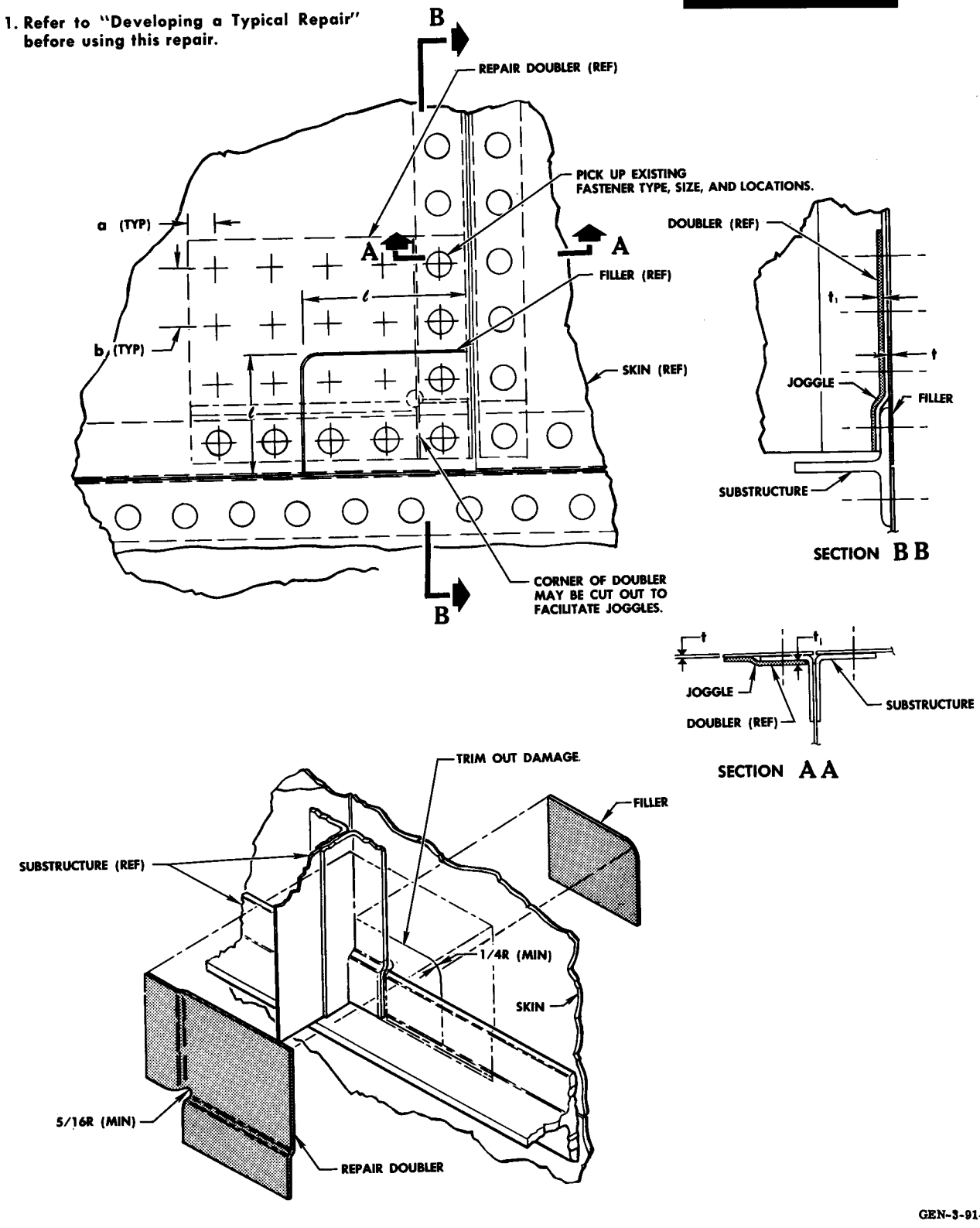
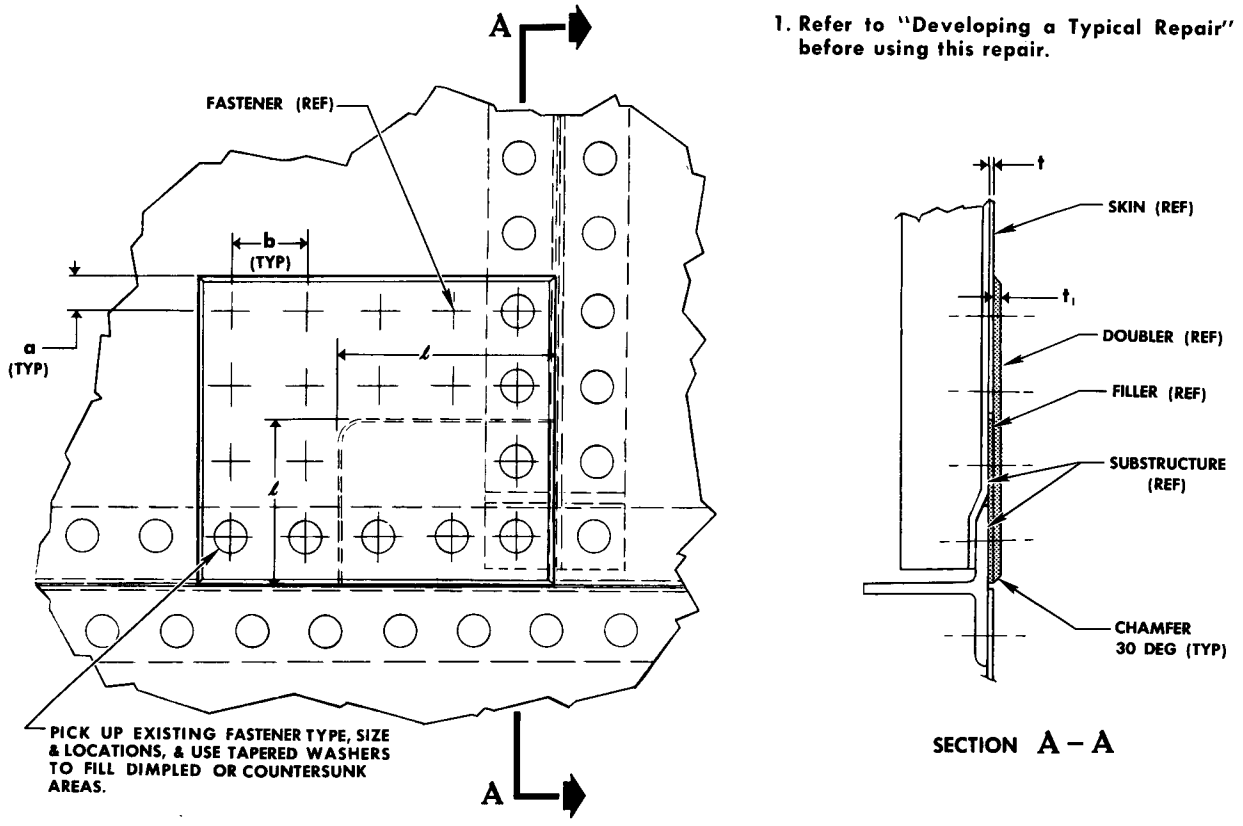
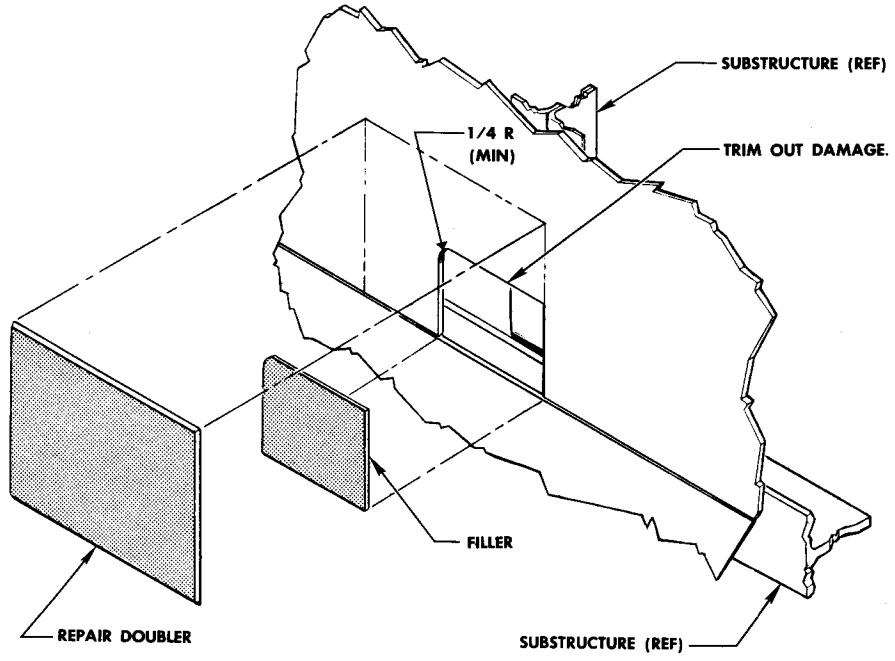


Figure 10-30. Skin End Repair—Corner, Shear and Tension

SKIN END REPAIR—SHEAR

TYPICAL REPAIR





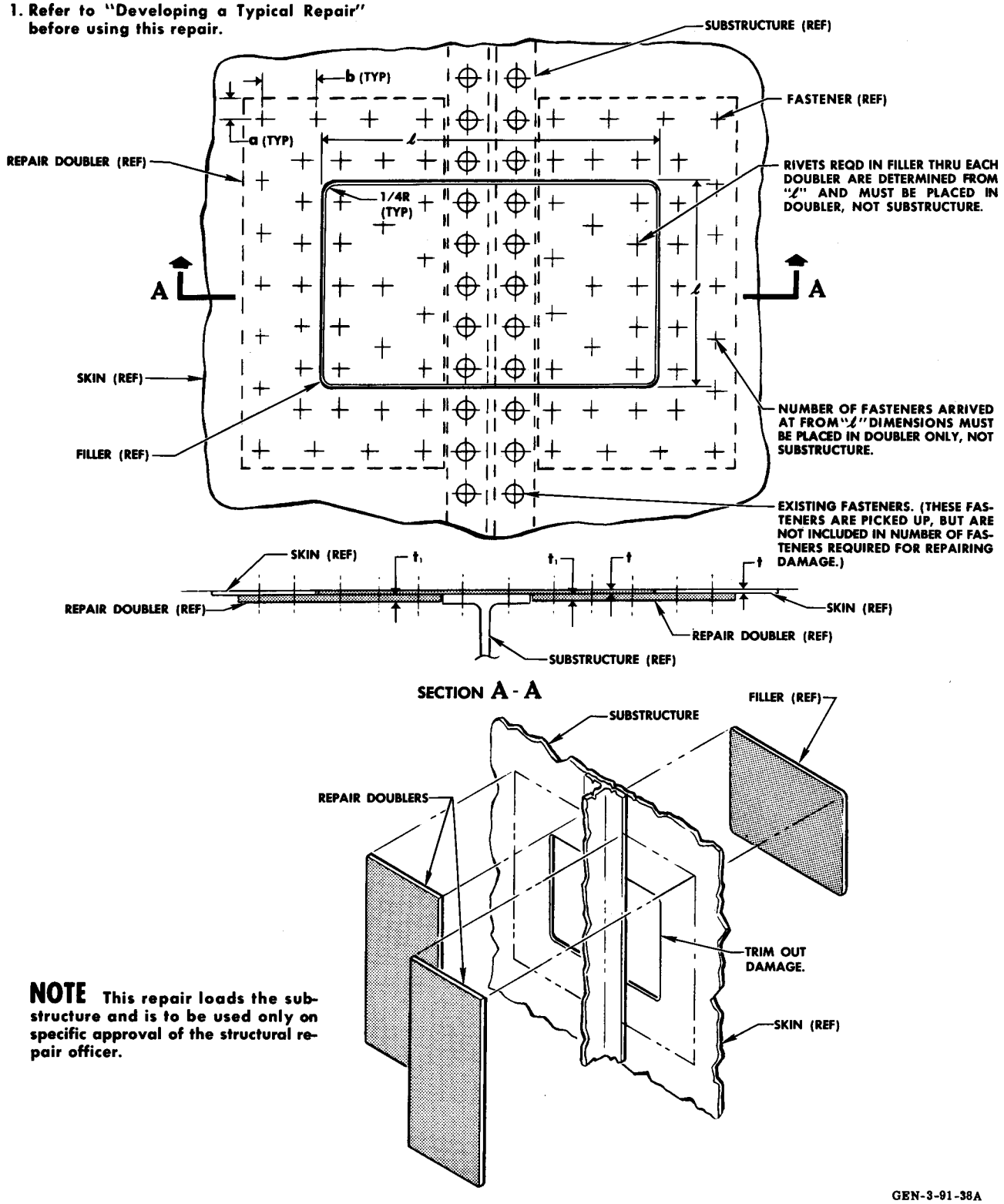
GEN-3-91-43A

Figure 10-31. Skin End Repair—Corner Shear and Tension, Temporary

SKIN OVER SUBSTRUCTURE - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



NOTE This repair loads the substructure and is to be used only on specific approval of the structural repair officer.

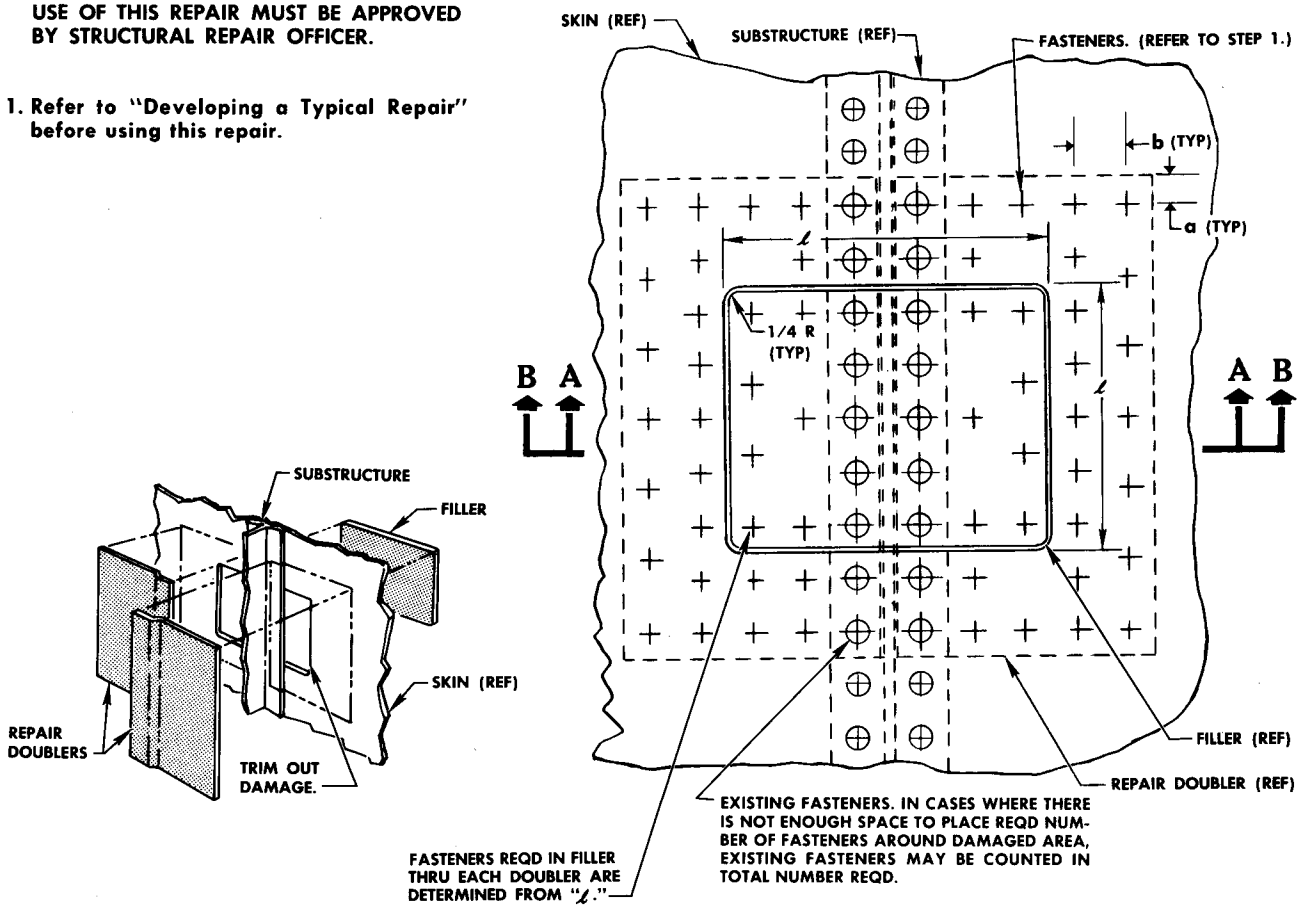
Figure 10-32. Skin Over Substructure Repair—Shear and Tension (Sheet 1 of 3)

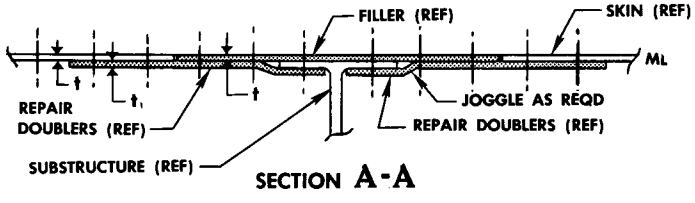
SKIN OVER SUBSTRUCTURE - SHEAR AND TENSION

TYPICAL REPAIR

USE OF THIS REPAIR MUST BE APPROVED BY STRUCTURAL REPAIR OFFICER.

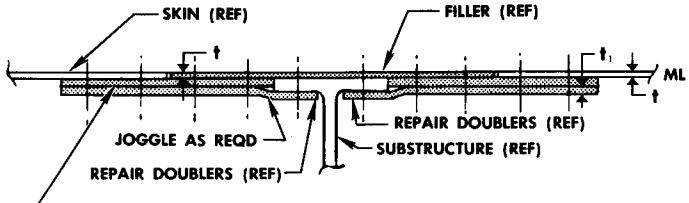
1. Refer to "Developing a Typical Repair" before using this repair.





SECTION A-A

SECTION B-B



WHEN THICKNESS OF EXISTING SUBSTRUCTURE CAUSES JOGGLE RUNOUT TO INTERFERE WITH RIVET INSTALLATION, SPACERS MAY BE PLACED BETWEEN DOUBLERS & SKIN

GEN-3-91-39A

Figure 10-32. Skin Over Substructure Repair—Shear and Tension (Sheet 2 of 3)

SKIN OVER SUBSTRUCTURE - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

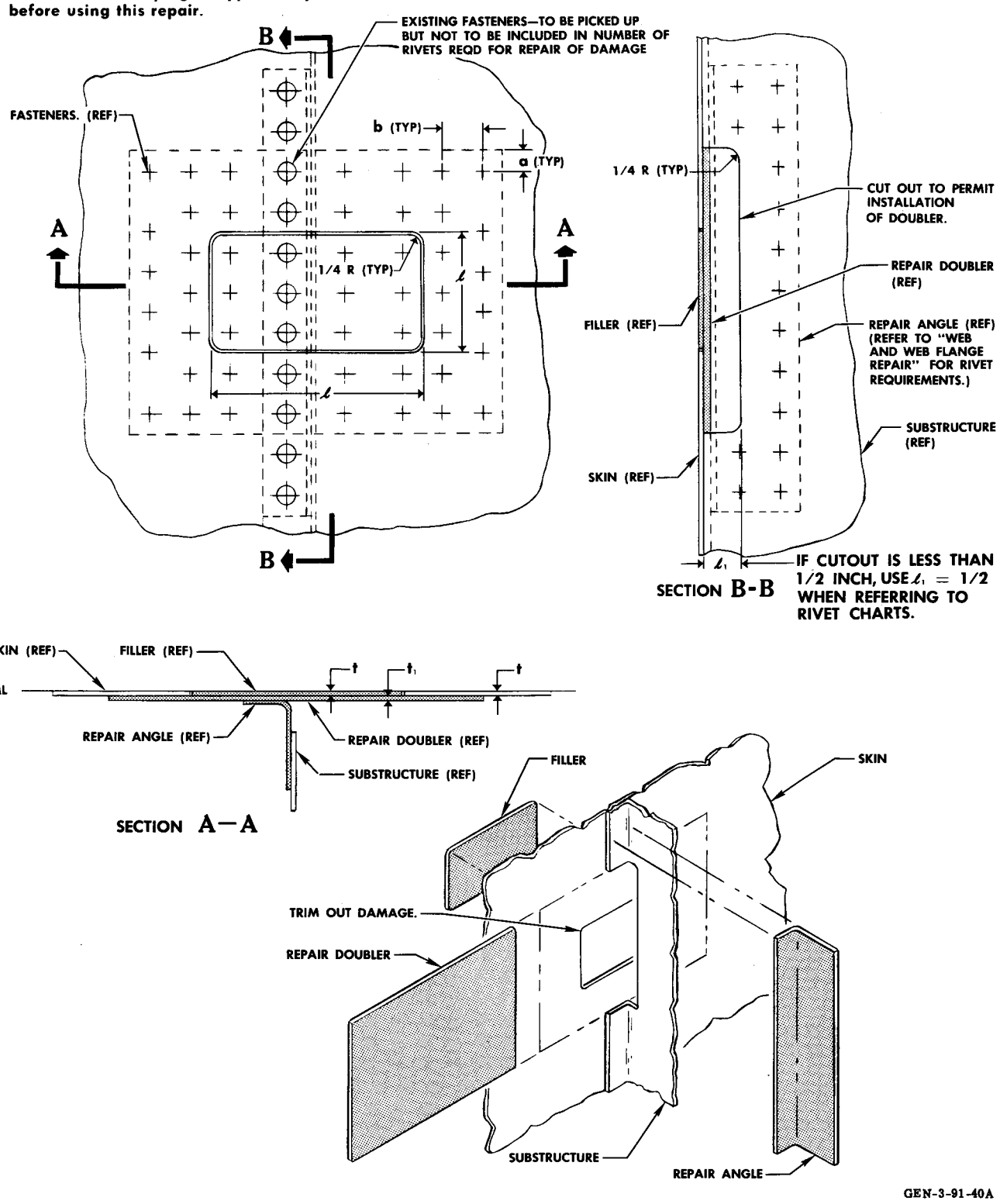


Figure 10-32. Skin Over Substructure Repair—Shear and Tension (Sheet 3 of 3)

SKIN AT SUBSTRUCTURE - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

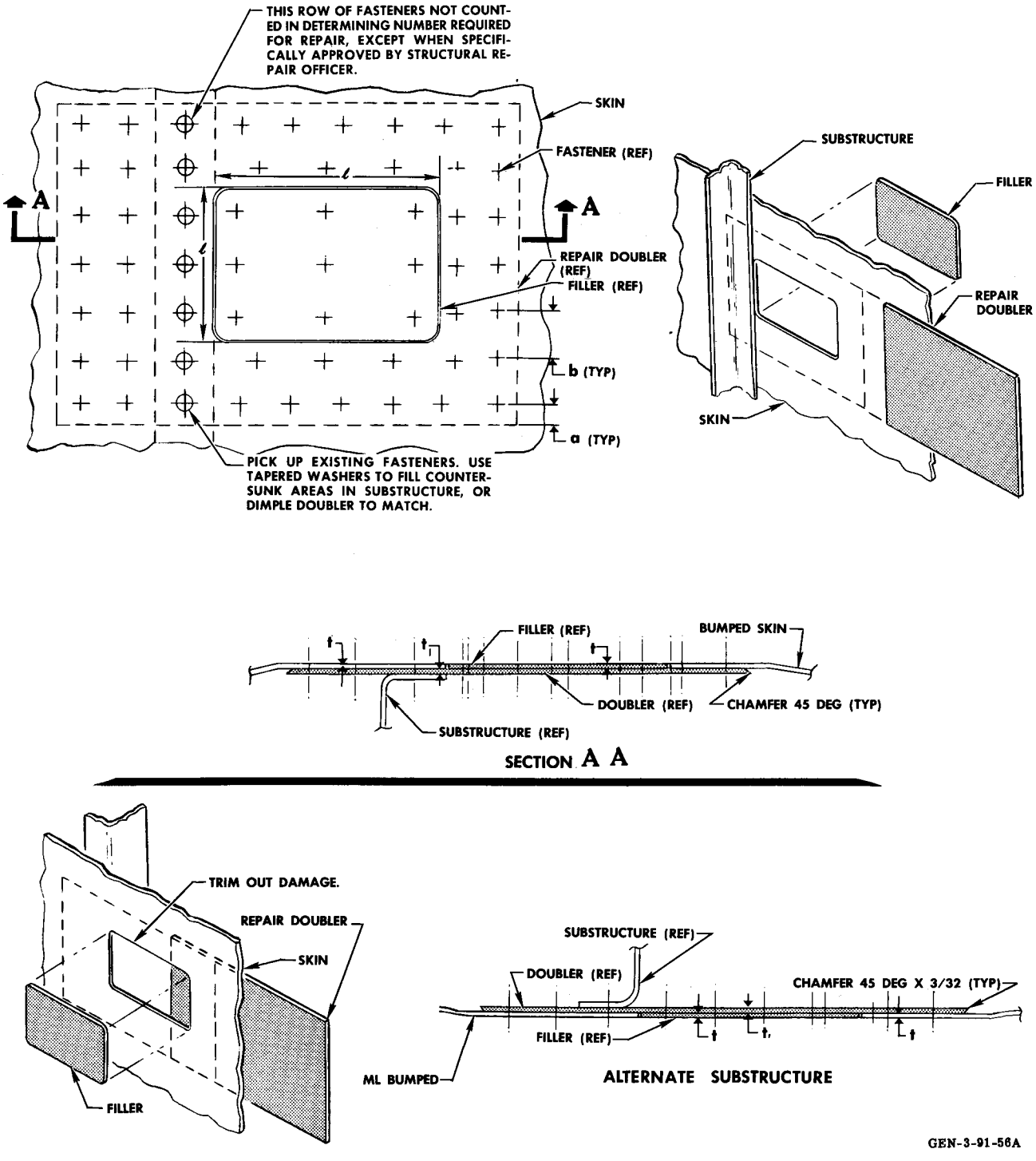


Figure 10-33. Skin Repair Adjacent to Substructure—Shear and Tension (Sheet 1 of 3)

SKIN AT SUBSTRUCTURE - SHEAR AND TENSION

TYPICAL REPAIR

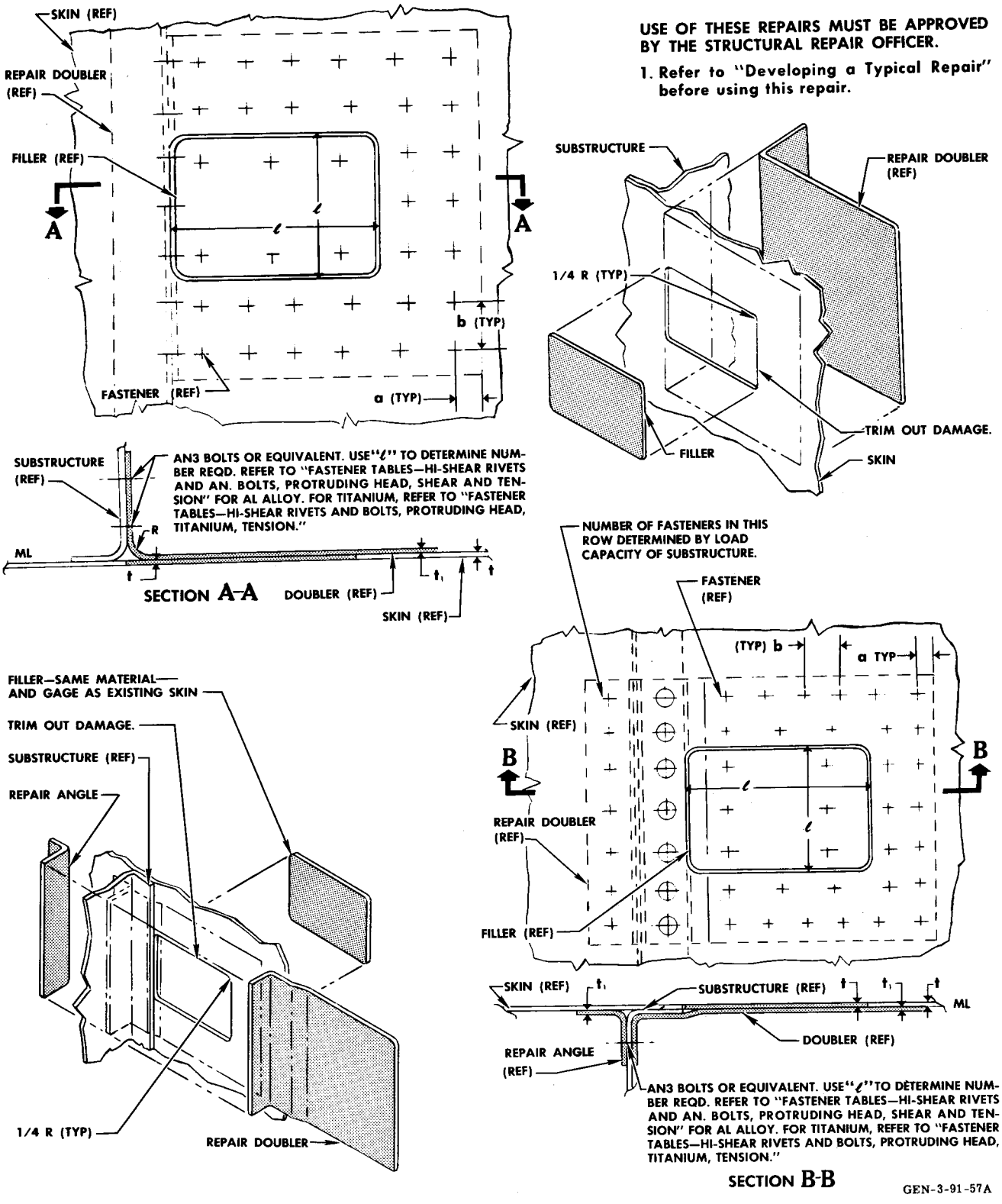


Figure 10-33. Skin Repair Adjacent to Substructure—Shear and Tension (Sheet 2 of 3)

SKIN AT SUBSTRUCTURE—SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

USE OF THESE REPAIRS MUST BE APPROVED BY STRUCTURAL REPAIR OFFICER.

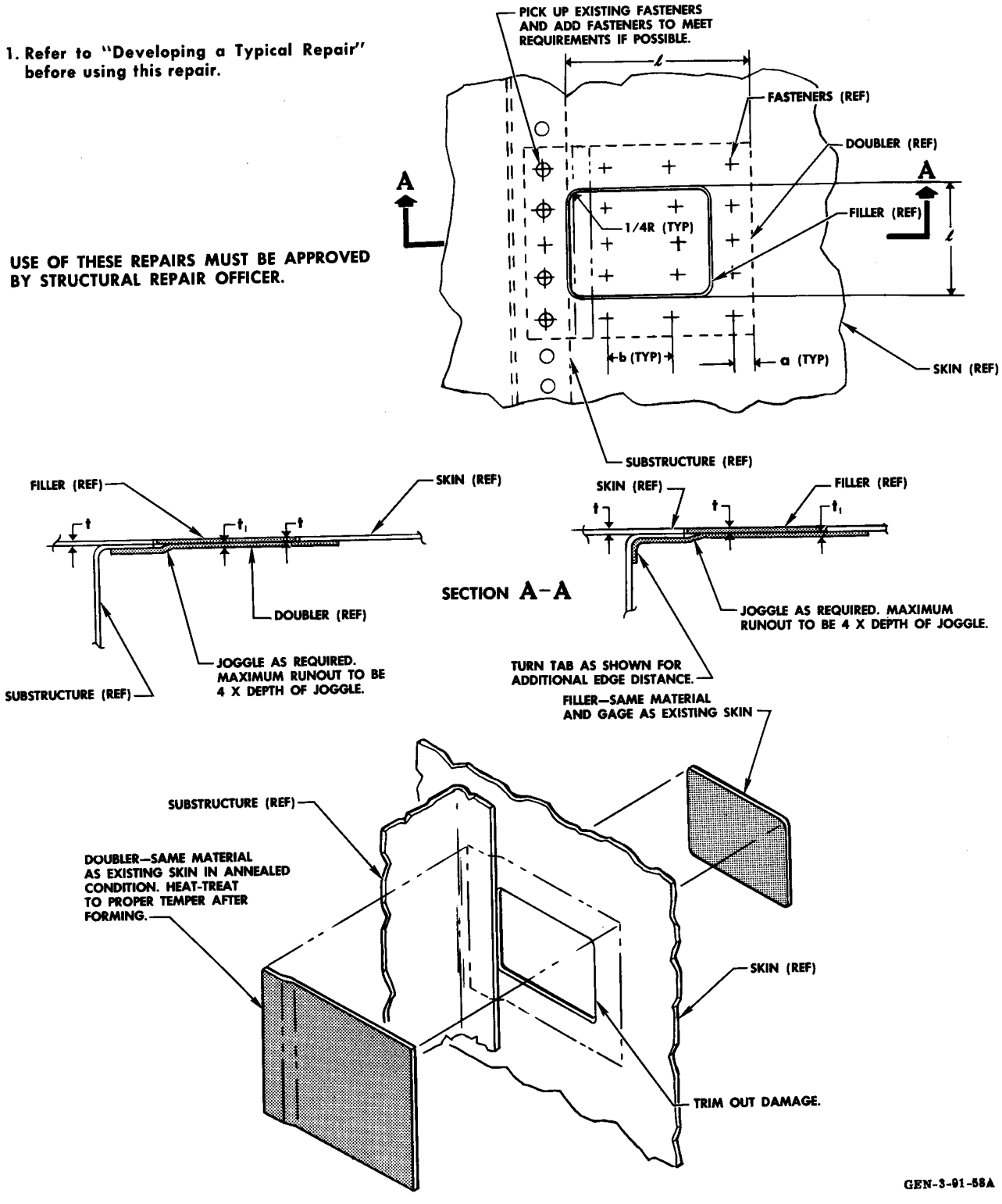


Figure 10-33. Skin Repair Adjacent to Substructure—Shear and Tension (Sheet 3 of 3)

SKIN AND WEB - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.

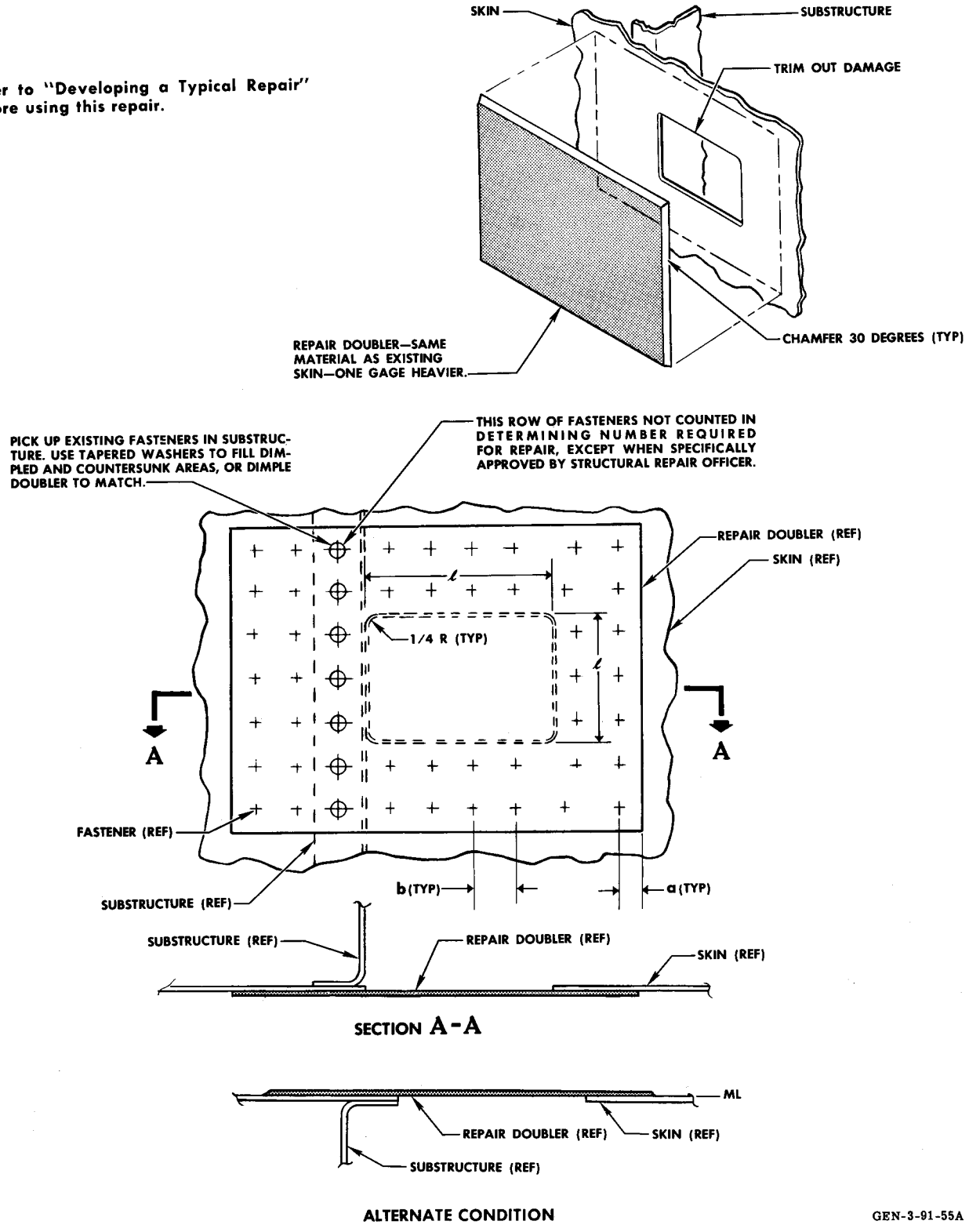
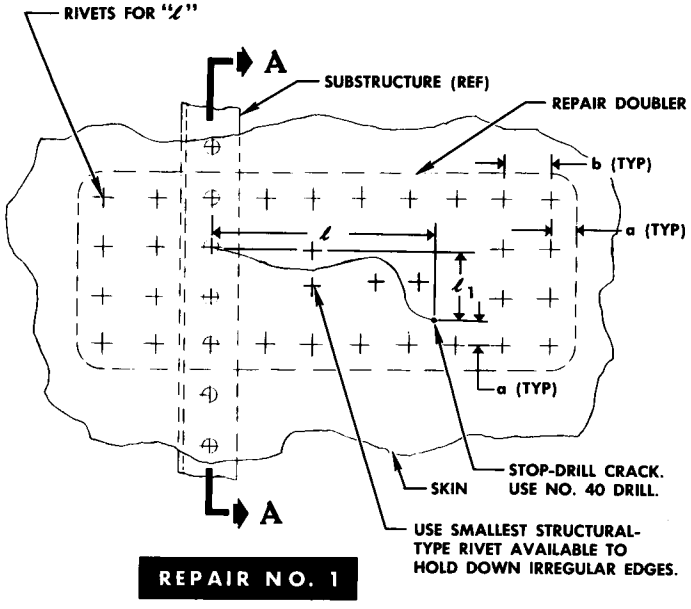


Figure 10-34. Skin Repair Adjacent to Substructure—Shear and Tension, Temporary

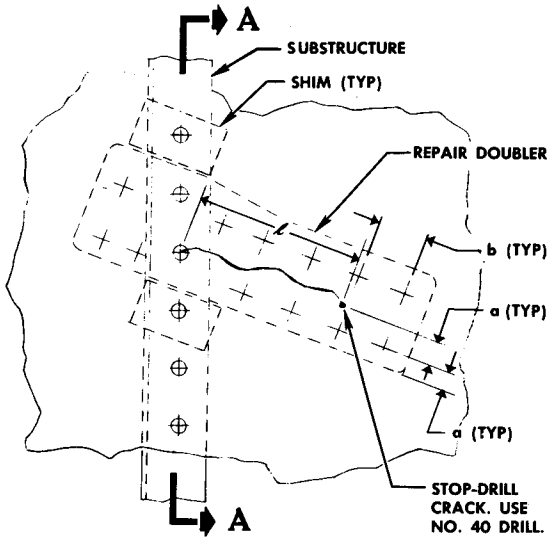
SKIN AND WEB

TYPICAL REPAIR

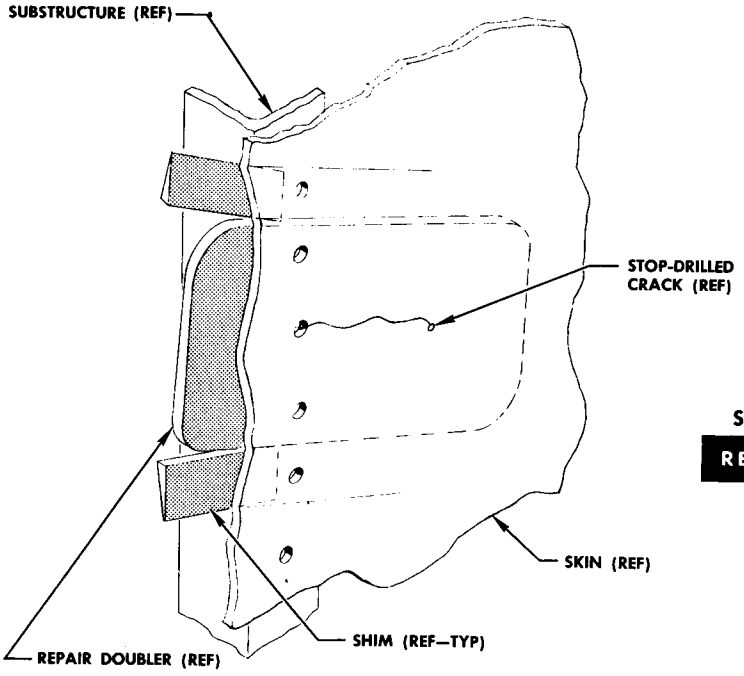
1. Refer to "Developing a Typical Repair" before using this repair.

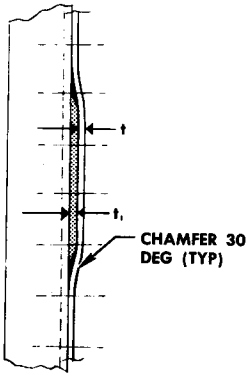


REPAIR NO. 1



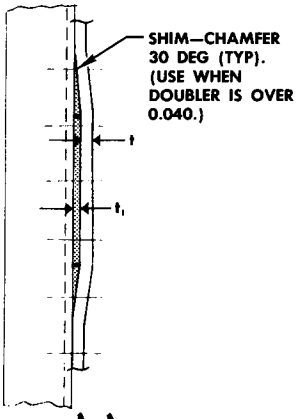
REPAIR NO. 2





SECTION A-A

REPAIR NO. 1



SECTION A-A

REPAIR NO. 2

NOTE Distribute rivets approximately as shown. When the crack deviates from a generally straight line, it should be treated as a rectangle with two "ℓ" dimensions.

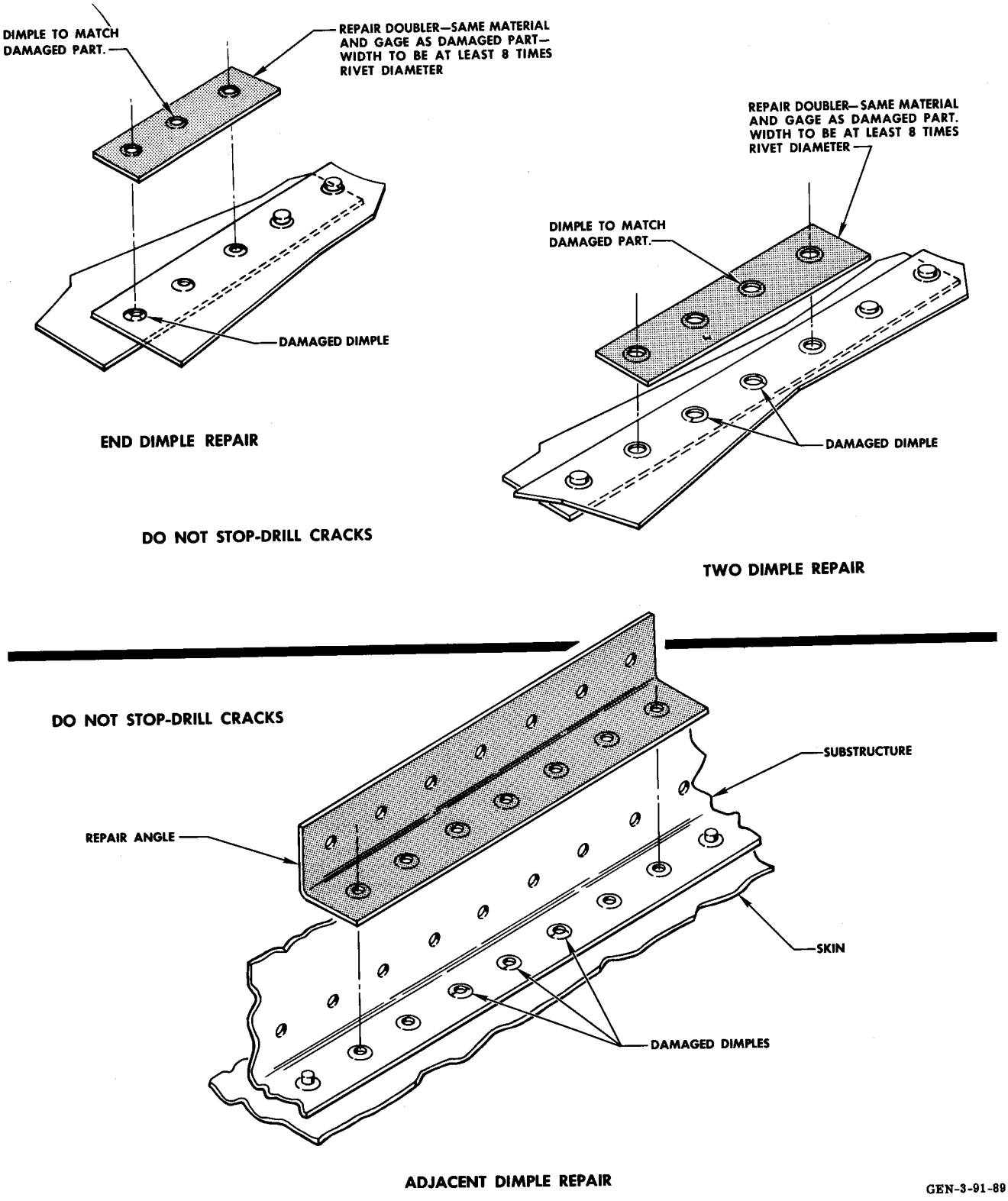
- When doubler material is 0.040 inch or less, chamfer edges as shown in repair No. 1. When the doubler material is over 0.040 inch, shims should be used, as shown in repair No. 2.

GEN-3-91-81A

Figure 10-35. Cracked Skin and Web Repair

CRACKED DIMPLE

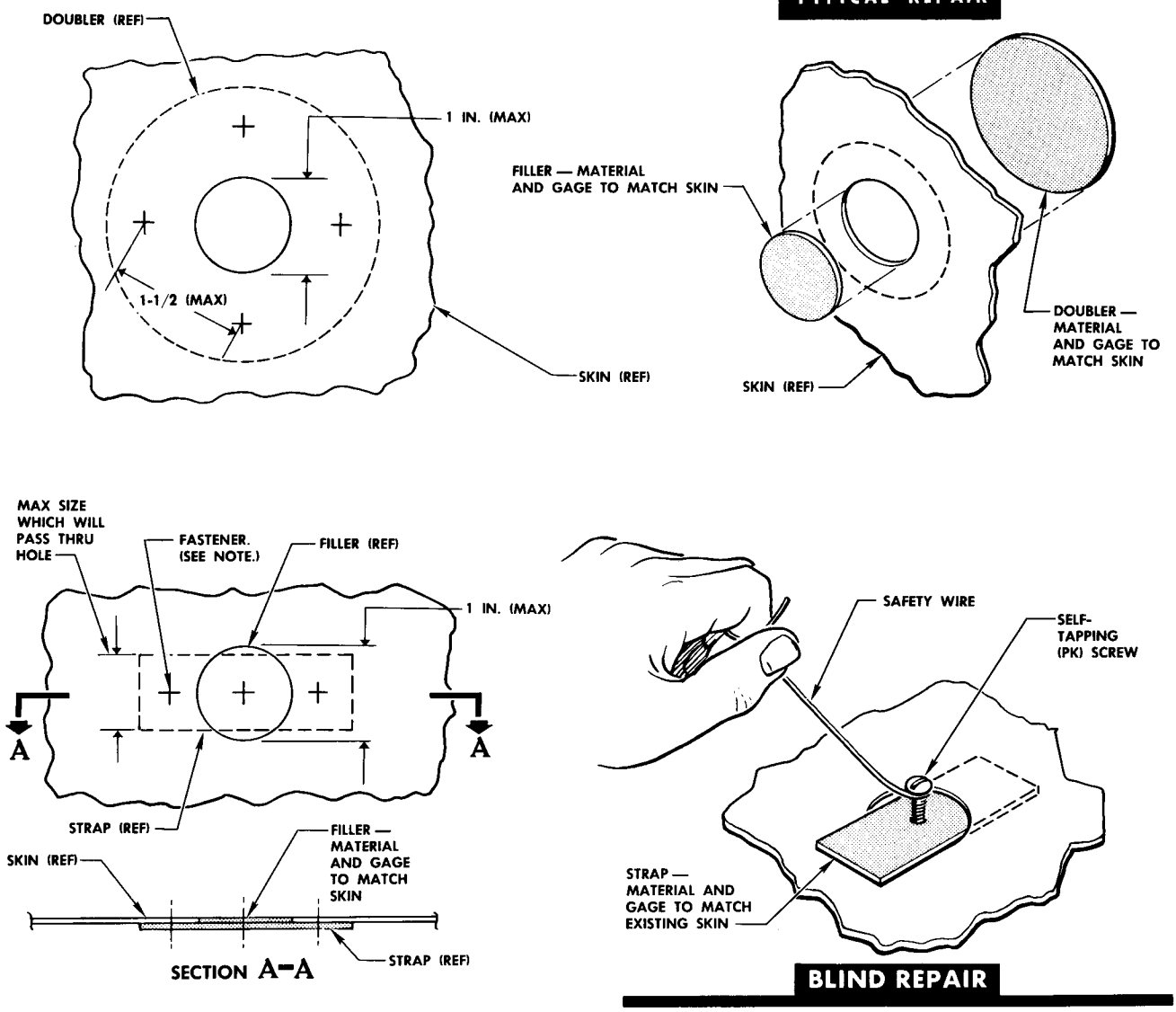
TYPICAL REPAIR



GEN-3-91-89

Figure 10-36. Cracked Dimple Repair

MINOR HOLES



NOTE These repairs are for holes shown as *negligible*, on negligible damage illustrations.

- (The self-tapping screw and safety wire are suggested as means for holding the strap while it is being positioned, drilled, and riveted.)
- Use 3/32- or 1/8-inch flush rivets. (Refer to minimum countersink thickness table.) When 3/32-inch hollow Cherry rivets are used, the hole in the rivet must be plugged with aerodynamic skin filler. Use 3/32- or 1/8-inch universal head rivets in materials under 0.032 inch when material cannot be dimpled.
- Minor hole repairs in sealed areas must be sealed the same as the surrounding area.
- Aerodynamic sealer may be used around the edge of the filler to obtain a smooth outer surface.
- The circular repair doubler shown may be used on the outside of the skin as a temporary repair. Install with flush rivets and chamfer edges of the patch.

MINIMUM MATERIAL THICKNESS FOR COUNTERSINKING

RIVET			MATERIAL
DIA	AN426	CHERRY	
3/32	AD3	CR-116-3	0.032
1/8	AD4	AF462-4	0.040

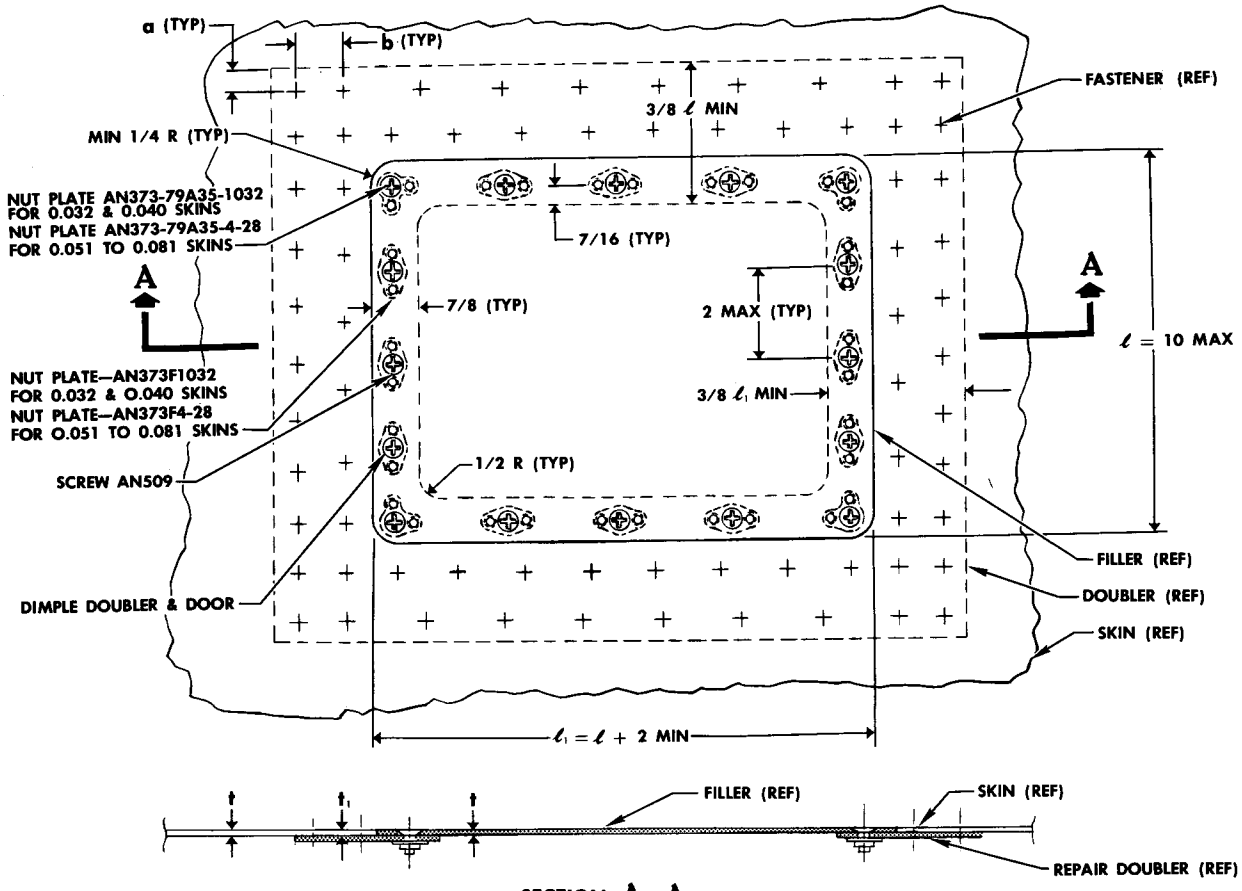
GEN-3-91-90

Figure 10-37. Minor Hole Repairs

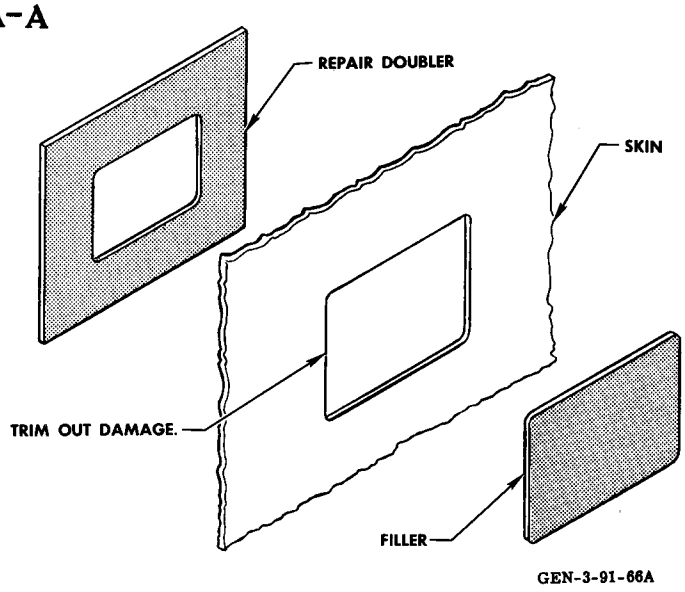
STRUCTURAL DOOR - SHEAR AND TENSION

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



SECTION A-A



NOTE This repair applies to 0.032- through 0.081-inch skins.

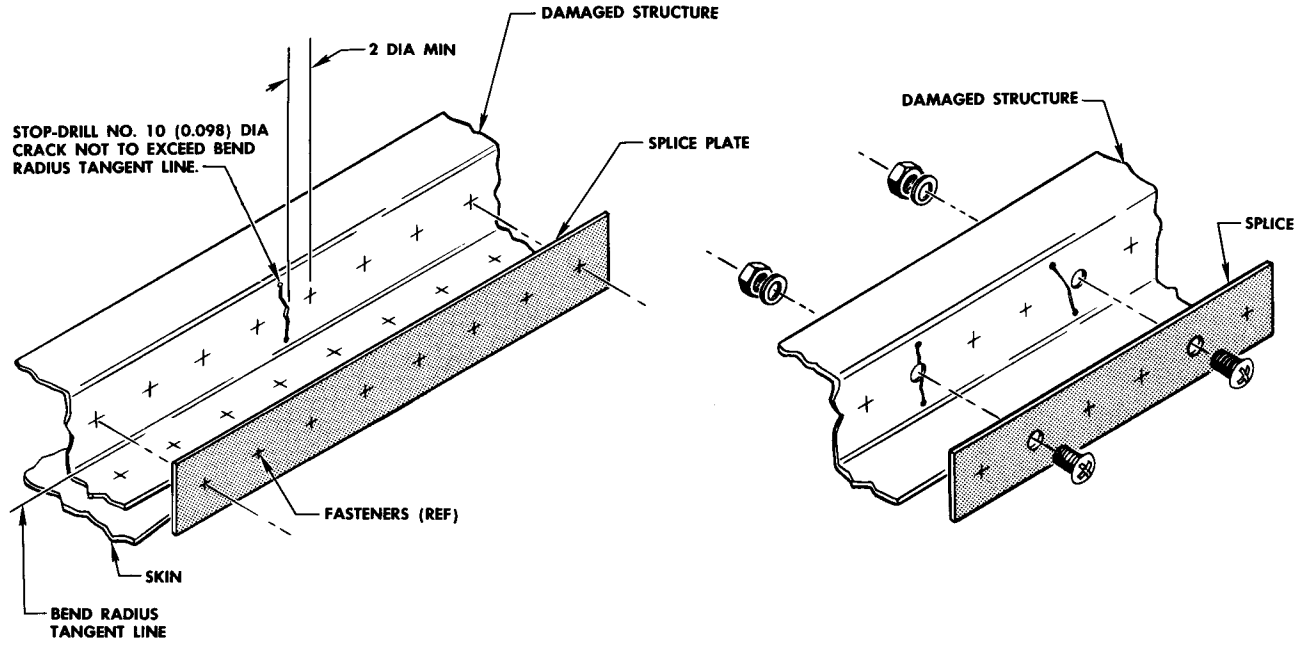
• For both "ℓ" and "ℓ1" dimensions, refer to "ℓ" column in fastener tables.

Figure 10-38. Access Door Installation—Structural Shear and Tension

CRACKS—SHAPES AND EXTRUSIONS

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



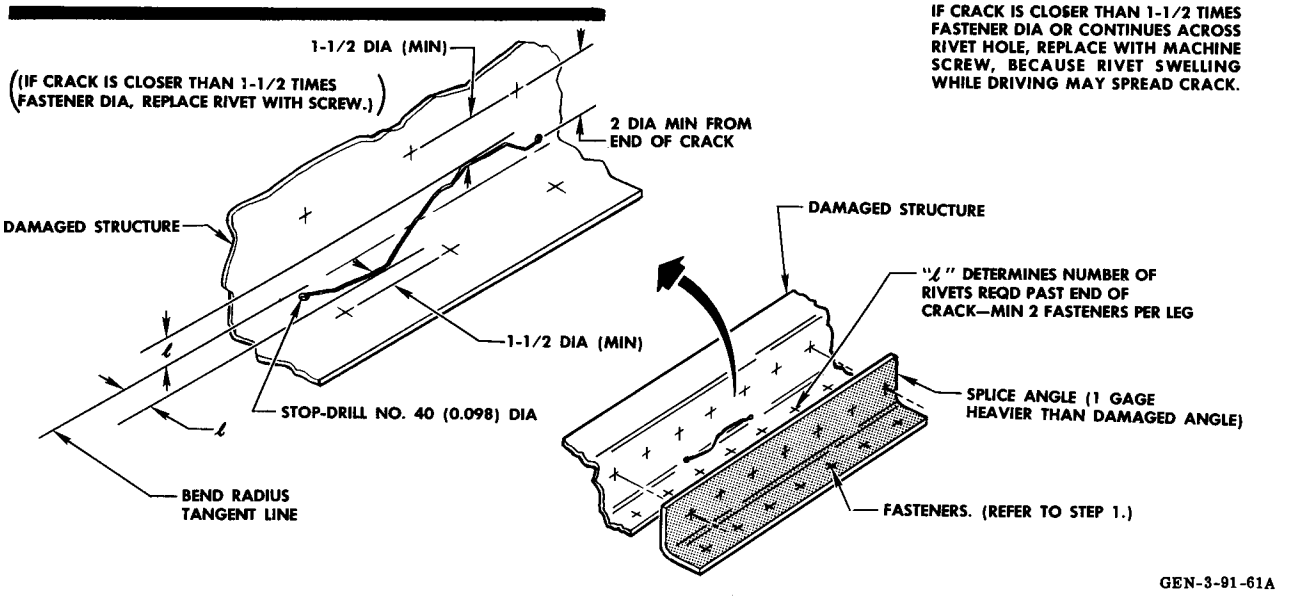


Figure 10-39. Sheet-metal Shapes and Extrusions—Crack Repairs (Sheet 1 of 2)

CRACKS—SHAPES AND EXTRUSIONS

TYPICAL REPAIR

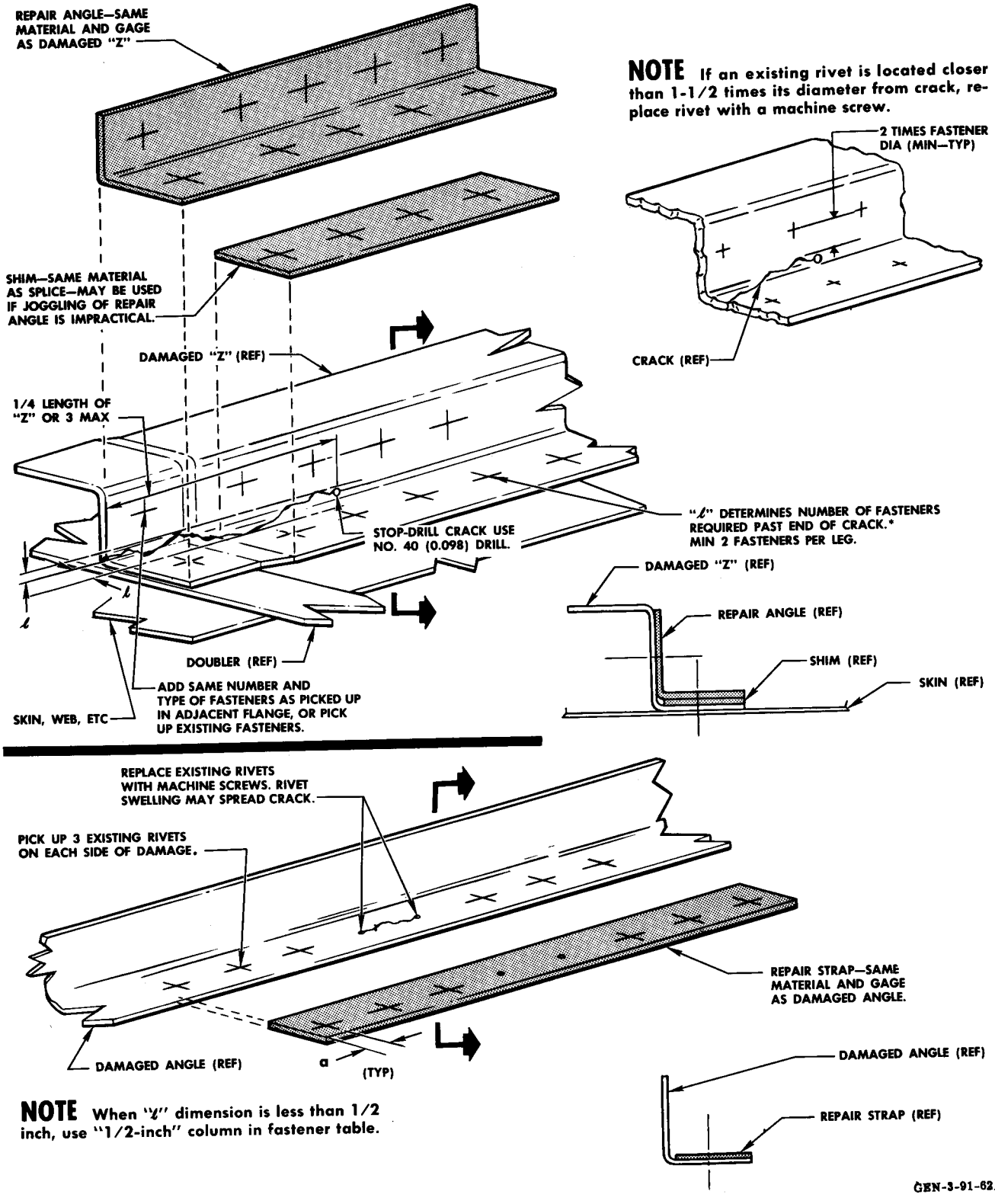
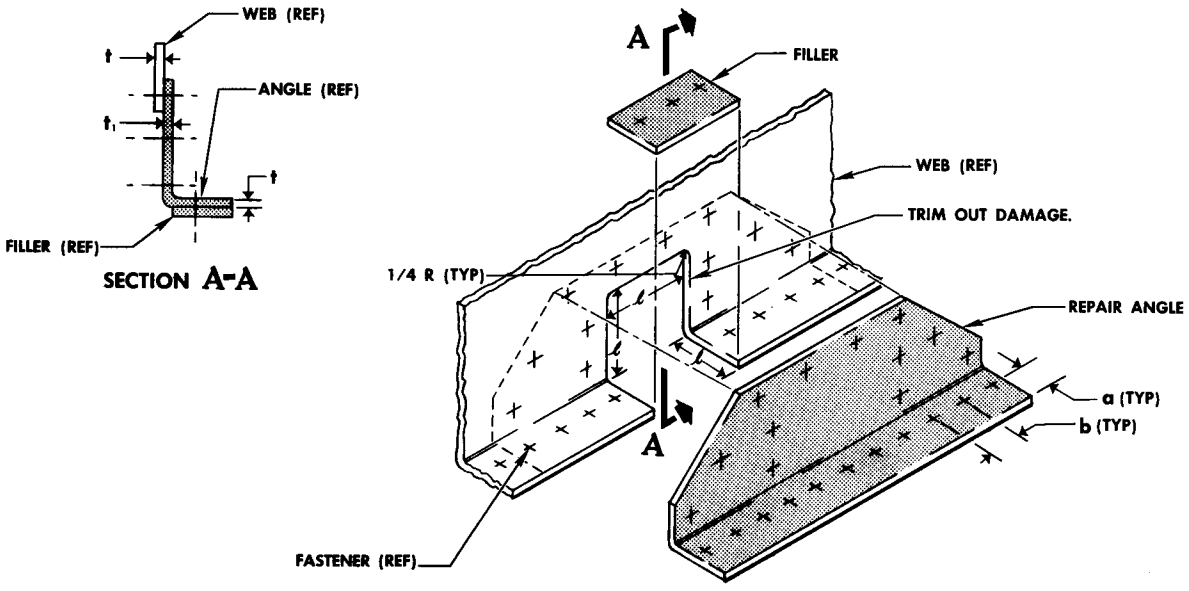


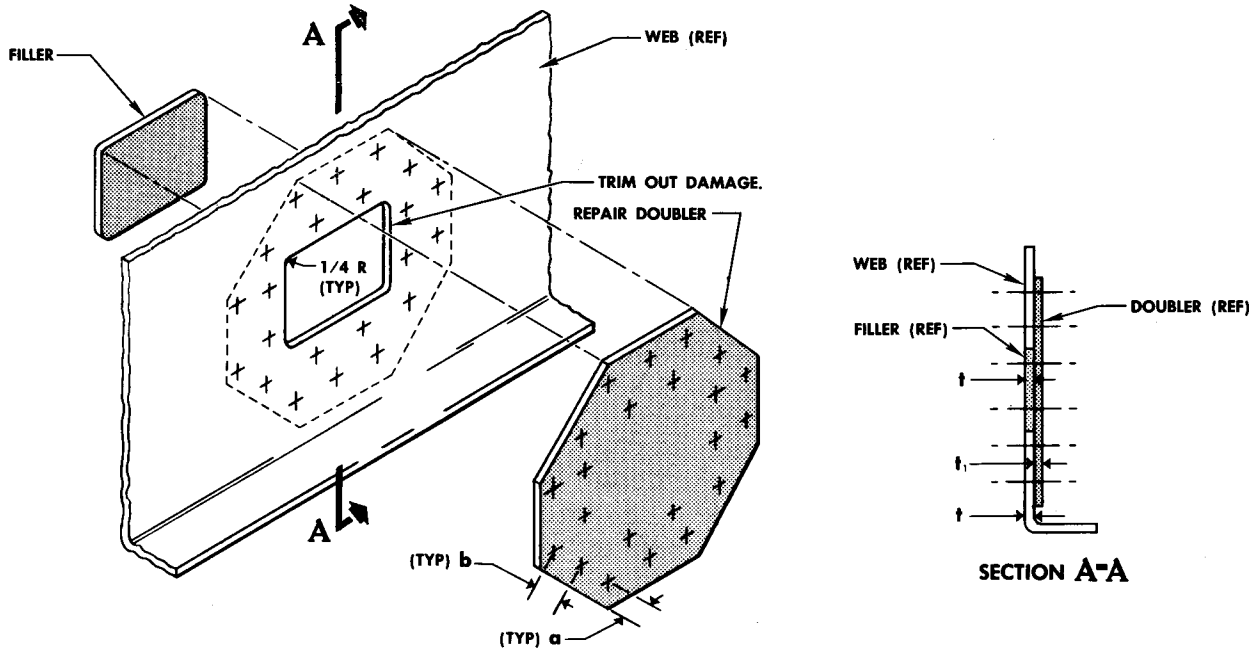
Figure 10-39. Sheet-metal Shapes and Extrusions—Crack Repairs (Sheet 2 of 2)

WEB AND FLANGE

TYPICAL REPAIR

1. Refer to "Developing a Typical Repair" before using this repair.



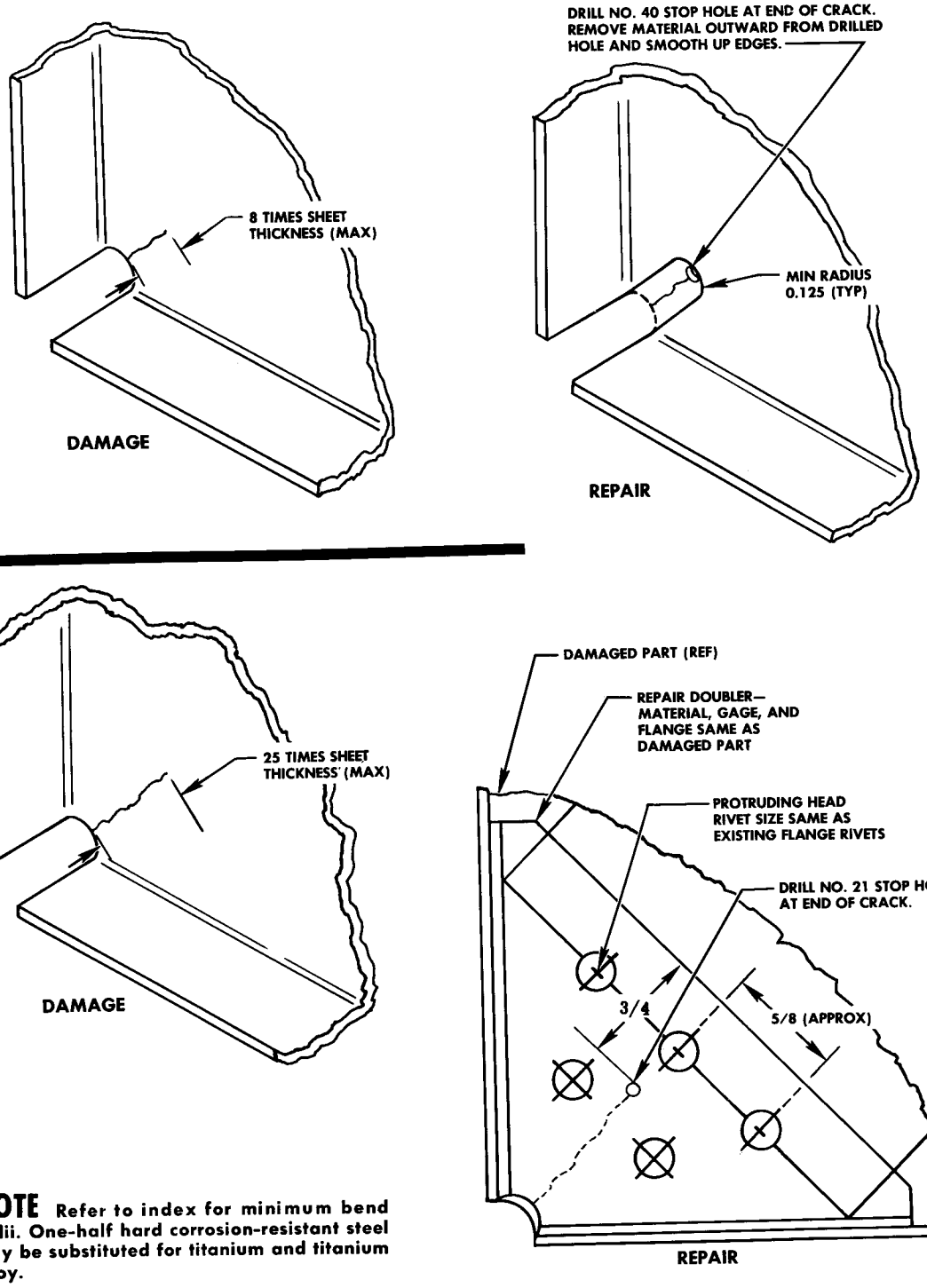


GEN-3-91-63A

Figure 10-40. Web and Web Flange Repair

DOUBLE FORMED FLANGES

TYPICAL REPAIR



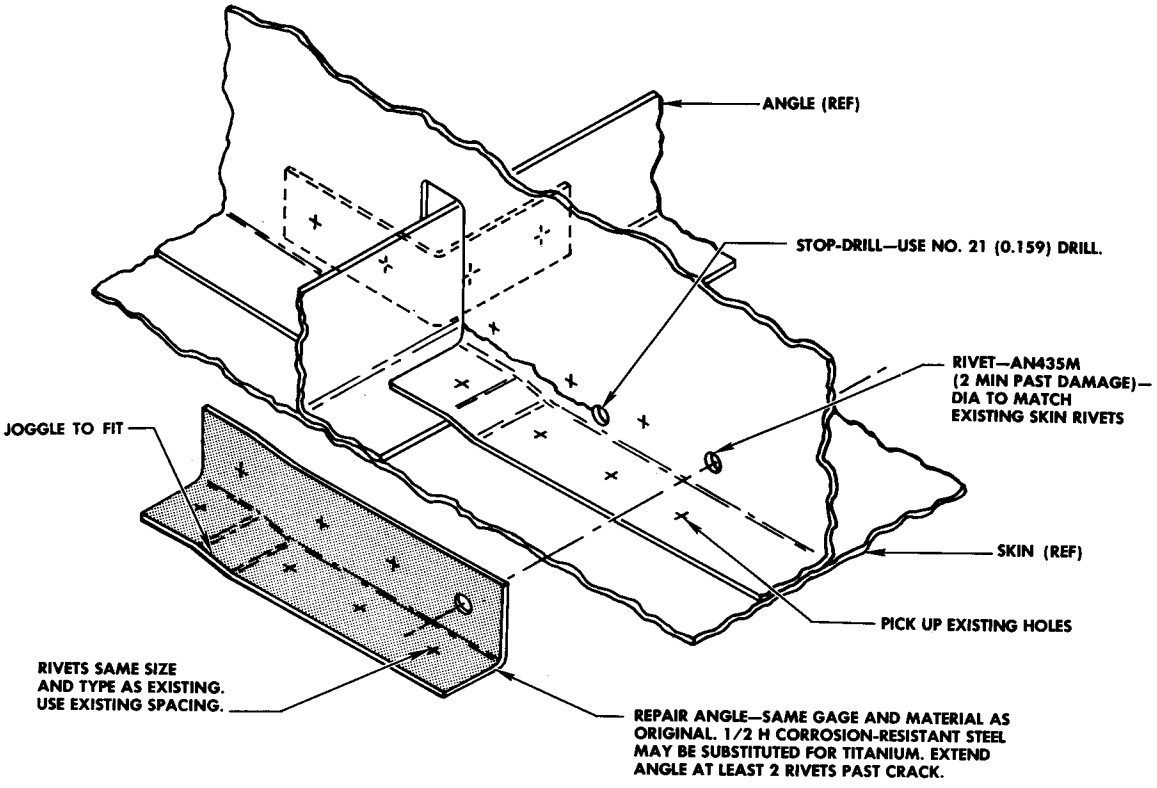
NOTE Refer to index for minimum bend radii. One-half hard corrosion-resistant steel may be substituted for titanium and titanium alloy.

GEN-3-91-69A

Figure 10-41. Double Formed Flange Repairs

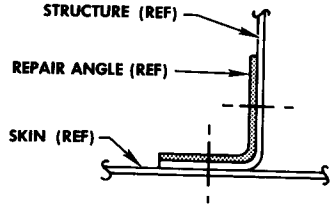
FORMED-FLANGE CRACKS

TYPICAL REPAIR



NOTE This repair is applicable to material of 0.063-inch maximum thickness.

- For minimum bend radii, refer to index.
- The minimum fastener edge distance is two times the fastener diameter.

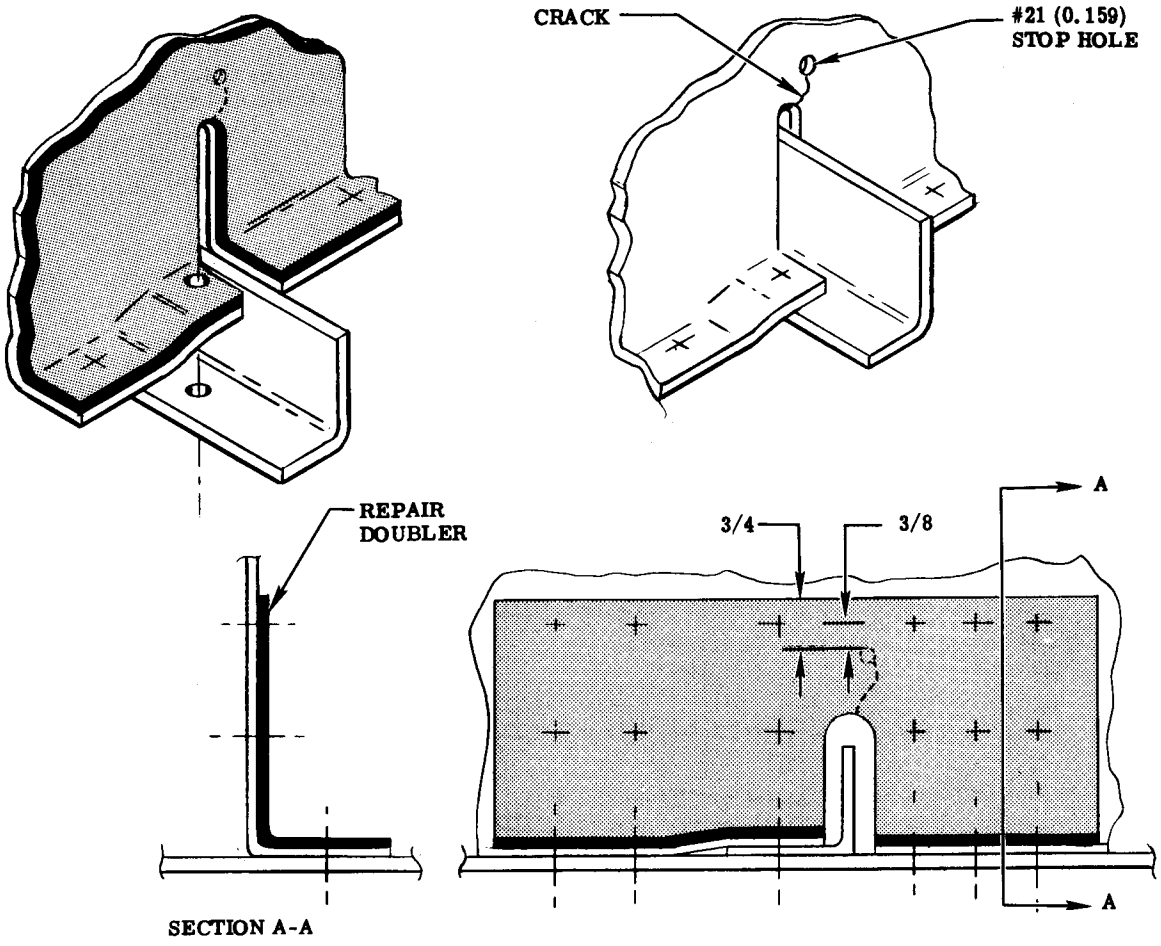


GEN-3-91-70A

Figure 10-42. Formed Flange Crack Repairs

WEB CRACK

TYPICAL REPAIR



For repair of cracks in formed members at stringer cutouts in flanges: The length of crack is limited to 20 times the sheet thickness. The repair may be made by using a flanged doubler as shown in sketch.

1. Drill No. 21 (0.159) stop hole at end of crack.
2. Flanged doubler is to be one gage heavier than web thickness. Flange on doubler is to be same size as flange on damaged part.
3. Doubler-to-web rivets are to be in two rows at same spacing as existing flange rivets.
4. Doubler is to pick up a minimum of three flange rivets on either side of cutout.
5. Doubler-to-web rivets are to be same size as existing flange rivets.

GEN-3-91-71

Figure 10-43. Damaged Stringer Cutout

ALUMINUM ALLOY SHAPES

TYPICAL REPAIR

- The typical repair methods shown apply to 2024 and 7075 extruded and roll-formed shapes.

- Two types of repairs are shown: angle (preferred) repairs and strap (alternate) repairs. Angles should be used when the bend radius permits the minimum fastener edge distance, when the fastener head is clear of the radius, and when necessary tooling and time for proper forming are available.

- The repair materials and the heat-treatment must match the part being repaired.

- Repair angle thicknesses are shown in sheet No. 2. Repair strap thicknesses are shown in the chart on this page. Repair strap thicknesses over 0.090 inch may be built up of two straps of approximately equal gages; however, their combined net gage thickness must be at least one gage thicker than the minimum single strap required.

- Symbols used in the chart and on the drawing are as follows:
  t — thickness of damaged part
  t_1 — thickness of *outside* repair strap
  t_2 — thickness of *inside* repair strap
  l — length of cut across damaged angle
  f — width of flange
  g — length of cut parallel to edge of flange.
  d — fastener diameter.

- Flanges to be repaired by the strap method must have the following minimum widths:

  0.025 in. through 0.040 in. material = 5/8 in.
  Material thicker than 0.040 in. = 3/4 in.

- The minimum fastener edge distance for all parts is 1-1/2 times the fastener diameter.

- Use the following fastener tables with these repairs:
  "Fastener Tables—Conventional and Hi-Shear, Flush, 2024-T4, Tension"
  "Fastener Tables—Conventional and Hi-Shear, Flush, 7075-T6, Tension"
  "Fastener Tables—Conventional, Protruding-head, 2024-T4 and 7075-T6, Tension"
  "Fastener Tables—Conventional, Protruding-head, Extrusions, Tension"

REPAIR STRAP THICKNESS CHART

	2024-T4 CLAD SH		2024-T4 EXTR		7075-T6 CLAD SH		7075-T6 EXTR	
t	t_1	t_2	t_1	t_2	t_1	t_2	t_1	t_2
0.025	0.040	0.040	0.040	0.040	0.050	0.040	0.050	0.040
0.032	0.050	0.050	0.050	0.050	0.063	0.050	0.063	0.050
0.040	0.063	0.063	0.063	0.063	0.071	0.063	0.071	0.063
0.050	0.071	0.071	0.071	0.071	0.100	0.090	0.100	0.090
0.063	0.090	0.090	0.090	0.090	0.125	0.100	0.125	0.100
0.071	0.100	0.125	0.100	0.125	0.160	0.125	0.160	0.125
0.080	0.125	0.160	0.125	0.160	0.160	0.125	0.160	0.125
0.090	0.160	0.190	0.160	0.190	0.160	0.160	0.160	0.160
0.100	0.160	0.250	0.160	0.250	0.160	0.250	0.160	0.250
0.125	0.250	0.250	0.160	0.250	0.190	0.250	0.190	0.250
0.160			0.190*	0.250			0.190*	0.250
0.190			0.250	0.250			0.250	0.250

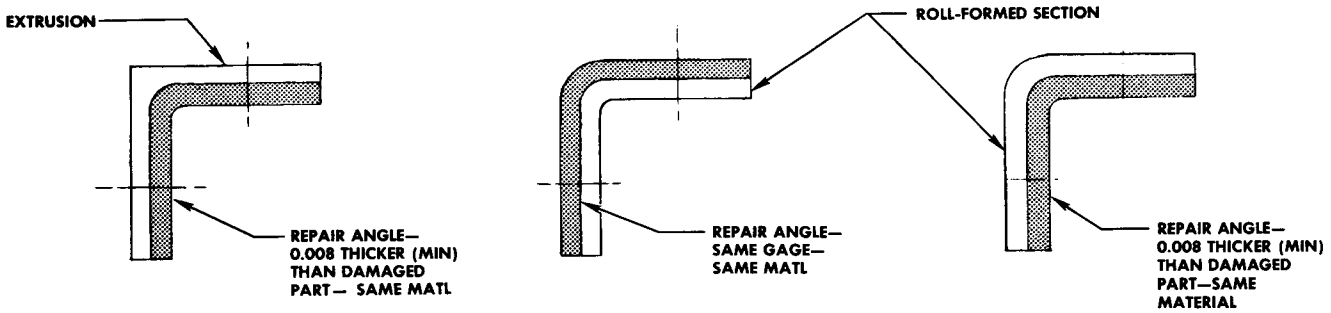
* If 0.190 gage is not available for strap, a strap built up from two sheets of 0.100-inch gage material may be used.

GEN-3-91-93A

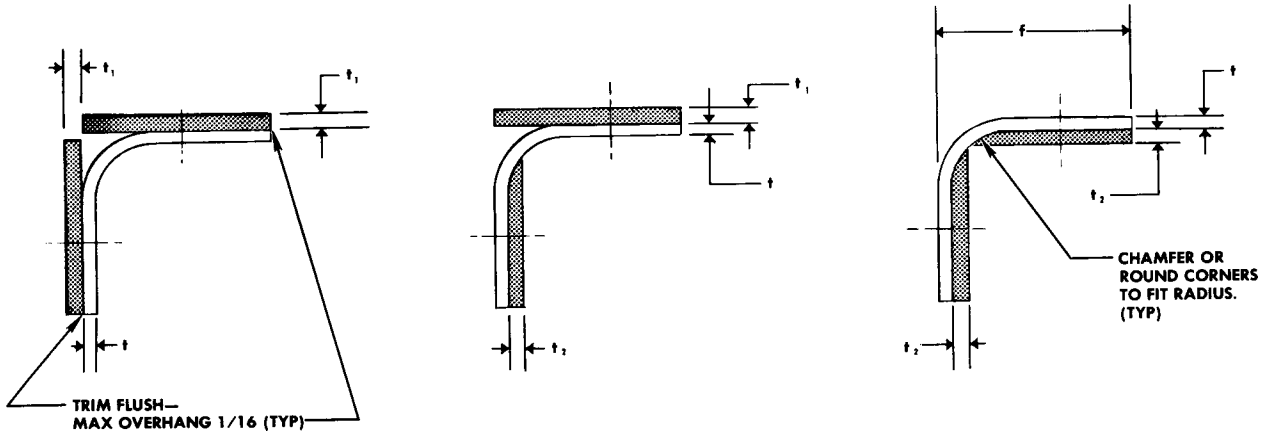
Figure 10-44. Angle Repairs (Sheet 1 of 3)

ALUMINUM ALLOY SHAPES

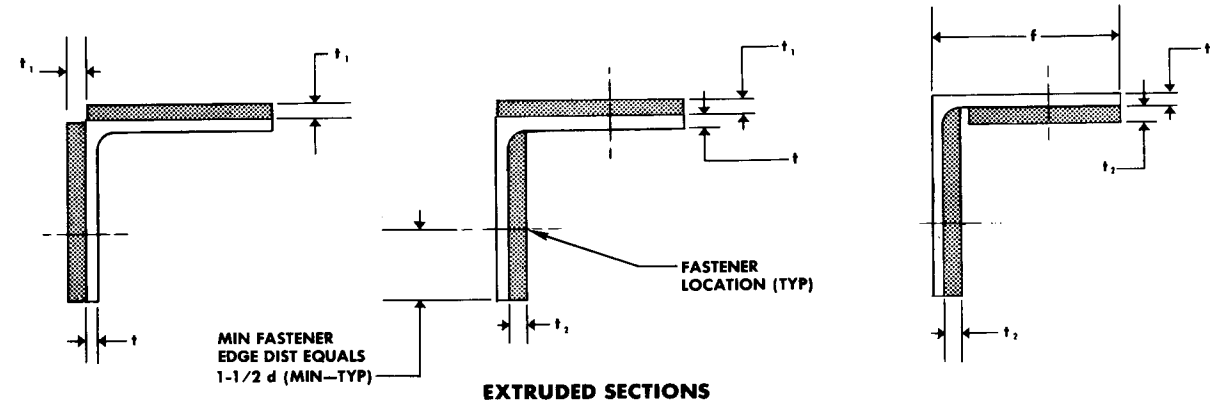
TYPICAL REPAIR



ANGLE (PREFERRED) REPAIRS



ROLL-FORMED SECTIONS



EXTRUDED SECTIONS

STRAP (ALTERNATE) REPAIRS

- t = thickness of damaged part.
- t_1 = thickness of outside repair strap.
- t_2 = thickness of inside repair strap.

DAMAGED PART

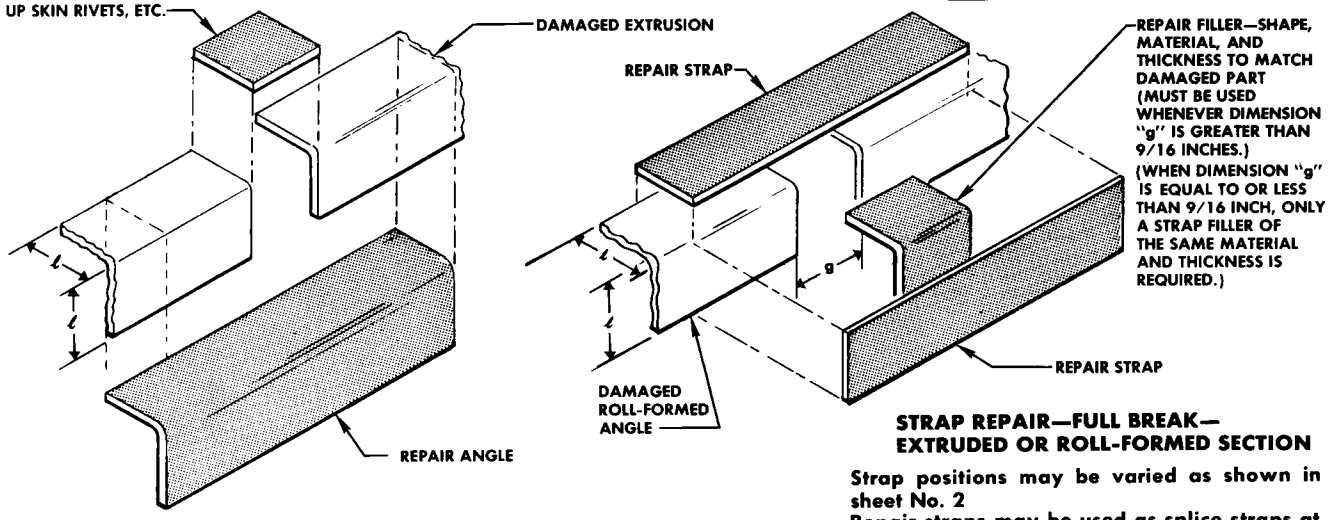
REPAIR STRAP

GEN-3-91-94A

Figure 10-44. Angle Repairs (Sheet 2 of 3)

ALUMINUM ALLOY SHAPES

TYPICAL REPAIR

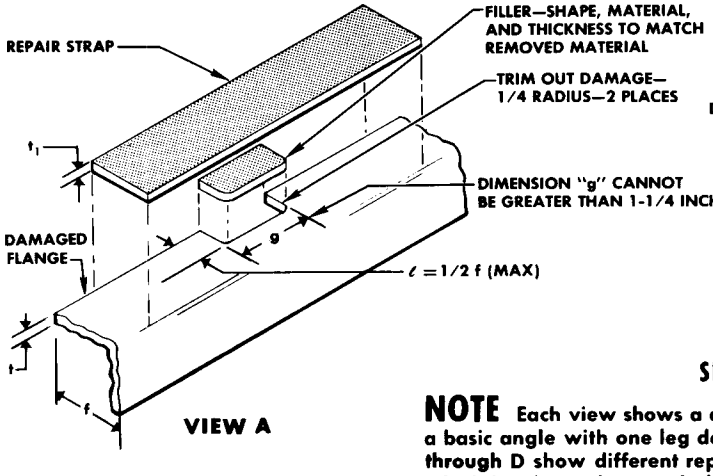


ANGLE (PREFERRED) REPAIR—FULL BREAK—EXTRUDED OR ROLL-FORMED SECTION

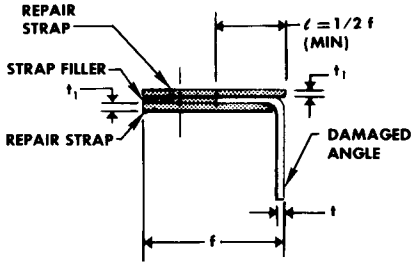
This repair should be used when (1) bend radius allows normal fastener installation and (2) when time and tooling are available for proper fabrication.

STRAP REPAIR—FULL BREAK—EXTRUDED OR ROLL-FORMED SECTION

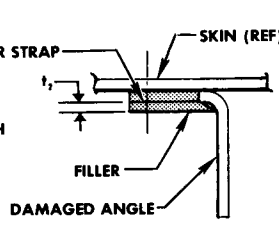
Strap positions may be varied as shown in sheet No. 2
Repair straps may be used as splice straps at each end of an insert member.

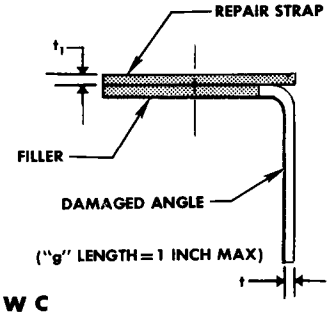


VIEW A

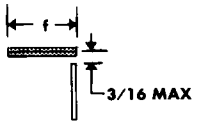


VIEW B

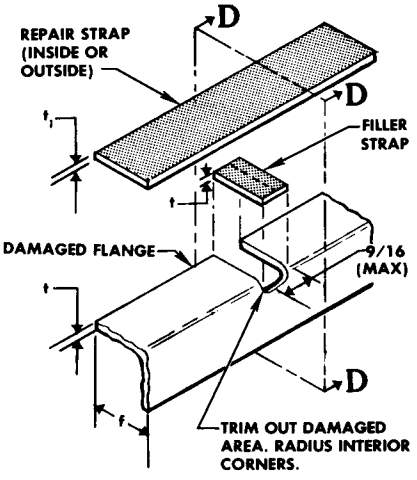




VIEW C



SECTION D–D



VIEW D

NOTE Each view shows a different repair for a basic angle with one leg damaged. Views A through D show different repairs for the angle which has been damaged along one leg only, the damage penetrating up to a depth of one half the length of the leg. View A is the repair for a maximum "g" length of 1-1/4 inches. View B is the repair for a minimum "g" length of 1-1/4 inches and a maximum "g" length of 3 inches. Views C and D show repairs for deeper leg damage. They differ in that the angle radius is not damaged in view C, while it is completely removed in view D. In view C, the maximum "g" length is one inch; in view D, the maximum "g" length is 9/16 inch. If both legs of an angle are damaged as shown in views C and D, and if this damage occurs in the same general area, cut out the damaged area and make repair as shown above (full break).

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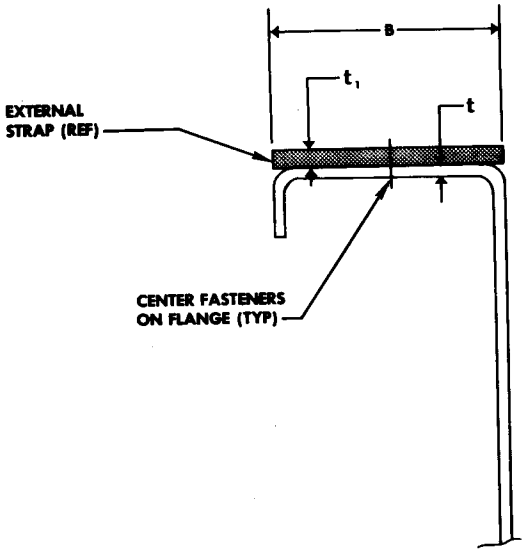
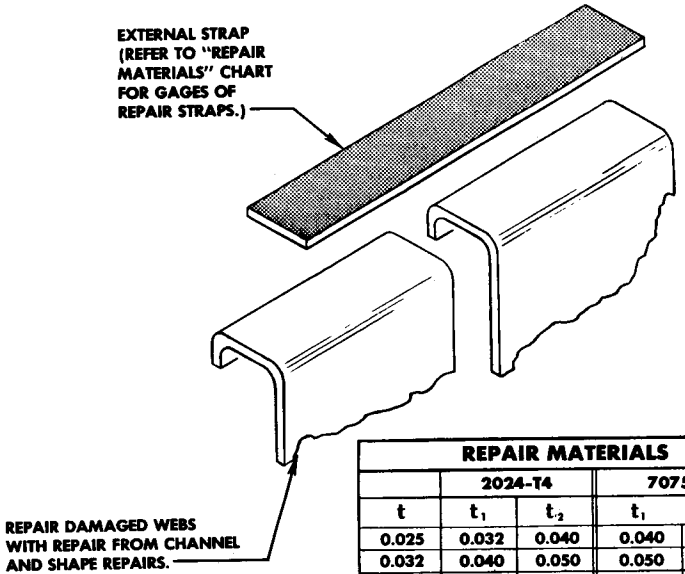
Figure 10-44. Angle Repairs (Sheet 3 of 3)

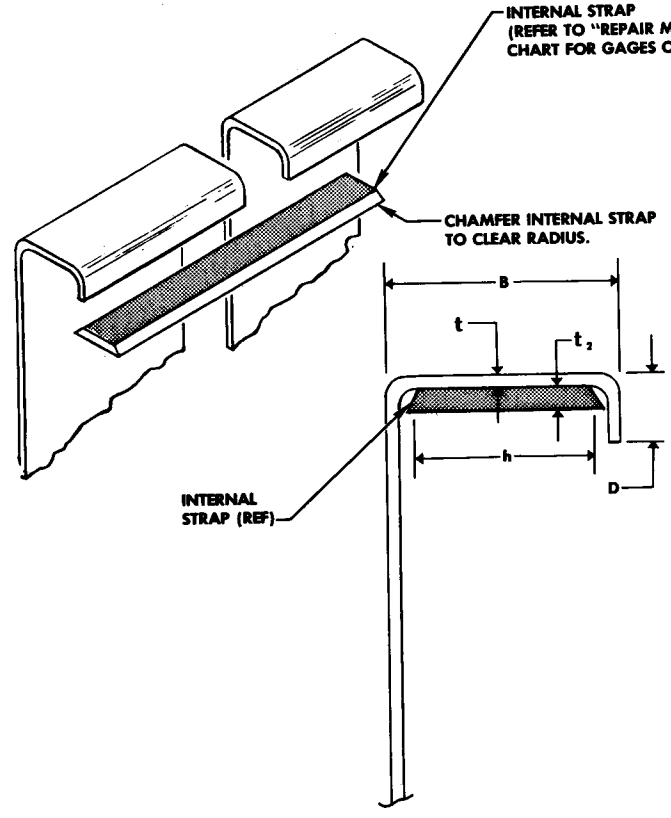
ALUMINUM ALLOY SHAPES

TYPICAL REPAIR





REPAIR MATERIALS				
	2024-T4		7075-T6	
t	t_1	t_2	t_1	t_2
0.025	0.032	0.040	0.040	0.050
0.032	0.040	0.050	0.050	0.063
0.040	0.050	0.071	0.063	0.071
0.050	0.071	0.100	0.080	0.090
0.063	0.090	0.125	0.100	0.125
0.071	0.100	0.190	0.125	0.125
0.080	0.125	0.250	0.160	0.160
0.090	0.160		0.160	0.190
0.100	0.160		0.160	0.190
0.125	0.250		0.250	0.250



SYMBOLS USED IN CHART AND ON ILLUSTRATIONS ARE AS FOLLOWS:

B is width of damaged flange, also width of external strap.
h is average width of internal strap (B-2t-R).
t is gage of damaged part.
t_1 is gage of external strap.
t_2 is gage of internal strap.

On 2024-T4 repairs, do not use internal strap repair when gage of damaged part exceeds 0.080 inch.

Do not use internal strap repair when flange width (B) of damaged part is less than 1-1/4 inch for 7075-T6, and one inch for 2024-T4.

Use the fastener tables given on sheets 2 and 3 for the type of repairs shown on this page.

Make repair straps from the same material as damaged part.

Length of repair straps is determined by the number of fasteners required.

When strap thickness is built up of more than one gage, use a maximum of two separate gages of about equal thickness, each equal to at least 2/3 the thickness of the required strap.

Maximum width of "D" is to be no greater than (2R+t) for sheet and no greater than (5t) for extrusion.

Maximum width of "B" is to be no greater than 3 inches.

GEN-3-91-98A

Figure 10-45. Angles and Flanges With Return Flanges (Sheet 1 of 3)

ALUMINUM ALLOY SHAPES

TYPICAL REPAIR

FASTENERS REQUIRED FOR 2024-T4 MATERIAL

MAT'L	t	STRAP MAT'L	1 INCH						1-1/2 INCHES						3 INCHES					
			AD4	AD5	DD6	DD8	3/16 HS	1/4 HS	AD4	AD5	DD6	DD8	3/16 HS	1/4 HS	AD4	AD5	DD6	DD8	3/16 HS	1/4 HS
TABLE 1	PROTRUDING HEAD																			
2024-T4	0.020	2024-T4	6	5	4	3	3	2	8	7	6	5	6	5	15	13	11	9	9	8
2024-T4	0.025	2024-T4	6	5	4	3	4	3	8	7	6	5	6	5	16	14	11	9	11	9
2024-T4	0.032	2024-T4	6	5	4	3	4	3	9	7	6	5	6	5	18	14	11	9	11	9
2024-T4	0.040	2024-T4	7	5	4	3	4	3	11	8	6	5	6	5	22	15	12	9	12	9
2024-T4	0.050	2024-T4	9	6	4	3	4	3	13	8	6	5	6	5	26	16	12	10	12	10
2024-T4	0.063	2024-T4	11	7	4	3	4	3	16	10	6	5	6	5	30	19	12	10	12	10
2024-T4	0.071	2024-T4	13	9	5	4	5	4	19	13	7	5	7	5	36	25	14	10	14	10
2024-T4	0.080	2024-T4	16	11	6	4	5	4	22	15	8	5	7	5	40	30	16	10	14	10
2024-T4	0.090	2024-T4	19	13	7	5	5	4	25	18	10	6	7	5		36	20	12	14	10
2024-T4	0.100	2024-T4	22	15	8	5	6	4	29	21	11	6	7	6			22	14	14	12
2024-T4	0.125	2024-T4	32	21	10	6	6	5	41	27	14	8	8	6			28	16	16	12
TABLE 2	COUNTERSUNK																			
2024-T4	0.032	2024-T4	6	5	4	3	3	3	10	8	7	6	6	5	19	15	13	11	11	9
2024-T4	0.040	2024-T4	8	6	4	3	4	3	12	9	7	6	6	5	23	17	13	11	11	9
2024-T4	0.050	2024-T4	10	6	5	4	4	3	14	10	7	6	6	5	28	19	14	12	12	9
2024-T4	0.063	2024-T4	12	8	5	4	4	3	17	12	7	6	6	5	34	23	14	12	12	9
2024-T4	0.071	2024-T4	15	10	6	5	5	4	20	14	8	6	7	5	40	28	16	12	14	10
2024-T4	0.080	2024-T4	17	12	7	5	5	4	23	16	9	6	7	5		32	18	12	14	10
2024-T4	0.090	2024-T4	20	14	8	6	5	4	26	19	10	7	7	5		38	20	14	14	10
2024-T4	0.100	2024-T4	24	16	9	6	5	4	30	21	11	8	7	6			22	16	14	11
2024-T4	0.125	2024-T4	32	21	11	7	6	5		27	14	9	8	6			28	18	16	12
2024-T4	0.160	2024-T4		29	15	9	8	7		37	19	12	10	8			38	23	20	16
TABLE 3	DIMPLED																			
2024-T4	0.020	2024-T4	5	4	3	3	6	6	6	5	4	4	7	7	12	9	7	7	13	13
2024-T4	0.025	2024-T4	5	4	3	3	5	6	6	5	4	4	6	6	13	10	8	7	11	11
2024-T4	0.032	2024-T4	5	4	3	3	4	5	7	5	4	4	6	6	14	11	9	8	11	12
2024-T4	0.040	2024-T4	6	4	3	3	3	4	8	6	4	4	5	5	16	12	9	8	9	10
2024-T4	0.050	2024-T4	7	5	4	3	4	4	10	7	5	4	5	5	19	14	10	9	10	9
2024-T4	0.063	2024-T4	9	6	4	3	4	3	13	9	5	4	5	4	25	17	11	9	10	9
2024-T4	0.071	2024-T4	11	7	4	4	4	3	15	10	6	5	6	4	30	20	12	10	11	9
2024-T4	0.080	2024-T4	13	8	5	4	5	3	18	12	7	5	7	4	36	23	13	10	13	9
2024-T4	0.090	2035-Y4	16	10	6	5	6	4	21	14	8	5	8	5		27	15	10	15	10

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Figure 10-45. Angles and Flanges With Return Flanges (Sheet 2 of 3)

ALUMINUM ALLOY SHAPES

TYPICAL REPAIR

FASTENERS REQUIRED FOR 7075-T6 MATERIAL

TABLE 4 PROTRUDING HEAD

MAT'L	t	STRAP MAT'L	1 INCH						1-1/2 INCHES						3 INCHES					
			AD4	AD5	DD6	DD8	3/16 HS	1/4 HS	AD4	AD5	DD6	DD8	3/16 HS	1/4 HS	AD4	AD5	DD6	DD8	3/16 HS	1/4 HS
7075-T6	0.020	7075-T6	4	4	3	3	4	3	6	5	4	4	5	4	12	10	8	7	10	8
7075-T6	0.025	7075-T6	6	5	3	3	4	3	9	7	5	4	5	4	18	13	9	7	10	8
7075-T6	0.032	7075-T6	8	6	4	3	4	3	11	8	5	4	5	4	22	15	10	7	10	8
7075-T6	0.040	7075-T6	11	7	4	3	4	3	15	10	6	4	5	4	28	19	10	8	10	8
7075-T6	0.050	7075-T6	14	9	5	4	4	3	19	13	7	5	6	5	35	25	13	9	10	8
7075-T6	0.063	7075-T6	18	12	6	4	5	4	24	16	9	5	6	5		30	15	9	11	8
7075-T6	0.071	7075-T6	20	13	7	5	5	4	27	18	10	6	6	5		34	18	11	11	8
7075-T6	0.080	7075-T6	23	15	8	5	5	4	31	20	11	6	6	5		38	20	12	11	8
7075-T6	0.090	7075-T6	26	17	9	6	5	4	35	23	12	7	6	5			22	13	11	8
7075-T6	0.100	7075-T6	28	19	10	6	5	4	39	25	13	8	6	5			24	13	11	8

TABLE 5 COUNTERSUNK

7075-T6	0.032	7075-T6	12	11	7	5	4	4	17	16	10	8	6	5	33	31	18	14	12	9
7075-T6	0.040	7075-T6	14	12	8	6	5	4	20	17	10	8	6	5	34	32	19	15	12	9
7075-T6	0.050	7075-T6	16	13	8	6	5	4	23	18	11	9	7	6	40	33	20	16	13	10
7075-T6	0.063	7075-T6	19	15	8	6	5	4	27	19	11	9	7	6		34	20	16	13	10
7075-T6	0.071	7075-T6	21	16	9	7	5	4	30	20	12	9	7	6		36	22	16	13	10
7075-T6	0.080	7075-T6	23	17	9	7	5	4	32	22	13	9	7	6		40	23	16	13	10
7075-T6	0.090	7075-T6	26	18	10	7	6	4	36	24	13	10	8	6			24	17	13	10
7075-T6	0.100	7075-T6	29	20	10	8	6	4	39	26	14	10	8	6			26	17	13	10
7075-T6	0.125	7075-T6	35	23	12	9	7	4		32	17	11	9	6			30	19	16	10
7075-T6	0.160	7075-T6		29	15	12	9	5			22	13	11	7			40	23	20	12

TABLE 6 DIMPLED

7075-T6	0.020	7075-T6	4	4	4	4	4	4	6	5	5	5	6	5	12	10	10	10	12	10
7075-T6	0.025	7075-T6	5	5	4	4	4	4	8	6	6	5	6	5	16	12	12	10	12	10
7075-T6	0.032	7075-T6	7	5	4	4	4	4	10	7	6	5	6	5	20	14	11	9	12	9
7075-T6	0.040	7075-T6	9	6	5	4	5	4	13	9	7	5	7	5	24	16	11	9	12	9
7075-T6	0.050	7075-T6	12	8	5	4	5	4	16	11	7	5	7	5	30	20	12	9	12	9
7075-T6	0.063	7075-T6	15	9	6	4	5	4	20	12	8	5	8	5	37	21	14	9	12	10
7075-T6	0.071	7075-T6	17	10	7	4	5	4	22	13	9	6	8	5	40	24	16	11	12	10
7075-T6	0.080	7075-T6	18	11	7	4	5	4	25	14	9	6	8	5		26	17	11	13	10
7075-T6	0.090	7075-T6	20	11	8	4	5	4	29	16	10	6	9	5		30	19	12	14	10

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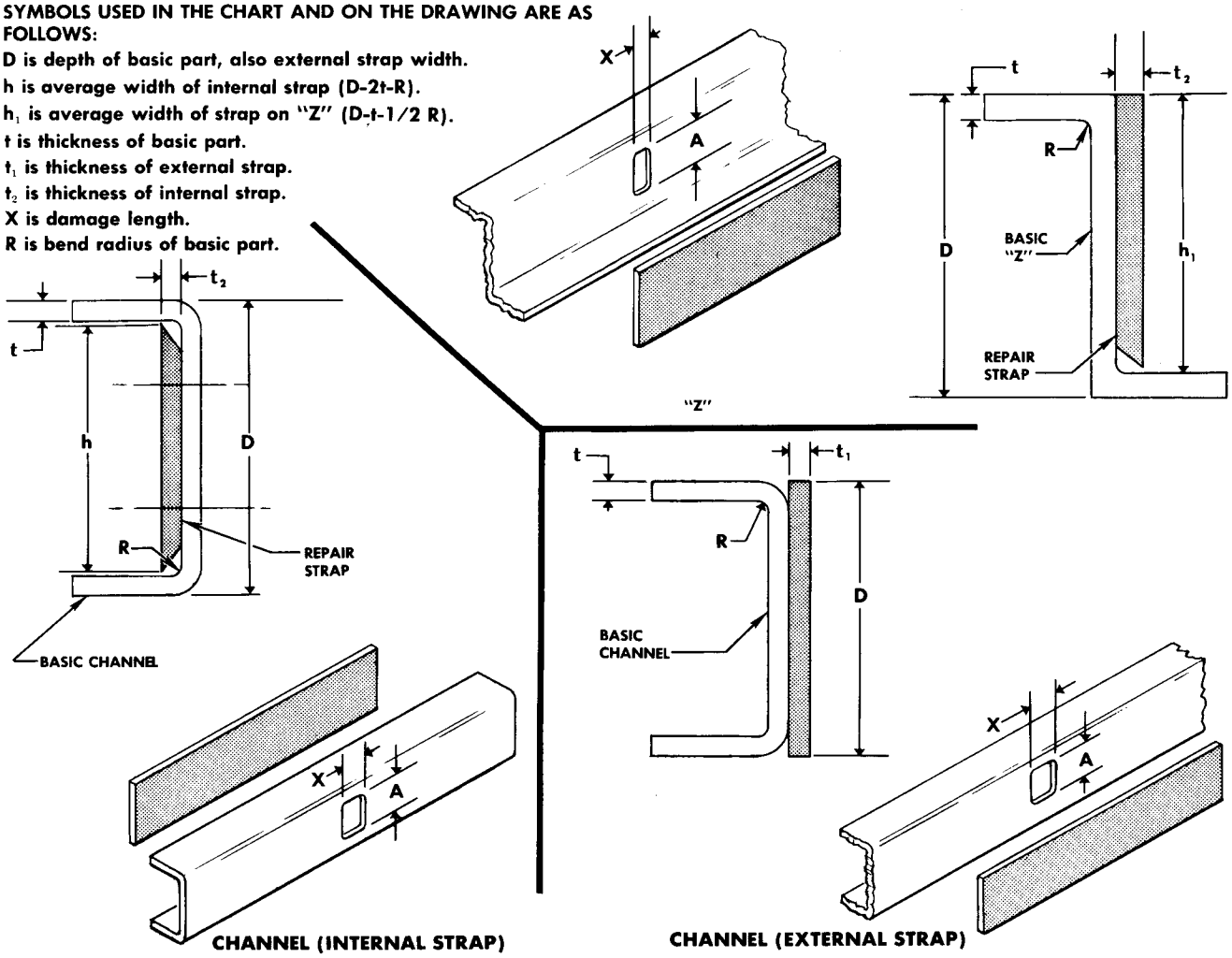
Figure 10-45. Angles and Flanges With Return Flanges (Sheet 3 of 3)

ALUMINUM ALLOY SHAPES

TYPICAL

SYMBOLS USED IN THE CHART AND ON THE DRAWING ARE AS FOLLOWS:

D is depth of basic part, also external strap width.
h is average width of internal strap (D-2t-R).
h_1 is average width of strap on "Z" (D-t-1/2 R).
t is thickness of basic part.
t_1 is thickness of external strap.
t_2 is thickness of internal strap.
X is damage length.
R is bend radius of basic part.



Use the following fastener tables from Section X with these repairs:

"Fastener Tables—Conventional and Hi-Shear, Flush, 2024-T4, Tension"

"Fastener Tables—Conventional and Hi-Shear, Flush, 7075-T6, Tension"

"Fastener Tables—Conventional, Protruding-head, 2024-T4 and 7075-T6, Tension"

"Fastener Tables—Conventional, Protruding-head, Extrusions, Tension"

If repair strap width is greater than one inch, use two or more rows of fasteners. The fasteners may be spaced either by staggering them or by placing them opposite one another, whichever is required, to maintain the correct number of fasteners with the proper edge distances.

When damage length (X) is less than 9/16 inch, use repair strap gages as shown in table. A filler is not required. When damage length (X) is more than 9/16 inch but less than 1-1/2 inches, use a repair strap 0.008 inch thicker than the gage required in the table. (The next standard gage may be used if it is thicker by 0.008 inch. A filler is not required.

When damage length (X) is greater than 1-1/2 inches, cut out the damaged area completely, and replace it with a new section. Fasten the new section in place with standard strap repairs. The new section is to have the same cross section, material, and gage as the original part.

REPAIR MATERIALS		
	2024-T4 or 7075-T6	
t	t_1	t_2
0.025	0.025	0.032
0.032	0.032	0.040
0.040	0.040	0.050
0.050	0.050	0.063
0.063	0.063	0.071
0.071	0.071	0.080
0.080	0.080	0.090
0.090	0.090	0.100
0.100	0.100	0.125
0.125	0.125	0.160
0.160	0.160	0.190
0.190	0.190	0.250

NOTE "A" equals the maximum damage width (D-2T-R), and the damage is not to extend into the fillet radius. If the damage extends into the fillet, repair as shown in sheet 2.

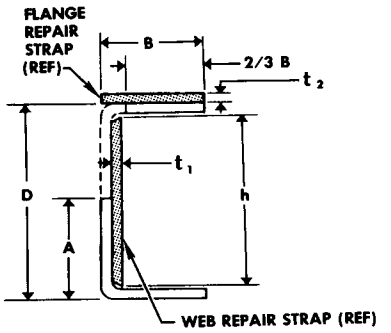
GEN 3-91-96

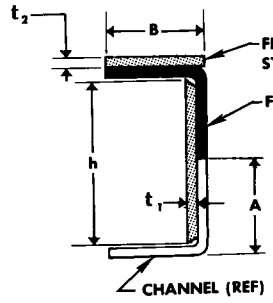
Figure 10-46. Flange and Web Strap Repairs (Sheet 1 of 2)

ALUMINUM ALLOY SHAPES

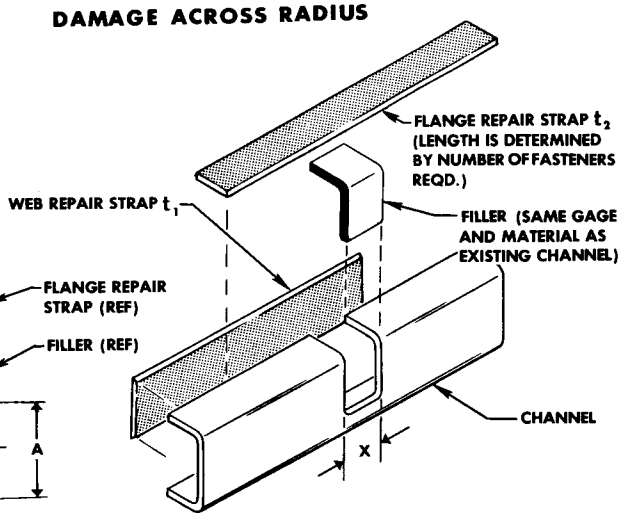
TYPICAL REPAIR

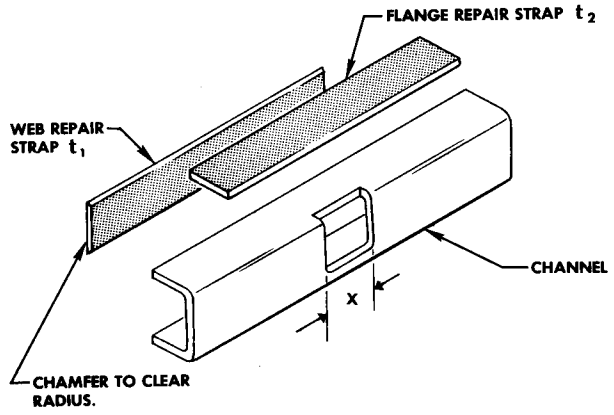
B is width of damaged flange, also width of external doubler.
h is average width of internal strap (D-2t-R).
t is gage of damaged part.
t_1 is gage of external straps.
t_2 is gage of internal straps.
D is depth of channel.
A = 1/2 D min. If less than 1/2 D, cut out web and both flanges and repair as a full section.





DAMAGE ACROSS RADIUS





REPAIR MATERIALS		
	2024-T4 or 7075-T6	
0.025	0.025	0.032
0.032	0.032	0.040
0.040	0.040	0.050
0.050	0.050	0.063
0.063	0.063	0.071
0.071	0.071	0.080
0.080	0.080	0.090
0.090	0.090	0.100
0.100	0.100	0.125
0.125	0.125	0.160
0.160	0.160	0.190
0.190	0.190	0.250

Use the following fastener tables from Section X for these repairs:
"Fastener Tables—Conventional and Hi-Shear, Flush, 2024-T4, Tension"
"Fastener Tables—Conventional and Hi-Shear, Flush, 7075-T6, Tension"
"Fastener Tables—Conventional, Protruding-head, 2024-T4 and 7075-T6, Tension"
"Fastener Tables—Conventional, Protruding-head, Extrusions, Tension"

If the damage occurs across the radius and is less than 9/16 inch in length, and if the damage extends less than 1/3 of the flange width, the *flange repair* strap may be equal to t_1. The *web strap* must be of the same thickness as that shown in the table. If the damage extends across more than 1/3 of the flange width, use the strap gages directly from the table.

NOTE When strap thickness is built up of material of more than one gage, use a maximum of two separate gages of about equal thickness, but each must be at least equal to 2/3 of the thickness of the required strap.

If the damage length (X) is more than 9/16 inch but less than 1-1/2 inches, and if the damage does not extend more than 1/3 of the distance across the flange, the flange repair strap is equal to t_1 and the web strap must be equal to t_2. A filler is not required. However, if the damage does extend more than 1/3 of the flange width, the damage must be cut out and a filler inserted. The tabulated strap repair gages must then be used on both the flange and the web.

If the damage length (X) is greater than 1-1/2 inches, cut out the damaged area completely, and replace with a suitable filler made to the same cross section as the original, and using the same material and gage as the original part. Fasten the new section in place using the standard strap repairs.

GEN 3-91-97

Figure 10-46. Flange and Web Strap Repairs (Sheet 2 of 2)

(All data deleted from page 10-72, pages 10-73 through 10-88 deleted, figures 10-47 through 10-58 deleted)

Section XI

ELECTROPLATING

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INTRODUCTION.

Stylus electroplating is a high current-density electrical method of depositing metals on any conductive surface without having to immerse the part in a plating bath. Unserviceable parts that may have oversize holes, or grooved surfaces, or loss of metal on any close-tolerance fit can be plated and brought back to the original tolerance for restoration as a serviceable item.

STYLUS ELECTROPLATING.

Stylus electroplating is a high-speed selective electroplating operation that can be accomplished with portable equipment. The actual technique more closely resembles arc welding than it does the more familiar bath plating. The necessary equipment consists of an electrical power source, an electrical rectifier, two power supply cables, a working tool called a stylus, and the necessary cleaning and plating solutions. The area to be plated is cleaned with trichloroethylene or Freon TF, and the surrounding area is protected with electroplater's tape.

The stylus consists of a handle attached to an anode (normally made of graphite). (See figure 11-2.) The anode is wrapped with an absorbent pad, held in place by surgical tube gauze, a spot tie, or rubber bands. (See figure 11-1.) A flexible cathodic cable (−) from the electrical rectifier is connected to the part to be plated, and a second flexible anodic cable (+) is connected to the stylus. The stylus anode is periodically dipped into the desired solution; then, by use of a circular motion, the solution is brushed over the area where the metal deposit is required.

This brushing operation completes the electrical circuit, and metal from the solution in the absorbent pad is deposited onto the surface being plated. No pressure is required; however, a constant motion is absolutely necessary to prevent burning or rough deposits. A constant, consistent pattern of movement must be followed to deposit an even amount of metal. Thickness of metal deposited is strictly a function of time and current density, controlled by an ampere-hour meter on the electrical rectifier.

PARTS REPARABLE BY STYLUS PLATING METHOD.

NOMENCLATURE	PART NO.
Wing outer panel slat tracks	180-17107 thru -17114
Horizontal stabilizer center section beam assembly	180-21403-101, -121
Engine forward right-hand trunnion locking support	192-31922
Aileron outboard surface control actuator rod	223-52321-3

WING OUTER PANEL SLAT TRACKS.

Slat tracks with nickel plating worn through on bearing surfaces may be restored to a serviceable condition by the stylus plating process. Plating limits and original drawing dimensions are shown in figure 11-3. Slat tracks are made of 4140 steel with a heat treat of 180,000 to 200,000 psi. The following materials, equipment, and tools are necessary to stylus-plate slat tracks.

ITEM	PART NO.	SUPPLIER
Anodes		Vanguard Pacific, Inc 1655 9th St. Santa Monica, Calif. 90404

ITEM	PART NO.	SUPPLIER
Electroplater's tape	Scotch No. 470	Minnesota Mining & Mfg Co 3M Center St. Paul, Minn. 55101
Cotton batting, USP, long-fiber, lint-free, surgical-grade		
Gauze, surgical tube		
Silicone carbide abrasive paper, grit 120 through 400		Vanguard Pacific, Inc
Filter paper	Whatman No. 41 and 41H	
Trichloroethylene		Shell Chemical Co 9901 S. Paramount Blvd. Downey, Calif.
Freon TF		Shell Chemical
Electroclean solution	SCM410	Vanguard Pacific, Inc
Activating No. 1 solution	SCM420	Vanguard Pacific, Inc
Nickel special solution	SPS563	Vanguard Pacific, Inc
Copper acid high-speed solution	SPS526	Vanguard Pacific, Inc
Nickel-tungsten solution	SAS570	Vanguard Pacific, Inc
Rectifier and accessories	10040	Vanguard Pacific, Inc
Arc gage		
Hand grinder		

PLATING BEARING SURFACES OF WING OUTER PANEL SLAT TRACKS.

1. Degrease fitting, using either of the following methods:

- Immerse in trichloroethylene vapors.
- Wipe with cloth soaked in Freon TF.

2. Remove pits from worn area with a hand grinder. Fair smooth to contour with 400 grit carborundum paper.

2A. Fluorescent-penetrant-inspect for cracks.

3. Mask area adjacent to surface to be plated, using electroplater's tape.

4. Choose suitably sized and shaped anode and handle for area that needs plating. (See figure 11-2.) Wrap each anode securely with cotton batting; then enclose in gauze tubing. (See figure 11-1.) Prepare a separate anode for each solution to be used.

Note

During cleaning, activating, and plating operations, always keep anode covering thoroughly saturated with fresh solution. When plating, move stylus continually in a circular motion. Never rest stylus on area being plated unless stylus is in motion.

5. Using Electroclean solution, clean area somewhat larger than area to be plated. Use 10 to 20 volts direct current for 15 to 60 seconds.

6. Rinse with clean, cold water. If there are water breaks, repeat step 5 and repeat rinse procedure.

7. Activate surface to be plated with activating No. 1 solution, using 10 to 18 volts dc for about 75 seconds. Do not rinse.

Note

To obtain best results, always follow manufacturer's instructions when using a cleaning, activating, or plating solution.

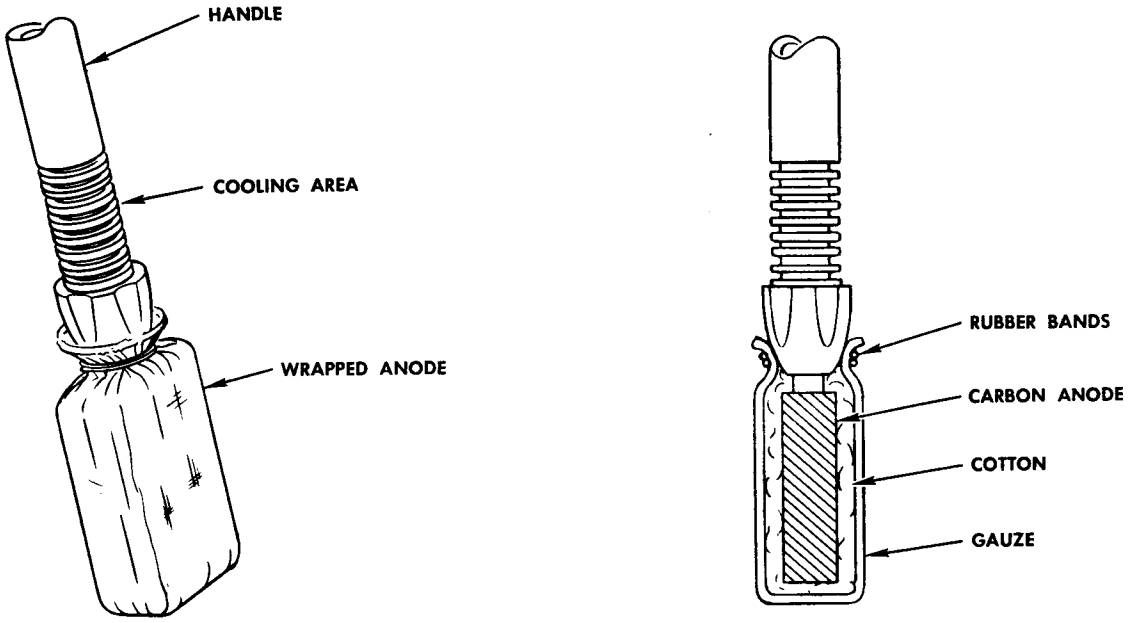
8. Plate immediately with nickel special solution to obtain a strike deposit of 0.00005 to 0.0001 inch; then rinse with clean, cold water. Use 10 to 25 volts dc.

Note

Use ampere-hour meter on rectifier for gaging plating thickness during plating. Use an adjustable arc gage on all slat tracks to determine plating thicknesses and as a guide for main-

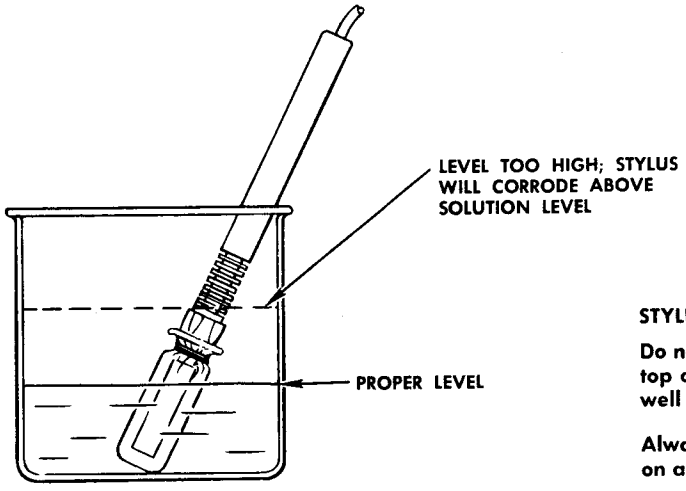
STYLUS PLATING

ELECTROPLATING



ANODE PREPARATION

Use a stylus with a carbon anode wrapped with surgical cotton. Cover cotton with Dynel or surgical gauze folded over and fastened in place with rubber bands. Make a neat, smooth-faced contact side.



STYLUS IN BEAKER

Do not let plating stylus remain in brush plating solution above top of carbon anode, as stylus will corrode and be ruined, as well as contaminate solution.

Always be sure stylus is rinsed out after each job and stored on a clean shelf or in a clean cabinet.

100A-3-91-27

Figure 11-1. Preparation of Stylus Prior to Plating Procedure

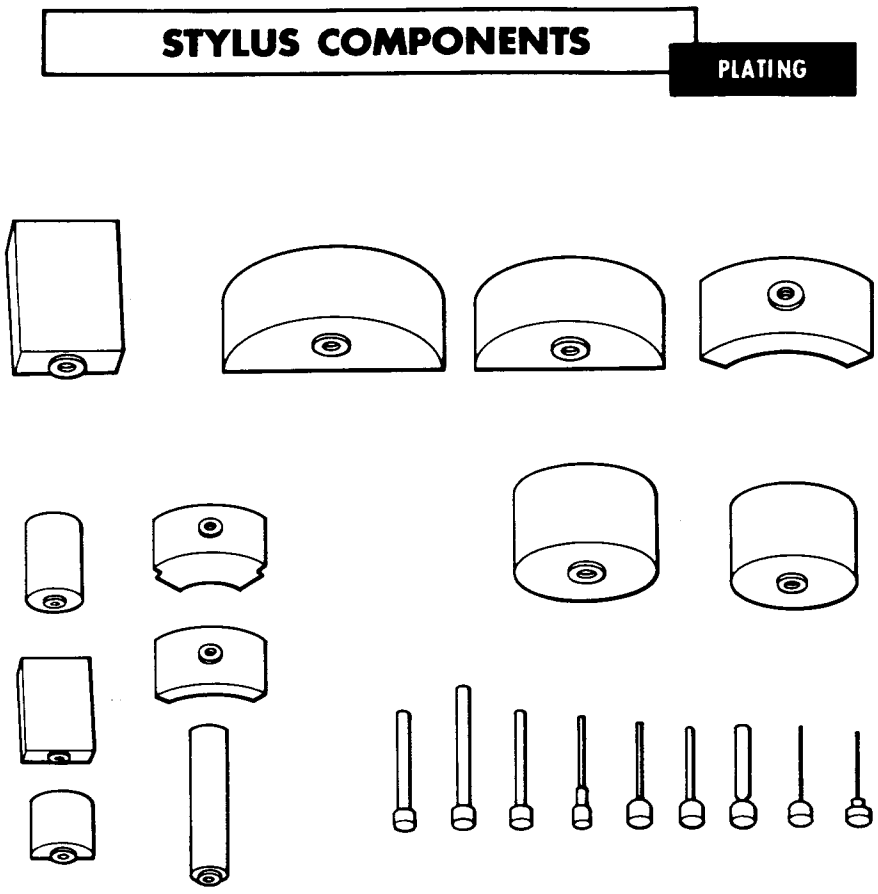


Figure 11-2. Stylus Components

taining arc of plated slat tracks to print specifications.

9. Plate immediately with copper acid high-speed solution at 3 to 25 volts dc to obtain a deposit about 0.001 inch thick; then rinse with clean, cold water. Measure plating thickness with arc gage.

10. Plate immediately with nickel special solution at 10 to 25 volts dc to obtain a deposit about 0.001 inch thick; then rinse with clean, cold water. Measure plating thickness with arc gage.

11. Build up worn area by alternately repeating steps 9 and 10 until area is about 0.001 inch higher than surrounding area. Let copper be the final deposit of the laminate.

12. Remove excess plating by hand-sanding with silicone carbide abrasive paper, grit 400. Sand until plated deposit has been smoothed and blended into surrounding surface.

13. Check that ID of slat track is 0.002 inch from finished dimension. (See figure 11-3.) Use arc gage to determine dimension.

14. Clean plated area with Electroclean solution. Repeat steps 5 and 6.

15. Deposit a strike coating of nickel, using nickel special solution, to a thickness of 0.00005 to 0.0001 inch; then rinse with clean, cold water.

16. Plate with copper acid high-speed solution to obtain a deposit about 0.001 inch thick; then rinse with clean, cold water.

17. Remove excess plating, if present, as noted in step 12. If excess plating has to be removed, repeat step 5.

18. Plate entire bearing area with nickel-tungsten solution at 8 to 15 volts dc to obtain proper ID. (See figure 11-3.) Use an arc gage to measure plating thickness.

19. Remove excess plating, if present, as noted in step 12.

21. Remove electroplater's masking tape.

22. Fluorescent-penetrant-inspect. Examine plating under 10 magnifications; there shall be no evidence of burning, stress, blisters, pits, nodules, porosity, excessive edge build-up, or other defects. Evidence of any such defect shall be cause for rejection.

22. Check that plating will not pull away when electroplater's tape is applied to plated surface with heavy hand pressure, then pulled away rapidly at right angles to plated surface. Any plate adhering to the tape will be cause for rejection of the stylus plating operation.

23. The plating shall meet the quality requirements of the following specifications:

SPECIFICATION	TITLE
T.O. 42C2-1-7	Metal Treatments
Fed Spec QQ-N-290	Nickel Plating (Electrodeposited)
Spec MIL-C-14550	Copper Plating (Electrodeposited)

HORIZONTAL STABILIZER CENTER SECTION BEAM ASSEMBLY.

Oversize bearing and bolt holes in horizontal stabilizer center section beams may be returned to proper dimensions by the stylus plating process. Plating limits and original drawing dimensions are shown in figure 11-4. The following materials, equipment, and tools are necessary to stylus-plate the beam assembly oversize holes.

ITEM	PART NO.	SUPPLIER
Anodes		Vanguard Pacific, Inc 1655 9th St. Santa Monica, Calif. 90404
Cotton batting, USP, long-fiber lint-free, surgical-grade		
Gauze, surgical tube		Vanguard Pacific, Inc
Electroplater's tape	Scotch No. 470	Minnesota Mining & Mfg Co, 3M Center St. Paul, Minn. 55101
Silicone carbide abrasive paper, grit 120 through 400		
Filter paper	Whatman No. 41 & 41H	
Electroclean solution	SCM410	Vanguard Pacific, Inc
Activating No. 1 solution	SCM420	Vanguard Pacific, Inc

ITEM	PART NO.	SUPPLIER
Activating No. 2 solution	SCM430	Vanguard Pacific, Inc
Nickel special solution	SPS563	Vanguard Pacific, Inc
Nickel neutral solution	SPS565	Vanguard Pacific, Inc
Cadmium low hydrogen embrittling (LHE) solution	SPS507	Vanguard Pacific, Inc
Trichloroethylene		Shell Chemical Co 9901 S. Paramount Blvd. Downey, Calif.
Freon TF		Shell Chemical Co
Rectifier and accessories	10040	Vanguard Pacific, Inc
Porcelain evaporating dishes		Vanguard Pacific, Inc

PLATING OVERSIZE HOLES IN HORIZONTAL STABILIZER CENTER SECTION BEAM ASSEMBLY.

1. Degrease fitting, using either of the following methods:

- Immerse in trichloroethylene vapors.
- Wipe with cloth soaked in Freon TF.

2. Remove oxide and corrosion products by vapor blasting or some other suitable means.

3. Mask area adjacent to surface to be plated, using electroplater's tape.

4. Choose suitably sized and shaped anode and handle for area that needs plating. (See figure 11-2.) Wrap each anode securely with cotton batting; then enclose in gauze tubing. (See figure 11-1.) Prepare a separate anode for each solution to be used.

Note

During cleaning, activating, and plating operation, always keep anode covering thoroughly saturated with fresh solution. When plating, move stylus continually in a circular motion. Never rest stylus on area being plated unless stylus is in motion.

5. Pour a quantity of each solution to be used into a separate dish or tray.

6. Clean area to be plated with a stylus saturated with Electroclean solution. Use 10 to 20 volts dc for 10 to 15 seconds.

7. Rinse with clean, cold water. If there are water breaks, repeat step 6 and repeat rinse procedure.

8. Plate each oversized hole with cadmium LHE solution. Use ampere-hour meter and a bore gage to determine amount of cadmium deposited. Use 8 to 20 volts dc and plate each affected hole as follows:

a. Plate crossbeam pivot bearing seat hole to obtain a deposit of 0.0015 inch thick maximum per side or until diameter of hole is 1.9989 (+0.0009, −0.00) inches, whichever is obtained first.

b. Place actuator attaching point bearing seat hole to obtain a deposit 0.0015 inch thick maximum per side or until diameter of hole is 2.3112 (+0.0010, −0.00) inches, whichever is obtained first.

c. Plate stabilizer attaching point holes to obtain a deposit 0.0015 inch thick maximum per side or until diameter of hole is 1.2809 (+0.0020, −0.00) inches, whichever is obtained first.

Note

If 0.0015 inch or less of cadmium plate brings diameter of plated hole to respective diameter specified in step 8, skip steps 9 through 13. If 0.0015 inch of cadmium plate does not bring hole to respective diameter specified in step 8, proceed with step 9.

9. Remove high spots or excess plating as necessary with no-load silicone carbide abrasive paper.

10. Clean with Electroclean solution, using 4 to 8 volts dc for 4 to 6 seconds; then rinse with clean, cold water. Keep surface wet until next step.

11. Plate immediately with nickel neutral solution. Use 10 to 25 volts dc and moderate stylus motion to obtain a deposit 0.0004 to 0.0005 inch thick per side. Use ampere-hour meter to determine amount of nickel deposited. Do not deposit more than 0.0005 inch of nickel neutral solution continuously. Rinse with clean, cold water. Keep surface wet until next step.

12. Plate immediately with cadmium LHE solution. Repeat step 8.

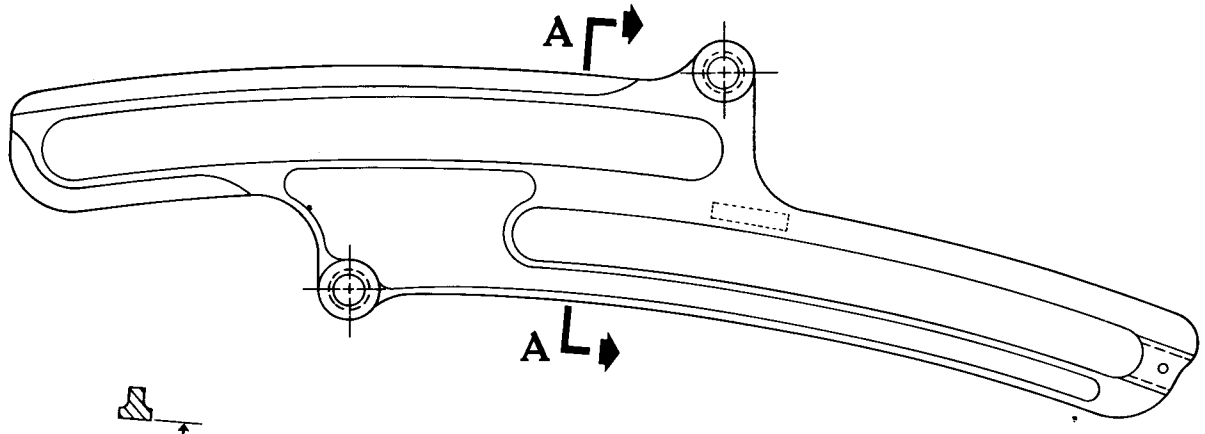
13. Continue cadmium-nickel laminate build-up by repeating steps 10 through 12 until diameter of respec-

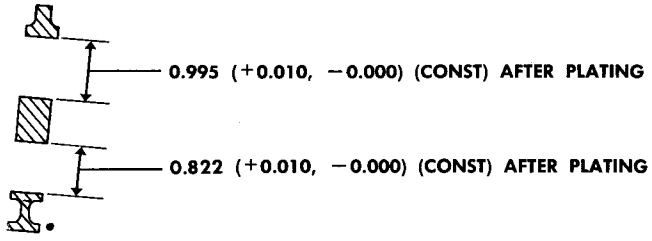
PERMANENT—CATEGORY B

WING

NOTE Refer to "Wing Outer Panel Slat Tracks" text for plating instructions.

- Maximum plating thickness for all areas is 0.015 inch.

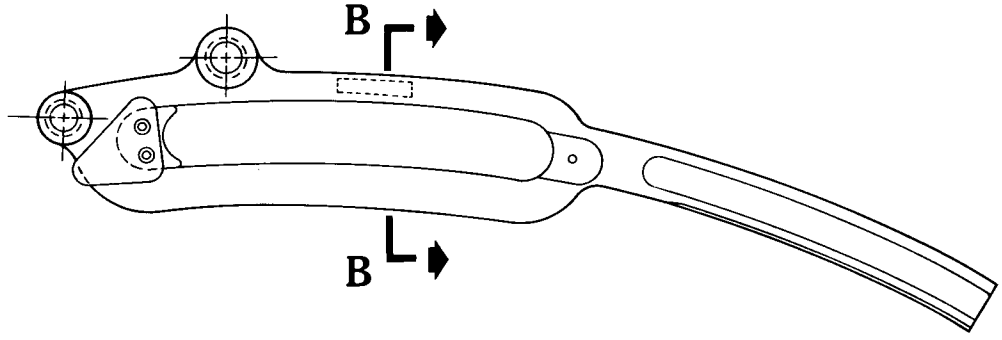


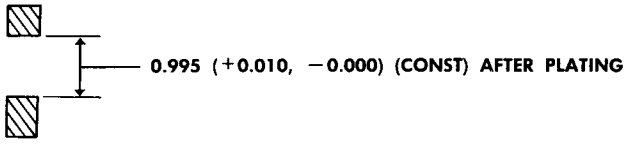


SECTION A-A

THIS VIEW AND DIMENSION REFLECT THE FOLLOWING SLAT TRACKS:

180—17107
180—17108
180—17109
180—17110
180—17111





SECTION B-B

THIS VIEW AND DIMENSION REFLECT THE FOLLOWING SLAT TRACKS:

180—17112
180—17113
180—17114

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Figure 11-3. Wing Outer Panel Slat Tracks

tive hole is as specified in step 8. Lay down laminate so that cadmium will be the final layer deposited.

14. Remove excess plating as necessary with no-load silicone carbide abrasive paper; then repeat step 10.

15. Complete plating with nickel neutral solution, using 10 to 25 volts dc to obtain a deposit 0.0003 inch thick per side. Rinse with clean, cold water; then measure diameter of holes with a bore gage. (See figure 11-4.) The diameter of the plated hole shall be as follows:

- Crossbeam pivot bearing seat hole, 1.9992 (+0.0009, −0.00) inches
- Actuator attaching point bearing seat hole, 2.3115 (+0.0010, −0.00) inches
- Stabilizer attaching point holes, 1.2810 (+0.0020, −0.00) inches

16. Remove electroplater's tape.

17. Examine plating under 10 magnifications; there shall be no evidence of burning, stress, blisters, pits, nodules, porosity, excessive edge build-up, or other defects. Evidence of any such defect shall be cause for rejection.

18. Check that plating will not pull away when electroplater's tape is applied to plated surface with heavy hand pressure, then pulled away rapidly at right angles to plated surface. Any plate adhering to the tape will be cause for rejection of the stylus plating operation.

19. The plating shall meet the quality requirements of the following specifications:

SPECIFICATION	TITLE
T.O. 42C2-1-7	Metal Treatments
Fed. Spec QQ-N-290	Nickel Plating (Electrodeposited)
Fed. Spec QQ-P-416	Cadmium (Electrodeposited)

ENGINE FORWARD RIGHT-HAND TRUNNION LOCKING SUPPORT.

Worn trunnion locking supports may be returned to proper dimensions by the stylus plating process. Allowable wear before plating is required, plating limits, and original drawing dimensions are shown in figure 11-5. Trunnions are made of 4140 steel with a heat treat of 160,000 to 180,000 psi. The following materials, equipment, and tools are necessary to stylus-plate the worn area.

ITEM	PART NO.	SUPPLIER
Anodes		Vanguard Pacific, Inc 1655 9th St. Santa Monica, Calif. 90404
Cotton batting, USP, long-fiber, lint-free, surgical grade		
Gauze, surgical tube		Vanguard Pacific, Inc
Electroplater's tape	Scotch No. 470	Minnesota Mining & Mfg Co 3M Center St. Paul, Minn. 55101
Silicone carbide abrasive paper, grit 120 through 400		
Filter paper	Whatman No. 41 & 41H	
Freon TF		Shell Chemical Co 9901 S. Paramount Blvd. Downey, Calif.
Electroclean solution	SCM410	Vanguard Pacific, Inc
Activating No. 1 solution	SCM420	Vanguard Pacific, Inc
Nickel neutral solution	SPS565	Vanguard Pacific, Inc
Copper acid high-speed solution	SPS526	Vanguard Pacific, Inc
Nickel-tungsten solution	SAS570	Vanguard Pacific, Inc
Tin alkaline solution	SPS595	Vanguard Pacific, Inc
Tin acid solution	SPS590	Vanguard Pacific, Inc

ITEM	PART NO.	SUPPLIER
Rectifier and accessories	10040	Vanguard Pacific, Inc
Glass beakers & flasks		Vanguard Pacific, Inc
Porcelain evaporating dishes		Vanguard Pacific, Inc
Abrasive dispersing unit	S. S. White Airbrasive unit, Model F, single-jet	Vanguard Pacific, Inc
Depth gage	67SCB6095	Sacramento Air Materiel Area

PLATING INSIDE DIAMETER OF ENGINE FORWARD TRUNNION LOCKING SUPPORT.

1. Remove excess oil and grease with absorbent towels or rags.

2. Degrease trunnion with Freon TF.

3. Mask area adjacent to surface to be plated, using electroplater's tape.

4. Choose suitably sized and shaped anode and handle for area that needs plating. (See figure 11-2.) Wrap each anode securely with cotton batting; then enclose in gauze tubing. (See figure 11-1.) Prepare a separate anode for each solution to be used.

Note

During cleaning, activating, and plating operations, always keep anode covering thoroughly saturated with fresh solution. When plating, move stylus continually in a circular motion. Never rest stylus on area being plated unless stylus is in motion.

5. Pour a quantity of each solution to be used into a separate dish or tray.

6. Clean area to be plated with a stylus saturated with Electroclean solution. Use 10 to 15 volts dc for 20 to 60 seconds.

7. Rinse with clean, cold water. If there are water breaks, repeat step 6 and repeat rinse procedure.

8. Plate immediately with nickel neutral solution. Use 10 to 25 volts dc to obtain a deposit about 0.0005 inch thick. Use a depth gage to measure plating thickness.

9. Plate immediately with copper acid high-speed solution, using 3 to 25 volts dc; then rinse with clean, cold water. Do not deposit more than 0.005 inch of copper continuously. Use a depth gage to measure plating thickness.

10. Laminate copper deposit by immediately plating with nickel neutral solution. Use 10 to 25 volts dc to obtain a deposit about 0.0005 inch thick. Use a depth gage to measure plating thickness.

11. Repeat steps 9 and 10 until ID of trunnion is 1.6880 to 1.6882 inches. Use a depth gage to measure plating thickness.

12. Sand copper-nickel laminate with silicone carbide abrasive paper until surface is smooth.

13. Clean or activate top deposit of copper-nickel laminate as follows:

a. Clean copper with Electroclean solution, using 8 to 12 volts dc for 15 to 30 seconds; then rinse with clean, cold water.

b. Activate nickel with activating No. 1 solution, using 6 to 12 volts dc for 20 to 120 seconds. Do not rinse. Flash coat immediately with nickel special solution, using 6 to 12 volts dc; then rinse with clean, cold water.

14. Plate with nickel-tungsten solution, using 8 to 15 volts dc until ID of trunnion is 1.6875 (+0.002, −0.0001) inches.

15. Activate nickel-tungsten deposit with activating No. 1 solution, using 6 to 12 volts dc for 20 to 120 seconds. Do not rinse.

16. Flash coat immediately with nickel special solution, using 6 to 20 volts dc. Rinse with clean, cold water.

17. Plate immediately with tin alkaline solution, using 8 to 25 volts dc or with tin acid solution at 5 to 15 volts dc. Plate until trunnion ID is about 1.6869 inches (0.0006 inch under listed tolerance). (See figure 11-5.)

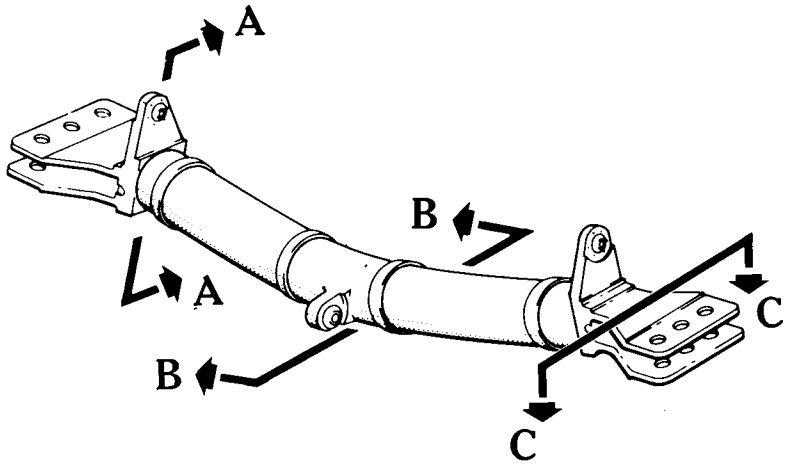
18. Rinse with clean, cold water and dry.

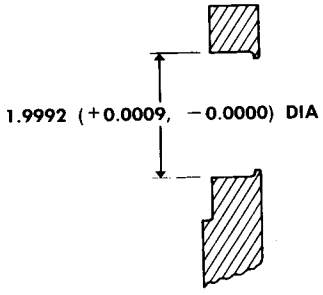
19. Remove electroplater's tape.

20. Examine plating under 10 magnifications; there shall be no evidence of burning, stress, blisters, pits, nodules, porosity, excessive edge build-up, or other defects. Evidence of any such defect shall be cause for rejection.

21. Check that plating will not pull away when electroplater's tape is applied to plated surface with heavy hand pressure, then pulled away rapidly at right angles to plated surface. Any plate adhering to the tape will be cause for rejection of the stylus plating operation.

PERMANENT—CATEGORY B **EMPENNAGE**

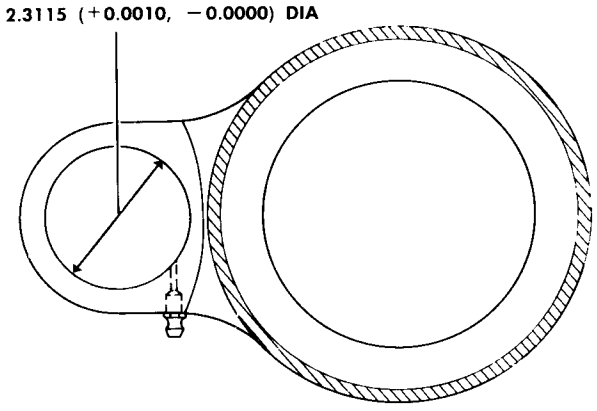




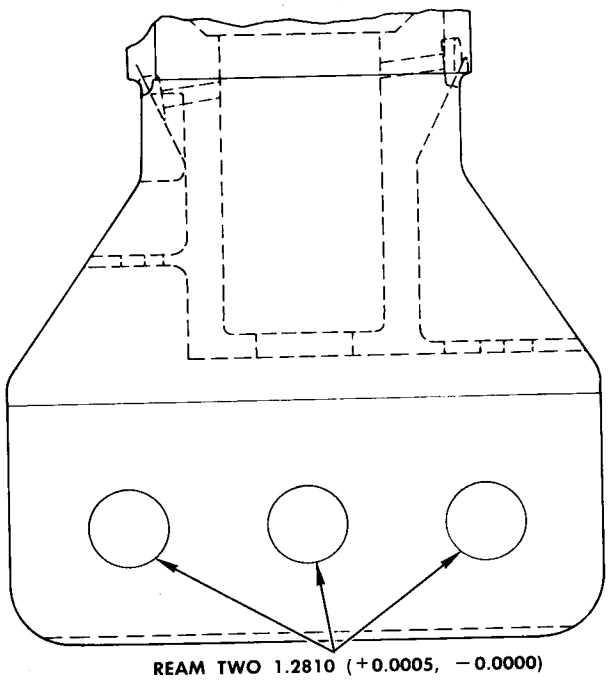
SECTION A-A

NOTE Refer to "Horizontal Stabilizer Center Section Beam Assembly" text for plating instructions.

- Maximum plating thickness for all holes is 0.015 inch.



SECTION B-B



REAM TWO 1.2810 (+0.0005, −0.0000) HOLES IN LINE, 6 PLACES TO MATCH.

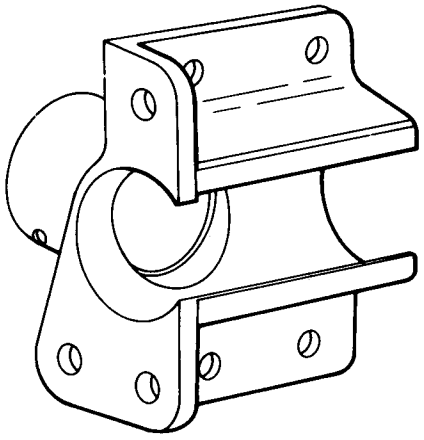
DETAIL C-C

100A-3-91-29A

Figure 11-4. Horizontal Stabilizer Center Section Beam Assembly

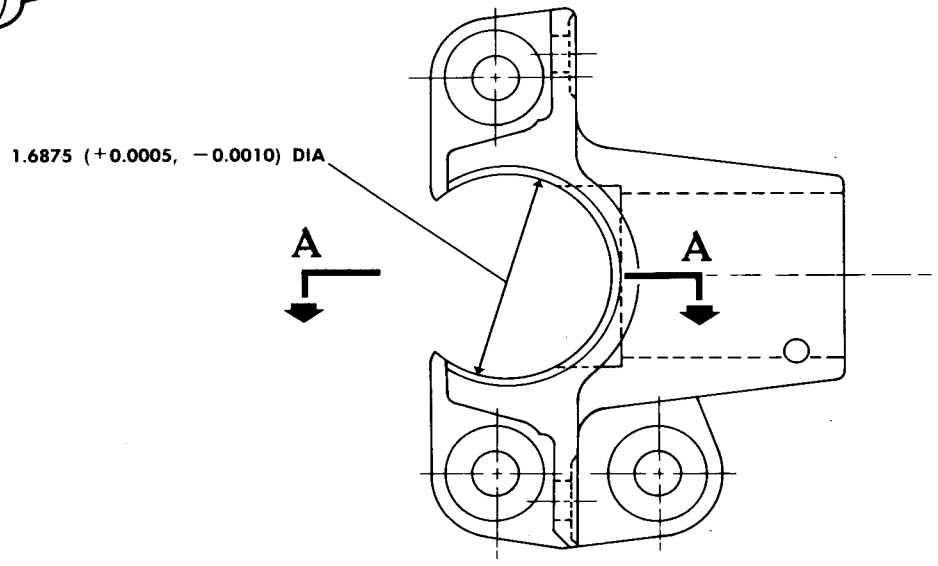
PERMANENT—CATEGORY B

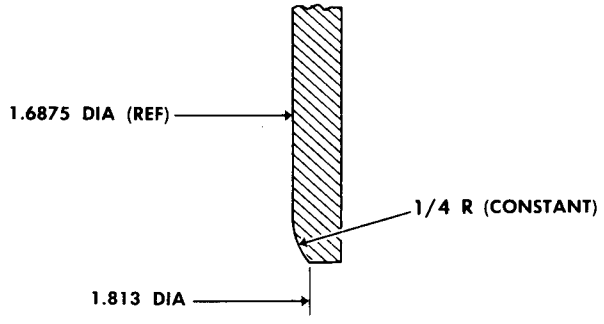
FUSELAGE



NOTE Wear limit at any point before plating is required is 0.005 inch.

- Refer to "Engine Right-hand Forward Trunnion Locking Support" text for plating instructions.
- Maximum plating thickness at any point is 0.040 inch.





SECTION A-A
TWICE SIZE

100A-3-91-30A

Figure 11-5. Engine Forward Right-hand Trunnion Locking Support

22. The plating shall meet the quality requirements of the following specifications:

SPECIFICATION	TITLE
T.O. 42C2-1-7	Metal Treatments
Fed Spec QQ-N-290	Nickel plating (Electrodeposited)
Spec MIL-T-10727	Tin Plating, Electrodeposited or Hot Dipped for Ferrous and Nonferrous Metals
Spec MIL-C-14550	Copper Plating (Electrodeposited)

AILERON SURFACE CONTROL OUTBOARD ACTUATOR ROD.

Some aileron surface control outboard actuator rods are not serviceable because they have oversize hinge and bearing holes. These fittings can be restored to a serviceable condition by the stylus plating process. Repair is limited to holes not more than 0.049 inch oversize, and stylus plating on an oversize hole is limited to 0.025 inch maximum thickness. Fittings with holes more than 0.049 inch oversize should be replaced. The following materials, equipment, and tools are necessary to stylus-plate this actuating rod.

ITEM	PART NO.	SUPPLIER
Anodes		Vanguard Pacific, Inc 1655 9th St. Santa Monica, Calif. 90404
Cotton batting, USP, long-fiber, lint-free, surgical-grade		
Gauze, surgical tube		Vanguard Pacific, Inc
Electroplater's tape	Scotch No. 470	Minnesota Mining & Mfg Co 3M Center St. Paul, Minn. 55101
Silicone carbide abrasive paper, grit 120 through 400		
Filter paper	Whatman No. 41 & 41H	
Electroclean solution	SCM410	Vanguard Pacific, Inc
Nickel neutral solution	SPS565	Vanguard Pacific, Inc
Nickel low-stress solution	5640	Vanguard Pacific, Inc
Tin alkaline solution	SPS595	Vanguard Pacific, Inc
Cadmium low hydrogen embrittling (LHE) solution	SPS507	Vanguard Pacific, Inc
Rectifier and accessories	10040	Vanguard Pacific, Inc
Porcelain evaporating dishes		Vanguard Pacific, Inc
Abrasive dispensing unit	S. S. White Airbrasive Model F, single-jet	Vanguard Pacific, Inc
Bore gage		
Glass beakers and flasks		Vanguard Pacific, Inc

PLATING HINGE HOLE.

1. Degrease fitting, using either of the following methods:

- Immerse in trichloroethylene vapors.
- Wipe with cloth soaked in Freon TF.

2. Remove oxide and corrosion, using either of the following methods:

- Vapor blasting.
- Alumina grit blasting with S. S. White Airbrasive unit.

3. Choose suitably sized and shaped anode and handle for hinge holes that need plating. (See figure 11-2.) Wrap each anode securely with cotton batting; then enclose in gauze tubing. (See figure 11-1.) Prepare a separate anode for each solution to be used.

Note

During cleaning, activating, and plating operations, always keep anode covering thoroughly saturated with fresh solution. When plating, move stylus continually in a circular motion. Never rest stylus on area being plated unless stylus is in motion.

4. Clean holes with Electroclean solution. Use 10 to 18 volts reverse current for 20 to 60 seconds.

5. Rinse with clean, cold water. If there are water breaks, repeat step 4 and repeat rinse procedure.

6. Using electroplater's tape, mask area adjacent to holes being plated, to prevent plating build-up on outer surfaces.

7. Deposit a strike coating of cadmium. Use cadmium LHE solution at 8 to 20 volts dc; then rinse with clean, cold water.

8. Plate with nickel neutral solution at 10 to 25 volts dc to obtain a deposit about 0.0005 inch thick. Use ampere-hour meter or bore gage to determine amount of plate deposited.

9. Plate with nickel low-stress solution at 6 to 12 volts dc to obtain a plate thickness of about 0.0005 inch; then rinse. Use ampere-hour meter or bore gage to determine amount of plate deposited.

10. Alternately repeat steps 7 and 8 until ID of hole is 0.5620 to 0.5622 inch.

11. Ream plated holes to 0.5625 (+0.0005, −0.000) inch. (See figure 11-6.)

12. Rinse with clean, cold water. Remove electroplater's tape.

13. Repaint to original color.

14. Examine plating under 10 magnifications; there shall be no evidence of burning, stress, blisters, pits, nodules, porosity, excessive edge build-up, or other defects. Evidence of any such defect shall be cause for rejection.

15. Check that plating will not pull away when electroplater's tape is applied to plated surface with heavy hand pressure, then pulled away rapidly at right angles to plated surface. Any plate adhering to the tape will be cause for rejection of the stylus plating operation.

16. The plating shall meet the quality requirements of the following:

SPECIFICATION	TITLE
T.O. 42C2-1-7	Metal Treatments
Fed. Spec QQ-N-290	Nickel Plating (Electrodeposited)
Fed. Spec QQ-P-416	Cadmium (Electrodeposited)
Spec MIL-T-10727	Tin Plating, Electrodeposited or Hot Dipped, for Ferrous and Nonferrous Metals

PLATING BEARING HOLES.

1. Degrease fitting, using either of the following methods:

- Immerse in trichloroethylene vapors.
- Wipe with cloth soaked in Freon TF.

2. Remove oxide and corrosion, using either of the following methods:

- Vapor blasting.
- Alumina grit blasting with S. S. White Airbrasive unit.

3. Choose suitably sized and shaped anode and handle for bearing holes that need plating. (See figure 11-2.) Wrap each anode securely with cotton batting; then enclose in gauze tubing. (See figure 11-1.) Prepare a separate anode for each solution to be used.

Note

During cleaning, activating, and plating operations, always keep anode covering thoroughly saturated with fresh solution. When plating, move stylus continually in a circular motion. Never rest stylus on area being plated unless stylus is in motion.

4. Clean holes with Electroclean solution. Use 10 to 18 volts reverse current for 20 to 60 seconds.

5. Rinse with clean, cold water. If there are water breaks, repeat step 4 and repeat rinse procedure.

6. Using electroplater's tape, mask area adjacent to holes being plated, to prevent plating build-up on outer surfaces.

7. Plate with cadmium LHE solution to bring ID of holes to 1.0625 inches; then rinse with clean, cold water. Measure plated hole with bore gage.

8. Remove excess plating, if any, with silicone carbide abrasive paper.

9. Clean cadmium-plated hole with Electroclean solution. Use 4 to 8 volts direct current for about 10 seconds. Rinse with clean, cold water.

PERMANENT—CATEGORY B **WING**

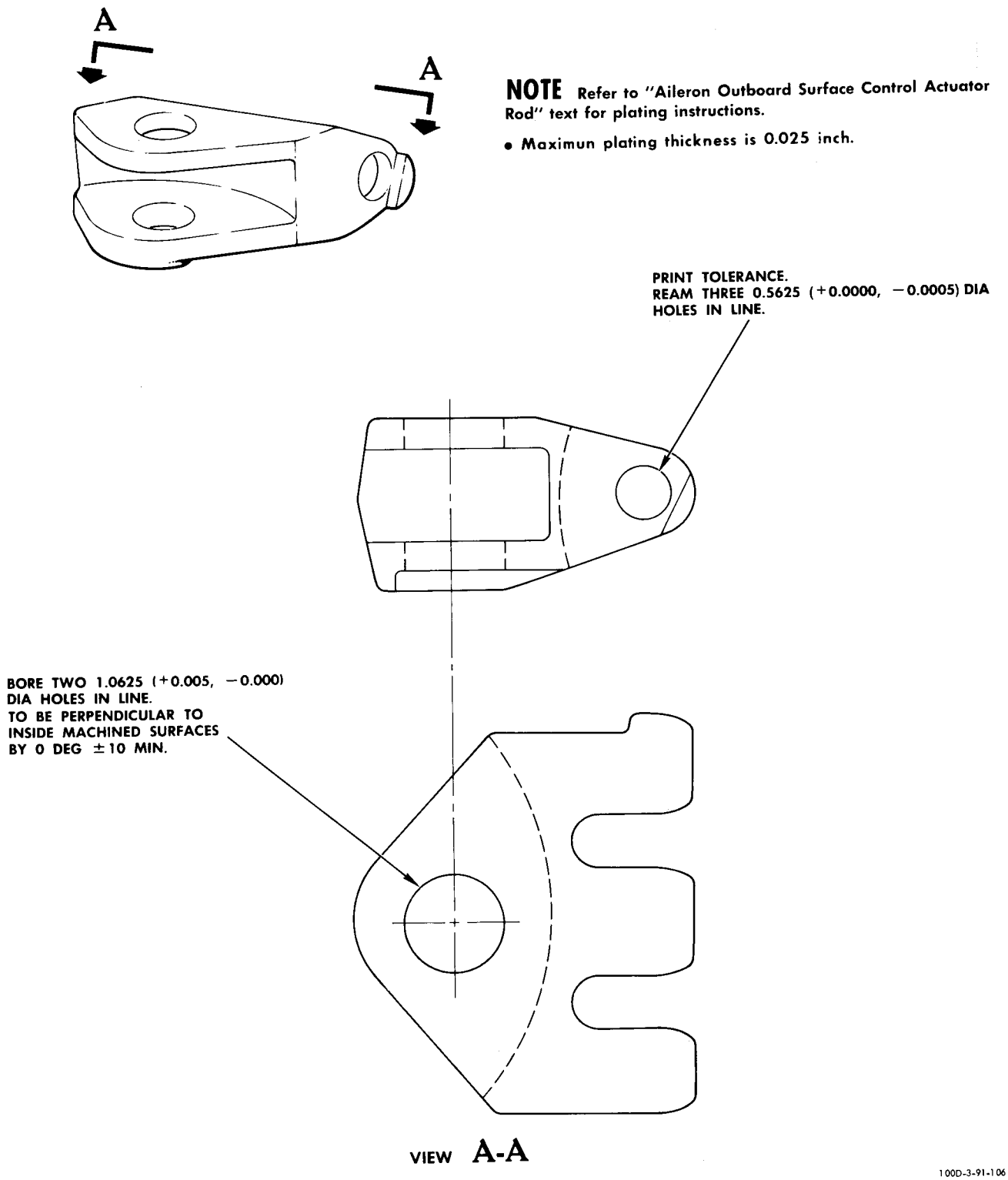


Figure 11-6. Aileron Surface Control Outboard Actuator Rod

10. Plate hole immediately with tin alkaline solution, using 10 to 20 volts direct current. Deposit enough tin so that there will be an excess of about 0.0003 inch per side beyond hole tolerance. (See figure 11-6.)

11. Rinse with clean, cold water and dry. Use bore gage to determine hole ID of 1.0619 inches for press fit.

Note

Do not deposit more than 0.025 inch of plating in any of the holes.

12. Remove electroplater's tape; then repaint to original color.

13. Examine plating under 10 magnifications; there shall be no evidence of burning, stress, blisters, pits, nodules, porosity, excessive edge build-up, or other defects. Evidence of any such defect shall be cause for rejection.

14. Check that plating will not pull away when electroplater's tape is applied to plated surface with heavy hand pressure, then pulled away rapidly at right angles to plated surface. Any plate adhering to the tape will be cause for rejecton of the stylus plating operation.

15. The plating shall meet the quality requirements of the following:

SPECIFICATION	TITLE
T.O. 42C2-1-7	Metal Treatments
Fed. Spec QQ-N-290	Nickel Plating (Electrodeposited)
Fed. Spec QQ-P-416	Cadmium (Electrodeposited)
Spec MIL-T-10727	Tin Plating, Electrodeposited or Hot Dipped for Ferrous Metals

Section XII
HEAT AND VENT DUCTING

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INTRODUCTION.

This section covers negligible (permissible) damage limits and repair of ducting in various airplane heating, cooling, and ventilation systems.

ENGINE BLEED-AIR HEAT AND VENT DUCTS.

The following instructions are for the visual inspection and repair of the sheathing, bellows, inner tubes, and flanges of the heat and vent ducts from the engine to the primary heat exchanger.

INSPECTION.

All hot section reparable ducts must be inspected as follows:

1. Pressure-check inner tube at 35 psig.
2. Pressure-check outer sheathing at 2 psig.
3. Check bellows for damage to guards and convoluted tubes.
4. Check that dents in guards do not restrict movement of flexible portion of bellows.
5. Check braided guards and determine that they do not exceed the following limits:
   - Any one plait may contain only two strands, provided the adjacent plaits contain six or more strands. More than one two-strand plait is allowed.
   - Any plait may contain four or more strands.
6. Dents, nicks, scratches, bends, and cracks in the outer layer of the convoluted bellows tubes are acceptable within the following limits:
   - Dents and bends must have a minimum radius of at least 0.020 inch. Dents and bends with less than a 0.020-inch radius are cause for rejection of the part.
   - Nicks and scratches occurring in an axial direction (±45 degrees) are acceptable if damage is less than 0.006 inch deep. All other nicks and scratches are cause for rejection.
   - Cracks are not acceptable.
7. Check flanges for nicks, dents, cracks, and/or distortion. All sealing edges must be free of nicks and dents, and flat within 0.015 inch.
8. Check inner tube for damage. If inner tube is dented, or if wall is gouged, scratched, or nicked, the part must be rejected.
9. For inspection of insulation, see figure 12-1.
10. Visually inspect for excessive and deep pitting caused by corrosion. Any pits 0.003 inch deep in the longitudinal direction or 0.005 inch deep in a direction perpendicular to the axis (±45°) that can not be removed by polishing without metal loss in excess of 0.005 inch, will be cause for rejection.

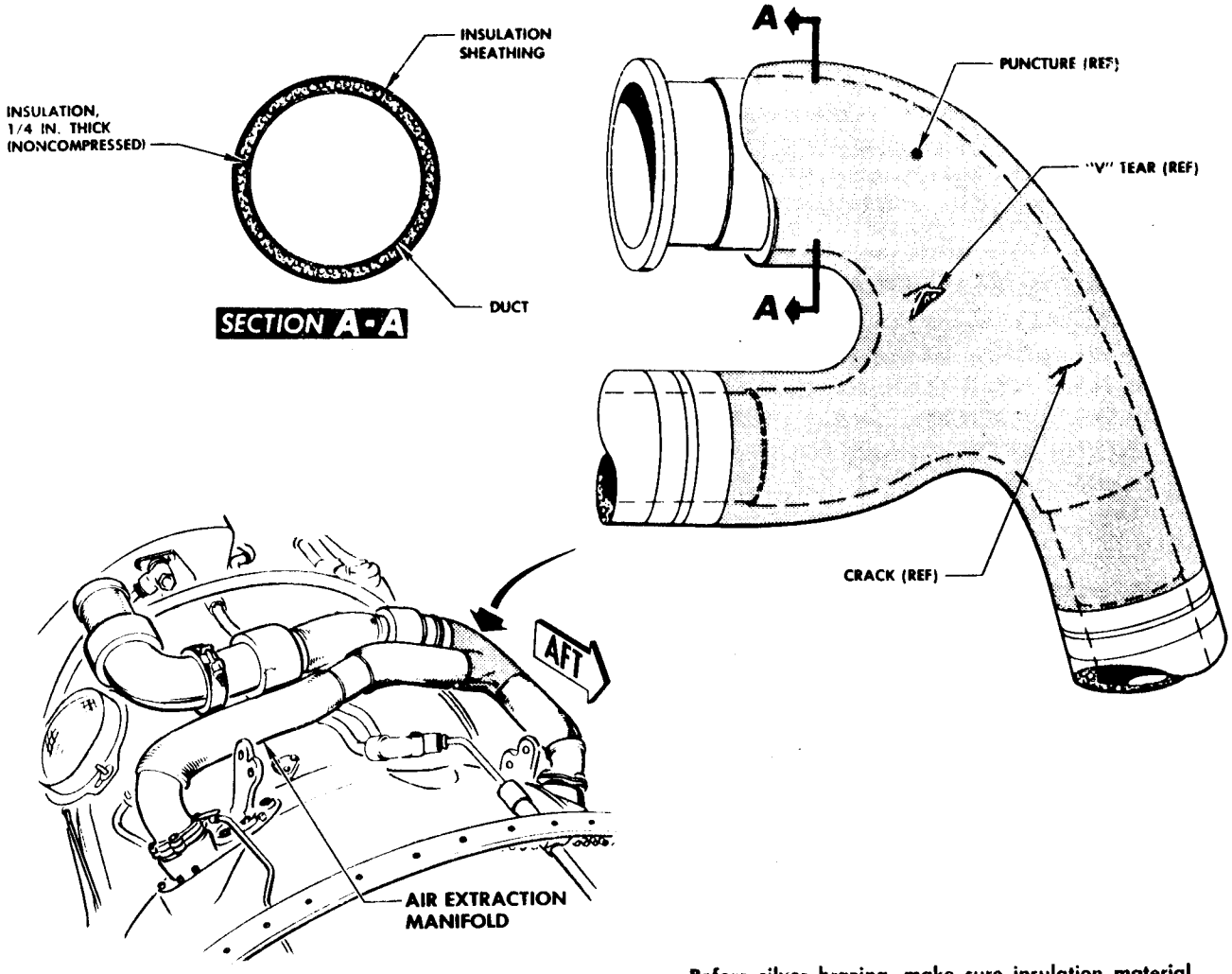
REPAIRS.

Repair of all reparable heat and vent ducts is to be accomplished using the following process specifications:

SPECIFICATION	PROCESS
MIL-W-8611	Fusion welding
MIL-W-6858	Resistance welding

PERMANENT - CATEGORY B

HEAT AND VENT



There are no limitations as to repair size on the insulation sheathing on 223-53549 and 243-953325 heat and vent ducts, provided the contour is maintained, and provided the repaired duct can withstand the 2 psi pressure check.

- Dents which do not show signs of cracking may be left as is.
- Punctures, cracks, and tears may be repaired by silver brazing according to Specification MIL-B-7883.

Before silver brazing, make sure insulation material is not parted to reveal outside wall of duct assembly. Caution should also be taken to prevent damage to the insulation material during the silver brazing procedure. To determine the adequacy of insulation when not visually exposed, flex outer cover. The maximum lateral movement should not exceed 1/8 inch and no metal-to-metal contact should be encountered. See Section A-A for material thickness.

Repair areas with missing sheathing and insulation by trimming away sheathing to allow replacement of insulation. Obtain replacement insulation from similar damaged part.* Repair sheathing by using a scab patch obtained from a similar damaged part and silver-soldering in place. Pressure-check repaired sheathing at 2 psig for 2 minutes.

*If new insulation material is needed, refer to "Insulation Requirements" text.

100D-3-40-78

Figure 12-1. Heat and Vent Duct Insulation Check and Sheathing Repair

SPECIFICATION	PROCESS
MIL-B-7883 (alloy per Fed. Spec QQ-S-561, Class 4)	Silver brazing

Note

Weld repairs on inner tubes must be followed by pressure-checking to ensure the structural integrity of the tube, and by passivating and painting to minimize subsequent corrosion.

During cleaning of the duct before rework, and during passivating following rework, all outlets should be sealed to prevent entrapment of the fluid in the bellows and overlap joints.

Discrepant flange sealing surfaces may be metalized and refinished if damage is not severe. If damage is severe, the flange may be replaced by fusion butt-welding in a section of duct equal to the amount removed and using a new flange.

REPAIR OF SOLID TUBES AND FLANGES.

1. Repair leaks in inner tube and at flanges on assemblies similar to Part No. 103007-39, 223-53549, 800176, and 103243 by Heliarc weld. Use filler rod (Specification MIL-R-5031, Class 5A).

2. Following repair, pressure-check tube at 35 psig, at room ambient temperature. There must be no leakage. Then proof-pressure-check tube at 510 psig, at room ambient temperature. No deformation shall occur.

3. Passivate tube per Specification MIL-S-5002.

4. Paint tube as directed under "Painting Exterior Duct Walls and Bellows."

5. Replace insulation and sheathing. If new insulation is required, refer to "Insulation Requirements."

REPAIR OF DAMAGED BELLOWS.

Repair of damaged bellows assemblies is not allowed except for the following minor rework: Broken strands of plaits may be removed and cut off at the collars, and displaced plaits and strands may be rearranged to as near normal position as possible. A bellows may be replaced in a manner similar to replacing a flange.

REPAIR OF OUTER SHEATHING.

See figure 12-1.

INSULATION REQUIREMENTS.

The insulation material must be one that does not have an organic binder. A ceramic fiber such as Kaowool, or equivalent, is recommended. (Kaowool may be purchased from The Babcock & Wilcox Co, 450 Mission St, San Francisco, California.) The thermal conductance from the ID of the tube to the OD of the outer cover, excluding air film coefficient, shall not exceed 2.2 Btu/hr/ft^2, based on an average temperature of 675°F at the ID of the tube.

PAINTING EXTERIOR DUCT WALLS AND BELLOWS.

MATERIALS REQUIRED.

MATERIAL	SPECIFICATION OR SOURCE
Paint, HR-25	Kansas Paint and Color Co
or	Wichita, Kan.
Paint, G665C8	Sherwin-Williams Co
	Los Angeles, Calif.
Methyl ethyl ketone (MEK)	Fed. Spec TT-M-261
or	
Acetone	Fed. Spec O-A-51
Xylene	Fed. Spec TT-X-916
or	
Toluene	Fed. Spec TT-T-548

CLEANING.

Surfaces to be painted must be thoroughly cleaned by wiping with methyl ethyl ketone or acetone. Follow all safety precautions established for use of flammable liquids.

APPLYING PAINT.

Note

No primer is to be used.

For spray application, reduce with one volume of xylene to four volumes of paint, using xylene in accordance with Federal Specification TT-X-916. Add additional xylene if required to produce satisfactory spray application. Toluene (Federal Specification TT-T-548) may be used in lieu of xylene only until xylene becomes available.

The air supply must be clean, free of water, oil, or other contaminants. The air line ahead of the spray gun shall be equipped with a DeVilbiss Air Transformer No. P-HLG-501, or equivalent.

Apply paint by spray in as thin a coat as possible to achieve complete hiding without runs or sags. Only the exteriors of the duct walls and the bellows are to be painted. Slight overspray on the flange exterior is permissible except in and around boltholes and on sealing surfaces.

After application of paint, the ducts must be handled with care to minimize damage to the paint. The paint must be cured by exposing the duct to 400°F for 30 minutes.

ENGINE AFTERBURNER COOLING-AIR DUCTS.

AFTERBURNER REMOVABLE COOLING-AIR DUCT.

See figure 12-2 for repairs.

AFTERBURNER COOLING SHROUD ASSEMBLY (PART NO. 102873).

The wall thickness of the original 102873-1 and -2 tubes was 0.011 inch. An alternate tube with an 0.016-inch-thick wall is also authorized. The alternate tube material is 321 or 347 CRES (Specification MIL-T-8606).

10-TUBE FLANGE (PART NO. 154661).

This tube flange was originally fabricated from one piece of material. Fabrication from two pieces of material is now authorized.

ENGINE AFTERBURNER COOLING-AIR FORWARD DUCT ASSEMBLY (PART NO. 192-942027).

The station 382 tube flange on this assembly was originally fabricated from one piece of material. Fabrication from two pieces of material is now authorized.

The following instructions are for repair procedures and crack limitations on ducts, part Nos. 192–42057, 180–52068–10, 180–42069 and 180–40048:

a. Stop drill cracks less than two inches long with No. 40 drill and cover with RTV 106 sealant.

b. Cracks near open end may be ground out to one-fourth inch radius fillet.

c. Ducts with more than ten cracks or cracks within three inches of another crack or two inches of a mounting point must be removed and weld repaired per MIL–W–8611.

HEAT AND VENT AIR INLET SCREENS.

The following instructions are for repair procedures and broken mesh limitations on primary heat exchanger inlet and electronic compartment ram air cooling inlet screens, located in the engine air intake duct:

a. Screens with broken mesh strands may be repaired by removing broken strands to the point of intersection. The following limitations apply:

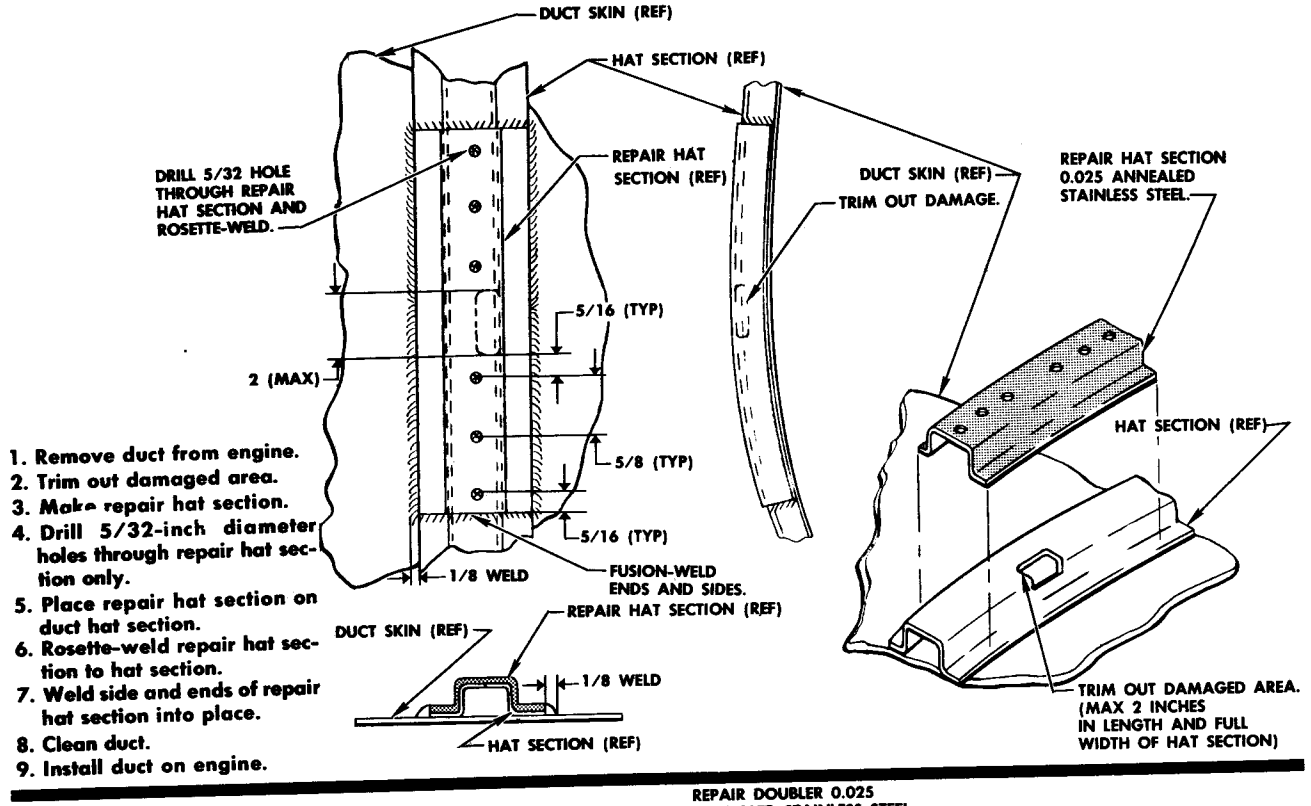
(1) Each mesh with a broken strand must be completely encompassed by intact meshes and/or outer frame of screen.

(2) The number of mesh strands broken within a screen shall not exceed three.

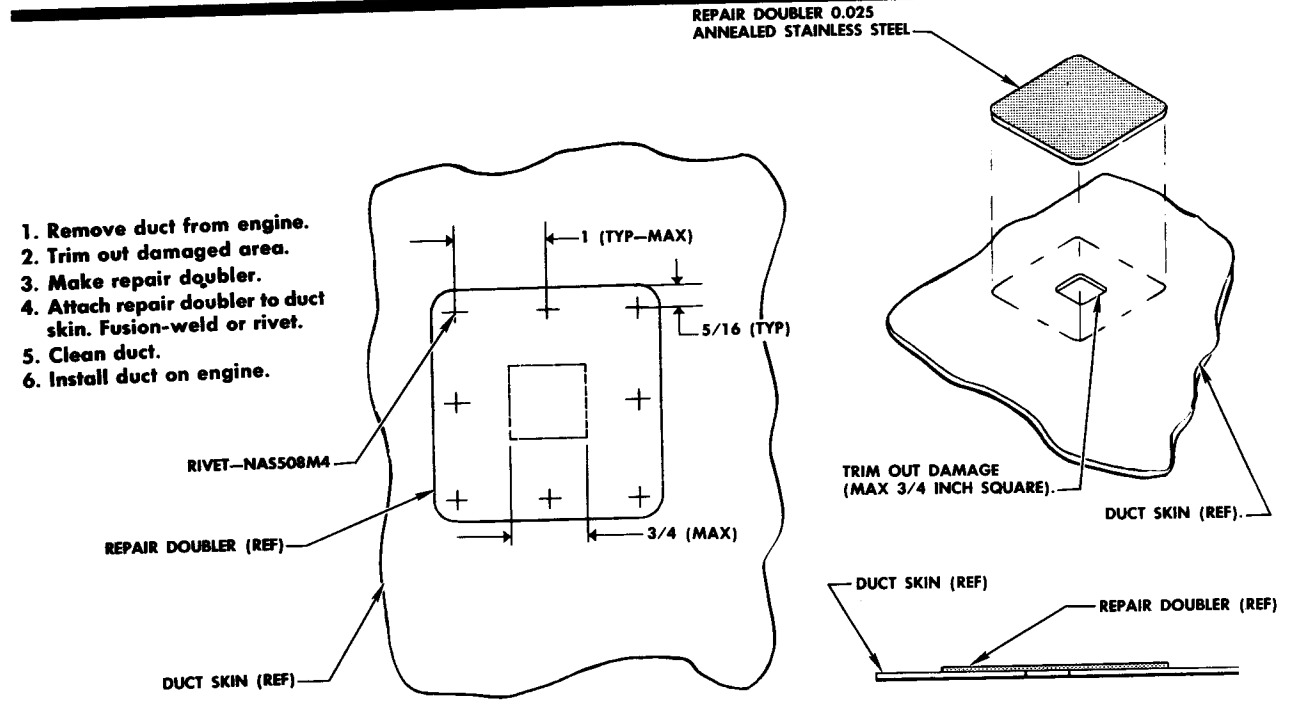
b. Screens exceeding above limitations may be repaired by the silver brazing in of new strands of wire. Stainless steel wire 18-8 is a suitable substitute for the monel wire, if monel wire can not be obtained.

PERMANENT—CATEGORY A

HEAT AND VENT



1. Remove duct from engine.
2. Trim out damaged area.
3. Make repair hat section.
4. Drill 5/32-inch diameter holes through repair hat section only.
5. Place repair hat section on duct hat section.
6. Rosette-weld repair hat section to hat section.
7. Weld side and ends of repair hat section into place.
8. Clean duct.
9. Install duct on engine.



1. Remove duct from engine.
2. Trim out damaged area.
3. Make repair doubler.
4. Attach repair doubler to duct skin. Fusion-weld or rivet.
5. Clean duct.
6. Install duct on engine.

100D-3-40-3A

Figure 12-2. Afterburner Removable Cooling-air Duct

Appendix I
REPAIR MATERIALS

REPAIR METALS

STOCK DESCRIPTION	GAGE	COMMERCIAL DESIGNATION	SPECIFICATION
		ALUMINUM	
Sheet	0.016	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.020	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.025	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.032	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.040	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.050	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.063	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.071	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.080	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.090	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.100	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.125	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.156	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.188	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.250	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	0.375	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	0.500	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	0.625	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	0.750	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	1.000	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	1.250	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362

STOCK DESCRIPTION	GAGE	COMMERCIAL DESIGNATION	SPECIFICATION
Plate	1.500	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	1.750	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Plate	2.000	2024-T4 clad or 2024-O clad	Federal Spec QQ-A-362
Sheet	0.016	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.020	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.025	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.032	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.040	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.050	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.063	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.071	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.080	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.090	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.100	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.125	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.156	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.188	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Sheet	0.250	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	0.375	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	0.500	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	0.625	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	0.750	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	1.000	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	1.250	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	1.500	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287

STOCK DESCRIPTION	GAGE	COMMERCIAL DESIGNATION	SPECIFICATION
Plate	1.750	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
Plate	2.000	7075-O clad or 7075-T6 clad	Federal Spec QQ-A-287
		MAGNESIUM ALLOY	
Magnesium alloy	0.020	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.025	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.032	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.040	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.050	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.063	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.071	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.080	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.090	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.100	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.125	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.190	Magnesium alloy	Federal Spec QQ-M-31
Magnesium alloy	0.250	Magnesium alloy	Federal Spec QQ-M-31
		STEEL	
Sheet	0.010	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.012	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.016	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.018	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.020	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.024	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.025	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.030	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.032	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.035	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.036	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.040	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.045	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.050	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.063	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.080	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.090	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.125	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.160	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.190	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.250	½ hard (CRES)	Spec MIL-S-5059
Plate	0.312	½ hard (CRES)	Spec MIL-S-5059
Plate	0.375	½ hard (CRES)	Spec MIL-S-5059
Plate	0.500	½ hard (CRES)	Spec MIL-S-5059
Plate	0.625	½ hard (CRES)	Spec MIL-S-5059
Plate	0.750	½ hard (CRES)	Spec MIL-S-5059

STOCK DESCRIPTION	GAGE	COMMERCIAL DESIGNATION	SPECIFICATION
Plate	1.000	½ hard (CRES)	Spec MIL-S-5059
Plate	1.250	½ hard (CRES)	Spec MIL-S-5059
Plate	1.500	½ hard (CRES)	Spec MIL-S-5059
Sheet	0.012	4130 (norm)	Spec MIL-S-18729
Sheet	0.016	4130 (norm)	Spec MIL-S-18729
Sheet	0.020	4130 (norm)	Spec MIL-S-18729
Sheet	0.025	4130 (norm)	Spec MIL-S-18729
Sheet	0.032	4130 (norm)	Spec MIL-S-18729
Sheet	0.036	4130 (norm)	Spec MIL-S-18729
Sheet	0.040	4130 (norm)	Spec MIL-S-18729
Sheet	0.045	4130 (norm)	Spec MIL-S-18729
Sheet	0.050	4130 (norm)	Spec MIL-S-18729
Sheet	0.063	4130 (norm)	Spec MIL-S-18729
Sheet	0.080	4130 (norm)	Spec MIL-S-18729
Sheet	0.090	4130 (norm)	Spec MIL-S-18729
Sheet	0.125	4130 (norm)	Spec MIL-S-18729
Sheet	0.160	4130 (norm)	Spec MIL-S-18729
Sheet	0.190	4130 (norm)	Spec MIL-S-18729
Sheet	0.250	4130 (norm)	Spec MIL-S-18729
Plate	0.312	4130 (norm)	Spec MIL-S-18729
Plate	0.375	4130 (norm)	Spec MIL-S-18729
Plate	0.500	4130 (norm)	Spec MIL-S-18729
Plate	0.625	4130 (norm)	Spec MIL-S-18729
Plate	0.750	4130 (norm)	Spec MIL-S-18729
Plate	1.000	4130 (norm)	Spec MIL-S-18729
Plate	1.250	4130 (norm)	Spec MIL-S-18729
Plate	1.500	4130 (norm)	Spec MIL-S-18729

SOURCES OF COMMERCIAL PRODUCTS

PRODUCT	SOURCE
Accelerator, PR341A	Products Research Corp Los Angeles, Calif.
Adhesive epoxy base, Bondmaster 611	Rubber and Asbestos Corp Bloomfield, N. J.
Adhesive, epoxy base, Epon VIII	Shell Chemical Corp New York, N. Y.
Bolt, EWB-18 Series	Standard Pressed Steel Co Jenkintown, Pa.
Bolt, North American Aviation specification	Voi-Shan Manufacturing Co, Inc Culver City, Calif. Briles Manufacturing Co El Segundo, Calif.
Catalyst, DDM	Lucidol Division Novadel-Agene Corp Buffalo, N. Y.
Cement, EC-613B	Minnesota Mining and Mfg Co St. Paul, Minn.

PRODUCT	SOURCE
Chemical film, Alodine 1200	American Chemical Paint Co Ambler, Pa.
Chemical film, Iridite 14	Allied Research Products, Inc Baltimore, Md.
Chuck, drill extension	Beaver Tool Company New York City, N. Y.
Cleaning compound, LAR 1634	Turco Products, Inc Los Angeles, Calif.
Cleaning compound, Deseal A	Kelite Products, Inc Los Angeles, Calif.
Coating compound, fuel- and oil-resistant, Stabond C-136	American Latex Products Corp Hawthorne, Calif.
Coating, Vulcabond V-36	American Latex Products Corp Hawthorne, Calif.
Coating, A56B	B. F. Goodrich Co Akron, Ohio
Coating, plastic, abrasion- & corrosion-resistant Everlube 1329, Everlube 1329A	Everlube Corp North Hollywood, Calif.
Curing agent, epoxy adhesive, CH-1 hardener	Rubber and Asbestos Corp Bloomfield, N. J.
Curing agent, epoxy adhesive, diethylamino propylamine	Shell Chemical Corp New York, N. Y.
Curing agent, epoxy adhesive, diethylene triamine (DTA)	Carbide and Carbon Chemicals Corp Division of Union Carbide and Carbon New York, N. Y.
Detonator, friction, explosive rivet	Sheridan Products, Inc Inglewood 3, Calif.
Developer, dry powder, Zyglo (ZP-2)	Magnaflux Corp Chicago, Ill.
Developer, Zyglo (ZP-3)	Magnaflux Corp Chicago, Ill.
Drill extension, "Little Beaver"	Beaver Tool Co New York, N. Y.
Drill, high-speed, 135-degree, split-point, Type C	National Twist Drill and Tool Co Rochester, Michigan
Fastener, quick-acting, structural	Cam-Loc Fastener Corp Paramus, N. J.
Filler, aerodynamic skin filler PR341	Products Research Corp Los Angeles, Calif.
Filler, epoxy adhesive, concrete aggregate	Panacalite Pacific, Inc Los Angeles, Calif.
Filler, epoxy adhesive, Perlite No. 1	Paramount Perlite Co Paramount, Calif.
Insert, threaded, Heli-Coil	Heli-Coil Corp Danbury, Conn.
Insert, threaded, Keensert	Newton Insert Co Los Angeles, Calif.
Insert, threaded, Rosan	Rosan, Inc Newport Beach, Calif.
Jo-bolt	National Screw Manufacturing Co of Calif. Los Angeles, Calif.
Light, black, inspection, Magnaflux ZB-26	Magnaflux Corp Chicago, Ill.
Nut, EB series	Elastic Stop Nut Corp of America Union, N. J.

PRODUCT	SOURCE
Penetrant, inspection, Zyglo	Magnaflux Corp Chicago, Ill.
Remover, LAC1981C Noncresylic, Type I (for spray application)	Turco Products, Inc Los Angeles, Calif.
Remover, 713C, Type II (for brush-on application)	Turco Products, Inc Los Angeles, Calif.
Rivet, Cherry	Townsend Co, Cherry Rivet Div Santa Ana, Calif.
Rivet, Deutsch drive pin	Huck Manufacturing Co Detroit 7, Mich.
Rivet, explosive	E. I. du Pont de Nemours and Co (Inc) Explosives Department Wilmington 98, Del.
Rivet, Hi-Shear	Hi-Shear Rivet Tool Co Los Angeles 45, Calif.
Rivet, Huck	Huck Manufacturing Co Detroit 7, Mich.
Rivet, Olympic	Olympic Screw and Rivet Corp Downey, Calif.
Screw, Hi-Shear	Hi-Shear Rivet Tool Co Los Angeles 45, Calif.
Screw, Phillips head	American Phillips Recess American Screw Co Willimantic, Conn.
Screw, Reed & Prince	Reed & Prince Mfg Co Worcester, Mass.
Screw, Torq-Set	American Screw Co Willimantic, Conn.
Sealant gun, disposable-cartridge	Pyles Industries, Inc Detroit 2, Mich.
Sealing compound, PR341	Products Research Corp Los Angeles, Calif.
Sealing compound, high-temperature HT-23	American Latex Products Corp Hawthorne, Calif.
Sealing compound, high-temperature EC-1548	Minnesota Mining and Mfg Co St. Paul, Minn.
Sealing compound, general-purpose	Coast Pro-Seal & Mfg Co Los Angeles, Calif. Minnesota Mining and Mfg Co St. Paul, Minn. Products Research Corp Los Angeles, Calif.
Sealing compound, Pro-Seal 414	Coast Pro-Seal & Mfg Co Los Angeles, Calif.
Sealing compound, EC-612P	Minnesota Mining and Mfg Co St. Paul, Minn.
Sealing compound, Pro-Seal 714	Coast Pro-Seal & Mfg Co Los Angeles, Calif.
Sealing compound, Pro-Seal 714A	Coast Pro-Seal & Mfg Co Los Angeles, Calif.
Sealing tape, unfused Teflon 547	Minnesota Mining and Mfg Co St. Paul, Minn.
Tape, general-purpose, pressure-sensitive	Industrial Tape Corp New Brunswick, N. J.
Tape, general-purpose, pressure-sealing	Bauer & Black Division Kendall Co Chicago, Ill.

PRODUCT	SOURCE
Tester, hardness, portable, Barcol	Barber-Colman Co Rockford, Ill.
Tester, hardness, portable, Ernst	Newage International, Inc New York 17, N. Y.
Tester, hardness, portable, Riehle	Riehle Testing Machine Div American Machine & Metals, Inc East Moline, Ill.
Tool, Jo-bolt driving	Lok-Fast, Inc Torrance, Calif.

REPAIR MATERIALS AND USES

MATERIAL	STOCK NO.	SPECIFICATION	REMARKS
		ADHESIVES	
Adhesive, EC-1469		Commercial	Used in aluminum honeycomb repair.
Aerodynamic fillers			Used to fill dents, nicks, and gouges for aerodynamic smoothness.
PR-341 filler and	8030-888-0534	Commercial	
PR-341A accelerator	8030-NSL	Commercial	
B-88	8040-NI217115	Commercial	
Preslite 108	8030-693-2311	Commercial	
EC-967	8030-693-2311	Commercial	
Metlite 70-30	8030-693-2311	Commercial	
Cement, EC-613B		Commercial	Used for liquidtight sealing in areas where temperature is between 400°F and 500°F.
Cement, transparent plastic	8040-266-0815	Commercial	Used in making repairs on transparent plastic.
Cement and accelerator, top-coat base		Spec MIL-C-7439	Used in making repairs on glass fabric.
Coating, A-56B		Commercial	Used in sealing pressurized areas.
Coating, Vulcabond V-36		Commercial	Used in sealing pressurized areas.
Coating compound, C-136		Commercial	Used for liquidtight sealing in areas where temperature is between 400°F and 500°F.
Epoxy filler, Corefil 615		Commercial	Used in aluminum honeycomb repair.
Methylene chloride		Spec MIL-M-6998	Used as a substitute cement for transparent plastic.
Methyl isobutyl ketone		Fed. Spec TT-M-268	Used as a thinner for abrasion-resistant coating.
Neoprene coating, Class I	8030-506-4030	Spec MIL-C-7439	Rain-resistant coating.
Neoprene coating, Class II	8030-NSL	Spec MIL-C-7439	Rain-resistant coating.
Resin, low-pressure laminating		Spec MIL-R-7575	Used for making radome repairs.
Sealing compound, general-purpose		Spec MIL-S-7502, Class B-4	Pressure sealant.
Sealing compound, Pro-Seal high-temperature 714, and Pro-Seal cataylst 714A		Commercial	Used for liquidtight sealing where temperature is between 225°F and 400°F.

MATERIAL	STOCK NO.	SPECIFICATION	REMARKS
Sealing compound, high-temperature EC-1548		Commercial	Used for liquidtight sealing in areas where temperature is between 400°F and 500°F.
Sealing compound (PR-1221)		Commercial	Used as a corrosion barrier.
Sealing compound, EC-612P		Commercial	Used for sealing pressurized areas.
Sealing compound, Pro-Seal 414		Commercial	Used for sealing pressurized areas.
CLEANERS AND SOLVENTS			
Acid, chromic		Fed. Spec O-C-303	Used in chemical identification of metals.
Acid, hydrochloric		Fed. Spec O-A-86	Used in chemical identification of metals.
Acid, sulfuric, specific gravity 1.84		Commercial	Used in aluminum honeycomb repair.
Acid, nitric		Fed. Spec O-A-88	Used in chemical identification of metals.
Cleaning compound, Deseal A.		Commercial	Used as a cleaning agent.
Cleaner, phosphoric acid		Spec MIL-C-5410	Metal cleaner and brightener.
Cleaning compound, LAR 1634		Commercial	Used as a cleaning agent.
Methyl ethyl ketone	6810-281-2785	Fed. Spec TT-M-261	Used as a cleaning agent.
Naphtha		Fed. Spec TT-N-95	Used as a cleaning agent.
Polish and cleaner, transparent plastic		Spec MIL-C-18767	Used to clean and polish transparent plastics.
Sodium dichromate		Commercial	Used in aluminum honeycomb repair.
Sodium hydroxide		Spec MIL-S-11795	Used as a cladding check.
Toluol	6810-281-2002	Fed. Spec TT-T-548	Used as a cleaning agent.
Thinner, wash		Fed. Spec P-S-661	Used as a paint remover.
Xylene, Grade B	6810-598-6600	Fed. Spec TT-X-916	Used as a spraying thinner or reducer.
FINISHES			
Alodine 1200		Spec MIL-C-5541	Prepaint surface treatment for aluminum alloys.
Coating		Spec MIL-C-27725	For integral fuel tanks.
Iridite 14		Spec MIL-C-5541	Identical to Alodine 1200.
Lacquer, aluminized		Spec MIL-L-7178	Exterior finish.
Lacquer, medium green		Spec MIL-L-6805	Interior finish.
Primer, rain-erosion-resistant coating material (Bostic 1007)		Commercial	Used as a protective coating for fiber glass structure.

MATERIAL	STOCK NO.	SPECIFICATION	REMARKS
Primer, wash		MIL-C-8514	Used as a primer pretreatment coating on metal surfaces before painting.
Primer, zinc-chromate		Spec MIL-P-8585A	Primer coating used over wash primer or on clean metal surfaces of all types.
		MISCELLANEOUS	
Balsa wood		MIL-S-7998	Used in aluminum honeycomb repair.
Black light, Magnaflux ZB-26		Commercial	Used for locating and determining size of cracks in metal.
Butyl cellosolve		Commercial	Used in corrosion- and abrasion-resistant coating.
Cadmium sulphate		Commercial	Used in chemical identification of metals.
Calcium carbonate		Spec JAN-C-293	Used to detect cracks in metal surfaces.
Cellophane sheet		Commercial	Used in aluminum honeycomb repair.
Chamois skin	8330-257-2492	Fed. Spec KK-L-167	Used to clean transparent plastic.
Cloth, polishing		Fed. Spec CCC-F-466	General-purpose cleaning cloth.
Cobalt naphthenate (6 percent)		Commercial	Used in making glass fabric repairs.
Curing agent U		Commercial	Used in aluminum honeycomb repair.
Felt sheet (0.250 inch)		Commercial	Used in making glass fabric repairs.
Glass fabric, Type VIII		Spec MIL-F-9084	Used in making glass fabric repairs.
Glass wool		Commercial	Used in aluminum honeycomb repair.
Honeycomb core, aluminum, $\frac{1}{8} \times 0.002$ inch		Spec MIL-C-7438	Used as core material when making aluminum honeycomb repairs.
Petrolatum		Fed. Spec VV-P-236	Used to prevent sealing compound from sticking to screws and removable fasteners.
Plastic, transparent		Spec MIL-P-5425	Used to make repairs on transparent plastic canopies.
Polyvinyl alcohol sheet, 0.003 to 0.005 inch		Commercial	Used in making glass fabric repairs.
Sealing tape, unfused Teflon No. 547		Commercial	Used for liquidtight sealing in areas where temperature is between 400°F and 500°F.
Silver nitrate crystals		Commercial	Used in chemical identification of metals.
Sodium chloride		Commercial	Used in chemical identification of metals.
Talc, white powder, technical grade		Commercial	Used in aluminum honeycomb repair.

MATERIAL	STOCK NO.	SPECIFICATION	REMARKS
Tape, general-purpose, pressure-sensitive, Type II, Grade III		Fed. Spec PPP-T-60	Used as backup, to prevent pressure from forcing sealants through openings in structure being sealed.
Tape, Vinyl No. 473		Commercial	Used as a corrosion barrier between magnesium and other metals.
Zyglo developer ZP-3		Commercial	Used to detect cracks in metal surfaces.
Zyglo dry developer powder ZR-2		Commercial	Used to detect cracks in metal surfaces.
Zyglo penetrant		Commercial	Used to detect cracks in metal surfaces.

Appendix II

GLOSSARY OF AIRPLANE AND AIRPLANE REPAIR TERMS

INTRODUCTION.

The definitions in this glossary correspond to the meanings of words and phrases as used in this manual. Nontechnical explanations are used whenever possible.

DEFINITIONS.

abrasion. Wearing away by friction, scratching.

abrasive. A substance used for grinding, polishing, etc, as emery paper.

aerodynamic. 1. Designed for favorable reaction when moved through the air. **2.** The result from the reaction of the moving object to the forces in the surrounding air, as in aerodynamic drag or performance.

aft. Toward the rear or tail.—**aft of.** Behind.

airfoil. A surface or body designed to obtain a reaction from the air through which it moves.

airframe. The total airplane structure, including the framework (substructure) and skins.

air load. The aerodynamic load or force imposed upon a moving airfoil or other object.

air-load rib. A rib which derives its principal loads from aerodynamic forces.

annealing. A process by which metals are stress-relieved and softened. This is done by carefully controlled heating and cooling (quenching).

anodize. An electrochemical treatment or bare aluminum alloys resulting in corrosion-resistant surface. Anodized surfaces also offer an excellent bond for primer and paint.

axis. A straight line used as a reference for orienting a part or about which a part could bend, twist, or rotate.

baffle. A plate or wall, as around an engine or in a tank, the purpose of which is to direct or retard the flow of a liquid or gas.

bead. 1. A pressed or rolled-in molding in sheet metal used for stiffening. **2.** The fused area in fusion welding. **3.** A narrow strip of sealant laid over or along a seam.

beam. The principal structural members running from root to tip in the stabilizers and rudder and from inboard to outboard in the ailerons and flaps. (Corresponding members in the wing are called spars.)

beam cap. The beam edge member to which the control surface skins are attached.

bend allowance. The flat pattern dimension that must be allowed for each bend in sheet-metal layout.

blind hole. A hole which does not pass through the material.

blind repair. A repair that can be made with access to but one side of the damage. Blind rivets or other fasteners that can be installed from one side are necessary for close-out.

blind rivet. A rivet that is installed from one side—pull rivet, drive-pin rivet, etc.

bolting angle. An angle provided for bolting one assembly to another.

bolting bar. A heavy bar-like member provided for bolting one assembly to another.

brake, *n.* A tool used for bending sheet metal.—*v.* to bend.

braze. To join metal with a hard solder, usually a copper alloy.

bulkhead. A solid or closed frame in the fuselage such as at each end of the pressurized compartment.

burring. The removal of sharp edges which result from cutting, drilling, and punching sheet metal.

cantilever. A beam or other member supported at one end only.

centerline. A line running along the longitudinal center of an object. In the fuselage, it is the theoretical line dividing the left and right sides.

chamfer. The diagonal surface formed when a corner is removed. External patches are chamfered to blend the patch with the skin to reduce drag.

channel. A three-sided (generally U-shaped) member. It may be formed, machined, or extruded.

chord. A straight line between the leading and trailing edges of an airfoil section.

chordwise. In the direction of the chord of an airfoil. See **chord.**

clad. A general term applied to aluminum alloy sheet metal coated with pure aluminum.

close-out. Installation of the final panel, skin, or cover. This operation is often accomplished to some extent by the use of blind fasteners.

doubler. An added thickness of material, usually at the edge of a sheet. See **repair doubler.**

drag. The resistance produced by a body when moved through the air. An airplane is designed to produce the least possible drag. Any object added outside its mold line (skin) will increase the drag.

empennage. The group of fixed and movable control surfaces at the aft end of the airplane.

extrusion. A metal shape formed by forcing the material in a softened condition through a die.

fair. To streamline a *part* of an airplane, especially for reducing wind resistance or drag.

fairing. A nonstructural part of an airplane designed to blend with the surrounding structure to minimize drag.

fastener. A general term given to attaching hardware, such as bolts, screws, rivets, quick-acting fasteners, etc.

fatigue. The weakening of material caused by the repeated loading or bending movements applied to it.

faying surface. The surface area where one sheet of metal (or other material) overlaps another.

fillet. A concave area formed where two surfaces meet, or a member added to produce a concave area at a joint.

flutter. A vibrating movement of a wing or control surface caused by aerodynamic forces.

frame. Lateral members in the fuselage which support the skin and longitudinal members.

grid. A plate lightened by a pattern of holes, usually used for strength and/or stiffness inside the skin.

hat section. An extruded or formed shape, the cross section of which resembles that of a hat.

joggle. An offset or step in a sheet-metal part, usually used to fit against parts in different planes.

land. 1. The surface on, or against, which an object rests, as a door land. **2.** In milled-skin construction, a thickened area where other members are attached, as a *rib land*, a *spar land*, *etc.*

longeron. A fore-and-aft member of the framing of an airplane fuselage, usually continuous across a number of points of support.

magnaflux. A patented method of inspecting steel parts for cracks or flaws. The part is first magnetized, then immersed in a solution of iron oxide and kerosene, thus making flaws visible to the naked eye.

manufactured splice. A skin splice made during the manufacture of the airplane.

mold line. The outside surface of an airplane. External patches are said to be "outside the mold line."

nonstructural. A non-load-carrying member, but subjected to air loads when used as an external part.

plane. A real or imaginary surface of infinite length and width, but having zero thickness.

prime. To apply the base or *primer* coats of finish in paints, lacquers, etc.

primer. A material used for the base coat (under the finish coat) of paint, lacquer, etc.

repair doubler. The added thickness of sheet or plate material that carries the loads across a hole or crack.

rib. A generally chordwise member of an airfoil which gives shape to the component. In addition, it usually carries structural loads.

spanwise. Across the ribs of an airfoil.

spar. A principal spanwise member of the structural framework of the wing.

spar cap. The upper and lower edge members of a spar. The skins are usually attached to this member.

specific gravity. The ratio of a weight or mass of a given volume to that of an equal volume of another substance. Water at 4°C is often used as a standard.

spotface. The removing of a small amount of the surface material around a hole providing a flat surface for a fastener head.

spring back. The amount or degree that metal will return when the bending forces are released. Spring-back must be taken into consideration when making form blocks or when making bends on a power brake.

static. At rest, without motion, influenced by no force other than its own weight.

statically balanced surface. A control surface that is in balance about its hinge axis when not subjected to aerodynamic forces.

strain. A condition in the material of a part expressed as the amount of dimensional distortion of the material (stretching, squeezing, etc) caused by the amount of stress being carried by the material. The higher the stress, the higher the strain.

stress. An internal condition of the material of a part, which is dependent upon the intensity of the load being carried by the part at that instant. Stress is always accompanied by strain. See **strain.**

stressed skin. A skin that works with the inner structure in carrying airframe loads.

stringer. Stiffening members usually running longitudinally in the fuselage and spanwise in the wing used to reinforce thin sheets and also to resist bending and shear loads.

substructure. A general term for any structure behind a skin or web.

tension. The effect of loads acting on a part that tend to pull it apart or lengthen the part.

Appendix III

INSPECTION AND REPAIR OF WING ROOT RIB ATTACH FITTINGS

I. INTRODUCTION.

a. To provide inspection and repair methods for stress corrosion damage on the wing root rib attach fittings forward and aft, left and right panels, on F-100D and F Aircraft.

b. This inspection will apply only to Depot and/or contract maintenance as a category "B" requirement and will not be required unless the wings are removed for depot or contract repair, IRAN or MISTR programs, and will be included as Appendix III to this manual.

II. INSTRUCTIONS.

a. Accomplish the following inspections on the forward and aft attach fittings left and right hand, located on the wing root rib, part No. 192-12371, on F-100D and F aircraft.

(1) Remove all parts and attaching hardware from the wing root rib forward attach fitting, (see figure 3A-1) and retain for reinstallation.

(2) Remove the outer bushing, part No. 180-14709, on the rear fitting using bushing removing tool, part No. 63SCD6250 (see figure 3A-2).

(3) Remove and forward fitting inner bushing (sleeve and shoulder bushing) using removing tool, part No. 63SCD6250.

(4) Remove the outer shoulder bushing, part No. 4B20-B18-020, and the sleeve bushing, part No. 4B19-B18-022, on the aft wing attach fitting, using bushing removing tool, part No. 63SCD6250.

b. Removing Tool Instructions. (See figure 3A-2 (sheet 1 thru 4).

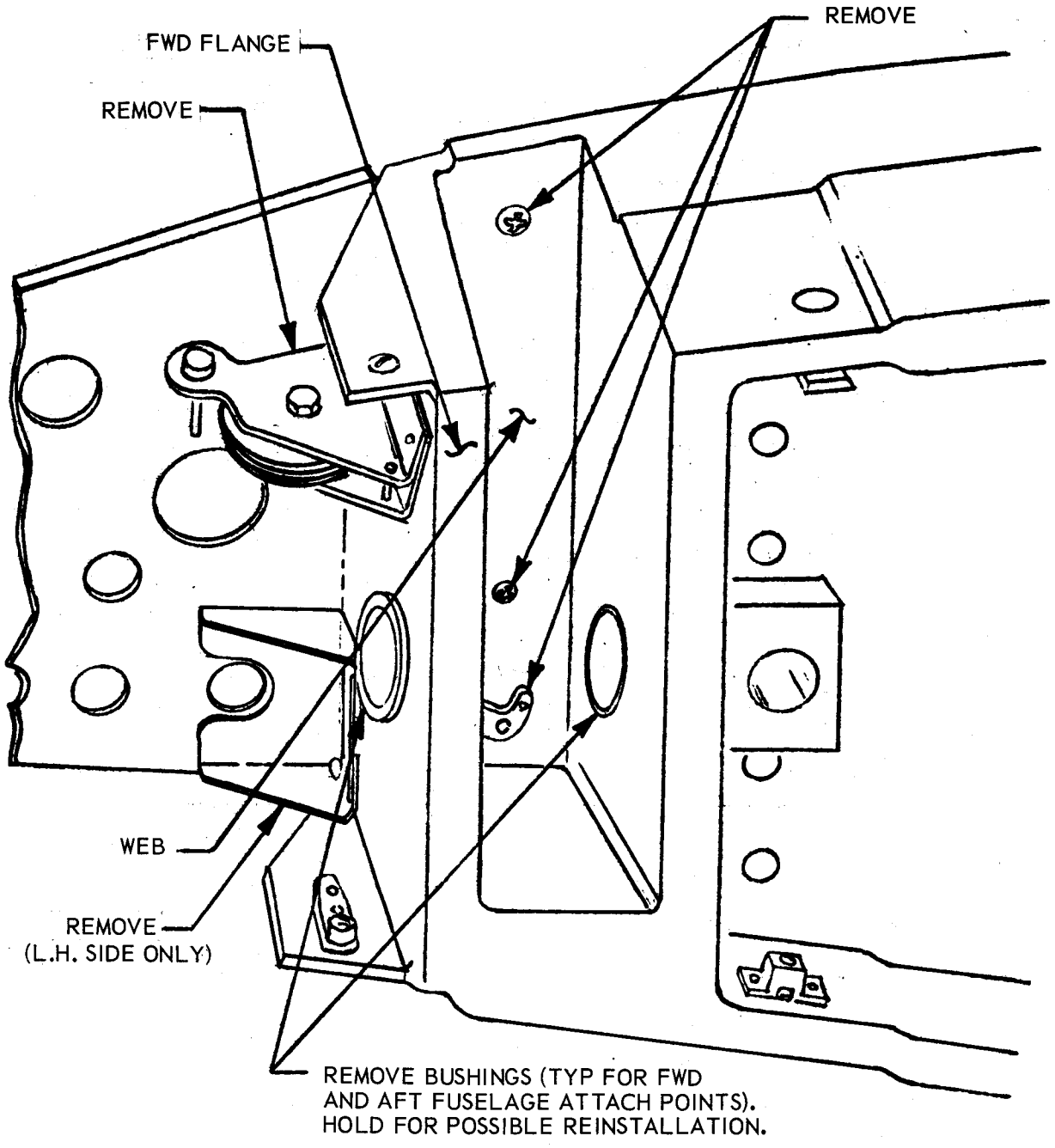
(1) Loosen puller nut, part No. AN335-12, with 1 1/8 inch wrench.

(2) Turn -2 stem counterclockwise (7/16 inch wrench) in -1 head until its fingers are completely collapsed (head is at its minimum diameter).

(3) Insert puller assembly tool through bushing. The lips of the puller head should be just past end of bushing. In case of the inner bushings, in blind holes, proper location of the head must be determined by feel.

(4) Hold -5 handle and tighten -2 stem until fingers of head are tight in bushing.

(5) Pull bushing by holding the -2 stem and tightening the puller nut, part No. AN335-12, against the -4 sleeve. Continue turning puller nut until bushing is free of housing.



NOTE:
REMOVE ALL DETAILS AND HARDWARE FROM WEB AND FWD FLANGE. HOLD ALL REMOVED ITEMS FOR REINSTALLATION.

Figure 3A-1. Parts and Hardware Removal – Fwd Attach Point

PULLER ASSY X63SCD6250 (BREAKDOWN)

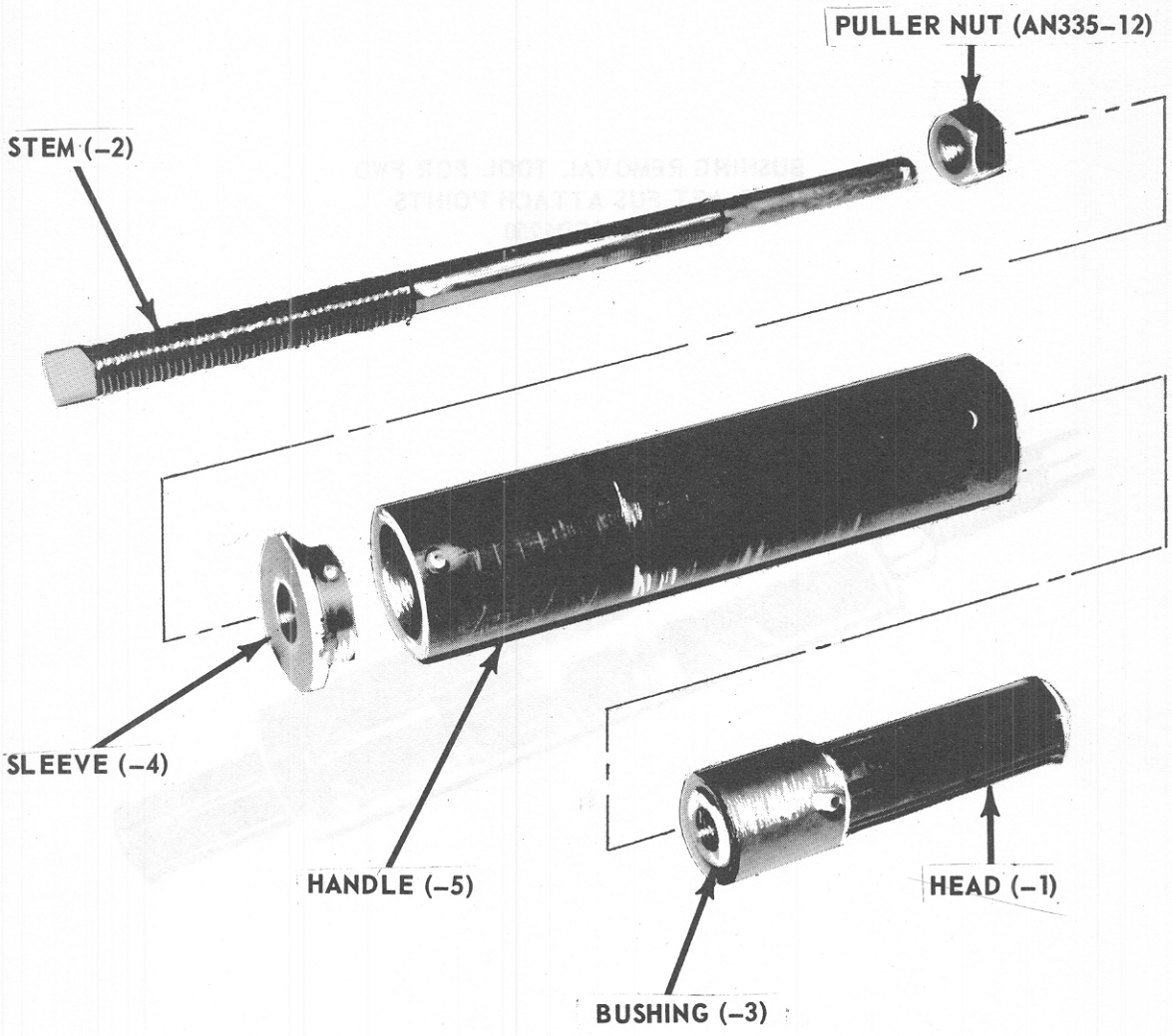


Figure 3A-2. Bushing Removal Tool (Sheet 1 of 4)

BUSHING REMOVAL TOOL FOR FWD
AND AFT FUS ATTACH POINTS
P/N X63SCD6250

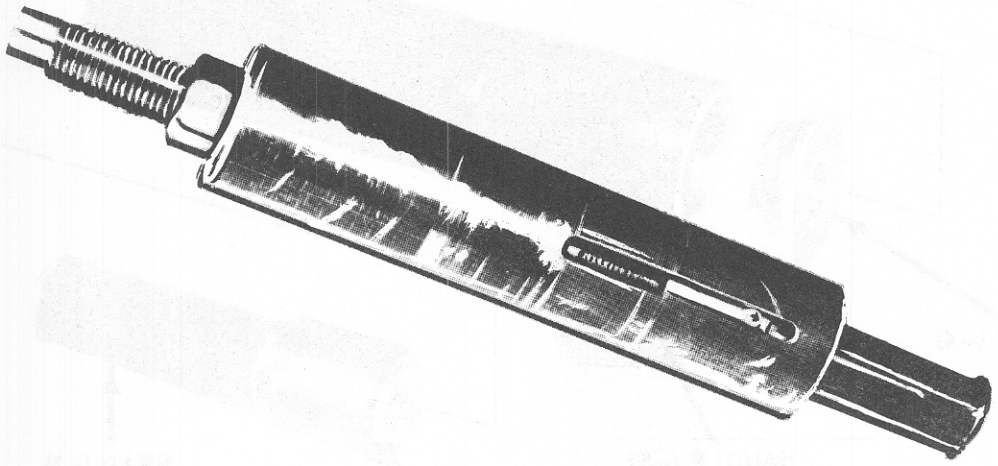


Figure 3A-2. Bushing Removal Tool (Sheet 2 of 4)

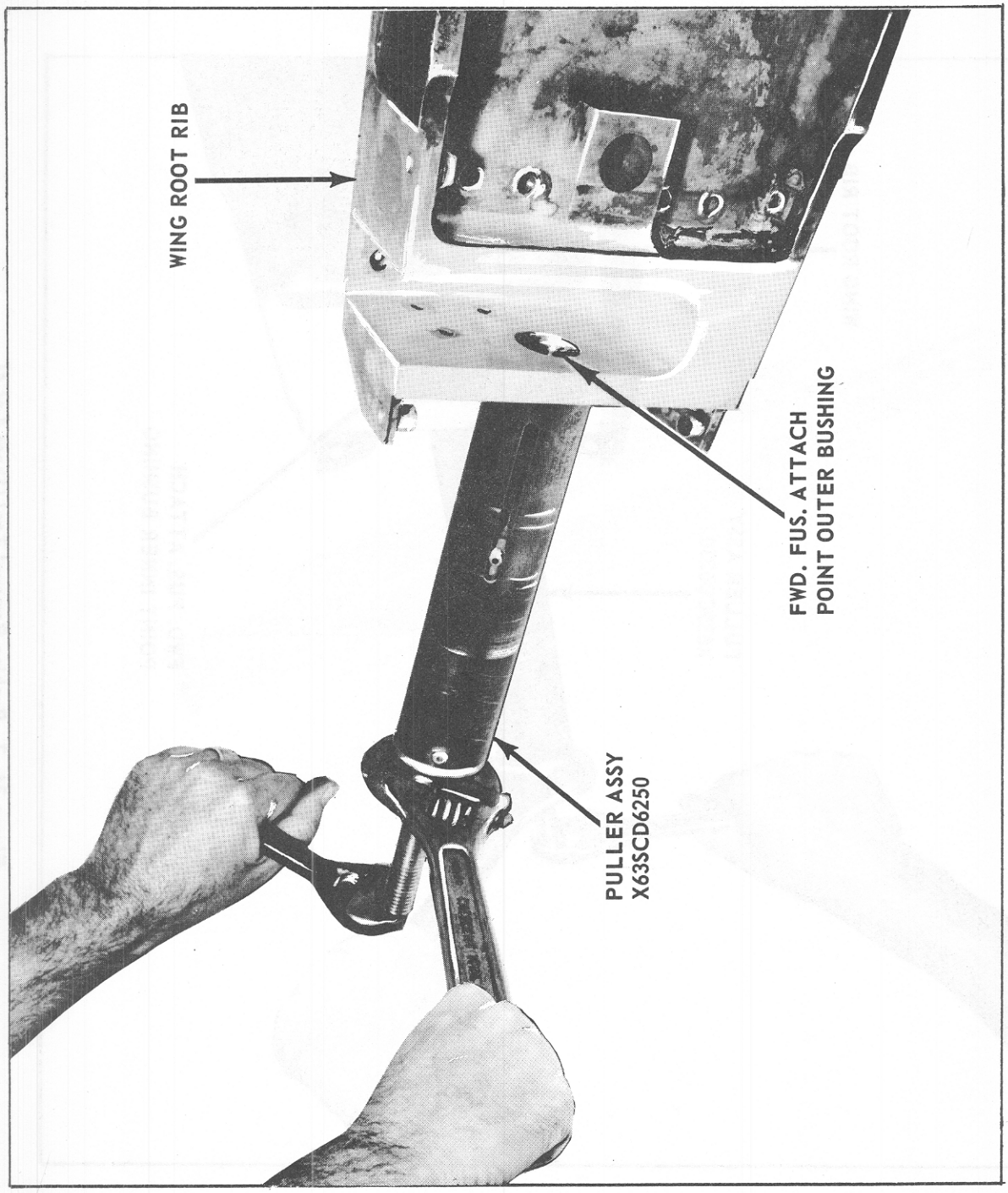


Figure 3A-2. Bushing Removal Tool (Sheet 3 of 4)

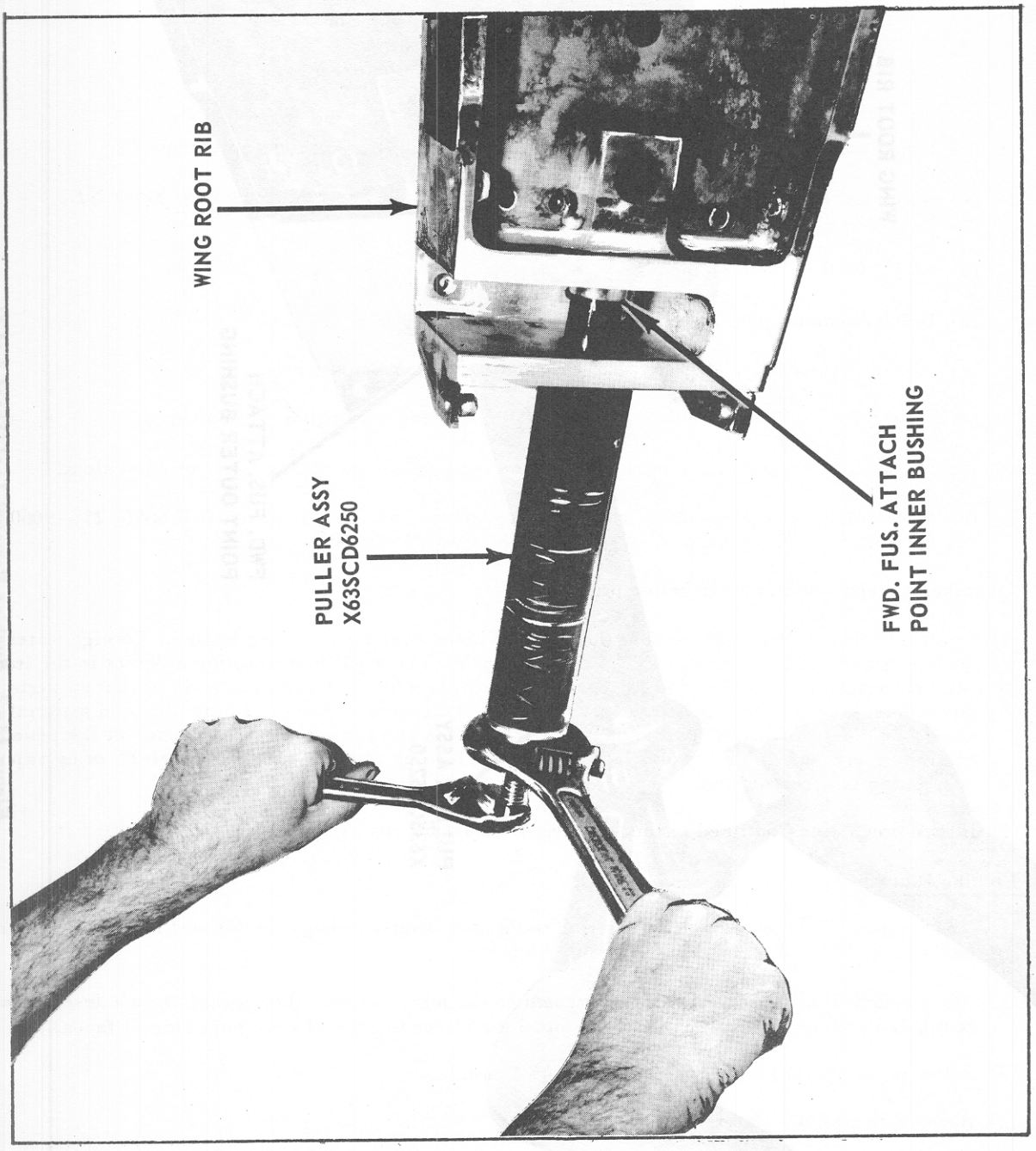


Figure 3A-2. Bushing Removal Tool (Sheet 4 of 4)

c. Penetrant Inspection.

(1) Penetrant Equipment.

NOTE

A portable fluorescent penetrant inspection kit, Model ZA-43, Federal Stock No. 4920-703-7406, or equivalent, can be used in lieu of this equipment.

(a) Inspection Penetrant, pressurized can, ZL-22 Magnaflux Corp. or equivalent.

(b) Black light, portable-Model ZB-26 Magnaflux Corp, or equivalent (Federal Stock No. 6635-611-5617, as specified in TA No. 455).

(c) Cloth, flannel, 18 x 18 inches.

(d) Brush aluminum wire (no other metal permitted).

(e) Inspection penetrant cleaner, ZC-7 Magnaflux Corp, or equivalent.

(f) Inspection penetrant developer, powder type, ZP Magnaflux Corp, or equivalent.

(g) Inspection penetrant developer, suspension type, ZP-9 Magnaflux Corp, or equivalent.

(h) Magnifying glass (Magnaflux, monocular type 5X to 20X magnification, R/S 6650-256-9060 and 6650-255-8268, Stock A-X 6650-530-1880 from TA No. 455).

d. Cleaning. (Preparation of parts before processing)

(1) Successful inspection requires that the parts be clean before processing begins. Foreign materials such as grease, oil, or water which would prevent the penetrant from entering a defect in the item, must be removed. Paint or plating should be removed before processing. Scale or rust on parts may cover defects or may trap penetrant causing false indications. Strong caustic and acid materials should be cleaned from the part because the chemicals can reduce the fluorescence of the penetrant band therefore making the results less reliable. Cleaning of parts with naptha, MEK, or by vapor degreasing is recommended.

(2) Health precautions outlined in this supplement ***must be*** observed.

e. Etching Procedure.

(1) To prevent etchant material from entering the Rosan inserts through the forward face of the fitting, cover these holes with masking tape on both faces.

(2) Apply a 20% sodium hydroxide paste etchant to the inside of both the forward and aft fuselage attach bolt holes and approximately 1/2-inch around the circumference of each hole face (3 faces).

(3) Allow to remain on parts for a period of 3 to 5 minutes.

(4) Rinse with potable water.

(5) Apply small amounts of Oakite 34 to etched surfaces to neutralize etchant. (Alternate method: Neutralize etchant with 50% nitric acid solution).

NOTE

Oakite Products, N. Y., N. Y.

(6) Wash thoroughly with potable water and allow to dry.

f. Developing and Applying Penetrant.

(1) Vigorously shake the can of wet developer ZP-9, until agitators rattle inside. Spray the part or section with developer, enough to wet the part thinly and evenly, ***NO MORE!*** Proper thickness will dry to a thin white translucent layer. Approximately 30 minutes drying time is required.

g. Removing Penetrant.

(1) When sufficient penetration time has been allowed, spray cleaner on the flannel cloth and remove the surface penetrant. Repeat, using clean areas of cloth, until clean. Do not clean by spraying cleaner on surface.

NOTE

Before performing light inspection, plug the light assembly into the 115 volt AC supply. Allow the light 5 or 10 minutes to warm up to maximum brilliance. Leave the light on throughout this inspection. If turned off, it must cool before it will relight.

(2) View with black light to be sure all fluorescence is removed from the surface only. The surface must be cleaned of surface penetrant. ***DO NOT OVERCLEAN!*** Excessive use of cleaner can dissolve penetrant out of cracks.

(3) The powder type developer, ZP-4, may be used in inaccessible areas.

(4) A short developing time, 3 to 5 minutes, should be allowed before inspecting the part to allow the developer to draw back to the surface penetrant which may be in defects. For rough surfaces, dry developing powder gives superior results.

h. Inspect.

(1) After development of indications, the parts are viewed under high intensity "black light" in a darkened area.

(2) Fluorescent indications, fluoresce brilliantly and the extent of the defect. Pores will show as glowing spots, but cracks will show as fluorescent lines.

(3) The operator should allow his eyes to become accustomed to the darkness before inspecting parts.

(4) After inspection, the developer, either wet or dry, should be removed from the parts.

i. Operating Precautions.

(1) The pressurized can may explode if the temperature rises over 130° F.

(2) Do not operate the black light with the transformer in a cabinet with the door closed. Either open the door or preferably, remove the transformer from the cabinet.

j. Health Precautions.

(1) The cleaner and developer must be used with adequate ventilation. Avoid prolonged or repeated breathing of the vapor. Avoid prolonged or repeated contact with the skin.

(2) Avoid contact of penetrant with skin by wearing neoprene gloves.

(3) Wash with soap and water to remove penetrant from skin.

(4) Black light entering the eyes will cause a cloudy effect because the liquid in the eyeballs fluoresce. This lasts as long as the light enters the eye and is ***not*** harmful. This does cause fatigue and care should be taken to avoid direct or reflected light.

k. Confirmation of Defects.

(1) The presence of stress corrosion or fatigue checks should be confirmed by means of visual inspection. A 5 to 20X magnifying glass under good light will provide proper amplification of the check.

l. Inspection Results.

(1) Data Reporting – all inspection results will be noted on applicable forms for each aircraft inspected as follows:

(a) Total aircraft flying hours.

(b) Aircraft type, model and serial number.

(c) Date of inspection.

(d) Location of Base, Depot, or contract facility.

(e) Inspection Method –(give details including materials and any deviations).

(2) Detailed Inspection Results.

Indicate results of inspection on sketch of paper. If no defects are noted, indicate the part is acceptable (OK). Send copy of results to SMAMA (NBTA), McClellan AFB, California, 95652.

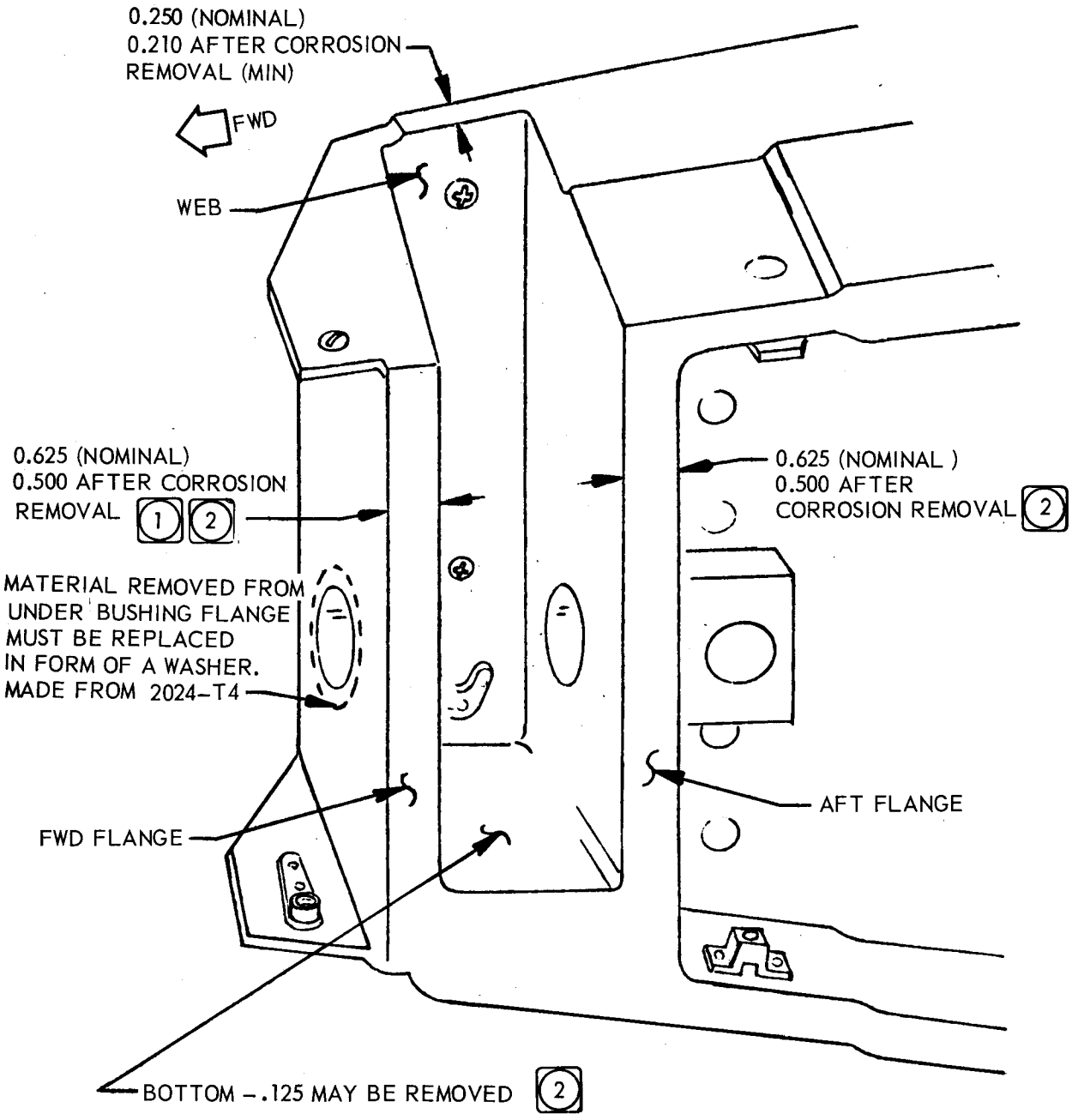
(3) Corrosion Removal.

(a) Surface corrosion will be removed in accordance with T.O. 1–1–2 or 1–1–5.

(b) Material removal limits from various surfaces of the forward and aft fittings are shown in figures 3A–3, 3A–4 and 3A–5.

(4) Reaming and Spot Face Requirements:

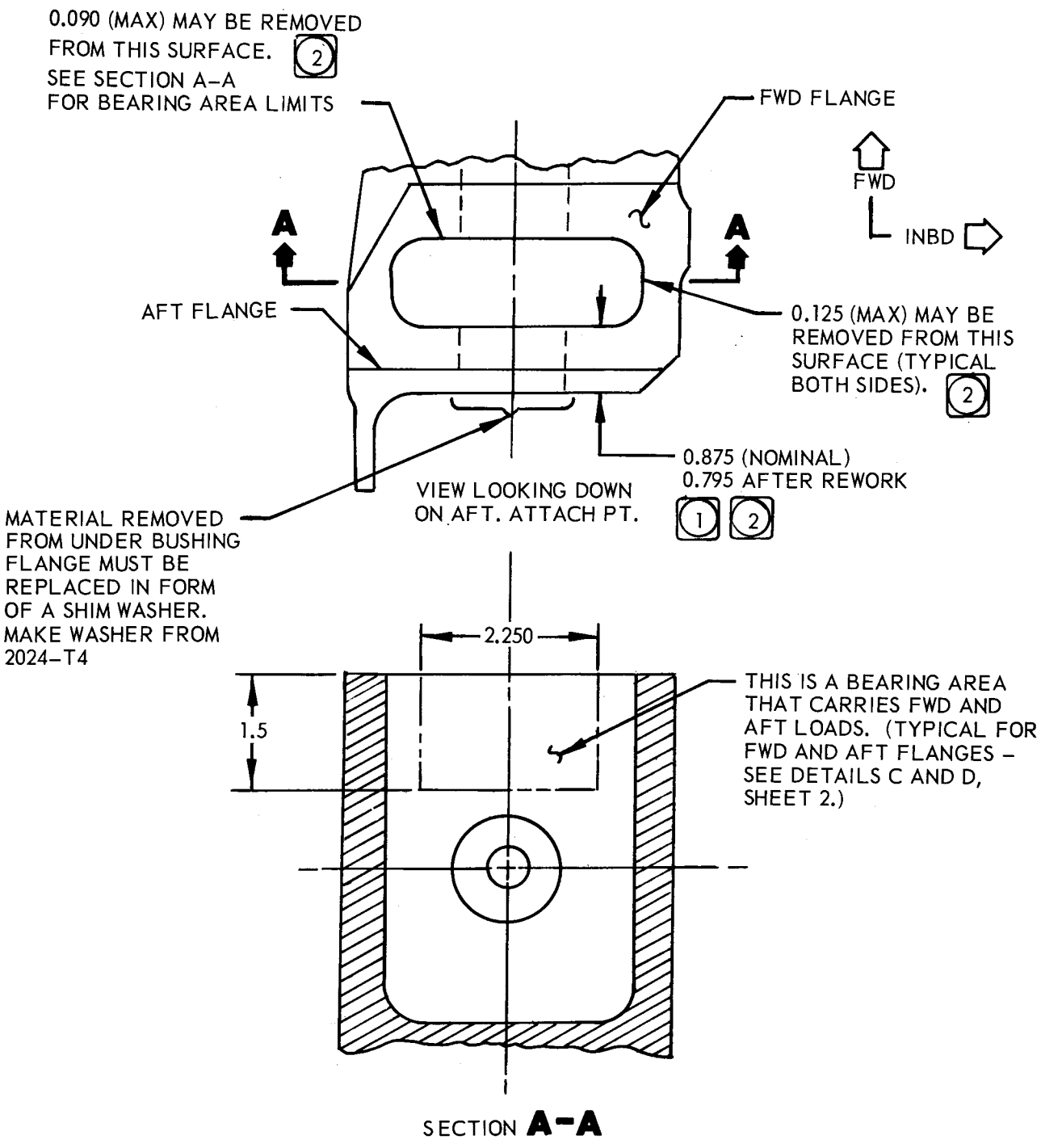
(a) If inspection reveals cracks of less than 1/8-inch depth into the material, perpendicular to the hole bearing surface, the cracks may be removed by boring the hole to the necessary oversize diameter. Boring should be made in 1/16-inch increments and reinspected after each cut. The hole should be reamed only to the oversize diameter required to remove corrosion and fitting. Remaining cracks should be removed in accordance with NAR drawing 9443–120021, (sheets 1 and



NOTES:

1. REFER TO TEXT FOR FURTHER INSTRUCTIONS.
2. THE MAXIMUM AMOUNT OF MATERIAL TO BE REMOVED FROM ONE SURFACE WITH HAND TOOLS IS 0.060 INCH, EXCEPT AS NOTED.
3. MATERIAL MAY BE REMOVED FROM ONE OR BOTH FACES.
4. MATERIAL MAY BE REMOVED, WITHIN LIMITS SHOWN, FROM ALL SURFACES SIMULTANEOUSLY.

Figure 3A-3. Rework of Forward Attach Bolt

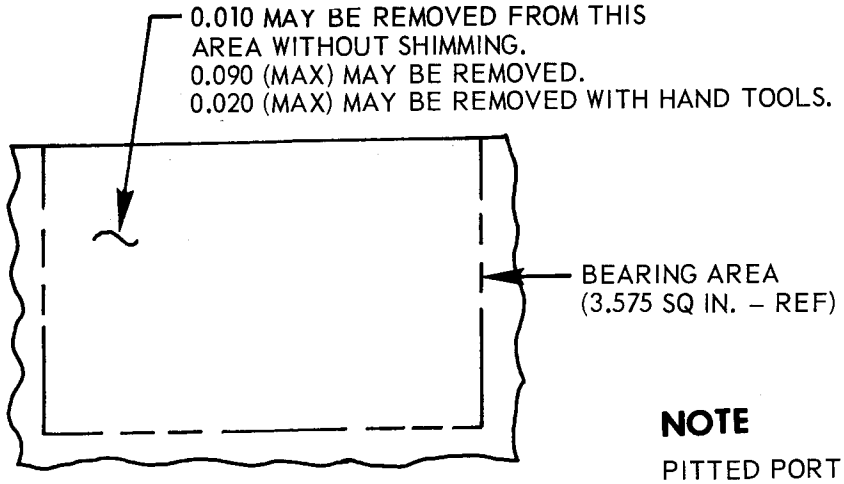


NOTES:

1. REFER TO NOTES ON SHEET 2.

REWORK OF AFT ATTACH POINT

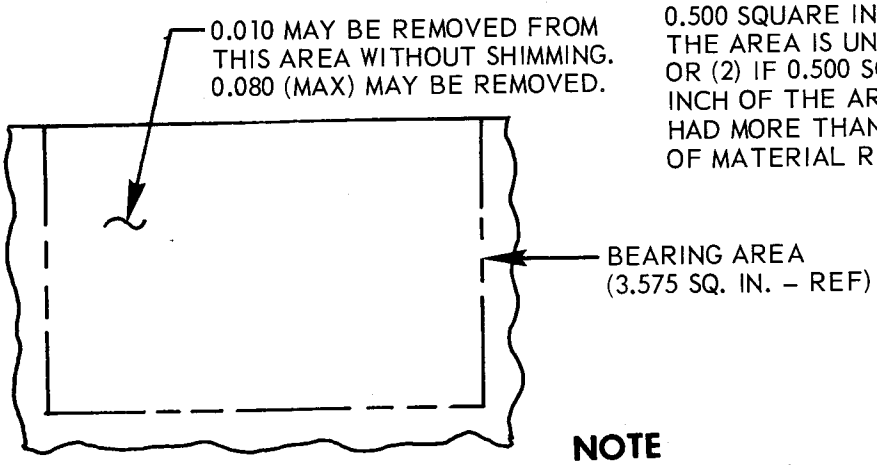
Figure 3A-4. Rework of Aft Attach Point (Sheet 1 of 2)



DETAIL C (INNER FACE OF FWD FLANGE)

NOTE

PITTED PORTIONS OF BEARING AREAS MAY BE CLEANED UP WITHOUT SHIMMING TO A DEPTH OF 0.090 (MAX), IF (1) 0.500 SQUARE INCH OF THE AREA IS UNAFFECTED OR (2) IF 0.500 SQUARE INCH OF THE AREA HAS NOT HAD MORE THAN 0.010 INCH OF MATERIAL REMOVED.



DETAIL D (INNER FACE OF AFT FLANGE)

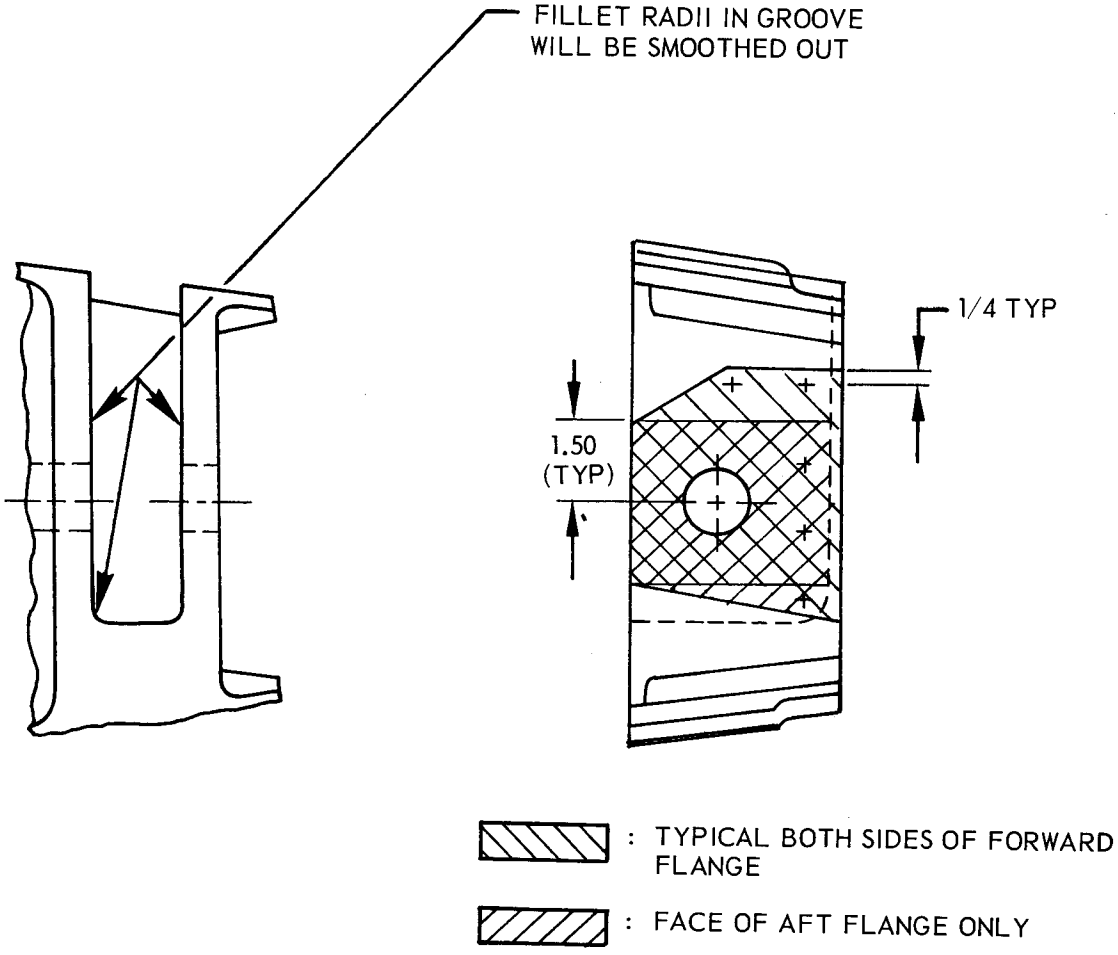
NOTE

IF THE CORROSION DAMAGE EXCEEDS THE LIMITS NOTED, SUBMIT DAMAGE CONDITON TO SMAMA ENGINEERING FOR EVALUATION.

NOTE:

1. REFER TO TEXT FOR FURTHER INSTRUCTIONS.
2. (2) THE MAXIMUM AMOUNT OF MATERIAL TO BE REMOVED FROM ONE SURFACE WITH HAND TOOLS IS 0.060 INCH. EXCEPT AS NOTED.
3. (1) MATERIAL MAY BE REMOVED FROM ONE OR BOTH FACES.
4. MATERIAL MAY BE REMOVED, WITHIN LIMITS SHOWN, FROM ALL SURFACES SIMULTANEOUSLY.

Figure 3A-4. Rework of Aft Attach Point (Sheet 2 of 2)



NOTE:

1. PITS & CORROSION IN CROSS-HATCHED AREAS WILL BE REMOVED MECHANICALLY AND SMOOTHED TO A MAXIMUM 125 R.M.S. FINISH.
2. PITS & CORROSION IN OTHER AREAS MAY BE CLEANED UP CHEMICALLY OR WITH DRY HONING. ALLOWABLE DEPTH OF PITS WITHOUT REQUIRING MECHANICAL SMOOTHING IS 0.060 INCH, OR 1/2 THE ALLOWWBLE REMOVAL DEPTH FOR THE PARTICULAR AREAS SHOWN IN FIGURE 3A-3 WHICHEVER IS LESS.
3. PITS ARE NOT ALLOWABLE ON THE BEARING SURFACES OF THE MAIN BOLT HOLES.

Figure 3A-5. Surface Corrosion Removal Requirements

2). Additional copies may be obtained from SMAMA, SMNBTA, McClellan AFB, California. (See bushing sizes and manufacturing information shown in figures 3A-6 and 3A-7. Do not exceed 1/4-inch oversize.)

(b) If a crack or cracks exceed the 1/8-inch depth, limitations as stated above, a doubler repair will be required as the crack cannot be removed by reaming.

(c) Fabricate doubler plate in accordance with figure 3A-8.

(d) Fabricate modified pulley brackets in accordance with figures 3A-9 and 3A-10, limitations as stated above, a doubler repair will be required. When reaming, the cutting should be to the next largest oversize diameter in 1/16-inch increments to remove the surface corrosion and fitting.

(e) Clean all bore aluminum alloy surfaces and adjacent painted surfaces by wiping with clean cloths dampened with methyl ethyl ketone (MEK) to remove grease. ***Wipe surfaces dry*** with a ***clean, dry cloth.*** If the MEK is allowed to evaporate, no cleaning has been accomplished.

NOTE

It is essential that clean cloths be used and that the solvent in the container be kept clean. Never dip cleaning cloth into the solvent. Pour it from the container onto the cloth.

Repeat the solvent cleaning operation until it is certain that no trace of oil or grease remains on the surface to be chemical filmtreated and painted.

(f) Deoxidize the cleaned base aluminum alloy surfaces using a solution of one part (by volume) of deionized or distilled water and one part (by volume) of cleaning compound (MIL-C-5410, Type II). Apply the deoxidizing solution with a clean soft bristle brush or with a clean cloth.

WARNING

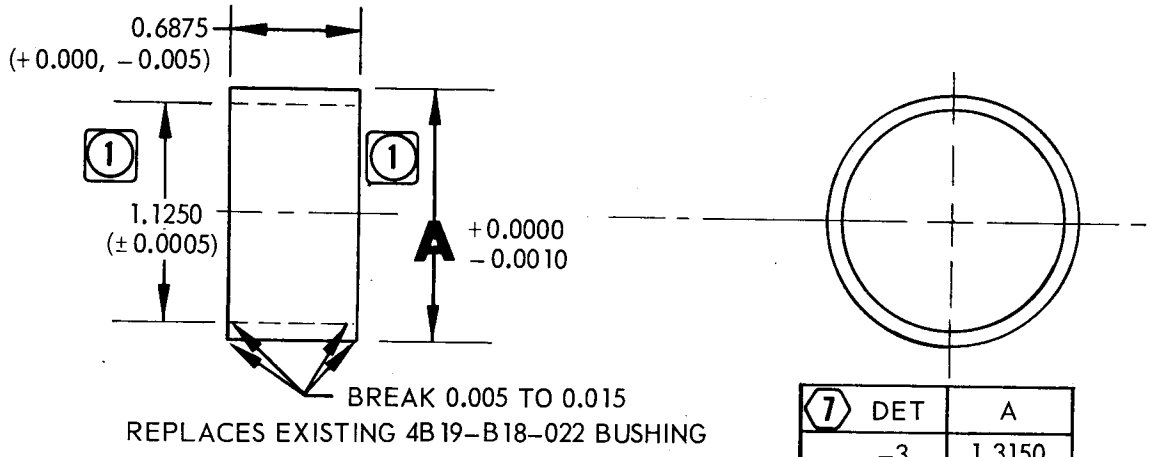
The cleaning compound is a mild acid. Wear rubber gloves to protect the hands. The surface must be kept wet with deoxidizing solution for a period of from five to ten minutes.

(g) Thoroughly rinse the deoxidizing solution from all affected parts with potable water. Wipe with clean, wet cloths frequently rinsed and wrung out in clean water. A water break free surface, must be obtained on the part. If water breaks appear on rinsing, repeat cleaning process.

(h) Dry the cleaned areas thoroughly with clean, dry cloths. Do not touch areas to be chemical treated with ***bare*** hands, or otherwise, permit them to become contaminated.

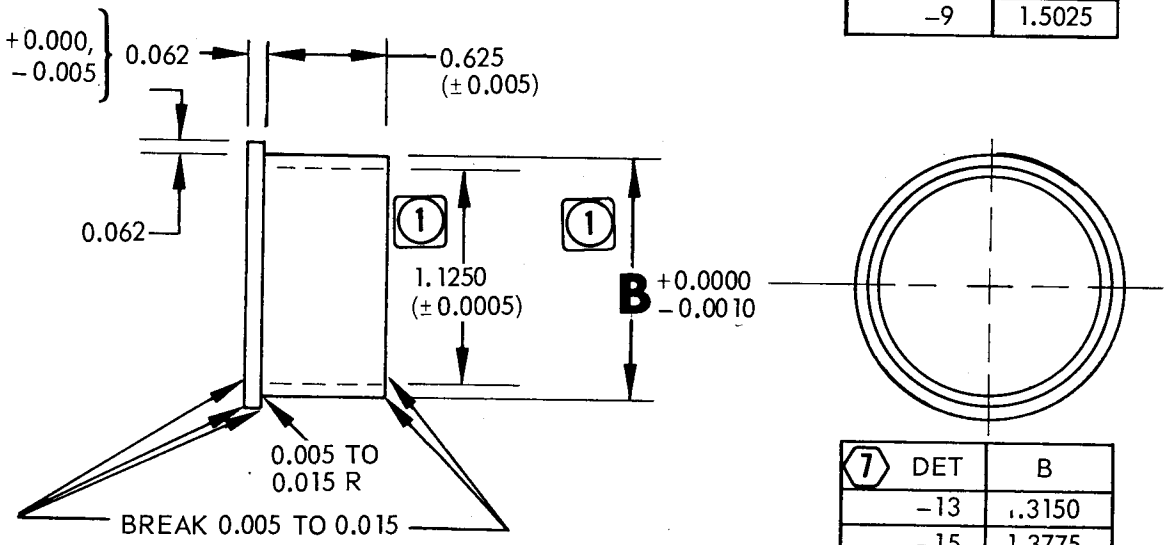
(5) Chemical Film Treatment:

(a) Alodine Number 1200 or Iridite No. 14 (materials which conform to Spec MIL-C-5541, Type I) must be used to chemically treat all base aluminum alloy surfaces. Instructions for both materials are provided.



REPLACES EXISTING 4B19-B18-022 BUSHING

7 DET	A
-3	1.3150
-5	1.3775
-7	1.440
-9	1.5025



REPLACES EXISTING 4B20-B18-020 BUSHING

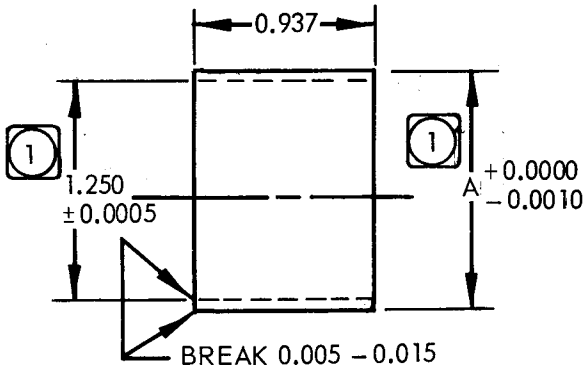
7 DET	B
-13	1.3150
-15	1.3775
-17	1.440
-19	1.5025

NOTES:

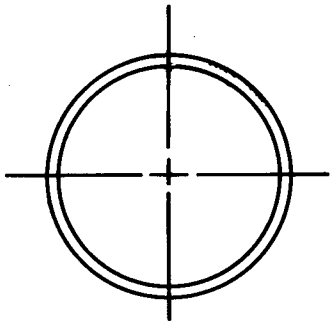
1. CADMIUM PLATE PER SPEC. QQ-P-416, TYPE II, CLASS 3, ALL SURFACES EXCEPT ID
2. 125 ALL MACHINED SURFACES.
3. MATERIAL-AL BRONZE BAR, AMS4631.
4. (1) MAX. ECCENTRICITY 0.003 TIR
5. AFTER INSTALLATION IN RIB ASSEMBLY REAM ID TO $1.1250^{+0.0005}_{-0.0015}$ DIAMETER IF NECESSARY

(7) THE DIMENSIONS SHOWN ARE FOR REFERENCE ONLY. THE DASH NUMBERS NOTED ARE IDENTICAL TO NAA DRAWING 9443-120011.

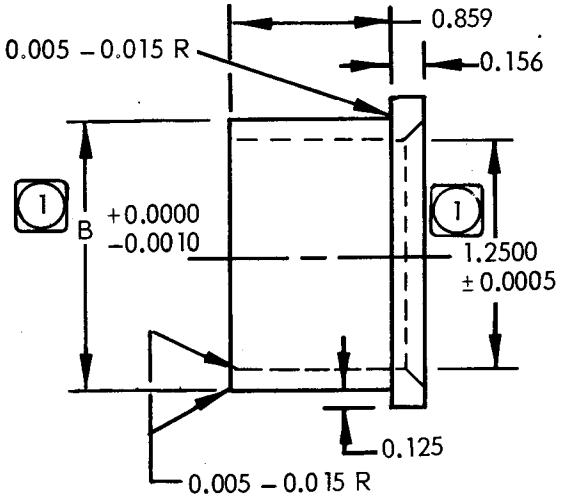
Figure 3A-6. Oversize Bushings for Wing Root Rib Fwd Fuselage Attach Point

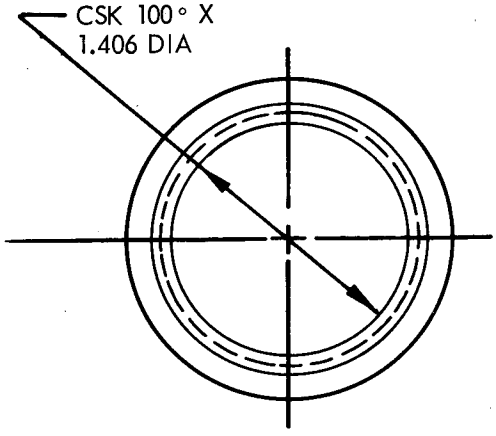


REPLACES EXISTING 180-14710 BUSHING



⟨8⟩ DET	A
-3	1.440
-5	1.5025
-7	1.5650
-9	1.6275





REPLACES EXISTING 180-14709 BUSHING

⟨8⟩ DET	B
-13	1.440
-15	1.5025
-17	1.5659
-19	1.6275

⟨8⟩ THE DIMENSIONS SHOWN ARE FOR REFERENCE ONLY. THE DASH NUMBERS NOTED ARE IDENTICAL TO NAA DWG 9443-120012

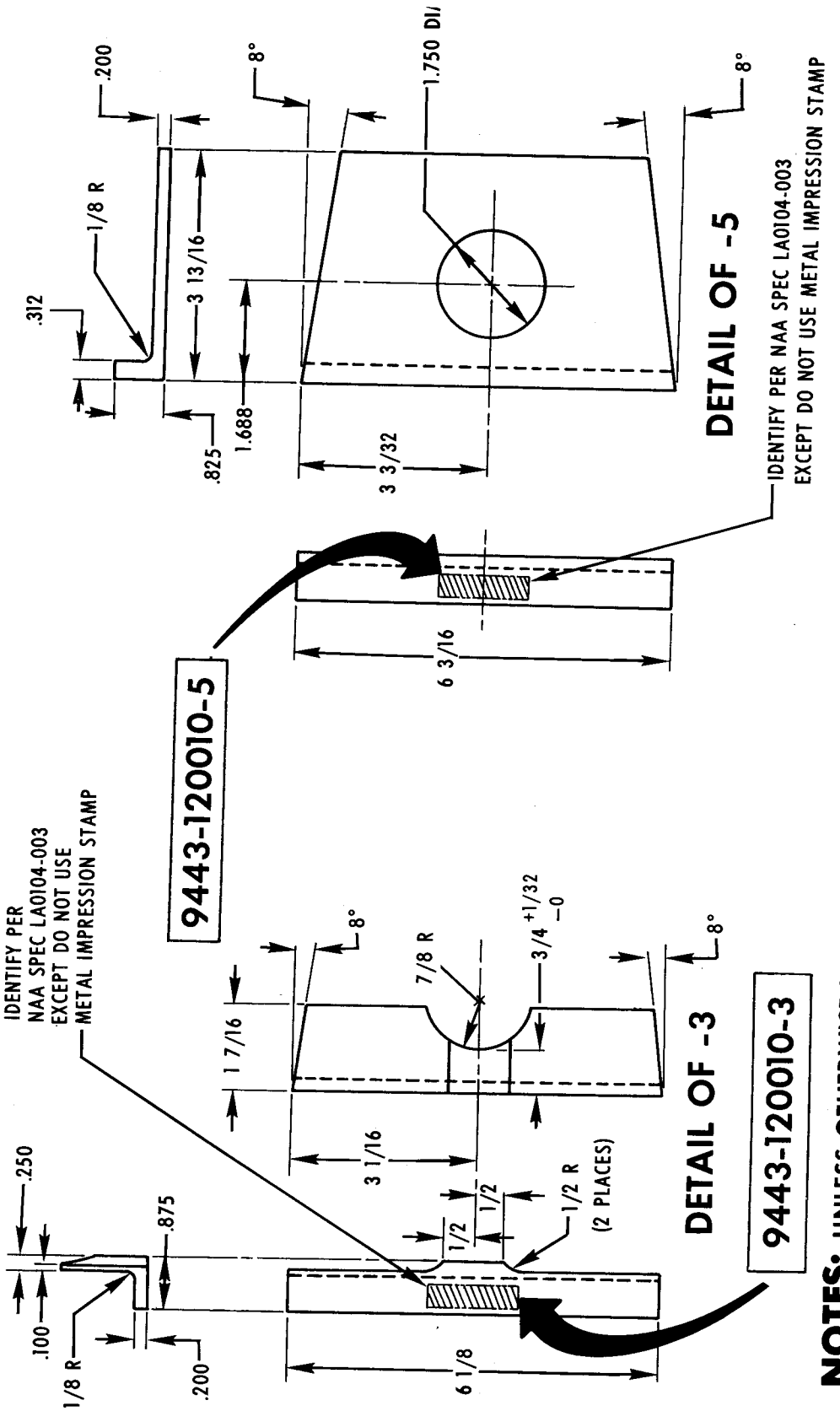
NOTES:

1. CADMIUM PLATE PER SPEC QQ-P-416, TYPE II, CLASS 3, ALL SURFACES EXCEPT ID.
2. 125√ ALL MACHINED SURFACES.
3. MATERIAL – BERYLLIUM COPPER, SPEC QQ-C-539, COND A. HEAT TREAT TO COND H AFTER MACHINING.

⟨1⟩

4. CONCENTRIC WITHIN 0.003 TIR
5. AFTER INSTALLATION IN RIB ASSEMBLY REAM ID TO 1.250 $^{+0.0005}_{-0.0015}$ DIAMETER IF NECESSARY

Figure 3A-7. Oversize Bushings for Wing Root Rib Aft Fuselage Attach Point

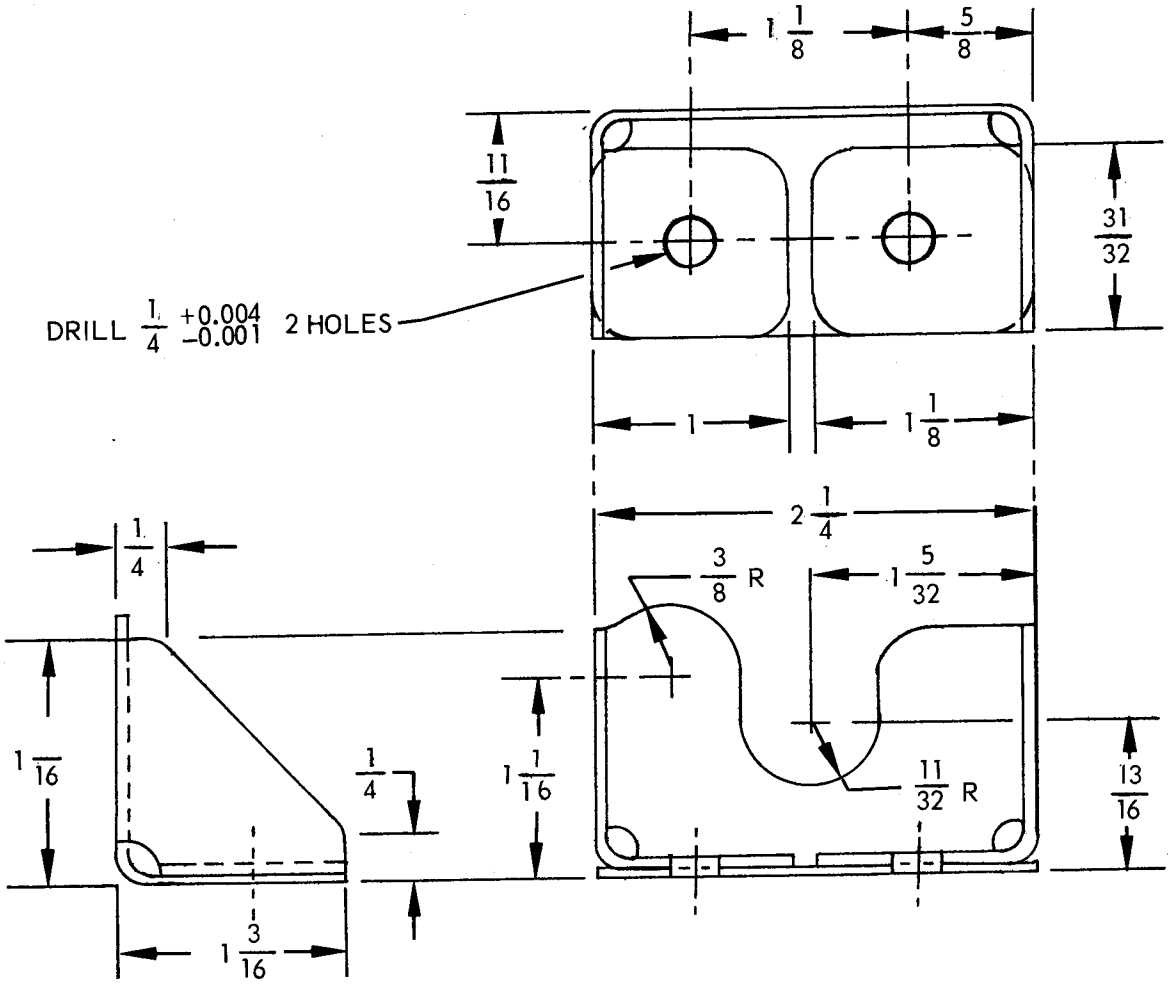


NOTES: UNLESS OTHERWISE NOTED

1. 125 ALL MACHINED SURFACES
2. CAD PLATE -3 AND -5 PER SPEC QQ-P-416, TYPE II, CLASS 3
3. THIS PART TO BE USED IN THE REWORK OF F-100 SERIES A/C WING OUTER PANEL ROOT RIBS WITH CORROSION OR STRESS-CORROSION DAMAGE
4. MACHINE PER NAA SPEC LA0103-004
5. ALTERNATE MATL-4340 STL BAR, SPEC-MIL-5000 COND C4
6. DELETED ALL PILOT HOLES IN -3 DETAIL

DWG NO 9443-120010

Figure 3A-8. Plate, Reinforcing, Wing Root Rib, Outer Panel, Fwd Attach Pt.

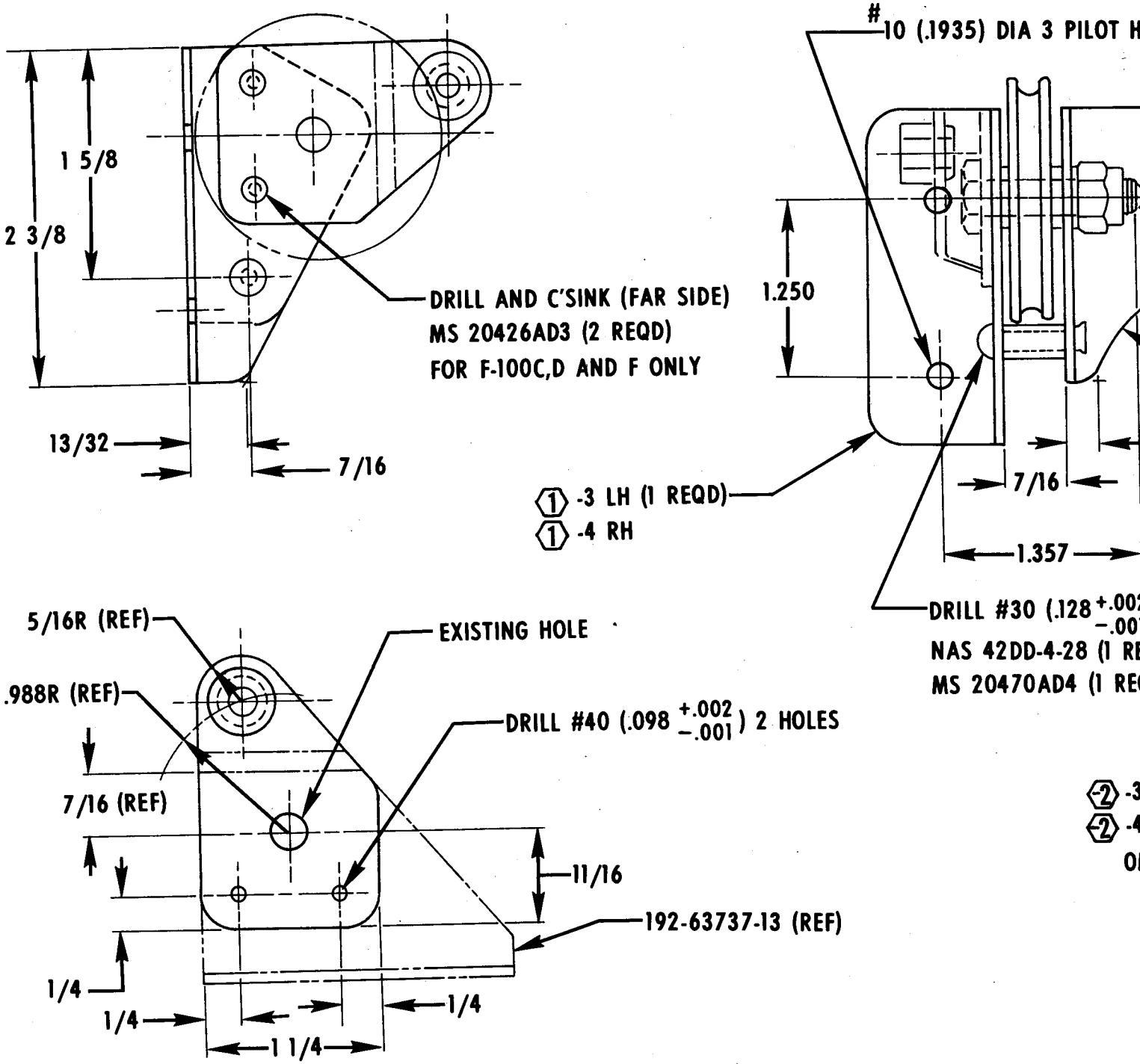


DETAIL -3 ⟨1⟩

NOTES: UNLESS OTHERWISE NOTED

1. BEND RADIUS 1/16
2. OUTSIDE CORNER RADII 1/4
3. INSIDE CORNER & BEND RELIEF RADII 5/32
4. CHEM FILM PER SPEC. MIL-C-554 & APPLY 2 COATS ZINC CHROMATE PRIMER PER MIL-P-8585.
5. MAKE FROM ALCLAD 2024-0 AL. SHEET, PER QQ-A-362. SIZE: 0.051 X 2-5/8 X 4-5/8
6. HEAT TREAT TO TEMP T42 PER MIL-H-6088 AFTER FORMING.
⟨1⟩ 7. THE DIMENSIONS SHOWN ARE FOR REFERENCE ONLY. THIS PART IS IDENTICAL TO NAA DWG 9443-120015.

Figure 3A-9. Modified Support Bracket

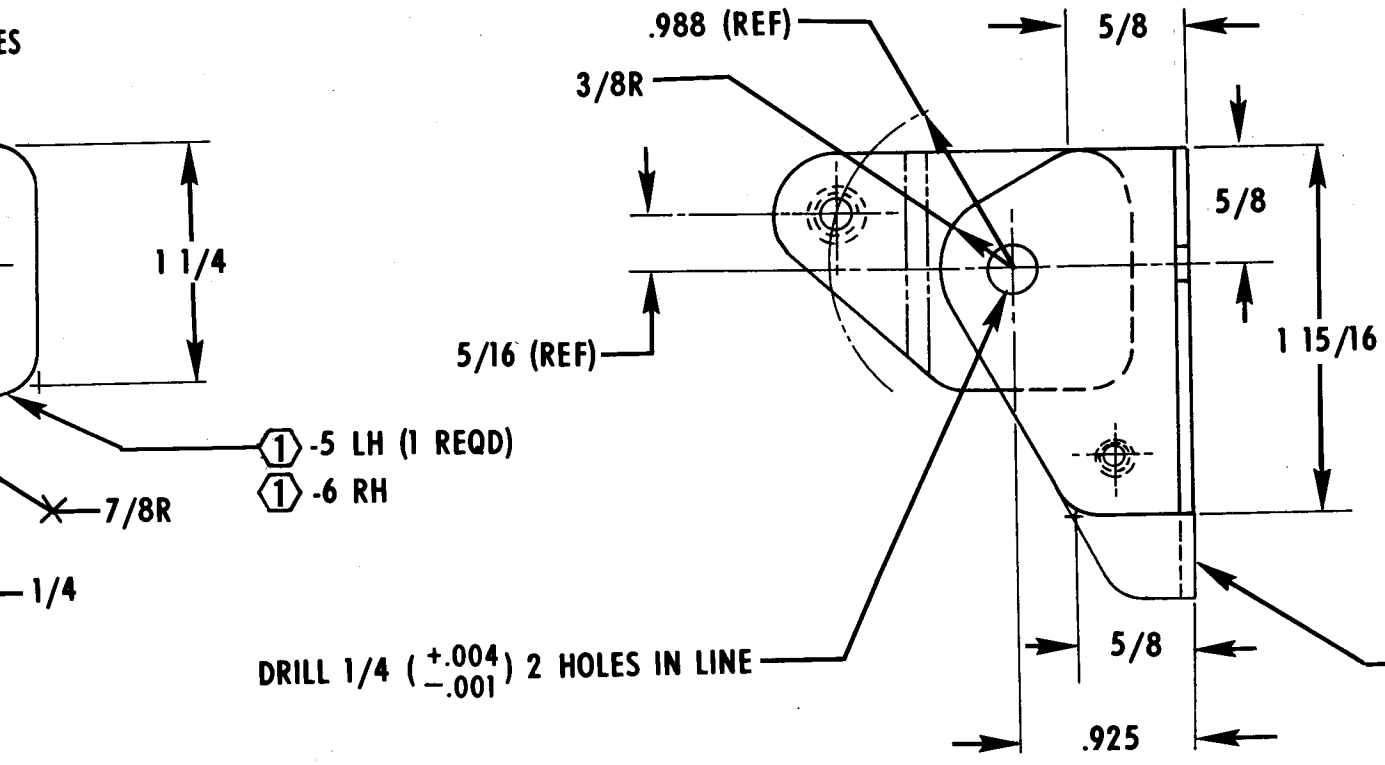


DETAIL OF ② -3 SHOWN ② -4 OPPOSITE

MAKE FROM 192-63737-13 FOR LH, -14 FO

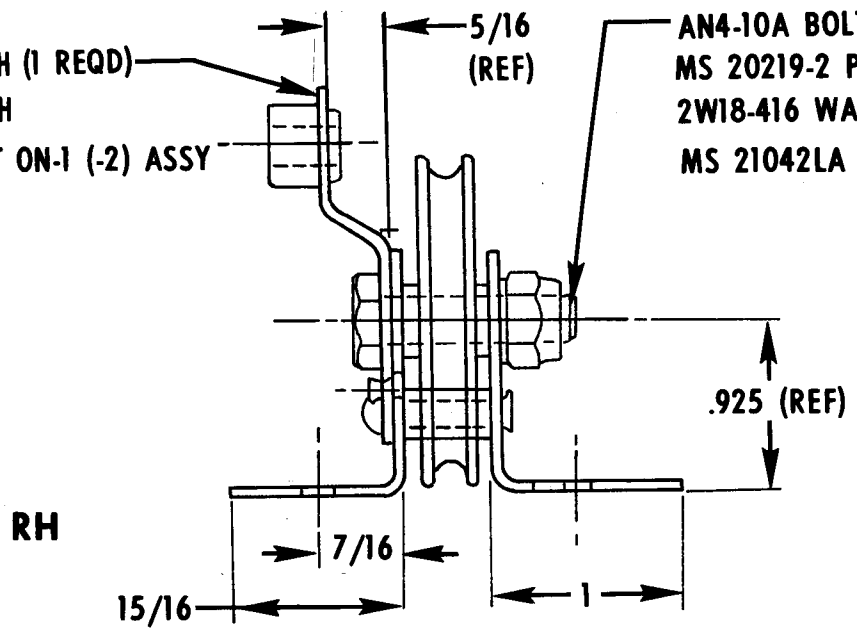
FOR F-100C D, AND F [ON -11 (-12) ASSY]

ES



LH SH
RH OP

2 HOLES IN LINE
)



NOTES: UNLESS OTH

1. BEND RADIUS 3/32
2. OUTSIDE CORNER RADII
3. MAKE 1 -3, -4, -5 AND
   HEAT TREAT TO TEMP T4
4. CHEM FILM ALL PARTS P
   ZINC CHROMATE PRIME

LH SHOWN
RH OPPOSITE

①	ASSY				
-12	-11	-2	-1	DASH NO.	DESC
	1			-1	ASSY
1				-2	ASSY
			1	-3	ANGLE
		1		-4	ANGLE
			1	-5	ANGLE
		1		-6	ANGLE
	1			-3	BRACKET
1				-4	BRACKET

-1 ASSY LH (1 REQD)
-2 ASSY RH (1 REQD) } USED ON F-100A

-11 ASSY LH (1 REQD)
-12 ASSY RH (1 REQD) } USED ON F-100C,D AND F

NG PULLEY ASSY MAY BE REUSED

LESS OTHERWISE NOTED
3/32
IER RADII 1/4
, -5 AND -6 FROM .064, 2024-0 ALCLAD AL SH QQ-A-362
TEMP T42 PER MIL-H-6088 AFTER FORMING FOR ② -3 AND -4 SEE DETAIL
L PARTS PER SPEC MIL-C-554 AND APPLY (2) COATS
ATE PRIMER PER MIL-P-8585

Figure 3A-10. Modified Pulley Brackets

WARNING

Wear rubber gloves when mixing and appyling either of the chemical film treatments. Brushes and cloths used for applying the solutions must not be allowed to dry, they constitute a fire hazard. Immediately after use, all such materials shall be thoroughly rinsed in clear potable water; after such rinsing they may be re-used or discarded.

Breathing of dust from the make-up powders and inhalation of vapors from the solutions shall be avoided as much as possible.

Make-up chemicals shall be stored near easily oxidizable material such as paints, solvents, lubricants, etc.

Alodine No. 1200

MATERIAL	SOURCE, OR SPECIFICATION
Alodine No. 1200	MIL-C-5541
Nitric Acid 61HNO_3 By Wt	Fed Spec O-N-350
Nonmetallic Brush or Cloth Pad	Commercial

Mix three ounces by weight of Alodine No. 1200 (Spec MIL-C-5541) and 0.5 fluid ounce of nitric acid (Fed Spec O-N-350) with one gallon of water.

Apply solution with a nonmetallic brush or small cloth pad. Keep surface wet with solution for three to five minutes.

If no color develops, repeat cleaning and chemical film treatments.

Rinse by flushing with potable water and allow area to air-dry. Avoid rubbing as the protective coating is easily damaged.

A continuous, unscratched film is necessary. Alodine No. 1200 will vary in shade from a light, brassy iridescent to a golden brown or a heavy, dark, dull brown. The light shades are superior to the darker but identical color matches between different surfaces need not be maintained. These variations in color tone are not indicative of poor quality of a chemical film. A gray appearance, or a powdery nonadherent coating, indicates films of unsatisfactory quality and surfaces with these films must be stripped and recoated.

Iridite No. 14

MATERIAL	SOURCE, OR SPECIFICATION
Iridite No. 14	MIL-C-5541
Alkyl Sufonate	Fed Stock No. 6550-250-2483
NonMetallic brush or cloth pad	Commercial
Spatula (or equivalent)	Commercial

Mix six ounces with a nonmetallic brush or small cloth pad. Keep surface wet with solution for three to five minutes.

If no color develops, repeat cleaning and chemical film treatments.

A continuous, unscratched finish is necessary. Iridite No. 14 will vary in shade from a dark brassy iridescence to a golden brown or a heavy, dark, dull brown. The light shades are superior to the darker but identical color matches between different surfaces need not be obtained. These variations in color tone are not indicative of poor quality in a chemical film. A gray appearance or a powdery nonadherent coating indicates a film of unsatisfactory quality and surfaces with this film must be stripped and recoated.

Priming

MATERIAL	SPECIFICATION
Stromtium Primer	MIL-P-23377B
Toluol	TT-T-S48 Federal Stock No. 6810-281-2002

Apply two light coats of thinned stromtium chromate primer to all bare surfaces (except bushing holes) that have been chemically treated. Masking of adjacent stromtium chromate primed areas is not required.

Plug bushing holes with suitable corks.

Thin primer by mixing one part (by volume) primer with from two to two and one-half parts (by volume) thinner.

Spray a uniform coat of thinned primer to a depth that provides complete hiding. Avoid any "build-up" of the coating, giving particular attention to build-up in inside corners.

Allow primer to dry for a minimum of four hours at room temperature before installing bushings or doubler.

Install fabricated doubler plate (when required) in accordance with NAR Drawing 9443-120021, sheet 1 and sheet 2 previously furnished by the Air Force. Additional copies can be obtained from SMAMA, SMNBTA, McClellan AFB, California.

Install modified pulley brackets, as directed in paragraph 1.(4)(d) in accordance with figure 3A-9 and 3A-10.

m. Installation of Bushings.

(1) Bushings are to be installed after aluminum housing has been refinished as directed earlier in this rework procedure. It is mandatory that a 0.001 to 0.003 interference fit be maintained.

TOOLS AND MATERIALS REQUIRED	SOURCE OF SPECIFICATIONS
Bushing installation Tool AF part No. 61SCD6107 (figure 3A-11)	AF Supply
Boring Tool (Reamer) 66D44553	AF Supply
Methyl Ethyl Ketone (MEK)	TT-M-261
Primer, Chromate	MIL-P-23377B
Cloth, Cleaning	Commercial

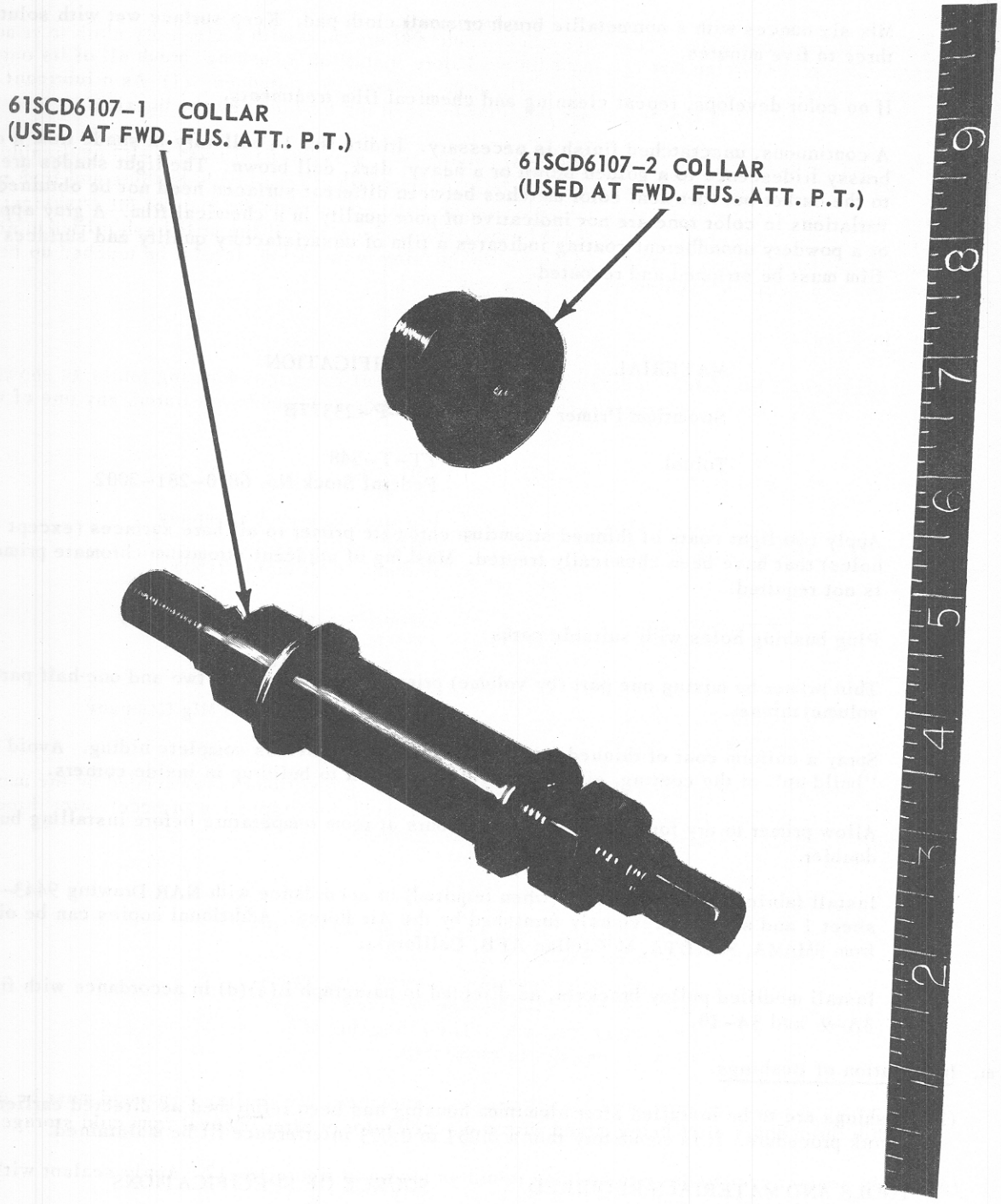


Figure 3A-11. Bushing Installation Tool (61SCD6107)

(2) Wash bushing thoroughly with methyl ethyl ketone and dry them with a clean dry cloth to remove grease and other foreign matter. Immediately before installing a bushing, brush all of its outer contact surfaces with a coat of primer. The primer serves two purposes: (1) As a lubricant for installation, and (2) When dry, seals the joint between the bushing and the housing.

(3) Press the primer-lubricated bushing in place using tool, part No. 61SCD6107 (with applicable collar), as shown in figure 3A-11. After pressing, wipe excess primer from inside bushing and from surrounding areas. Be extremely careful, however, not to wipe primer from the ends of the bushings; thereby, destroying the joint seal. Check ID of bushing for proper size. If the still fresh primer in areas around the bushing has been marred during the bushing installation, it must be touched up before the bushing end joints are sealed with sealing compound.

n. Sealing Bushings

(1) When the primer has dried; approximately four hours, seal bushing-to-housing joints as shown in figure 3A-12. Three commercial general purpose sealing compounds are listed, any one of which may be used.

MATERIAL	SOURCE
PR 1222 Compound and PR1222A Accelerator	Products Research Company 2919 Empire Avenue Burbank, California, 91504
707 Compound and 707 Accelerator	Coast Proseal and Mfg Company 2235 Beverly Blvd. Los Angeles, California
EC-1528 Compound and EC-1528 Accelerator	Minnesota Mining and Mfg Company St Paul, Minnesota

(2) Mix the base compound and the accelerator together in proportions recommended by the manufacturer. The two components must be blended until there is a uniform dispersion of accelerator through the mixture. Improperly mixed sealant will not cure properly.

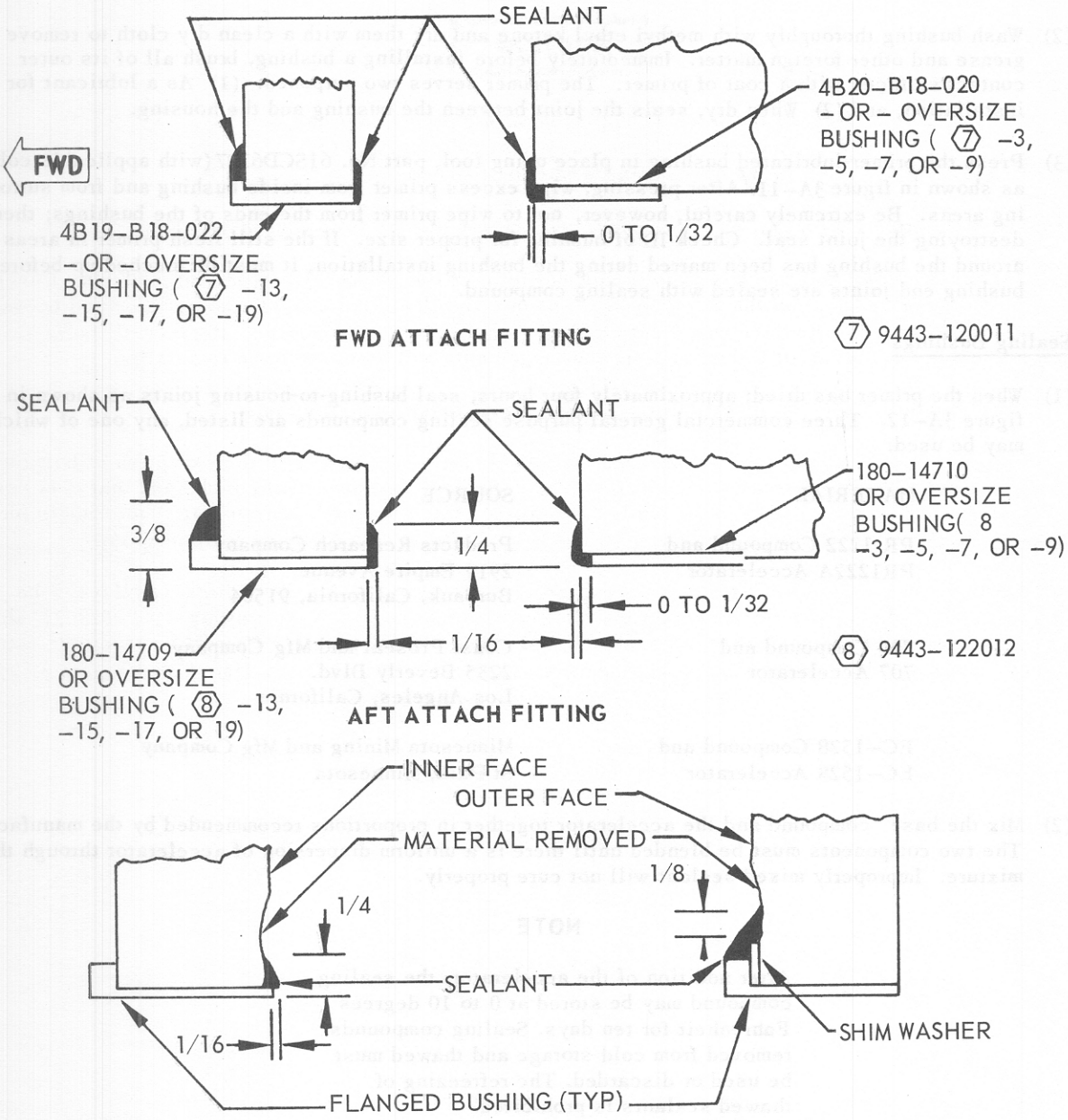
NOTE

After addition of the accelerator, the sealing compound may be stored at 0 to 10 degrees Fahrenheit for ten days. Sealing compounds removed from cold storage and thawed must be used or discarded. The refreezing of thawed sealants is prohibited.

(3) At room temperature (approximately 77 degrees Fahrenheit) the sealing compound must be used within three hours after being mixed with the accelerator or after removal from cold storage.

(4) Apply sealant over bushing-to-housing joints as shown in figure 3A-12. Apply sealant with a small spatula or equivalent tool.

(5) Allow sealant to dry for a minimum of eight hours before subjecting it to handling.

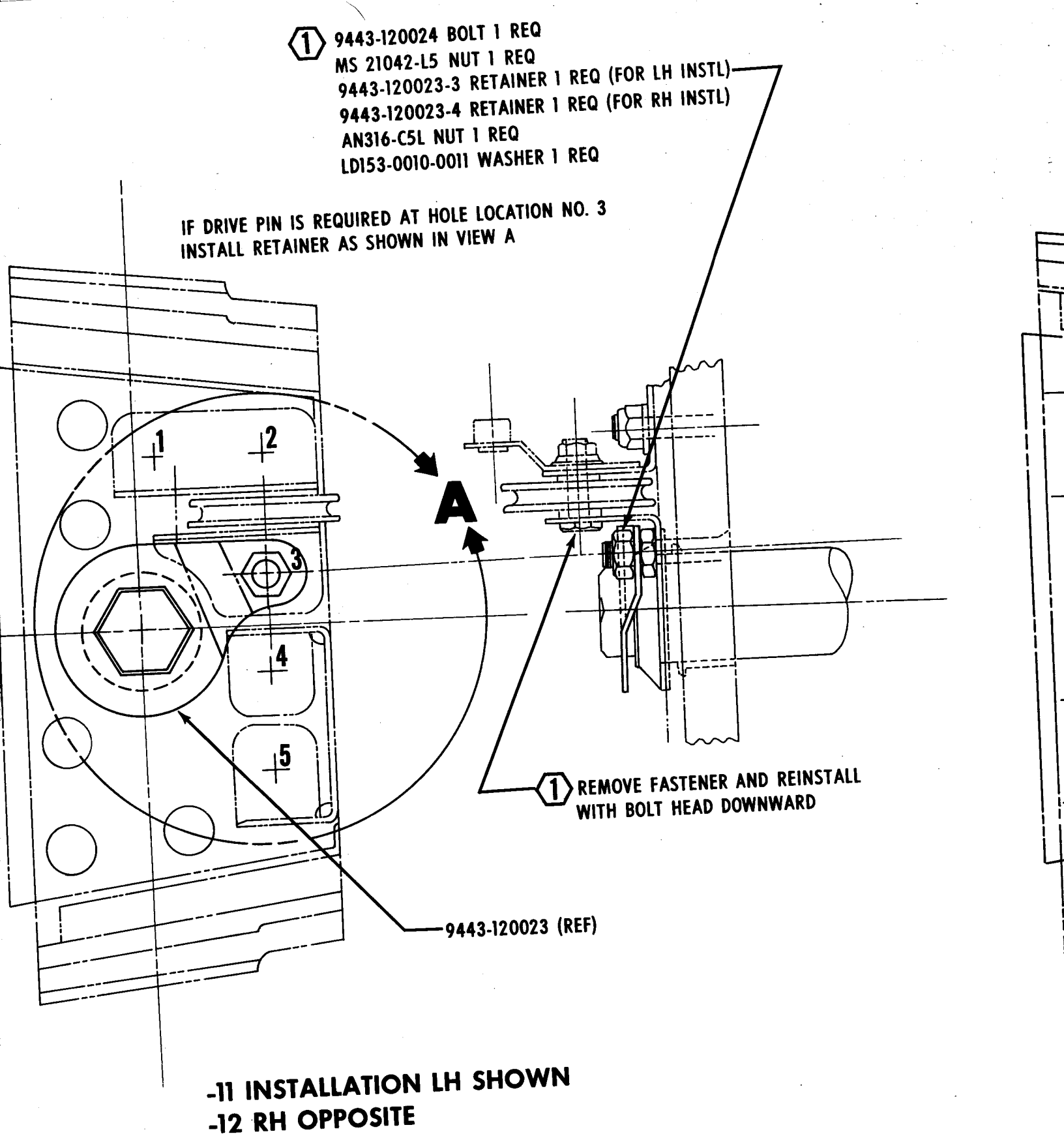


POSSIBLE SEALING CONDITIONS AFTER REWORK

NOTE

- SEALANT BEAD DIMENSIONS ARE APPROXIMATE.
- DIMENSIONS SHOWN FOR AFT ATTACH FITTING APPLY TO FWD ATTACH FITTING ALSO
- REFER TO "SEALING" IN TEXT FOR MATERIALS AND FURTHER INSTRUCTIONS.

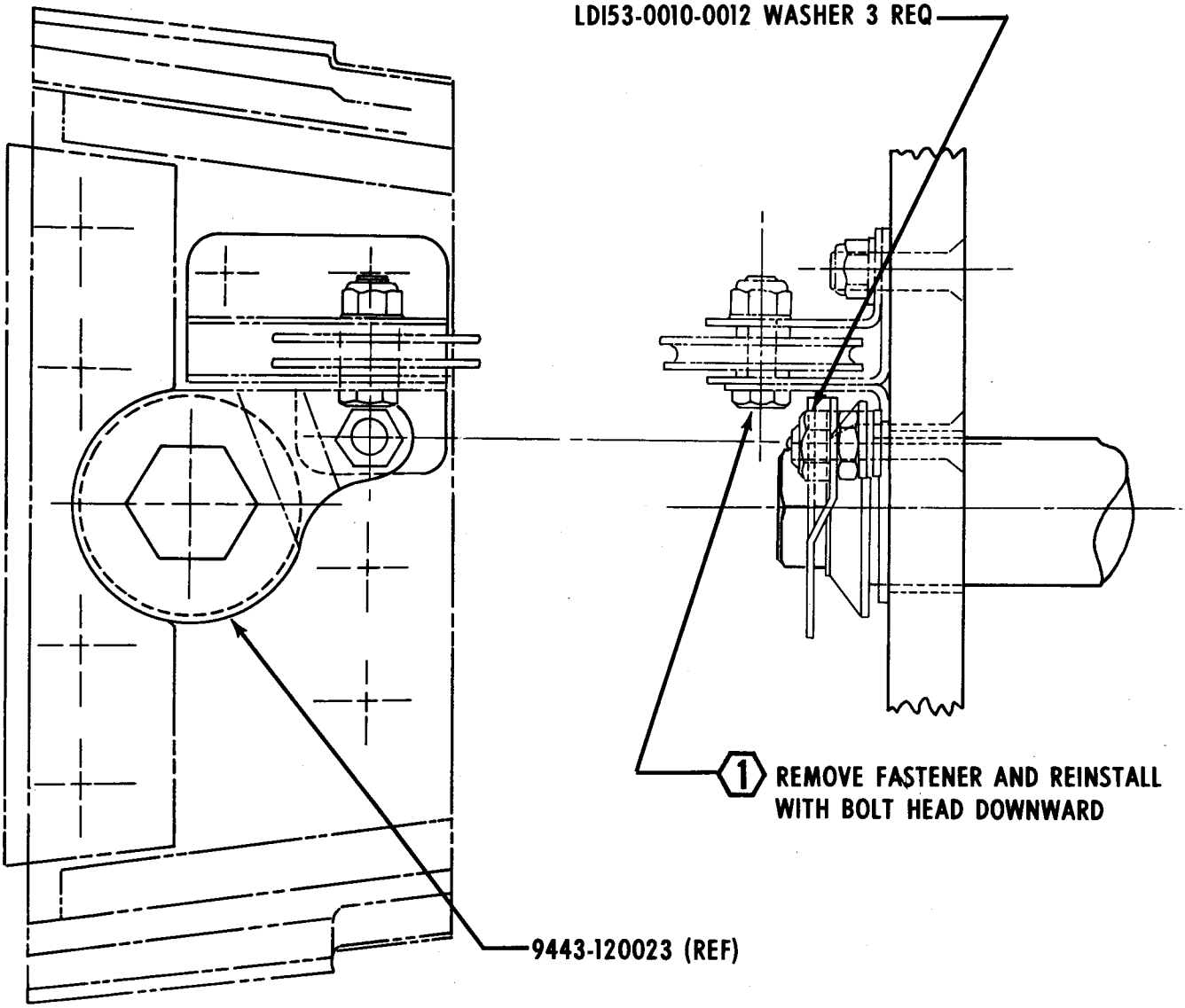
Figure 3A-12. Sealing Bushings



-11 INSTALLATION LH SHOWN
-12 RH OPPOSITE

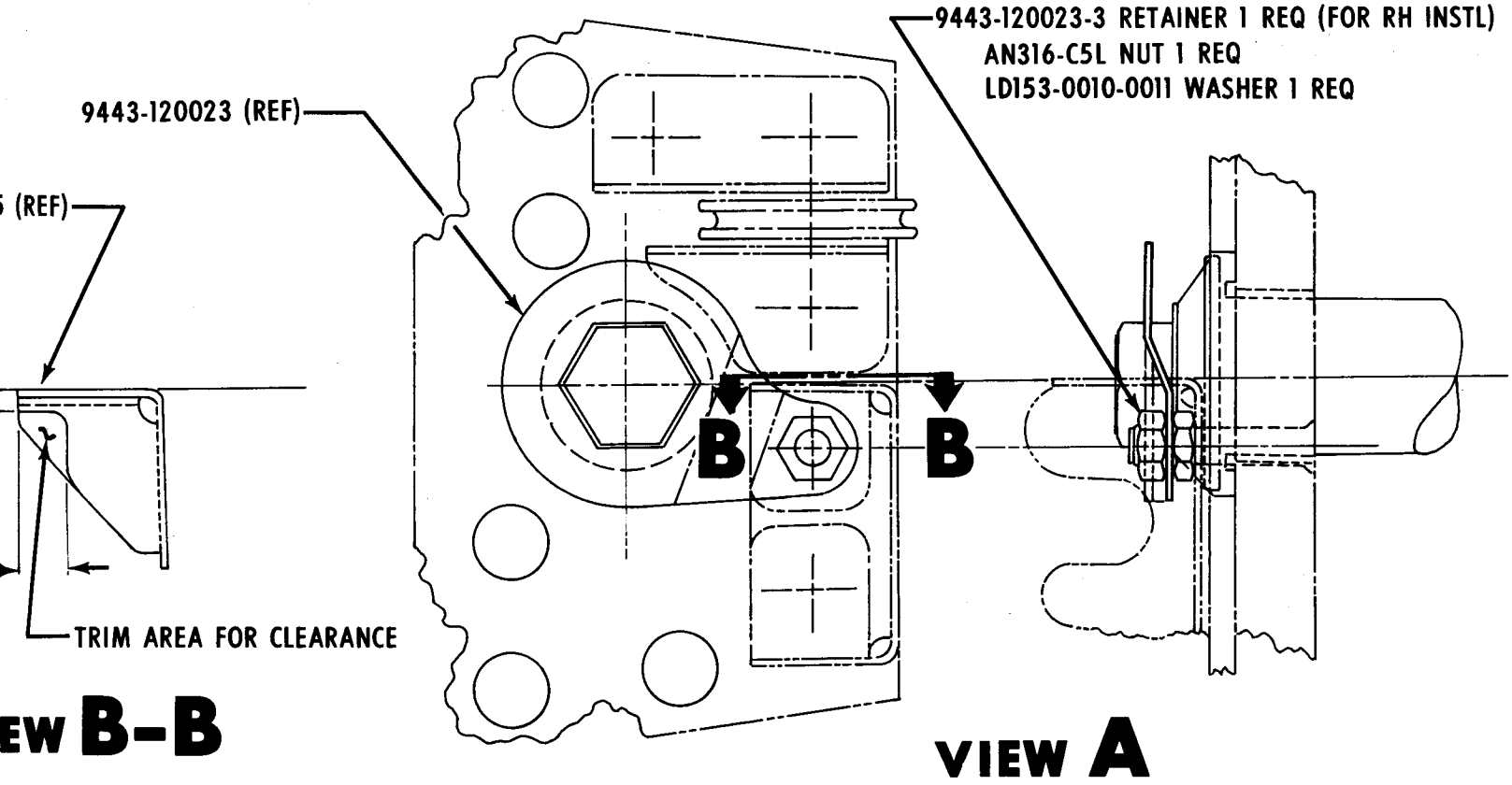
REMOVE EXISTING NAS221-9 SCREW
REMOVE ROSAN INSERT FROM HOLE IN ROOT RIB

(1) 9443-120024 BOLT 1 REQ
MS 21042-L5 NUT 1 REQ
9443-120023-3 RETAINER 1 REQ (FOR LH INSTL)
9443-120023-4 RETAINER 1 REQ (FOR RH INSTL)
AN316-C5L NUT 1 REQ
LDI53-0010-0012 WASHER 3 REQ



-21 INSTALLATION LH SHOWN
-22 RH OPPOSITE

NOTE ON THIS INSTALLATION USE THE
RH RETAINER (9443-120023-4) ON THE LH INSTALLATION
AND THE LH RETAINER 9443-120023-3 ON THE RH INSTALLATION



S:
SS OTHERWISE NOTED
DARD DETAIL PER LA0102-012
ALL BOLTS PER T.O. 1F-100-1006

DWG NO. 9443-120022

QTY REQ	QTY REQ	QTY REQ	QTY REQ	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION
-22	-21	-12	-11	9443-120022	INSTALLATION
	1		1	9443-120023-3	RETAINER
1		1		9443-120023-4	RETAINER
1	1	1	1	9443-120024	BOLT
		1	1	LD153-0010-0011	WASHER
3	3			LD153-0010-0012	WASHER
1	1	1	1	AN316-C5L	NUT
1	1	1	1	MS 21D42-L5	NUT

Figure 3A-13. Retainer Instl., Bolt – Fwd Wing Attaching

PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	MATERIAL	DATA SPECIFICATION SIZE NOTES SUPPLIERS
9443-120022			
-3	RETAINER	301 CRES SH	MIL-S-5059 ½ H .063 X 3.25 X 2.0
-4	RETAINER	301 CRES SH	MIL-S-5059 ½ H .063 X 3.25 X 2.0

NOTES:

UNLESS OTHERWISE NOTED

1. BREAK SHARP EDGES
2. BEND RADII 1/8
3. FABRICATE PER LA0102-012
4. MACHINE PER LA0103-004

DWG NO. 9443-120023

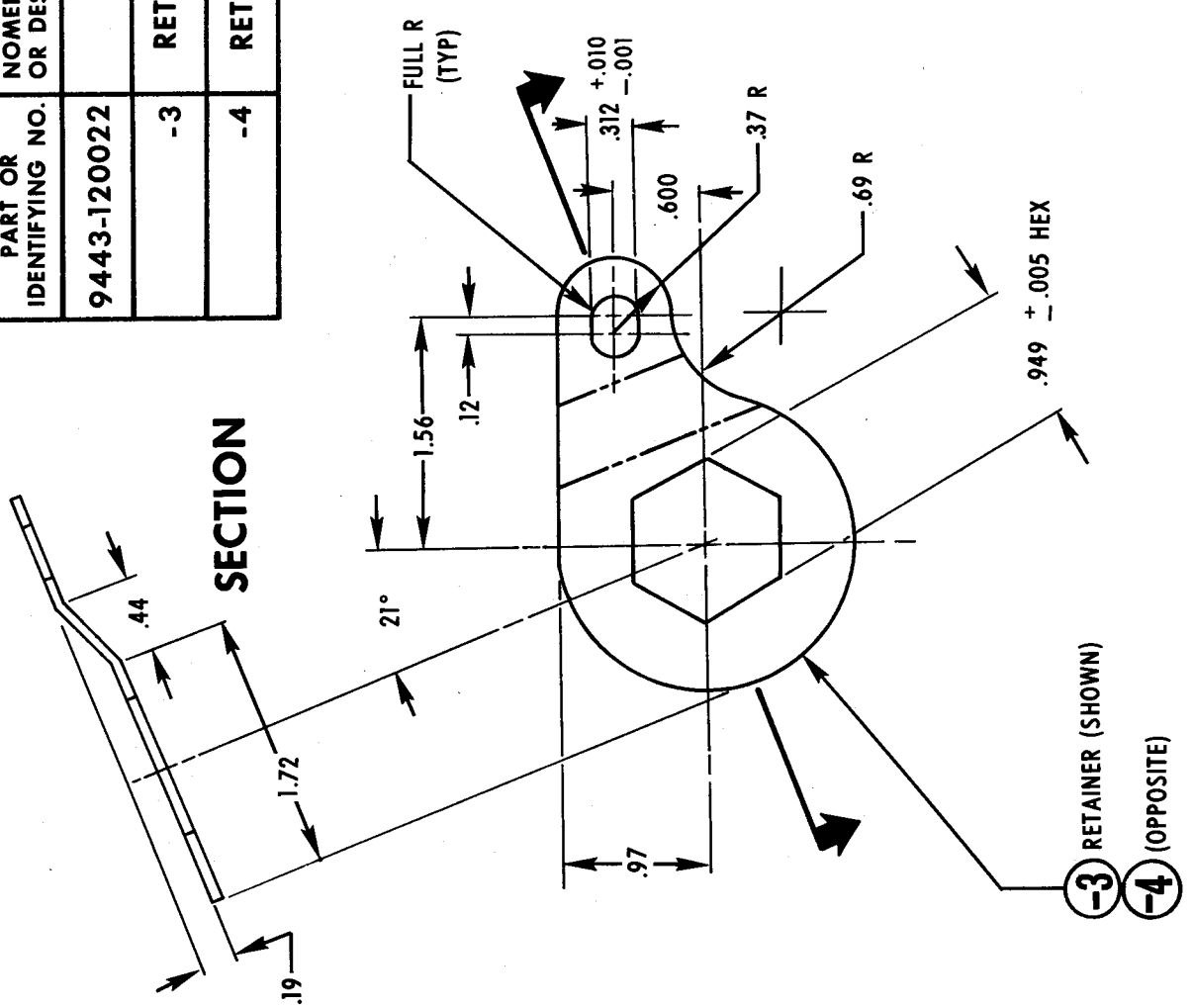


Figure 3A-14. Retainer Bolt – Fwd Wing Attaching

CAUTION

If the fresh stromtium chromate primer coat has been marred in any way during the sealing (or prior) operations, it must be touched up.

(6) All wing attach bolts that have been removed will require the following inspection:

(a) Examine the bolt and barrel nut using magnetic or fluorescent dye-penetrant methods. Check for wear or fractures at the thread base. The nut fibre lock ring should also be examined for wear or distortion. Replace barrel nuts having thread or lock ring damage. Replace all bolts having thread distortion or fractures. All undamaged nuts and bolts are to be reinstalled.

(b) Reinstall wiring on fuselage in accordance with existing instructions.

(c) Install the attaching bolts per T.O. 1F-100( )-2-1, Section 9 with the exception of bolt torque values. Make certain that the weight of the fuselage is supported by jacks and that it is not resting on the bolts or joints while the bolts are being torqued. The torque requirements for these bolts are shown in T.O. 1F-100D(I)-2-1, Page 9-5, Changed 9 October 1967.

(d) After torquing, install the bolt lock in accordance with T.O. 1F-100D-1006C on all four bolts. However, on wings having a doubler installed in accordance with NAR drawing 9443-120021, sheets 1 and 2 (previously furnished by the Air Force), a special locking clip will be required. Parts and drawings for the clip can be obtained from SMAMA (SMNBTA), McClellan AFB. Figure 3A-13, 3A-14, and 3A-15, will be used when installing doubler.

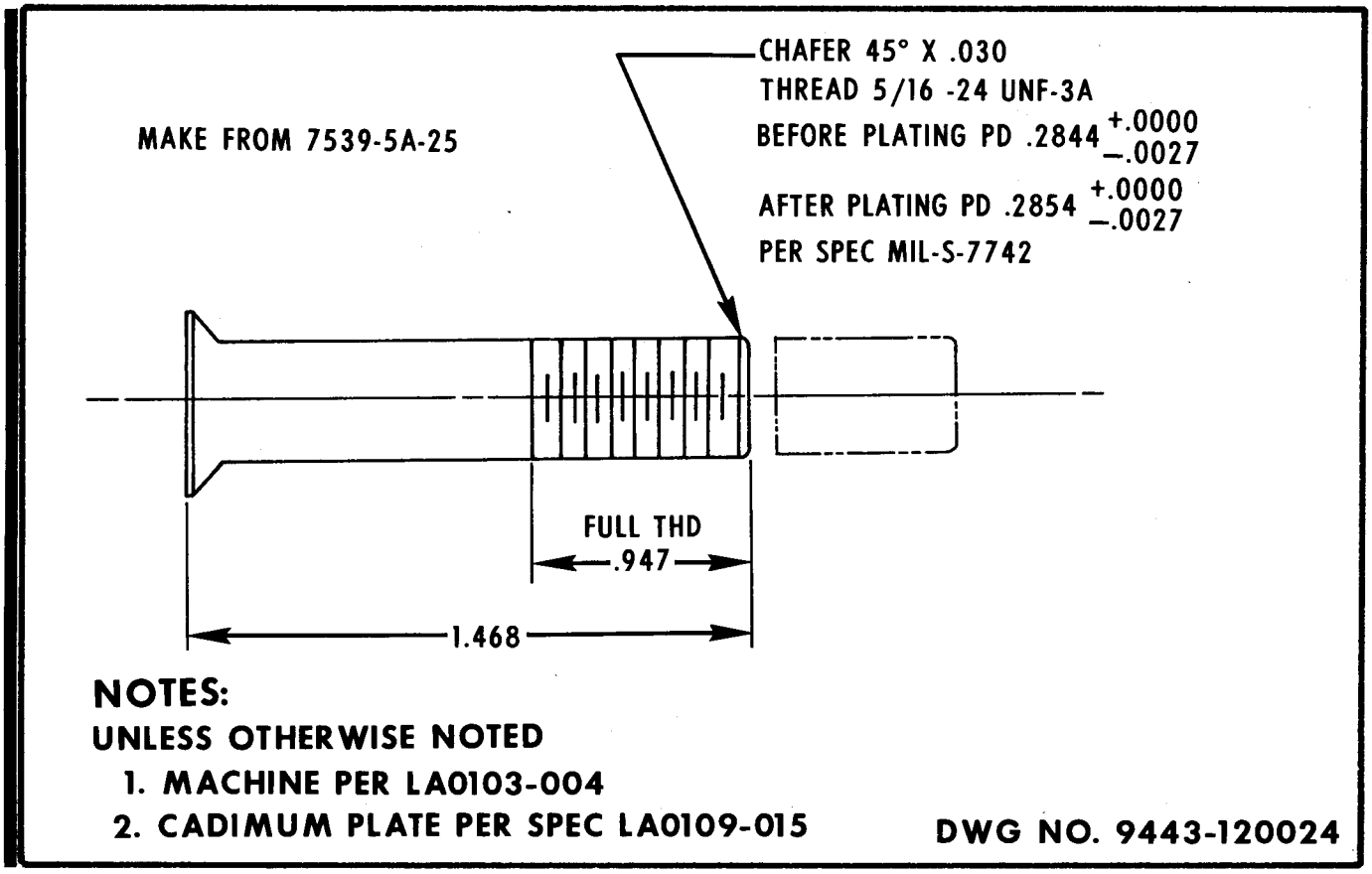


Figure 3A-15. Retainer Bolt - Fwd Wing Attaching

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