SEQUENTIAL COPY
NO. 149 OF 345

T.O. 1F-111(Y)A-1

UTILITY FLIGHT MANUAL

F-111A

USAF SERIES AIRCRAFT
SERIAL NO. AF 63-9766 THRU 63-9783
Contract AF 33(657)8260

THIS PUBLICATION IS INCOMPLETE WITHOUT CLASSIFIED SUPPLEMENTS T.O. 1F-111(Y)A-1A AND T.O. 1F-111(Y)A-1B.

THIS PUBLICATION REPLACES SAFETY SUPPLEMENTS -3 AND -4.
SEE NUMERICAL INDEX AND REQUIREMENT TABLE WEEKLY SUPPLEMENT T.O. 0-1-1-4A FOR CURRENT STATUS OF SAFETY SUPPLEMENTS.

Commanders are responsible for bringing this publication to the attention of all Air Force personnel cleared for operation of subject aircraft.

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Insert changed pages into basic publication. Destroy superseded pages.

THIS IS A SERVICE TEST MANUAL

A0000000-019C

3 DECEMBER 1965

Changed 23 December 1966

LIST OF EFFECTIVE PAGES

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CURRENT FLIGHT CREW CHECKLISTS

AIRCRAFT COMMANDER'S	PILOT'S
1F-111(Y)A-1CL-1	1F-111(Y)A-1CL-2
3 DEC 1965 CHANGED 23 DEC 1966	3 DEC 1965 CHANGED 23 DEC 1966

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*6-3 thru 6-6 Added . . .	23 Dec 66
*7-1 thru 7-6 . .	23 Dec 66
*7-7 Added . .	23 Dec 66
*7-8 Blank Added . . .	23 Dec 66
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SAFETY SUPPLEMENT SUMMARY

Safety Supplements will be identified with "SS" immediately following the -1 contained in the basic publication number and will be assigned consecutive dash numbers. Example: T.O. 1F-111(Y)A-1SS-1, -1SS-2, etc. The supplements you receive should follow in sequence and if you are missing one, check the weekly Safety Supplement Index T.O. 0-1-1-4A to see if the supplement was issued and, if so, is still in effect. That supplement may have been replaced or rescinded before you received your copy. If the supplement is still active, see your Publication Distribution Officer and get your copy. Existing Safety Supplements will not be renumbered and will remain active until rescinded or replaced. The following list contains: the previously cancelled or incorporated Safety Supplements; the outstanding Safety Supplements, if any; and the Safety Supplements incorporated in this issue. In addition, space is provided to list those Safety Supplements received since the latest issue.

NUMBER	DATE	SUBJECT OR DISPOSITION
1F-111(Y)A-1SS-1 & -2		Previously incorporated.
1F-111(Y)A-1SS-3		Nose wheel steering malfunction, Section III
1F-111(Y)A-1SS-4	16 Sept. 1966	Flight control system gain settings, Section II

OPERATIONAL SUPPLEMENT SUMMARY

Operational Supplements will be identified with an "S" immediately following the -1 contained in the basic publication number and will be assigned consecutive dash numbers. Example: T.O. 1F-111(Y)A-1S-1, -1S-2, -1S-3, etc. The supplements you receive should follow in sequence and if you are missing one, check the weekly Safety Supplement Index T.O. 0-1-1-2A to see if the supplement was issued and, if so, is still in effect. The supplement may have been replaced or rescinded before you received your copy. If the supplement is still active, see your Publications Distribution Officer and get your copy. Existing Operational Supplements will not be renumbered and will remain active until rescinded or replaced. The following list contains: the previously cancelled or incorporated Operational Supplements; the outstanding Operational Supplements, if any; and the Operational Supplements incorporated in this issue. In addition, space is provided to list those Operational Supplements received since the latest issue.

NUMBER	DATE	SHORT TITLE	DISPOSITION

AIRPLANE RETROFIT TECHNICAL ORDER INFORMATION.

Time Compliance Technical Order (TCTO) numbers, along with the signs "+" and "-", are used to distinguish between airplanes that have been modified and those that have not. This list includes the applicable TCTO numbers that have been issued up to the date of this publication. Those issued after that date will appear in the next change/revision. This is not a complete TCTO listing. Refer to the Basic Index (T.O. 0-1-1-1) for the complete listing of TCTO's which are applicable to this airplane. EXAMPLE: T.O. 1F-111A-575, Replace Windshield Wash/Rain Removal Selector Switch. Information applicable to airplanes modified by T.O. 1F-111A-575 will be coded + T.O. 1F-111A-575. Information applicable to airplanes not modified by T.O. 1F-111A-575 will be coded - T.O. 1F-111A-575.

T.O. NO.	SHORT TITLE	SYSTEM/EQUIPMENT AFFECTED
1F-111A-575	Replace Windshield Wash/Rain Removal Selector Switch	Anti-Icing and Defog Systems Section I

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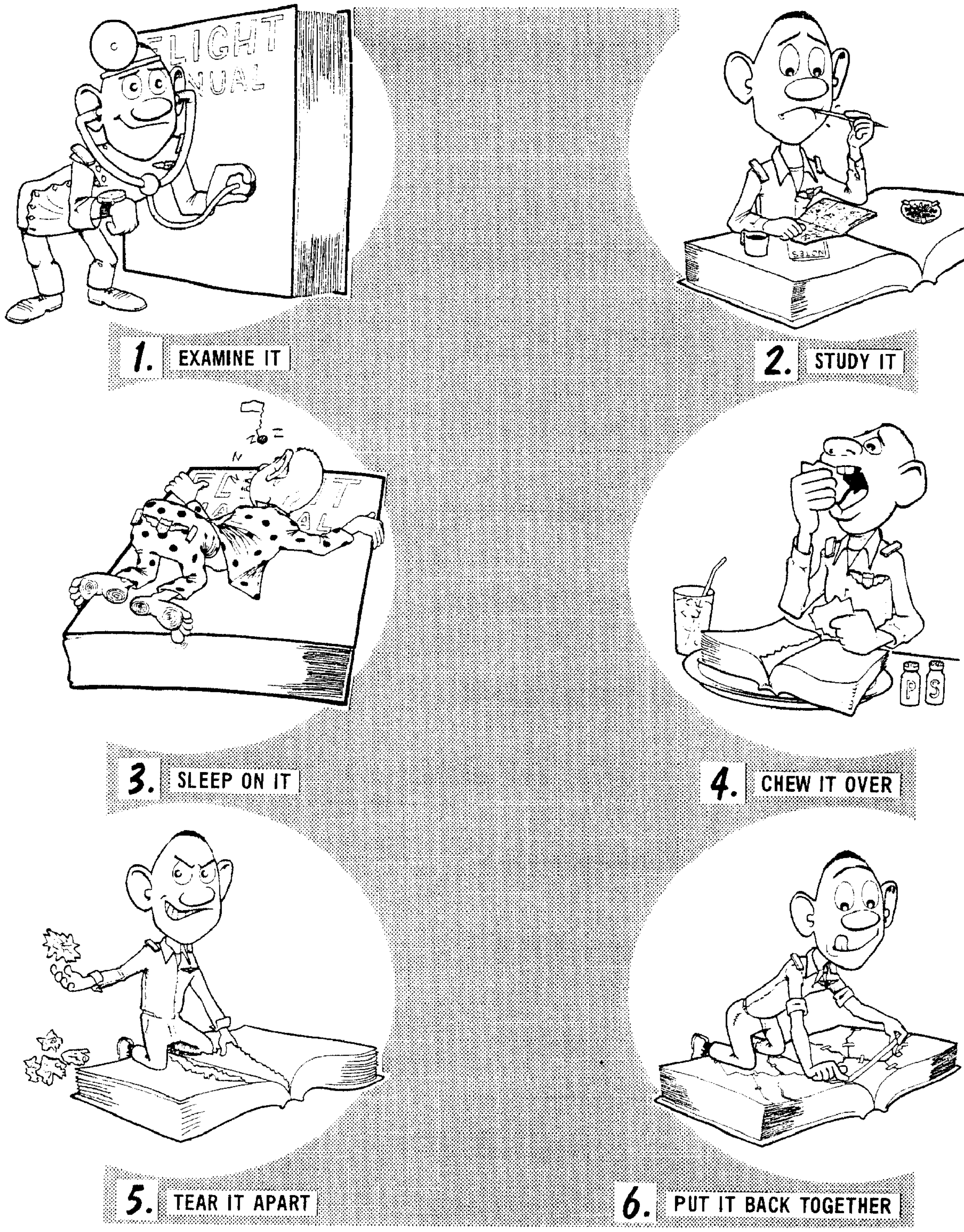
SOMETHING NEW...

This Flight Manual is arranged in a new format. It combines Sections I, IV and VII into a new Section I, "Description and Operation". It is arranged according to component function and its relationship to other components. Please,



7. *Then* — let us know (Form 847) how we can improve it (Comment on Section III also).

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SCOPE.

This manual contains the necessary information for safe and efficient operation of the F-111A. These instructions provide you with a general knowledge of the airplane, its characteristics, and specific normal and emergency operating procedures. Your experience is recognized, and, therefore, basic flight principles are avoided. Instructions in this manual are for a crew inexperienced in the operation of this airplane. This manual provides the best possible operating instructions under most circumstances. Multiple emergencies, adverse weather, terrain, etc. may require modification of the procedures.

PERMISSIBLE OPERATIONS.

The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations are prohibited unless specifically covered herein. Clearance must be obtained before any questionable operation, which is not specifically permitted in this manual, is attempted.

HOW TO BE ASSURED OF HAVING LATEST DATA.

Refer to T.O. 0-1-1-4A which is issued weekly and devoted solely to the listing of all current Flight Manuals, Safety Supplements, Operational Supplements, and Checklists. Its frequency of issue and brevity assures an accurate, up-to-date listing of these publications.

ARRANGEMENT.

The manual is divided into seven fairly independent sections to simplify reading it straight through or using it as a reference manual.

Note

Performance data normally included in Appendix I is contained in Classified Supplement 1F-111(Y)A-1A.

SAFETY SUPPLEMENTS.

Information involving safety will be promptly forwarded to you by Safety Supplement. Supplements covering loss of life will get to you within 48 hours by TWX, and those covering serious damage to equipment within 10 days by mail. The title page of the Flight Manual and the title block of each Safety Supplement should be checked to determine the effect they may have on existing supplements. You must remain constantly aware of the status of all supplements. Current supplements must be complied with, but there is no point in restricting your operation by complying with a replaced or rescinded supplement.

OPERATIONAL SUPPLEMENTS.

Information involving changes to operating procedures will be forwarded to you by Operational Supplements. The procedure for handling Operational Supplements is the same as for Safety Supplements.

CHECKLISTS.

The Flight Manual contains only amplified checklists. Checklists are issued as separate documents, see the back of the title page for the date of your latest checklist. Line items in the Flight Manual and checklists are identical with respect to arrangement and checklist number. Whenever a Safety Supplement affects the checklist, write in the applicable change on the affected checklist page. As soon as possible, a new checklist page, incorporating the supplement will be issued. This will keep handwritten entries of Safety Supplement information in your checklist to a minimum.

HOW TO GET PERSONAL COPIES.

Each flight crewmember is entitled to personal copies of the Flight Manual, Safety Supplements and Checklists. The required quantities should be ordered before you need them to insure their prompt receipt. Check with your supply personnel - it is their job to fulfill your Technical Order requests. Requests for copies will be made to SMAMA (SMNSTA).

FLIGHT MANUAL BINDERS.

Looseleaf binders and sectionalized tabs are available for use with your manual. These are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part 1). Check with your supply personnel for assistance in procuring these items.

WARNINGS, CAUTIONS, AND NOTES.

The following definitions apply to "Warnings", "Cautions", and "Notes" found throughout the manual.

WARNING

Operating procedures, techniques, etc., which will result in personal injury or loss of life if not carefully followed.

CAUTION

Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.

Note

An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY-TO LET US KNOW.

Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. However, we cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded through your Command Headquarters to ASD, Wright-Patterson AFB, Ohio, Attn; ASZO.

AIRPLANE DESIGNATION CODES.

Major differences between airplanes covered in this manual are designated by number symbols which appear in the text or on illustrations. Symbol designations for individual aircraft, and groups of aircraft are as follows:

(1)	63-9766	(10)	63-9775
(2)	63-9767	(11)	63-9776
(3)	63-9768	(12)	63-9777
(4)	63-9769	(13)	63-9778
(5)	63-9770	(14)	63-9779
(6)	63-9771	(15)	63-9780
(7)	63-9772	(16)	63-9781
(8)	63-9773	(17)	63-9782
(9)	63-9774	(18)	63-9783

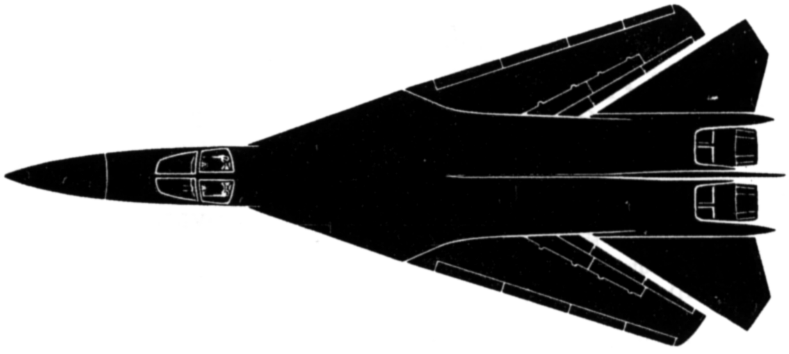
→ "through" or "and on".

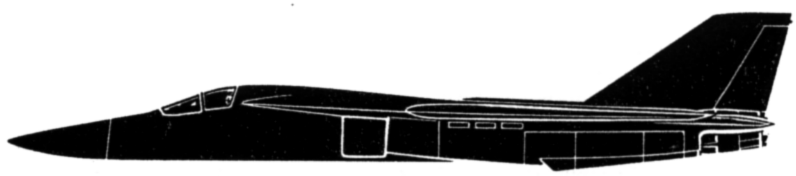
GLOSSARY

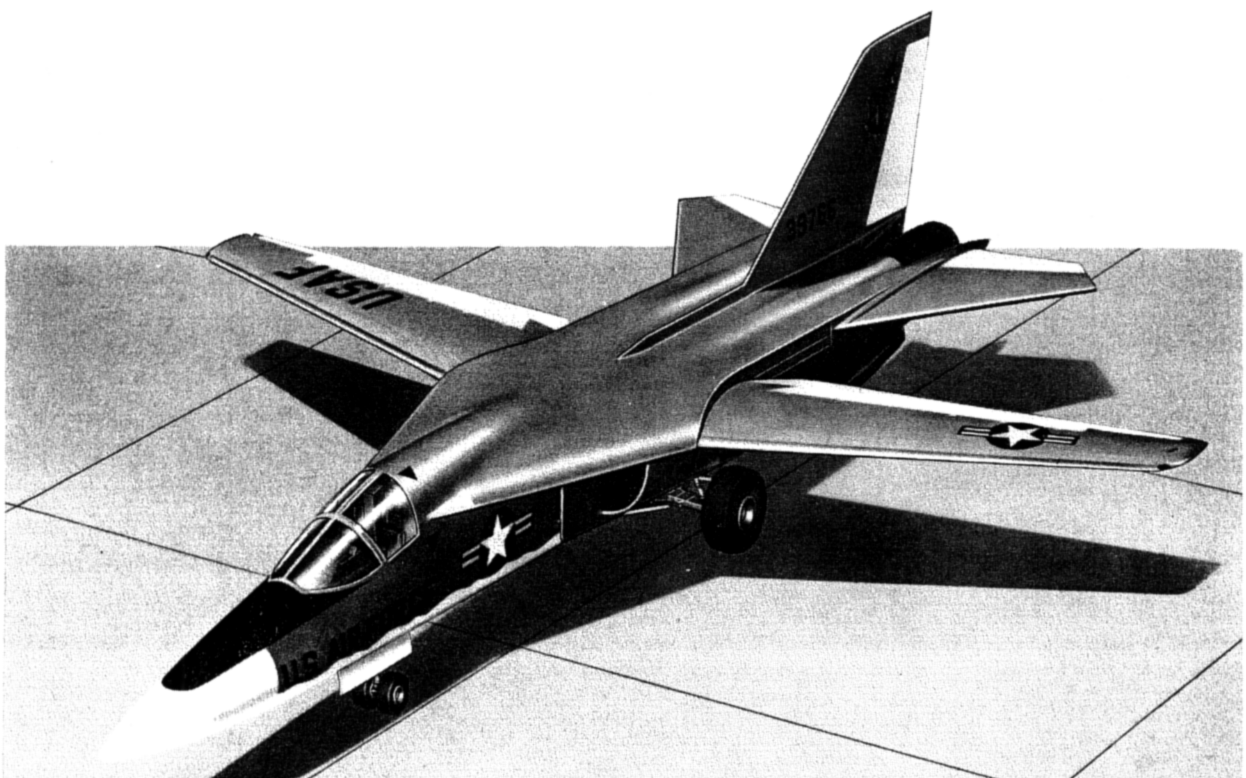
AA GUN	Air-to-Air Gunnery
AA RKT	Air-to-Air Rocketry
A/B	Afterburner
AC	Aircraft Commander
ac	Alternating Current
ADF	Automatic Direction Finder
ADI	Altitude Director Indicator
AFC	Automatic Frequency Control
AFRS	Auxiliary Flight Reference System
AG GUN	Air-to-Ground Gunnery
AILA	Airborne Instrument Low Approach
AIR FF	Air Freefall
AIR RET	Air Retard
ALT CAL	Altitude Calibration
ALT HLD	Altitude Hold
ALTM	Altimeter
ALT REF	Altitude Reference
AMI	Airspeed-Mach Indicator
ANT CAGE	Antenna Cage
ANT TILT	Antenna Tilt
ATF	Automatic Terrain Following
ATT GYRO	Attitude Gyro
AUX NAV	Auxiliary Navigation
AUX ATT	Auxiliary Attitude
AVVI	Altitude-Vertical Velocity Indicator
B/C	Biological/Chemical
BDHI	Bearing Distance Heading Indicator
BNDTI	Bomb Nav Distance Time Indicator
CADC	Central Air Data Computer
CCM	Counter-Counter Measures
CCW	Counterclockwise
CIR	Circular
CMDS	Countermeasures Dispenser Set
CMRS	Countermeasures Receiver Set
COMP	Compass
cps	Cycles Per Second
COM	Common
CKT	Circuit
CRS SEL NAV	Course Select Navigation
CW	Clockwise
DBT	Dual Bombing Timer
dc	Direct Current
DEST	Destination
DG	Directional Gyro
DISP	Dispenser
DIV BOMB	Dive Bombing

EBL	Emergency Boom Latching
EPR	Engine Pressure Ratio
FDC	Flight Director Computer
FOD	Foreign Object Damage
FTC	Fast Time Constant
GND/GRD	Ground
GND MAN	Ground Manual
GND VEL	Ground Velocity
GRD FF	Ground Freefall
GRD RET	Ground Retard
HF	High Frequency
HOM	Homing
HSI	Horizontal Situation Indicator
IF	Intermediate Frequency
IFF	Identification Friend or Foe
IFIS	Integrated Flight Instrument System
ILS	Instrument Landing System
INPH	Interphone
I/P	Identification of Position
ISC	Instrument System Coupler
IRU	Inertial Reference Unit
JETT	Jettison
LEV BOMB	Level Bombing
LCOSS	Lead Computing Optical Sight System
LOF BOMB	Loft Bombing
LNCH	Launch
LSB	Lower Side Band
MACH HLD	Mach Hold
MAG VAR	Magnetic Variation
MAN CRS	Manual Course
MAN FIX	Manual Fix
MAN HDG	Manual Heading
MED	Medium
MFC	Manual Frequency Control
MIC	Microphone
MI/DIA	Miles/Diameter
MON	Monitor
MRT	Modulator-Receiver Transmitter
MSMA	Maximum Safe Mach Assembly
NC	Navigation Computer
NORM	Normal
NWS/AR	Nosewheel Steering/Air Refueling
OMS	Off, Monitor, Safe
OVRD	Override
P	Pilot
PP/PRES POS	Present Position
pph	Pounds Per Hour
PPI	Plan Position Indicator
psi	Pounds Per Square Inch
RAT	Ram Air Turbine
REC	Receive
REF ENGAGE	Reference Engage
RHAWS	Radar Homing and Warning System
RKT	Rocket
RPM	Revolutions Per Minute
RT	Receiver-Transmitter
SIF	Selective Identification Frequency
SIT	Situation Display
SLC	Side Lobe Cancellation
SP	Stabilization Platform
STAB AUG	Stability Augmentation
STBY	Standby
STC	Sensitivity Time Control
TAS	True Airspeed
TF	Terrain Following
TFR	Terrain Following Radar
TIT	Turbine Inlet Temperature
T.O. & LAND	Takeoff and Land
T/R	Transmit/Receive
TRANS	Transmit
TTI	Total Temperature Indicator
UHF	Ultra High Frequency

THE F-111A AIRPLANE







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SECTION I

DESCRIPTION & OPERATION

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THE AIRPLANE.

The F-111A is a two place (side-by-side) long range fighter bomber built by General Dynamics, Fort Worth Division. The airplane is designed for all-weather supersonic operation at both low and high altitude. Mission capabilities include: long range high altitude intercepts utilizing air-to-air missiles and/or guns; long range attack missions utilizing conventional or nuclear weapons as primary armament and close support missions utilizing a choice of missiles, guns, bombs and rockets. An automatic low altitude terrain following system enhances penetration capability. Power is provided by two TF-30 axial-flow, dual-compressor turbo-fan engines equipped with afterburners. The wings, equipped with leading edge slats and trailing edge flaps, may be varied in sweep, area, camber, and aspect ratio, by the selection of any wing sweep angle between 16 and 72.5 degrees. A selective forward wing sweep provides takeoff and landing capabilities at minimum speeds. For all other regimes the wings are manually swept in accordance with desired mach number. This feature provides the airplanes with a highly versatile operating envelope. The empennage consists of a fixed vertical stabilizer with rudder for directional control, and a horizontal stabilizer that is moved symmetrically for pitch control and asymmetrically for roll control. Stability augmentation incorporates triple redundant features which enhance system reliability. The tricycle-type forward retracting landing gear is hydraulically operated. The main landing gear consists of a single common trunnion upon which two wheels are singly mounted, and contains but one extending/retracting/locking system which ensures symmetrical main gear operation. Also ground loads imposed upon the gear tend to extend the drag strut to the locked position. Stores are carried in a fuselage-enclosed weapons bay and externally on both pivoting and fixed wing-mounted pylons. The fuel system incorporates both inflight and single point ground refueling capabilities.

AIRPLANE DIMENSIONS.

Length (overall including pitot static boom) - 73 feet, 10.6 inches
Wing span (wings swept) - 31 feet, 11.4 inches
Wing span (wings extended) - 63 feet, 0.0 inches
Height (to top of vertical tail) - 17 feet, 1.4 inches

Refer to Section II for turning radius and ground clearance dimensions.

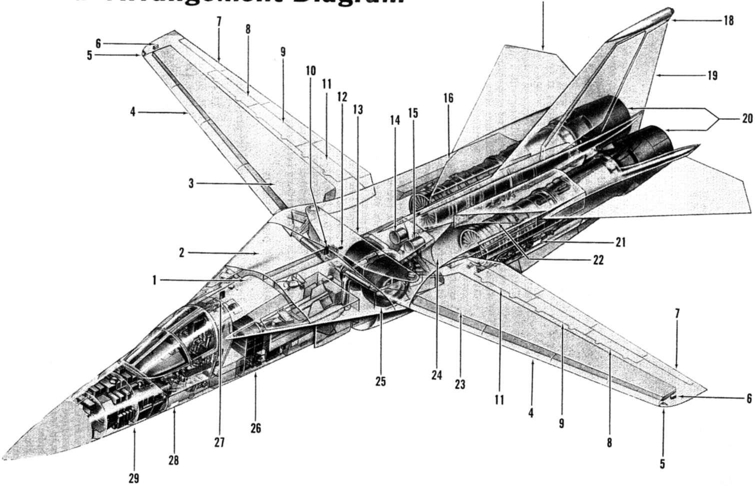
AIRPLANE WEIGHT.

Refer to T.O. 1F-111(Y)A-1A for basic gross weight of the airplane. Refer to Section V, Weight Limitations, for maximum allowable gross weights.

FLIGHT CREW.

The flight crew consists of two pilots seated side-by-side. The crew member assigned to the left crew station normally serves as aircraft commander. The pilot at the right crew station, in addition to his normal pilot duties, operates the offensive and defensive equipment associated with the controls at that station.

General Arrangement Diagram



1. FORWARD FUEL TANK (BAY F-1)
2. FORWARD FUEL TANK (BAY F-2)
3. RIGHT WING FUEL TANK
4. LEADING EDGE SLATS
5. WING POSITION LIGHT
6. WING FORMATION LIGHT (UPR & LWR)
7. WING FLAPS
8. OUTBOARD SPOILER
9. INBOARD SPOILER
10. TACAN/UHF ANTENNA
11. GROUND ROLL SPOILER
12. ANTI-COLLISION LIGHT (UPR & LWR)
13. TRAP TANK
14. PRIMARY HYDRAULIC SYSTEM RESERVOIR
15. UTILITY HYDRAULIC SYSTEM RESERVOIR
16. RIGHT SADDLE FUEL TANK
17. HORIZONTAL STABILIZER
18. TAIL POSITION LIGHT
19. RUDDER
20. ENGINES
21. AFT FORMATION LIGHT (2)
22. LEFT SADDLE FUEL TANK
23. LEFT WING FUEL TANK
24. AFT FUEL TANK
25. MAIN LANDING GEAR
26. AFT EQUIPMENT BAY ELECTRONICS
27. IFF ANTENNA
28. FORWARD FORMATION LIGHT
29. FORWARD EQUIPMENT BAY ELECTRONICS

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Figure 1-1.

Crew Station General Arrangement (Typical)

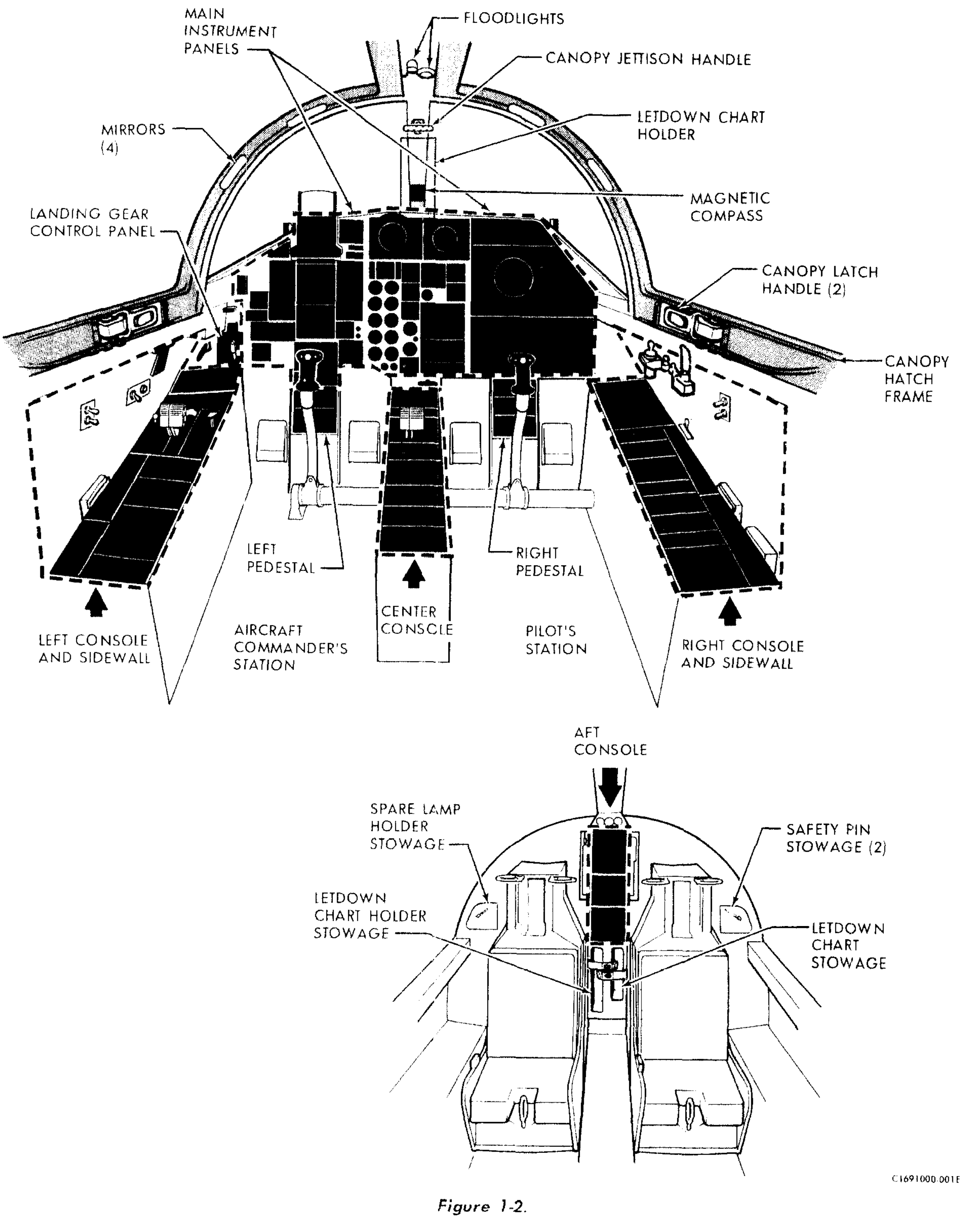


Figure 1-2.

"Fuel Quantity Data," and "Gross Weight Limitations", refer to Confidential Supplement T.O. 1F-111(Y)A-A. For specific aircraft weight, refer to the associated handbook of Weight and Balance Data, T.O. 1-1B-40.

FLIGHT CREW.

The flight crew consists of two pilots seated side-by-side. The crew member assigned to the left crew station normally serves as aircraft commander. The pilot at the right crew station, in addition to his normal pilot duties, operates the offensive and defensive equipment associated with the controls at that station.

ENGINES.

The airplane is powered by two Pratt and Whitney YTF30-P-1 (or TF30-P-1) sixteen-stage axial flow turbofan engines equipped with afterburners. See figure 1-3. The engines are mounted side by side in the fuselage and are interchangeable. The sea level, standard day thrust rating of the engine is in the 10,000 pound class in military power and in the 18,000 pound class in afterburner. Either engine may be started without the aid of ground support equipment by means of a pyrotechnic cartridge. Provisions are also made for starting the engines with an external pneumatic ground starter cart. With either engine operating, the other engine can be started by using bleed air from the operating engine. Electrical power is supplied for the engine igniter plugs by an engine-driven alternator. Each engine is supplied a flow of air through a separate inlet duct located below the intersection of the wing and fuselage. An automatically controlled, movable spike is used in each inlet duct to control airflow to the engines. Additional engine inlet air is provided during ground operation and at low airspeeds through openings in the outboard side of each nacelle, when the translating cowls are in the extended position. Splitter vanes are used at the front of the inlet ducts to remove the low energy air from the fuselage and the lower surface of the wing glove, thus preventing boundary layer air from disturbing engine inlet air. These features allow optimum engine performance throughout a wide range of airplane operating conditions. Air from the inlet of each engine is routed through a single duct for both the basic engine section and the fan section. Three compressor stages provide the initial pressurization of the air flowing into the engine and into the fan duct. The fan duct is a full-annular duct which directs flow aft to join the engine airflow coming from the turbine discharge. The fan air develops a significant portion of total engine thrust. Engine air is compressed by 9 stages of the low pressure compressor (N1) of which three stages are the fan, and 7 stages of the high pressure compressor (N2). The air is then diffused into the combustion section which contains 8 combustion chambers. The turbine section of the engine consists of a single-stage turbine to drive the high pressure compressor and a three-stage turbine to drive the low pressure compressor. The turbines are mechanically independent of each other. High pressure compressor speed is indicated by a tachometer. Speed of the low pressure compressor is not monitored except by an overspeed caution lamp. After leaving the turbine section of the engine, the air is joined with the fan air in the afterburner section. Bleed air from the engine compressors is used for cockpit and equipment bay air conditioning and pressurization, hydraulic system pressurization, fuel tank pressurization, hydraulic oil cooling, engine vortex destroyers, generator/CSD cooling, ground oil cooling, and windshield rain removal. Also, hot bleed air is used for engine inlet and guide vane anti-icing (Refer to "Anti-Icing and Defog Systems", this section). Procedures for starting and operation of the engines are contained in Section II. Refer to Section V for Engine Operating Limitations.

ENGINE FUEL CONTROL SYSTEM.

Each engine fuel control system (figure 1-3) automatically provides optimum fuel flow for any throttle setting. This system responds to several engine operating parameters and makes it unncessary to adjust the throttle in order to compensate for variations in inlet air temperature, altitude or airspeed. The engine fuel system consists of a two-stage engine-driven fuel pump, fuel control unit, flowmeter, filter, a pressurizing and dump valve, nozzles, and a fuel-oil heat exchanger. Fuel from the tanks is routed through the flowmeter to the centrifugal stage of the engine fuel pump, through a filter, and back to the gear stage of the pump. Bypass valves route fuel past the filter or first pump stage in event of failure of these components. The second pump stage delivers fully pressurized fuel to the fuel control unit which provides metered fuel flow through the fuel-oil heat exchanger to the fuel pressurizing and dump valve. This dual function valve directs the fuel through the primary and secondary fuel manifolds to eight fuel nozzles which spray the fuel into the eight engine combustion chambers. When the fuel pressure drops during engine shutdown, the fuel pressurizing and dump valve automatically opens and drains the primary fuel manifold.

Fuel Control Unit.

The engine fuel control unit is a hydromechanical device incorporating an engine-driven, flyball-type speed governor. The control unit consists of a fuel metering system and a computing system which operates as a function of throttle setting, main combustion chamber pressure, high pressure rotor N2 speed, compressor inlet pressure, compressor inlet temperature, and flight mach number. The metering system selects the rate of fuel flow to be supplied to the engine in response to the throttle setting. However, metering sections are regulated by the fuel control computing system which monitors the various engine operating parameters. Fuel enters the fuel control through a filter that is provided with a spring-loaded bypass. Fuel metering is accomplished by maintaining a constant pressure across a variable valve area which is controlled by the computing system. The constant pressure is maintained by means of a pressure regulating bypass valve. This valve consists of a servo-operated valve and a spring-loaded valve. Normally, the servo maintains constant valve regulation; but in the event of servo malfunction, the spring valve alone will provide adequate regulation. Deviations from the desired metering pressure are sensed in the valve regulating unit which varies the bypass flow area, thereby restoring the desired pressure by returning excess fuel to the pump inlet.

ENGINE AFTERBURNERS.

The afterburner (A/B) augments engine thrust by injecting fuel into the engine exhaust stream in the afterburner section where it is ignited by a hot streak ignition system. Operation is controlled by the throttle. When the throttle is moved forward within the afterburner range, the afterburner fuel control pressurizes the afterburner first fuel manifold, (zone 1) schedules light-off flow, and activates the variable nozzle system. This system senses a pressure change and controls the exit area of the afterburner exhaust nozzle. Six free-floating, blow-in doors are located near the aft end of the afterburner. These doors open any time outside air pressure is greater than pressure inside the duct, allowing outside air to enter, thus increasing the total engine thrust. The trailing edge of the afterburner consists of free-floating leaves which reduce drag at the aft end of the engine by directing the exhaust gases into the slipstream with minimum turbulence.

Afterburner Fuel System.

The afterburner fuel system (figure 1-3) consists of the following major components: an exhaust nozzle pump, an afterburner fuel pump, an afterburner fuel control unit with integral exhaust nozzle control, and fuel spray rings. Fuel from the tanks flows through the flowmeter to the afterburner fuel pump. The exhaust nozzle pump is supplied fuel from the boost stage of the engine main fuel pump. The exhaust nozzle pump supplies fuel to the afterburner fuel control until a predetermined fuel flow rate is exceeded. At this flow rate, the afterburner fuel pump inlet is opened and begins to supply fuel to the afterburner fuel control unit. Fuel from the afterburner pump passes through a fuel-oil cooler before entering the afterburner fuel control unit. This unit includes a computer and a high pressure flow section. Fuel is then directed to the spray rings where it is atomized and ignited in the afterburner combustion chamber. Five zones of afterburning can be selected through the afterburner fuel control unit which schedules fuel to the spray rings in the various zones of the afterburner as a function of throttle setting. When the throttle is advanced for afterburner initiation and when high pressure compressor speed exceeds approximately 80 percent rpm, the afterburner initiation valve schedules light-off fuel flow until afterburner light-off occurs, as sensed by the exhaust nozzle control.

Afterburner Ignition.

The function of the afterburner ignition system is to provide a means of igniting fuel in the afterburner combustion chamber to initiate afterburner operation. When the system is actuated, fuel from the afterburner fuel system is injected into the aft end of No. 4 engine combustion chamber, thereby creating a local excessively rich fuel-air mixture. This mixture results in a longer flame which burns past the turbines to provide hot streak ignition for a second injection of fuel into the engine in the vicinity of the turbine exhaust section. This second hot streak continues aft and ignites the fuel that is injected into the afterburner combustion chamber.

ENGINE INLET SPIKES.

Engine inlet air velocity is regulated throughout the entire aircraft speed range in order to maintain maximum engine performance. This regulation of the air inlet velocity is accomplished by a movable spike located in the inlet of each engine. Each spike is a quarter circle, conical-shaped, variable diameter body that is independently movable forward and aft. The spikes are located in each air intake at the intersection of the wing lower surface and the fuselage boundary plate. Position and shape of the spikes are changed automatically to vary the inlet geometry and to control the inlet shock wave system. Local air pressure changes due to variations in inlet local mach, cowl lip shock wave position, and diffuser exit mach number are measured by sensors in the spike control unit. Signals from the control unit operate hydraulic actuators which are powered by the utility hydraulic system to position the spike fore and aft (extend or retract) and adjust the spike cone angle by contracting and expanding the spike as required. In the event the system malfunctions, a pneumatic override is provided to position and lock the spike full forward and fully contracted. An electronic anti-icing system prevents ice formation on the sensors. Refer to "Anti-Icing and Defog System" this section.

ENGINE TRANSLATING COWLS.

During ground operation and low speed flight, an additional amount of air is required for optimum engine performance. This additional air is provided by extension of movable cowls which form the leading edge of each inlet duct. When a cowl is in the extended position, an opening is exposed aft of the cowl in the engine inlet duct to provide a path for additional outside air to the engine. The translating cowl system consists of a movable cowl for each engine, an electrical actuator for each cowl, two flip-flop type cowl position indicators, a cowl caution lamp and switches for manual or automatic operation. On airplanes 1 → 11 the position of the cowls is controlled manually. On airplanes 12 → the cowls will drive to the extended position any time the landing gear handle is placed in the down position. With the landing gear handle in the up position the cowls can be manually retracted or extended or controlled automatically as a function of airplane speed. In the automatic mode the cowls will retract above mach 0.5 and extend below mach 0.44.

FOREIGN OBJECT DAMAGE (FOD) PREVENTION DOORS AND VORTEX DESTROYERS.

On airplanes 12 → , the ingestion of foreign objects into the engine is prevented by FOD prevention doors which are deployed ahead of each engine inlet, and by an aerodynamic screen of engine bleed air, which is directed down and outboard beneath each inlet through vortex destroyer air jets. The FOD prevention doors consist of two parts for each inlet. An upper door extends down from beneath the glove and a side door extends outward from the splitter vane to form a shield ahead of the inlet. The vortex destroyers are on all airplanes and serve to prevent the formation of vortexes below the inlet, thereby preventing foreign objects from being entrained in a vortex and sucked into the engine. A ground safety switch, located on the landing gear activates the vortex prevention air screen. The doors are operated by an independent pneumatic system and are electrically controlled by an FOD prevention door switch in the cockpit. The supply of pneumatic pressure provides for two cycles of operation (open and close twice) of the doors.

ENGINE VARIABLE EXHAUST NOZZLES.

The variable nozzle system incrementally opens and closes the engine exhaust nozzle for afterburner modulation. The control is a hydromechanical computing device that determines and sets the nozzle area required to maintain a desired turbine pressure ratio during afterburner operation. The nozzle position is scheduled by the throttle setting and governed by turbine pressure ratio. The nozzle is closed for all ranges of nonafterburner operation except for ground engine idle at which time it is positioned fully open for minimum thrust. The nozzle closes when either throttle is advanced 3 degrees above IDLE. If afterburner blowout occurs, the blowout signal

Engine Fuel Control System

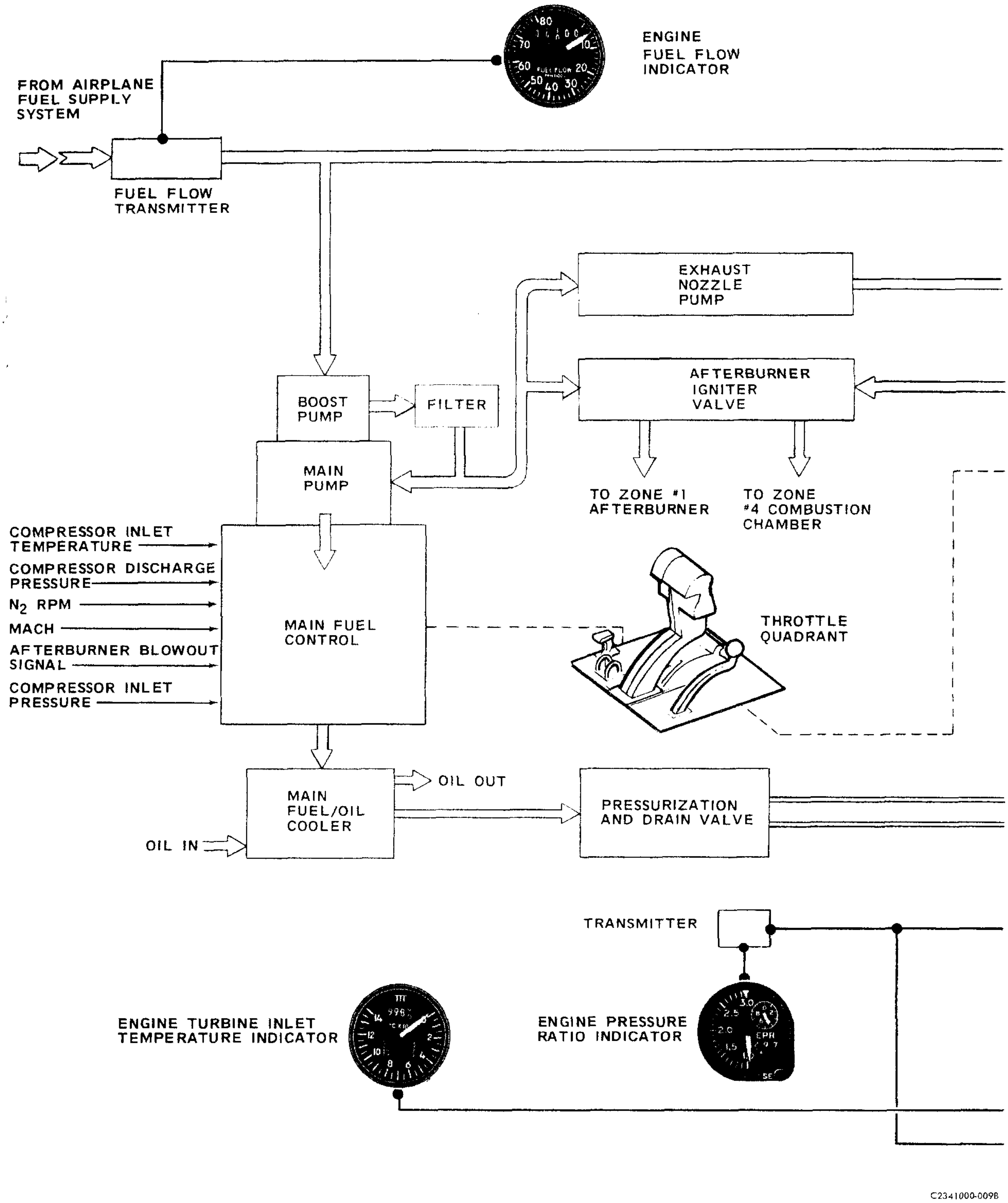


Figure 1-3. (Sheet 1 of 2)

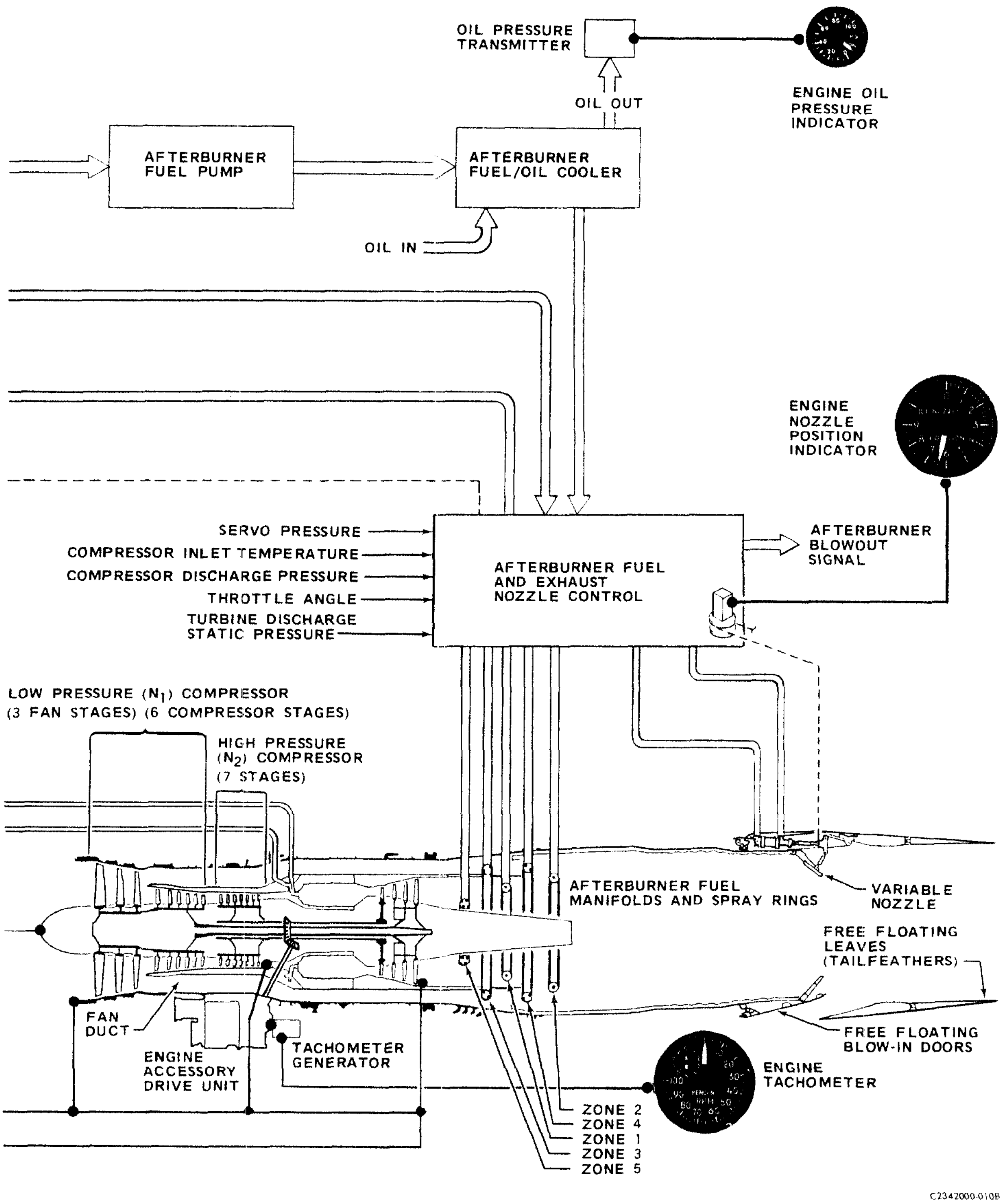


Figure 1-3. (Sheet 2 of 2)

valve is actuated, and the nozzle closes. In addition, the afterburner fuel selector valve closes off fuel flow to all afterburner zones, and a signal is directed to the engine main fuel control to reduce fuel flow to the main combustion chamber. When the nozzle has moved to the closed position, the blowout signal is removed. Afterburner operation can again be initiated; however, the throttles must first be moved to a nonafterburning position.

ENGINE IGNITION SYSTEM.

The functions of the engine ignition system are to provide a means of initiating combustion in the combustion chambers during the starting cycle and to provide a means for furnishing an engine ignition source in the event of a flameout. Each engine has a dual main ignition system including two ignition exciters, two igniter plugs, an ignition alternator, and an automatic restart switch. The alternator is engine driven and is capable of providing sufficient energy to both exciters of the ignition system for ground starting or for windmill starts during all flight conditions. Ignition alternator voltage is stepped up by transformer and capacitor circuits within the exciters to provide ionizing voltage for the igniter plugs. The alternator incorporates two independent current generating circuits for increased reliability. The automatic restart circuit energizes the ignition system in the event of a combustion chamber flameout by sensing the rate of change of burner pressure. Engine ignition is accomplished by the two spark igniters located in the lower combustion chambers (No. 4 and No. 5) of each engine. Advancing the throttle more than 3 degrees from OFF position actuates the throttle ignition switch for that engine. This action provides ignition when the engine start switch is in PNEU or CARTRIDGE. Electrical ignition is cut off when the ground start switch returns to OFF. This normally occurs when the starter centrifugal cutout switch, of the last engine to be started, opens at approximately 40 percent engine rpm. Ignition is also cut off when the throttle is retarded to less than approximately 3 degrees from OFF position.

ENGINE STARTER SYSTEM.

The engines may be started either by a pyrotechnic starter cartridge or by an external pneumatic pressure source. Also an engine may be started by pneumatic crossbleed provided one engine is running. Electrical power required for starting may be obtained from an external ground source or from the airplane battery. Each engine is provided with a combination cartridge-pneumatic starter that may be activated by a solid propellent cartridge, by air obtained from an auxiliary air cart, or by routing bleed air from the other engine (if operating). Starter operation is basically the same for cartridge or pneumatic operation. The cartridge-pneumatic starter is composed of: turbine, gear train, overrunning clutch with a speed sensing device, an overspeed disengagement mechanism with shear pin, a breech chamber with cap and locking handle, a two-stage pressure relief valve, an aerodynamic brake, and cartridge ignition electrical components. For a cartridge start, placing the ground start switch to CARTRIDGE and lifting the throttles out of OFF position completes an electrical circuit to ignite the cartridge. The two-stage pressure relief valve maintains proper operating pressure and, in case of malfunction, relieves pressure to safe limits. As engine rpm exceeds starter rpm, an overrunning clutch releases to prevent the starter turbine from being driven to an overspeed condition. In case the overrunning clutch fails to release, the reverse torque disengagement mechanism will disengage the starter gear train and turbine from the engine. An aerodynamic brake is attached to the starter turbine to prevent turbine overspeed. When compressed air is used to start the engine, placing the ground start switch to PNEU and lifting the throttle out of the OFF position opens the starter pressure shutoff valve and allows compressed air to operate the starter. When approximately 40 percent rpm is reached, a centrifugal switch breaks the starter control circuit, which allows the control valve to close, shutting off starter air. Provision has been made to carry two spare cartridges in the main landing gear wheel well for starting the engines when auxiliary power is not available.

ENGINE CONTROLS AND INDICATORS.

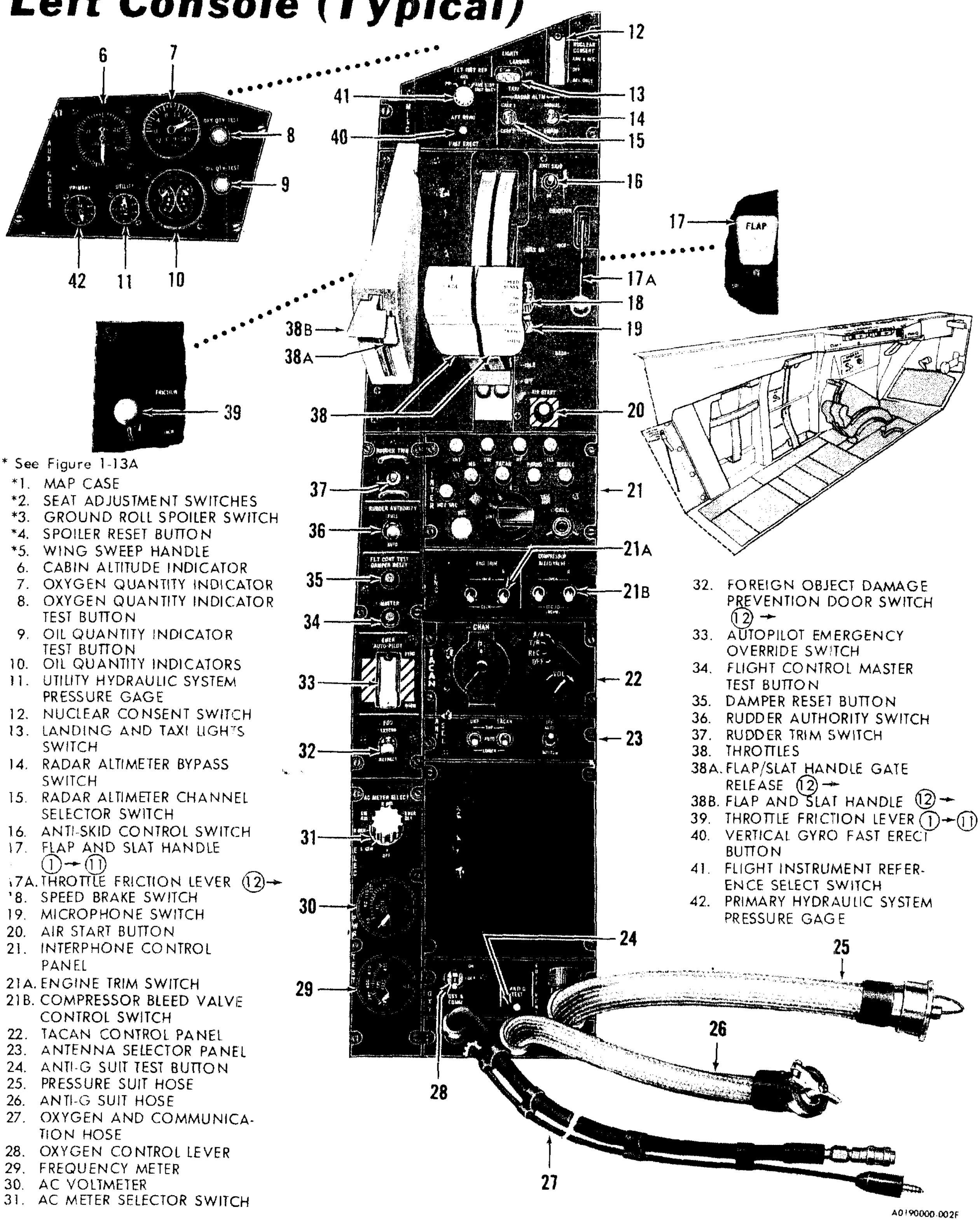
Throttles.

A set of throttles (38, figure 1-4 and 5, figure 1-17), is provided for both the aircraft commander and the pilot. The respective left and right throttles in each set are mechanically linked together. Each throttle provides thrust setting adjustment for its respective engine. Throttle friction for both sets of throttles is controlled by means of the friction lever located adjacent to the left set. Moving the lever toward INCR increases throttle friction, and moving the lever toward DECR decreases the friction. Force required to move the throttles varies from approximately 5.0 pounds when the friction lever is in the full DECR position to 15.0 pounds when the friction lever is in the full INCR position. Both sets of throttles have positions marked OFF, IDLE, MIL, and MAX AB, respectively. Only the aircraft commander's throttles can be raised to go into or from the OFF position. The pilot's throttles cannot be used for engine starting or shutdown. When the aircraft commander's throttles are lifted to move them out of the OFF position, the throttle starter switches are actuated. If the ground start switch is in CARTRIDGE or PNEU, the starter cartridges are automatically fired or the starter pneumatic pressure shutoff valves opened if an external starting air source is used. Movement of either set of throttles past approximately three degrees forward of OFF activates the engine ignition system. The left throttle of the aircraft commander's set includes a cage-gyro switch. The right throttle of each set includes a microphone switch and a speed brake switch.

Engine Ground Start Switch.

The engine ground start switch (4, figure 1-17), located on the center console, has three positions

Left Console (Typical)



* See Figure 1-13A

*1. MAP CASE
*2. SEAT ADJUSTMENT SWITCHES
*3. GROUND ROLL SPOILER SWITCH
*4. SPOILER RESET BUTTON
*5. WING SWEEP HANDLE
6. CABIN ALTITUDE INDICATOR
7. OXYGEN QUANTITY INDICATOR
8. OXYGEN QUANTITY INDICATOR TEST BUTTON
9. OIL QUANTITY INDICATOR TEST BUTTON
10. OIL QUANTITY INDICATORS
11. UTILITY HYDRAULIC SYSTEM PRESSURE GAGE
12. NUCLEAR CONSENT SWITCH
13. LANDING AND TAXI LIGHTS SWITCH
14. RADAR ALTIMETER BYPASS SWITCH
15. RADAR ALTIMETER CHANNEL SELECTOR SWITCH
16. ANTI-SKID CONTROL SWITCH
17. FLAP AND SLAT HANDLE (1)→(11)
17A. THROTTLE FRICTION LEVER (12)→
18. SPEED BRAKE SWITCH
19. MICROPHONE SWITCH
20. AIR START BUTTON
21. INTERPHONE CONTROL PANEL
21A. ENGINE TRIM SWITCH
21B. COMPRESSOR BLEED VALVE CONTROL SWITCH
22. TACAN CONTROL PANEL
23. ANTENNA SELECTOR PANEL
24. ANTI-G SUIT TEST BUTTON
25. PRESSURE SUIT HOSE
26. ANTI-G SUIT HOSE
27. OXYGEN AND COMMUNICATION HOSE
28. OXYGEN CONTROL LEVER
29. FREQUENCY METER
30. AC VOLTMETER
31. AC METER SELECTOR SWITCH
32. FOREIGN OBJECT DAMAGE PREVENTION DOOR SWITCH (12)→
33. AUTOPILOT EMERGENCY OVERRIDE SWITCH
34. FLIGHT CONTROL MASTER TEST BUTTON
35. DAMPER RESET BUTTON
36. RUDDER AUTHORITY SWITCH
37. RUDDER TRIM SWITCH
38. THROTTLES
38A. FLAP/SLAT HANDLE GATE RELEASE (12)→
38B. FLAP AND SLAT HANDLE (12)→
39. THROTTLE FRICTION LEVER (1)→(11)
40. VERTICAL GYRO FAST ERECT BUTTON
41. FLIGHT INSTRUMENT REFERENCE SELECT SWITCH
42. PRIMARY HYDRAULIC SYSTEM PRESSURE GAGE

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Figure 1-4.

marked PNEU, OFF, and CARTRIDGE. The switch is spring loaded to OFF; however, when the switch holding coil is energized, the ground start switch remains in the position selected until the coil is de-energized. The holding relay of the switch is de-energized when, during starting, the centrifugal cutout switch in the starter of the last engine to be started opens the relay control circuit. This occurs when high pressure compressor speed reaches 36 to 39 percent rpm. If the ground start switch locks in the OFF position, the toggle must be lifted to reposition it. When the switch is placed in PNEU or CARTRIDGE, 28 volt dc electrical power is supplied to arm the throttle starter switches. Power is also directed to energize the engine start relays and the ground start switch holding relay. Two throttle starter switches, one for each engine throttle, are actuated by throttle movement. When a throttle is lifted or moved out of OFF, the starter switch directs electrical power from the ground start switch to fire the respective engine start cartridge if the ground start switch is in CARTRIDGE or through the engine start relay to open the starter air shutoff valve if the ground start switch is in PNEU. When a throttle is in OFF and down, the throttle starter switch electrically grounds the starter cartridge circuit, thereby preventing cartridge activation through inadvertent movement of the ground start switch. When the ground start switch is in the OFF position, electrical power is isolated from the engine starter system. Engine start counters, located in the left forward equipment bay, record separately the number of cartridge and pneumatic starts for each engine.

Air Start Buttons.

Two airstart pushbuttons (20, figure 1-4 and 8, figure 1-17), one located on the left console and the other located on the center console respectively, provide a means of obtaining ignition for air starting the engines. The buttons are marked AIR START. When either airstart button is depressed, the airstart timer relay actuates and allows ignition generator power to operate the ignition exciters for both engines. The relay will remain energized for approximately 55 seconds after the airstart button is released, thereby providing ignition for this length of time.

Ignition Cutoff Switch.

The ignition cutoff switch (8, figure 1-16), located on the aft console, is labeled GRD IGNITION and has two positions marked NORM and OFF. When the switch is in OFF, a relay, which deactivates the engine electrical ignition system for both engines by grounding the ignition alternator output, is energized. When the switch is in the NORM position, the relay is deactivated and the ignition circuits are not grounded through this relay.

Translating Cowl Switches. 1 → 11 SS-8

~~Two translating cowl switches (8A, figure 1-17), located on the center console are provided for opening or closing the translating cowls. The switches are~~ labeled L COWL and R COWL and have three positions marked EXTEND, OFF and RETRACT. The switches are the lock lever type which must be lifted out of the lock before they can be moved. Placing either switch to EXTEND or RETRACT will cause the respective cowl to drive to the position called for by the switch. With the switches in the OFF position power is removed from the system.

Translating Cowl Switches. 12 →

Two translating cowl switches (8A, figure 1-17), located on the center console are provided for manual or automatic control of the translating cowls. The switches are labeled L COWL and R COWL and have three positions marked EXTEND, AUTO and RETRACT. The switches are the lock lever type which must be lifted out of the lock before they can be moved. The switches are activated when the landing gear is in the UP position. When the switches are placed to AUTO and the landing gear is in the up position the position of the cowls is controlled automatically as a function of airspeed. When the airspeed is below mach 0.44 the cowls are extended. When the airspeed is above mach 0.5 the cowls are retracted. Automatic operation of the cowls is disabled at speeds above mach 0.8 to prevent inadvertent opening at high speeds. Placing the switches to EXTEND or RETRACT will override the automatic control and drive the cowls to the position called for. When the landing gear handle is placed to the DN position the cowls will drive to the extend position regardless of switch position or airplane speed. Operation of the translating cowls can be ground checked by holding the translating cowl and mach trim test switch to the TEST position and placing the translating cowl switches to the EXTEND or RETRACT position.

Translating Cowl and Mach Trim Test Switch. 12 →

The translating cowl and mach trim test switch (4A, figure 1-16) located on the aft console, has two positions marked NORM and TEST. Placing the switch to the TEST position will allow checking the operation of the translating cowls on the ground. The switch is also used for ground checking engine mach lever operation.

Engine Trim Switches.

Two engine trim switches (21A, figure 1-4), located on the left console, are provided to make fine adjustments in engine fuel flow at MIL or A/B power settings to maintain turbine inlet temperature at or below the TIT limit. The switches are labeled ENG TRIM with an L and R for the respective engine. Each switch has three positions marked INCR (increase), DECR (decrease) and are spring loaded to an unmarked center position.

Compressor Bleed Valve Control Switches.

Two engine compressor bleed valve control switches (21B, figure 1-4), are located on the left console. The switches are labeled COMPRESSOR BLEED VALVE with and L an R for the respective engine. Each switch has two positions marked OPEN and CLOSED (NORM). At speeds above mach 1.6 the switches are placed to the OPEN position to bleed air from the engine compressor to prevent compressor stalls. On some airplanes the switches are labeled 5TH STAGE BLEED VALVE.

Foreign Object Damage (FOD) Prevention Door Switch. (12)→

The FOD prevention door switch (32, figure 1-4), located on the left console, is labeled FOD and has two positions marked EXTEND and RETRACT. The FOD prevention doors are extended or retracted when the switch is placed to the corresponding position. The switch controls 28 volt dc power from the main dc bus for operation of the doors.

Spike Control Switch.

Two spike control switches (3, figure 1-17), located on the center console, are labeled L SPIKE and R SPIKE respectively. Each switch has three positions marked OVERRIDE, AUTO and OFF. When a switch is positioned to AUTO, 28 volt dc power is directed to open the associated spike hydraulic shutoff valve to initiate automatic operation. When the switch is positioned to OFF, the valve closes, shutting off hydraulic power to the spike actuator. When the switch is positioned to OVERRIDE pneumatic pressure is applied to the spike actuators to position the spike full forward and fully contracted. On airplane (12)→, this switch is guarded from the OVERRIDE position.

Spike Test Buttons.

Two spike test buttons (19, figure 1-16), located on the aft console ground check panel, are provided to check operation of the spikes. The buttons are marked L SPIKE and R SPIKE. Depressing and holding either button will cause the respective spike to move to the full aft, fully expanded position. The spike caution lamps will light while the spikes are in transit. When the buttons are released the spikes will move to the full forward, fully contracted position.

Engine Tachometers.

Two engine tachometers (27, figure 1-5), located on the left main instrument panel, indicate the percent of RPM of the high pressure compressor (N2) in each engine. Each tachometer main dial is graduated from 0 to 100 percent rpm in increments of 2 percent, the subdial is graduated from 0 to 10 percent in increments of 1 percent.

Engine Fuel Flow Indicators.

Two engine fuel flow indicators (29, figure 1-5), located on the left main instrument panel, show fuel flow for each engine in pounds per hour. The indicators are calibrated from 0 to 80,000 pph in increments of 2000 pph. A digital readout of fuel flow is displayed on the face of the indicator. This readout shows fuel flow to the nearest 50 pph.

Engine Nozzle Position Indicators.

Two engine nozzle position indicators (30, figure 1-5), located on the left main instrument panel, show nozzle position. The indicators are calibrated from 0 (smallest nozzle area) to 10 (largest nozzle area). The indicators use 115 volt ac power from the essential ac bus.

Engine Oil Pressure Indicators.

Two engine oil pressure indicators (31, figure 1-5), located on the left main instrument panel, indicate engine oil pressure in pounds per square inch. The indicators are calibrated from zero to 100 psi in increments of 5 psi. The oil pressure indicating system operates on 26 volt ac which has been reduced by a transformer which has an input of 115 volts ac from the ac essential bus.

Engine Pressure Ratio Indicator.

Two engine pressure ratio (EPR) indicators (26, figure 1-5), located on the left main instrument panel, indicate the ratio of turbine discharge pressure to engine inlet pressure. The main dial of each indicator is calibrated from 1.0 to 3.0 in 0.1 increments. A smaller circular dial (subdial) on the indicator face is calibrated in 0.01 increments for precise reading. A set button on the lower right of each indicator permits movement of a reference pointer on the perimeter of the indicator to serve as an index for computed EPR. The precise EPR position of the reference pointer is displayed by a digital readout window on the indicator face. 115 volt ac power to the indicators is supplied from the essential ac bus.

Engine Turbine Inlet Temperature Indicators.

Two engine turbine inlet temperature (TIT) indicators (28, figure 1-5), located on the left main instrument panel, show turbine inlet temperature in degrees centigrade. The indicator dials are graduated from 0 to 1400 degrees in 50 degree increments. In addition, a digital readout of turbine inlet temperature in 1 degree increments is displayed. On some indicators installed on airplanes (1)→(12), a red flag with the letters HOT is displayed on the face of the indicator if TIT for the respective engine exceeds the allowable limit during engine start. Power to the TIT indicators is supplied from the 28 volt dc engine start bus. A flag marked OFF appears on the face of the indicator when power to the indicator is interrupted.

Spike Caution Lamps.

Two amber spike caution lamps, one for the spike in each engine inlet, are located on the main caution lamp panel (figure 1-21A). When lighted, the letters

Left Main Instrument Panel (Typical)

1. LEFT WARNING AND CAUTION PANEL
2. LANDING GEAR CONTROL PANEL
3. ANGLE-OF-ATTACK INDEXERS
4. GUN SWITCH
5. AIRSPEED MACH INDICATOR

5A. ROUNDS COUNTER

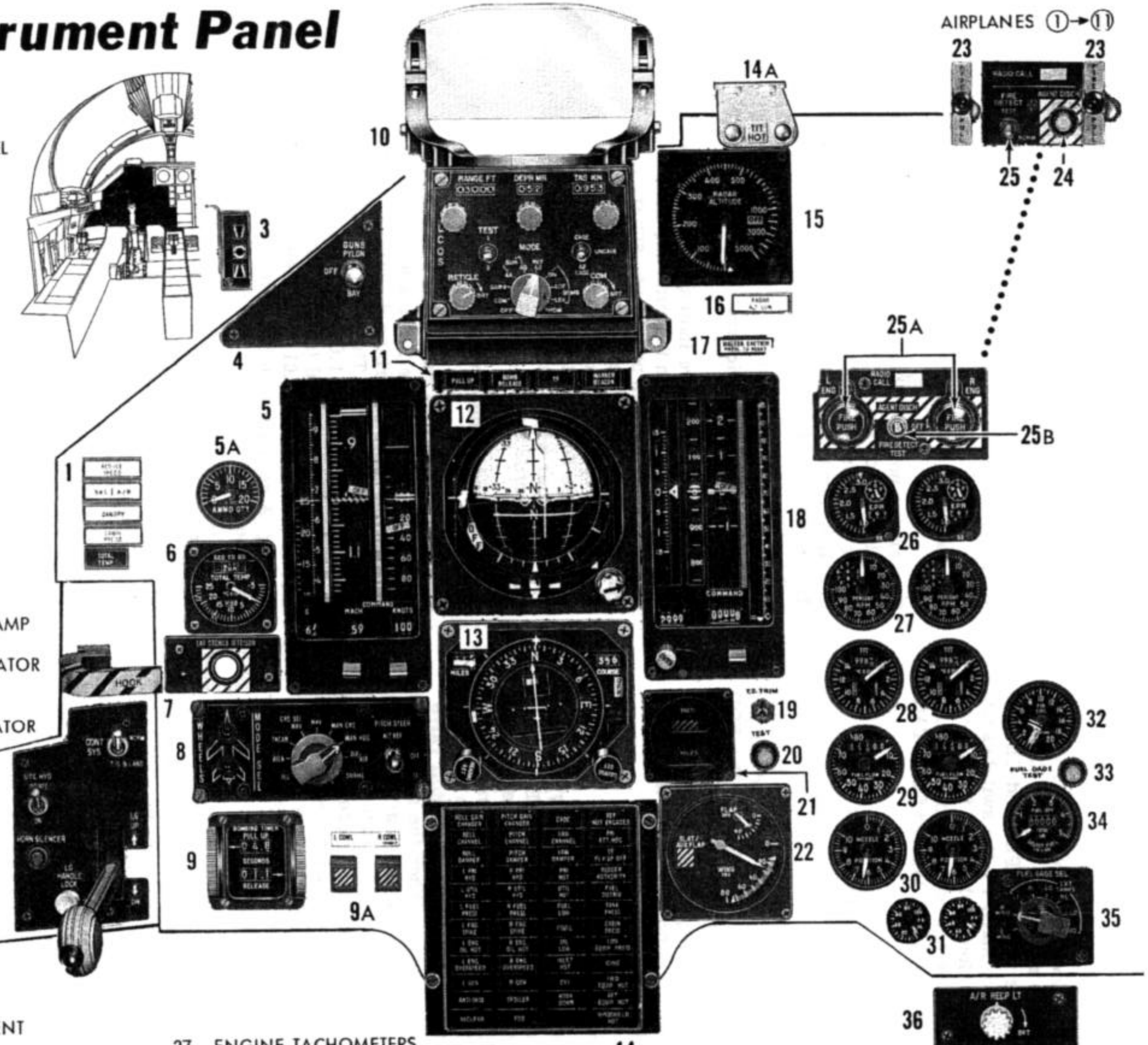
6. TOTAL TEMPERATURE INDICATOR
7. EXTERNAL STORES JETTISON BUTTON
8. INSTRUMENT SYSTEM COUPLER CONTROL PANEL
9. DUAL BOMBING TIMER

9A. TRANSLATING COWL POSITION INDICATORS

10. LEAD COMPUTING OPTICAL SIGHT AND CONTROL PANEL
11. UPPER WARNING AND CAUTION LAMP PANEL
12. ATTITUDE DIRECTOR INDICATOR
13. HORIZONTAL SITUATION INDICATOR
14. MAIN CAUTION LAMP PANEL

14A. TIT CAUTION LAMPS

15. RADAR ALTIMETER
16. RADAR ALTITUDE LOW WARNING LAMP
17. MASTER CAUTION LAMP
18. ALTITUDE-VERTICAL VELOCITY INDICATOR
19. TAKEOFF TRIM INDICATOR LAMP
20. TAKEOFF TRIM TEST BUTTON
21. BOMB NAV. DISTANCE TIME INDICATOR



22. SURFACE POSITION INDICATOR
23. ENGINE FIRE PULL HANDLES
24. ENGINE FIRE EXTINGUISHING AGENT DISCHARGE BUTTON
25. ENGINE FIRE DETECTION TEST SWITCH

25A. ENGINE FIRE PUSHBUTTON WARNING LAMPS

25B. AGENT DISCHARGE/FIRE DETECT TEST SWITCH

26. ENGINE PRESSURE RATIO INDICATORS
27. ENGINE TACHOMETERS
28. ENGINE TURBINE INLET TEMPERATURE INDICATORS
29. ENGINE FUEL FLOW INDICATORS
30. ENGINE NOZZLE POSITION INDICATORS
31. ENGINE OIL PRESSURE INDICATORS
32. FUSELAGE FUEL QUANTITY INDICATOR
33. FUEL QUANTITY INDICATOR TEST BUTTON
34. TOTAL FUEL QUANTITY INDICATOR
35. FUEL QUANTITY INDICATOR SELECTOR KNOB
36. AIR REFUELING RECEPTACLE LIGHTS CONTROL KNOB

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Figure 1-5.

L ENG SPIKE and R ENG SPIKE, respectively are visible. A spike caution lamp lights when the aircraft mach number is less than 0.3 and the respective spike is not full forward and fully contracted. When the spike control switch is placed to OVERRIDE the spike caution lamp will light and remain on until the spike has reached the full forward and fully contracted position. During spike self test the lamps will light until the spike has reached its full aft and full expanded position. The lamps operate on 28 volt dc electrical power from the essential dc bus.

Translating Cowl Position Indicators.

Two translating cowl indicators (9A, figure 1-5), located on the left main instrument panel, indicate the position of the translating cowls. The indicators are labeled L COWL and R COWL for the respective cowl. When the cowls are full open the indicators display the word OPEN. When the cowls are in transit the indicators will show cross hatched. With the cowls closed the indicators will display the word CLOSED.

Translating Cowl Caution Lamp.

An amber translating cowl caution lamp located on the main caution lamp panel (figure 1-21A), will light if one or both translating cowls are not extended at speeds below mach 0.2 (mach 0.35 on airplanes 12→) or if one or both cowls are not retracted at speeds above mach 0.6 (0.5 for more than 15 seconds on airplanes 12→). When the lamp lights, the letters COWL will be visible. The lamp operates on 28 volt dc power from the dc essential bus.

Foreign Object Damage (FOD) Prevention Door Caution Lamp. 12→

The FOD prevention door caution lamp (figure 1-21A), located on the main caution lamp panel, will light when the FOD doors are extended. When lighted the letters FOD are visible.

Engine Oil Hot Caution Lamps.

The two engine oil hot caution lamps are located on the main caution lamp panel (figure 1-21A). When the oil temperature of either engine exceeds 245°F, the associated lamp will light. When lighted, the following letters will be visible in the lense of the respective lamp: L ENG OIL HOT; and R ENG OIL HOT.

ENGINE FIRE DETECTION AND EXTINGUISHING SYSTEM.

Engine fire detection is provided by sensing elements routed throughout each engine compartment. Should a fire or overheat condition occur the rise in temperature is detected by the sensors which light the respective left or right engine fire warning lamp. Shutoff valves are provided to isolate fuel and hydraulic fluid from the affected engine. After the shutoff valves are closed fire extinguishing agent can be discharged into the affected engine compartment to put out the fire. The extinguishing agent is contained in a single container with a separate discharge valve for each engine. Self test features are incorporated in the system for maintenance checks and troubleshooting.

Fire Pull Handles. 1→11

Two fire pull handles (23, figure 1-5), marked FIRE PULL, are located on the left main instrument panel. Pulling either handle will close the engine fuel shutoff valve and the utility and primary hydraulic system shutoff valves to the respective engine and will arm the extinguishing agent discharge circuit to the affected engine. When the handle is pushed in the fuel shutoff valve is opened, the hydraulic shutoff valves will remain closed. A fire warning lamp is located in each handle.

Agent Discharge Button. 1→11

An agent discharge button (24, figure 1-5), marked AGENT DISCH, is located between the fire pull handles on the left main instrument panel. Once a fire pull handle has been pulled out depressing the button will discharge fire extinguishing agent into the engine compartment selected.

Fire Pushbutton Warning Lamps. 12→

Two fire pushbutton warning lamps (25A, figure 1-5), marked L ENG and R ENG, are located on the left main instrument panel. When a fire is indicated by a warning lamp, depressing either button will close the engine fuel shutoff valve and utility and primary hydraulic system shutoff valves to the respective engine and will arm the extinguishing agent discharge valve to the affected engine. Depressing the button again will open the fuel shutoff valve.

Agent Discharge/Fire Detect Test Switch. 12→

The agent discharge/fire detection test switch (25B, figure 1-5), located on the left main instrument panel, is a three position switch marked AGENT DISCH, OFF and FIRE DETECT TEST. The switch is spring loaded to the OFF position. Holding the switch to the AGENT DISCH position will discharge fire extinguishing agent into the engine compartment of the engine selected after depressing the affected engine fire pushbutton warning lamp.

Holding the switch to the FIRE DETECT TEST position will light both fire warning lamps if the fire detection system is operational.

Fire Detect Test Switch. 1→11

The fire detect test switch (25, figure 1-5), located on the left main instrument panel, is marked TEST and NORM. The switch is spring loaded to the NORM position. In the NORM position the system will function to detect any engine fire or overheat condition that may occur. Holding the switch to the TEST position will light both fire warning lamps if the system is operational.

Fire Detection System Short Test Button. (1)→(11)

The fire detection system short test button (17, figure 1-16), located on the aft console, is marked SHORT TEST. The button is provided to ground check the system short discriminating test circuit during maintenance or troubleshooting.

Fire Detection System Ground Test Switch. (1)→(11)

The fire detection system ground test switch (18, figure 1-16), located on the aft console, is a five position switch marked CONTROL BOX - L ENGINE, CONTROL BOX - R ENGINE, ELEMENT - L ENG and ELEMENT - R ENG. The switch is spring loaded to an unmarked NORM (center) position. The switch is used to ground check the system circuitry during maintenance or troubleshooting.

Fire Detection System Ground Test Switches. (12)→

Two fire detection system ground test switches (18, figure 1-16), located on the aft console, are labeled R ENG and L ENG. The switches have three positions marked CONTROL BOX, NORM and ELEMENT. The switches are spring loaded to the NORM (center) position and are used to ground check the system circuitry during maintenance or troubleshooting.

ENGINE OPERATION.

The following paragraph, containing information pertinent to engine operation will aid in the evaluation and correction of engine malfunctions.

Afterburner Control Malfunction.

In the event an afterburner fails to shut down when the associated throttle is retarded to MIL, move the throttle to IDLE. If this corrects the problem, normal operation of the engine up to MIL power may be restored, however, do not advance throttle beyond MIL. If retarding the throttle to IDLE does not correct the malfunction, shut the engine down. Following such an afterburner shut down, the engine may be restarted, but it is recommended that throttle settings above MIL be avoided for remainder of the flight.

OIL SUPPLY SYSTEM.

Each engine is equipped with an oil supply system which consists of an oil tank, a main supply pump, six scavenger pumps, a deoiler, two filters, an overboard breather pressurizing valve, a pressure valve, and three oil coolers (air-oil, fuel-oil, and afterburner fuel-oil). The air-oil cooler operates with engine bleed air. Oil is fed to the main oil supply pump from the oil tank. It is then pumped in series through the two filters, the air-oil cooler, fuel-oil cooler, and afterburner fuel-oil cooler. Oil flow through the fuel-oil coolers is controlled by temperature and pressure sensing bypass valves. The oil is then directed to the engine bearings and to the accessory gearbox. Scavenger pumps return the oil to the oil tank. Capacity of the tank is five gallons, four gallons of which are usable. For oil quantities and specification, see figure 1-50, Servicing Diagram.

Engine Oil Quantity Indicator. (8)→

The engine oil quantity indicator (10, figure 1-4), located on the left console, is a dual indicating instrument with two displays labeled L and R for the left and right engine respectively. Each display is graduated from 0 to 16 in one quart increments. A pointer for each display provides an indication of the number of quarts of oil remaining in each oil tank. The indicator operates on 115 volt dc power from the left main ac bus.

Engine Oil Quantity Indicator Test Button. (8)→

The engine oil quantity indicator test button (9, figure 1-4), located on the left console beside the oil quantity indicator, provides a means of checking the indicator. When the button is depressed and held the pointers will drive to predetermined values of 5 quarts on the left display and 5.7 quarts for the right display. When the button is released the pointers will return to their previous indications.

Oil Low Caution Lamp. (8)→

An oil low caution lamp, (figure 1-21A), located on the main caution lamp panel, lights any time the oil level in either the left or right engine oil supply tank drops to four (4) quarts. When the lamp is lighted the letters OIL LOW are visible.

FUEL SUPPLY SYSTEM.

The fuel supply system (figure 1-6), consists of a forward and aft integral fuselage tank, two integral wing tanks, an integral vent tank, and the associated fuel pumps, controls, and indicators. During normal operation, the left engine receives fuel from the forward fuselage tank, and the right engine receives fuel from the aft fuselage tank. Fuel from the wing tanks is transferred to the fuselage tanks before being delivered to the engines. The fuel system employs ten fuel pumps, of which six deliver fuel to the engine and four are used to transfer fuel from the wing tanks to the fuselage tanks. Provisions are made for inflight refueling of the internal and external fuel tanks from a boom-type tanker aircraft. Single-point refueling is provided for ground servicing and is accomplished through a standard ground refueling receptacle on the left side of the fuselage. All tanks are equipped with refuel automatic shutoff valves. On airplanes (12)→ gravity refueling can be accomplished through filler caps in the wings and fuselage. For fuel tank capacities, refer to T.O. 1F-111(Y)A-1A.

FUEL TANKS.

The forward fuselage tank extends from the aft bulkhead of the crew module to the bulkhead forward of the main wheel well. The forward tank is divided into three separate bays; F-1, F-2, and the reservoir tank. The bays are interconnected by standpipes and one-way flow flapper valves. The flapper valves allow fuel to flow from F-1 into F-2 and from F-2 to the reservoir. The reservoir tank is an integral part of the forward tank and retains 2538 pounds of fuel after all other fuel has been used. A float switch in the reservoir tank lights the FUEL LOW caution lamp when fuel remaining drops below approximately 2300 pounds. The aft fuel tank extends from aft of the main gear wheel well to a bulkhead at the rear of the fuselage structure. The aft tank is divided into two bays, A-1 and A-2. Each wing has an integral fuel tank that extends from the wing pivot structure to near the wing tips. The wing tanks cannot feed fuel directly to the engines. To use wing tank fuel, the fuel must first be transferred to the fuselage tank. A vent tank located in the vertical fin is provided for fuel expansion and for venting the fuselage and wing tanks.

FUEL PUMPS.

There are ten fuel pumps in the fuel system that operate on 115 volt, three phase, 400 cycle ac power. The six fuselage fuel pumps are dual inlet booster pumps, and the four wing fuel pumps are single inlet transfer pumps. Boost pumps 1 and 3 are in bay F-2, 2 and 4 are in the reservoir tank, and 5 and 6 are in bay A-1. Transfer pumps 7 and 9 are in the left wing, and transfer pumps 8 and 10 are in the right wing. Pumps 3, 4, 5 and 6 are the primary engine feed pumps, and 1 and 2 are standby engine feed pumps. Number 1 boost pump is a standby pump and operates continuously with the engine feed selector switch in any position except OFF. When not needed for engine

Fuel System (Typical)

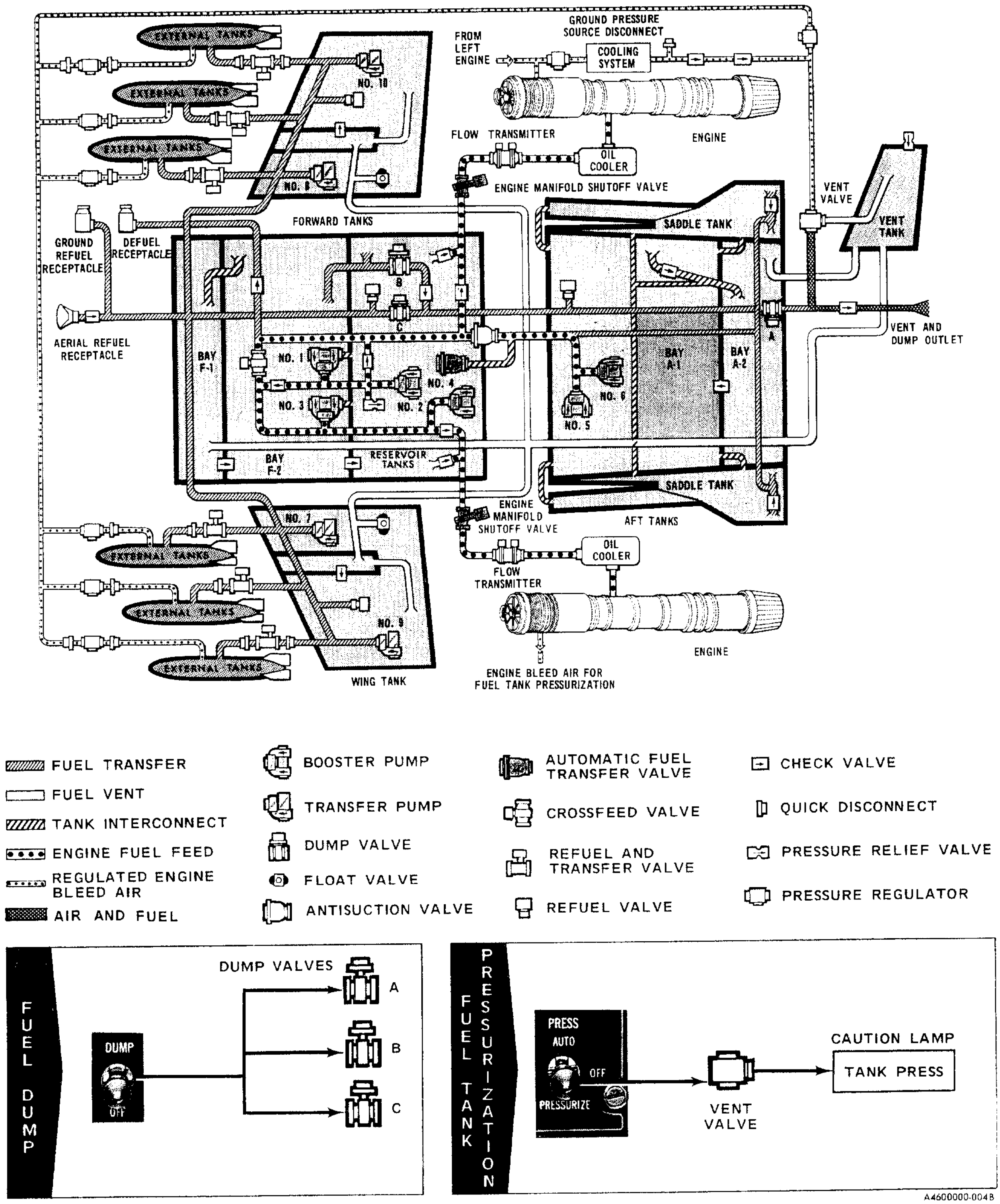


Figure 1-6. (Sheet 1 of 2)

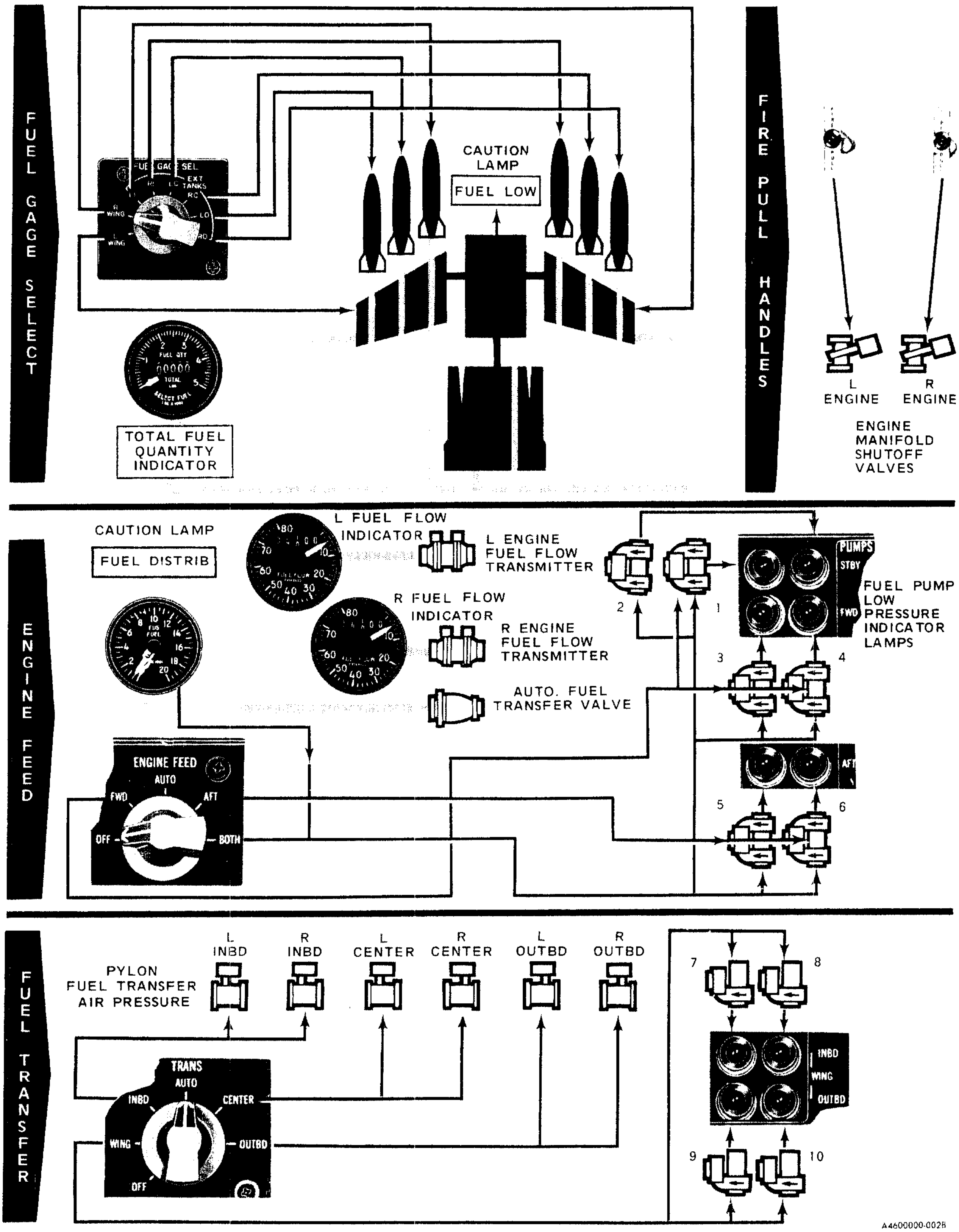


Figure 1-6. (Sheet 2 of 2)

fuel supply, the fuel provided by pump 1 is circulated into the reservoir tank through a pressure relief valve. The number 2 standby pump is energized by a pressure sensing switch whenever the manifold pressure falls below 16.1 psi above tank pressure.

ENGINE FUEL SUPPLY SYSTEM.

The engine fuel supply system controls the sequence of fuel flow to the engines and the transfer of fuel between the fuel tanks. The engine fuel supply system, when functioning in the automatic engine feed mode, maintains a predetermined fuel quantity difference between the forward fuel tank and the aft fuel tank in order to control the airplane center of gravity. There are four modes of engine fuel feed (AUTO, BOTH, FWD and AFT) which are controlled by the engine feed selector knob on the fuel control panel. The first mode (AUTO) is a feed condition in which the left engine receives fuel from the forward tank and the right engine receives fuel from the aft tank. In this mode a fuel differential between the forward and aft tank is automatically monitored and maintained by the gaging system. The second mode (BOTH) of operation is the same as the first mode except that the fuel differential between forward and aft tanks is not maintained by the gaging system. The third and fourth modes (FWD and AFT) are those in which both engine fuel manifolds can be fed by either the forward or aft tank alone.

FUEL TRANSFER.

In order to use the fuel in the wing internal tanks, it must be transferred to the fuselage tanks. The forward and aft refuel valves will open during transfer operation any time the tank is not full. Refuel valves cannot be controlled from the cockpit. The activation of the fuel transfer system is controlled by the fuel transfer switch on the fuel control panel. The fuel level in the fuselage tanks is maintained by float valves which open or close refuel valves. If AUTO engine feed is selected, an 8200 (±300) pound fuel differential is maintaintained between the forward fuselage tank and the aft fuselage tank, with the greater amount being in the forward tank. This differential is maintained by an automatic transfer of fuel from the aft tank to the forward tank when the differential is less than 8200 pounds. If the differential is greater than 8200 pounds, the aft fuel pumps are shut off and fuel is used from the forward tank until the differential is re-established. When transferring from the external tanks and wing tanks the fuel pump low pressure indicator lamps should be used in conjunction with the fuel quantity indicator to determine when the particular tank is empty. When emptying the wing tanks, the wing transfer fuel pump low pressure indicator lamps may not light simultaneously depending on wing sweep angle. With the wings forward the outboard transfer pumps will run out of fuel ahead of the inboard pumps. With the wings aft the reverse will occur.

FUEL PRESSURIZATION AND VENT SYSTEM.

The fuel pressurization system air is obtained from the engine compressor bleed line and is used to provide pressure for the fuel tanks. The system maintains a pressure between 5.0 and 6.0 psig in the tanks by means of the fuel tank vent and pressurization control valve. Should the fuel tank pressure approach 6.0 psig, the vent valve opens to vent the excess air overboard through the vent/dump outlet located at the lower aft end of the fuselage. If the pressure decreases, the valve controls air into the tank to maintain pressure.

FUEL DUMP SYSTEM.

The airplane is equipped with a fuel dump system capable of dumping fuel at a rate of 2300 pounds per minute minimum. During dumping operation, all fuel is automatically transferred to the forward fuselage tank. Fuel tank pressure then forces the fuel overboard through the fuel dump line. All fuel except that in the reservoir tank can be dumped. The fuel dump outlet is located between and aft of the engines directly beneath the rudder.

Note

- Fuel dumping should normally be accomplished with engines at MIL power or less. If dumping operation is necessary during afterburner operation, the fuel may ignite behind the airplane. This should cause no concern however since the fire will be behind the airplane. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.
- To eliminate prolonged fuel dripping from the fuel dump outlet after dumping is discontinued, the fuel system may be momentarily depressurized to clear residual fuel from the fuel dump lines. (This will occur automatically when the landing gear is extended for landing.)

AIR REFUELING SYSTEM.

The airplane is equipped with an air refueling system capable of receiving fuel from a flying-boom type tanker aircraft. The system consists of a hydraulically actuated receptacle and slipway door, a signal amplifier, and the associated controls and indicators. Hydraulic pressure for operation of the receptacle and its latch mechanism is supplied by the utility hydraulic system. The receptacle is located on top of the fuselage immediately aft of the crew module. When the receptacle is extended, a mechanical linkage retracts the aft end of the slipway door into the fuselage forming a slipway into the receptacle. When retracted, the upper surface of the receptacle and the slipway door join to form a surface flush with the fuselage skin. The refueling receptacle is equipped with two lamps located one on each side. As the receptacle extends, the lamps will light the receptacle and the slipway area. During normal refueling operations, the refueling boom enters the receptacle and is automatically latched in place by a hydraulically actuated latching mechanism. When the boom is latched in place, fuel flows through the receptacle and the refuel manifold into the fuel tanks at a rate of approximately 5100 to 5800 pounds per minute. As the tanks are filled, float operated valves automatically close the tank refueling valves shutting off flow to the tanks. As the last tank is filled, a disconnect signal is automatically sent to the boom latching mechanism to disconnect the boom, thus stopping fuel flow. A disconnect signal can be manually initiated at any time during refueling by either receiver pilot or by the tanker boom operator. If a disconnect cannot be made by other methods, a brute force pull-out can be safely accomplished.

SINGLE POINT REFUELING SYSTEM.

The airplane is equipped with a single point refueling system which enables all airplane fuel tanks to be pressure filled simultaneously from a single refueling receptacle. During ground refueling operations, fuel flows through the refueling receptacle and refueling manifold into the fuel tanks. As each tank fills, a float operated valve automatically closes the refuel valve stopping flow to the tank. The single point refueling receptacle is located on the left side of the fuselage forward of the engine air intake.

GRAVITY REFUELING. (12)→

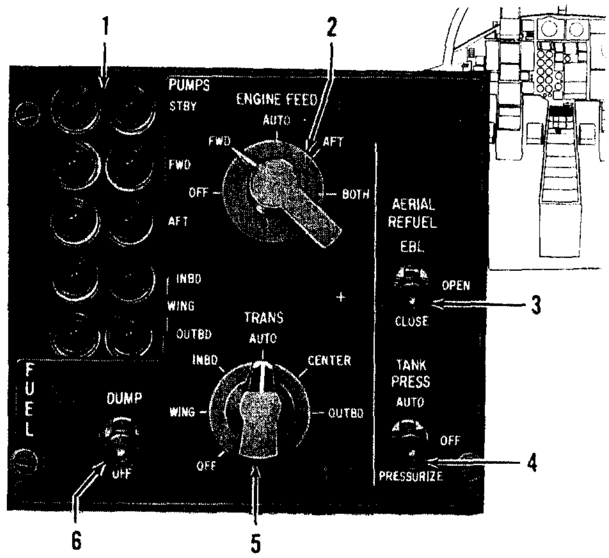
Gravity refueling is accomplished through filler caps in the top of the wing and fuselage. There is one filler cap in each wing on the trailing edge near the fuselage. There are five filler caps in the fuselage. One for F-1, F-2 and A-2 tanks and two for A-1 tank. Plug in type static ground receptacles for grounding the airplane during gravity refueling are provided under each wing and adjacent to the filler caps on the top of the fuselage. Each receptacle is marked GROUND HERE for identification.

FUEL SYSTEM CONTROLS AND INDICATORS.

Engine Feed Selector Knob.

The engine feed selector knob (2, figure 1-7), located on the fuel control panel, is a rotary, five-position detent switch marked OFF, FWD, AUTO, AFT, and BOTH. When the knob is rotated to OFF, all fuel boost pumps are de-energized. When the knob is rotated to FWD, boost pumps 1, 3, and 4 are energized, and boost pump 2 is placed on standby. In this configuration, both engines are fed from the forward fuel tank. When the knob is rotated to AUTO, boost pumps 1, 3, 4, 5, and 6 are energized, and boost pump 2 is on standby. In this configuration the left engine receives fuel from the forward tank and the right engine receives fuel from the aft tank with a differential of 8200 pounds automatically maintained between the two tanks. When the knob is rotated to AFT position, boost pumps 1, 5, and 6 are energized, boost pump 2 is on standby, and both engines are fed

Fuel Control Panel



1. FUEL PUMP LOW PRESSURE INDICATOR LAMPS
2. ENGINE FEED SELECTOR KNOB
3. AIR REFUEL SWITCH
4. FUEL TANK PRESSURIZATION SELECTOR SWITCH
5. FUEL TRANSFER KNOB
6. FUEL DUMP SWITCH

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Figure 1-7.

from the aft fuselage tank. When the knob is rotated to BOTH, boost pumps 1, 3, 4, 5, and 6 are energized and boost pump 2 is on standby. In this configuration the left engine is fed from the forward fuselage tank, and the right engine is fed from the aft fuselage tank. Fuel distribution will be maintained if fuel flow to each engine is equal, however the 8200 pound differential between tanks is not maintained by the gaging system.

Fuel Transfer Knob.

The fuel transfer knob (5, figure 1-7), located on the fuel control panel, is a six-position rotary switch marked WING, INBD, AUTO, CENTER, OUTBD, and OFF. When the knob is in the OFF position, all fuel transfer functions are off. When the knob is rotated to WING, four transfer pumps, two in each wing tank, are energized; and fuel is transferred from the wing tanks to the fuselage tanks. The INBD, CENTER, and OUTBD positions of the knob are for transferring fuel from external tanks when installed. The AUTO position automatically sequences the transfer of fuel from the outboard, center, and inboard external tanks and the wing tanks in that order. If an external tank is not installed, the sequence of transfer remains the same except the missing tank is skipped.

Fuel Dump Switch.

The fuel dump switch (6, figure 1-7), located on the fuel control panel, is a two position switch marked

DUMP and OFF. When the switch is in the OFF position, dump valves A and B are closed and C is open. When the switch is positioned to DUMP, dump valves A and B are opened, C is closed, the automatic transfer valve is opened, the tanks are pressurized and fuel booster pumps 5 and 6 and fuel transfer pumps 7, 8, 9, and 10 are energized, thereby transferring fuel from the wing and aft tanks to the forward tank to be dumped. The fuel tanks will pressurize when the dump switch is in DUMP regardless of the position of the fuel tank pressurization selector switch, the landing gear, or the air refueling door.

Fuel Tank Pressurization Selector Switch.

The fuel tank pressurization selector switch (4, figure 1-7), located on the fuel control panel, is a three-position, lever-lock toggle switch marked AUTO, OFF, and PRESSURIZE. When the switch is positioned to AUTO, the fuel tanks are pressurized, except when the landing gear is down, or the aerial refueling door is open. When the switch is placed to OFF, the pressurization airflow to the tanks is turned off and the tanks are vented. When the switch is placed to PRESSURIZE and pressurization air is available, fuel tank pressurization is maintained with the landing gear down or the refuel door open.

Air Refueling Switch.

The air refueling switch (3, figure 1-7), located on the fuel control panel, is a three position lever lock toggle switch marked EBL, OPEN and CLOSE. When the switch is positioned to OPEN, hydraulic pressure is directed to the refueling receptacle actuating cylinder to extend the receptacle and retract the fairing door. Positioning the switch to CLOSE will cause the refueling receptacle to retract. The EBL (emergency boom latch) position is used in the event that the automatic latching feature of the receptacle fails to latch the boom in place. Positioning the switch to EBL will provide a signal directly to the hydraulic latching mechanism through a boom contact switch in the receptacle, bypassing the signal amplifier. This will cause the hydraulic latching actuators to latch the boom in place. When the switch is in EBL, the automatic disconnect signal will not be available to unlatch the boom at completion of refueling. The boom must be manually disconnected by depressing and holding the nose wheel steering and air refuel button located on either control stick grip. If a disconnect cannot be made by other methods, a brute force pull-out can be safely accomplished. Electrical power is supplied to all external tank refuel valves when the switch is in the OPEN or EBL position.

Nose Wheel Steering/Air Refuel Buttons.

The nose wheel steering/air refuel buttons (4, figure 1-15), located one on each control stick grip, are labeled NWS and A/R DISC. The air refueling function of the button provides a means of manually disconnecting the air refueling boom. Depressing either button will interrupt power to the boom latching mechanism causing it to unlatch. For a description of the NWS function of the buttons, refer to "Nose Wheel Steering System," this section.

Precheck Selector Valves.

Four precheck selector valves, one for each internal fuel tank, are located on a recessed panel just aft and below the single point refueling receptacle. The valves are provided to functionally test the refuel valves. The valves are labeled FWD TK (forward tank), AFT TK (aft tank), LH WG TK (left wing tank), and RH WG TK. On airplanes 1→11 each valve has positions marked SEC (secondary), OFF, and PRI (primary). Positioning a selector valve to PRI while the aircraft is being fueled will cause fuel to flow into a bowl at the selected float valve and cause the float to be submerged, simulating a full tank. This will cause the refuel shutoff valve to close. The secondary portion of the precheck system is inactive. When the OFF position is selected, the bowl at the float valve will drain and the float is allowed to drop unless the tank is full. On airplanes 12→ the fwd and aft tank valves are labeled SEC, REFUEL, and PRI. The PRI and SEC positions check the primary and secondary floats respectively as described above. The REFUEL position is identical to the OFF position described above. The wing tank selector valves at the precheck selector panel are marked REFUEL and CK (check). In addition to the valves at the precheck panel, there is a selector valve in each wing tank. This valve appears as a rod with a screwdriver slot flush with the lower surface of the wing. This valve is used to select the float valve in the wing tank to be checked when the selector at the refuel station is in the CK position.

Stores Refuel Power Switch.

The stores refuel power switch (4, figure 1-16), located on the aft console, has positions marked BATTERY and NORM. The switch is provided to supply battery power to the external fuel tank fuel valves for single point ground refueling. Placing the switch to BATTERY will provide 28 volt battery power to the valves and permit the external tanks to be fueled. A float switch in the external tank will break the circuit and shut off the flow when the tank is full. Placing the switch to NORM, deenergizes the valves.

Position Lights/ Stores Refuel Battery Power Switch. 12→

The position lights/stores refuel battery power switch (4, figure 1-16), located on the aft console, has three positions marked POS LIGHTS, NORM and STORES REFUEL. Placing the switch to the STORES REFUEL position will supply battery power to the external fuel tank fuel valves for single point ground refueling. A float switch in the external tank will break the circuit and shut off the flow of fuel when the tank is full. Placing the switch to NORM deenergizes the circuit. The switch is mechanically held in the NORM position when the ground check panel door is closed. For a description of the POS LIGHTS position of the switch refer to Lighting System, this section.

Total/Select Fuel Quantity Indicator.

The total/select fuel quantity indicator (34, figure 1-5), located on the left main instrument panel, provides indications of total fuel in all tanks and the fuel remaining in individual wing or external pylon tanks. The indicator is graduated from zero to 5 (times 1000 pounds) in increments of 100 pounds and has a pointer and a five digit counter. The pointer will read the fuel remaining in the wing or external tank as selected by the fuel quantity indicator selector knob. The counter continuously reads the total fuel remaining in all tanks. Due to fuel quantity indicating system tolerance it is possible to have a small amount of fuel remaining in the wing tanks when the select fuel indicator reads empty. The fuel pump low pressure indicator lamps for the wing transfer pumps provide the most positive indication that the wing tanks are completely empty. The select fuel quantity indicator circuit uses a single compensator sensor which is located in the aft fuel tank. If the aft tank is emptied while there is fuel in one or more of the wing or external tanks, the uncovering of the compensator will cause the select gage indications to read erroneously high.

Fuselage Fuel Quantity Indicator.

The fuselage fuel quantity indicator (32, figure 1-5), located on the left main instrument panel, indicates the amount of fuel remaining in the forward and aft fuselage tanks and the amount of fuel differential between the two tanks. The indicator is graduated from zero to 20 (times 1000 pounds) in 500 pound increments. The indicator has two pointers marked F (forward and A (aft) for the forward and aft tanks. When operating in automatic engine feed, the A pointer will be maintained approximately 8200 pounds below the F pointer. In this position the F pointer will be between two dot indices on the outer scale of the indicator. The two dots indicate the point at which aft to forward fuel transfer will occur to maintain the 8200 pound fuel differential. Two bar indices (2 additional dots on some airplanes) outboard of the dots indicate the point at which the fuel distribution caution lamp will light to indicate the fuel differential between the forward and aft tanks is out of tolerance. The indices move with the A pointer and thus provide a ready reference of fuel differential when operating in manual engine feed.

Fuel Quantity Indicator Test Button.

The fuel quantity indicator test button (33, figure 1-5), located on the left main instrument panel, is provided to test the fuselage fuel quantity and total/select fuel quantity indicators. When the test button is depressed each of the three pointers and the total fuel digital counter will drive to the following indications: Forward and aft tank pointers, 2000 (±400). Select tank pointer, 2000 (±100). Total fuel digital counter, 2000 (±1250). The indicators will either increase or decrease to the 2000 pound reading depending on the quantity of fuel in the tanks. A normal confidence check of fuel quantity indicator operation may be made by depressing the test button long enough to observe movement of the pointers and counter. When the button is released the pointers and counter will return to their original readings.

Fuel Quantity Indicator Selector Knob.

The fuel quantity indicator selector knob (35, figure 1-5), located on the left main instrument panel, has eight positions marked L WING, R WING, LI (left inboard external tank), RI, LC (left center external tank), RC, LO (left outboard external tank) and RO. Placing the knob to the desired tank enables reading the amount of fuel remaining in that tank on the total/select fuel quantity indicator.

Fuel Manifold Low Pressure Caution Lamps.

Two amber fuel manifold low pressure caution lamps (figure 1-21A), are located on the main caution lamp panel. The letters R FUEL PRESS or L FUEL PRESS are visible when the respective lamp is lighted. The applicable lamp lights any time the fuel pressure in the right or left fuel manifold is less than 15.5 psia.

Fuel Low Caution Lamp.

The amber fuel low caution lamp (figure 1-21A), located on the main caution lamp panel, lights any time fuel in the reservoir tank is less than 2350 (±235) pounds. Fuel quantity indication will be between 1700 and 3000 pounds in the forward tank when the lamp lights. When the lamp is lighted, the letters FUEL LOW are visible.

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Fuel Pump Low Pressure Indicator Lamps.

Ten green lamps (1, figure 1-7), located on the left side of the fuel panel, are fuel pump low pressure indicator lamps. When a fuel pump is energized, whether by automatic or manual tank selection, and the pump is not generating at least 3.5 psi, the corresponding green lamp lights. The uppermost two lamps are for the standby booster pumps in the forward tank. The next two lamps are for the forward fuselage tank booster pumps. The next two lamps are for the aft fuselage tank booster pumps. The next two lamps reflect the wing inboard transfer pumps, and the lower two lamps are for the wing outboard transfer pumps. The fuel pump low pressure indicator lamps may not light simultaneously depending on wing sweep angle. With the wings forward the outboard transfer pumps will run out of fuel before the inboard pumps. With the wings aft the reverse will occur.

Fuel Tank Pressurization Caution Lamp.

The amber fuel tank pressurization lamp (figure 1-21A), located on the main caution lamp panel, lights when fuel tank air pressure drops below approximately 3.5(±.5) psi during flight with the landing gear and the air refueling receptacle retracted. The lamp also lights any time the fuel tanks are pressurized and the landing gear or air refueling receptacle is extended. When the lamp lights the letters TANK PRESS are visible.

Fuel Distribution Caution Lamp.

The amber fuel distribution caution lamp (figure 1-21A), located on the main caution lamp panel, is provided to indicate when the fuel distribution between the forward and aft tanks is out of tolerance. On airplanes 1 → 11 when the fuel distribution is less or greater than 8200 (±800) pounds differential, the lamp will light. On airplanes 12 → the lamp will light at 8200 (plus 1800 minus 800) pounds. When lighted, the letters FUEL DISTRIB are visible.

Nose Wheel Steering/Air Refueling Indicator Lamp.

The green nose wheel steering and air refueling indicator lamp (1, figure 1-5), located on the left main instrument panel, is labeled NWS A/R. For air refueling, the lamp provides an indication when the air refueling circuitry is set to receive the refueling boom. As the receptacle extends into place, the lamp will light. When the boom is latched in the receptacle, the lamp will go out. When the boom disconnects, the lamp will again light. On airplanes 1 → 11 the intensity of the lamp can be controlled with the malfunction and indicator lamp dimming switch. On airplanes 12 → the lamp cannot be dimmed. For a description of the NWS function of the lamp, refer to "Nose Wheel Steering System", this section.

Fuel Tank Pressure Gage.

The tank pressure gage, located adjacent to the single point refueling receptacle is provided to monitor tank pressure during ground refueling. The gage is graduated from 0 to 15 psi, in 0.5 psi increments.

FUEL SYSTEM OPERATION.

Automatic Fuel System Management.

Normal fuel system management is accomplished by positioning the engine feed selector switch to AUTO and the fuel transfer switch to AUTO. This establishes a mode of operation whereby fuel is automatically transferred in sequence from the external tanks and wing fuel tanks to the fuselage fuel tanks. The remaining fuel in each tank, other than the fuselage tanks, may be monitored at any time by positioning the fuel gage selector switch to the desired tank position and observing the indication on the select fuel quantity indicator.

Manual Fuel System Management.

In the event that either the automatic engine feed or automatic fuel transfer modes of operation become inoperative or manual control of the fuel system is desired, several alternate modes of fuel flow control may be selected.

Manual Engine Fuel Supply. When the engine feed selector switch is in the FWD, AFT, or BOTH position, the automatic fuel distribution system between the forward and aft fuselage fuel tank is inoperative. If a change in fuel distribution between the forward and aft fuselage tanks is needed, it is obtained by alternately switching from BOTH to FWD or AFT as necessary. The relative fuel levels will be indicated on the fuselage fuel indicator.

Manual Fuel Transfer. If the automatic fuel transfer mode should become inoperative or manual transfer of fuel is desired, the fuel transfer switch may be positioned to OUTBD, CENTER, INBD, or WING as necessary. During a normal manual transfer of fuel, the fuel transfer switch is positioned first to OUTBD (outboard pylon tank) and the fuel gage selector switch is positioned to EXT TANKS - LO or RO. It will be necessary to periodically switch the fuel gage selector switch back and forth between LO and RO to monitor the fuel level in each outboard pylon tank. When the fuel quantity gage indicates that both tanks are empty, the fuel transfer switch is positioned to CENTER and the fuel gage selector switch is positioned to EXT TANKS - LC or RC. When the center pylon tanks are empty, the inboard pylon tanks are selected. After the inboard pylon tanks are emptied the fuel transfer switch is positioned to WING and the fuel gage selector switch is positioned to L WING or R WING. During fuel transfer from the wings and external tanks there is a tendency for more of the fuel to be transferred to the aft tank on airplanes 1 → 7 9 and 11 or the forward tank on airplanes 8 10 12. This tendency normally has no effect on the fuselage tank fuel distribution since these tanks are maintained full during the transfer operation. If the fuel levels in the fuselage tanks are lowered before the wing and external fuel is transferred, AUTO engine feed should be used to achieve the proper differential between the forward and aft tanks. On airplanes (1)→(7) (9) and (11) the fuel distribution system will make corrections by transferring fuel forward. This will be accomplished at a rate sufficient to prevent the fuel distribution caution lamp from lighting. On airplanes (8) (10) (12) the distribution is corrected by burning fuel from the forward tank. During low fuel consumption rates, the fuel distribution caution lamps may light indicating excessive fuel in the forward tank.

Fuel Dumping.

During fuel dumping operations it should be noted that the automatic center-of-gravity control will not operate normally. If the engine feed selector switch is in AUTO during dumping, the No. 5 fuel pump in the aft tank will shut off when the 8200 pound fuel differential is exceeded. The No. 6 pump will continue to run. Assuming that fuel is also being transferred from the wing tanks, the forward fuselage tank will remain nearly full while the aft fuselage and wing tanks are emptying. This will cause the center-of-gravity to gradually shift forward. When the wing tanks are emptied, fuel from the forward fuselage tank will be dumped at a faster rate than that being transferred from the aft fuselage tank. This will cause the center-of-gravity to shift aft until the 8200 pound fuel differential is reestablished. From this point until the aft fuselage tank is empty, the No. 5 fuel pump in the aft tank will cycle on and off to maintain the 8200 pound fuel differential.

Air Refueling.

For air refueling system operation, refer to Section II.

Single Point Refueling.

1. Airplane and refueling equipment - Grounded.
   Insure that the airplane and all refueling equipment are statically grounded.

2. Nose gear chocks - Removed.
   Remove all entrance ladders and equipment under the aircraft which might cause damage when the landing gear shock struts compress due to increased fuel load.

3. Precheck selector valves - OFF or REFUEL. (as applicable)

3A. (1)→(11) Stores refuel power switch - BATTERY.
   If external tanks are installed, place the stores refuel power switch to BATTERY.

3B. 12 Position lights/stores refuel battery power switch - STORES REFUEL.
   If external tanks are installed, place the position lights/stores refuel battery power switch to STORES REFUEL.

4. Fueling hose ground cable - Connected.
   Connect the grounding cable from the fueling hose to the airplane.

5. Ground refueling receptacle cap - Removed.

6. Fuel nozzle - Connected to refueling receptacle.

7. Start fuel servicing unit and open fuel nozzle.

8. Precheck selector valves - PRI or CK. (as applicable)
   Within a few seconds after fuel flow is indicated, position all precheck selector valves to PRI or CK as applicable. The fuel flow should drop to less than 10 gpm indicating that all primary valves have closed.

CAUTION

Do not allow fuel flow to the aft tank or wing tanks for more than a few seconds when the forward tank quantity is below 7500 pounds. To do so may cause a longitudinal unbalance and cause the airplane to tip up.

Note

If fuel flow drops to 5 gpm or less, proceed to step 9. If fuel flow does not drop, determine which refuel valve has malfunctioned as follows: On airplanes (1)→(11) connect electrical power and observe the fuel quantity gages while all tanks are prechecked to determine which tank is filling. On airplanes (12)→ select the aft tank valve to SEC and observe the flowmeter for 30 seconds, then select PRI. If flow did not drop below 5 gpm when SEC position was selected, repeat the test for the forward tank. If flow is not stopped when SEC position was selected for the forward tank, repeat the test for each wing by changing positions for the wing precheck selector valve located on the lower surface of each wing. The defective valve will be indicated by a drop of flow.

9. Fuselage tank precheck selector valves - OFF or REFUEL then SEC.
   Individually rotate the fuselage tank precheck selector valves to OFF or REFUEL as applicable and then to SEC while observing the flowmeter. Flow should rise at least 100 gpm while in OFF or REFUEL position indicating that the selected refuel valve has opened. The valve should then close when the SEC position is selected.

Note

On airplanes (1)→(11) the secondary portion of the precheck system is inactive, therefore it is necessary to return the precheck selector valves to PRI instead of SEC after it has been determined that the refuel valve has opened.

10. Precheck selector valves - OFF or REFUEL. (as applicable)
 Continue refueling operations.

11. Tank pressure gage - Monitor.
 If pressure exceeds 3 psi, discontinue refueling operation and determine the cause. The tanks should be depressurized and air should flow from the vent during fueling.

Note

Fuel tanks are full and valves are closed when the flowmeter on the fuel truck falls to zero.

12 thru 13. (Deleted)

14. Fuel nozzle - Closed.
 At completion of refueling, close the fuel nozzle and stop the refueling truck pump.

15. Fuel nozzle and grounding cable - Disconnected.

16. Refueling receptacle cap - Installed.

17. Single point refueling control access doors - Closed and latched.

18. Position lights/stores refuel battery power switch or stores refuel selector switch - NORM (if external tanks were fueled).

ELECTRICAL POWER SUPPLY SYSTEM.

The electrical power supply system provides 115 volt, three-phase, 400 cycle ac power and 28 volt dc power. Two ac generator drive assemblies, one mounted on each engine, supply ac power. Two transformer rectifier units provide 28 volt dc power. (See figure 1-8.)

ALTERNATING CURRENT POWER SUPPLY SYSTEM.

AC power is supplied by two 60 kva generating systems. Each generator is driven by a constant-speed drive assembly which regulates generator frequency at 400 cycles per second. Voltage regulation and system protective functions are performed by generator control units. There are three ac buses: a left main ac bus, a right main ac bus, and an essential ac bus. During normal operation, the right generator supplies power to the right main ac bus, and the left generator power the left main ac bus and the essential bus. Each generator is connected to its associated bus with multiple wire generator feeders. Power transfer contactors located near the main ac buses are used to switch the buses from one generator to another. Each main ac bus is normally individually powered and isolated from the other. The power contactors provide a bus tie function automatically in the event of a generator failure. If a fault or malfunction occurs causing an undervoltage, overvoltage, underfrequency, or overfrequency, the associated ac generator control unit removes the generator from the bus. Undervoltage or overvoltage de-excites the generator and disconnects it from the bus. Underfrequency or overfrequency does not de-excite the gencessive amount of heat occurs in the constant-speed drive (CSD) unit, a thermal device in the unit automatically decouples the drive from the engine. Once decoupled, the drive cannot be recoupled during flight. An emergency generator with a 10 kva output is provided to generate electrical power in the event of failure of both main ac generators. The emergency generator is driven by a hydraulic motor which receives power from the utility hydraulic system. In the event of loss of both primary generating systems, a solenoid-operated valve is deenergized, allowing hydraulic pressure to operate the emergency generator. Emergency generator power is applied to the ac and dc essential buses and to the 28 volt dc engine start bus.

Generator Switches.

The two generator switches (1, figure 1-10), located on the electrical control panel, are lever-lock type toggle switches with positions marked OFF, ON, and TEST. In the OFF position, the generator is not excited; the power contactor is open; and the generator system is reset.

Note

If a generator is de-excited while connected to the bus, it will not automatically reset, even though the fault condition is cleared. The generator switch must be positioned to OFF to reset the system.

Positioning the switch to ON will excite the generator and connect it to its respective ac bus. In the TEST position the generator will be disconnected from its bus and will be excited. The TEST position can be used to check generator operation without connecting it to a bus.

Generator Decouple Pushbuttons.

The generator decouple pushbuttons (7, figure 1-10), located on the electrical control panel, are provided to actuate the constant-speed drive decoupler. When a pushbutton is depressed, the constant-speed drive will be decoupled. Once decoupled, the constant-speed drive cannot be reconnected during flight.

Electrical Power Flow Indicator.

The electrical power flow indicator (2, figure 1-10), located on the electrical control panel, is a flip-flop type indicator labeled AC BUSSES and displays the various bus configurations. If both buses are receiving power from their respective generator, the indicator will display ISOL, indicating that the buses are isolated from each other and are operating normally. If only one generator is providing power for both buses, the indicator will display TIE. When the emergency generator is operating and supplying power to

Electrical Power Supply System

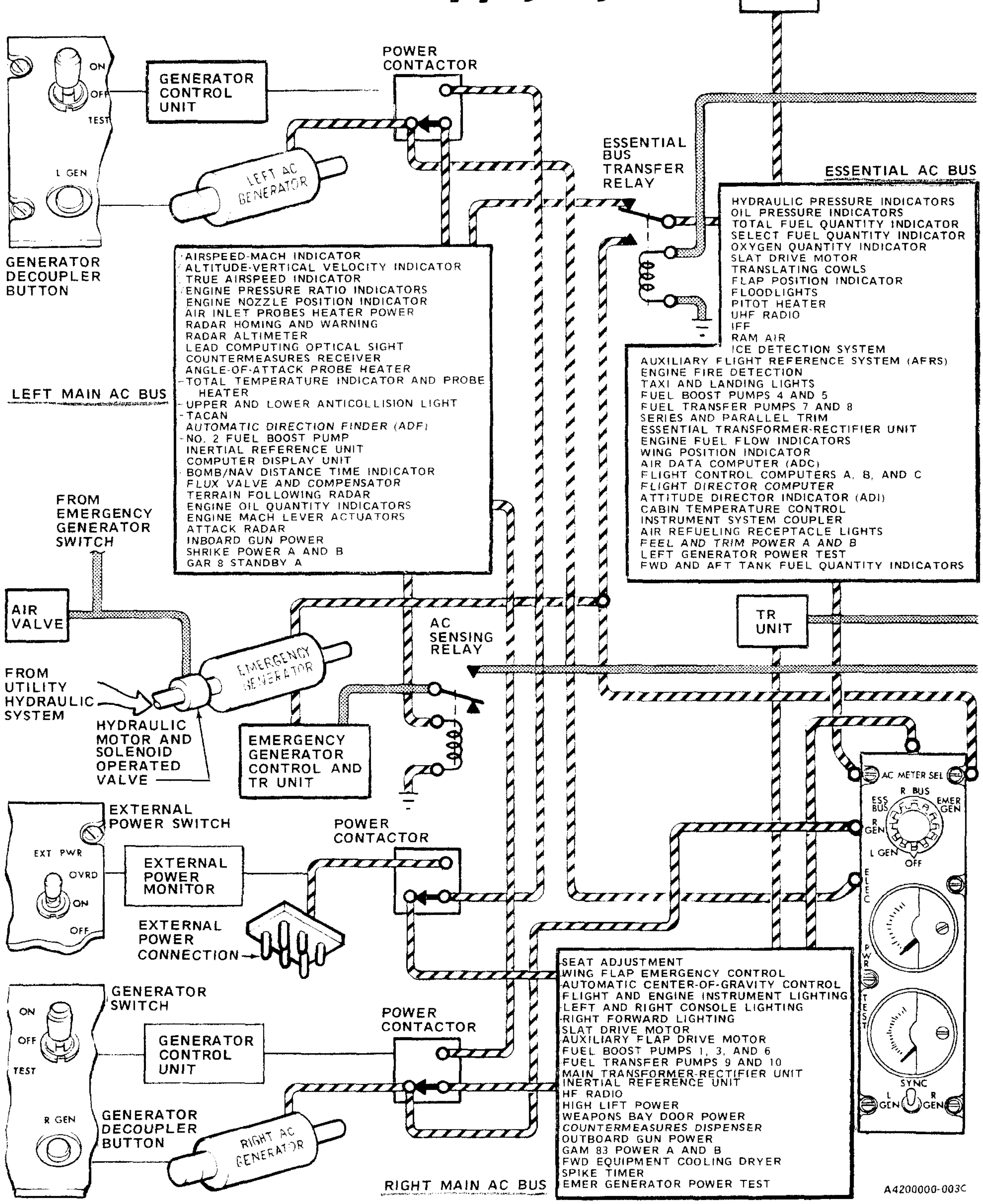


Figure 1-8. (Sheet 1 of 2)

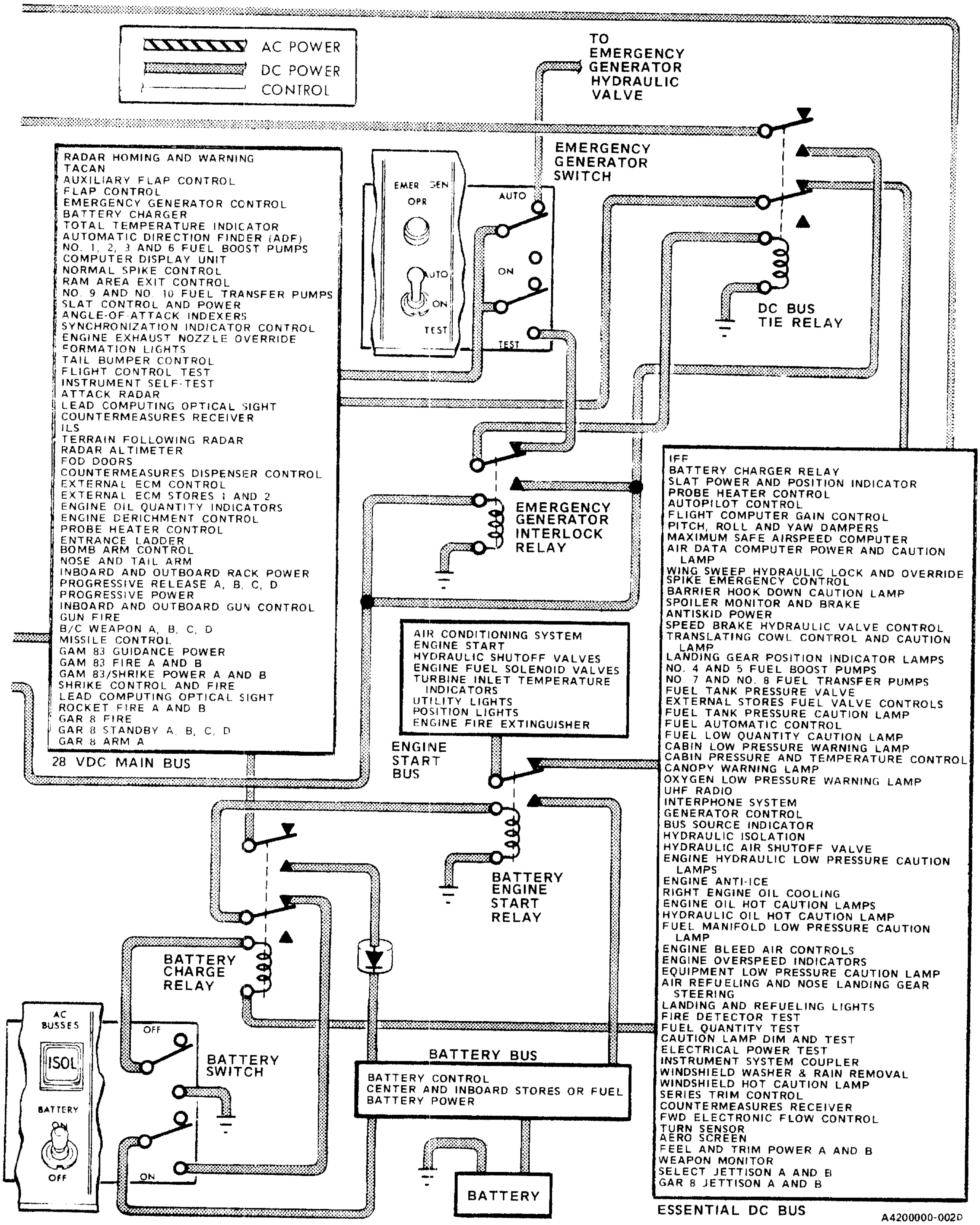


Figure 1-8. (Sheet 2 of 2)

the ac essential bus, the indicator will display EMER. When ground power is connected to the airplane and supplying power to the ac buses, the indicator will display TIE. When there is no ac power being applied to the airplane, the indicator will display a crosshatched surface.

Emergency Generator Switch.

The emergency generator switch (5, figure 1-10), located on the electrical control panel, is a toggle switch having positions marked ON, AUTO, and TEST. When the switch is in the ON position, the hydraulically driven emergency generator is operating, but not connected to the essential ac bus unless all ac power is lost. In the AUTO position, if all ac power is lost, the emergency generator hydraulic valve will open, the emergency generator will operate, and the ac essential bus transfer relay will be energized, thereby connecting the emergency generator to the essential ac bus.

Note

In the event that the emergency generator does not come on within 3 seconds, manually place the switch to the ON position.

When the switch is in the TEST position, the emergency generator operates and the emergency generator indicator lamp lights. The TEST position will not connect the emergency generator to the essential ac bus. The TEST position also opens the dc bus tie contactor to provide a method of checking operation of the two 28 volt dc converters. If the main and essential dc buses remain energized, both converters are operating and the electrical power flow indicator will display a crosshatch. If either converter has failed and the battery switch is ON, the power flow indicator will cycle or operate erratically.

External Power Switch.

The external power switch (4, figure 1-10), located on the electrical control panel, is a toggle switch having positions marked OFF, ON and OVRD. In the OFF position, external power cannot be supplied to the airplane ac buses. In the ON position with neither engine operating, external power supplies total airplane power. With the left engine operating, the left main ac generator will supply total airplane electrical load, and external power is disconnected from the ac buses. With only the right engine operating, the right main ac generator supplies power to the right main ac bus, and external power feeds the left main ac and essential buses. Associated with the external power is a power monitor which measures external power voltage, frequency and phase sequence. Should any one of these parameters be out of tolerance, the monitor prevents closing of the external power contactor. When the external power switch is in the OVRD position, the external power monitor circuit is bypassed, thus allowing external power which is out of voltage and frequency tolerance to be applied to airplane buses. The override position does not override external power with improper phase sequence.

AC Meter Selector Switch.

The ac meter selector switch (31, figure 1-4), located on the left console, is a rotary switch having positions marked OFF, L GEN, R GEN, ESS BUS, R BUS, and EMER GEN. When the switch is rotated from OFF to any one of the other positions, the frequency meter and ac voltmeter display the frequency and voltage of the bus or generator selected.

Emergency Generator Indicator Lamp.

The green emergency generator indicator lamp (3, figure 1-10), located on the electrical control panel, lights when the emergency generator is operating. The lamp receives power from the emergency generator through the emergency generator control unit.

Generator Caution Lamps.

Two amber generator caution lamps (figure 1-21A), are located on the main caution lamp panel. Either lamp lights when the respective generator is disconnected from the ac bus. When lighted, the letters L GEN are visible in the left lamp and R GEN in the right lamp.

Frequency Meter.

The frequency meter (29, figure 1-4), located on the left console, displays frequency in cycles per second (cps) of the various buses and generators as selected

Circuit Breaker Panel

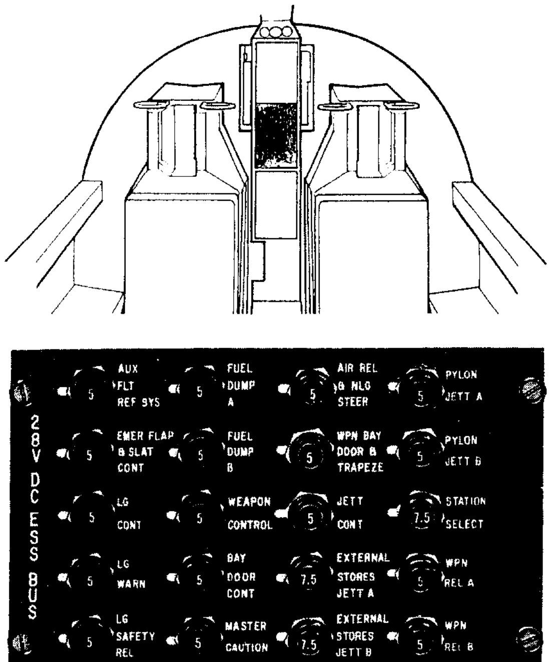
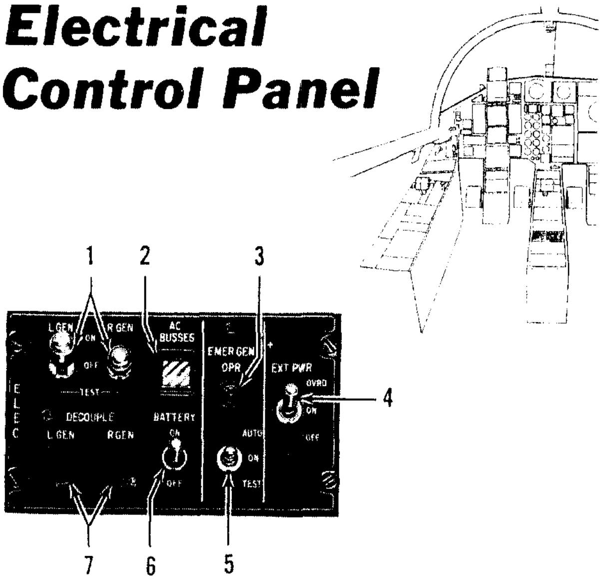


Figure 1-9. (1)→(11)

Electrical Control Panel



1. GENERATOR SWITCHES
2. ELECTRICAL POWER FLOW INDICATOR
3. EMERGENCY GENERATOR INDICATOR LAMP
4. EXTERNAL POWER SWITCH
5. EMERGENCY GENERATOR SWITCH
6. BATTERY SWITCH
7. GENERATOR DECOUPLE PUSHBUTTONS

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Figure 1-10.

dc bus during normal operation. Normally the outputs of the two transformer-rectifier units supply the total dc load in parallel.

Battery Switch.

The battery switch (6, figure 1-10), is located on the electrical control panel. The two position switch is marked OFF and ON. Positioning the switch to ON connects the engine start bus to the airplane 24 volt battery, provided the essential dc bus is not energized. If the essential dc bus is energized, the battery is connected to the main dc bus through the battery charger circuit. When the battery switch is positioned to OFF, the engine start bus is connected to the essential dc bus, and the battery charger circuit is disconnected from the main dc bus.

ELECTRICAL SYSTEM OPERATION.

For normal operation of the electrical system, refer to Section II, Normal Procedures.

HYDRAULIC POWER SUPPLY SYSTEM.

Hydraulic power is supplied by two independent, parallel hydraulic systems designated as the primary and utility systems (figure 1-11). Both systems operate simultaneously to supply hydraulic power for the flight controls and wing sweep. If one or the other system should fail, either system is capable of supplying sufficient power for wing sweep and flight control operation. In addition to supplying wing sweep and flight control hydraulic power, the utility system also supplies power for operation of the landing gear, tail bumper, nose wheel steering, wheel brakes, speed brakes, flaps, air inlet control, weapons bay doors, trapeze, and emergency electrical generator. Hydraulic pressure for each system is supplied by two engine-driven, variable delivery pumps. To assure hydraulic pressure in the event of single engine failure, one pump in each system is driven by the right engine, and one pump in each system is driven by the left engine. Pressurized accumulators are installed in the system to supplement engine-driven pump delivery during transient hydraulic power requirements. Each system has a piston-type reservoir for hydraulic fluid storage that also acts as a surge damper for return line pressures. These reservoirs are pressurized with nitrogen to insure critical pump inlet pressure for all operating conditions. Hydraulic pressure of each system is displayed on the left console. Low pressure caution lights for each of the four pumps are displayed on the caution light panel. An automatic isolation valve reserves all utility power output for flight control and wing sweep operation by isolation of the other utility functions in the event of a primary system failure. Isolation of the utility functions is automatic upon loss of primary system pressure. Normal isolation, controlled by the aircraft commander, is a manual switch-selected operation to depressurize those functions of the utility hydraulic system used only during takeoff, landing, and ground operation.

HYDRAULIC PUMPS.

Four variable delivery pumps are employed. Normal power for the primary and utility systems is provided by two engine-driven pumps in each system. One pump in each system is driven by each engine. The pumps are each rated at 36 gpm, 5800 rpm, and 3100 (±50) psi.

HYDRAULIC ACCUMULATORS.

On airplanes (1)→(11) nine accumulators, three in the primary hydraulic system and six in the utility hydraulic system, are provided. Each system has two accumulators for the horizontal stabilizer and one for the autopilot damper servos. The wheel brake system has two accumulators, and, on airplanes (1)→(11), the utility system has one accumulator for the remaining utility system functions. Airplanes (12)→ have eight accumulators, three in the primary system and five in the utility system. See figure 1-50 for servicing data.

Primary Hydraulic Power Supply System

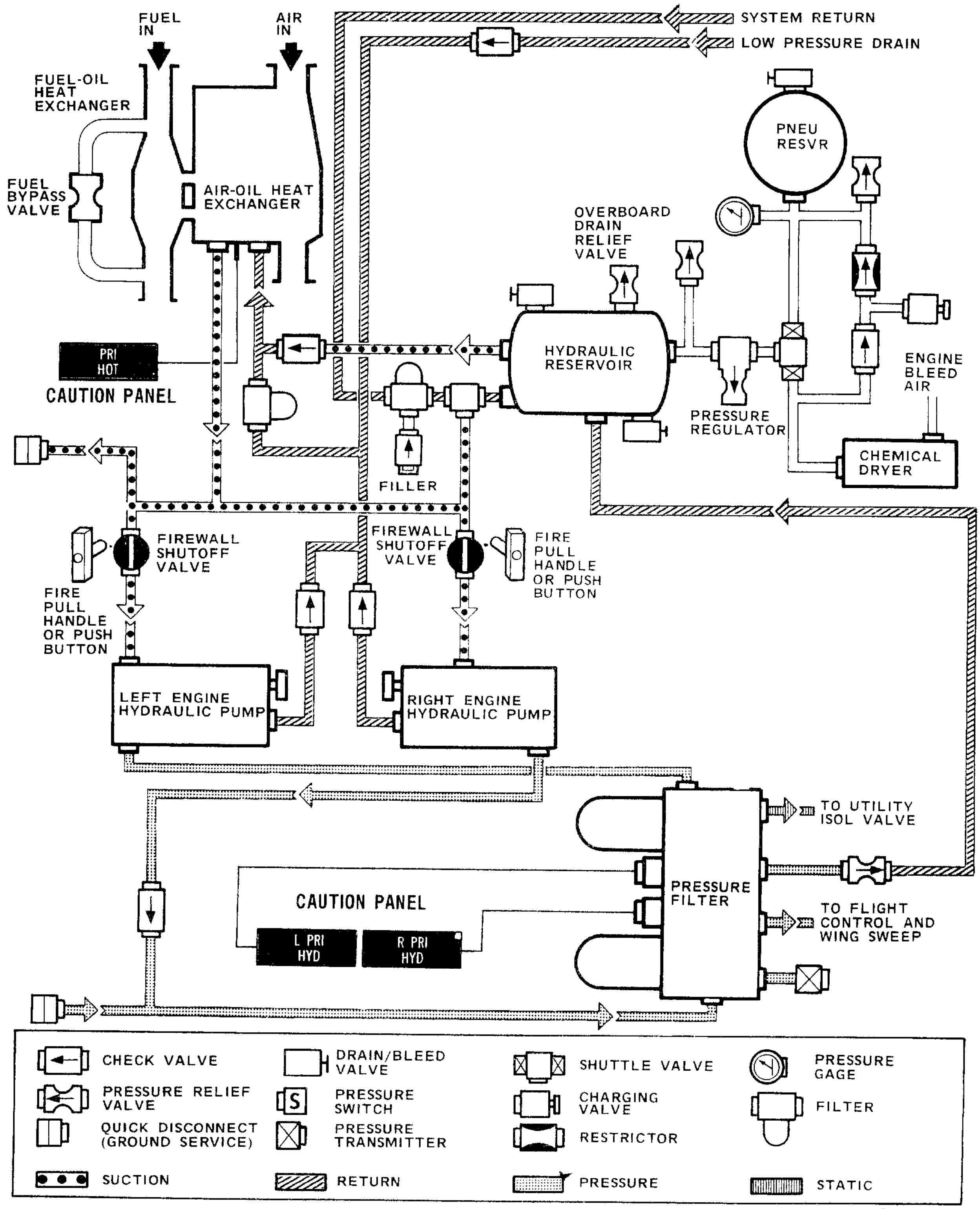

Figure 1-11. (Sheet 1 of 2)

Utility Hydraulic Power Supply System

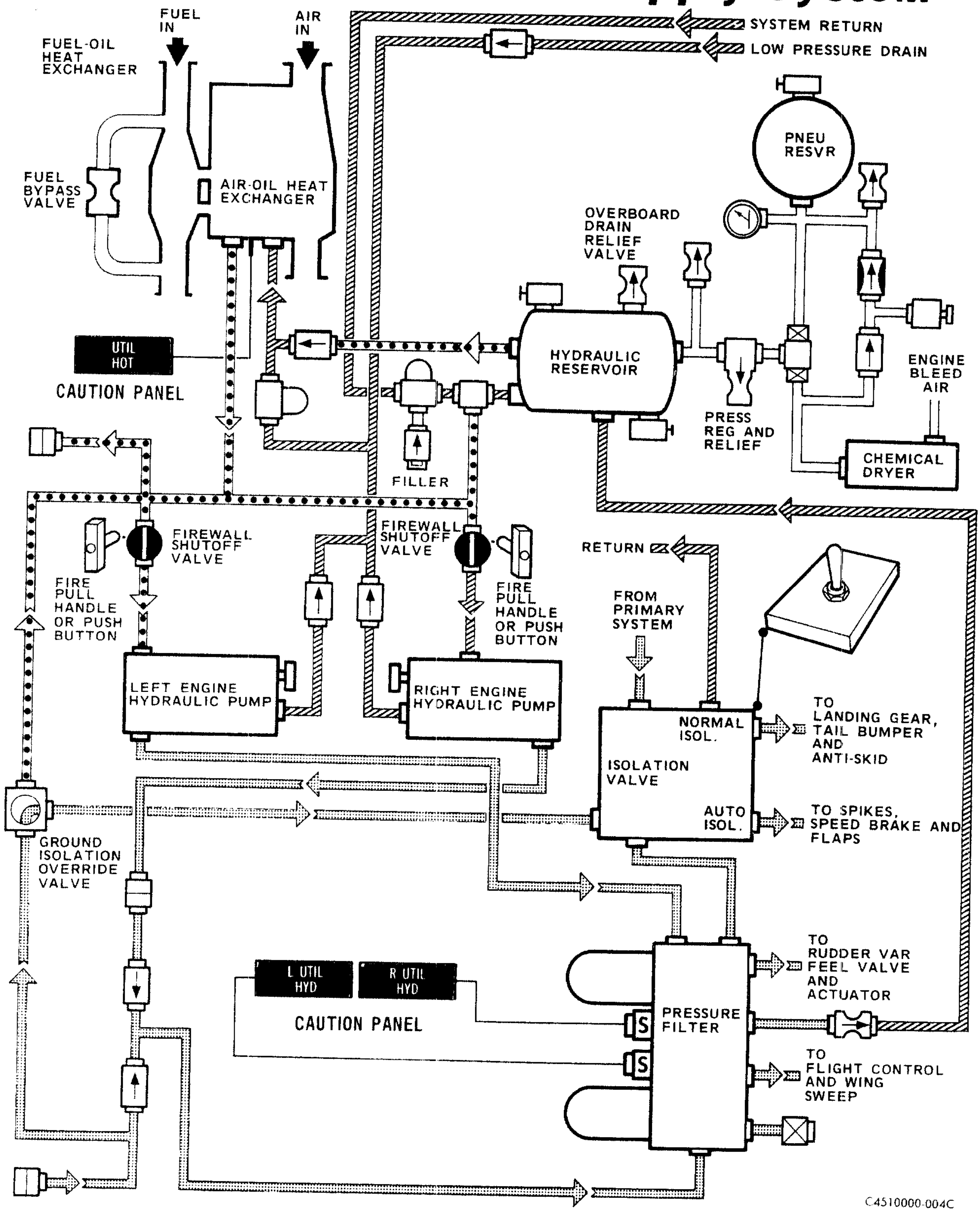

Figure 1-11. (Sheet 2 of 2)

HYDRAULIC FLUID RESERVOIRS.

Both primary and utility hydraulic reservoirs are floating piston, air-oil separated type using air pressure on one side of the piston to maintain hydraulic pressure on the other. Pneumatic pressure is supplied from pneumatic storage reservoirs located on the forward end of each hydraulic reservoir, and, as an alternate source, from the engine bleed air system. A pressure operated hydraulic relief valve prevents over pressurization by venting excess fluid overboard when reservoir pressure exceeds 135 psi. Steady-state fluid flow is passed through the reservoir to maintain reservoir warmth and to remove air from the fluid. During high flow rates, the fluid is bypassed around the reservoir and cooler loop directly to the pumps by means of the suction bypass valve. A 15-micron bypass type filter is located upstream of the reservoir. The reservoir also acts as a surge damper for return line impulse pressures. See figure 1-50 for servicing data.

HYDRAULIC COOLING SYSTEM.

Cooling is provided by an air-to-hydraulic heat exchanger and a fuel-to-hydraulic heat exchanger in each hydraulic system. The controls are arranged so that the cooling medium is air only at low speeds, fuel and air at intermediate speeds, and fuel only at high speeds.

HYDRAULIC ISOLATION VALVE.

The isolation valve is a safety feature to protect the utility hydraulic flight controls, pumps, and reservoir from fluid loss in the event of a hydraulic rupture in the remote parts of the system. This arrangement isolates the utility functions not essential for flight and thereby provides the protection equivalent to two independent primary flight control systems. Two modes are provided, automatic isolation and normal isolation. Automatic isolation occurs only during a malfunction in the primary system and is activated by loss of primary system pressure, without pilot choice, isolating all utility functions except flight control. Normal isolation, controlled by the utility hydraulic system isolation switch, is standard procedure during flight and depressurizes all utility functions used only during takeoff, landing, and ground operation. Both modes of operation, automatic and normal, are controlled by one valve. The group of hydraulic functions isolated by the automatic mode is larger and includes the group isolated by the normal mode. Isolated functions may still be operated by emergency pneumatic or emergency electric power where provided. No hydraulic interconnection between the primary and utility system is employed.

UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH.

The utility hydraulic system isolation switch (7, figure 1-12), with positions marked ISOLATE and ON, is located on the landing gear control panel. Positioning the switch to the ISOLATE position isolates the landing gear, tail bumper, nose wheel steering, and the wheel brakes from the utility hydraulic system. Positioning the switch to ON supplies utility hydraulic pressure to these systems. The switch is automatically positioned to ON when the landing gear handle is placed in the DN position.

HYDRAULIC PRESSURE INDICATORS.

Two 0-4000 psi pressure indicators (11 and 42, figure 1-4), one each for the primary and utility systems, are located on the left console. Pressure is measured mechanically and transmitted electrically by pressure transmitters in the system pressure lines.

LOW PRESSURE CAUTION LAMPS.

Four amber low pressure caution lamps (figure 1-21A), energized by pressure switches in each pump pressure line, are located on the main caution lamp panel. These lamps light when the individual pump output pressure falls below 500 (±100) psi. When lighted, the following letters will be visible in the respective lamp lense: L PRI HYD; L UTIL HYD; R PRI HYD; and R UTIL HYD.

HYDRAULIC FLUID OVERHEAT CAUTION LAMPS.

Two hydraulic fluid overheat caution lamps (figure 1-21A), one for each system, are located on the main caution lamp panel. A lamp lights when the hydraulic fluid temperature of the associated system exceeds 240±10°F (110±6°C). When lighted, the following letters will be visible in the respective lamp lense: PRI HOT; and UTIL HOT.

HYDRAULIC SYSTEM OPERATION.

For normal operation of the hydraulic system, refer to Section II, Normal Procedures.

PNEUMATIC POWER SUPPLY SYSTEMS.

There are four independent pneumatic power supply systems which provide pressure for emergency operation of the landing gear, spike system, weapon bay, trapeze, and for pressurization of the hydraulic reservoirs. Pressure for emergency extension of the landing gear is provided by a pneumatic reservoir located in the main landing gear wheel well. Each spike is provided with a separate pneumatic reservoir located in the main landing gear wheel well. Each trapeze is provided with a separate pneumatic reservoir located in the weapons bay. Two pneumatic reservoirs, one for each hydraulic system reservoir, provide pneumatic pressure for hydraulic system operation. For a functional description of each pneumatic system, refer to the associated system descriptions, this section. For servicing information on the pneumatic systems, see figure 1-50.

LANDING GEAR SYSTEM.

The landing gear is tricycle-type, forward retracting, and hydraulically operated. The main landing gear consists of a single common trunnion upon which two wheels are singly mounted. This arrangement of the main landing gear provides symmetrical main landing gear operation. Fusible metal, thermal pressure relief, plugs are incorporated in the main landing gear wheels to relieve tire pressure in the event of maximum performance braking. The nose landing gear has two dual-mounted wheels. The landing gear system is normally powered by the utility hydraulic system. A pneumatic system is provided as an alternate means of extending the gear in the event the normal system fails. The nose landing gear retracts into the nose wheel well, and the main landing gear retracts into a fuselage well. Procedures for normal operation of the landing gear system are contained within the appropriate portion of Section II, Normal Procedures.

MAIN GEAR.

Three hydraulic actuators are provided for operation of the main landing gear. A single-acting linear actuator retracts the main landing gear. Two double-acting linear actuators, one for an uplock and one for a downlock, are provided to lock the landing gear in the retracted or extended position. There are two main landing gear doors. The aft door is mechanically linked to the main landing gear and opens and closes with movement of the gear. The forward door, which also serves as the speed brake, is hydraulically operated. A mechanical connection between the main landing gear and the speed brake selector valve causes the main landing gear door to open and close in the proper sequence during landing gear operation. A ground safety switch, located on the lateral trunnion beam, prevents normal gear retraction while the airplane is on the ground.

NOSE GEAR.

Three hydraulic actuators are provided for operation of the nose landing gear and nose wheel well doors. A single-acting actuator retracts the nose landing gear. An uplock actuator locks the nose landing gear in the retracted position and also, through linkages, opens and closes the two nose wheel well doors. A downlock actuator locks the nose landing gear drag strut when the nose landing gear is extended.

LANDING GEAR CONTROLS AND INDICATORS.

Landing Gear Handle.

The landing gear handle (3, figure 1-12), located on the landing gear control panel, has two positions marked UP and DN. The handle has a gear unsafe warning lamp in the end. Moving the handle to the UP or DN position will cause the following actions to occur.

Gear Up

When the handle is moved to the UP position, an electrical signal actuates a solenoid-powered valve, sending hydraulic pressure to the nose gear downlock actuator, nose gear retract actuator, nose gear uplock door actuator, and the speed brake door actuator.

Landing Gear Control Panel

1. ARRESTING HOOK HANDLE
2. CONTROL SYSTEM SWITCH
3. LANDING GEAR HANDLE
4. LANDING GEAR HANDLE LOCK RELEASE BUTTON
5. LANDING GEAR WARNING HORN SILENCER BUTTON
6. AUXILIARY BRAKE HANDLE
7. UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH

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Figure 1-12.

The nose gear unlocks and retracts. When it is almost retracted, it mechanically triggers the nose gear uplatch which then locks the gear up and closes and locks the doors. The main gear forward door (speed brake) actuator extends the door. When the door is sufficiently open to allow the main gear to retract, a linkage from the door opens a valve which sends hydraulic pressure to the main gear downlock actuator, main gear retract actuator, and the uplock actuator. The gear then unlocks and retracts. When it is almost retracted, it mechanically triggers the uplatch which locks the gear up and also actuates a valve to close the speed brake door.

Gear Down

When the handle is moved to the DN position, an electrical signal actuates a solenoid-powered valve, sending hydraulic pressure to the nose gear uplock actuator, nose gear downlock actuator, and the speed brake door actuator. The nose gear uplock actuator unlocks and drives the nose gear doors open and locked, at which time the nose gear is allowed to free fall (extend) against the snubbing of its retract actuator. When the gear is almost extended, the downlock actuator drives it fully extended and locked. The speed brake door actuator opens the door until the door clears the main gear. A linkage then actuates a valve to pressurize the main gear uplock actuator and downlock actuator. The uplock opens, allowing the gear to free fall (extend) against the damping of its retract actuator. When the gear is extended, the downlock actuates. This causes the speed brake door actuator to position the door in the partially retracted (trail) position.

The landing gear handle is locked in the DN position by a spring loaded electrical solenoid when the weight of the airplane is on the landing gear. A landing gear safety switch controls 28 volt dc power to the solenoid. The weight of the airplane compresses the shock strut and opens the safety switch which breaks the circuit to the solenoid. When the solenoid is de-energized, the solenoid extends a mechanical lock holding the landing gear handle in the DN position. Removing the weight from the landing gear closes the safety switch on the landing gear and energizes the solenoid. The energized solenoid retracts the lock and frees the landing gear handle.

Landing Gear Handle Lock Release Button. The landing gear handle lock release button is located on the landing gear control panel (4, figure 1-12). The button must be depressed to release the landing gear handle from the up position to lower the gear. Normally, it is not necessary to depress the button when retracting the gear since the gear handle is locked in the down position by a solenoid which will release the handle as the weight of the airplane comes off the gear on takeoff. Should the solenoid malfunction, depressing the button will release the handle to allow gear retraction.

Note

> Any time it is necessary to depress the landing gear handle lock release button to move the handle to the UP position, the crew member should immediately suspect a malfunction of the landing gear ground safety switch. A failure of this switch, which left it in the closed position, would render ineffective the AUTO position of the fuel tank pressurization switch and cause all spoilers to remain armed even with the landing gear retracted. If a malfunction of the landing gear safety switch is suspected, the fuel tank pressurization switch should be placed to PRESSURIZE and the spoiler switch to OFF.

Landing Gear Alternate Release Handle.

The landing gear alternate release handle (9, figure 1-21), located on the right main instrument panel, is provided to extend the landing gear in the event the normal hydraulic system fails. When the handle is pulled pneumatic pressure is directed to simultaneously open the speed brake door and unlock the nose and main gear uplocks. The gear will free fall to the extended position, then pneumatic pressure will actuate the nose and main gear downlocks and retract the speed brake door to the trail position. Once the gear has been extended by the alternate method it cannot be retracted. On airplanes 1 → 11 the speed brake door may fail to retract to the trail position. This will be indicated by the landing gear handle warning lamp remaining on after the gear is extended and locked. Should this occur, pushing the handle back in will relieve the pressure in the system and allow the air load to push the speed brake door to the trail position.

CAUTION

- As the airplane slows down after landing, the weight of the door and lack of air load will cause the door to extend and drag the ground. Stopping the airplane as soon as possible will prevent extensive damage to the door.
- Any time this handle is pulled, it must be pushed back in before removing electrical power from the airplane. Otherwise, pressure in the speed brake door actuator will extend the door causing damage from ground contact.

Landing Gear Warning Horn.

The landing gear warning horn provides an audible signal in the crew members' headsets when an unsafe landing gear condition exists. The horn sounds when all of the following conditions exist: The nose and main landing gear are not down and locked and/or speed brake door is not in trail position, indicated airspeed is below 160 (±12) knots, airplane altitude is less than 10,000 (±350) feet, and one or both throttles are set below minimum cruise setting. The malfunction and indicator lamp test button located on the lighting control panel may be used to test the landing gear warning horn. The warning horn may be silenced by depressing the horn silencer button adjacent to the landing gear handle (5, figure 1-12).

Landing Gear Position Indicator Lamps.

A planform silhouette of the airplane having two green indicator lamps is located on the left main instrument panel (8, figure 1-5). The lamps are positioned to represent the nose and main landing gear. When the landing gear is down and locked, the lamps are lighted. In-transit positions of the landing gear and unsafe landing gear conditions are indicated by lighting of the red warning lamp in the landing gear handle knob. A safe up-and-locked landing gear condition is indicated when both the green indicator lamps and the red warning lamp are off.

TAIL BUMPER SYSTEM.

The tail bumper protects the control surfaces, engines, and portions of the airframe from damage that might occur if the tail inadvertently contacts the ground during ground handling. The tail bumper also provides limited protection during overrotation on take-off and during landings. In flight, the tail bumper is held in the fully retracted position by hydraulic pressure in the tail bumper lift cylinder. The hydraulic pressure is ported to the tail bumper lift cylinder from the speed brake control valve. When the landing gear is extended and the speed brake returns to trail position, the lift cylinder pressure is relieved and the tail bumper is extended by the pneumatic action of the tail bumper dashpot. The dashpot, which functions as the impact shock absorber, has its own separate reservoir that is charged with compressed nitrogen. Retraction of the landing gear allows hydraulic pressure to again be ported to the tail bumper lift cylinder to retract the bumper and hold it in this position.

Note

On some airplanes, the tail bumper will extend when the speed brake is extended. This is caused by a slight pressure drop in the tail bumper lift cylinder when the speed brake is actuated and does not indicate a malfunction.

NOSE WHEEL STEERING SYSTEM.

The nose wheel steering system provides directional control of the airplane for taxiing and during takeoff and landing. The system is electrically engaged, hydraulically actuated and controlled by the rudder pedals. Hydraulic pressure is supplied from the utility hydraulic system. When the system is energized, movement of the rudder pedals at either crew station is mechanically transmitted through a system of push-pull rods to a hydraulic steering actuator which turns the nose wheels. A mechanical linkage at the actuator provides a nonlinear increase in steering angle, as the pedals are displaced, to prevent overcontrolling. Maximum deflection of the rudder pedals steers the nose wheels 30 degrees either side of center on airplanes 1→11 (40 degrees on airplanes 12→). The steering system automatically centers the nose gear during retraction. Nose wheel shimmy damping is accomplished by restricting the displacement of hydraulic fluid in the steering actuator. The flight control system switch must be in the T.O. & LAND position, or the rudder authority switch must be in the FULL position to permit sufficient rudder pedal travel for steering. Power for engaging the system is furnished from the 28 volt dc essential

bus. Procedures for normal operation of the nose wheel steering system are contained in the appropriate portions of Section II.

Note

Nose wheel steering will be inoperative if the landing gear is extended using the landing gear alternate release handle.

NOSE WHEEL STEERING/AIR REFUEL BUTTONS.

A nose wheel steering/air refuel button (4, figure 1-15), is located on each control stick grip. The buttons are labeled NWS and A/R DISC. With the weight of the airplane on the gear, depressing either button actuates a holding relay to engage the system. The button can then be released and the system will remain engaged until the button is again depressed and released to open the relay and disengage the system. When the system is disengaged, the nose wheels will still react in response to rudder pedal displacement but at a reduced rate and with a reduced steering force. With the system disengaged there will not be sufficient steering force to hold the nose wheels in position with large side loads applied. The button receives 28 volt dc power from the essential bus. For a description of the A/R DISC function of the buttons, refer to "Fuel Supply System", this section.

NOSE WHEEL STEERING/AIR REFUELING INDICATOR LAMP.

A green nose wheel steering/air refueling indicator lamp labeled NWS/A/R is located on the left warning and caution lamp panel (1, figure 1-5). The lamp will light when the nose wheel steering system is energized. On airplanes 1→11 the intensity of the lamp can be controlled with the malfunction and indicator lamp dimming switch. On airplanes 12→ the lamp cannot be dimmed. For a description of the A/R DISC function of the lamp, refer to "Fuel Supply System", this section. The lamp receives power from the 28 volt dc essential bus.

BRAKE SYSTEM.

Each main landing gear wheel is equipped with a hydraulically operated multiple disc brake. Pressure for operation of the brakes is supplied by the utility hydraulic system for normal operation and by two hydraulic accumulators in the event of utility hydraulic system failure. Anti-skid control, automatic braking during landing gear retraction, and an auxiliary brake are provided. Normal brake operation is controlled by conventional brake pedals, each mechanically connected to brake metering valves. The brake hydraulic system is a dual-normal type, separated into two circuits. Each circuit operates independently of the other. One circuit operates one half of the pressure pistons on the left brake and one half the pressure pistons on the right brake. The other circuit operates the other half of the pistons on each brake. During normal operation of the brakes, pressure is metered to the brakes from both hydraulic circuits in proportion to applied force on the brake pedals. Full braking effectiveness is achieved with approximately 60 percent of full brake pedal travel. If one hydraulic circuit becomes inoperative, the brake system can provide sufficient increased pressure to the operative circuit for 90 percent of normal braking effectiveness. This is accomplished by application of greater than normal brake pedal travel and slightly higher pedal force. The dual-normal type brake hydraulic system provides emergency brake operation automatically; therefore, actuation of an emergency brake control handle is not required. Two hydraulic accumulators are provided in the system to supply brake system pressure in the event of failure of the utility hydraulic system. Each accumulator is precharged and supplies pressure to only one of the individual brake circuits. Fully charged accumulators will provide 10-14 full-pressure brake applications or one full-pressure brake application with 32 anti-skid cycles. A priority valve, which limits the quantity of fluid which can be displaced from the brake accumulator through the brake metering valves by actuating the brake pedals, is included in each hydraulic circuit. If the brake accumulators are not replenished as fluid is displaced by repetitive brake applications or by anti-skid cycling, the priority valves will close when accumulator pressure has been reduced to approximately 1000 psi.

CAUTION

Do not inadvertently actuate the brake pedals inflight. When utility hydraulic pressure is isolated from the brake system there is no way to replenish the brake accumulators. If the utility hydraulic system fails after the brake accumulators are bled off to below 1000 psi there will be no braking available with the brake pedals on landing.

When accumulator pressure is 1000 psi, sufficient fluid volume for 5-10 auxiliary brake applications is remaining. After the priority valves close, the remaining fluid can be utilized only by pulling the auxiliary brake handle. No braking action can be achieved by actuating the brake pedals. Procedures for normal operation of the brake system are contained in the appropriate portions of Section II.

ANTI-SKID SYSTEM.

Anti-skid control is provided for normal braking. Solenoid operated valves in each brake and anti-skid control valve assembly function to release brake pressure in response to electrical signals received from the anti-skid control system as impending wheel skids are detected. The solenoid valves will reapply brake pressure upon being de-energized after the wheel returns to normal speed.

Anti-Skid Control Switch.

The anti-skid control switch (2, figure 1-5), is located on the aircraft commander's throttle panel and labeled ANTI-SKID. The switch has two positions, one marked OFF and an unmarked ON (up) position. Placing the switch to ON will provide anti-skid control during normal braking. With the switch in OFF, anti-skid control will not be available and brake pressure will be in direct response to pedal pressure.

Anti-Skid Caution Lamp.

An amber caution lamp labeled ANTI-SKID is located on the main caution lamp panel (14, figure 1-5). The lamp will light when the anti-skid switch is in ANTI-SKID and a malfunction has caused the anti-skid system to become deenergized. When the lamp is lighted, anti-skid control is not available and braking will be in direct response to pedal pressure.

AUXILIARY BRAKE HANDLE.

The auxiliary brake handle (6, figure 1-12), labeled AUX BRAKE, is located on the landing gear control panel. When the handle is pulled out, a mechanical linkage opens a selector valve which admits pressure from the hydraulic accumulators directly into the brake lines downstream of the brake control valve. The primary function of the auxiliary brake control handle is to apply the brakes while the airplane is parked. The auxiliary brake control can be used to set the brakes for engine run-up. A secondary function of the auxiliary brake control is to serve as a supplemental emergency brake in the event that accumulator pressure is reduced sufficiently to cause the priority valves to close and prevent normal brake application by pedal actuation. Brake pressure cannot be metered by the auxiliary brake handle. The total accumulator pressure is ported directly to the brake cylinders, bypassing the metering valves and the anti-skid valves. Therefore, the auxiliary brake handle should not be pulled while the airplane is in motion except when braking cannot be achieved by pedal actuation.

CAUTION

Pulling the auxiliary brake handle while the airplane is moving will cause the wheels to lock and result in tire skidding or blowout.

BRAKE HYDRAULIC HANDPUMP.

A hydraulic handpump, located in the main landing gear wheel well, is provided to replenish brake accumulator pressure during ground handling operation.

AIRCRAFT ARRESTING SYSTEM.

The arresting hook system provides for emergency arrestment of the airplane. The system consists of an arresting hook, arresting hook dashpot, a dashpot air bottle, an uplock latch, arresting hook controls, a pressure gage, and an air filler valve. Except for the controls the arresting hook components are located in the lower aft end of the fuselage tail cone.

ARRESTING HOOK HANDLE.

The arresting hook handle (1, figure 1-12), located adjacent to the left main instrument panel, is connected to a low friction push-pull type mechanism contained in a flexible metal housing. The handle is labeled HOOK on diagonal stripes. The mechanism provides a direct mechanical linkage from the handle to the arresting hook uplatch mechanism in the tail cone. The arresting hook is released by grasping the handle and pulling aft. The total travel of the handle from retract to extend position is approximately four inches. Approximately one second is required for the arresting hook to extend. The hook must be raised manually to its stowed position.

ARRESTING HOOK CAUTION LAMP.

The amber arresting hook caution lamp, labeled HOOK DOWN, is located on the main caution lamp panel (14, figure 1-5). The caution lamp lights to indicate hook down position only.

AERODYNAMIC DECELERATION EQUIPMENT.

SPEED BRAKE.

The speed brake, which also serves as the main landing gear forward door, is provided as an aid to deceleration during flight. The speed brake is hydraulically operated and may be used as a speed brake only when the landing gear is up and locked. For operation of the speed brake as a landing gear door refer to "Landing Gear System" this section.

Speed Brake Switches.

A three-position speed brake switch (18, figure 1-4 and 6, figure 1-17), marked IN, OFF, and OUT, is located on the right throttle at each crew station. The switches are thumb actuated and slide forward (IN) and aft (OUT). The aircraft commander's switch is detented in all positions. The pilot's switch is spring loaded to OFF from both the IN and OUT positions and will override the aircraft commander's switch.

Gun Selector Switch.

The gun selector switch (4, figure 1-5), located on the left main instrument panel, is labeled GUNS and has three positions marked PYLONS, BAY, and OFF. With the switch in the OFF position the weapon bay gun and gun pods cannot be fired. Placing the switch to PYLONS enables firing the gun pods mounted on the pivot pylons. Placing the switch to BAY enables firing the weapons bay gun.

Rounds Counter.

The rounds counter (5A, figure 1-5), located on the left main instrument panel, provides an indication of the amount of ammunition remaining in the weapon bay gun. The counter is graduated from 0 to 20, times 100, in increments of 100. Ammunition counters are not provided for the pylon guns.

Gun Trigger Switch.

Two gun trigger switches (5, figure 1-15), one located on each control stick grip, are provided to fire the guns. Depressing either switch will fire either all pylon gun pods or the weapon bay gun depending on the position of the gun selector switch.

Operation of the Weapon Bay Gun or Pylon Gun Pods.

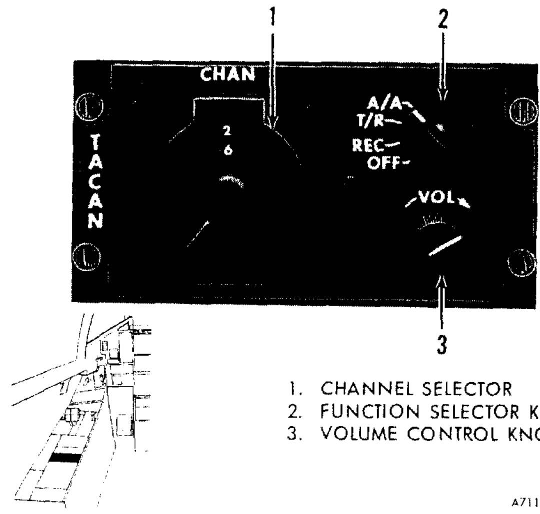
Except under actual combat conditions the guns will not be fired unless over a cleared gunnery range.

1. Master power switch - ON.
2. Gun selector switch - As required.
3. LCOS mode selector knob - GUN-AA or GUN-AG. (as applicable)
4. LCOS range set knob - Set range.
5. LCOS true airspeed knob - Set TAS.
6. LCOS aiming reticle brightness knob - Set as desired.
7. Center pipper on the target.
8. Gun trigger switch - Depress when in range.

TACTICAL AIR NAVIGATION SYSTEM (AN/ARN-52).

The tactical air navigation system (TACAN) enables the airplane to receive continuous indications of its distance and bearing from any selected TACAN station located within a line-of-sight distance of approximately 300 nautical miles. There are 126 channels available for selection. The equipment con-

Tacan Control Panel



1. CHANNEL SELECTOR
2. FUNCTION SELECTOR KNOB
3. VOLUME CONTROL KNOB

A7111200-002

Figure 1-25.

sists of the TACAN receiver-transmitter and its control panel. Two antennas, one on top of the fuselage and the other beneath the fuselage (figure 1-40), function to keep the TACAN receiver locked on to the antenna receiving a usable signal. The TACAN equipment also has an air-to-air mode and can be used between two aircraft having TACAN with air-to-air capability for range information only. The TACAN works in conjunction with the instrument system coupler, the bearing distance heading indicator, the lead computing optical sight, the horizontal situation indicator, the attitude director indicator, and through the interphone control panel for audio output. The system operates on 28 vdc from the main dc bus and 115 vac from the left main ac bus. The TACAN control panel (figure 1-25) is located on the left console.

TACAN FUNCTION SELECTOR KNOB.

The function selector knob (2, figure 1-25), located on the TACAN control panel, has four positions marked OFF, REC, T/R, and A/A. In the OFF position, electrical power to the TACAN system is off. In any of the other three positions, electrical power is supplied and the TACAN set is on. In the REC position, the set will receive bearing and audio identity signals only. In REC position, range information will not be displayed because the TACAN transmitter is not on. In the T/R position, both the receiver and the transmitter are operative, the system will receive and display both range and bearing of the station being interrogated, and audio identity signals are fed into the interphone system. In the A/A (air-to-air) position, the set will transmit and receive to and

WING SLATS. (12)→

Each wing is equipped with a leading edge slat. Each slat is divided into four sections which are connected and operate as one unit. The slats operate in conjunction with the main flaps and are connected to the main flap drive assembly by flexible drive shafts. On the extend cycle, the slats will extend to the full down position before the main flaps start to extend. On the retract cycle, the flaps will fully retract before the slats start to retract. Asymmetrical slat travel is prevented by an asymmetry device which when sensing asymmetrical slat travel will close the main flap drive control valve. Once the flap drive control valve has closed, the flaps and slats cannot be extended or retracted by either the normal or the emergency mode.

ROTATING GLOVES. (12)→

The outboard edges of the wing gloves, adjacent to the wing inboard leading edges are equipped with movable surfaces to allow full forward movement of the inboard slats. These surfaces are called rotating gloves. A door forms the lower surface of each rotating glove. Each rotating glove and its associated door are operated by a mechanical actuator and linkage which is connected to the slat drive flexible shaft. When the slats are extended, the rotating gloves automatically rotate (leading edge down and trailing edge up) and the doors open to allow full extension of the slats.

Flap and Slat Handle. (1)→(11)

The flap and slat handle (17, figure 1-4), located on the left console, has three positions marked UP, LOITER, and FULL DOWN. When the handle is positioned to FULL DOWN, a mechanical linkage opens the flap drive control valve, directing hydraulic pressure to the flap drive assembly to extend the flaps. Positioning the flap handle to extend the flaps more than 15 degrees will also close a contact to provide electrical power to the auxiliary flap actuators and the slat drive actuators. In the FULL DOWN position, both main and auxiliary flaps extend to 30 degrees down. A manually operated gate is provided to stop flap and slat handle travel at a position which provides main flap full aft (horiz) travel and zero deflection. The gate must be manually released to move the handle up or down. During the flap extension, the flaps do not extend beyond 15 degrees until the slats have extended 70 percent of travel. When the handle is placed in the UP position, the main and auxiliary flaps fully retract. Normal extension of the flaps and slats takes approximately 15 seconds and retraction takes approximately 12 seconds.

Flap and Slat Handle. (12)→

The flap and slat handle (38B, figure 1-4), located on the left console, has three positions marked UP, SLAT DOWN, and FLAP DOWN. A manually operated gate (38A, figure 1-4), located between the SLAT DOWN and FLAP DOWN areas, must be released before the handle can be moved from one area to another. When the handle is moved from UP to any position in the SLAT DOWN area, a mechanical linkage opens the flap drive control valve, directing hydraulic pressure to the flap drive motor. The flap drive assembly rotates the flexible shafts connected to the slat drive mechanism to position the rotating glove and to extend the slats to a position corresponding to handle position. Moving the handle down to the gate will cause the slats to fully extend. When the gate is released and the handle is moved into the FLAP DOWN area, the flap drive assembly will rotate the flexible shafts connected to the main flap actuators, extending the main flaps to a position corresponding to handle position. The flap and slat drive assembly is so designed that it will not extend the flaps until the slats are fully extended. When the handle is moved down to a position corresponding to 28 degrees or more of flaps, a contact closes providing electrical power to the auxiliary flap actuators. Full down position of the flap and slat handle will provide 37.5 degrees of flap deflection. The retraction cycle sequence is just the opposite from the extension cycle. Moving the handle from the full FLAP DOWN position to the full UP position will first cause the flaps to retract and then the slats to retract. It should be noted that at no time will the flaps extend until the slats are fully extended nor will the slats retract until the flaps are fully retracted regardless of flap and slat handle position. Normal extension or retraction of the flaps and slats takes approximately 12 seconds.

Flap and Slat Switch.

The flap and slat switch (24, figure 1-17), located on the center console, has two positions marked EMER and NORM. On aircraft (1)→(11), when the switch is in EMER, either the flaps or slats may be electrically extended or retracted to the desired position by holding the emergency flap or emergency slat switch to RETRACT or EXTEND as applicable. On aircraft (12)→, when the switch is in EMER, the flaps and slats may be extended or retracted electrically by holding the emergency flap and slat switch to EXTEND or RETRACT as applicable. On all aircraft, when the switch is in EMER, the flap drive control valve is closed, disabling the flap drive motor. When the flap and slat switch is in the NORM position, the flaps and slats are actuated normally by use of the flap handle. The EMER position is used in the event of utility hydraulic system failure.

Emergency Flap Switch. (1)→(11)

The emergency flap switch (13, figure 1-17), is a three-position switch located on the center console. The switch has two positions marked EXTEND and RETRACT and is spring loaded to an unmarked center OFF position. With the flap and slat switch positioned to EMER, holding the emergency flap switch to either RETRACT or EXTEND applies electrical power to the emergency electrical flap motor which

Wing Sweep Hydraulic System (Typical)

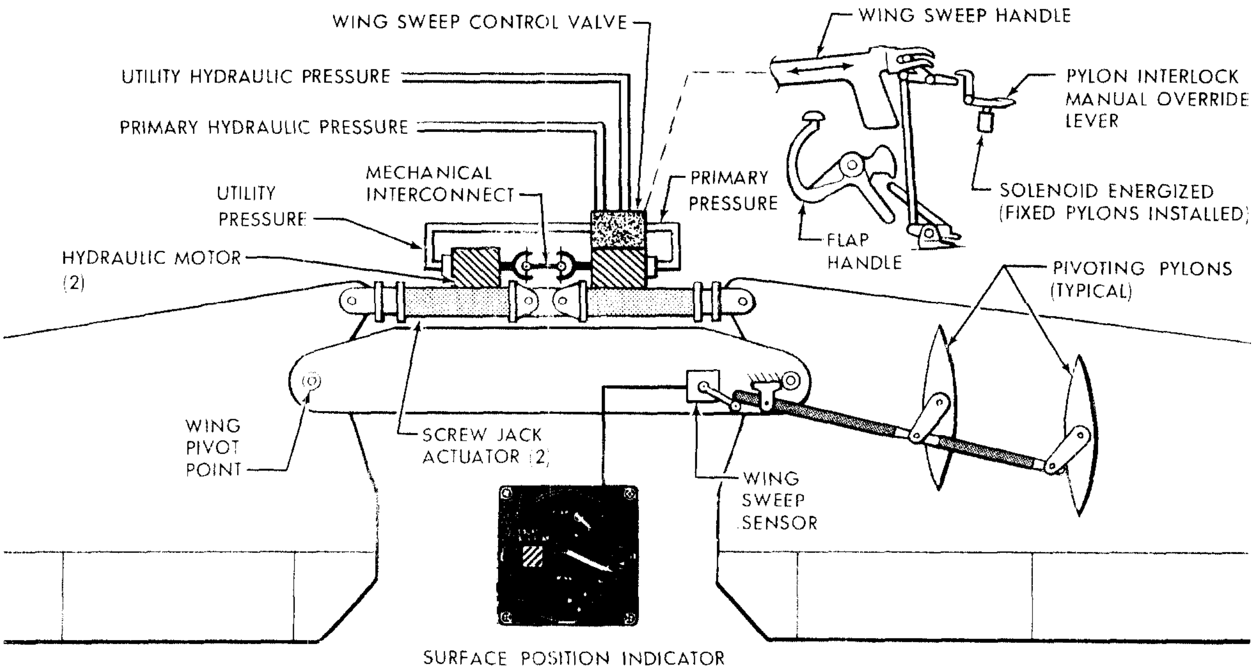


Figure 1-13.

drives the main flaps to the selected position. The slats must be extended to 70 percent of travel before the flaps will extend beyond 15 degrees. An electrical interlock prevents flap extension when the wings are swept past 26-1/2 degrees. It should be noted that the emergency flap switch does not control the auxiliary flaps since the auxiliary flap actuators are energized only when the flap and slat handle is positioned to move more than 15 degrees. Emergency flap extension takes approximately 150 seconds and retraction takes approximately 120 seconds.

Emergency Slat Switch. (1)→(11)

The three-position emergency slat switch (12, figure 1-17) is located on the center console. The switch provides an alternate source of 28 volt dc power to operate the contactors which apply 115 volt ac power to the slat drive motors. The switch has positions marked RETRACT and EXTEND and is spring loaded to an unmarked center OFF position. The switch is used when normal operation of the flap handle fails to extend or retract the slats. With the flap and slat switch positioned to EMER, placing the emergency slat switch to EXTEND will extend the slats. The slats must be extended beyond 70% travel before the flaps will extend beyond 15 degrees.

Emergency Flap and Slat Switch. (12)→

The emergency flap and slat switch (12 and 13, figure 1-17), located on the center console, has positions marked EXTEND and RETRACT and is spring loaded to the center unmarked OFF position. The switch is provided as an emergency method of operating the main flaps and slats in the event of a utility hydraulic system failure. Operation of the flaps and slats using this switch is identical to that when using the flap and slat handle except that electric power is used to operate the flap drive motor instead of hydraulic power. It should be noted that the emergency flap and slat switch does not control the auxiliary flaps since the auxiliary flap actuators are energized only when the flap and slat handle is positioned to more than 28 degrees. Emergency flap extension or retraction takes approximately 60 seconds.

Flap and Slat Position Indicators.

The flap and slat position indicators are a part of the surface position indicator (22, figure 1-5), located on the left main instrument panel. The indicators display main flap position in degrees and slat and auxiliary flap position in a window as either UP or DN (down). When the slats or auxiliary flaps are in transit or when electrical power is turned off a crosshatch is displayed in the indicator window.

WING SWEEP SYSTEM.

The variable sweep wings are moved to and held in position by two hydraulic, motor-driven, linear actuators. The actuators are mechanically interconnected to insure positive synchronization (figure 1-13). The right actuator is furnished power by the primary hydraulic system, and the left actuator is furnished power by the utility hydraulic system. In the event of failure of either hydraulic system, the remaining system, by utilizing the load transfer capability of the mechanical interconnect will still provide wing actuation. However, actuation under this condition will be at a reduced rate commensurate with actuator loading. Wing position is controlled by a closed loop mechanical servo system in response to an input signal from the wing sweep handle. The maximum rate at which the wings extend or retract is controlled by flow-limiting devices in the hydraulic lines. Directional reversal, due to aerodynamic loads, is prevented by the non-reversing (Acme-type) threads in the actuator. The wing sweep handle is locked in the 16 degree position by a solenoid operated latch whenever the auxiliary flaps are out of the zero position. Also, a mechanical interlock prevents the wing sweep handle from being moved past the 26-1/2 degree position when either the flap and slat handle is out of the UP position or the main flaps are out of the fully retracted position. If aircraft (1)→(11) are equipped with fixed pylons, an interlock solenoid is de-energized, preventing the wing sweep handle from being moved past 26-1/2 degrees while the pylons are installed.

WING SWEEP CONTROL HANDLE.

The wing sweep control handle (5, figure 1-13A), is shaped like a pistol grip and is spring loaded to a stowed position under the canopy sill to the left of the crew module. Teeth in the top of the handle lock it to serrations in the handle support, when it is stowed, to prevent inadvertent movement. To adjust wing sweep, the handle must be rotated to the vertical position to unlock it; then it can be moved forward or aft as necessary. The handle is mechanically linked to the wing sweep control valve. On aircraft (1)→(11), the handle is pushed forward to sweep the wings aft and pulled aft to sweep the wings forward. On aircraft (12)→, the handle is pulled aft to sweep the wings aft and pushed forward to sweep the wings forward.

MANUAL OVERRIDE LEVER. (1)→(11)

A manual override lever, marked WS OVRD, is provided directly aft of the wing sweep handle to permit the wings to be swept past 26-1/2 degrees when the fixed external pylons are installed. For ground maintenance or in case of emergency, the pylon interlock can be overridden by depressing the lever. When the fixed pylons are jettisoned, the circuit to the pylon interlock solenoid is closed, releasing the wing sweep handle.

WING SWEEP HANDLE LOCKOUT CONTROLS. (12)→

Two wing sweep handle lockout controls (6, figure 1-13A), one labeled FIXED STORES and the other labeled WEAPONS, are located just above and aft of the wing sweep control handle. When either control is moved forward, the word ON is visible, and a latch extends which prevents aft movement of the wing sweep handle past the latch. When either control is moved aft, the word OFF is visible and the latch retracts. The fixed stores lockout control, when ON, prevents the wing sweep handle from being moved aft past the 26 degree position. This is the sweep angle at which the fixed pylons and stores are in a streamlined configuration. The weapons lockout control restricts aft movement of the wing sweep handle to 55 degrees. This is the wing sweep angle past which, certain weapons on the inboard pivot pylons would strike the fuselage. The wing sweep handle lockout controls restrict aft movement of the wing sweep handle only. Forward motion is unrestricted.

WING SWEEP HANDLE 26 DEGREE FORWARD GATE. (12)→

A wing sweep handle 26 degree forward gate (4, figure 1-13A), located above the wing sweep handle, is provided to stop forward motion of the wing sweep handle at 26 degrees. The gate is thumb-actuated and is spring loaded to the latched position. Depressing the gate will retract a latch, allowing the wing sweep handle to be moved forward past the 26 degree position.

WING SWEEP POSITION INDICATOR.

The wing sweep position indicator (22, figure 1-5), is a part of the surface position indicator located on the left main instrument panel. The indicator displays the wing position in degrees and is graduated in 2 degree increments from 16 to 72 degrees. The angle of wing sweep is monitored by a transmitter which mechanically follows the change in wing position and converts this information to an electrical signal which drives the wing sweep indicator.

FLIGHT CONTROL SYSTEM.

The flight control system (figure 1-14), provides control of the airplane by movement of the primary control surfaces. The primary control surfaces consist of a pair of movable horizontal stabilizers, rudder, and spoilers. Movement of the control surfaces is controlled by the control stick and rudder pedals. Rate gyros and accelerometers, in conjunction with electronic computers and damper servoactuators provide continuous automatic damping about the three axes of the airplane. Separate channels of mechanical linkage control hydraulic servo actuators, which produce control surface movement. Yaw control of the airplane is accomplished by deflection of a rudder surface located on the trailing edge of the vertical stabilizer. Pitch attitude of the airplane is controlled by symmetrical deflection of the horizontal stabilizer surfaces. Roll attitude is controlled by asymmetrical deflection of the horizontal stabilizer surfaces. When the wing sweep angle is less than 45 degrees, roll control is aided by action of two spoilers on top of each wing. The stability augmentation system employs use of triple-redundant sensors, electronic circuitry and electro-hydraulic dampers. Automatic failure detection and correction, as well as self-test features, are provided in the system. The flight control system functions in conjunction with the terrain following radar (TFR) to provide the capability of automatic terrain following. Signals from the TFR, control the pitch axis of the flight control system through the pitch damper to maintain the airplane at a preselected altitude above the terrain. When operating in either automatic or manual terrain following, a failure in the TFR will generate a 2g pull up signal to the pitch damper. For description of the TFR refer to "Terrain Following Radar," this section. Procedures for normal operation of the flight control system are contained in the appropriate portions of Section II.

PITCH CHANNEL.

Manual control of the aircraft in pitch is achieved by fore and aft movement of the control stick. This movement is transmitted along the pitch channel push-pull tubes and bellcranks to the left and right horizontal stabilizer actuator control valves. These control valves control the flow of hydraulic fluid to the actuators, thus causing the horizontal stabilizers to move symmetrically. Stick throw is limited by mechanical stops to prevent interference with the seat and control panel knobs. With the series trim actuator at zero, these stops allow the pilot to command 25 degrees trailing edge up and 10 degrees trailing edge down elevator motion of the horizontal stabilizer. The pitch command input limits are set at the input to the pitch-roll mixer which limit the sum of manual inputs, series trim inputs, and damper servo inputs to 25 degrees trailing edge up and 10 degrees trailing edge down motion of the horizontal tail. Since airloads on the surfaces are not transmitted back to the controls, artificial feel has been incorporated into the system to give the pilot the desired feel. The artificial feel is provided by a spring in parallel with the mechanical linkage. With the pitch damper off, total stick travel from neutral to full aft is 7 inches and from neutral to full forward is 2.8 inches. The force required to move the stick from neutral to full aft, ranges from the initial breakout force of 1.7 pounds to a force of 78 pounds. The force required to move the stick from neutral to full forward, ranges from 1.7 pounds to 34 pounds.

ROLL CHANNEL.

Lateral movement of the control stick is transmitted along the roll channel push-pull tubes and bellcranks to the horizontal stabilizer actuator control valves. This motion operates the horizontal stabilizer actuators in opposite directions, causing an asymmetrical movement of the horizontal stabilizers. Stick displacement is limited by stick stops so that stick command is limited to ±8 degrees of differential horizontal stabilizer motion. Stick travel from neutral to hardover is 4.8 inches. The stick forces and surface displacement in the roll channel are nonlinear with respect to stick travel. Differential horizontal stabilizer command, which consists of damper servo commands and stick commands, is limited to ±8 degrees of differential motion by mixer stops in the pitch-roll mixer. As the stick is moved laterally from neutral, the stick forces increase from a breakout force of 1.3 pounds to a force of 15 pounds at one half the total stick travel (2.4 inches). At this point a force detent is encountered and the total command of the horizontal stabilizer is 2 degrees of differential horizontal stabilizer motion. Force to the stick must be increased from 15 pounds to 23 pounds to pass the force detent. From this point to hardover, the stick force increase is linear to the maximum of 31 pounds. When the wings are forward of 45 degrees, roll control is aided by action of two spoilers on the top of each wing. Each spoiler surface is actuated by a hydraulic servo actuator. The outboard pair of spoiler actuators are supplied pressure by the utility hydraulic system. The inboard pair of spoiler actuators are supplied pressure by the primary hydraulic system. The actuators receive their command signals from the stick position transducers located in the roll channel linkage. Lateral movement of the control stick causes the stick position transducers to generate command signals which are sent through the wing sweep sensor assembly to the spoiler actuators. Both commanded spoilers extend to a maximum of 45 degrees in response to one half lateral stick displacement. The spoiler extension is in non-linear proportion to stick displacement. The spoilers are operable only when the wing sweep angle is between 16 degrees and 45 degrees. When the wing sweep angle is between 45 and 47 degrees, the spoiler command signals are zeroed and the spoilers are locked down. Between 47 and 49 degrees, the hydraulic supply to the spoilers is cut off. A spoiler monitor is provided in the roll channel to deactivate or lock down either pair of spoilers should one of that pair of spoilers malfunction. If a spoiler inadvertently extends without being commanded and the aircraft starts a roll, the pilot would apply an opposite stick command to maintain wings level. Simultaneous extension of spoilers on each wing will cause the monitor, through a voting process, to cut off hydraulic pressure to the malfunctioning spoiler and its mate on the opposite wing. This action will retract and lock the pair of spoilers in the down position and cause the spoiler caution lamp to light. Flight is continued using the remaining pair of spoilers and asymmetrical horizontal tail operation for roll control. The spoiler monitor may be reset by depressing a spoiler reset button. This will cause the spoiler caution lamp to go out and will restore hydraulic pressure to the pair of spoilers that is locked down. If the malfunction still exists, the faulty spoiler will again extend and the previous sequence of events will be repeated. One attempt to reset a faulty spoiler is sufficient. In the event a spoiler extends because of a failure while roll autopilot is engaged, the wings must be held level by the pilot. Roll autopilot does not move the control stick and the pilots control stick corrective motion will be required to operate the monitor. When the pilot moves the control stick to hold wings level, the monitor will vote and the failed spoiler will be locked down as previously described. For deceleration during ground roll, the flight control spoilers are used in conjunction with the ground roll spoilers. In which case, they are controlled with the ground roll spoiler switch (refer to "Aerodynamic Deceleration Equipment", this section).

YAW CHANNEL.

Manual control of the aircraft in yaw is achieved by using conventional rudder and rudder pedals. Movement of the rudder pedals is transmitted to the rudder actuator control valve by a combination of control cables, push-pull tubes, and bellcranks. The control valve controls the flow of hydraulic fluid to the rudder actuator. The actuator moves the rudder in the direction commanded by the rudder pedal movement. Rudder authority is either of two configurations: 30 degree authority or 7-1/2 degree authority. In one configuration, rudder pedal travel is approximately 2-1/2 inches, full pedal force is 86 pounds, and full rudder command is 30 degrees. In the other configuration, full rudder pedal travel is approximately one inch, and full rudder command is 7-1/2 degrees. Rudder authority configuration is determined by the position of either the control system switch or the rudder authority switch. Rudder breakout force is approximately 17 pounds and approximately 86 pounds is required to achieve the available pedal travel.

PITCH-ROLL MIXING.

Combined roll and pitch movements of the control stick are transmitted by the linkage of their respective channel to pitch-roll mixer assembly where they are combined and converted into left and right horizontal stabilizer actuator command signals. The mixer pitch channel input stops are set at 25 degrees up and 10 degrees down symmetrical horizontal stabilizer command. The mixer roll channel input stops are set at ±8 degrees of differential horizontal stabilizer command. Therefore, the combined mixer stops limit individual actuator commands to 33 degrees up or 18 degrees down. However, the horizontal actuators are limited to 30 degrees trailing edge up and 15 degrees trailing edge down. Some of the excess horizontal stabilizer actuator command will be absorbed as overtravel within the valve spool when the pilot is commanding full pitch and roll. Any channel input command in excess of the pitch-roll mixer input stops will result in limiting of control stick motion.

STABILITY AUGMENTATION.

The stability augmentation system is triple redundant in that each pitch, roll, or yaw command generates three redundant signals. Each of the pitch, roll, and yaw channel electronics incorporates three signal selectors. Whenever a pitch, roll, or yaw command is generated, each of the three signal selectors for the particular channel, anaylzes each of the three redundant signals. Through a process of majority logic voting, the signal selectors select the middle value signal and send it to the appropriate damper servo. Should one of the three redundant signals to the selectors be erroneous, each selector will select one of the remaining two good signals and send it to the damper servo. Of the three signals sent to the damper servos, one is used by the damper as a model or standard which the other two are compared against. Should one of these two signals be erroneous, it will be voted out by the damper logic circuitry. A self-adaptive gains system continuously varies the gain of the signals sent to the pitch and roll damper servos as flight conditions change to optimize airplane response.

Command Augmentation.

The effectiveness of the control surfaces varies with the flight condition. At low speed and high altitude several degrees of elevator are required to command one g while at high speed and low altitude it may take less than a degree. Since stick force and surface movement are directly related to stick motion an unaugmented system will require heavy stick forces at low speed and very light forces during high speed low altitude flight. The "feel" would then vary continuously with flight condition. The command augmentation system augments the control stick command through the damper. The damper moves directly proportional to the stick input (for a particular gain). This damper input is then reduced proportional to the resulting airplane response. At a flight condition where the control surface effectiveness is high the aircraft response will be large and the damper contribution will be reduced greatly. Likewise, in a low response flight condition the damper contribution will not be reduced as much. The surface motion will then vary with flight condition so that the resulting airplane response will always be very nearly the same for the stick force. The continuously adapting gain helps the system to approach the ideal of a constant stick force and aircraft response relationship.

Left Sidewall (Typical)

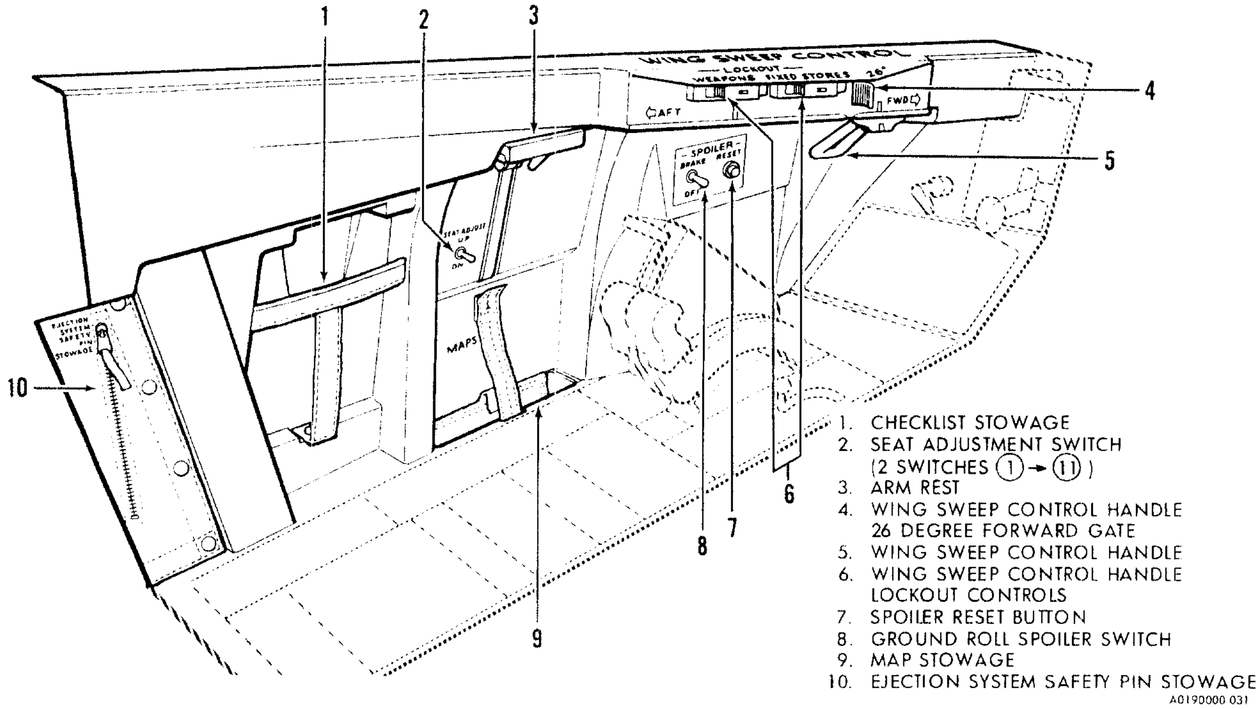


Figure 1-13A.

TRIM.

Yaw Trim.

Yaw trim is accomplished by an electrically driven actuator which mechanically positions the rudder linkage. Since the yaw trim actuator is in series with the rudder linkage, there is no movement of the rudder pedals as trim is applied. Yaw trim is controlled by either of two rudder trim switches, one located on the auxiliary flight control panel and one located on the flight control test panel. Hardover yaw trim that will not respond to either of the rudder trim switches will require about 80 pounds of rudder pedal force to hold the rudder centered. In event of such a malfunction, the rudder authority switch is used to increase rudder pedal authority and reduce the required force to approximately 30 pounds.

Roll Trim.

Roll trim is accomplished through the roll damper servo. Roll trim command signals operate roll trim relays in the feel and trim assembly. The relays supply 26 volts ac to the roll trim integrator motor for manual control. The output of the roll trim integrator supplies a signal which is summed with the roll rate in the roll computer and sent to the roll damper servo which positions the horizontal stabilizer. Therefore roll trim commands a given roll rate rather than a given amount of control surface displacement. Since the output of the roll damper servo is in series with the roll channel linkage, the control stick does not move as trim is applied. Manual roll trim is controlled by a trim button located on either control stick.

Pitch Trim.

Pitch Trim Series (Autotrim). Pitch series trim is incorporated into the system to automatically return the pitch damper to neutral, when operating in stability augmentation. This will provide full damper authority at all times in the event it is needed and prevent damper disengage transients. Displacement of the control stick causes a signal to be sent from the stick transducer to the pitch damper causing the damper to displace. This damper displacement is sensed and causes the pitch series trim to return the damper to neutral. In this manner full damper authority is always available to the pilot and disengage transients are eliminated when the damper is turned off or fails.

Manual Pitch Trim. Manual pitch trim is accomplished by either a pitch trim series actuator or a pitch trim parallel actuator. Pitch trim command signals operate pitch trim relays in the feel and trim assembly, thus supplying 115 volts ac to one of the pitch trim actuators. Output of the pitch trim parallel

Flight Control System Schematic

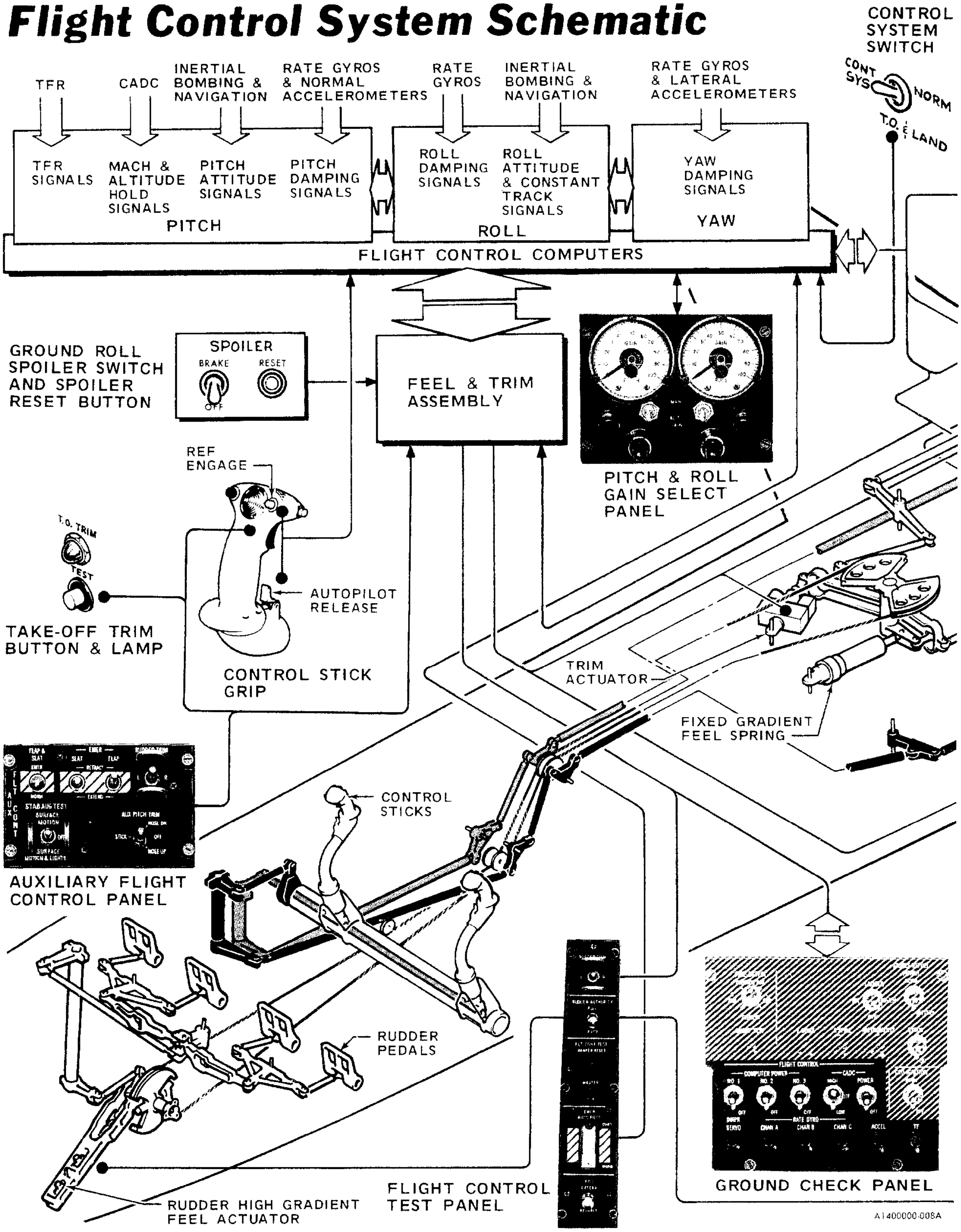


Figure 1-14. (Sheet 1 of 2)

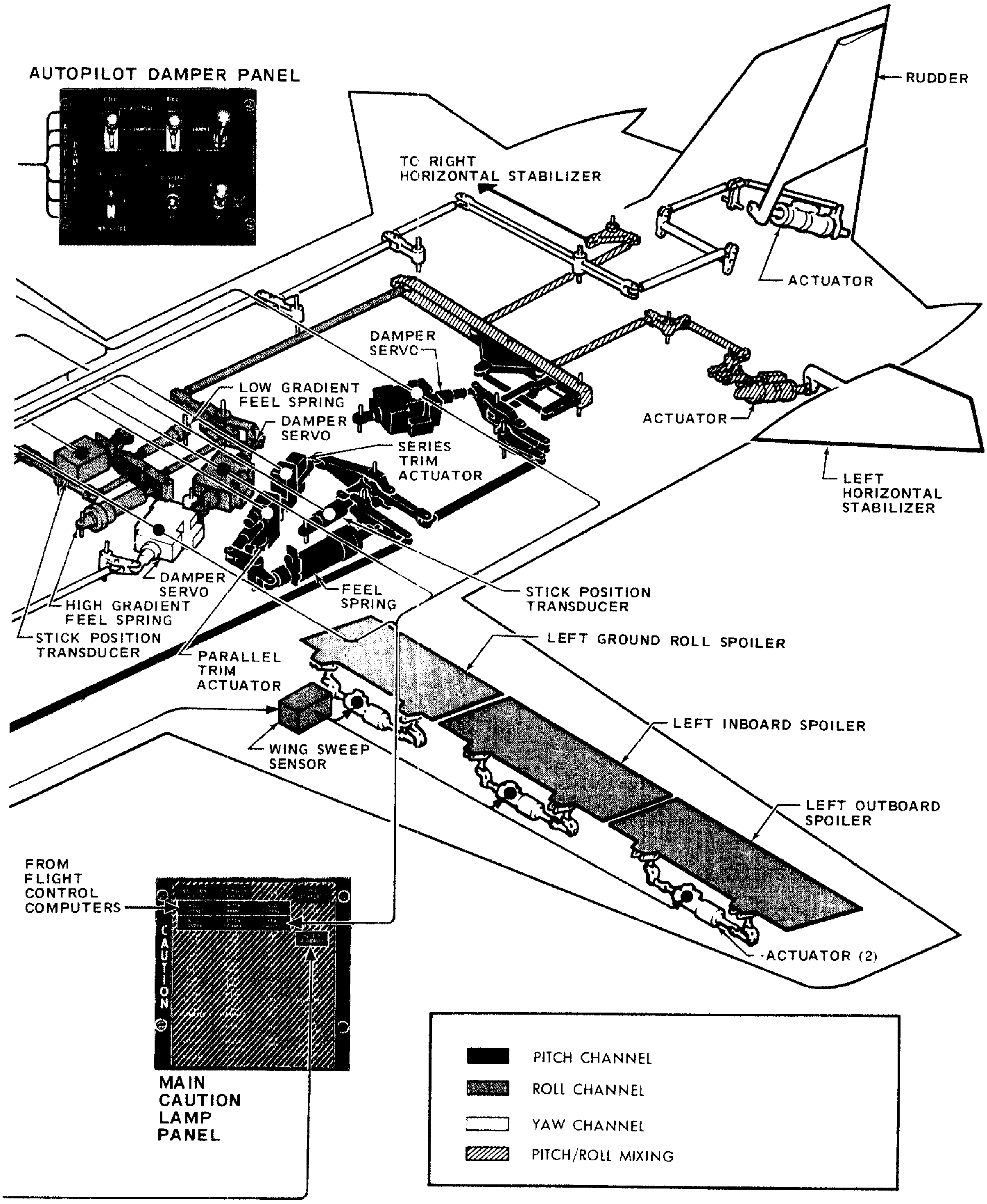


Figure 1-14. (Sheet 2 of 2)

actuator causes the control stick to move as trim is applied. Output of the pitch trim series actuator does not cause the control stick to move. Manual pitch trim is controlled by either a trim button on the control stick or by an auxiliary pitch trim switch (series trim) on the auxiliary flight control panel. The pitch trim configuration changes as a function of positions of the pitch damper switch, auxiliary pitch trim switch, and the control system switch. The configurations are as follows:

1. The pitch parallel trim actuator drives to neutral and is locked when:

   a. The pitch damper is off or:

   b. The auxiliary pitch trim switch is out of the STICK position.

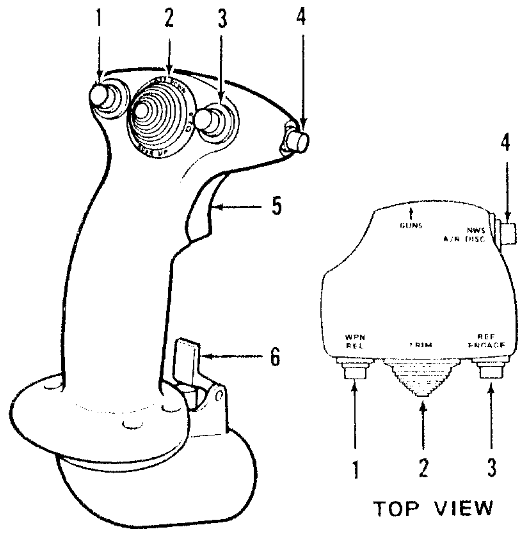
2. The pitch series trim actuator stops at its present position and no longer keeps the damper at neutral when:

   a. The pitch damper is turned off or:

   b. The control system switch is in T.O. & LAND.

When using the auxiliary pitch trim switch with the pitch damper on, pitch trim is applied by electrically positioning the pitch damper. The pitch trim series actuator is used for trim when the pitch damper is off by use of the stick trim button.

Control Sticks



1. WEAPON RELEASE BUTTON
2. TRIM BUTTON
3. REFERENCE ENGAGE BUTTON
4. AERIAL REFUEL AND NOSE WHEEL STEERING BUTTON
5. GUN TRIGGER
6. AUTOPILOT RELEASE LEVER

Figure 1-15.

FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS.

Control Sticks.

The two control sticks, one located at each crewmembers station, are mechanically interconnected. Each stick grip (figure 1-15), contains a trim button, weapon release button, reference engage button, aerial refuel and nose wheel steering button, a gun trigger and an autopilot release lever. On airplane (12)→, the control sticks serve as a means of actuating the crew module bilge/flotation bag inflation pump. Refer to "Crew Module Escape System," this section.

Rudder Pedals.

Rudder control is provided by two sets of rudder pedals, one set located at each crewmembers station. The two sets of rudder pedals are mechanically interconnected and in addition to controlling the rudder, each pedal operates the respective wheel brake in the conventional manner.

Trim Button.

A trim button (2, figure 1-15), located on each control stick grip, is provided to control trim in the pitch and roll axes. The button has positions marked LWD, RWD, NOSE UP, NOSE DOWN, and is spring loaded to the center unmarked OFF position. Moving the button to NOSE UP or NOSE DOWN causes the pitch trim actuator to position the horizontal stabilizer surfaces symmetrically with trailing edge either up or down as selected. Moving the button to LWD or RWD causes the roll damper servo to position the horizontal stabilizer surfaces asymmetrically as selected. The aircraft commander's trim button can always override the pilot's trim button control. Maximum travel of the horizontal stabilizer using the trim button is 10 degrees up and 8 degrees down through the parallel trim with the pitch damper on. Maximum command for roll trim using the trim button is equivalent to 32 degrees roll rate per second. The resultant horizontal stabilizer travel is, that travel required at the particular flight conditions to cause the airplane to roll at 32 degrees per second.

Auxiliary Pitch Trim Switch.

An auxiliary pitch trim switch (15, figure 1-17), with positions marked STICK, NOSE DN, NOSE UP, and OFF, is located on the center console. The switch is provided to control the pitch trim series actuator. When the switch is in the STICK position, pitch trim signals can be commanded only by the trim buttons on the control sticks. When the switch is held in NOSE DN or NOSE UP position, the pitch trim series actuator and the pitch damper move the horizontal stabilizer symmetrically as selected until the switch is released to OFF or the limits are reached. With the switch in the OFF, NOSE DN, or NOSE UP position, the trim buttons on the control sticks are inoperative. When the switch is not in the STICK position, the roll trim command is zero and roll trim cannot be commanded from the control stick trim button.

Rudder Trim Switches.

Two rudder trim switches (37, figure 1-4 and 14, figure 1-17), located on the left and center consoles respectively, are provided for rudder trim control. The switches have positions marked L and R and are spring loaded to the center unmarked OFF position. Holding either switch in L or R causes the rudder trim actuator to drive the rudder in the selected direction until the switch is released to OFF or a maximum deflection of 7-1/2 degrees is reached.

Takeoff Trim Button.

The takeoff trim button (20, figure 1-5), is located on the left main instrument panel. When the button is depressed, the takeoff trim relay is energized; the pitch parallel trim and yaw trim actuators are driven to 0 degrees; the roll trim integrator is synchronized so that the output to the roll damper is zero; the auxiliary pitch trim integrator is driven to a null; and the pitch trim series actuator is driven to a noseup position of 3.8 degrees.

Autopilot/Damper Switches.

Three switches (9, figure 1-17), one each for the pitch, roll, and yaw channels, are located on the center console. The pitch and roll damper switches are three position switches marked AUTOPILOT, DAMPER and OFF and are solenoid held in the AUTOPILOT and OFF positions and are spring-loaded to the DAMPER position. The yaw damper is a two position switch marked DAMPER and OFF. It is solenoid held in the OFF position and is spring-loaded to the DAMPER position. Placing any of the switches to DAMPER turns the respective damper on. The pitch and roll channels come on with the automatic gain at a low value and then begin setting the correct gain for that flight condition. Placing either the pitch or roll switch to AUTOPILOT will engage autopilot attitude stabilization. Placing a switch to OFF disengages the damper system of the respective channel and causes the respective damper caution lamp to light. These switches are also used to engage the autopilot. For description of that function, refer to "Autopilot System," this section.

Auto Terrain Following Switch.

The auto terrain following (auto TF) switch (11, figure 1-17), located on the center console, is a three position switch marked AUTO TF, FLY UP ONLY and OFF. The switch is solenoid held in AUTO TF and OFF positions and is spring loaded to FLY UP ONLY. With the terrain following radar operating, the pitch damper can be engaged to maintain a preselected altitude by placing the switch to AUTO TF and depressing either reference engage button. With the switch in either AUTO TF or FLY UP ONLY positions a TFR failure provides a signal to the pitch damper to perform a 2g pull up maneuver. With the switch in the OFF position the pitch damper cannot receive TFR signals and the TF fly up off caution lamp will light. For additional information on the auto TF switch and TF fly up off caution lamp refer to "Terrain Following Radar," this section.

Damper Reset Button.

The damper reset button (35, figure 1-4), located on the left console, is a momentary pushbutton switch labeled DAMPER RESET. When the button is depressed, the pitch, roll and yaw damper caution lamps and their respective channel caution lamps on the main caution lamp panel will go out and the dampers and their respective electronic channels will be simultaneously reset to accept inputs for logic voting. If a malfunction is present at the time the reset button is released, the appropriate caution lamps will light.

Rudder Authority Switch.

The rudder authority switch (36, figure 1-4), located on the left console, has positions marked FULL and AUTO. When the switch is in AUTO, full rudder authority of 30 degrees either side of center is available provided the control system switch is in T.O. & LAND or rudder authority of 7-1/2 degrees either side of center if the control system switch is in NORM. When the rudder authority switch is in FULL position, full rudder authority is available regardless of the position of the control system switch.

Control System Switch.

The control system switch (2, figure 1-12), located on the landing gear control panel, is a two position switch marked T.O. & LAND and NORM. When the switch is placed in the T.O. & LAND position, the following actions occur: the rudder variable authority actuator moves to the 30 degree authority position, the TFR fly-up signals to the pitch damper are locked out, the pitch and roll computer gains are driven to preset values if AUTO gains are selected on the pitch and roll gain select panel. Roll gain is driven to 100 percent, and pitch gain will be driven to either 30 percent or 100 percent depending on the computer configuration in the airplane. Also when the control system switch is in T.O. & LAND, the pitch series trim actuator is locked. When the switch is in NORM, the rudder authority is 7-1/2 degrees, the pitch and roll gains are as determined by the flight control computers (if AUTO gains are selected on the pitch and roll gain select panel), the series trim actuator is operable and the pitch damper will respond to TFR fly-up signals. The T.O. & LAND position is used only during takeoff and landing operations.

Flight Control Master Test Button.

The flight control master test button (34, figure 1-4), located on the left console, provides a source of power to the flight control test switches and buttons on the ground check panel and to the stability augmentation test switch on the center console. Depressing the button closes a switch, thus allowing power to be applied to the flight control test switches and buttons. When the button is released, these switches ana buttons are inoperable.

Stability Augmentation Test Switch.

The stability augmentation test switch (23, figure 1-17), located on the center console, is a three position switch marked SURFACE MOTION, SURFACE MOTION & LIGHTS with an unmarked center OFF position. This switch, when used in conjunction with the control system switch and the flight control master test button, provides a means of ground checking the stability augmentation system. With the control system switch in NORM and the master test button depressed, selecting the following positions of the stability augmentation test switch will obtain the results as indicated:

1. SURFACE MOTION;

   a. Right horizontal stabilizer trailing edge moves full down

   b. Left stabilizer trailing edge moves down slightly

   c. Rudder trailing edge moves right then left.

2. SURFACE MOTION & LIGHTS;

   a. Same horizontal stabilizer motion as items 1 (a) and (b) above.

   b. Rudder trailing edge moves right and back to neutral.

   c. Three (3) damper and three (3) channel caution lamps light

   d. On some airplanes, the roll and pitch gain changer caution lamps will light.

Spoiler Reset Button.

The spoiler reset button (7, figure 1-13A), located on the crew module left sidewall, is a momentary push-button labeled SPOILER RESET. The button is provided to reset the spoiler monitor in the event that a malfunction has caused a pair of spoilers to be voted out and locked down. If a pair of spoilers has been locked down and the spoiler caution lamp is lighted, depressing the spoiler reset button will cause the caution lamp to go out and the spoiler circuitry to be reset to accept signals from the spoiler transducers. If the malfunction still exists the faulty spoiler will again extend and the corrective control stick motion will cause the spoilers to again lock down and the spoiler caution lamp to light.

Pitch and Roll Gain Selector Switches.

Two gain selector switches, one for pitch and one for roll, are located on the pitch and roll gain select panel (8, figure 1-21). The switches are labeled GAIN and have positions AUTO and MAN. When either switch is in AUTO the stability augmentation damping signal gain of the selected channel will be automatically determined by the flight control computer. The MAN position is provided to enable the pilot to manually adjust the output gain of the selected computer and thus control stability augmentation damping response.

Pitch and Roll Manual Gain Control Knobs.

Two manual gain control knobs, one for pitch and one for roll, are located on the pitch and roll gain select panel (8, figure 1-21). The knobs are provided to manually adjust the gain of the stability augmentating damping signal. Each knob consists of an inner and outer scale. The outer scale is graduated from 0 to 10 representing 100 percent of gain control. The inner scale is graduated from 0 to 10 representing 10 percent of total gain control. One complete revolution of the inner scale will cause the outer scale to rotate 10 percent.

Computer Power Switches.

The computer power switches (16, figure 1-16) labeled NO 1, NO 2, and NO 3 are located on the aft console. When any one of the switches is placed in the UP position, activating power is applied to the selected branch in each pitch, roll, and yaw computer. The switches are held in this position when the door to the panel is closed.

Damper Servo Button.

The damper servo button (15, figure 1-16), labeled DMPR SERVO, is located on the aft console. When the damper servo, rate gyro channel B and channel C buttons and flight control test master switch are depressed and held, the electrical power to valve No. 1 on each damper servo is interrupted, resulting in an electrical command signal from each computer, causing the damper servos to vote hydraulically. This causes the pitch, roll, and yaw damper and channel caution lamps to light.

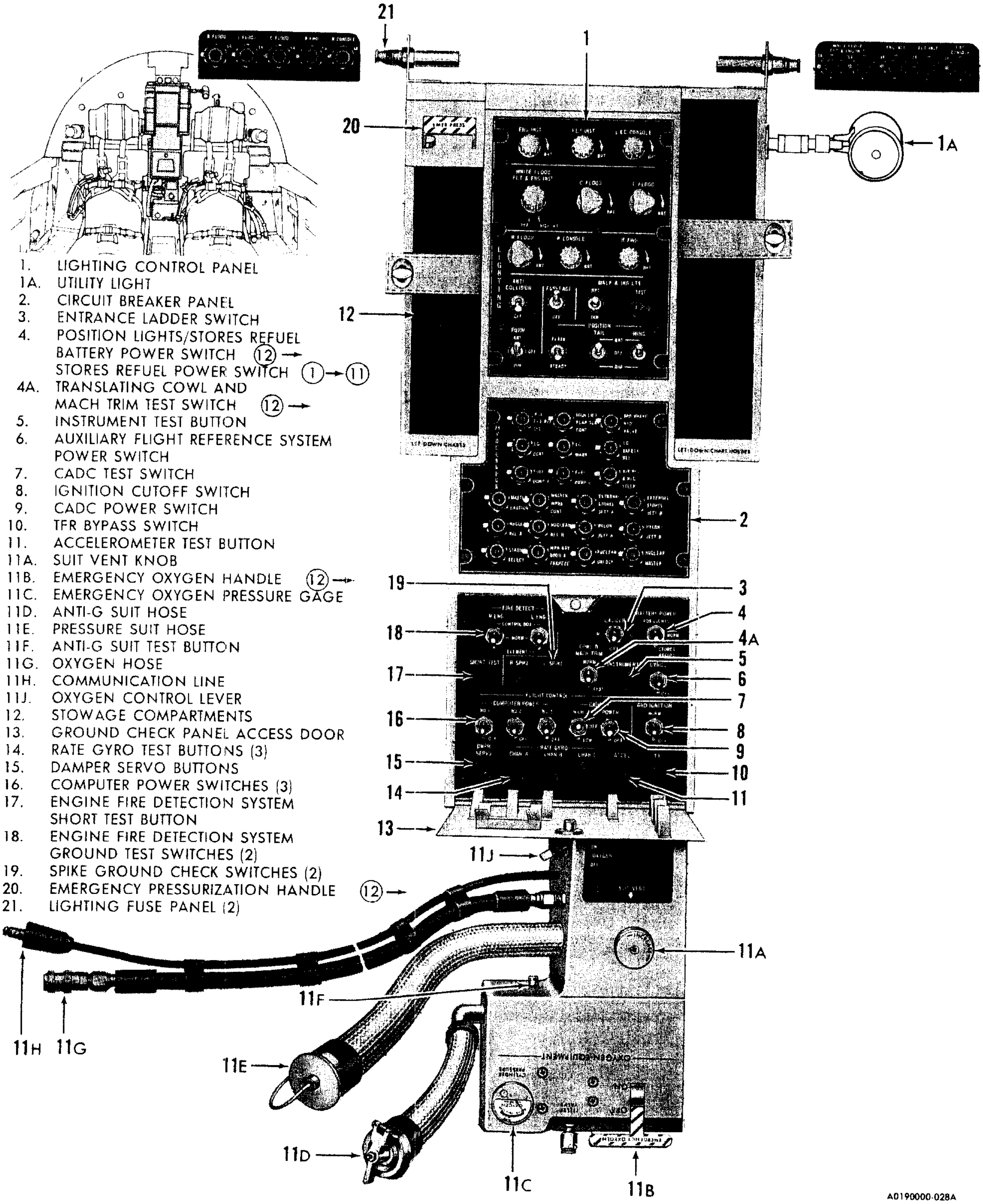
Rate Gyro Test Buttons.

The rate gyro test buttons (CHAN A, CHAN B, and CHAN C) (14, figure 1-16), are located on the aft console. When two or more of the buttons are depressed in conjunction with the flight control master test button, the respective rate gyros are torqued, resulting in a predetermined displacement of the primary flight control surfaces. The CHAN A button, when depressed, torques the "A" gyros in the pitch, roll, and yaw channels. The CHAN B and CHAN C buttons, when depressed, torque their respective gyros.

Takeoff Trim Indicator Lamp.

A takeoff trim indicator lamp (19, figure 1-5), located on the left main instrument panel, is provided to indicate when the horizontal stabilizer and rudder are in the proper trim position for takeoff and the aux-

Aft Console (Typical)



1. LIGHTING CONTROL PANEL
1A. UTILITY LIGHT
2. CIRCUIT BREAKER PANEL
3. ENTRANCE LADDER SWITCH
4. POSITION LIGHTS/STORES REFUEL BATTERY POWER SWITCH (12) →
   STORES REFUEL POWER SWITCH (1)→(11)
4A. TRANSLATING COWL AND MACH TRIM TEST SWITCH (12) →
5. INSTRUMENT TEST BUTTON
6. AUXILIARY FLIGHT REFERENCE SYSTEM POWER SWITCH
7. CADC TEST SWITCH
8. IGNITION CUTOFF SWITCH
9. CADC POWER SWITCH
10. TFR BYPASS SWITCH
11. ACCELEROMETER TEST BUTTON
11A. SUIT VENT KNOB
11B. EMERGENCY OXYGEN HANDLE (12) →
11C. EMERGENCY OXYGEN PRESSURE GAGE
11D. ANTI-G SUIT HOSE
11E. PRESSURE SUIT HOSE
11F. ANTI-G SUIT TEST BUTTON
11G. OXYGEN HOSE
11H. COMMUNICATION LINE
11J. OXYGEN CONTROL LEVER
12. STOWAGE COMPARTMENTS
13. GROUND CHECK PANEL ACCESS DOOR
14. RATE GYRO TEST BUTTONS (3)
15. DAMPER SERVO BUTTONS
16. COMPUTER POWER SWITCHES (3)
17. ENGINE FIRE DETECTION SYSTEM SHORT TEST BUTTON
18. ENGINE FIRE DETECTION SYSTEM GROUND TEST SWITCHES (2)
19. SPIKE GROUND CHECK SWITCHES (2)
20. EMERGENCY PRESSURIZATION HANDLE (12) →
21. LIGHTING FUSE PANEL (2)

Figure 1-16.

iliary pitch trim integrator is zeroed. When the takeoff trim button is depressed and all affected surfaces reach their proper position, the lamp lights. When the takeoff trim button is released, the lamp goes out.

Roll, Pitch, and Yaw Channel Caution Lamps.

Three amber caution lamps, one each for the roll, pitch, and yaw channels, are located on the main caution lamp panel (14, figure 1-5). Lighting of any one of the lamps indicates that a malfunction has been sensed in the computer of the respective roll, pitch, or yaw channel. Since the electronics in each channel is triple redundant, lighting of one of these caution lamps indicates that one of the three sets of electronics is in error (passive first failure) and does not indicate a complete failure.

Roll, Pitch, and Yaw Damper Caution Lamps.

Three amber caution lamps, one each for the roll, pitch, and yaw dampers, are located on the main caution lamp panel (14, figure 1-5). Lighting of any one of the lamps indicates that a malfunction has been sensed in its respective damper. Since each damper has two active valves and a model valve, lighting of one of the caution lamps does not indicate a complete damper failure.

Flight Control Spoiler Caution Lamp.

The flight control spoiler caution lamp, located on the main caution lamp panel (14, figure 1-5), is provided to indicate when a malfunction in the spoiler circuitry has occurred causing a symmetric pair of flight control spoilers to be locked down.

Roll and Pitch Gain Changer Caution Lamps.

Two amber gain changer caution lamps, one each for the roll and pitch gain changer, are located on the main caution lamp panel (14, figure 1-5). Lighting of either of these lamps indicates that a portion of the triple redundant gain setting in the respective channel is in error. Depressing the damper reset button will reset the lamp for a temporary error. Since the gain changer circuitry in each channel is triple redundant, lighting of one of these caution lamps indicates that one of these three sets of electronics is in error and does not indicate a complete failure.

Pitch and Roll Gain Indicators.

Two gain indicators, one for pitch and one for roll, are located on the pitch and roll gains select panel (8, figure 1-21). Each indicator displays, in percentage of total gain available, the gain of the stability augmentation damping signal coming from the flight control computer. The indicators are graduated from 0 to 100 percent and display gain in either AUTO or MAN position.

Gain Selector Switch Indicator Lamps.

Two gain selector switch indicator lamps, one for pitch and one for roll, are located on the pitch and roll gains select panel (8, figure 1-21). The lamps are push-to-test type and are provided to indicate when the gain control is in the manual mode. When either the pitch or roll gain selector switch is in MAN, the corresponding lamp will light.

Rudder Authority Caution Lamp.

An amber rudder authority caution lamp (14, figure 1-5), is located on the main caution lamp panel. Lighting of the lamp indicates the rudder authority actuator is not in the position commanded by the control system switch.

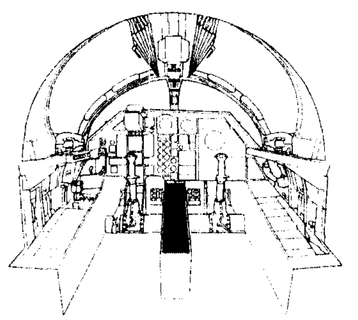
AUTOPILOT SYSTEM.

The autopilot system consists of electronic circuitry that, in conjunction with the primary flight control system, controls the aircraft during the four modes of autopilot flight. The autopilot system receives input signals from other systems and computes command signals to the pitch and roll dampers to control the aircraft. The autopilot modes are attitude stabilization, mach hold, altitude hold, and constant track. Incompatible mode selection is prevented by circuit interlocks. Attitude stabilization is normally in effect when the autopilot is engaged. Attitude stabilization will hold the aircraft at the reference roll and/or pitch attitude until selection of another autopilot mode or until pilot initiation of control stick steering. The aircraft may be manually maneuvered at any time by use of control stick steering without disengaging the autopilot. During operation of the autopilot, the control stick will not reflect the position of the surfaces. Pitch autopilot disengage transients are reduced by the pitch series trim actuator. Engage transients are prevented by continuous synchronization of the pitch and roll attitude input signals so that their commands to the dampers are zero at the time of autopilot engagement.

ATTITUDE STABILIZATION MODE.

The attitude stabilization mode is the initial control mode established when the autopilot is engaged. Attitude stabilization can be engaged in either or both the roll and pitch channels. Attitude reference signals are received by the pitch and/or roll computers from the inertial reference unit of the inertial bombing navigation system. Resultant signals from the pitch and/or roll computer then controls the pitch and/or roll damper position, thus holding the aircraft at the reference attitude existing at the time of autopilot engagement; however, roll angles of less than 3 (±1) degrees will result in a wings level attitude command upon engagement of this mode. If the constant track mode is selected, the roll damper will control the aircraft according to the new reference. However, when the constant track mode is discontinued, the autopilot will revert back to attitude stabilization and will maintain the attitude that existed at the time of disengagement. For example, if pitch and roll autopilot are engaged with the aircraft in a twenty-degree bank, this bank angle will be held. If the constant track mode is then selected, the aircraft will respond

Center Console (Typical)



* See Figure 1-16

1. FUEL CONTROL PANEL
2. TERRAIN FOLLOWING RADAR CONTROL PANEL
2A. EJECTION HANDLE (2) 12 →
3. SPIKE CONTROL SWITCHES
4. ENGINE GROUND START SWITCH
5. THROTTLES
6. AIR START BUTTON
7. SPEED BRAKE SWITCH
8. MICROPHONE SWITCH
8A. TRANSLATING COWL SWITCH
9. AUTOPILOT/DAMPER SWITCHES
10. CONSTANT TRACK SWITCH
11. AUTO TERRAIN FOLLOWING RADAR SWITCH
12.&13. EMERGENCY FLAP AND SLAT SWITCH 12 → (2 SWITCHES 1 → 11)
14. RUDDER TRIM SWITCH
15. AUXILIARY PITCH TRIM SWITCH
16. ELECTRICAL CONTROL PANEL
17. AIR CONDITIONING CONTROL PANEL
*18. ANTI-G SUIT TEST BUTTON
*19. PRESSURE SUIT HOSE
*20. ANTI-G SUIT HOSE
*21. OXYGEN AND COMMUNICATION HOSE
*22. OXYGEN CONTROL LEVER
23. STABILITY AUGMENTATION TEST SWITCH
24. FLAP AND SLAT SWITCH
25. ALTITUDE HOLD/MACH HOLD SELECTOR SWITCH

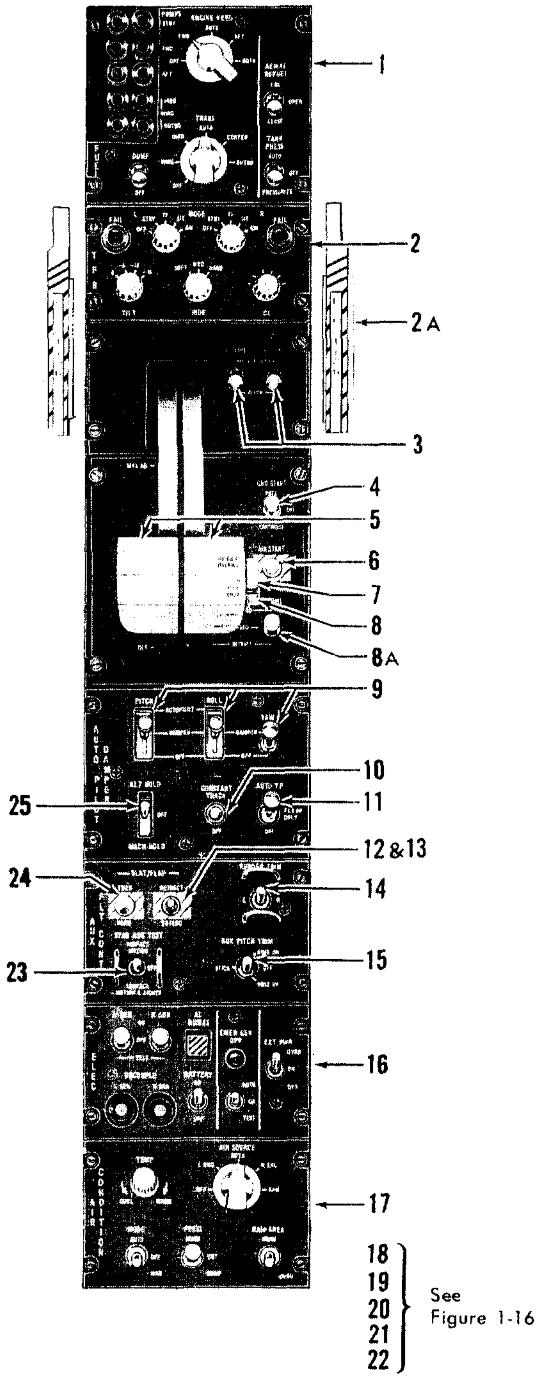


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Figure 1-17.

by rolling to wings level and holding the constant ground track. The original pitch attitude will continue to be controlled by attitude stabilization and will remain unchanged. If the constant track mode is subsequently discontinued, the autopilot will revert back to attitude stabilization in roll. The pilot may change the attitude stabilization pitch and roll references at any time by using control stick steering. The mode selector switch on the bomb nav control panel must be in one of its primary modes and the autopilot emergency override switch must be in NORM to engage the autopilot.

MACH HOLD MODE.

The mach hold mode maintains constant mach. In this mode, throttle position is fixed and mach is controlled by aircraft pitch attitude through operation of the horizontal stabilizer surfaces. Upon engagement of this mode, a mach reference is set up in the central air data computer (CADC). Any deviation of mach from this reference results in an error signal being sent to the pitch computer from the CADC. If mach increases above the reference, the resulting mach error signal will command a nose up attitude through the pitch damper causing the aircraft to return to the referenced mach number. An opposite command is used for a decrease in mach.

ALTITUDE HOLD MODE.

The altitude hold mode automatically maintains constant altitude. Upon engagement of this mode, an altitude reference is established in the central air data computer (CADC). Any deviation in altitude by the aircraft results in an altitude error being fed to the pitch computer from the CADC. If the aircraft altitude increases above the reference, the resulting altitude error signal will command a nose down attitude through the pitch damper until the desired altitude is obtained. An opposite command is given for a decrease in altitude.

CONSTANT TRACK MODE.

The constant track mode maintains the airplane on a constant ground track. When this mode is engaged, the existing ground track is sensed in the inertial bombing navigation system and is set up as a mode reference. Any deviation from this reference by the aircraft results in an error signal being sent from the inertial bombing navigation system to the roll computer. The roll computer, in turn, sends a command to the roll damper, correcting the deviation.

CONTROL STICK STEERING.

When any autopilot mode is engaged, including basic attitude stabilization, the reference controlling the aircraft can be disengaged by use of control stick steering. Control stick steering is activated in the pitch channel by applying a force greater than 1.7 pounds, in a forward or aft direction, to the top of the control stick. This mode is activated in the roll channel by applying a force of 1.3 pounds laterally to the control stick. When this force is applied in either or both channels, the reference or references are disengaged, a caution lamp will light, and the pilot can maneuver the aircraft to a new reference. When the force to the control stick is reduced below 1.7 pounds in the pitch channel or 1.3 pounds in the roll channel, attitude stabilization will automatically re-engage in the affected channel or channels provided the attitude limits are not exceeded. The reference engage button must be depressed to re-engage submodes. The attitude limits are ±30 degrees in pitch and ±60 degrees in roll. Should these limits be exceeded in one or both channels, attitude stabilization will not re-engage in that channel until its attitude angle is reduced to less than its limit. In addition, the roll channel can not be engaged if either the pitch attitude is greater than ±30 degrees or the yaw damper is turned OFF.

CONTROLS AND INDICATORS.

Computer Power Switches.

Three computer power switches (16, figure 1-16), located on the aft console, control electrical power for autopilot and certain flight control system operations. (Refer to "Flight Control System," this section). The autopilot is normally ready to engage after the power switches are placed in the ON position and the inertial reference unit of the inertial navigation bombing system is properly erected.

Autopilot/Damper Switches.

Two autopilot/damper switches (9, figure 1-17), one each for pitch and roll, are located on the center console. The switches have three positions marked AUTOPILOT, DAMPER, and OFF and are solenoid held by 28 volt dc power in the AUTOPILOT and OFF position and are spring-loaded to the DAMPER position. If while operating in the AUTOPILOT or OFF positions, this 28 volt dc holding power is lost, the switches will return to the DAMPER position. The switches operate independently of each other. When the switches are in the AUTOPILOT position, attitude stabilization is engaged and the airplane will maintain constant attitude. When the switches are moved to DAMPER or OFF, all other mode switches will move to OFF and the airplane will then revert to pilot-controlled flight. For additional information on these switches, as related to control of the roll and pitch dampers, refer to "Flight Control System," this section.

Constant Track Switch.

The constant track switch (10, figure 1-17), located on the center console, is marked CONSTANT TRACK and OFF. The switch is solenoid held by 28 volt dc power in CONSTANT TRACK and is spring-loaded to OFF. When the switch is placed in the CONSTANT TRACK position and the reference engage button is depressed, the airplane will be held on a constant ground track. The switch will not latch in CONSTANT TRACK if the roll autopilot/damper switch is not in the AUTOPILOT position. If while operating in CONSTANT TRACK position, 28 volt dc power to the holding solenoid is lost, the switch will return to the OFF position. The reference not engaged lamp will not light for this malfunction. When the switch is positioned to OFF, the autopilot will discontinue controlling the airplane in a constant track mode and will revert to attitude stabilization in the roll channel.

Altitude Hold/Mach Hold Selector Switch.

The altitude hold/mach hold selector switch (25, figure 1-17), located on the center console, is a three position switch marked ALT HLD, OFF, and MACH HLD. The switch is solenoid held by 28 volt dc power to ALT HLD or MACH HLD and is spring-loaded to OFF. When the switch is in the ALT HLD position and the reference engage button is depressed, the autopilot will control the airplane to maintain the altitude present at the time the mode was engaged. When the switch is positioned to MACH HLD and the reference engage button is depressed, the autopilot will control the airplane to maintain the mach number present at the time of mode engagement. The switch will not latch in either ALT HLD or MACH HLD position if the pitch autopilot/damper switch is not in AUTOPILOT position. Selection of the ATF MODE is incompatible with altitude hold or mach hold mode and will cause the switch to move to the OFF position. If while operating in MACH HLD or ALT HLD positions, 28 volt dc power to the holding relay is lost, the switch will return to the OFF position. The reference not engaged lamp will not light for this malfunction.

Reference Engage Buttons.

A reference engage button (3, figure 1-15), marked REF ENGAGE is located on each control stick grip. When any autopilot mode is selected, other than attitude stabilization, one of the reference engage buttons must be depressed before the mode will engage. Either button may be used to engage the autopilot.

Autopilot Release Lever.

The autopilot release lever (6, figure 1-15), located at the base of the stick grip, permits either pilot to disengage certain functions of the autopilot without removing his hand from the stick. Depressing the lever will return the autopilot/damper switches to DAMPER. This disengages all autopilot functions and places the airplane under pilot control. When auto TF is engaged and the autopilot release lever is depressed, the auto TF switch will return to the FLY UP ONLY position and the terrain following radar will no longer control the aircraft. Should the terrain following radar fail and the 2g climb command be generated, this climb command can be interrupted by depressing the autopilot release lever. The command will reappear, however, when the lever is released unless the TFR system is turned off or the auto TF switch is turned OFF.

Autopilot Emergency Override Switch.

The autopilot emergency override switch (33, figure 1-4), located on the left console, has positions marked NORM and OVRD. The switch is guarded to the NORM position. If the switch is positioned to OVRD, certain inputs are removed from the roll and pitch damper systems. These signals are roll and pitch autopilot commands, roll trim commands, and pitch damper trim inputs. The switch disconnects these command signals from the roll and pitch damper channels and causes the reference not engaged caution lamp to light.

Reference Not Engaged Caution Lamp.

The reference not engaged caution lamp, located on the main caution lamp panel (14, figure 1-5), will light when the selected autopilot reference is not engaged. When control stick steering is activated while attitude stabilization is controlling the airplane, the lamp will go out when stick force becomes so low as to allow attitude stabilization re-engagement. When any mode other than attitude stabilization is engaged and control stick steering is activated, the reference not engaged caution lamp will remain lighted until the reference engage button is depressed.

AUTOPILOT OPERATION.

Engaging the Autopilot.

1. Bomb nav mode selector knob - Primary mode.
   The bomb nav system must be in a primary mode for autopilot operation.

2. ADI - Check for normal indications.

3. Attain a safe altitude and trim the airplane to the desired attitude.

Note

Autopilot operation cannot be engaged at attitudes exceeding ±30 degrees in pitch and ±60 degrees in roll.

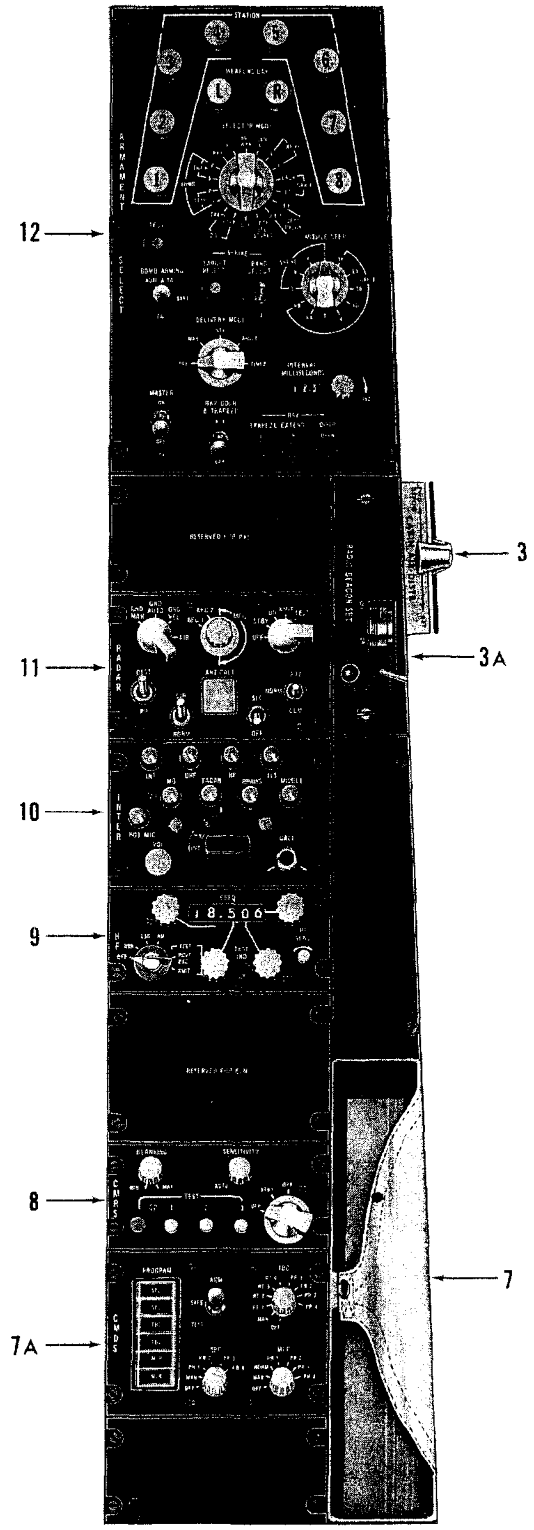
4. Pitch, roll autopilot/damper and yaw damper switches - DAMPER.
   Check that all dampers are operating properly.

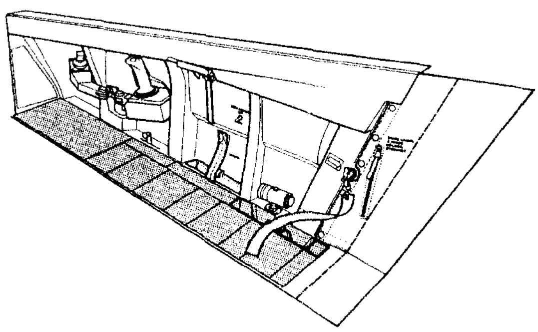
5. Flight instrument reference select knob - PRI.

6. Roll and pitch autopilot/damper switches - AUTOPILOT.

7. Reference not engaged caution lamp - Out.
   Check that the reference not engaged caution lamp goes out with no force applied to the control stick.

Right Console (Typical)





* See Figure 1-45A

*1. GAM-83 CONTROL STICK
*2. ATTACK RADAR TRACKING CONTROL HANDLE
3. CABIN AIR DISTRIBUTION CONTROL LEVER
3A. AN/URT-21 RADIO BEACON SET
*4. ARM REST
*5. SEAT ADJUSTMENT SWITCHES
*6. MAP STOWAGE
7. DATA STOWAGE CASE
7A. COUNTERMEASURES DISPENSER CONTROL PANEL
8. COUNTERMEASURES RECEIVER CONTROL PANEL
9. HF CONTROL PANEL
10. INTERPHONE CONTROL PANEL
11. ATTACK RADAR CONTROL PANEL
12. ARMAMENT SELECT PANEL

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Figure 1-18.

Selecting the Autopilot Control Modes.

After the autopilot is initially engaged in attitude stabilization, the pilot may select a single control mode or a combination of compatible modes by means of the mode switches on the autopilot/damper panel. A mode affecting the pitch channel (mach hold or altitude hold) may be selected simultaneously with a mode affecting the roll channel (constant track). However, two modes in the same channel cannot be selected simultaneously. The following procedures are for selecting each control mode after attitude stabilization has been engaged.

Selecting the Mach Hold or Altitude Hold Mode. Manually maneuver the aircraft to achieve the desired mach number.

1. Mach hold/altitude hold switch - MACH HLD or ALT HLD. (as applicable)
2. Reference engage button - Depress.
   If it is desired to increase or decrease the reference mach number, it will be necessary to disengage this mode while the desired change to airspeed is made. If it is desired to increase or decrease the reference altitude, use control stick steering to maneuver the aircraft to the new reference and depress the reference engage button.

Selecting the Constant Track Mode. Manually maneuver the aircraft until it is following the desired ground track.

1. Constant track switch - CONSTANT.
2. Reference engage button - Depress.
   If it is desired to change course, use control stick steering to maneuver the aircraft until the new ground track is established. Release the force to the stick and depress the reference engage button.

Disengaging the Autopilot.

To disengage all autopilot functions and place the aircraft under pilot control, either depress the autopilot release lever or place the pitch and roll autopilot/damper switches to DAMPER. In either case, all the switches will move to OFF.

CENTRAL AIR DATA COMPUTER SYSTEM (CADC).

Note

Until flight test data is available from which to incorporate correction cams, the CADC outputs of static pressure and angle-of-attack will be calibrated values rather than true values.

The airplane is equipped with a central air data computer system which provides aerodynamic intelligence to various control systems. The system consists basically of an electromechanical computer which processes raw data from the angle-of-attack transducer, pitot-static probe, and a temperature sensor probe located on the right side of the fuselage above the nose wheel well. The computer utilizes the following raw data: indicated static pressure, pitot pressure, total temperature, and indicated angle of attack. When this data reaches the computer, it is transformed into electrical signal outputs through an arrangement of transducers, mechanical linkage, and servo repeaters. The central air data computer is equipped with a failure monitoring system which continually monitors the computing functions. Should a computing function fail, a CADC caution lamp on the main caution lamp panel will light. If a computing function should fail, which affects the pressure altitude or indicated airspeed displays on the integrated flight instrument system, a warning flag will appear on the associated instrument. In addition, mach, angle-of-attack, and pressure altitude data good signals are supplied from the CADC monitor system to be used as failure monitor interlocks for hydraulic cooling, terrain following radar, and the flight director computer system respectively. The computer requires 115 volt ac and 28 volt dc power. Listed below are the various airplane systems served by the air data computer system, followed in parentheses by the computer outputs which go to the systems:

1. Altitude-vertical velocity indicator (pressure altitude and vertical velocity).
2. Airspeed mach indicator (mach number, indicated airspeed, true angle of attack).
3. Maximum safe mach assembly (pressure altitude, mach number, true air temperature).
4. Flight control system (angle of attack, indicated airspeed, incremental mach number and incremental LOG_{10} static pressure).
5. Translating cowl (mach number).
6. Engine mach lever actuator (mach number).
7. Engine bleed air ejector (mach number).
8. Spike caution lamp (mach number).
9. Bomb nav system (pressure altitude, pressure altitude rate, true airspeed).
10. True airspeed indicator (true airspeed).
11. Lead computing optical sight (pressure altitude, total pressure, true airspeed).
12. Terrain following radar (true angle of attack, true airspeed).

13. Angle-of-attack indexer (true angle of attack).
14. Environmental control (indicated airspeed, true air temperature).
15. Hydraulic system coolers (mach number).
16. Flight director (incremental pressure altitude and pressure altitude).
17. Landing gear warning (indicated airspeed, pressure altitude).
18. Marker beacon (pressure altitude).

CADC POWER SWITCH.

The CADC power switch (9, figure 1-16), with positions POWER and OFF, is located on the aft console. When the switch is in the OFF position, no aircraft power is supplied to the CADC or the maximum safe mach assembly. Also the CADC caution lamp will light and the OFF warning flags in the airspeed indicator and altimeter will appear. When placed in the POWER position 115 volt, 400 cycle, single phase ac power is supplied to the CADC and the maximum safe mach assembly.

CADC TEST SWITCH.

The CADC test switch (7, figure 1-16), with positions HIGH, OFF, and LOW, is located on the aft console. The test switch activates a self test system in the CADC. The normal system inputs are disconnected from the CADC, and a set of pre-selected test inputs are fed into the CADC. The HIGH position of the switch is used in conjunction with the pitot heater switch to ground check the total temperature probe heater. Normally this switch is used by the flight crew only during functional or acceptance check flights.

MAXIMUM SAFE MACH ASSEMBLY.

The maximum safe mach assembly (MSMA) receives mach number, pressure altitude and true air temperature signals from the central air data computer and wing sweep position from the wing sweep sensor and provides outputs to the maximum safe mach indicator and to the reduce speed warning lamp. The MSMA computes the maximum continuous safe mach of the airplane regardless of whether the limitation is due to structural loading limitation or temperature and provides a signal to the maximum safe mach indicator. It also continuously compares the airplane mach number to the maximum safe mach, computed as a function of structural load limit, and provides a signal to light the reduce speed warning lamp when the airplane reaches maximum allowable speed. The MSMA utilizes 115 volt ac power from essential ac bus through the central air data computer power switch and 28 volt dc power from the essential dc bus.

AUXILIARY FLIGHT REFERENCE SYSTEM (AFRS).

The auxiliary flight reference system (AFRS) provides standby or backup attitude and directional information. The system consists of a number of electronic packages which receive, compute, and transmit gyroscopic attitude and directional reference signals. Basic components of the system include vertical and directional gyros, a coupler, a compass controller, and a remote compass transmitter (flux valve). The vertical gyro is unlimited in roll but is limited to ±82 degrees in pitch. Any change in airplane attitude with respect to the vertical reference is detected by the vertical gyro and electrically transmitted to the standby attitude indicator at all times and to the attitude director indicator when the system is operating in the auxiliary mode. The directional gyro and the flux valve operate together as the compass set to provide heading signals for the BDHI at all times and to the HSI and the ADI when the system is operating in the auxiliary mode. The compass set operates either as a gyro-stabilized magnetic compass (slaved mode) or as a directional gyro (DG mode). The two modes, slaved and DG, provide accurate heading reference for all latitudes. In the slaved mode, the system is basically a directional gyro slaved to the remote compass transmitter. This mode is designed for use at latitudes up to 70 degrees. In the polar regions, the direction of the earth's magnetic field becomes more vertical rather than horizontal to such an extent that the slaved mode is not reliable and the DG mode should be used. In the DG mode, the system is freed from the remote compass transmitter and operates as a free gyro indicating an arbitrary gyro heading. In the DG mode, apparent drift due to earth rotation is corrected. The random drift (precession rate) of the gyro in the DG mode will not exceed ±1.5 degrees per hour. This mode may be used at all latitudes but is most useful when operating in the polar regions or when the magnetic field is weak or distorted. On airplanes (14)→ a third mode, the compass (COMP) mode, provides unstabilized compass heading. The purpose of this mode is to permit continued operation of the AFRS in the event of a malfunction of the gyros. The AFRS operates on 115 volt ac power from the ac essential bus and 28 volt dc power from the dc essential bus.

FLIGHT INSTRUMENT REFERENCE SELECT SWITCH.

The flight instrument reference select switch (41, figure 1-4), located on the left console, has three positions marked PRI, AUX and AUX STBY INST ONLY. Placing the switch to the PRI (primary) position supplies pitch, roll and heading information from the bomb nav system to the following subsystems, as applicable.

- Autopilot
- Attitude Director Indicator
- Horizontal Situation Indicator
- Flight Director Computer
- Terrain Following Radar
- Lead Computing Optical Sight
- Attack Radar

Placing the switch to the AUX (auxiliary) position supplies pitch, roll and heading information from the AFRS (auxiliary flight reference system) to all the above subsystems except autopilot. Placing the switch to the AUX STBY INST ONLY position supplies heading information from the AFRS to the above subsystems and roll and pitch information from the AFRS to the standby attitude indicator only. This position prevents degradation of the standby attitude indicator information in the event of a malfunction of any of the various subsystems supplied by the AFRS. The AUX STBY INST ONLY position should be selected whenever a malfunction occurs in the bomb nav system, and when other subsystems that receive signals from the AFRS are not required to fly the airplane. Also this position should be used when making crosschecks or malfunction checks between the bomb nav system and AFRS. On airplanes (14)→ the standby attitude indicator receives isolated pitch and roll signals at all times, therefore the AUX STBY INST ONLY switch position has been deleted.

AUXILIARY FLIGHT REFERENCE SYSTEM POWER SWITCH.

The auxiliary flight reference system power switch (6, figure 1-16), located on the aft console, has positions GYROS and OFF. Placing the switch to GYROS supplies power to the AFRS gyros, the compass set, the BDHI, and the standby attitude indicator. Placing the switch to OFF de-energizes these components.

AFRS GYRO FAST ERECT BUTTON.

The auxiliary flight reference system gyro fast erect button (40, figure 1-4), located on the left console, provides a means for fast erection of the AFRS. The button is labeled ATT GYRO FAST ERECT. During initial turn on of the system (initial erection), the gyro will automatically erect at the fast rate. If re-erection is required after initial erection due to the limits of the gyro being exceeded, fast erection may be accomplished by depressing and holding the fast erect button until the attitude indicators return to normal. During initial erection or when the fast erect button is depressed the AUX ATT lamp (airplanes (14)→) on the main caution lamp panel will light, the OFF flag on the standby attitude indicator will come into view and, if the flight instrument reference select switch is in the AUX position, the OFF flag on the ADI will come into view. During initial erection or whenever the fast erect button is depressed, the displacement gyroscope erects at a rate of approximately twelve (12) degrees per minute.

COMPASS MODE SELECTOR KNOB. (1)→(13)

The compass mode selector knob, located on the compass control panel (2, figure 1-19), is used to select the mode of operation of the compass set. The knob has positions SLAVED, DG-N, and DG-S. Placing the knob in the SLAVED position selects the magnetic mode of operation. Placing the knob in DG-N or DG-S position selects the directional gyro mode. Since the direction of drift correction is determined by DG-N (north) or DG-S (south) position, the switch position selected must correspond to the hemisphere in which the airplane is operating.

COMPASS MODE SELECTOR KNOB. (14)→

The compass mode selector knob (5, figure 1-19A), located on the compass control panel, is used to select the mode of operation of the auxiliary flight reference system compass. The knob has three positions marked SLAVED, COMP, and DG. When the SLAVED mode is selected, gyro-stabilized magnetic heading from the remote compass transmitter is provided. In the DG mode, the remote compass transmitter information is removed from the system and the system operates as a free gyro indicating an arbitrary gyro heading. In the COMP mode, the compass heading is obtained directly from the remote compass transmitter without stabilization by the directional gyro and is used in event of an attitude malfunction of the auxiliary flight reference system.

Note

> When moving the knob from the SLAVED position to COMP the compass cards on the HSI and BDHI and the altitude sphere of the ADI will rotate off the heading and immediately return. This is normal. When moving the knob from COMP back to the SLAVED position the compass cards of the HSI and BDHI and the attitude sphere of the ADI will rotate off the heading but will not return until the heading set knob is depressed and held to null the synchronization indicator.

LATITUDE CORRECTION KNOB.

The latitude correction knob, located on the compass control panel (3, figure 1-19) and (4, figure 1-19A), is marked with latitudes from 0 degrees to 90 degrees. Setting the knob to the latitude at which the flight is being made determines the rate of gyro drift correction when operating in DG mode and on airplanes (14)→ improves accuracy when operating in SLAVED mode.

SYNCHRONIZER-SET KNOB. (1)→(13)

The synchronizer-set knob, located on the compass control panel (4, figure 1-19) provides a means of manually synchronizing the AFRS gyro with the remote compass when the compass mode selector knob is in the SLAVED mode, and to set in desired heading on the BDHI when the knob is in the DG-N or DG-S mode. The knob is marked - SYNC +. Rotating the knob in a clockwise direction, toward the plus (+) position, will increase the heading on the BDHI. The opposite will occur if the knob is moved to the minus (-) position. When in the DG-N or DG-S mode of operation, the direction of movement of the knob is determined by whether an increased or decreased heading is desired. When in the SLAVED mode, the direction of rotation of the knob is determined by the synchronization indicator pointer.

HEADING SET KNOB. (14)→

The heading set knob (6, figure 1-19A), located on the compass control panel, provides a means of rapidly synchronizing the AFRS gyro with the remote compass transmitter when operating in the SLAVED mode, and to set in desired heading on the BDHI when operating in the DG mode. When the compass is operated in the SLAVED mode, fast synchronization is accomplished by depressing and holding the knob depressed until the synchronization indicator on the compass control panel becomes centered. When the compass is operated in the DG mode, system heading is changed by depressing and turning the knob to the right to increase the heading and left to decrease the heading. The rate of heading change is determined by the amount the knob is turned. When the compass is operated in the COMP mode, the system continuously tracks the remote compass transmitter and it is not necessary to use the knob.

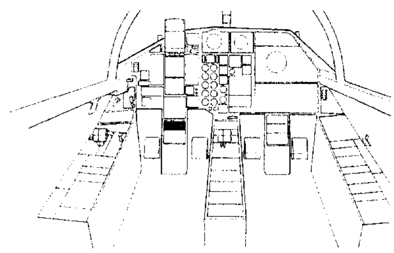
HEMISPHERE SELECTOR SWITCH. (14)→

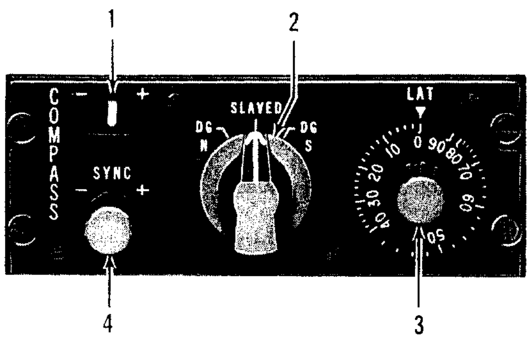
The hemisphere selector switch (3, figure 1-19A), located on the compass control panel, has two positions marked N (North) and S (South). The switch must be positioned to the correct hemisphere in which the airplane is operating to provide the proper polarity of the earth's rate correction.

SYNCHRONIZATION INDICATOR.

The synchronization indicator, located on the compass control panel (1, figure 1-19) and (2, figure 1-19A), indicates whether or not the AFRS gyro and remote compass are synchronized. During operation in the SLAVED mode the pointer will normally fluctuate slightly when the compass set is synchronized with the gyro. Should the compass get out of synchronization, the pointer will deflect toward either the plus or minus sign on the face of the indicator. To synchronize the system on airplanes (1)→(13) the synchronizer set knob must be turned in the opposite direction from the pointer indication until the pointer is centered (nulled). On airplanes (14)→ the heading set knob must be depressed and held until the pointer is centered to synchronize the system.

Compass Control Panel





1. SYNCHRONIZATION INDICATOR
2. COMPASS MODE SELECTOR KNOB
3. LATITUDE CORRECTION KNOB
4. SYNCHRONIZER – SET KNOB

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Figure 1-19. (1)→(13)

Note

On airplanes (1)→(13) the pointer can be centered (nulled) 180 degrees out of phase.

The indicator is de-activated when operating in the DG mode on airplanes (1)→(13) and when operating in the DG or COMP modes on airplanes (14)→.

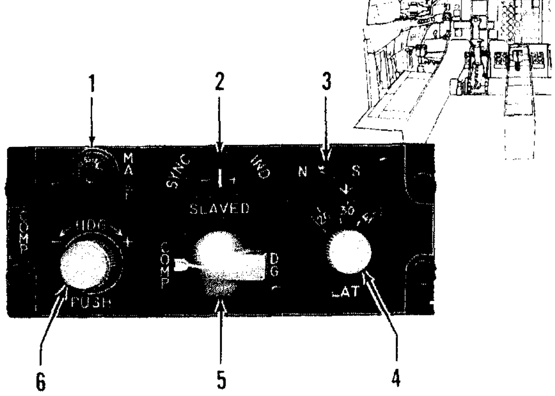
AUXILIARY ATTITUDE (AUX ATT) CAUTION LAMP. (14)→

The auxiliary attitude caution lamp, (figure 1-21A), located on the main caution lamp panel, will light if attitude information from the AFRS becomes unreliable. The lamp will also light during initial erection and when the fast erect button is depressed. When lighted, the amber letters AUX ATT are visible. Should the lamp light and remain on, the flight instrument reference select switch should be positioned to the PRI position. The standby attitude indicator will be unreliable when the lamp is lighted.

HEADING MALFUNCTION CAUTION LAMP. (14)→

An amber heading malfunction caution lamp (1, figure 1-19A), located on the compass control panel, is provided to indicate that the AFRS heading is unreliable. A push-to-test circuit is provided to check the lamp.

Compass Control Panel



1. HEADING MALFUNCTION CAUTION LAMP
2. SYNCHRONIZATION INDICATOR
3. HEMISPHERE SELECTOR SWITCH
4. LATITUDE CORRECTION KNOB
5. COMPASS MODE SELECTOR KNOB
6. HEADING SET KNOB

A5116000 297

Figure 1-19A. (14)→

AUXILIARY FLIGHT REFERENCE SYSTEM OPERATION.

The AFRS gyro, which provides auxiliary roll and pitch information, is limited to 82 degrees in pitch. Climbing or diving the airplane at angles greater than 82 degrees may cause large errors in pitch and roll indications. Therefore, after executing such maneuvers the fast erect button on the miscellaneous control panel should be depressed and held until attitude indications return to normal. If this is not accomplished, the gyro will erect to a normal position at a rate of 1.5 degrees per minute, on airplanes (1)→(13) or at a rate of 5 degrees per minute on airplanes (14)→. On airplanes (1)→(13) sustained fore and aft accelerations will also cause erroneous pitch indications. After prolonged acceleration intervals the fast erect button should be depressed to rapidly remove these errors. It should also be noted that prolonged acceleration followed by a 90 degree turn will transfer any pitch error that may have developed into roll indication error. The directional gyro is limited to 85 degrees in pitch and roll on airplanes (1)→(13) and to 85 degrees in pitch on airplanes (14)→. Exceeding these pitch and roll limits may induce large heading errors. These errors are readily detectible by observing the synchronization meter on the compass control panel when operating the compass in the slaved mode. Manual resynchronization of the compass can be accomplished by depressing and rotating the synchronization set knob on airplanes (1)→(13) or by depressing the heading set knob on airplanes (14)→ to center the synchronization pointer between - and +. If manual synchronization is not accomplished, the directional gyro will slave to the synchronized position at a rate of 3.5 degrees per minute on airplanes (1)→(13) or at a rate of 1.5 degrees per minute on airplanes (14)→. Procedures for normal operation of the AFRS are contained in the appropriate portions of Section II.

PITOT-STATIC SYSTEM.

The airplane is equipped with a single pitot-static system which provides pitot and static pressures required for operation of standby instruments and the central air data computer (CADC). The system consists of the pitot-static tube, mounted on an adapter installed on the forward tip of the radome, and the tubing required for connection to the operating components. The tubing includes two sets of drains, and a static system manifold just forward of the instrument panel. Connections of both pitot and static pressures are made at the CADC unit and at the standby airspeed indicator. The other standby instruments, the altimeter, and the vertical velocity indicator are connected only to the static system. The pitot-static probe is equipped with a heating element for anti-icing. Refer to "Anti-icing and Defog Systems" this section. Pitot-static system instrument error and difference between primary and standby instruments are as follows:

Airspeed and Altimeter Instrument Errors

INSTRUMENT CHECK POINT	PRIMARY INSTRUMENT SYSTEM ACCURACY	STANDBY INSTRUMENT ACCURACY	MAX DIFFERENCE BETWEEN PRIMARY AND STANDBY
AIRSPEED (Knots)			
100	*±8	±3	11
150	±6	±3	9
300	±7	±10	17
400	±7	±15	22
600	±7	±20	27
ALTITUDE (Feet)			
0	*±50	±50	100
500	±50	±50	100
1000	±50	±70	120
4000	±50	±100	150
10,000	±50	±130	180
20,000	±50	±225	275
30,000	±70	±325	395
40,000	±80	±500	580

*CADC plus instrument error.

NOTE: 1. Instrument accuracies are based upon perfect pitot-static input information.

2. Maximum differences in altimeter readings are with both altimeters set to 29.92. For altimeter settings other than 29.92, add 40 feet to maximum altimeter differences.

Figure 1-20.

INSTRUMENTS.

The instruments consists of the total temperature indicator (TTI), true airspeed (TAS), standby instruments and the integrated flight instrument system (IFIS). Procedures for normal operation of the standby instruments and the integrated flight instrument system, are contained in the appropriate portions of Section II.

TOTAL TEMPERATURE INDICATOR.

The total temperature indicator (6, figure 1-5) located on the left main instrument panel, provides indications of aerodynamic heating. The indicator is an electrical resistance type instrument that uses a remote temperature sensing probe, an amplifier and a motor to position the indicator pointer. The total temperature sensing probe is equipped with a heating element for anti-icing. Refer to "Anti-Icing and Defog System," this section. The face of the indicator is graduated in 10 degree increments from -50 degrees to +250 degrees centigrade, with a critical temperature index mark of 153.3 degrees and a maximum temperature index mark at 214.3 degrees. A digital readout counter in the face of the indicator, marked SEC TO GO, indicates the time remaining for operation in the critical temperature range between 153.3 and 214.3 degrees. The counter will start to drive down from 300 seconds toward zero and an amber total temperature caution lamp will light when the critical temperature of 153.3 degrees is reached. The counter will continue to drive until one or more of the following conditions are met: until it reaches zero; until the temperature is reduced below 153.3 degrees, or until the maximum temperature index of 214.3 degrees is reached. When the maximum temperature index is reached or when the counter drives to zero, a red reduce speed lamp will light. The counter will reverse and drive back to 300 seconds any time the temperature falls and remains below 153.3 degrees. If the reduce speed warning lamp is on, it will go out as the counter starts to drive back. The total temperature caution lamp will go out when the counter has driven back to 300 seconds. An OFF flag will appear in the face of the indicator when power is removed from the instrument. The indicator operates on 115 volt ac power from the essential ac bus.

Total Temperature Caution Lamp.

The total temperature caution lamp (1, figure 1-5), located on the left main instrument, will light any time the airplane is operated above the critical temperature of 153.3 degrees centigrade. When lighted, the words TOTAL TEMP appear on the lamp face. Once lighted the lamp will remain on until the total temperature counter has reversed and driven back to 300 seconds.

Reduce Speed Warning Lamp.

The reduce speed warning lamp (1, figure 1-5), located on the left main instrument panel, functions in conjunction with the total temperature indicator to indicate that the airplane has flown for at least 300 seconds in the critical temperature range of from 153.3 to 214.3 degrees centigrade or that the maximum temperature index of 214.3 degrees has been reached or exceeded. When lighted the words REDUCE SPEED are visible in red on the face of the lamp. If the lamp was lighted due to the expiration of 300 seconds in the critical temperature range it will remain on until the temperature is reduced to below 153.3 degrees and the total temperature counter has reversed and started to drive back to 300 seconds. If the lamp was lighted upon reaching the maximum temperature index of 214.3 degrees as the counter was driving to zero it will go out as soon as the temperature is reduced below 214.3 degrees. On airplane (1)→(11) the intensity of the lamp can be controlled with the malfunction and indicator lamp dimming switch. On airplanes (12)→ the lamp cannot be dimmed. The lamp also functions in conjunction with the maximum safe mach assembly; refer to "Maximum Safe Mach Assembly" this section.

TRUE AIRSPEED INDICATOR.

The true airspeed indicator (4, figure 1-21), located on the right main instrument panel, provides a digital readout of true airspeed. The instrument displays true airspeed on a servo-driven 4-digit counter within the range of 40-1750 knots. The indicator is operated by electrical signals from the CADC.

STANDBY INSTRUMENTS.

The standby instruments include the airspeed indicator, altimeter, vertical velocity indicator, magnetic compass, attitude indicator and bearing distance heading indicator (BDHI). These instruments provide backup indications in the event of failure of the integrated flight instrument system.

Airspeed Indicator.

The airspeed indicator (2, figure 1-21), located on the right main instrument panel, is operated by pitot and static pressures direct from the pitot-static system. The instrument is graduated from 0.6 to 8 times 100 knots.

Altimeter.

The altimeter (5, figure 1-21), located on the right main instrument panel, is a barometric type which operates on static pressure direct from the pitot-static system. A barometric pressure set knob located on the left corner of the instrument provides a means of adjusting the barometric scale on the instrument.

Vertical Velocity Indicator.

The vertical velocity indicator (12, figure 1-21), located on the right main instrument panel, provides rate of climb and descent information. The instrument operates on static pressure from the pitot-static system.

Magnetic Compass.

The magnetic compass (figure 1-2), located on the windshield center beam, provides magnetic heading information. A deviation correction card for the compass is located below the center of the glare shield.

Attitude Indicator.

The attitude indicator (11, figure 1-21), located on the right main instrument panel, provides backup attitude information in the event of malfunction or failure of the attitude director indicator. The indicator displays pitch and roll information on an attitude sphere in relation to a miniature aircraft. Pitch and roll signals are received from the auxiliary flight reference system (AFRS). The indicator receives 115 volt ac power from the ac essential bus. In the event of power failure or an AFRS malfunction, an OFF warning flag will appear on the lower left face of the indicator. A pitch trim knob on the lower right side of the instrument is provided to adjust the attitude sphere to the proper pitch attitude.

Bearing-Distance-Heading Indicator.

The bearing-distance and heading indicator (BDHI) (3, figure 1-21), is located on the right main instrument panel. The instrument is a remote type heading indicator with a rotating compass card. Automatic direction finding (ADF) and TACAN bearing information is displayed by means of pointers. A synchro driven range indicator is provided which receives signals from the TACAN set. Range of the distance display is 0 - 999 nautical miles. A red and black striped range warning flag partially obscures the range indicator when distance-to-station signals are too weak or there is a loss of lock-on to TACAN distance signals. Magnetic heading of the airplane is shown by the index at the top of the instrument and the compass card. A pointer designated as number one is servo driven and receives signals from a TACAN coupler. Bearing information is read from the compass card under the pointer tip. A pointer designated as number 2 is also servo driven and, when required, receives signals by selection from the ADF set. When ADF signals are desired, the function selector knob on the UHF radio control panel is positioned to ADF. When deenergized, the pointer is positioned concurrent with the number one pointer and rotates with it. The BDHI receives heading information from the auxiliary flight reference system (AFRS). The set index knob located on the lower right side of the indicator is used to set the heading index to a desired magnetic heading. Once set, the index rotates with the compass card. A flag marked OFF will appear in the window when the BDHI is not energized or when power is not available to the compass card.

INTEGRATED FLIGHT INSTRUMENT SYSTEM.

The integrated flight instrument system takes outputs from the following systems and integrates them into usable displays on the integrated flight instruments.

- Central air data computer (CADC)
- Auxiliary flight reference system (AFRS)
- Instrument landing system (ILS)
- Tactical air navigation system (TACAN)
- Terrain following radar (TFR)
- Radar homing and warning system (RHAWS)
- Bomb nav system
- Attack radar system
- Radar altimeter
- Lead computing optical sight system (LCOSS)
- Dual bombing timer (DBT)
- Shrike

The primary components of the system are the integrated flight instruments; consisting of the airspeed-mach indicator (AMI), altitude-vertical velocity indicator (AVVI), attitude director indicator (ADI) and horizontal situation indicator (HSI), a flight director computer (FDC) and an instrument system coupler (ISC). The four integrated flight instruments are grouped together on the left main instrument panel to provide actual and command flight and navigational information in a clear concise manner. Altitude, airspeed, acceleration mach, vertical velocity, and angle of attack are displayed on moving tapes on the AMI and AVVI. The ADI and HSI display attitude, heading and navigational information from various other systems in the airplane. The lead computing optical sight (LCOS) command steering bars operate in conjunction with the system to provide the same pitch and bank steering commands as the ADI. The instrument system coupler accepts inputs from other airplane systems and channels them through the flight director computer for display on the ADI, LCOS and HSI. The system incorporates self test features to check reliability and isolate malfunctions. The system operates on 115 volt and 26 volt ac power from the essential ac bus and 28 volt dc power from the essential dc bus.

Airspeed Mach Indicator.

The airspeed mach indicator (AMI) (5, figure 1-5), located on the left main instrument panel, provides remote reading vertical presentations of angle of attack, "g" acceleration, mach and airspeed on vertical moving scales. Readout windows below each moving scale present digital values for "g" acceleration, mach, and airspeed. Slewing switches for setting reference mach and airspeed markers are located on the bottom of the indicator. Signals for operation of the various scales are provided from the central air data computer and remote accelerometer. In the event of a malfunction or power failure a spring-loaded OFF warning flag will appear across the face of the mach scale. Presentations on the face of the indicator are from left to right.

Angle-of-Attack Indicating System.

Note

> Until flight test data is available to incorporate CADC correction cams, the angle of attack indicator and indexers will display uncalibrated rather than true angle-of-attack.

The angle-of-attack indicating system provides an indication of the angular position of the wing chord in relation to the airplane flight path. This indication is used for approach speed monitoring and to warn of an approaching stall. The system includes a vane type transmitter, indicator, and two sets of indexers. The indicator and indexers are electrically slaved to the sensor vane transmitter. In flight, the vane, which is located on the left side of the fuselage, will align itself with the airflow. Rotation of the vane generates an indicated angle-of-attack signal to the central air data computer. The central air data computer converts the signal to true angle-of-attack and sends this signal to the angle-of-attack indicator and also lights the appropriate angle-of-attack indexer. A damper assembly prevents rotational overshoot and flutter of the vane due to turbulence. Vane anti-icing is provided by means of a 115 volt ac heating element in the leading edge of the vane. The heating element receives power from the left main ac bus and is controlled by the pitot heat switch.

Angle-of-Attack Indicator. The angle-of-attack indicator, located on the airspeed-mach indicator (5, figure 1-5), indicates in degrees the angular position of the wing chord in relation to the airplane flight path. The vertically moving tape displays angle of attack from minus 10 degrees to plus 25 degrees. The angle-of-attack indicator is operated by means of synchro signal received from the central air data computer.

Angle-of-Attack Indexer. An angle-of-attack indexer (3, figure 1-5, and 14, figure 1-21) is located on either side of the glare shield. Each indexer consists of 3 red lamps arranged vertically. The low speed symbol (top V-shaped lamp) lights when the angle of attack exceeds 8 degrees. The on speed symbol (center donut-shaped lamp) lights between 8.5 and 6.5 degrees. The high speed symbol (bottom inverted V-shaped lamp) will light when the angle-of-attack is less than 7 degrees. The indexer lamps function only when the landing gear is in the down position. A dimming rheostat, located on the side of the indexer controls the intensity of the lamps which receive 28 volt dc power from the main dc bus.

Accelerometer. The accelerometer located adjacent to the angle-of-attack indicator, provides normal G (load factor) information. The G forces being sustained by the aircraft are continuously shown by the acceleration scale read against a fixed index line. The tape scale is graduated from -4 to +10 Gs. The presentation on the digital readout is from 0.0 to 9.9 G's. The accelerometer and readout window are actuated by electrical signals from the remote accelerometer.

Mach Indicator. The mach scale in the center of the airspeed-mach indicator indicates true mach number which is shown on a moving scale and is read against the fixed index. The scale is calibrated in hundredths and shows numbers in tenths from 0.4 through 3.5. At speeds below mach 0.4, the scale will continue to read 0.4. The moving scale is operated by electrical signals from the CADC. A command mach marker and command mach readout window indicate manually selected command mach. The command mach marker remains at the top or bottom of the display column until the selected command mach comes into view on the mach scale, at which time it will synchronize and move with the scale. The selected true mach is numerically displayed in the command mach readout window at all times. Command mach setting is controlled manually by the command mach slewing switch under the command mach readout window. When selecting a command mach number, slewing speed is proportional to the amount the slewing switch is displaced from its normal center position. The maximum allowable mach is indicated by a diagonally-striped maximum allowable mach marker which normally rests at the bottom of the mach scale. When maximum allowable speed is approached, the marker will climb toward the fixed index line. The maximum allowable mach marker will show on the scale depending on the airplane configuration, air density, and temperature. The maximum allowable mach marker is operated by electrical signals from the maximum safe mach assembly (MSMA).

Airspeed Indicator. The airspeed scale on the right column of the airspeed-mach indicator indicates airspeed on a moving scale read against a fixed index. The scale is calibrated in 10 knot increments and displays numerals at each 20 knot interval from 100 to 200 knots and each 50 knot interval from 200 through 1000 knots. At speeds below 50 knots, the scale will continue to read 50. The airspeed scale is operated by electrical signals from the CADC. If there is an airspeed signal failure from the CADC, the IAS monitoring flag marked OFF will appear across the airspeed scale. A command airspeed marker and a command airspeed readout window below the scale indicates selected command airspeed. Command airspeed setting is controlled by the command airspeed slewing switch under the command airspeed readout window. When selecting a command airspeed, slewing speed is proportional to the amount the slewing switch is displaced up or down from the center position. Once the command airspeed is set into the command airspeed readout window, the command airspeed marker remains at the top or bottom of the display column until the selected command airspeed comes into view on the moving scale, at which time it will synchronize and move with the reading on the scale. This will be the same reading as shown in the readout window.

Note

> If the slewing switch is moved to the detented position on the right, the command airspeed marker will align with the fixed index and continuous digital presentation of the airspeed will then be displayed on the moving scale and in the readout window.

Altitude-Vertical Velocity Indicator.

The altitude-vertical velocity indicator (AVVI) (18, figure 1-5), located on the right main instrument panel, provides remote reading presentations of altitude and vertical velocity on vertical moving scales. Readout windows across the bottom of the indicator present digital readout of barometric pressure and command altitude. A barometric pressure set knob and command altitude slewing switch are also located on the bottom of the indicator. Signals for operation of the moving scales, markers and readouts are provided from the CADC. A spring-loaded OFF warning flag will appear across the face of the coarse altitude scale in the event of malfunction or power failure to the indicator. The barometric pressure reading is set by a knob marked BARO located on the lower left corner of the indicator and is numerically displayed in the barometric pressure readout window above the knob.

Note

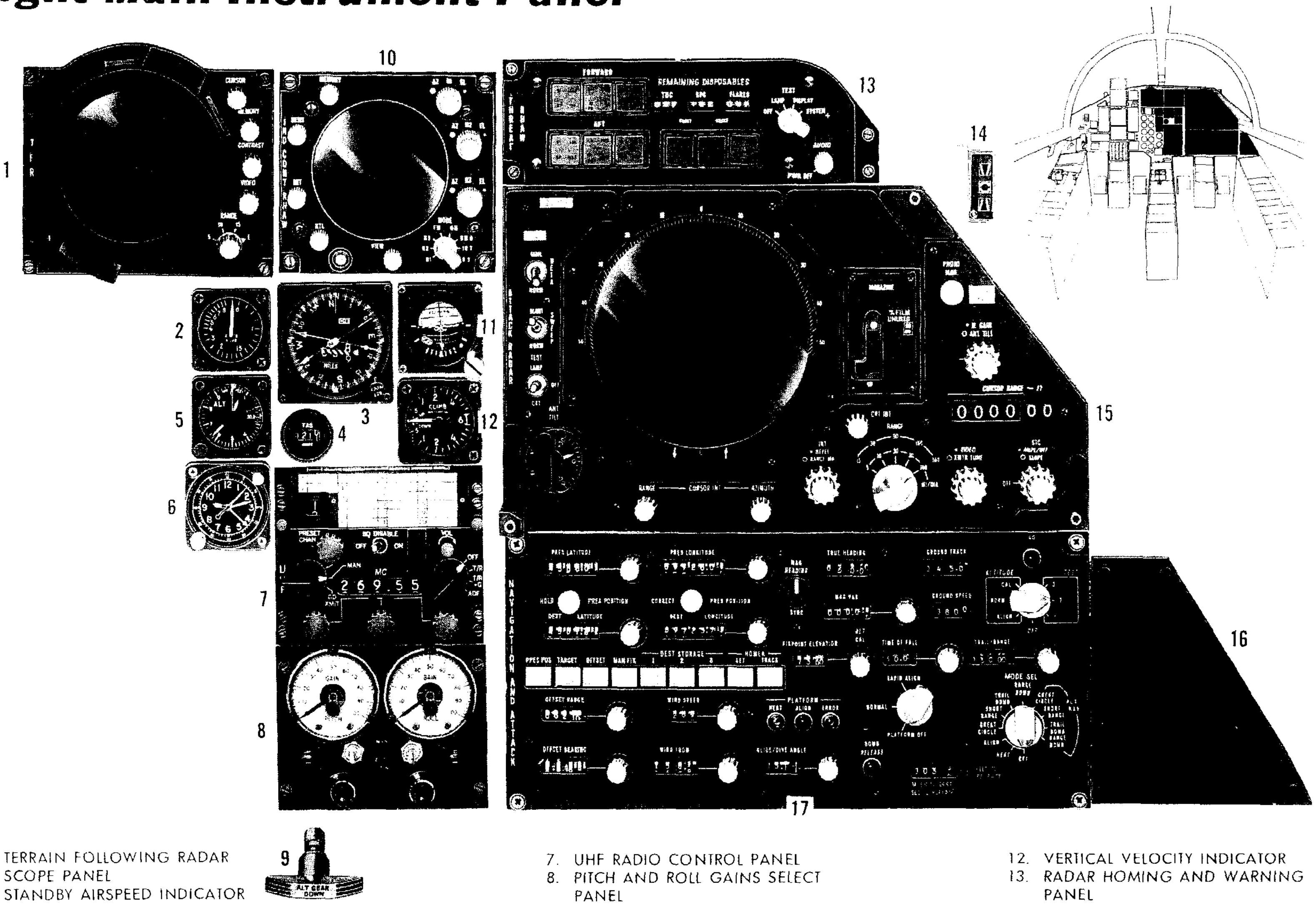
> A mechanical failure within the altitude-vertical velocity indicator may not cause the flag to appear even though the indicator reading will be unreliable. If a failure is suspected, rely on the standby altimeter.

Presentations on the face of the indicator are from left to right as follows:

Vertical Velocity Indicator. The vertical velocity indicator is located on the left side of the altitude-vertical velocity indicator. The instrument indicates climb or dive velocities from 0 to 1500 feet per minute by means of a moving index pointer to the right of a vertical fixed scale. The scale is graduated in increments of one hundred feet from 0 to 1.5 thousand. When the vertical velocity exceeds this scale the pointer index will move to the top or bottom of the instrument to a readout window where a moving scale, graduated in thousands of feet from 2 to 40 thousand feet per minute, will indicate the rate of climb or descent. The instrument receives information from the CADC.

Vernier Altimeter. The altitude scales in the center of the altitude-vertical velocity indicator indicate aircraft pressure altitude which is read on the alti-

Right Main Instrument Panel



1. TERRAIN FOLLOWING RADAR SCOPE PANEL
2. STANDBY AIRSPEED INDICATOR
3. BEARING-DISTANCE HEADING INDICATOR
4. TRUE AIRSPEED INDICATOR
5. STANDBY ALTIMETER
6. CLOCK
7. UHF RADIO CONTROL PANEL
8. PITCH AND ROLL GAINS SELECT PANEL
9. LANDING GEAR EMERGENCY RELEASE HANDLE
10. RADAR HOMING AND WARNING SCOPE PANEL
11. STANDBY ATTITUDE INDICATOR
12. VERTICAL VELOCITY INDICATOR
13. RADAR HOMING AND WARNING PANEL
14. ANGLE-OF-ATTACK INDEXER
15. ATTACK RADAR SCOPE PANEL
16. NOT USED
17. BOMB NAV CONTROL PANEL

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Figure 1-21.

tude scales against a fixed index line. The vernier scale is calibrated in 50 foot graduations and indicates each hundred foot level from 0 to 1000 feet. The coarse scale is calibrated in 500 foot graduations and indicates each thousand foot level from -1000 through +120,000 feet. Both the vernier and coarse scales are operated by electrical signals from the CADC. A command altitude marker and the command altitude readout window below the scale indicate manually selected command altitude. The command altitude numerals are controlled manually by the command altitude slewing switch under the command altitude readout window. When selecting a command altitude, slewing speed of the command marker and readout window numerals is proportional to the amount the slewing switch is displaced from center. The command altitude remains at the top or bottom of the display column until the selected command altitude comes into view on the altitude scale, at which time it will synchronize and move with the scale. The selected command altitude is numerically shown in hundreds in the altitude readout window at all times.

Gross Altimeter. The gross altimeter located on the right side of the altitude-vertical velocity indicator is a thermometer-type altitude index which shows aircraft altitude against a gross altitude scale. It is operated by electrical signals from the CADC. The gross altitude scale is calibrated in thousands of feet and numerically indicates 10,000 foot levels from 0 to 120,000 feet. Command altitude is indicated by a double line command altitude marker and is simultaneously shown and operated in conjunction with the command altitude marker on the vernier altimeter.

Attitude Director Indicator.

The attitude director indicator (ADI) (12, figure 1-5), located on the left main instrument panel, is a remote indicating instrument which displays attitude, heading, turn and slip, glide slope deviation, "g" deviation, and bank and pitch steering information. The indicator includes an attitude sphere, turn and slip indicator, pitch and bank steering bars, miniature aircraft, glide slope indicator, warning flags and a pitch trim knob. The attitude sphere displays pitch, bank and heading in relation to the miniature aircraft. These signals are received directly from either the bomb nav system stabilization platform or the auxiliary flight reference system depending on the position of the flight instrument reference select switch. The pitch reference of the attitude sphere to the miniature aircraft may be adjusted with the pitch trim knob. The turn and slip indicator, located in the bottom of the ADI, receives turn signals directly from a remotely located rate-of-turn transmitter and is designed for a 4 minute turn. Pitch and bank steering commands from other systems are processed by the instrument system coupler and routed through the flight director computer to the pitch and bank steering bars and glide slope deviation indicator. (Refer to "Instrument System Coupler Mode Selector Knob" and "Pitch Steering Selector Switch," this section, for ADI indications during various modes of operation). An OFF warning flag indicates loss of power to the ADI or an AFRS malfunction, other warning flags indicate the loss of signal to the bank steering bar and loss of signal to the glide slope deviation indicator.

CAUTION

The attitude warning flag will not appear with a slight electrical power reduction or a failure of other components within the system. Failure of certain components can result in erroneous or complete loss of pitch and bank presentations without a visible flag.

The ADI operates on 115 volt ac power from the essential ac bus.

Horizontal Situation Indicator.

The horizontal situation indicator (HSI) (13, figure 1-5), located on the left main instrument panel, is a remote indicating instrument which displays course, heading, distance and bearing information. The indicator includes a compass card, course and heading set knobs, course arrow, to-from indicator, lubber lines, bearing pointer, course deviation indicator and scale, range indicator and course selector windows, warning flags and an aircraft symbol. The compass card is servo driven and receives magnetic heading signals directly from either the bomb nav system or auxiliary flight reference system depending on the position of the flight instrument reference select switch. Airplane heading or its reciprocal are read under an upper and lower lubber line. The aircarft symbol is fixed and is oriented to the nose of the airplane. A heading set knob is provided to set a heading marker to the desired heading. Once it is set the marker rotates with the compass card. A course set knob is provided to set the course arrow and digits in the course selector window to the desired course. Once set, the arrow will rotate with the compass card. The shaft of the course arrow provides course deviation indications. The reciprocal course may be read off the tail of the arrow. An unreliable course signal or loss of the course signal to the indicator will cause a warning flag to appear in the upper center of the indicator. The bearing and distance to TACAN stations are displayed by the bearing pointer and range indicator window. The to-from indicator indicates the relative bearing from the airplane to the selected station. Loss of the TACAN signal or an unreliable signal will cause a range warning flag to appear in the range indicator window. Loss of power to the HSI will cause an OFF warning flag to appear on the right side of the instrument. (Refer to "Instrument System Coupler Mode Selector Knob," this section, for HSI indications during various modes of operation). The HSI operates an 115 volt ac power from the ac essential bus.

Instrument System Coupler Pitch Steering Mode Switch.

The instrument system coupler pitch steering mode switch, located on the instrument system coupler control panel (8, figure 1-5), is a three position switch marked ALT REF (altitude reference), OFF and TF (terrain following). The switch is solenoid held in either the ALT REF or TF position, when used with a compatible position of the instrument system coupler mode selector knob. When the switch is placed in the ALT REF position, pitch steering commands, referenced to the pressure altitude at the time the switch is engaged, will be displayed on the pitch steering bars on the attitude director indicator (ADI) and lead computing optical sight (LCOS). The ALT REF position is compatible with all positions of the instrument system coupler mode selector knob except AIR/AIR, however, when making an ILS or AILA approach the switch will automatically return to OFF when the glide slope is intercepted. When the switch is placed to the TF position pitch steering commands referenced to the altitude setting of the terrain following radar will be displayed on the pitch steering bars on the ADI and LCOS. The TF position is compatible with all positions of the instrument system coupler mode selector knob except ILS, AILA and AIR/AIR. However, with the knob in CRS SEL NAV, NAV and SHRIKE position, the switch will return to OFF when a pull-up signal is generated by the armament system.

Instrument System Coupler Mode Selector Knob.

The instrument system coupler mode selector knob, located on the instrument system coupler control panel (8, figure 1-5) has twelve positions. Nine positions of the knob are activated and are marked ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS, MAN HDG, AIR/AIR and SHRIKE. Three unmarked positions provide space for the installation of new equipment. The knob must be depressed to change positions. The knob positions provide the following functions:

- The ILS (instrument landing system) position provides the capability of making letdowns and approaches to runways equipped with localizer and glide slope transmitters. Localizer steering commands are displayed by the bank steering bars on the attitude director indicator (ADI) and lead computing optical sight (LCOS) and course deviation information is displayed on the course deviation indicator on the horizontal situation indicator (HSI). Pitch steering commands will be displayed on the pitch steering bars on the ADI and LCOS if the pitch steering mode switch is in the ALT REF position. When the glide slope beam is intercepted the pitch steering mode switch, if on, will return to OFF and glide slope steering commands will then be displayed on the pitch steering bars on the ADI and LCOS and glide slope deviation will be displayed on the glide slope deviation indicator on the ADI. If the radar altimeter is operating and is set for a minimum altitude penetration, the pitch steering bars on the ADI and LCOS will indicate a fly-up command when this altitude is reached. If a pull-up is initiated the fly-up command can be terminated when the airplane is above minimum penetration altitude by one of the following methods:

  By momentarily depressing the instrument system coupler mode selector knob.

  By placing the instrument system coupler mode selector knob to another mode.

  By placing the instrument system coupler pitch steering mode switch to ALT REF when level off altitude is reached.

- The AILA (airborne instrument landing and approach) position provides the capability of making instrument letdowns and approaches to runways not equipped with ground based letdown systems. The bomb nav and attack radar systems furnish simulated localizer and glide slope information to provide the same indications on the ADI, LCOS and HSI as when using the ILS position.

- The TACAN (tactical air navigation) position provides the capability of making instrument letdowns and flying a selected course to or from a TACAN station. Course steering commands are displayed on the bank steering bars on the ADI and LCOS and course deviation information is displayed on the course deviation indicator and bearing pointer on the HSI. Distance from the TACAN station is displayed in the range indicator window on the HSI. The bearing pointer will indicate the direction to the station.

- The CRS SEL NAV (course select navigation) position provides the capability of approaching a bomb nav system computed destination along a manually selected course other than the most direct route. This may be used to avoid weather, obstacles or sensitive enemy areas. The pilot sets the HSI course pointer and course selector window to the desired course using the course set knob. This establishes a course error signal to the bomb nav system to provide a steering command to the bank steering bars on the ADI and LCOS and course deviation information to the course deviation indicator on the HSI.

- The NAV (basic navigation) position provides course information from the bomb nav system when it is used in any one of four modes of operation. When the bomb nav mode selector knob is in either the GREAT CIRCLE, SHORT RANGE, BOMB TRAIL or BOMB RANGE positions, course steering commands to a destination set into the bomb nav system are displayed by the bank steering bars on the ADI and LCOS and course deviation is displayed on the course deviation indicator on the HSI. If the bomb nav system is inoperative, course information can be obtained from the auxiliary flight reference system (AFRS) by placing the bomb nav mode selector knob to AUX NAV in one of the above positions. Distance to destination or time/distance to target and mode position of the bomb nav mode selector knob are displayed on the bomb nav time distance indicator (BNTDI).

- The MAN CRS (manual course) position provides the capability of flying a manually selected course instead of a bomb nav system computed course. This position can be utilized to fly a constant course while taking a fix, changing destination or working a navigation problem. The compass card is set to the desired course to be flown, using the course set knob and course selector window. The selected course is compared with actual course by the bomb nav system and an error signal is provided to display course steering commands on the bank steering bars on the ADI and LCOS and course deviation information on the course deviation indicator on the HSI.

- The MAN HDG (manual heading) position provides the capability of flying any desired heading when use of the bomb nav system is impractical or inefficient or when the system is inoperative. The heading marker on the HSI is set to the desired heading on the compass card by using the heading set knob and then turning the airplane to the desired heading. Any deviation from this heading will generate a steering command on the bank steering bars on the ADI and LCOS. If the bomb nav system is inoperative the course set knob should be used to set the desired heading in the course selector window. This will provide a digital readout of the heading and align the course arrow with the heading marker to reduce the possibility of heading confusion.

- The AIR/AIR position provides the steering capability to a target being tracked by the attack radar system. In this mode the HSI heading marker is driven by a bearing signal from the attack radar and provides a signal to indicate the necessary steering commands on the bank steering bars on the ADI and LCOS to steer the airplane to the target. The pitch steering bars on the ADI and LCOS will be activated and indicate the necessary pitch steering correction (airplane angle of attack plus radar antenna tilt angle) to be on target.

- The SHRIKE position provides the steering capability for making attacks on ground radar transmitters. In this mode the shrike receiver furnishes direction finding and pitch deviation information to provide target deviation indications on the ADI and LCOS. The pitch steering bars will display "g" command steering signals when a pull-up condition is desired.

Instrument Test Button.

The instrument test button (5, figure 1-16), located on the aft console, is provided for ground checking and trouble shooting of the integrated flight instruments, the instrument system coupler, and the total temperature indicator. The button must be used in conjunction with the central air data computer (CADC) power switch when checking the airspeed-mach indicator (AMI), altitude-vertical velocity indicator (AVVI), or the total temperature indicator (TTI). Depressing and holding the button will provide a set of predetermined indications on the above instruments. (Refer to "CADC Test Switch," this section, for AMI, AVVI and TTI test indications.) Test indications on the ADI and HSI will be compatible with the normal indications expected for each mode selected by the instrument system coupler mode selector knob.

WARNING CAUTION AND INDICATOR LAMPS.

In order to keep instrument surveillance to a minimum, warning, caution, and indicator lamps are located throughout the cockpit. All of these lamps except the master caution lamp are described under their respective systems. For a grouping of lamps located on the main caution lamp panel, and on, or adjacent to, the aircraft commanders left main instrument panel. See figure 1-21A.

MASTER CAUTION LAMP.

A shutter type master caution lamp (17, figure 1-5), located on the left main instrument panel, will light to alert the crew that a malfunction exists when any of the individual caution lamps on the caution lamp panel light to indicate a malfunction. The lamp will remain lighted as long as an individual caution lamp is on; however, it should be reset as soon as possible by depressing the face of the lamp so that other caution lamps can be monitored should additional malfunctions occur. The intensity of the lamp cannot be adjusted. The lamp can be checked by depressing the malfunction and indicator lamp test button.

Malfunction and Indicator Lamp Dimming Switch.

The malfunction and indicator lamp dimming switch (2, figure 1-42), located on the lighting control panel, is a three position switch marked BRT (bright) and DIM and is spring-loaded to an unmarked center position. The switch controls the light intensity, either bright or dim, of all the warning, caution and indicator lamps in the cockpit except the master caution lamp. The lamps are automatically set to bright when: the internal lighting control knob (FLT INST) is off; the lighting control knob (WHITE FLOOD FLT & ENG INST) is on; or when aircraft power is turned off. On airplanes 12→ the radar altitude low warning lamp, reduce speed warning lamp, canopy unlock warning lamp, cabin pressure warning lamp and the nose wheel steering/air refueling indicator lamp are shutter type lamps that cannot be dimmed.

Malfunction and Indicator Lamp Test Button.

The malfunction and indicator lamp test button (3, figure 1-42), located on the lighting control panel, is provided to check all warning, caution and indicator lamps in the cockpit for burned out bulbs and the landing gear warning horn.

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Warning, Caution and Indicator Lamps

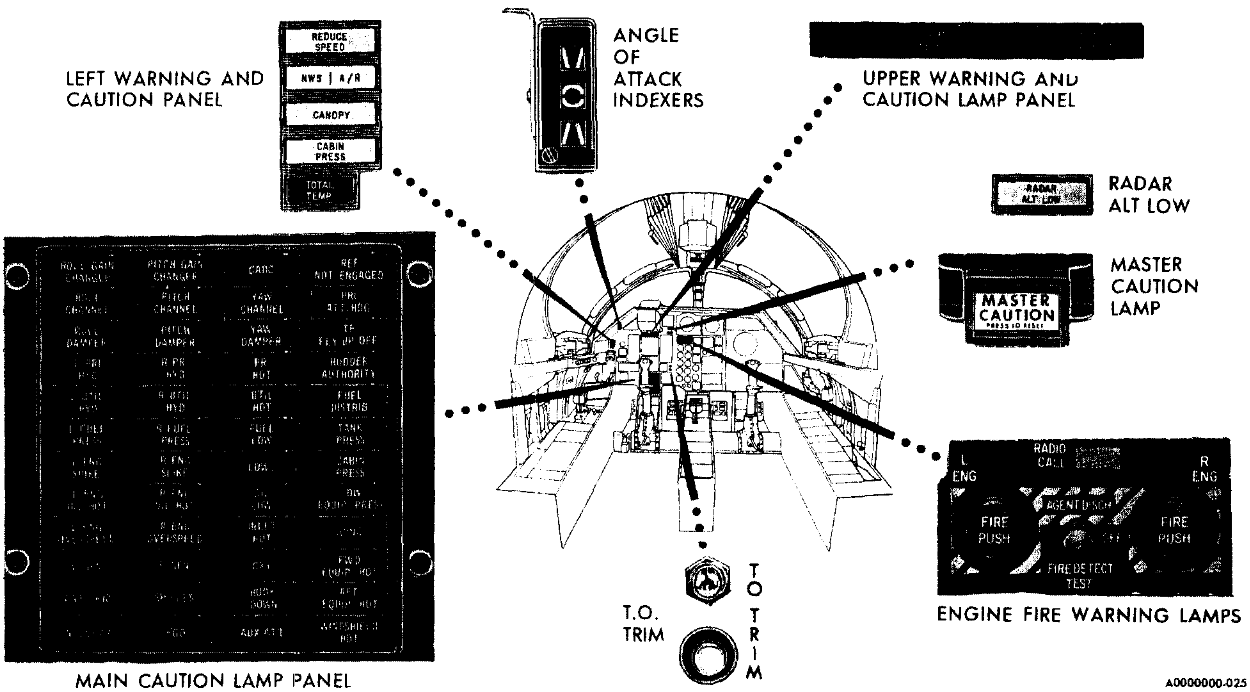


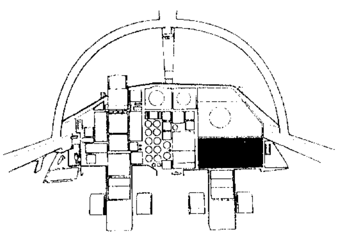
Figure 1-21A.

BOMBING-NAVIGATION SYSTEM (AN/AJQ-20).

The bombing-navigation (bomb nav) system is a self-contained dead reckoning analogue inertial system. The system consists of a stabilization platform (SP), a navigational computer (NC), a remotely located flux valve and a remotely located bomb nav distance time indicator. The system provides the following functions:

- Computed aircraft position in latitude and longitude.
- Range and bearing to target or destination for navigation steering and/or bombing.
- Continuously computed and displayed values of ground speed, ground track, true heading, wind speed, and wind direction.
- Aircraft pitch and roll attitude.
- A stabilized magnetic heading to the pilot's flight instruments.
- Automatic steering signals to the attitude director indicator, the horizontal situation indicator, and the lead computing optical sight for navigation and homing.
- Constant ground track steering signals to autopilot.
- Drift Angle.
- Slant range and bearing to a fixpoint for attack radar crosshair laying.
- Provisions for attacking no-show radar targets by off-set radar sighting.
- Position correction via radar fix-taking.
- Determination of the coordinates of unknown radar locations detected by the homing and warning radar.
- Pushbutton fly over fix-taking capability for present position correction.
- Altitude calibration by use of attack radar or radar altimeter.
- Up to three alternate or intermediate destination storages. New destinations may be inserted into storages at any time by the operator.
- The capability of airborne instrument landing or dive bombing in conjunction with the horizontal situation indicator.
- A backup capability in case of stabilization platform failure.

Bomb Nav Control Panel



1. PRESENT POSITION HOLD BUTTON
2. PRESENT POSITION COUNTERS (2)
3. PRESENT POSITION CORRECTION BUTTON
4. MAGNETIC HEADING SYNCHRONIZATION INDICATOR
5. FIXPOINT ELEVATION COUNTER
6. TRUE HEADING COUNTER
7. MAGNETIC VARIATION COUNTER
8. TIME-OF-FALL COUNTER
9. GROUND TRACK AND GROUNDSPEED COUNTERS (2)
10. GO LAMP
11. ALTITUDE/TEST SELECTOR KNOB
12. TRAIL/RANGE COUNTER
13. BOMB NAV MODE SELECTOR KNOB
14. DESTINATION DISTANCE/TIME COUNTER
15. PLATFORM ALIGNMENT CONTROL KNOB
16. BOMB RELEASE LAMP
17. GLIDE/DIVE ANGLE COUNTER
18. PLATFORM INDICATOR LAMPS (3)
19. WIND SPEED AND WIND FROM COUNTERS (2)
20. BOMB NAV FIX MODE SELECTOR BUTTONS (9)
21. OFFSET RANGE AND OFFSET BEARING COUNTERS (2)
22. DESTINATION POSITION COUNTERS

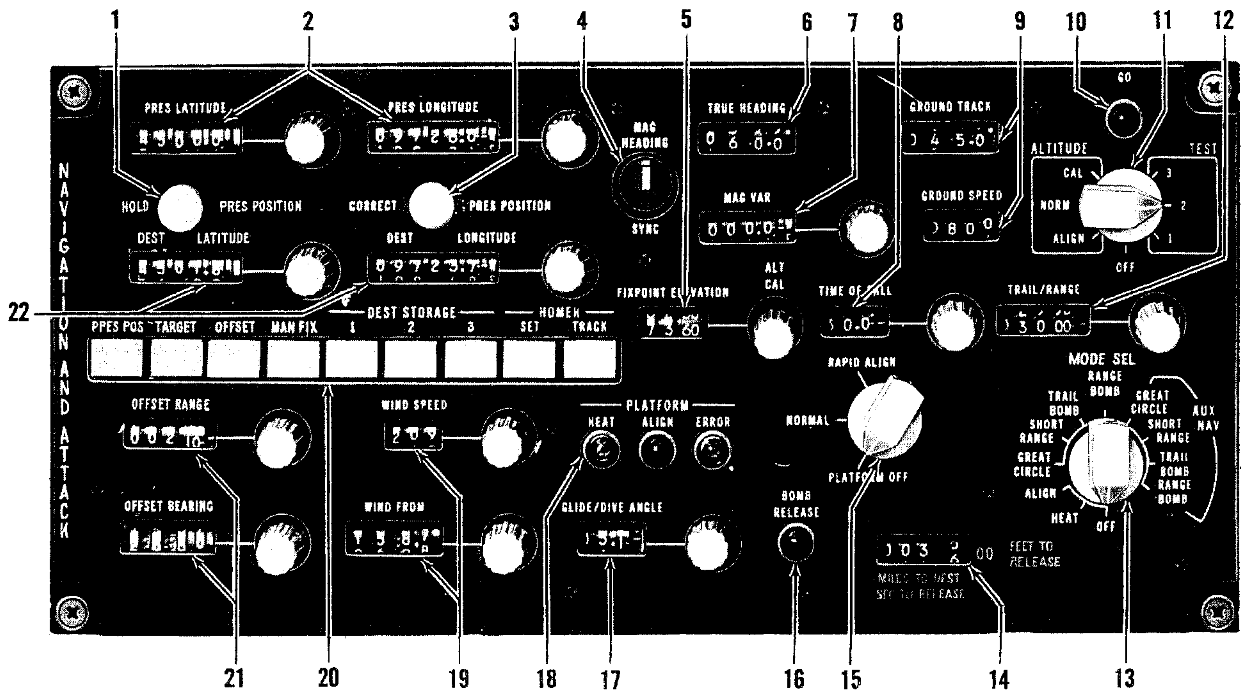


Figure 1-22.

- Simple self-test features to isolate system troubles to the stabilization platform or navigational computer while the system is still installed in the airplane.

STABILIZATION PLATFORM.

The stabilization platform (SP), located in the forward electronics bay, consists of a four-gimbal, all attitude inertially stabilized platform, and its associated electronics. The SP supplies outputs of pitch, roll, true heading, and north and east components of ground speed. Additionally, signals are provided to indicate (1) progress of initial alignment, (2) proper range of SP gyroscope temperatures, and (3) reliability of SP output data. Prior to flight, the SP is initially aligned by one of three methods: (1) Gyrocompass alignment (the normal method), (2) Alignment to stored magnetic variations, (3) Rapid alignment to stored gyrocompass heading. In normal gyrocompass alignment, the output signals of the inertial sensors (two gyroscopes with two degrees of freedom and two linear accelerometers) are utilized to drive the platform to a true north and plumb-bob level orientation. The orientation is such that the "East" gyroscope senses no angular rotation input (earth-rate) resulting from the earth's rotation and neither accelerometer senses any acceleration effects of the earth's gravity. Both of the other two methods of alignment are rapid alignment modes. Either of these two modes may be used when the correct aircraft true heading has been stored in the navigational computer (NC) from a previous operation or when the correct magnetic variation is known. These two modes bypass the relatively slow gyrocompass process to determine true north, and brings the SP to a ready condition in 90 seconds or less. After alignment, the SP is placed in the appropriate navigation mode. In navigational modes, the platform is maintained in a north-stabilized plumb-bob level orientation by signals from the gyroscopes, which are precessed by precisely computed signals to compensate for the earth's spin rate and aircraft movement relative to the earth. Accelerometers, mounted in the horizontal plane on the platform, are aligned so that one senses only north-south accelerations and the other senses only east-west accelerations. Their outputs after correction for Coriolis and centripetal effects, are then integrated to obtain signals proportional to instantaneous north and east velocities. These signals are used to develop gyroscope precession signals in the SP and also to generate ground track data, ground speed data and update aircraft position information by the NC. All the critical signal loops within the SP are constantly monitored by a go-no-go circuit which, in event of SP failure, turns the SP off and signals the operator by lighting the platform error lamp on the bomb nav control panel. In addition, automatic temperature controls signal the operator when temperatures in the SP are below the required level. When this condition exists, the SP is not required to perform to full accuracy. The SP requires inputs of 115 volt 3 phase ac power from the right main AC bus, 28 volt dc mode control signals from the NC, and synchro analogue data from the NC corresponding to aircraft latitude.

NAVIGATIONAL COMPUTER.

The navigational computer (NC), located in the right main instrument panel, is a self-contained navigation steering, radar sighting and bombing computer. The primary inputs to the NC are north and east components of ground speed, true heading and pitch angle from the stabilization platform (SP), true air speed and pressure altitude from the central air data computer (CADC) and magnetic heading from the system flux valve. The NC provides all the computing, control, and display functions for the bomb nav system. In event of SP failure the NC will continue to operate using computed or handset wind combined with true airspeed from the CADC and magnetic heading from the auxiliary flight reference system (AFRS) to substitute for SP data.

CONTROLS AND INDICATORS.

Bomb Nav Mode Selector Knob.

The eleven position bomb nav mode selector knob (13, figure 1-22), located on the bomb nav control panel is labeled MODE SEL. The knob controls warm-up, power turn on, and system operating modes. By rotating the knob clockwise, the 7 and 8 o'clock positions set the system up for operation. The 9 through 12 o'clock positions provide normal navigation modes and the 1 through 4 o'clock positions provide auxiliary navigation modes. The knob must be rotated counterclockwise to turn the system off. This knob has detents at all positions and requires a pull-out to rotate from ALIGN to any normal navigation position and from the normal navigation range to any auxiliary navigation position. The knob markings and functions are as follows:

1. OFF - All power off.
2. HEAT - All power off except for inertial platform heater power - provided the platform alignment control knob is in any position except PLATFORM OFF.
3. ALIGN - All power on and inertial platform sequenced through alignment cycle - provided the platform alignment control knob is in any position except PLATFORM OFF. Computer receives power and performs align mode functions.
4. GREAT CIRCLE - Normal navigation operating mode, used for ranges in excess of 200 nautical miles, in which the range and course computers solve for the Great Circle route from the geographic position indicated by the present position counters to the geographic position indicated by the destination position counters. In this mode, the radar sighting computer is inoperative.
5. SHORT RANGE - Normal navigation operating mode, used only within 200 nautical miles of the target, in which the computer assumes a flat-earth condition to compute range and course from the geographic positions indicated by the present position counters to the geographic position indicated by the destination position counters. In this mode, the radar sighting computer is also operative.
6. TRAIL BOMB - This attack mode is primarily a high altitude weapon delivery mode which utilizes the computer in the short range configuration to develop the bombing equation. In this mode, the radar sighting computer is also operative. Trail and time-of-fall parameters are handset and the bomb release lamp automatically lights when the computed time to release is zero. Time to release is continuously displayed on both the destination distance/time counter and the bomb nav distance/time indicator.
7. RANGE BOMB - This attack mode is a tactical mode used primarily for low-altitude weapon delivery. The computer operates in the basic short range mode and continuously displays the range to the release point, which

is the intersection of the course to target line with the release circle, which is a circle centered on the target with the radius selected by the operator for the particular weapons. The bomb release lamp lights when the pre-selected release range is reached. In this mode, the radar sighting computer is also operative.

8. AUX/NAV - During the auxiliary navigation modes, the computer display unit is used as a dead reckoning computer and performs essentially the same functions as during the corresponding normal navigation modes except the inputs from the stabilization platform are replaced by flux valve compass heading from the auxiliary flight reference system (AFRS), true airspeed input from the air-data computer, and handset magnetic variation and wind information. Also, during auxiliary navigation modes, the bomb nav system attitude ready signal is removed, causing other systems to be switched to the AFRS for attitude/heading reference data, and the primary attitude/heading caution lamp is lighted.

Bomb Nav Fix Mode Selector Buttons.

Nine bomb nav fix mode selector pushbuttons (20, figure 1-22) are located on the bomb nav control panel. Only one button can be depressed at a time. With each new mode selection, the preceding mode will disengage. In addition, a bar on the panel below the engaged button will become visible, indicating that the button has been depressed and the indicated mode selected. To disengage an operating mode without engaging another mode, lightly depress any other button. The buttons are labeled and function as follows:

1. PRES POS - Slaves the destination position counters to present position, removes attack radar cursors from display, and de-activates the target bearing, slant range, course angle, and distance to destination servos.
2. TARGET - Selects target set in the destination position counters as the radar sighting point. The present position counters may be corrected by the attack radar tracking handle or hand setting.
3. OFFSET - Selects offset aimpoint as the sighting point. The present position counters may be corrected by the attack radar tracking handle or hand setting.
4. MAN FIX - Applies energizing voltages to the present position hold and present position correction buttons. Removes attack radar cursors from display and de-activates the target bearing, slant range, course angle, and distance to destination servos.
5. DEST STORAGE 1, 2, and 3 - Slaves destination shafts to the stored 1, 2, or 3 destination. Removes attack radar cursors from display and de-activates the target bearing, slant range, course angle, and distance to destination servos. Stored information may be changed by hand-setting as desired.

Note

The destination position counters will not slew more than ±18 degrees of latitude or longitude from the latitude and longitude indicated at the time the fix mode DEST STORAGE selector button is depressed. The counter(s) will drive to an erroneous position should this range limitation not be observed during operation. However, even though the limitations are exceeded the stored information will not be lost, unless changed by hand-setting, and will drive the counters to the correct stored position when the range limitation is observed.

6. HOMER SET - Causes position of computer destination position relative to present position to be displayed on the radar homing and warning scope. This allows the coordinates of an interrogating radar to be determined by correcting the destination position counters with the attack radar tracking handle until the bomb nav system computer cursor on the radar homing and warning scope is in coincidence with the interrogating radar's indicated position.
7. HOMER TRACK - Selected to obtain second fix on interrogating radar set making use of triangulation to improve accuracy of fixing the interrogating radar coordinates. The computer cursor on the radar homing and warning scope will respond only to cross-track correction commands from the attack radar tracking handle.

Present Position Correction Button.

The present position correction button (3, figure 1-22), located on the bomb nav control panel, is labeled CORRECT PRES POSITION. This button is used, in conjunction with other controls, as a mode of updating the present position counters at the time of overflying a fixpoint. The mode cannot be activated until after the fix mode MAN FIX selector button has been depressed. Also, the destination position counters must first be set to the latitude and longitude of the fixpoint to be overflown. With the two preceding requirements satisfied, momentarily depressing the present position correction button at the instant of overflying the fixpoint, will slave the present position counters to the destination position counters. During this time the destination position counters will be tracking actual aircraft position while the present position counters are catching up. The present position up-dating is complete when the readings of the two counters agree, at which time, the mode may be deactivated (deenergized) by depressing any other fix mode selector button.

Present Position Hold Button.

The present position hold button (1, figure 1-22), located on the bomb nav control panel, is labeled HOLD PRES POSITION. This button is used, in conjunction with other controls, as a mode of up-dating present position when it is not desired to reset the destination position counters from distant destination to local fixpoint. The mode cannot be activated until after the fix mode MAN FIX selector button has been depressed. With the above requirement satisfied, depressing and holding the present position hold button while re-setting the present position counters to the latitude and longitude of the fixpoint, and then releasing the button at the instant of overflying the fixpoint will cause the present position counters to start tracking the corrected aircraft position. To deactivate the mode some other fix mode selector button must be depressed.

Platform Alignment Control Knob.

The platform alignment control knob (15, figure 1-22) located on the bomb nav control panel is a three-position rotary knob marked RAPID ALIGN, NORMAL and PLATFORM OFF. The knob must be pulled out before it can be rotated from any position. The knob controls the alignment modes of the inertial platform when the bomb nav mode selector knob is positioned to ALIGN. In the PLATFORM OFF position, the inertial system is completely de-energized. In the NORMAL position, the platform may be aligned in either of two ways, (1) gyrocompass or (2) alignment to stored magnetic variation. The gyrocompass alignment is the slower alignment, but is the more precise and may be used irrespective of the accuracy to which local magnetic variation is known. Alignment to stored magnetic variation may be used when local magnetic variation is accurately known and a rapid alignment is desired. In the gyrocompass alignment, the stabilization platform goes through two modes of azimuth alignment: (1) Alignment to true north as defined by the flux valve magnetic heading plus hand-set magnetic variation, and (2) Refinement of alignment or gyrocompassing to true north as precisely defined by the direction where the "East" gyroscope senses none of the earth's rotation. In the alignment to stored magnetic variation, the operator manually places the platform alignment control knob to RAPID ALIGN as soon as the navigational computer (NC) true heading has stabilized at the flux valve heading, plus magnetic variation. The decision to use this method is based on three considerations: (1) Urgency of accomplishing alignment rapidly, (2) Accuracy to which local magnetic variation is known, and (3) Whether or not gyrocompass heading information has previously been stored in the system for the rapid align to stored heading mode.

Note

> If gyrocompass heading information has previously been stored in the system and the airplane heading has not been changed since the information was stored, the RAPID ALIGN mode should be used. In this mode, the platform will align to the stored heading and will be ready for operation in 90 seconds or less.

Altitude/Test Selector Knob.

The altitude/test selector knob (11, figure 1-22), located on the bomb nav control panel, is labeled ALTITUDE TEST. The knob has three positions for computer self-testing marked 1, 2 and 3; three positions for altitude calibration marked CAL, NORM and ALIGN; and an OFF position. The knob must be pulled out to go into or out of the Test Sector. The altitude positions control a clutch that connects or disconnects the altitude calibration (ALT CAL) knob to the fixpoint elevation counter and either activates the GO lamp circuitry (in ALIGN) or commands the attack radar system to the altitude calibration mode (in CAL). The CAL position provides for manual calibration of pressure altitude above terrain when over terrain of known elevation with the attack radar system in operation. The ALIGN position works in conjunction with the GO lamp and provides for setting the pressure altitude correction term to zero for test purposes, or in lieu of calibration. When set to the NORM position, the signals applied in CAL or ALIGN positions are removed and the system is in the normal mode. Test 1, 2, and 3 positions are used in conjunction with the GO lamp for system operational ground checkout only.

Note

> Do not set the switch to any of these test positions in flight.

Platform Indicator Lamps.

Three platform indicator lamps (18, figure 1-22), located on the bomb nav control panel, are labeled HEAT, ALIGN and ERROR. The amber HEAT indicator lamp provides a monitor of the platform heat signal from the stabilization platform (SP). When the system is activated, the lamp will light until the gyroscope temperature reaches the factory-set level. When the lamp is lighted, the SP will not enter the gyrocompass cycle. When the lamp goes out, it will remain out until the system is recycled. The green ALIGN indicator lamp provides for monitoring platform alignment status during the alignment mode (mode selector knob in ALIGN). With the platform alignment control knob in NORMAL, the lamp will light when the SP switches into the gyrocompass phase of alignment. It will remain lighted continuously until the gyrocompass process has attained the required alignment accuracy, at which time it will begin flashing, signalling the operator that the gyrocompass process is complete to required accuracies.

The mode selector knob may then be left in ALIGN to allow alignment quality to improve. The knob must be advanced to a navigation mode for taxi. With the platform alignment control knob in RAPID ALIGN, the ALIGN indicator lamp will remain out until alignment is complete at which time (within 90 seconds of turn-on), the lamp will begin flashing. The amber ERROR indicator lamp provides a monitor of the platform reliability signal from the stabilization platform when the mode selector knob is in ALIGN or any of the normal navigation positions. The lamp will light when the SP is off level, unless the platform alignment control knob is in PLATFORM OFF and the altitude test selector knob is in any one of the three test positions. In either condition, the navigational computer is automatically in the auxiliary navigation mode, except when in Test 1, 2, or 3. The lamp will light, should the platform become unreliable, when the bomb nav mode selector knob is in any of the normal navigation modes and will go out when an auxiliary navigation mode is selected.

Primary Attitude/Heading Caution Lamp.

The primary attitude/heading caution lamp, located on the main caution lamp panel (14, figure 1-5), will light when either attitude or heading information from the stabilization platform is interrupted or becomes unreliable due to a platform malfunction. Placing the bomb nav mode selector switch to an auxiliary navigation position or placing the flight instrument reference select switch to AUX or AUX STANDBY INST ONLY will also cause the lamp to light. When the lamp lights the letters PRI ATT/HDG will be visible. Should the lamp light when operating with the flight instrument reference select switch in the PRI position the attitude director indicator and horizontal situation indicator will be automatically switched to the AFRS, however, the AUX position on the flight instrument reference select switch should be selected so that the proper source of signals will be indicated.

Go Lamp.

The green GO lamp (10, figure 1-22), located on the bomb nav control panel, is labeled GO and provides for monitoring the navigational computer self-test circuits and fixpoint elevation counter alignment. The lamp will light when the altitude/test selector knob is placed in ALIGN or Test positions 1, 2, or 3, provided the proper settings, as described elsewhere, have been set on the panel, and the bomb nav system is operational.

Present Position Counters.

Two present position counters (2, figure 1-22), located on the bomb nav control panel, are labeled PRES LATITUDE and PRES LONGITUDE. The counters display the geodetic latitude and longitude utilized as the airplane position coordinates. The counters are continuously and automatically up-dated by inputs of true north and east velocity components from the stabilization platform during all normal navigational modes. During all auxiliary navigational modes, the counters are similarly up-dated by north and east velocities as derived from airspeed, hand-set or last computed wind data, and auxiliary flight reference system heading plus hand-set magnetic variation. Control knobs provide for handsetting the counters to the initial position or to insert manual corrections. Rapid adjustment is accomplished electrically by merely turning the knobs. Fine adjustment is accomplished manually by depressing and turning the knobs. The counters may also be automatically driven with the attack radar tracking handle.

Fixpoint Elevation Counter.

The fixpoint elevation counter (5, figure 1-22), located on the bomb nav control panel is labeled FIXPOINT ELEVATION. An adjacent altitude calibration knob, labeled ALT CAL, provides control of the counter. The counter is handset to the elevation of the radar aiming point in use; either (1) the position indicated by the destination counters, or (2) the offset aimpoint when using offset sighting, or (3) elevation of terrain below the airplane when calibrating altitude.

Destination Position Counters.

Two destination position counters (22, figure 1-22), located on the bomb nav control panel, are labeled DEST LATITUDE and DEST LONGITUDE. The counters which display destination position in geodetic latitude and longitude, may be handset or automatically slaved to track the present position counters. The counters may also be automatically slaved to positions stored in any of three storage channels or automatically driven with the attack radar system tracking handle. Control knobs are provided to handset the destination latitude and longitude counters. Rapid adjustment is accomplished electrically by merely turning the knobs. Fine adjustment is accomplished manually by depressing and turning the knobs.

True Heading Counter.

The true heading counter (6, figure 1-22), located on the bomb nav control panel, is labeled TRUE HEADING. The counter continuously displays the computed airplane heading relative to true north as derived from (1) inputs of true heading from the stabilization platform during all normal navigation modes, and (2) magnetic heading input from the auxiliary flight reference system and hand-set magnetic variation during all auxiliary navigation modes.

Magnetic Variation Counter.

The magnetic variation counter (7, figure 1-22), located on the bomb nav control panel, is labeled MAG VAR. The counter displays manually inserted magnetic variation. The counter is varied by manually turning its control knob. In normal navigation modes, the counter is adjusted until the magnetic heading synchronization indicator indicates a null. At this time, the counter will indicate the local variation. In auxiliary navigation modes, the counter must be up-dated to settings specified from map data. This up-dating procedure has no effect on the synchronization meter but will simultaneously up-date the navigational computer true heading.

Magnetic Heading Synchronization Indicators.

The magnetic heading synchronization indicator (4, figure 1-22), located on the bomb nav control panel, is marked MAG HEADING SYNC. In normal navigation modes, the indicator provides an indication of agreement (or disagreement) between computed magnetic heading (which is also being transmitted to the flight instruments) and magnetic heading from a flux valve input. During the normal navigation modes, the indicator is maintained at null by periodic manual correction of handset magnetic variation to correct computed magnetic heading to agree with flux valve data. The indicator should be disregarded when operating in any of the auxiliary navigation modes. In platform align modes, the indicator automatically provides a status of azimuth alignment independent of other parameters.

Groundtrack and Groundspeed Counters.

The groundtrack and groundspeed counters (9, figure 1-22), located on the bomb nav control panel, are labeled GROUNDTRACK and GROUNDSPEED. The counters continuously display the computed true groundtrack and groundspeed as derived from (1) inputs of true north and east velocity from the stabilized platform during all normal navigation modes, and (2) airspeed input from the central air data computer, magnetic heading input from auxiliary compass set, handset magnetic variation, and either stored or handset wind information during all auxiliary navigation modes.

Wind Speed and Wind From Counters.

The wind speed and wind from counters (19, figure 1-22), located on the bomb nav control panel, are labeled WIND SPEED and WIND FROM. During all normal navigational modes, the counters continuously and automatically display the computed value of wind direction and magnitude as derived from inputs of true velocity and true heading from the stabilized platform and airspeed from the central air data computer. During auxiliary navigational modes, the counter readings are controlled by adjacent control knobs for manual up-dating wind information as corrected wind information becomes available to the operator.

Glide/Dive Angle Counter.

The glide/dive angle counter (17, figure 1-22), located on the bomb nav control panel, is labeled GLIDE/DIVE ANGLE. The counter displays the preselected glide or dive angle set into the computer in the vertical steering mode. The glide/dive angle is used to derive information proportional to the difference between the hand-set desired value and the computed value of either aircraft dive angle (in dive mode) or glide slope (in airborne instrument landing approach mode). The resultant steering information is displayed on the attitude director indicator (ADI) and lead computing optical sight (LCOS).

Offset Range and Offset Bearing Counters.

The offset range and offset bearing counters (21, figure 1-22), located on the bomb nav control panel, are labeled OFFSET RANGE and OFFSET BEARING. The counters are hand-set by the operator to values derived from map data. They represent the range and bearing from the position represented by the destination counters to the position of a preselected radar sighting point. The information entered into these counters is provided to the computer only when the fix mode OFFSET selector button is depressed. The computer utilizes this information to derive radar sighting parameters for the selected offset sighting point. Derivation of these parameters will allow the offset point to be used as a radar position fix reference while the bombing computer is solving equations for attack of the position indicated by the destination counters. This mode is used when the target is a poor or no-show radar target with a good radar target in the near vicinity.

Bomb/Nav Distance—Time Indicator.

The bomb/nav distance-time indicator (BNDTI) (21, figure 1-5), located on the left main instrument panel, is a remote indicating type instrument. The indicator displays digital time and distance to target or destination. The display elements are (1) a four-digit counter display and (2) a legend-type tape display. The digital display consists of three synchro drum counters which receive signals from the navigational computer (NC). The fourth digit is fixed at zero and is covered by a shutter except when operating in the great circle navigation mode. The operating modes of the NC are identified on a servo driven tape, containing legends positioned in a window, and functioning in synchronization with the digital display. When operating in great circle mode, the tape will display great circle/miles and the digital display will indicate distance in nautical miles to destination. When operating in short range mode, the tape will display short range/miles and the digital display will indicate distance in nautical miles to destination. When operating in trail bomb mode, the tape will display bomb trail/seconds and the digital display will indicate the time, in seconds, remaining before bomb release. When operating in range bomb mode, the tape will display bomb range/thousands of feet, and the digital display will indicate, in feet, the distance to the bomb release point. When the bomb nav system is off, or when the operator is changing system modes, the digital display is covered by a shutter.

Master Power Switch.

The master power switch (9, figure 1-23), located on the armament select panel, is a two position switch marked ON and OFF. When the switch is placed to the ON position electrical power is applied to the armament system.

Destination Distance/Time Counter.

The destination distance/time counter (14, figure 1-22), is located on the bomb nav control panel. The counter is marked MILES TO DEST, SEC TO REL and FEET TO REL. Each function of the counter is separately lighted so that only the correct marking for the quantity being displayed is visible to the operator. When the mode selector knob is positioned to GREAT CIRCLE or SHORT RANGE, MILES TO DEST will light and display continuous computation of the distance in nautical miles from present position to any destination set into the destination counters (provided the destination is within 4000 nautical miles of the present position). When the mode selector knob is placed to RANGE BOMB, the words, FEET TO REL and two zeros (00) will light, and display continuous computation of the range to the bomb release point in feet from zero to 400,000 feet (based on handset weapon release range). When the mode selector knob is placed to TRAIL BOMB, SEC TO REL will light, and display continuous computation of the seconds remaining before bomb release ranging from zero to 400 seconds (based on handset weapon trail and time-of-fall).

Trail/Range Counter.

The trail/range counter (12, figure 1-22), located on the bomb nav control panel, is labeled TRAIL/RANGE. The counter displays trail or range bombing parameter information manually set into the computer for trail bomb or range bomb mode. The trail/range knob is mechanically connected to the counter and a potentiometer. The output of the potentiometer is a voltage representing the trail or range value set into the counter for use by the computer during bombing. The counter setting has no effect on the great circle or short range modes.

Weapon Mode Selector Knob.

The weapon mode selector knob (3, figure 1-23), located on the armament select panel, is used to select the type of store to be released or type of release to be accomplished. The knob has 18 positions marked clockwise as follows:

OFF

DISP STEP - dispenser step.

DISP TRAIN - dispenser train.

BOMB STEP S - bomb step singles.

BOMB STEP P - bomb step pairs.

BOMB TRAIN S - bomb train singles.

BOMB TRAIN P - bomb train pairs.

RKT - rockets.

B/C - biological/chemical.

NUC WPN - nuclear weapons.

GAM 83 - GAM-83 (AGM-12B) missiles.

SHRIKE S - shrike missile single.

SHRIKE P - shrike missile pairs.

GAR 8 LNCH - GAR-8 (AIM-9B) missile launch.

GAR 8 JETT - GAR-8 (AIM-9B) missile jettison.

JETT PYLON 1 & 8 - jettison pylons 1 and 8.

JETT PYLON 2 & 7 - jettison pylons 2 and 7.

JETT SEL STORES - jettison selected stores.

Placing the knob to the GAM-83 position selects the GAM-83 (AGM-12B) missiles being carried for launch. An individual missile station must then be selected and a release signal generated to launch a missile. For additional information on the weapon mode selector knob positions related to missiles and pylons, refer to the associated paragraphs under "Armament System", this section.

Delivery Mode Selector Knob.

The delivery mode selector knob (10, figure 1-23), located on the armament select panel, provides a means of selecting the source of signal for weapon release. The knob has five positions marked OFF, NAV, MAN, ANGLE and TIMER to provide the following weapon delivery capabilities:

Note

> The weapon release button on either control stick must be depressed to complete a release circuit in any of the following knob positions.

OFF - prevents all weapon release capability except jettison.

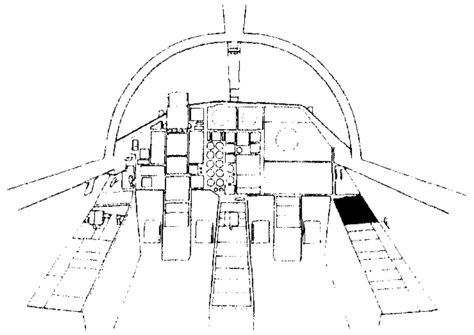
NAV - provides automatic weapon delivery by utilizing release signals generated by the bomb nav system.

MAN - provides manual release using the weapon release buttons mounted on the aircraft commander's and pilot's control stick grips.

ANGLE - provides loft type weapon delivery capability at various predetermined angles by utilizing release signals generated by the lead computing optical sight.

TIMER - provides loft and straight fly-over timed weapon delivery capability by utilizing pull-up and release signals generated by the dual bombing timer.

Armament Select Panel



1. PYLON WEAPON SELECTOR BUTTON (8)
2. WEAPONS BAY CONTROL BUTTON (2)
3. WEAPON MODE SELECTOR KNOB
4. MISSILE SELECTOR KNOB
5. INTERVALOMETER SET KNOB
6. WEAPONS BAY DOOR OPEN LAMP
7. TRAPEZE EXTEND LAMPS
8. WEAPONS BAY AUXILIARY CONTROL SWITCH
9. MASTER POWER SWITCH
10. DELIVERY MODE SELECTOR KNOB
11. SHRIKE BAND SELECTOR SWITCH
12. BOMB ARMING SELECTOR SWITCH
13. SHRIKE TARGET REJECT BUTTON
14. RELEASE SYSTEM TEST BUTTON

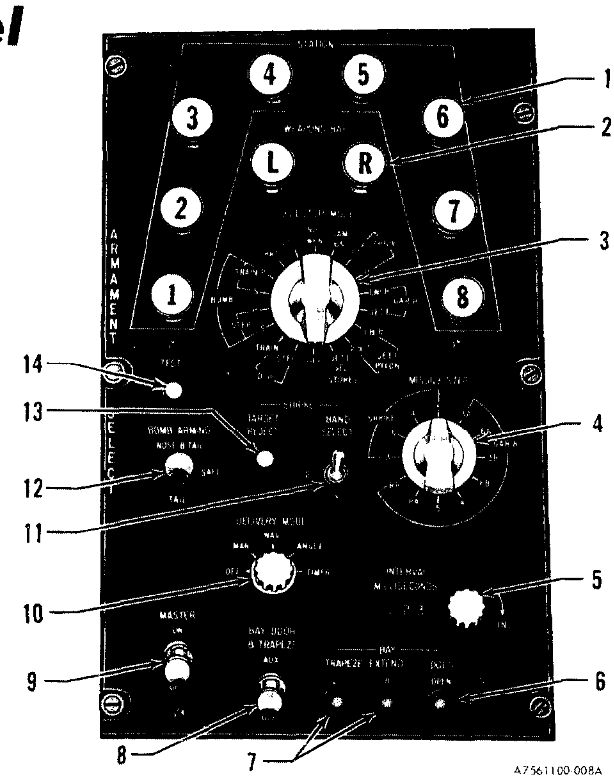


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Figure 1-23.

Pylon Weapon Selector Buttons.

Eight push-pull type buttons (1, figure 1-23), located on the armament select panel, provide for the selection of the external pylon stations. The buttons are numbered from 1 to 8. Buttons 1, 2, 7 and 8 are for the fixed pylon stations. Buttons 3, 4, 5 and 6 are for the pivot pylon stations. Depressing a button selects that station for release and a lamp in the button will light to indicate the presence of a store at that station. Pulling the button out will dearm the station and the lamp will go out.

Bomb Arming Selector Switch.

The bomb arming selector switch (12, figure 1-23), located on the armament select panel has three positions marked NOSE & TAIL, SAFE, and TAIL. The switch controls 28 volt dc power from the main dc bus to arm or safe the fusing systems of conventional bombs released from the weapons bay or pylons. The NOSE & TAIL position is used to arm bombs which utilize both a nose and tail fuse. The TAIL position is used to arm bombs which utilize only a tail fuse. Placing the switch to the NOSE & TAIL or TAIL positions will cause the arming solenoids in the bomb racks to retain the arming wires from each bomb fuze thereby arming the bomb fuzes as the bombs are released. Placing the switch to SAFE will allow the arming wires to pullout of the arming solenoids and stay in the fuzes as the bombs are released thereby rendering the weapons safe.

Release System Test Button.

The release system test button (14, figure 1-23), located on the armament select panel, provides a means of monitoring the presence of stores on the pylon and weapon bay stations. Placing the weapon mode selector knob to a position corresponding to the type of weapons being carried and depressing the test button will cause a lamp to light in each pylon weapon selector and weapon bay control button for stations that have weapons loaded.

Intervalometer Counter and Set Knob.

An intervalometer counter and set knob (5, figure 1-23), located on the armament select panel, are provided to set weapon release intervals when releasing more than one weapon. The counter is set by turning the knob clockwise to increase the time interval and vice-versa. The counter has three digits and can be set for weapon release intervals from 0.010 second to .999 second in increments of 1 millisecond.

Nuclear Consent Switch.

The nuclear consent switch (12, figure 1-4), located on the left console has three positions marked ARM & REL, OFF, and REL ONLY. Placing the switch to the ARM & REL position enables nuclear weapon arming and unlocking of the bomb racks at all stations. Placing the switch to the REL ONLY position enables unlocking of the bomb racks at all stations. With the switch in the OFF position all power is removed from the bomb rack and nuclear weapon arming circuits. A red guard must be raised to gain access to the switch. When nuclear weapons are carried the guard is safetied and sealed in the closed position.

Nuclear Weapons Monitor and Release Knob.

The nuclear weapons monitor and release knob (3, figure 1-23A), located on the nuclear weapons control panel has seven positions marked OFF, 3, 4, L, R, 5, and 6. Placing the knob to OFF opens the monitoring and release circuits to all stations. Positions 3, 4, L, R, 5, and 6 are for the pivoting pylon and weapon bay stations. Selecting one of these positions completes the monitoring and release circuits to the station selected.

Nuclear Weapons Arm Knob.

The nuclear weapons arm knob (5, figure 1-23A), located on the nuclear weapons control panel, has seven positions marked OFF, MON (monitor), SAFE, GRD RET (ground retard), AIR RET, GRD FF (ground freefall), and AIR FF. When the knob is in the OFF, all power is removed from all nuclear weapon arming and monitoring circuits. Placing the knob to the MON position allows monitoring of the condition (safe or armed) of the nuclear weapon selected by the nuclear weapons monitor and release knob. Placing the knob to SAFE provides power to safe all nuclear weapons simultaneously. The SAFE position also allows the knob lock lever to be moved from the OMS position to the S ARM position and return. The GRD RET, AIR RET, GRD FF, and AIR FF positions function in conjunction with the nuclear consent switch to arm the nuclear weapon selected for release to the option desired. The knob controls 28 vdc power from the essential dc bus.

Nuclear Weapons Arm Knob Lock Lever.

The nuclear weapons arm knob lock lever (4, figure 1-23A), located on the nuclear weapons control panel, has two positions marked OMS (off, monitor, safe) and S ARM (safe, arm). The lever is safetied and sealed to the OMS position when nuclear weapons are carried. With the lever in the OMS position, the nuclear weapons arm knob can be moved to the OFF, MON, and SAFE positions. Placing the nuclear weapons arm knob to SAFE allows movement of the lever to the S ARM position. The arm knob can then be moved from SAFE to one of the four fuzing option positions and returned.

Nuclear Weapons Monitor Lamps.

Seven nuclear weapons monitor lamps (1, figure 1-23A), located on the nuclear weapons control panel, provide monitoring of the bomb racks and nuclear weapons being carried. When lighted the lamps indicate the following condition exists at the station selected by the nuclear weapons monitor and release knob:

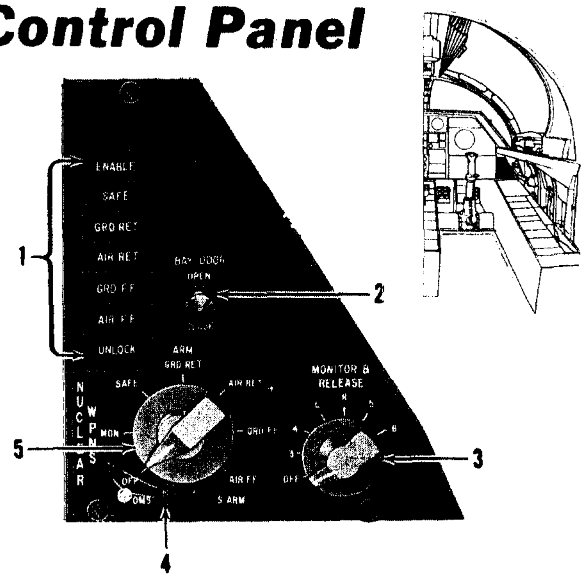
ENABLE - This lamp inoperative.

SAFE - Indicates the nuclear weapons circuits are safe.

GRD RET (ground retard)
AIR RET (air retard)
GRD FF (ground freefall)
AIR FF (air freefall)
} Indicate the fusing option setting of the nuclear weapon selected.

UNLOCK - Indicates the bomb rack is unlocked and a nuclear weapon is present at that station.

Nuclear Weapons Control Panel



1. NUCLEAR WEAPONS MONITOR LAMPS (7)
2. WEAPON BAY DOOR SWITCH
3. NUCLEAR WEAPONS MONITOR AND RELEASE KNOB
4. NUCLEAR WEAPONS ARM KNOB LOCK LEVER
5. NUCLEAR WEAPONS ARM KNOB

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Figure 1-23A.

Weapon Release Buttons.

Two weapon release buttons (1, figure 1-15), one on each control stick grip, initiate or enable normal weapon release from the pylon or weapon bay stations. The buttons are labeled WPN REL. The function of the button is described under each type of store release capability of the airplane.

External Stores Jettison Button.

The external stores jettison button (7, figure 1-5), located on the left main instrument panel, is a flush mounted push button labeled EXT STORES JETTISON. Depressing the button will jettison all external stores when the airplane is on the ground or inflight. Nuclear weapons, if installed, will jettison only if their racks are unlocked.

Time-of-Fall Counter.

The time-of-fall counter (8, figure 1-22), located on the bomb nav control panel, is labeled TIME OF FALL. The counter displays time-of-fall bombing parameter information hand-set into the counter for use during the trail bomb mode. The time-of-fall control knob is mechanically connected to the time-of-fall counter to provide a hand-setting method. Information displayed is entered into the computer only when the system is operating in the trail bomb mode.

Bomb Release Lamp.

The green bomb-release lamp (16, figure 1-22), located on the bomb nav control panel is labeled BOMB RELEASE. The lamp provides monitoring for the automatic release signal that is computed and provided by the navigational computer to other equipment for weapons release. The lamp will light when the computer is in the trail or range bomb configuration and the seconds or feet to release has driven to zero.

BOMB NAV SYSTEM OPERATION.

Normal Mode Operation.

During normal operation, the bomb nav mode selector knob is set to either the GREAT CIRCLE, SHORT RANGE, TRAIL BOMB, or RANGE BOMB positions as appropriate to phase of mission. The heading groundtrack, and ground speed counters are controlled by outputs from the stabilization platform. Wind speed and wind direction are computed and displayed on the wind speed and wind from counters. Present position is continuously and automatically updated by input velocity signals from the SP, and may be corrected, as required, by radar sighting and/or manual fix modes. Range and course to target or destination are continuously computed and transmitted to the flight instruments, with range or time to target or destination displayed on the destination distance/time counter. All other counters and controls are hand set, as required, by the operator. If the mode selector knob is in a normal navigation mode, the platform ERROR indicator light will illuminate if the SP fails, at which time the NC will automatically switch into auxiliary navigation.

Altitude Calibration. Altitude calibration is neccessary prior to position updating, radar bombing and AILA letdown. Altitude calibration is not affected by operations in the auxiliary navigation modes.

Using Attack Radar.

This procedure should be accomplished over level terrain of known elevation when flying at altitudes above 1000 feet.

1. Fixpoint elevation counter - Set.
   Set the fixpoint elevation counter to the known elevation of the terrain where the calibration is to be accomplished.
2. Bomb nav mode selector knob - SHORT RANGE.
3. Fix mode target selector button - Depress.
4. Attack radar mode selector knob - GND AUTO or GND VEL.
5. Attack radar beta switch - MAN.
6. Attack radar antenna tilt control knob - Adjust tilt full down.
7. Altitude/test selector knob - CAL.
8. Altitude calibration knob - Position range cursor coincident with first ground return.
   Turn altitude calibration knob until the range cursor is coincident with the first ground return.
9. Altitude/test selector knob - NORM.

Using Radar Altimeter.

This procedure should be accomplished over level terrain of known elevation when flying at altitudes below 1000 feet.

1. Fixpoint elevation counter - Set.
   Set the fixpoint elevation counter to the known elevation of the terrain where the calibration is to be accomplished.
2. Bomb nav mode selector knob - SHORT RANGE.
3. Fix mode target selector button - Depress.
4. Attack radar mode selector knob - GND AUTO or GND VEL.
5. Altitude/test selector knob - CAL.
6. Altitude calibration knob - Set attack radar cursor range counter to radar altimeter reading.
7. Antenna tilt indicator - Minus 30. Check that the antenna tilt indicator reads -30 to assure that a positive altitude value was set.
8. Altitude/test selector knob - NORM.

Magnetic Variation Updating. In normal navigation modes, the need for magnetic variation updating is indicated by an off-null condition on the magnetic heading synchronization indicator.

1. Magnetic variation counter control knob - Adjust.
   Adjust the knob until the magnetic heading synchronization indicator pointer is centered.

Note

When in auxiliary navigation modes, magnetic variation control knob adjustments update true heading and do not affect the magnetic heading synchronization indicator.

2. Magnetic Variation Counter - Check.
   Check that the magnetic variation counter indicates magnetic variation of the present airplane position.

Auxiliary Mode Operation.

Note

> Arbitrary selection of AUX NAV in flight, when the SP is good, should be kept to a minimum. The alignment of the platform will be unnecessarily subjected to possibly incorrect earth rate torquing signals due to degraded accuracy of present latitude updating in auxiliary navigation (AUX NAV) modes.

The auxiliary navigation (AUX NAV) modes are identical to the normal modes with the exception that the wind computation is stopped and the airspeed and stored winds are substituted for the stabilization platform outputs. Navigational computer true heading is derived from the back-up compass system and hand set magnetic variations. Magnetic heading for the horizontal situation indicator (HSI) and the attitude director indicator (ADI) is supplied directly from auxiliary flight reference system (AFRS). The magnetic heading synchronization indicator is not operative. The platform ERROR indicator light will be at off at all times when the mode selector knob is in an auxiliary navigation mode.

Airplane Position Updating.

The need for airplane position updating is indicated primarily by inaccuracies in attack radar crosshair laying. In addition it is indicated by other reference navigational accuracy information available from other systems. If the attack radar system is not operating, manual position updating should be accomplished periodically. Airplane position updating will be required more often when operating in an auxiliary navigation mode since nav system accuracy will be slightly degraded. Normal turn on, gyrocompassing platform alignment procedures are contained in this section as well as within the appropriate portions of Section II. Alignment to stored magnetic variation and rapid alignment to stored gyrocompass heading are covered as alternate procedures in this section. For Airborne Instrument Landing and Approach (AILA) procedures, refer to Section VII.

Radar Fix. The following procedure is applicable only if the attack radar is operating.

1. Altitude calibration - Completed.
2. Fix mode TARGET selector button - Depress.
   Depress the fix mode TARGET selector button then enter longitude and latitude coordinates of the fixpoint into the destination position counters.
3. Fixpoint elevation counter - Set to fixpoint elevation.
4. Bomb nav mode selector knob - Any position other than GREAT CIRCLE.
5. Attack radar mode selector knob - GND AUTO or GND VEL.
6. Destination/present position selector switch - PP.
7. Attack radar range selector knob - Use lowest range setting possible.
8. Depress the enable switch on the attack radar tracking handle and then move the radar cursors with the attack radar tracking handle until they are brought into coincidence with the fixpoint image. Discontinue tracking when the sighting angle reaches approximately 45 degrees.

Note

> At this point, the fix is complete and the present position counters are corrected.

9. Destination/present position selector switch - DEST.
10. Bomb nav mode selector knob - As desired.
11. Destination position counters - Enter original, or next destination coordinates into counters.

Manual Present Position Fix (Correct Present Position). Fly toward the fixpoint.

1. Destination position counters - Set fixpoint coordinates.
2. Fix mode MAN FIX selector button - Depress.
   Depress the fix mode MAN FIX selector button when approaching the fixpoint. Computed course and miles to destination will remain at the computed values existing when the MAN FIX selector button is depressed. The attack radar cursor will disappear from the scope.
3. Present position correction button - Depress.
   Depress the present position correction button at the instant of overflying the fixpoint as determined visually, or with the attack radar or TFR ground map scope displays. The fix is complete when the present position counters stop slewing and agree with the destination position counters. Both sets of counters will drive at the same rate.
4. Present position and destination position counters - Checked.
5. Fix mode TARGET selector button - Depress.

6. Destination position counters - Reset.
   Course and distance computations will resume to the new destination and the attack radar cursors will fall on the new destination if in range.

Manual Present Position Fix (Hold Present Position).
Fly toward the fixpoint.

1. Fix mode MAN FIX selector button - Depress.
   Depress the fix mode MAN FIX selector button when approaching the fixpoint. Computed course and miles to destination will remain at the computed values when the MAN FIX selector button is depressed. The attack radar cursors will disappear from the scope.
2. Present position hold button - Depress, and hold.
3. Present position counters - Set.
   Set the coordinates of the fixpoint in the present position counters.
4. Present position hold button - Release, over fixpoint.
   Release the present position hold button at the instant of overflying the fixpoint as determined visually, or with the attack radar or TFR ground map scope displays.

Note

The fix is complete. The present position counters will start to drive to track the aircraft position.

5. Fix mode TARGET selector button - Depress.
6. Destination position counters - Reset.
   Course and distance computations will resume to the new destination and the attack radar cursors will fall on the new destination if in range.

Gyrocompass Alignment Procedure.

1. Bomb nav mode selector knob - HEAT.
2. Platform heat indicator lamp - Lighted.

Note

The platform heat indicator lamp may not light if the SP has been operating within 30 minutes preceding this alignment.

3. Altitude/test selector knob - NORM.
4. Present position latitude - Checked.
   If latitude is incorrect proceed as follows:
   a. Platform alignment control knob - PLATFORM OFF.
   b. Bomb nav mode selector knob - ALIGN.
   c. Present position latitude counter - Set.
   d. Bomb nav mode selector knob - HEAT.
   e. Platform alignment control knob - NORMAL.
5. Magnetic variation counter - Check and set to local variation.
6. Platform heat indicator lamp - Out.
7. Bomb nav mode selector knob - ALIGN.
8. Present position longitude counter - Check and set if necessary.
9. Destination latitude and longitude counters - Set, as required.
10. Platform align indicator lamp - On steady after 1 minute, flashing after additional 4 minutes.
    The platform align indicator lamp should light within approximately one minute, then commence flashing within an additional 4 minutes. However, if the airplane is parked in an area where the normal earth's magnetic variation is significantly distorted (i.e. magnetic variation is not accurately known) more time may be required. A flashing platform align indicator lamp indicates the platform is sufficiently aligned to meet specification performance.
11. Magnetic heading synchronization indicator - Nulled and steady. (if time permits)
    A nulled and steady condition may not occur until after the platform align indicator lamp

begins flashing. If the indicator is not nulled, and time permits, the best possible alignment of the platform can be obtained by allowing the magnetic heading synchronization indicator to null. If the airplane is not to be moved immediately, the mode selector knob may be left in the ALIGN position until just before airplane movement. This will prevent any system error buildup during the waiting period.

Alternate Platform Alignment Procedures.

Alignment to Stored Magnetic Variation.

Pre-alignment Procedure

Position the airplane at the approximate location and heading where alignment to stored magnetic variation is anticipated.

1. Gyrocompass alignment - Completed.
2. Bomb nav mode selector knob - GREAT CIRCLE.
3. Magnetic heading synchronization indicator - Nulled.
4. Magnetic variation - Record for future reference.

Alignment Procedure

1. Altitude/test selector knob - NORM.
2. Platform alignment control knob - NORMAL.
3. Magnetic variation counter - Check and set pre-recorded value.
4. Bomb nav mode selector knob - ALIGN, and note time.
5. Present position counters - Checked. Check and set the present position latitude and longitude if necessary.

After approximately 110 seconds.

6. Bomb nav mode selector knob - GREAT CIRCLE or SHORT RANGE.
   At approximately 110 seconds after step 4 and before moving the airplane place the bomb nav mode selector knob to GREAT CIRCLE or SHORT RANGE.

Note

Any movement of the airplane such as that caused by the operation of the flight controls may induce a heading error in the system when switching from ALIGN to GREAT CIRCLE or SHORT RANGE.

Rapid Alignment to Stored Gyrocompass Heading.

Pre-Alignment Procedure

1. Gyrocompass alignment - Completed.
2. Platform alignment control knob - RAPID ALIGN.
3. Bomb nav mode selector knob - OFF.

Note

Once pre-alignment is complete the airplane must not be moved.

Alignment Procedure

1. Bomb nav mode selector knob - ALIGN.
2. Present position counters - Checked.
3. Platform alignment indicator lamp - Flashing, within 3 minutes.
4. Bomb nav mode selector knob - GREAT CIRCLE or SHORT RANGE.
   Place the bomb nav mode selector knob to GREAT CIRCLE or SHORT RANGE after the platform align indicator lamp starts flashing and before moving the airplane.
5. Rapid alignment control knob - NORMAL.

ARMAMENT SYSTEM.

The armament capability of the airplane includes the delivery of conventional and nuclear weapons in various configurations and air-to-ground and air-to-air gunnery. The stores are carried in the weapons bay and on eight wing pylons. Four of the wing pylons pivot to remain streamlined with different positions of the wing. The pivoting pylons are utilized for both stores and gunnery equipment in various configurations. Other gunnery equipment includes a weapons bay gun. Bombing and launching equipment (pylons, stores release system, and missile trapeze), weapons bay doors, the weapons themselves, and the gunnery equipment are herein, considered as the Armament System.

BOMBING AND LAUNCHING EQUIPMENT.

Bombing and launching equipment consists of the various bomb racks, missile trapeze, missile launchers, stationary and pivoting wing pylons, and the release systems. Where applicable, the controls and indicators and operating procedures for the bombing and launching equipment are covered in the following paragraphs, under the associated equipment headings. For bombing system controls and indicators and bombing procedures, refer to "Bombing - Navigation System," this section.

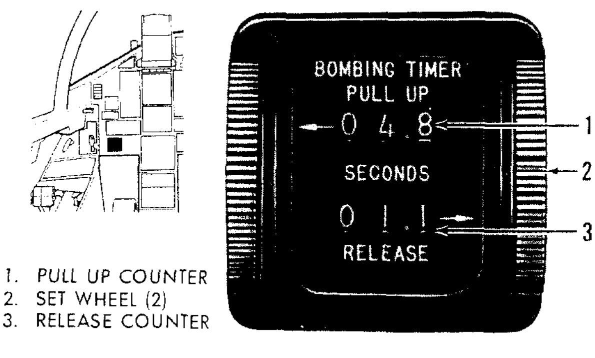
Pylons.

The airplane can be equipped with eight detachable pylons mounted along the lower surface of the wing. The pylons are designed to accommodate the MAU-12B/A bomb rack and the AERO-3B missile launcher. The pylons are numbered as stations 1 through 8, from left to right. Stations 1, 2, 7 and 8 are fixed pylon stations. The fixed pylons are streamlined at 26 degrees wing sweep angle only. A fixed stores lockout in the wing sweep handle prevents sweeping the wings more than 26 degrees with fixed pylons installed. The fixed pylons can be jettisoned by placing the weapons mode selector knob to 1 & 8 and 2 & 7 JETT PYLON positions and depressing the external stores jettison button to allow sweeping the wings more than 26 degrees. Stations 3, 4, 5 and 6 are pivoting pylon stations. These pylons are mechanically linked to keep the pylons streamlined as the wings are swept forward or aft. A weapons lockout in the wing sweep handle prevents sweeping the wings more than 55 degrees with certain weapons loaded on the inboard pivoting pylons to prevent damage to the fuselage. Pylon weapon selector buttons, located on the armament select panel, are provided to select individual pylon stations for release or launch. Each button is numbered corresponding to the station it controls.

Stores Release System.

The various stores carried on the airplane are released by electrical signals generated by the bomb nav system, lead computing optical sight or dual bombing timer. To accomplish a release, except jettison, the weapon mode selector knob must be positioned to the appropriate store or type release desired, the station to be released must be selected by depressing the appropriate pylon weapon selector button or weapon bay control button and either weapon release button on the control stick grips must be held depressed to initiate or enable the release. Refer to "Bombing - Navigation System" this section, for the type of releases that can be made for the various stores carried.

Dual Bombing Timer



1. PULL UP COUNTER
2. SET WHEEL (2)
3. RELEASE COUNTER

Figure 1-24.

Dual Bombing Timer. The dual bombing timer (figure 1-24), mounted on the left main instrument panel, provides a manual method of accurately timed weapon release for back-up weapon delivery. The timer has two digital readout windows marked PULL UP and RELEASE. Each may be set by means of thumb-actuated knurled wheels on each side of the timer to an accuracy of one-tenth of a second. The pull up window can be set from 0.2 to 60.0 seconds and the release window from 0.2 to 30.0 seconds. The timer is used with the delivery mode selector knob (10, figure 1-23) in either the TIMER or ANGLE mode positions when employing loft bombing delivery tactics. For a loft bombing delivery in the TIMER mode, precomputed values of time from initial point (IP) to pull up point and from pull up point to weapon release point are set in the PULL UP and RELEASE windows and the weapon is released at the expiration of these two times. For a loft bombing delivery in the ANGLE mode, a precomputed value of time from IP to pull up point is set in the pull up window, and the weapon is released at a predetermined pitch angle during the pull up maneuver. A fly over release is made with the delivery mode selector knob in the TIMER mode position only. For a fly over release the precomputed time from IP to target is divided and set in both windows in any manner so that the time set in the release window is more than 0.2 seconds and the total of the two is equal to the time from IP to target. This is necessary since a release signal will not be generated with the release window set at zero. When making a bomb run, either weapon release button (1, figure 1-15), located on either the left or right control stick, must be depressed when over the IP to start the timer, and held until after weapon release. Altitude, heading and airspeed must then be maintained at the predetermined values used for computing the times set into the timer. At the expiration of the time set in the PULL UP window a lamp (11, figure 1-5) on the left main instrument panel will light displaying the words PULL UP to indicate the point at which the pull up maneuver should commence if a loft bombing delivery is being made. At this time the pilot will initiate a manual constant 4g pull up using the airspeed-mach indicator accelerometer for "g" reference and the attitude director indicator and lead computing optical sight pitch steering bars for pitch steering commands. If the delivery mode selector switch is set to the TIMER position the weapon will be released at the expiration of release time and a green release lamp (11, figure 1-5), located on the left main instrument panel, will light displaying the words BOMB RELEASE. If the ANGLE position is used, the release portion of the timer is not used and the weapon will be released by a signal from the lead computing optical sight when a predetermined pitch angle is reached. If a straight fly over laydown delivery is being accomplished, disregard the pull up lamp and continue to hold altitude, heading, and airspeed through the expiration of release time to obtain weapon release. The timer receives 28 volt dc power from the 28 volt dc essential bus through the master power switch on the armament select panel.

Missile Trapeze.

Two trapeze are mounted side by side in the weapons bay to carry missiles internally. The trapeze functions in conjunction with the weapon bay doors. To launch a missile the weapons bay doors are opened and the trapeze is extended to fire the missile from below the airplane. Normal operation of the system is accomplished with hydraulic power from the utility hydraulic system. Alternate operation, in the event of failure of the utility hydraulic system, is accomplished by operating the weapons bay doors electrically and operating the trapeze with a self contained pneumatic system. Two weapon bay control buttons, located on the armament select panel, marked L and R are provided to open the doors and extend the respective trapeze. An auxiliary bay door and trapeze switch provides for alternate operation of the doors and trapeze in conjunction with the weapon bay control buttons. Electrical interconnects prevent; (1) Either trapeze from extending if both weapon bay control buttons are depressed, (2) Trapeze extension when the weapon bay doors are not fully open, (3) Closing the weapon bay doors when the trapeze is not fully retracted and (4) Launching a missile when the trapeze is not fully extended.

Missile Trapeze Controls. The missile trapeze is controlled by the weapons bay door controls; refer to "Weapons Bay Doors" this section.

Trapeze Extend Lamps. Two trapeze extend lamps (7, figure 1-23), located on the armament select panel, are marked L and R for the left and right trapeze. When lighted they indicate the respective trapeze is fully extended.

Missile Trapeze Operation. The missile trapeze operates in conjunction with the weapons bay docrs. Refer to "Weapons Bay Doors" this section.

WEAPONS BAY DOORS.

The weapons bay doors enclose the weapons bay area located between the nose and main landing gear. The doors are constructed in left and right clam shell halves which fold outward as they are opened. Normal and alternate systems are provided to operate the doors. The normal system utilizes hydraulic power from the utility hydraulic system to drive a hydraulic motor. The alternate system uses 115 volt ac power from the right main a-c bus to power an electric motor. Either motor drives a gear reduction mechanism, which through a series of drive shafts interconnected to hinges on the inside of the weapons bay, to open and close the doors. Normal time to open or close is 2-1/2 seconds. The alternate system takes approximately 30 seconds to open or close the doors. The weapon bay doors are controlled by either of two weapon bay control buttons which also control the trapeze and weapon stations in the weapon bay and by a weapons bay door switch.

Weapon Bay Control Buttons.

Two weapon bay control buttons (2, figure 1-23), located on the armament select panel, marked L and R are provided to open the weapon bay doors and extend the respective trapeze. Each button will open both weapon bay doors but will extend only its respective trapeze. If both buttons are depressed, an electrical interconnect prevents either trapeze from extending. A lamp in each button indicates the presence of a store on that trapeze station. A door open lamp and individual trapeze lamps will light when the weapon doors are open and respective trapeze is extended.

Weapons Bay Auxiliary Control Switch.

The weapons bay auxiliary control switch (8, figure 1-23), located on the armament select panel, is labeled BAY DOOR & TRAPEZE. The switch has two positions marked AUX and OFF. With the switch in the OFF position the weapons bay doors and trapeze operate on hydraulic pressure from the utility hydraulic system. Placing the switch to AUX provides electrical power to operate the weapons bay doors and pneumatic pressure from self contained pneumatic reservoirs in the weapons bay to extend and retract each trapeze. Refer to "Pneumatic Power Supply Systems" this section.

Weapons Bay Door Switch.

The weapons bay door switch (2, figure 1-24A), located on the nuclear weapons control panel, has two positions marked OPEN and CLOSE. The switch is used to open or close the weapons bay doors only when the weapon mode selector knob is in the NUC WPN (nuclear weapons) position.

Weapons Bay Door Open Lamp.

The weapon bay door open lamp (6, figure 1-23), located on the armament select panel, will light when the weapons bay doors are fully open.

Normal Operation of Weapons Bay Doors and Trapeze.

For normal operation of the weapons bay doors and trapeze the utility hydraulic system must be functioning and electrical power must be on.

To open doors and extend trapeze:

1. Master power switch - ON.
2. L or R weapon bay control button - Depressed.
3. Weapon bay door open lamp - On, after 2-1/2 seconds.
4. L or R trapeze extend lamp - On.

To retract trapeze and close doors:

1. L or R weapon bay control button - Pulled.
2. L or R trapeze extend lamp - Out.
3. Weapon bay door open lamp - Out.

Alternate Operation of Weapon Bay Doors and Trapeze.

For alternate operation of the weapons bay doors and trapeze electrical power must be on.

To open doors and extend trapeze:

1. Master power switch - ON.
2. Weapons bay auxiliary control switch - AUX.
3. L or R weapons bay control button - Depressed.
4. Weapons bay door open lamp - On, after 30 seconds.
5. L or R trapeze extend lamp - On.

To retract trapeze and close doors:

1. L or R weapons bay control button - Pulled.
2. L or R trapeze extend lamp - Out.
3. Weapons bay door open lamp - Out.

CAUTION

Do not use the alternate system to operate either the left or right trapeze more than one cycle (extend and retract). To do so will result in severe structural damage since there will be insufficient hydraulic fluid left in the trapeze actuator to dampen a second pneumatic extension of the trapeze.

MISSILES.

The airplane is equipped to carry AIM-9B (GAR-8), AGM-45A (SHRIKE) and AGM-12B (GAM-83) missiles.

AIM-9B (GAR-8) Missile.

The AIM-9B missile is a passive infrared-homing air-to-air supersonic guided missile. Since the missile homes on infrared energy radiated by heated parts of the target, it does not need to transmit a signal for guidance and is therefore relatively impervious to jamming. The missile is composed of four sections; guidance and control section, warhead, influence fuse, and rocket motor. The guidance and control section contains the optical tracking system, a gas operated control servo with four movable control fins to control the flight of the missile to the target, electronic components to convert target signals into missile control signals, a gas driven generator to supply electrical power during missile flight and the contact fuse. When the missile control system has acquired a target, an audio tone is transmitted on the interphone to the crew. The warhead, a 25 pound fragmentation type, is detonated by either the contact or influence fuse. The influence fuse will detonate the warhead if the missile passes within 30 feet of the target. The rocket motor is a standard 5 inch HVAR rocket motor. It will accelerate the missile to mach 1.7 above the speed of the launching aircraft. Four fixed stabilizer fins are mounted around the rocket motor section. The missile is carried on and launched from the AERO-3B launcher. The airplane is equipped to carry eight AIM-9B missiles as follows: Two on each outboard pivot pylon, one on each inboard pivot pylon and two on the right weapons bay trapeze station. The missiles are launched individually by selecting the desired missile to be fired and depressing the weapon release button on either control stick. All missiles can be jettisoned simultaneously by depressing the external stores jettison button.

Weapon Mode Selector Knob. The weapon mode selector knob (3, figure 1-23), has 18 positions, two of which are used in conjunction with the AIM-9B missile. These knob positions are labeled GAR-8 and are individually marked LNCH (launch) and JETT (jettison). Placing the knob to LNCH will allow the missile selected by the missile step selector knob to be launched when either weapon release button is depressed. Placing the knob to JETT will allow all missiles to be jettisoned when the external stores jettison button is depressed. Other positions of the weapon mode selector knob are described under their associated equipment paragraphs. Refer to "Bombing-Navigation System" and "AGM-45A (Shrike) Missile", this section.

Missile Step Selector Knob. The missile step selector knob (4, figure 1-23), located on the armament select panel, controls the launching sequence of SHRIKE or GAR-8 missiles. The knob has twelve positions. Four position are labeled SHRIKE and are marked 3, 6, 4, and 5 for the respective pylon weapons stations equipped to carry the SHRIKE missile. Eight positions of the knob are labeled GAR-8 and are marked 3A, 6A, 3B, 6B, 4, 5, RA and RB for the respective stations equipped to carry GAR-8 missiles. Pylon weapon station 3 and 6 and the right weapon bay station have the capability of carrying two GAR-8 missiles each therefore the A and B designates individual missiles at these stations. Each time a missile is launched the knob will step one position clockwise to sequence missile launchings from the left station to right station then back to left until all missiles are expended. When launching SHRIKE missiles in pairs the knob will step two positions.

AGM-45A (Shrike) Missile.

The AGM-45A is a supersonic air-to-ground guided missile. The missile is composed of four sections; guidance, armament, control, and propulsion. The guidance section consists of a radome, an RF receiver, and a computer, to detect targets, and the electronic portion of the proximity fuze. The armament section contains the warhead and fuze. The control section positions the missile wings as directed by the guidance system to steer the missile to the target. The propulsion section houses the rocket motor and igniter. Four identical wings, mounted on the control section provide control of the missile. Four tail fins mounted on the propulsion section provide flight stability. The missile functions in conjunction with the integrated flight instrument system and radar homing and warning system prior to launch to provide steering signals to the target on attitude director indicator and lead computing optical sight. An audio tone is provided through the interphone to inform the crew when in range of the target prior to launch and that the missile is functioning normally after launch. The airplane has the capability of carrying four missiles; one on each pivot pylon. The missiles can be launched individually or in pairs by placing the weapon mode selector knob and missile step selector knob to their respective positions and depressing either weapon release button. All missiles and their launchers can be jettisoned simultaneously by depressing the external stores jettison button.

Weapon Mode Selector Knob. The weapon mode selector knob (3, figure 1-23), has 18 positions, two of which are used in conjunction with the AGM-45A missile. These knob positions are labeled SHRIKE and are individually marked S (single) and P (pairs). Placing the knob to S will allow the missile selected by the missile step selector knob to be launched when either weapon release button is depressed. Placing the knob to P will allow a pair of missiles to be launched when either weapon release button is depressed. In this event the first missile of the pair being launched will be the one selected by the missile step selector knob. The second missile of the pair launched will be the symmetrical pair to the first and the missile step selector knob will step two positions clockwise. The second missile will be launched at the time interval set in the intervalometer. Other positions of the weapon mode selector knob are described under their associated equipment paragraphs. Refer to "Bombing-Navigation System" and "AIM-9B (GAR-8) Missile", this section.

Missile Step Selector Knob. The missile step selector knob (4, figure 1-23), controls the launching sequence of both the Shrike and GAR-8 missiles. For description of this knob, refer to "AIM-9B (GAR-8) Missile", this section.

Shrike Band Selector Switch. The shrike band selector switch (11, figure 1-23), located on the armament select panel, has three positions marked 1, 2, and 3. Each position represents a frequency band for missile target acquisition.

Shrike Target Reject Button. The shrike target reject button (13, figure 1-23), is located on the armament select panel. The button can be depressed to reject a target that the shrike missile has acquired, thereby providing a selection and rejection capability for the missile.

GAM-83 (AGM-12B) Missile.

The GAM-83 (AGM-12B) is a supersonic air-to-ground guided missile which utilizes radio controlled guidance signals to intercept a visually acquired target. The missile is composed of three sections; nose, body and tail. The nose section contains the missile guidance and control equipment and the warhead triggering device. The body section contains the warhead and fuze. The tail section contains the rocket motor, tracking flares and the missile wings. The tracking flares are ignited after launch to aid in visually guiding the missile to the target. The guidance system provides a means of after launch line-of-sight guidance of the missile from airplane to target. The system consists of a command guidance transmitter, a control stick and the necessary armament controls in the airplane to launch the missile and a command guidance receiver and a self contained guidance control equipment in the missile. The system utilizes the UHF frequency band with 24 transmitter channels. When activated the system uses the lower UHF antenna to transmit guidance command signals to the missile. The airplane is capable of carrying a GAM-83 (AGM-12B) on each of the four pivot pylons and the two inboard fixed pylons and on each trapeze in the weapons bay. Missiles are launched individually. After launch the missile is visually guided with the control stick. When the stick is moved to make corrections a signal is sent to the transmitter which codes the signal and transmits it to the receiver in the missile. The missile then decodes the signal and activates its control vanes in the proper proportion and direction in response to the signal. The system utilizes 28 volt dc power from the main dc bus and 115 volt ac power from the right main ac bus. Power is applied to the transmitter and to all missiles whenever power is on the airplane. Procedures for operation of the GAM-83 (AGM-12B) missile guidance system are contained within the appropriate portion of Section II.

Control Stick. The control stick (1, figure 1-45A) located on the right sidewall, is used to control the GAM-83 (AGM-12B) missile after launch. Moving the control stick in any direction initiates a signal proportional to the amount and rate of displacement of the control stick. This signal is sent to the transmitter which in turn transmits a coded radio frequency guidance command to the missile receiver.

Weapon Mode Selector Knob. The weapon mode selector knob (3, figure 1-23) has 18 positions, one of which marked GAM-83 is used in conjunction with launching the AGM-12B (GAM-83) missile. Placing the knob to the GAM-83 position selects the AGM-12B missiles being carried for launch. An individual missile station must then be selected and a release signal generated to launch a missile. Other positions of the weapon mode selector knob are described under their associated equipment paragraphs.

GUNNERY EQUIPMENT.

The airplane is equipped to carry a SUU-16/A gun pod on each pivoting pylon station and a weapons bay gun in the right side of the weapons bay.

SUU-16/A Gun Pod.

The SUU-16/A gun pod is provided primarily for air-to-ground gunnery. The pod contains the M-61 gun, a ram air turbine (RAT) drive assembly, and a linkless ammunition feed system. The M-61A1 is a 20 MM gun which has a rotating cluster of six barrels. The gun fires electrically primed ammunition at a nominal rate of 6000 rounds per minute. Hydraulic power to operate the gun and ammunition feed mechanism is furnished by the self contained RAT. A minimum of 300 KIAS (mach 0.5) is required to drive the RAT at a speed of 12,000 RPM for gun pod operation. Electrical power from the airplane system is used to extend the RAT into the slipstream prior to firing the gun. The RAT will free-wheel while the gun is held fixed by a brake. Then, when either gun trigger switch is depressed, the brake is released, the RAT is clutched into the gun mechanism, and electrical power is furnished to fire the gun. The gun pod carries 1200 rounds of ammunition of which approximately 1150 rounds can be expended. Expended ammunition cases are discharged overboard. A switch is provided for selecting pylon guns or weapon bay gun for firing. All pods will fire simultaneously each time the trigger is depressed. A safety switch on the main landing gear prevents firing the gun pods on the ground. The gun pods receive 28 vdc power from the main dc power panel to fire the guns.

Weapon Bay Gun Module.

The weapon bay gun module provides both air-to-air and air-to-ground gunnery capability. The module is packaged to facilitate installation and removal of components. When installed it supplants all other stores carrying capability in the right side of the weapons bay but does not alter the stores carrying capability in the left side of the bay. The module contains the M61A1 gun, a linkless ammunition feed system and an expended ammunition storage bin. The M61A1 is a 20 MM gun which has a rotating cluster of six barrels. The gun fires electrically primed ammunition at a nominal rate of 6,000 rounds per minute. Hydraulic power from the utility hydraulic system is used to operate the feed mechanism and the gun. The ammunition drum holds 2,050 rounds of which approximately 2000 rounds can be expended. Expended ammunition cases are retained in the storage bin which must be emptied on the ground. A switch is provided for selecting the weapon bay gun or pylon guns for firing. Depressing either gun trigger switch will fire the gun or guns as selected. A safety switch on the main landing gear prevents firing the gun on the ground. The guns are fired by 28 vdc power from the main dc power panel. A round counter provides an indication of remaining ammunition.

Gun Selector Switch.

The gun selector switch (4, figure 1-5), located on the left main instrument panel, is labeled GUNS and has three positions marked PYLONS, BAY, and OFF. With the switch in the OFF position the weapon bay gun and gun pods cannot be fired. Placing the switch to PYLONS enables firing the gun pods mounted on the pivot pylons. Placing the switch to BAY enables firing the weapons bay gun.

Rounds Counter.

The rounds counter (5A, figure 1-5), located on the left main instrument panel, provides an indication of the amount of ammunition remaining in the weapon bay gun. The counter is graduated from 0 to 20, times 100, in increments of 100. Ammunition counters are not provided for the pylon guns.

Gun Trigger Switch.

Two gun trigger switches (5, figure 1-15), one located on each control stick grip, are provided to fire the guns. Depressing either switch will fire either all pylon gun pods or the weapon bay gun depending on the position of the gun selector switch.

Operation of the Weapon Bay Gun or Pylon Gun Pods.

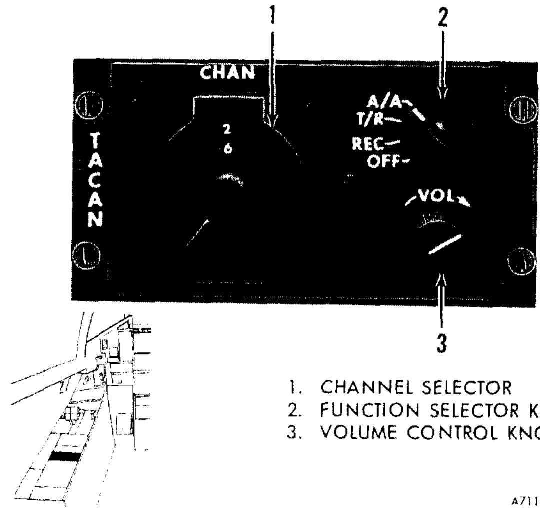
Except under actual combat conditions the guns will not be fired unless over a cleared gunnery range.

1. Master power switch - ON.
2. Gun selector switch - As required.
3. LCOS mode selector knob - GUN-AA or GUN-AG. (as applicable)
4. LCOS range set knob - Set range.
5. LCOS true airspeed knob - Set TAS.
6. LCOS aiming reticle brightness knob - Set as desired.
7. Center pipper on the target.
8. Gun trigger switch - Depress when in range.

TACTICAL AIR NAVIGATION SYSTEM (AN/ARN-52).

The tactical air navigation system (TACAN) enables the airplane to receive continuous indications of its distance and bearing from any selected TACAN station located within a line-of-sight distance of approximately 300 nautical miles. There are 126 channels available for selection. The equipment consists of the TACAN receiver-transmitter and its control panel. Two antennas, one on top of the fuselage and the other beneath the fuselage (figure 1-40), function to keep the TACAN receiver locked on to the antenna receiving a usable signal. The TACAN equipment also has an air-to-air mode and can be used between two aircraft having TACAN with air-to-air capability for range information only. The TACAN works in conjunction with the instrument system coupler, the bearing distance heading indicator, the lead computing optical sight, the horizontal situation indicator, the attitude director indicator, and through the interphone control panel for audio output. The system operates on 28 vdc from the main dc bus and 115 vac from the left main ac bus. The TACAN control panel (figure 1-25) is located on the left console.

Tacan Control Panel



1. CHANNEL SELECTOR
2. FUNCTION SELECTOR KNOB
3. VOLUME CONTROL KNOB

A7111200-002

Figure 1-25.

TACAN FUNCTION SELECTOR KNOB.

The function selector knob (2, figure 1-25), located on the TACAN control panel, has four positions marked OFF, REC, T/R, and A/A. In the OFF position, electrical power to the TACAN system is off. In any of the other three positions, electrical power is supplied and the TACAN set is on. In the REC position, the set will receive bearing and audio identity signals only. In REC position, range information will not be displayed because the TACAN transmitter is not on. In the T/R position, both the receiver and the transmitter are operative, the system will receive and display both range and bearing of the station being interrogated, and audio identity signals are fed into the interphone system. In the A/A (air-to-air) position, the set will transmit and receive to and from another aircraft having air-to-air capability. To operate in this mode, the air-to-air mode in both aircraft must be selected and the channels selected must be 63 channels apart. As an example, if the TACAN in one aircraft is on channel 10, the TACAN in the other aircraft must be selected to channel 73. In the A/A mode, the TACAN will provide range between aircraft information only (no identity or bearing).

TACAN CHANNEL SELECTOR.

The channel selector (1, figure 1-25), located on the TACAN control panel, consists of inner and outer adjustment controls for selecting any one of the available 126 TACAN channels. The selected channel is digitally displayed on the selector. The outer control is used to select the first two digits of the desired channel and the inner control to select the last digit.

TACAN VOLUME CONTROL KNOB.

A volume control knob (3, figure 1-25), located on the TACAN control panel, provides a means for controlling the volume of the audio identity code.

TACAN ANTENNA SELECTOR SWITCH.

The three position TACAN antenna selector switch (2, figure 1-30), located on the antenna select panel, controls the selection of the upper and lower TACAN antennas. The switch is marked UPPER, AUTO and LOWER. Placing the switch to AUTO causes the antenna selector to control the antenna switching relay to select the correct antenna. Placing the switch to UPPER or LOWER controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

TACAN OPERATION.

1. Function selector knob - As required (REC, T/R, and A/A).
2. Antenna selector switch - AUTO.
3. Channel selector - As required.
4. Volume control knob - Adjust for desired volume level.
5. Horizontal situation indicator (HSI) course selector window - Set.
   Set the desired TACAN course in the HSI course selector window.
6. Instrument system coupler mode selector knob - TACAN.
7. Monitor ADI, LCOS and HSI for proper indications.

INSTRUMENT LANDING SYSTEM.

The instrument landing system (ILS) provides the capability of making instrument letdowns and approaches to runways equipped with localizer, glide slope and marker beacon equipment. The system consists of three receivers, one each for localizer, glide slope and marker beacon; four antennas, two for localizer and one each for glide slope and marker beacon, a control panel and a marker beacon light. The localizer and glide slope receivers operate on 20 fixed frequency channels which may be selected on the control panel. Glide slope frequencies are paired with localizer frequencies so that selection of a localizer channel automatically provides for glide slope reception. Localizer identification signals are supplied to the headset for station identification. Localizer and glide slope steering and deviation signals are provided to the instrument system coupler for display on the attitude director indicator (ADI), horizontal situation indicator (HSI) and lead computing optical sight (LCOS). Warning flags on the ADI become visible whenever the signal level on the selected frequency is too weak to be usable or is unreliable. Refer to "Instruments," this section, for the tie-in of the ILS and Integrated Flight Instruments. The marker beacon receiver operates on a fixed frequency of 75 megacycles and when over a beacon facility will provide a coded station signal to the headset and to the marker beacon lamp. Power is applied to the marker beacon receiver whenever power is on the airplane. The ILS operates on 28 volt dc power from the 28 volt dc main bus.

ILS FREQUENCY SELECTOR KNOB.

The frequency selector knob (2, figure 1-26), located on the ILS control panel, allows individual selection

ILS Control Panel

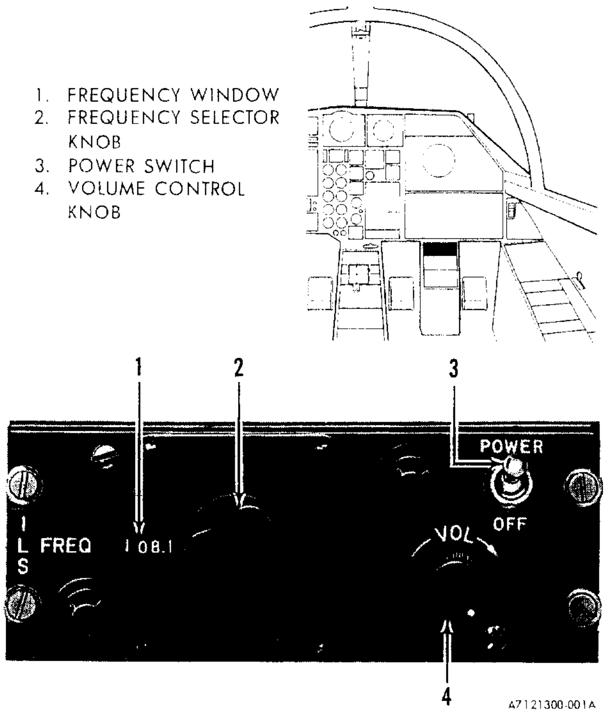


Figure 1-26.

of 20 ILS channels ranging in localizer frequencies from 108.1 to 111.9 mc in 0.2 mc increments. There is a detent position of the knob for each channel. One complete rotation of the knob covers the full range of frequencies. Each localizer frequency selected is paired with a glide slope frequency between 329.3 and 335.0 mc. The frequency of each channel selected is displayed in a digital window to the left of the knob.

ILS POWER SWITCH.

The power switch (3, figure 1-26), located on the ILS control panel, is a two position switch marked POWER and OFF. In the OFF position power is removed from the localizer and glide slope receivers. When the switch is placed to POWER 28 volt dc power is applied to the localizer and glide slope receivers.

ILS VOLUME CONTROL KNOB.

The volume control knob (4, figure 1-26), located on the ILS control panel, adjusts the volume of the localizer station identification signal. Clockwise rotation increases volume.

MARKER BEACON LAMP.

The marker beacon lamp (11, figure 1-5), located on the left main instrument panel, provides a visual coded station signal when the airplane is over a marker beacon facility. When lighted the words MARKER BEACON are displayed in green.

ILS OPERATION.

1. Power switch - POWER.
2. Frequency selector knob - Set.
   Set the frequency selector knob to the frequency of the localizer facility to be used.
3. Volume control - Adjusted.
   Adjust the volume to a comfortable level while identifying the station.
4. Instrument system coupler mode selector knob - ILS.
5. HSI course selector window - Set.
   Set the inbound localizer course in the HSI course selector window.
6. Monitor ADI, LCOS, and HSI for proper indications.

RADAR ALTIMETER SYSTEM (AN/APN-167).

The radar altimeter system is a dual channel low altitude radar system which provides precise absolute altitude, rate of altitude change and minimum altitude penetration information. Absolute altitude from 0 to 5000 feet is read on the radar altimeter. Rate of altitude change from 0 to 500 feet per second is furnished to the terrain following radar. Minimum altitude penetration fly-up signals are provided to the integrated flight instruments. The system is composed of two receiver-transmitter (RT) units; two antennas, one for transmitting and one for receiving; a distribution box; a radar altimeter and the necessary controls. The RT units are located in the forward electronic equipment bay. When the system is placed in operation, one RT unit is activated and the other is in standby for use in event the operating unit malfunctions. In the event of a malfunction the standby RT unit must be manually selected. The RT unit in operation is connected to the antennas and its outputs are distributed to other airplane systems by circuits in the distribution box. The system will break lock when 5000 feet above the terrain or when the bank or pitch limits are exceeded. A pressure operated switch in each RT unit will place the operating unit to standby when above 38,000 feet pressure altitude. The system incorporates a self-test feature for checking reliability. The system operates on 115 volt ac power from the main ac bus and 28 volt dc power from the main dc bus. Refer to figure 1-40 for antenna location.

RADAR ALTIMETER CHANNEL SELECTOR SWITCH.

The radar altimeter channel selector switch (15, figure 1-4), located on the left console, is labeled RADAR ALTM and has two positions marked CHAN 1 and CHAN 2. Placing the switch in either position will allow the RT unit in the respective channel to transmit and receive.

RADAR ALTIMETER BYPASS SWITCH.

The radar altimeter bypass switch (14, figure 1-4), located on the left console, is a two position switch marked NORMAL and BYPASS. Placing the switch to BYPASS when above 5000 feet over the terrain provides a signal to the TFR to permit automatic blind letdowns. As 5000 feet is passed during descent, the switch will go to NORMAL. When the switch is in the NORMAL position, automatic blind letdowns from below 5000 feet above terrain only, may be accomplished.

RADAR ALTIMETER.

The radar altimeter (15, figure 1-5), located on the left main instrument panel, provides absolute altitude indications from 0 to 5000 feet. Indications are provided by a pointer on a dial graduated in increments of 10 feet from 0 to 500, 50 feet from 500 to 1000, and 500 feet from 1000 to 5000. An OFF warning flag in a window on the right side of the dial will appear when power is removed from the system or any time the system breaks lock. The radar altimeter control knob on the lower right of the altimeter serves three functions; as an on-off control, to set a minimum altitude index pointer on the dial and as a test button to check the system. Initially turning the knob clockwise applies power to the system, further rotation of the knob rotates the index pointer from zero to any desired minimum altitude setting. Depressing and holding the knob activates the self-test feature of the system and provides an indication of 100 (±10) feet if the RT unit is operating properly. The self-test feature may be used at any time and at any altitude below 38,000 feet.

RADAR ALTITUDE LOW WARNING LAMP.

The radar altitude low warning lamp (16, figure 1-5), located on the right main instrument panel will light when one of the following conditions occur:

When the airplane descends below the minimum altitude index setting.

When the airplane descends through 5000 feet the lamp will light momentarily as the radar altimeter gains lock on and the pointer drives up through the minimum altitude index setting.

When lighted the letters RADAR ALT LOW are displayed on the face of the lamp in red. On airplanes (1)→(11) the intensity of the lamp can be adjusted with the malfunction and indicator lamp dimming switch. On airplanes (12)→ the lamp cannot be dimmed.

RADAR ALTIMETER OPERATION.

1. Radar altimeter control knob - On.
   The altitude pointer on the radar altimeter will deflect clockwise and then return to zero.
2. Minimum altitude index pointer - Set.
   Set the minimum altitude index pointer at 50 feet.
3. Channel selector switch - CHAN 1.
4. Power off warning flag - Out of view.
   After approximately 120 seconds warm-up time, the power off warning flag should disappear from view. The altitude pointer should read zero and the radar altitude low warning lamp should light.
5. Radar altimeter control knob - Depressed.
   Depress the radar altimeter control knob and observe that the altitude pointer drives to 100 (±10) feet and the radar altitude low warning lamp goes out.
6. Radar altimeter control knob - Release.
   Observe the altitude pointer returns to zero and the warning lamp lights.
7. Channel selector switch - CHAN 2.
   Repeat steps 5 and 6.
8. Set the minimum altitude index pointer to the minimum altitude to be flown.

TERRAIN FOLLOWING RADAR (AN/APQ-110).

The terrain following radar (TFR) provides low altitude terrain following, obstacle avoidance and blind letdown capability. The TFR consists of left and right antenna receivers, synchronizer transmitters, power supplies and computers in a dual channel configuration; a radar scope panel and a control panel. Each channel may be operated independently of the other in any one of three modes; terrain following (TF), situation display (SIT), or ground mapping (GM). The TFR receives inputs from the radar altimeter, attack radar, bomb-nav system or auxiliary flight reference system and central air data computer. The TFR operates on 115 volt ac power from the main ac bus and 28 volt dc power from the main dc bus.

TERRAIN FOLLOWING (TF) MODE.

The TF mode allows the airplane to be flown manually or automatically at a preselected terrain clearance. Climb and dive signals generated in this mode are furnished to the attitude director indicator (ADI), lead computing optical sight (LCOS) and to the flight control system. The set terrain clearance can be manually maintained by flying pitch steering commands on the ADI and LCOS. In auto TF operation the set terrain clearance will be automatically held. The TF mode can also be used to make blind letdowns to a preselected terrain clearance. When using this capability, descent can be made manually using the pitch steering commands on the ADI and LCOS, or automatically by placing the auto TF switch to the AUTO TF position. The descent is limited to a 12 degree dive. Only one channel at a time can be operated in TF mode. If both channels are placed to TF the second channel placed to TF will go to a standby condition as a backup and will automatically take over should the operating channel fail. A failure in the operating TF channel will provide a fly-up signal to either the pitch damper or ADI as follows: If the airplane is being flown with the auto TF switch in the AUTO TF position or FLY-UP ONLY position the pitch damper will initiate a "2g" pull up. If the airplane is being flown manually with the auto TF switch in the OFF position and with the instrument system coupler pitch steering mode switch in the TF position, a "2g" (incremental) referenced fly-up steering command will be displayed on ADI and LCOS pitch steering bars.

Note

- The fly up signal can be interrupted by holding the autopilot release lever depressed. To get rid of the fly-up signal the auto TF switch or the operating TFR channel must be turned OFF.
- The fly-up signal is locked out to prevent automatic fly-up when the landing gear is in the down position or when the flight control system switch is in the T.O. & LAND position.

Refer to "Flight Control System" this section, for information pertaining to operation of the TFR with the flight control system. In the TF mode, antenna scan is vertical and the scope display is in the form of a non-linear E type presentation. A cursor, displayed on the scope, provides a terrain clearance reference. The slope of the cursor will vary with the speed of the airplane, terrain clearance setting, the type of ride selected, pitch and angle of attack. Range displays on the scope are from left to right on a non-linear scale so that ranges up to two miles are displayed over three-fourths of the scope and the remaining one-fourth of the scope displays returns up to ten miles. Elevation of returns along the ground track are displayed vertically on the scope. In this manner the close returns are displayed in clearer definition than those at greater range.

WARNING

The TFR does not provide terrain avoidance information on either side of the flight path when operating in the TF mode. If coordinated turns are made while flying TF mode, do not exceed 10 degree of bank angle.

SITUATION (SIT) MODE.

This mode of operation is used in conjunction with TF mode for obstacle avoidance. Antenna scan is in azimuth, 30 degrees either side of ground track. Antenna tilt cannot be adjusted. Returns of the terrain that is higher than the airplane altitude are displayed on the radar scope in a one radius offset PPI presentation. Ground track is stabilized vertically along the center of the scope. Range graduations in this mode are linear.

GROUND MAPPING (GM) MODE.

The GM mode provides a scope presentation of the terrain that is ahead of the airplane below the altitude being flown. Antenna tilt can be adjusted for best picture. This mode is used primarily for navigation. The antenna scan and the type of scope display are the same as when operating in SIT mode.

TFR CHANNEL MODE SELECTOR KNOBS.

Two, five position rotary channel mode selector knobs, (2, figure 1-27), located on the TFR control panel, permit selection of the desired operating mode in each of the two channels. The knobs are labeled L and R for the respective channel and are individually marked OFF, STBY, TF, SIT and GM. In the OFF position, power is removed from the channel. In the STBY position, power is applied to the channel for warm-up. The TF, SIT and GM positions provide terrain following, situation display or ground mapping modes of operation respectively. Each channel may be operated in a different mode; however, one channel should always be in TF mode or a fail light and fly up signal will be generated. If both knobs are positioned to TF the second channel will automatically go to a standby condition, then, should the operating channel fail the one in standby will automatically take over.

AUTO TERRAIN FOLLOWING SWITCH.

The auto terrain following (auto TF) switch (11, figure 1-17), located on the center console, is a three position switch marked AUTO TF, FLY UP ONLY, and OFF. The switch is solenoid held in AUTO TF and OFF positions and spring-loaded to FLY UP ONLY. When the switch is in AUTO TF position and the reference engage button is depressed, the airplane will automatically maintain a preselected altitude above the terrain. With the switch in FLY UP ONLY, the pilot must fly the airplane manually but should the TFR malfunction, a signal will be sent to the pitch damper causing a "2g" (incremental) pull-up. When the switch is placed to OFF the pitch damper cannot receive TFR signals and the TF fly-up off caution lamp will light. The switch will not latch in AUTO TF or OFF positions unless the TFR is operating. If power to the holding solenoid is lost while operating in AUTO TF or OFF, the switch will return to the FLY UP ONLY position. The reference not engaged caution lamp will not light for this malfunction.

TERRAIN CLEARANCE KNOB.

The terrain clearance knob (3, figure 1-27), located on the TFR control panel, has six positions for setting terrain clearance altitudes. The marked positions of the knob is classified information. Rotating the knob clockwise increases the altitude clearance setting and vice versa.

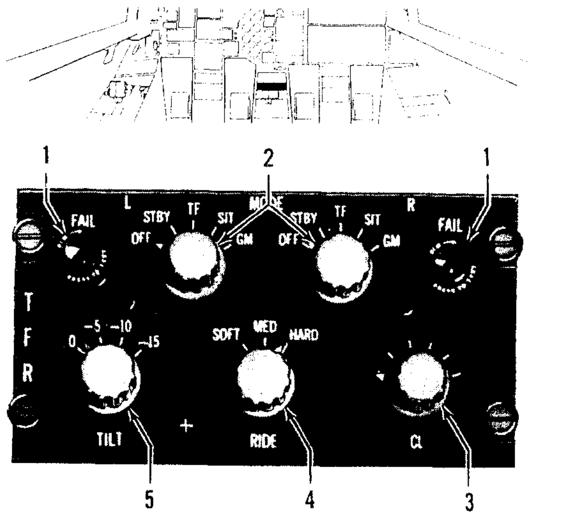
RIDE CONTROL KNOB.

The ride control knob (4, figure 1-27), located on the TFR control panel is a three position rotary knob marked SOFT, MED and HARD. The knob controls the magnitude of the negative "g" forces imposed on the airplane by the flight control system as it maintains a set altitude clearance above the terrain. Negative "g" (incremental) forces of -1.0, -0.5, and -0.25 will be experienced in the HARD, MED and SOFT positions respectively. The system will automatically provide a "2g" (incremental) pull up if necessary to avoid an obstacle regardless of the ride selected.

ANTENNA TILT CONTROL KNOB.

The antenna tilt control knob (5, figure 1-27), located on the TFR control panel is used to position antenna tilt between zero and -15 degrees for the best ground return when operating in the GM mode. The knob will continuously vary the antenna position between zero and -15 degrees. The knob has antenna tilt angles of 0, -5, -10 and -15 marked for reference.

Terrain Following Radar Control Panel



1. TFR CHANNEL FAILURE CAUTION LAMPS (2)
2. TFR CHANNEL MODE SELECTOR KNOBS (2)
3. TERRAIN CLEARANCE KNOB
4. RIDE CONTROL KNOB
5. ANTENNA TILT CONTROL KNOB

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Figure 1-27.

RANGE SELECTOR KNOB.

The range selector knob (5, figure 1-28), located on the TFR scope panel, has four positions marked 5, 10, 15 and E. The first three positions change range of the scope presentation when using SIT or GM modes. The E position is used with the TF mode only.

RADAR SCOPE CONTROL KNOBS.

Four radar scope control knobs (4, figure 1-28), located on the TFR scope panel provide a means of adjusting the scope to obtain the best display. The knobs are labeled CURSOR, MEMORY, CONTRAST and VIDEO from top to bottom. The cursor knob adjusts the brilliance of the range cursors. The memory knob increases or decreases scope storage retention time. The contrast knob adjusts scope contrast for optimum viewing. The video control adjusts the video return brightness to desired level.

RADAR SCOPE.

The radar scope (1, figure 1-28), located on the TFR scope panel, provides a direct viewing presentation of either an E (vertical scan) display when in TF mode or a sector PPI (azimuth scan) display when operating in SIT or GM modes. The scope overlays provide a rectangular grid with a 0 to 10 nautical mile scale at the bottom of the scope for TF mode and a "V" shaped grid for sector PPI presentations in SIT or GM modes. The polaroid filter controls around the face of the scope can be rotated to adjust polarization of light for the best display under various degrees of light. A red scope presentation for night vision adaptation can be obtained with the filter controls. The ear type handles on each side of the scope are provided to facilitate removal or installation of the unit.

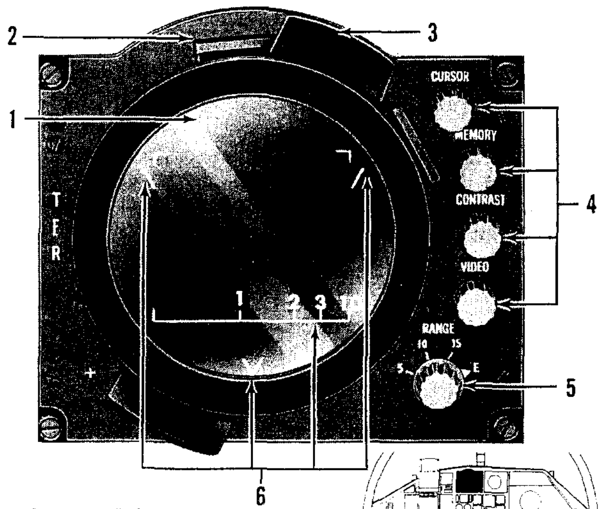
Note

> When the LCOS mode select knob is placed to the DIV BOMB, RKT AG or GUN AG position, the attack radar and TFR are used in conjunction with the LCOS for air-to-ground ranging. Under this condition the attack radar and TFR ground mapping or situation scope presentations will be unusable and should be ignored.

TFR CHANNEL FAIL CAUTION LAMPS.

Two amber channel failure caution lamps (1, figure 1-27), located on the TFR control panel, are individually marked FAIL and are labeled L and R for the respective left and right channels. When the channel

Terrain Following Radar Scope Panel



1. RADAR SCOPE
2. POLAROID FILTER CONTROL (2)
3. SCOPE REMOVAL HANDLE (2)
4. RADAR SCOPE TUNING CONTROL KNOBS (4)
5. RANGE SELECTOR KNOB
6. SCOPE OVERLAY

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Figure 1-28.

mode selector knob is placed from OFF to STBY the fail lamp will light to indicate that channel is not yet ready to operate. The lamp will go out after approximately 3 minutes indicating the channel is ready. After the channel is ready, a fail light with the mode selector switch in TF, SIT or GM position, indicates a malfunction in that channel. A press-to-test feature allows each lamp to be checked.

TF FLY-UP OFF CAUTION LAMP.

The TF fly-up off caution lamp, located on the main caution lamp panel (14, figure 1-5), will light when the auto TF switch is in the OFF position. The letters TF FLY-UP OFF are visible on the face of the lamp when it is lighted.

TFR FAILURE WARNING LAMP.

A TFR failure warning lamp (11, figure 1-5), located on the left main instrument panel, provides a more apparent indication of TFR channel malfunctions. If each channel is being operated in a different mode the lamp will light when the channel in TF mode malfunctions. If both channels are in TF mode, the lamp will momentarily light when the channel in operation fails and the backup channel takes over. Should the backup channel in turn fail, the lamp will light and remain on.

TERRAIN FOLLOWING RADAR OPERATION.

1. Left and right channel mode selector knobs - STBY.
2. Radar altimeter control knob - ON and set.
   Rotate the altimeter control knob clockwise to turn the altimeter on and to set altitude index pointer to minimum desired altitude.
3. Drift angle accuracy - Check. (AC-P)
   The following checks should be made prior to and during terrain following operation to determine if drift angle information is accurate.

WARNING

Before beginning terrain following and during terrain following operation it is essential that the TFR receive accurate drift angle information. Inaccurate drift angle information to the TFR could result in the airplane flying into the ground since the antenna search pattern will not be along the ground track being flown.

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a. Left channel mode selector knob - SIT or GM.

b. TFR scope display - Check.
Check that the radar returns move straight down the scope with no apparent drift to either side.

c. Left channel mode selector knob - TF.

d. Right channel mode selector knob - SIT or GM.

e. TFR scope display - Check.
Check that the radar returns move straight down the scope with no apparent drift to either side.

f. ISC mode select knob - NAV.

g. HSI - Check
Check that course arrow indicates a realistic drift angle as compared with the lubber line.

h. Attack radar scope display - Check. (P)
Check the display with the attack radar mode selector knob in one of the following position:

(1) GND MAN - With the antenna uncaged, check that the radar return moves straight down the scope with no apparent drift to either side.

(2) GND VEL - Check that rate of cursor drift is not excessive.

(3) GND AUTO - Check that radar return moves straight down the scope with no apparent drift to either side and that rate of cursor drift is not excessive.

i. Bomb nav system - Check. (P)

(1) Check for realistic computed drift angle by comparing groundtrack and true heading.

(2) Check wind speed counter for realistic value.

(3) Compare groundspeed counter with true airspeed for reasonable difference.

(4) Check for absence of gross error in present position by observing the position counters or the attack radar scope display.

4. Drift angle accuracy check - Complete. (AC-P)

5. Right channel mode selector knob - As desired.

6. Terrain clearance knob - Desired altitude.

7. Ride control knob - As desired.

8. Antenna tilt control knob - Adjust as required. (For ground map mode only)

9. Range selector knob - As desired.
For a scope presentation in TF mode, rotate the knob to the E position.

10. Radar scope tuning control knobs - Adjust for best scope presentation.

11. Polaroid filter - Adjust as required.

12. ISC pitch steering mode switch - TF.

13. Auto TF mode switch - As desired.
Place the auto TF mode switch to auto TF or FLY UP ONLY for the type operation desired.

14. Reference engage button - Depress.

WARNING

- The TFR does not provide terrain avoidance information on either side of the flight path when operating in the TF mode. If coordinated turns are made while flying TF mode, do not exceed 10 degrees of bank angle.
- Over calm water or flat terrain such as dry lake beds, dry wheat fields, or smooth sand, there will be little energy returned back to the radar. When forward video is lost while flying over smooth water or certain terrain, terrain following will be commanded by the radar altimeter. The radar altimeter looks only below the airplane and has no forward looking capability; therefore, it will provide safe flight only if the ground does not rise rapidly. Thus if forward video is lost on the scope from inadequate returns over certain terrain, the terrain following radar cannot be expected to provide safe flight.

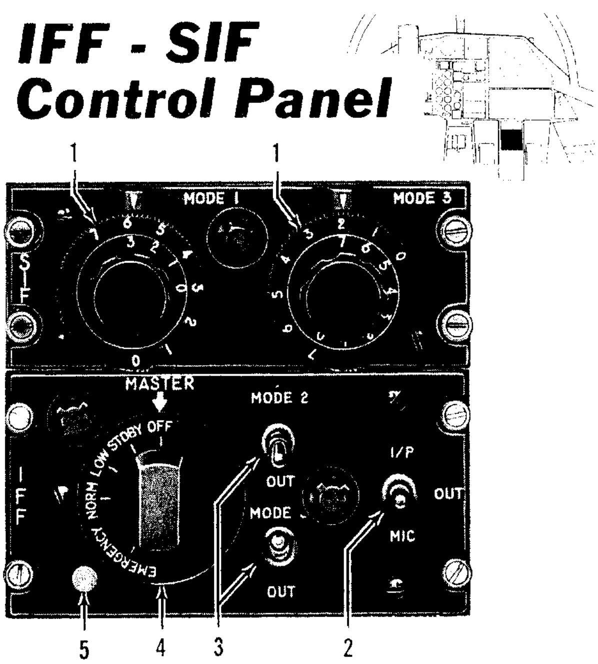
CAUTION

The back scatter from drizzle or rain and other forms of precipitation will often be visible on the scope. The operator should recognize that if the precipitation is so heavy that he cannot determine visually where the terrain ends and the precipitation begins, the automatic signal detection circuitry will also be incapable of the discrimination and a climb command will result.

IFF-SIF SYSTEM (AN/APX-46).

The air-to-ground IFF-SIF system provides the airplane with an automatic means of selective identification to ground, or shipboard recognition installations operating in the L-band frequency range. The system replies to proper interrogations from Mark X IFF systems and SIF (selective identification feature) stations. Operation is possible in any one of three modes, with the capabilities of I/P (identification of position) and emergency identification. The modes of operation have the following significance: Mode 1 - Security Identity, Mode 2 - Personal Identity and Mode 3 - Traffic Identity. The equipment consists of an IFF-SIF control panel, a transmitter-receiver, a decoder-coder, an antenna lobing switch, and two radiator-type antenna. The IFF-SIF control panel (figure 1-29) is located on the pilot's right pedestal. The receiver-transmitter is located in the electronic equipment bay. The equipment does not perform interrogation but only transmits coded replies to correctly coded interrogations. Two blade type antennas, an upper and lower, are provided. See figure 1-40 for antenna locations. The lobing switch rapidly transfers contact of the transmitter-receiver from one antenna to the other. This constant alternation eliminates blank spots in the antenna pattern caused by airplane structure. The receiver is sensitive to all signals within its frequency range; however, only those signals meeting the complete predetermined requirements of the code being used will be recognized and answered. This is a function of the decoder-coder, as directed and modified by switch settings within the decoder-coder and on the control panel. Mode 2 code settings are set into the receiver-transmitter on the ground and thus are fixed for any one flight. All other codes are set up at the control panel. Mode 1 is on whenever the equipment is operating. All other modes can be turned on or off at the control panel. Replies to modes 1, 2, and 3 interrogations, as well as to I/P and emergency replies, are shown on the ground station radar scope. In the case of the more complicated SIF codes, ground stations will use a plan position indicator (PPI) and letter symbol indicator to decode and indicate supplementary information, such as specific identification and location, and flight or airplane conditions. Mode 1 has 32 possible code combinations. Mode 2 has 4,096 combinations, though only a portion of these are usable with existing ground facilities. Mode 3 has 64 combinations. An optional low-power setting provision restricts sensitivity so that replies are made only to local interrogations. Airplane electrical power is supplied to the IFF system from the 115 volts ac essential bus and the 28 volts dc essential bus. See figure 1-37 for a listing of communications and avionics equipment.

IFF - SIF Control Panel



1. CODE SELECTOR KNOBS
2. IDENTIFICATION-OF-POSITION (I/P) SWITCH
3. MODE SWITCHES (2)
4. MASTER CONTROL KNOB
5. MASTER CONTROL KNOB LOCK RELEASE BUTTON C6510000-001A

Figure 1-29.

IFF MASTER CONTROL KNOB.

The five-position master control knob (4, figure 1-29), on the IFF-SIF control panel, controls operation of the IFF equipment. The knob positions are marked OFF, STDBY, LOW, NORM and EMERGENCY. When positioned to STDBY, the equipment is turned on and warmed up but will not transmit. When positioned to LOW, only local (strong) interrogations are recognized and answered. When positioned to NORM, full range recognition and replies occur. Transmitted power from the IFF system is the same for both the LOW and NORM positions. When the knob is positioned to EMERGENCY, an emergency-indicating pulse group is transmitted each time a Mode 1 or Mode 3 interrogation is recognized. The knob is prevented from being inadvertently moved to the EMERGENCY position by an internal lock. A lock release button, when depressed and held, allows the knob to be positioned to EMERGENCY. The knob can be moved out of the EMERGENCY position without pressing the lock release button.

IFF MODE SWITCHES.

The two-position mode switches (3, figure 1-29), marked MODE 2 and MODE 3 and OUT, are located on the IFF-SIF control panel. With either mode switch in the up (MODE 2 or MODE 3) position, the corresponding selected code will be transmitted to answer correctly coded interrogating reception. When either mode switch is positioned to OUT, its corresponding code is not transmitted.

IDENTIFICATION-OF-POSITION (I/P) SWITCH.

The identification-of-position (I/P) switch (2, figure 1-29), located on the IFF-SIF control panel, is used to control transmission of I/P pulse groups. The switch has three positions marked MIC, OUT, and

I/P. When the switch is momentarily held in the spring loaded I/P position, the I/P timer is energized for 30 seconds. If a mode 1 or mode 3 interrogation is recognized within this 30 second period, I/P replies will be made. When the switch is placed in the MIC position, the I/P pulse group will be transmitted in reply to a mode 1 or 3 interrogation as long as the microphone switch on either throttle is held to the TRANS position and for 30 seconds after the microphone switch is released. The transmitter selector knob, at the crew station being used, must be in the UHF position to allow transmission of I/P groups with the microphone switch. When the microphone switch is open, transmission of the I/P pulse groups will be withheld. Placing the switch to the OUT position prevents transmission of I/P groups.

SIF CODE SELECTOR CONTROL KNOBS.

Two code selector control knobs (1, figure 1-29), one for mode 1 and one for mode 3, are located on the IFF-SIF control panel. Each knob consists of an inner and outer selector ring each numbered counterclockwise. The inner selector ring of the mode 1 knob is marked 0, 1, 2, 3; the outer selector ring is marked 0, 1, 2, 3, 4, 5, 6, and 7. Each selector ring of the mode 3 selector knob is marked 0, 1, 2, 3, 4, 5, 6, and 7. Code numbers are read counterclockwise. For example, code 63 is selected by setting the 3 on the inner selector ring and the 6 on the outer selector ring to the index marker.

IFF ANTENNA SELECTOR SWITCH.

A two-position antenna selector switch (3, figure 1-30), located on the antenna select panel, is marked AUTO and LOWER. When the switch is placed in AUTO, the antenna lobing switch rapidly cycles contact of the receiver-transmitter between the upper and lower antenna to provide thorough antenna pattern coverage. If the lobing switch malfunctions or when the antenna selector switch is placed to LOWER only, the lower antenna will be used to receive and reply to interrogation signals.

IFF OPERATION.

1. Antenna selector switch - AUTO.
2. Code selector control knobs - As required.
3. Mode switches - As required.
4. I/P switch - OUT or MIC.
5. Master control knob - STDBY for approximately one minute then to LOW or NORM.

ATTACK RADAR (AN/APQ-113).

The attack radar provides all weather navigation, air-to-ground and air-to-air attack capability. The system may be operated independently or in conjunction with the bomb nav system in an air mode and three ground modes. Basic components of the

Antenna Select Panel

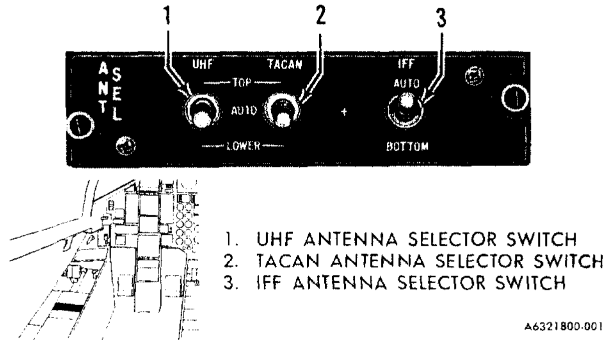


Figure 1-30.

system consists of an antenna, an antenna roll unit and antenna control, located in the radome and a modulator-receiver-transmitter (MRT) and synchronizer, located in the forward electronic bay. The radar scope and the controls, including the tracking control handle, are located at the pilot's station. A recording camera is provided to take radar scope photographs. The antenna is automatically stabilized in pitch and roll by signals from the bomb nav system or AFRS. Should the attitude signal fail the antenna can be caged in alignment with the airplane longitudinal and lateral axes. For location of the antenna, see figure 1-40. The MRT operates in the KU frequency band and has the capability of automatic frequency control (AFC) for normal operation and manual frequency control (MFC) for backup. When operating in AFC the transmitter is swept through the frequency band with random reversal. This provides a measure of immunity to many types of jamming and improves stability and legibility of returns. The synchronizer provides system timing, target declaration and range tracking. The radar scope panel contains the radar scope, recording camera and the necessary operating and tuning controls for the scope and camera. The recording camera is mounted behind the radar scope. A small window in the side of the cathode ray tube allows the camera to take exposures of the back of the radar scope. The image on the scope is reversed by optics so that the film exposure will represent the scope presentation as seen by the operator. A film exposure is taken automatically at weapon release on a signal from the bomb nav system or manually when desired. A lamp on the radar scope panel will blink each time an exposure is taken. A film magazine in the face of the radar scope panel provides a minimum of 500 exposures of 35 millimeter film. A readout window on the magazine shows film remaining. The magazine is installed or removed by means of a handle recessed in the front of the magazine. Simultaneous film exposure of a clock, data slate and 12 code lamps is made with each scope exposure to identify each frame of the film. The clock and slate provide time of exposure and operators name, date, mission etc. The code lamps are identified on the film and indicate the following:

1 - 5/15 miles diameter/range

2 - 10/30 miles diameter/range

3 - 30/90 miles diameter/range

4 - 80/160 miles diameter/range

5 - 160/160 miles diameter

6 - spare

7 - air mode

8 - ground manual mode

9 - ground auto mode

10 - ground velocity mode

11 - spare

12 - weapon release

The tracking control handle is used to position the azimuth and range cursors for fix taking, bombing and target tracking. Self test features incorporated into the system are used for preflight and maintenance malfunction analysis and troubleshooting. The system operates on 115 volt ac power from the left main ac bus and 28 volt dc power from the main dc bus. The system operates in two basic modes, air and ground.

ATTACK RADAR AIR MODE.

The air mode is used to search for, detect, acquire and track airborne targets until within missile range. In this mode, the antenna forms a pencil radar beam and scan is 90 degrees in azimuth (wide scan) for search and 20 degrees in azimuth (narrow scan) for tracking. Antenna elevation is controlled ±30 degrees by the tracking control handle. Once a target is detected the azimuth cursor is positioned with the tracking control handle to intercept the target and narrow scan is selected. In narrow scan a range cursor sweeps out in range automatically until the target is acquired, then range tracking is automatic. Both azimuth and elevation tracking must be manually performed with the tracking handle. On airplanes (10) (12)→ both range and azimuth tracking are automatic when a target is acquired. A target may be acquired more rapidly by overriding the automatic range sweep and positioning the range cursor on the target with the tracking control handle. A lock indicator lamp will light when the target is acquired. On airplanes (1)→(9) (11) elevation tracking arrows indicate the direction the antenna must be adjusted in tilt to track the target in elevation.

ATTACK RADAR GROUND MODES.

Three ground modes of operation are provided for radar navigation, fix-taking and fixed angle or automatic bombing. The ground modes are: ground manual (GND MAN), ground auto (GND AUTO) and ground velocity (GND VEL). In all three ground modes the antenna scans with a fan beam 45 degrees either side of the longitudinal axis. A drift stabilized map of the terrain is displayed on the scope at the range selected. Drift signals are provided from the bomb nav system. The ground manual mode operates independently of the bomb nav system and is used as the primary mode for radar navigation and as a back up method for fixed angle bombing in the event of a failure of the bomb nav system. Antenna tilt adjustment and azimuth and range cursor positioning is accomplished manually for navigation and fix-taking. If the antenna is caged the scope display is not ground track stabilized. The ground auto and velocity modes are used as primary modes of operation for fix taking and synchronous bombing. The system operates the same in either mode except that in the ground velocity mode the target or fix-point and the cursors are automatically maintained in the center of the scope. Antenna tilt is automatically adjusted by inputs from the bomb nav system, however corrections for scope display refinement can be made with the antenna tilt knob. The tracking control handle is used to drive the destination or present position counters in the bomb nav system for positioning the azimuth and range cursors.

ATTACK RADAR FUNCTION SELECTOR KNOB.

The attack radar function selector knob (3, figure 1-31), located on the attack radar control panel, has five positions marked OFF, STBY, ON, XMIT and TEST. In the OFF position the entire system is deenergized. Placing the switch to STBY supplies power to all system filaments for warm-up and energizes a 40 second warmup delay and a 5 minute transmitter high voltage delay. Also the antenna is caged in pitch and stowed full up in tilt and full left in azimuth. Placing the switch to ON energizes the entire system, except for the transmitter, after the 40 second warmup delay has expired. The XMIT position places the system in operation after the 5 minute high voltage delay has expired. The TEST position allows self test of the system for malfunction trouble shooting and ground maintenance.

ATTACK RADAR MODE SELECTOR KNOB.

The attack radar mode selector knob (1, figure 1-31), located on the attack radar control panel, has four positions marked GND MAN (ground manual), GND AUTO (ground auto), GND VEL (ground velocity) and AIR. In the GND MAN position the range and azimuth cursors are positioned with the tracking control handle independently of the navigational computer. In the GND AUTO position the cursors are automatically positioned by the navigational computer. The tracking control handle is used to correct the bomb nav system destination and present position counters. Operation in the GND VEL position is the same as in the GND AUTO position except the scope display is a ground velocity stabilized magnified picture and the intersection of the cursors remain in the center of the scope. In the AIR position the range and azimuth cursors are controlled by the tracking control handle for air-to-air search.

ATTACK RADAR FREQUENCY CONTROL KNOB.

The frequency control knob (2, figure 1-31), located on the attack radar control panel, has three positions marked AFC 1 (automatic frequency control), AFC 2, and MFC (manual frequency control). In the AFC 1 position the receiver operates in automatic frequency control and the transmitter operates in a frequency agility mode in which the transmitter sweeps through the frequency band with random reversal. The changing frequency and rapid scanning rate provided in this position provides immunity to many types of jamming and improves stability and legibility of the PPI display. In the AFC 2 position the receiver operates in automatic frequency control and the transmitter is manually tuned using the transmitter tuning control knob. The MFC position of the knob is variable over a range between the 12 and 6 o'clock positions. In this position the transmitter operates in a mid-band fixed frequency and receiver is manually tunable by adjusting the knob over the MFC range.

ATTACK RADAR ANTI-JAMMING SWITCH. (1)→(9) (11)

The anti-jamming switch (4, figure 1-31), located on the attack radar control panel, has three positions marked FTC (fast time constant), NORM and CCM (counter-countermeasures). The switch provides a means of selecting the desired receiver time response characteristics. The FTC and CCM positions are used to minimize the effects of jamming in any mode of operation; however, the CCM position has no effect when operating in the 160 or 160/80 nautical mile ranges. In the NORM position anti-jamming capabilities are inoperative.

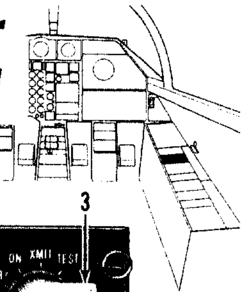
ATTACK RADAR FAST TIME CONSTANT SWITCH. (10)→ (12)

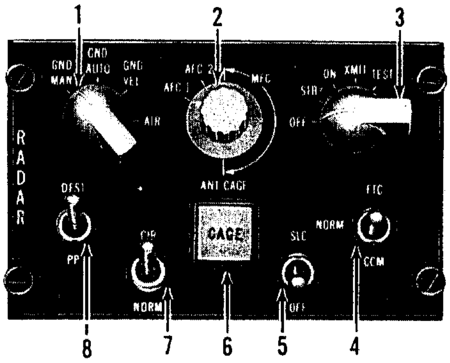
The fast time constant switch (4, figure 1-31), located on the attack radar control panel, has two positions marked FTC (fast time constant) and NORM. The switch provides a means of selecting the desired receiver time response characteristics. The FTC position is used to minimize the effects of jamming in any mode of operation. In the NORM position anti-jamming capabilities are inoperative.

ATTACK RADAR SIDE LOBE CANCELLATION SWITCH.

The side lobe cancellation (SLC) switch (5, figure 1-31), located on the attack radar control panel, is a two position switch marked SLC and OFF. Placing the switch to the SLC position will cancel the energy received from the side lobes of the radar beam to reduce ground clutter. The SLC position may be selected in any mode of operation, however, it is most effective when operating in the AIR mode at low altitudes.

Attack Radar Control Panel (Typical) (7)





1. MODE SELECTOR KNOB
2. FREQUENCY CONTROL KNOB
3. FUNCTION SELECTOR KNOB
4. ANTI-JAMMING SWITCH (1)→(9) (11)
   FAST TIME CONSTANT SWITCH (10)→(12)
5. SIDE LOBE CANCELLATION SWITCH
6. ANTENNA CAGE PUSHBUTTON INDICATOR LAMP
7. ANTENNA POLARIZATION SWITCH
8. DESTINATION/PRESENT POSITION SELECTOR SWITCH

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Figure 1-31.

ANTENNA POLARIZATION SWITCH.

The antenna polarization switch (7, figure 1-31), located on the attack radar control panel is a two position switch marked CIR (circular) and NORM. With the switch in the NORM position antenna polarization is horizontal when operating in the ground modes and vertical when operating in the air mode. Placing the switch to CIR changes antenna polarization to circular when operating in either ground or air modes. The CIR position may be used to reduce rain clutter interference on the scope.

DESTINATION/PRESENT POSITION SELECTOR SWITCH.

The destination/present position selector switch (8, figure 1-31), located on the attack radar control panel, is a two position switch marked DEST (destination) and PP (present position). The switch is used in the ground auto and ground velocity modes of operation when correcting the bomb nav system destination or present position counters.

ATTACK RADAR TRACKING CONTROL HANDLE.

The tracking control handle (figure 1-32) is mounted on a pivot pedestal on the right side of the pilot's station. The pedestal is stowed out of the way under the right canopy sill when not in use. To gain access to the tracking handle the pedestal is rotated outward. A blade type enable switch (3, figure 1-32), recessed in the front of handle must be depressed and held to activate the handle. When operating in the air mode

Attack Radar Tracking Control Handle

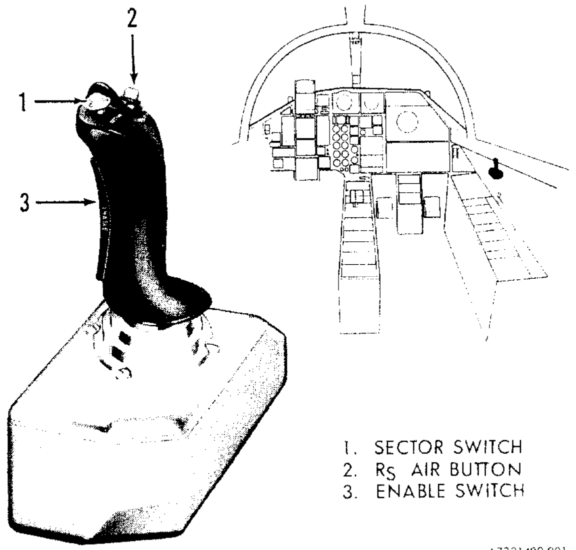


Figure 1-32.

with the range search button depressed or any of the three ground modes, fore and aft movement of the handle will slew the range cursor out or in respectively. Moving the handle fore and aft in the air mode without depressing the range search button will adjust antenna elevation down and up respectively. When operating in any mode left or right movement of the handle will slew the azimuth cursor left or right. Slewing speed is proportional to the amount of handle deflection.

ATTACK RADAR RANGE SEARCH BUTTON.

The range search button (2, figure 1-32), located on the right top of the tracking control handle, is used in the air mode of operation only. The button is labeled R_S AIR and must be depressed and held to break lock then the tracking control handle will override automatic range search when the range cursor is slewed fore and aft. When operating with the sector switch in the narrow sector position, range searching will resume from the point at which the range cursor was positioned by slewing. When operating with the sector switch in the wide sector position the range cursor will remain stationary after slewing.

ATTACK RADAR SECTOR SWITCH.

The sector switch (1, figure 1-32), located on the left top of the tracking control handle, is labeled SECTOR and is a two position, thumb actuated, sliding switch. The switch is used in either the ground or air modes of operation to change the sector of antenna sweep. In the aft position (wide scan) antenna sweep is 45 degrees either side of the longitudinal axis of the airplane. In the forward position (narrow scan) antenna sweep is 10 degrees either side of the azimuth cursor. When operating in the air mode and in narrow scan, automatic range searching is initiated.

ATTACK RADAR MANUAL PHOTO BUTTON.

The manual photo button (7, figure 1-33), located on the attack radar scope panel, provides a manual means of taking a film exposure of the radar scope display whenever desired. The button must be depressed each time a photo is desired.

INTERMEDIATE FREQUENCY GAIN KNOB.

The intermediate frequency gain knob (9, figure 1-33), located on the attack radar scope panel, is labeled IF GAIN and permits adjustment of receiver gain when operating in the ground modes.

ANTENNA TILT CONTROL KNOB.

The antenna tilt control knob (9, figure 1-33), located on the attack radar scope panel, provides a means of manually adjusting antenna tilt position when operating in the ground modes. The knob is labeled ANT TILT. In the ground manual mode the knob is the only means of adjusting antenna tilt. In the ground auto and ground velocity modes antenna tilt is automatically positioned by signals from the bomb nav system and the knob is used to refine this position. The knob has a detent corresponding to zero antenna tilt position for reference. Rotating the knob clockwise adjusts the tilt up to 30 degrees and vice versa. However, with the terrain following radar (TFR) operating the antenna can only be physically adjusted to -10 degrees to prevent interference with the TFR. Antenna position is indicated on the antenna tilt indicator (17, figure 1-33), located on the attack radar scope panel. The indicator is graduated in 5 degree increments from zero to ±30 degrees. The knob has no control over antenna tilt when operating in the AIR mode.

ATTACK RADAR SCOPE INTENSITY CONTROL KNOB.

The attack radar scope intensity control knob (14, figure 1-33), located on the attack radar scope panel, provides an adjustment of scope baseline intensity from zero to full brightness. Turning the knob clockwise increases brightness and vice versa. The knob is labeled CRT INT.

BEZEL/RANGE MARKS INTENSITY CONTROL KNOBS.

Two coaxial knobs (15, figure 1-33), located on the attack radar scope panel, provide an adjustment of bezel and range marks intensity. The knobs are labeled INT. The outer knob is marked BEZEL, and the inner knob is marked RANGE MK. Turning either knob clockwise increases intensity from zero to full brightness.

Attack Radar Scope Panel

1. SYSTEM MALFUNCTION LAMP
2. AZIMUTH BEZEL
3. RADAR SCOPE
4. GROUND ADJUSTMENT ACCESS DOOR
5. FILM MAGAZINE REMOVAL HANDLE
6. UNUSED FILM INDICATOR
7. MANUAL PHOTO BUTTON
8. PHOTO MALFUNCTION INDICATOR LAMP
9. INTERMEDIATE FREQUENCY GAIN/ANTENNA TILT CONTROL KNOBS
10. CURSOR RANGE COUNTER
11. SENSITIVITY TIME CONTROL KNOBS
12. VIDEO ADJUSTMENT/TRANSMITTER TUNING CONTROL KNOBS
13. RANGE SELECTOR KNOB
14. SCOPE INTENSITY CONTROL KNOB
15. BEZEL/RANGE MARKS INTENSITY CONTROL KNOBS
16. RANGE/AZIMUTH CURSOR INTENSITY CONTROL KNOBS
17. ANTENNA TILT INDICATOR
18. TEST SWITCH
19. SWEEP CONTROL SWITCH
20. BETA SWITCH
21. RANGE LOCK INDICATOR LAMP

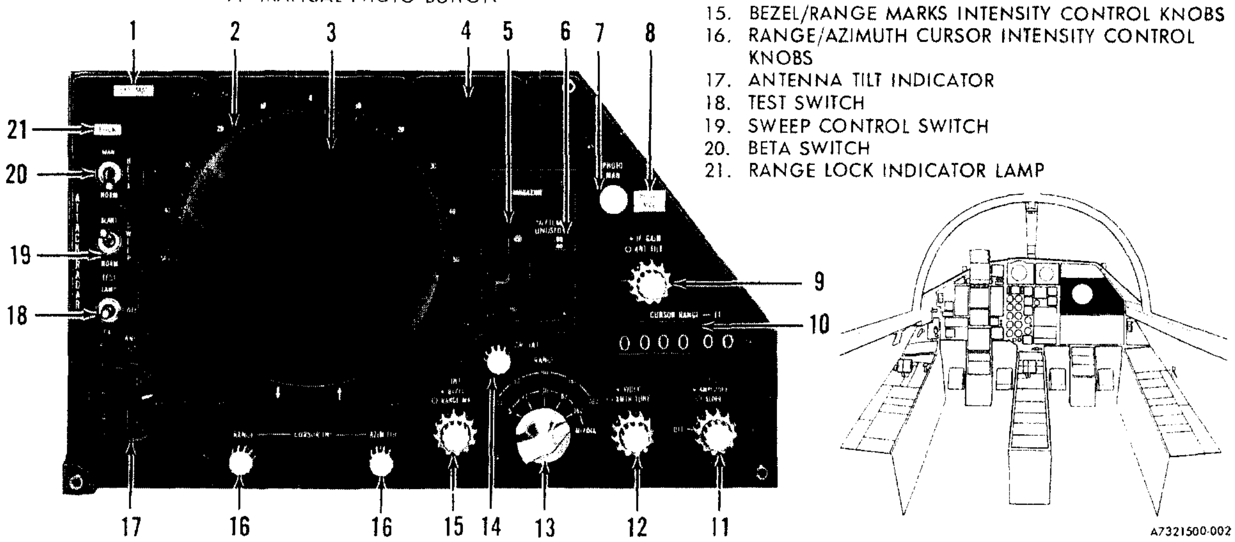


Figure 1-33.

RANGE AND AZIMUTH CURSOR INTENSITY CONTROL KNOBS.

Two cursor intensity control knobs (16, figure 1-33), located on the attack radar scope panel, provide an adjustment of range and azimuth cursor intensity. The knobs are labeled CURSOR INT and are individually marked RANGE and AZIMUTH. Turning either knob clockwise increases intensity of the respective cursor from zero to full brightness.

ATTACK RADAR TEST SWITCH.

The test switch (18, figure 1-33), located on the attack radar scope panel, is a three position switch marked LAMP, CKT (circuit) and OFF. The switch is used when performing preflight confidence and ground maintenance checks and is normally left in the OFF position.

ATTACK RADAR SWEEP CONTROL SWITCH.

The sweep control switch (19, figure 1-33), located on the attack radar scope panel, is a two position switch marked SLANT and NORM. The switch is used in the ground modes of operation to provide ground range in the NORM position and slant range in the SLANT position. The switch is inoperative in the air mode.

ATTACK RADAR BETA SWITCH.

The beta switch (20, figure 1-33), located on the attack radar scope panel, is a two position switch marked MAN (manual) and NORM. The switch functions in the ground auto and ground velocity modes to select automatic sighting angle in the NORM position and manual sighting angle in the MAN position. In the normal position sighting angle is automatically adjusted by signals from the bomb nav system. In the manual position sighting angle is adjusted with the antenna tilt knob. The switch is inoperative in the ground manual and air modes of operation.

SENSIVITY TIME CONTROL KNOBS.

Two coaxial rotary sensitivity time control (STC) knobs (11, figure 1-33), located on the attack radar scope panel, provide a means of equalizing radar intensity over the entire scope display when operating in the ground modes at low altitude. The outer knob labeled AMPL/OFF, has an OFF position at nine o' clock, and is used to obtain an initial adjustment of display intensity or to turn the STC function OFF in the event of a malfunction in the STC circuit. The inner knob, labeled SLOPE, is used to balance the display intensity throughout the sweep. The STC slope function is inoperative in the AIR mode.

VIDEO/TRANSMITTER TUNING CONTROL KNOBS.

Two coaxial rotary control knobs (12, figure 1-33), located on the attack radar scope panel, provide a means of adjusting video and tuning the transmitter. The outer knob, labeled VIDEO, is used to increase the amplitude of the video signal supplied to the attack radar scope when it is turned clockwise. The inner knob, labeled XMTR TUNE, allows continuous tuning of the transmitter over its entire frequency range when operating with the frequency control knob in the AFC-2 position.

ATTACK RADAR RANGE SELECTOR KNOB.

The range selector knob (13, figure 1-33), located on the attack radar scope panel, allows selection of various scope display ranges. The knob is marked RANGE with 15, 30, 90, 160, and 160 mile positions on an outer scale and miles/diameter (MI/DIA) with 5, 10, 30, 80 and 160 mile positions on an inner scale. The inner scale numbers and range are lighted when operating in air, ground manual and ground auto modes. When operating in the ground velocity mode both inner and outer scales and MI/DIA are lighted. This indicates that the scope presentation is the inner scale diameter of the outer scale range; for example with the knob in the 5/15 position the scope display is a 5 mile diameter portion of the 15 mile range.

ATTACK RADAR SCOPE.

The radar scope (3, figure 1-33) provides a sector scan plan position indicator (PPI) display with a fixed one radius offset sweep in all modes of operation except ground velocity mode. In ground velocity mode the sweep is a variable offset with a maximum displacement of six radii. The airplane position on the scope is at the bottom in vertical alignment with the center of the scope. The scope is 7 inches in diameter. The sector displayed is a 90 degree area ahead of the airplane when in wide scan and a 20 degree area centered on the azimuth cursor when in narrow scan. An azimuth bezel (2, figure 1-33), around the top of the scope, is graduated in one degree increments with each 10 degrees marked to show azimuth displacement up to 50 degrees either side of the airplane heading or ground track. When operating in the air mode, or when the antenna is caged in any mode, zero degrees on the scale represents airplane heading. In any of the ground modes the scan is displaced in azimuth to compensate for drift, and zero degrees represents ground track. In the air mode of operation two arrows in the bottom of the bezel indicate target position relative to antenna scan. When both arrows are lighted the target is in the center of the scan. Range and azimuth cursors are displayed on the scope for fix taking and target tracking. The cursors are positioned with the tracking control handle. Fixed range markers are provided for various ranges of operations. For 5, 10, 30, 80 and 160 ranges each range mark represents 1, 2, 5, 20 and 40 mile range increments respectively, except there are no range marks displayed in ground auto mode when in 5 and 10 mile range. Scope brilliancy and intensity of the bezel, cursors and range marks are controlled by knobs on the scope panel.

Note

> When the LCOS mode select knob is placed to the DIV BOMB, RKT AG or GUN AG position, the attack radar and TFR are used in conjunction with the LCOS for air-to-ground ranging. Under this condition the attack radar and TFR ground mapping or situation scope presentations will be unusable and should be ignored.

ATTACK RADAR CURSOR RANGE COUNTER.

The cursor range counter (10, figure 1-33), located on the attack radar scope panel automatically indicates slant range in all modes of operation. The counter has four digital readout windows which read distances up to 999,900 feet in increments of 100 feet.

ATTACK RADAR SYSTEM MALFUNCTION LAMP.

An amber system malfunction lamp (1, figure 1-33), located on the attack radar scope panel, provides the operator with an indication of a failure in the system. When lighted, SYS MAL is displayed on the face of the lamp.

Note

> If the system is not useable the function selector knob should be placed to STBY to stow the antenna.

ATTACK RADAR RANGE LOCK INDICATOR LAMP.

An amber range lock indicator lamp (21, figure 1-33), located on the attack radar scope panel, will light when a range lock is acquired on a target when operating in the air mode. When lighted, the word LOCK is visible on the face of the lamp.

PHOTO MALFUNCTION INDICATOR LAMP.

The photo malfunction indicator lamp (8, figure 1-33), located on the attack radar scope panel, provides an indication of camera operation or malfunctions. The lamp will blink each time a film exposure is made. The lack of a light indicates a camera shutter malfunction. A steady light indicates film breakage or failure of the film feed mechanism.

UNUSED FILM INDICATOR.

The unused film indicator (6, figure 1-33), located on the attack radar scope panel, is a digital readout indicator that displays the percent of film remaining in the magazine.

ANTENNA CAGE PUSHBUTTON INDICATOR LAMP.

The antenna cage pushbutton indicator lamp (6, figure 1-31), located on the attack radar control panel, provides a means of caging the antenna in the event of a failure in pitch or roll stabilization. The button is labeled ANT-CAGE. Depressing the button will cage the antenna pitch and roll axes and align the antenna with the longitudinal and lateral axes of the airplane, however, the antenna will continue to sector in azimuth, and tilt can be adjusted. When operating in the ground manual mode with the antenna caged the radar scope will display airplane heading at zero degrees azimuth and range sweep will be slant range instead of ground range. When the antenna is caged or when the function selector knob is in STBY position a lamp in the button will light displaying the word CAGE. Depressing the button again after the antenna has been caged will uncage the antenna and the lamp will go out. Should the bomb nav system stabilization platform fail the lamp will light and remain on until the flight instrument reference select switch is placed to AUX position.

ATTACK RADAR OPERATION.

Preflight.

Power Off.

1. Function selector knob - OFF.
2. Anti-jamming or fast time constant switch - NORM.
3. Side lobe cancellation switch - OFF.
4. Antenna polarization switch - NORM.
5. Destination/present position selector switch - DEST.
6. Mode selector knob - GND MAN.
7. Frequency control knob - AFC 1.
8. IF gain control knob - Full CCW.
9. Antenna tilt control knob - Detent.
10. Sensitivity time control knobs - Full CCW and OFF.
11. Video adjustment knob - Full CCW.
12. Range selector knob - 5/15 range.
13. Scope intensity control knob - Full CCW.
14. Bezel/range marks intensity control knobs - Full CCW.
15. Range/azimuth cursor intensity control knobs - Full CCW.
16. Test switch - OFF.
17. Sweep control switch - NORM.
18. Beta switch - NORM.

Power On.

1. Function selector knob - STBY.
   The function selector knob should remain in the STBY position for a minimum of 40 seconds for filament warmup.
2. Antenna cage pushbutton indicator lamp - Lighted.
3. Antenna tilt indicator - +30.
4. Function selector knob - ON.
   The function selector knob should remain in the ON position for a minimum of 5 minutes for system warmup.
5. Antenna cage pushbutton indicator lamp - OUT.
   If the antenna cage pushbutton indicator lamp remains on, depress the button to uncage the antenna. If the lamp still remains on check that the bomb nav mode selector knob is in a normal mode and that the flight instrument reference select switch is in the PRI position.
6. Antenna tilt indicator - Zero.
7. Scope intensity control knob - Desired intensity.
8. Bezel/range marks - Checked and set.
9. Range/azimuth cursors - Checked and set.
10. Function selector knob - XMIT.
11. Video adjustment knob - Tune for best picture.
12. IF gain knob - Tune for best picture.
13. Antenna tilt control knob - Adjust for best antenna coverage.
14. Range selector knob - Desired range.

LEAD COMPUTING OPTICAL SIGHT SYSTEM (AN/ASG-23).

The lead computing optical sight system (LCOS) provides the aircraft commander with information required to accurately deliver gun fire or missiles against aerial or ground targets and to deliver bombs or rockets against ground targets. The LCOS also provides homing, navigation, and landing information. The LCOS consists of a lead and launch computing amplifier, a lead computing gyro, and an optical sight and control panel. Information is displayed on the optical sight in the form of two presentations: an aiming reticle presentation and a command steering bar presentation. The LCOS utilizes 28 volt dc power from the main dc bus and 115 volt, three-phase, 400 cycle, ac power from the left main ac bus.

LCOS AIMING RETICLE.

The aiming reticle, (figure 1-34), which is a lighted image projected on the optical sight, consists of a 2 milliradian center pipper, a 30 milliradian circle, roll reference tabs, a 50 milliradian circle, fixed index reference tabs and range scale, an analog bar presentation, and two deviation indicators. All elements of the aiming reticle are lighted red and are fixed with respect to one another as the aiming reticle display moves about on the optical sight. Pitch limits of the aiming reticle are +2° to -11°. Azimuth limits are ±5.5°. The analog bar appears as a bar of light on the lower half of the 50 milliradian circle. The bar represents radar range deviation in gun, rocket, dive bomb, and GAR-8 modes and pitch angle deviation in loft bomb mode. When a target has been acquired by the attack radar system, the analog bar will appear. In the gun, rocket, and dive bomb modes, the 3 o'clock position of the analog bar represents the maximum range in feet as set on the control panel plus 3000 feet. The 6 o'clock position represents maximum range, and the 9 o'clock position represents maximum range minus 3000 feet. In the GAR-8 mode, the analog bar represents the range envelope of the GAR-8 missile. The 3 o'clock position represents the maximum missile firing range plus a fixed value. The 6 o'clock position represents the maximum range, and the 9 o'clock position represents minimum launch and breakaway range. In loft bomb mode, the analog bar represents deviation from a pitch angle preset into the glide/dive angle counter on the bomb nav control panel. During pull-up for a loft bomb delivery, the analog bar will recede from the 3 o'clock position (set angle minus 15°) toward the 9 o'clock position (set angle plus 15°). When the analog bar passes through the 6 o'clock position, the airplane pitch angle equals the preset angle and the LCOS will generate a weapon release signal. The three movable indices of the reticle display are the roll tabs. The roll tabs indicate airplane roll attitude. Roll tab reference indices are located at nine, twelve, and three o'clock positions on the 30 milliradian circle. The deviation indicators to the left and right of the reticle rings indicate aircraft performance and attitude deviations from preset conditions. The left hand deviation indicator displays deviation from G, pitch, or glide slope, depending on selected mode of operation. In the COM, GUN-AG, RKT-AG, DIVE BOMB, and HOM modes, the indicator displays deviation from the pitch angle set into the glide/dive angle counter on the bomb nav control panel. In the LOF BOMB mode, the indicator displays deviation from a fixed value of 4 G's. In the GAR-8, GUN-AA, and LEV BOMB modes, the indicator is zeroed. When the instrument system coupler mode selector switch is in ILS or AILA, the left deviation indicator will always present glide slope deviation regardless of

Aiming Reticle and Steering Bar Presentations

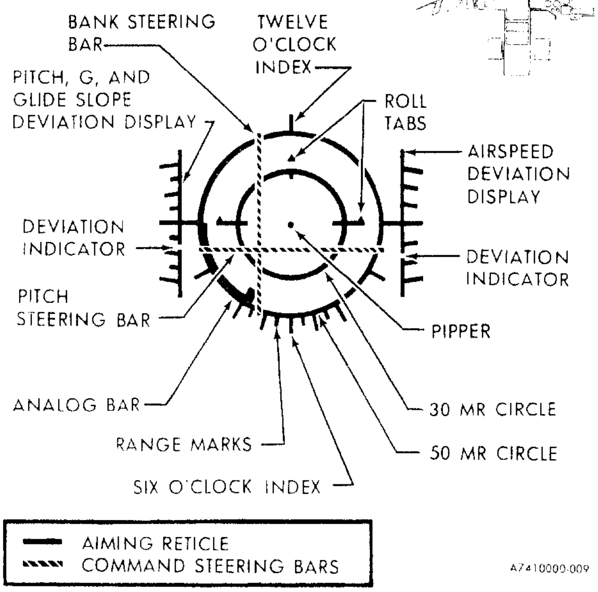


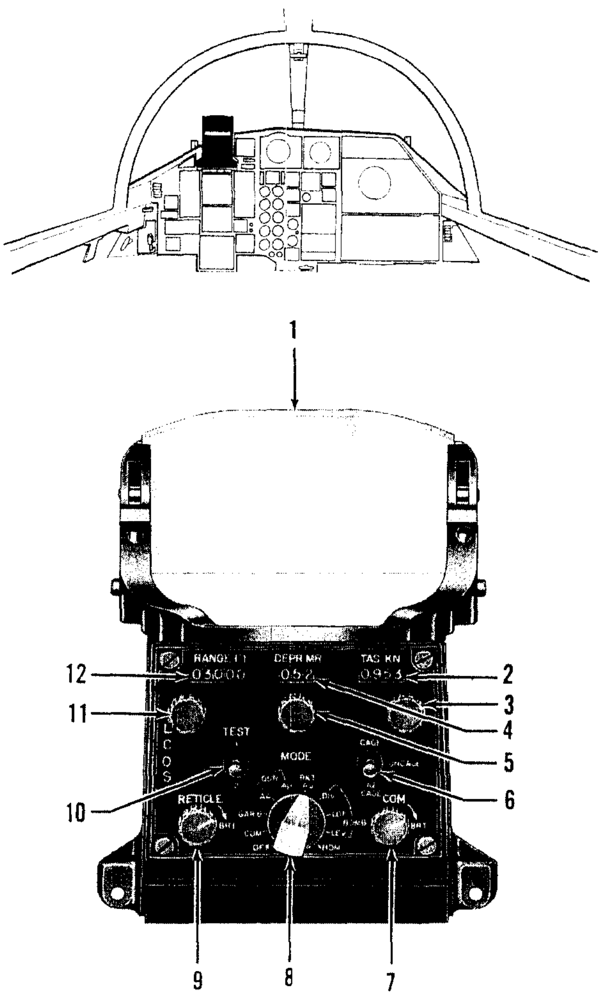
Figure 1-34.

the position of the LCOS mode select switch. The right hand indicator displays deviation from the airspeed preset into the true airspeed indicator on the LCOS control panel. In the GAR-8 and GUN-AA modes, the indicator is zeroed. In GAR-8 mode, a G limit indication is provided. The 6 o'clock index area of the 50 milliradian circle will be blanked out when the G limit of the GAR-8 missile is exceeded.

COMMAND STEERING BARS.

The command steering bars (figure 1-34) are presented as a vertical bank steering bar and a horizontal pitch steering bar. The steering bars are lighted green and are in parallel with the ADI steering bars. The aiming reticle pipper is used as the zero reference point for the steering bars. During a loft bomb delivery, at point of pull-up, the pitch steering bar will command a fly-up signal. The pitch steering bar indicates a deviation from 4 g's. Full deflection of the bar indicates a 3g deviation from the fixed 4g value. When operating the attack radar in the air mode, placing the instrument system coupler mode selector knob to the AIR/AIR position will provide both bank and pitch steering commands to the target. The command steering bars function independently of the LCOS mode select knob. The operator has the option of having the bars displayed by use of the command bar brightness knob.

Lead Computing Optical Sight and Control Panel



1. OPTICAL SIGHT
2. PRESET TRUE AIRSPEED INDICATOR
3. TRUE AIRSPEED SET KNOB
4. RETICLE DEPRESSION INDICATOR
5. RETICLE DEPRESSION SET KNOB
6. AIMING RETICLE CAGE SWITCH
7. COMMAND BAR BRIGHTNESS KNOB
8. MODE SELECT SWITCH
9. AIMING RETICLE BRIGHTNESS KNOB
10. TEST SWITCH
11. RANGE SET KNOB
12. PRESET RANGE INDICATOR

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Figure 1-35.

LCOS RANGE SET KNOB.

The range set knob (11, figure 1-35), located on the optical sight and control panel, is used to set in the six o'clock range index. A preset range indicator (12, figure 1-35), located directly above the knob will display the range in feet set in by the range set knob.

LCOS RETICLE DEPRESSION SET KNOB.

The reticle depression set knob (5, figure 1-35), located on the optical sight and control panel, is used to set in desired depression angles of the aiming reticle. A reticle depression indicator (4, figure 1-35), located directly above the set knob, indicates in milliradians the reticle depression set by the depression set knob.

TRUE AIRSPEED SET KNOB.

A true airspeed set knob (3, figure 1-35), located on the optical sight and control panel, is used to set in a desired true airspeed. A preset true airspeed indicator (2, figure 1-35), located directly above the set knob, indicates true airspeed in knots set in by the true airspeed set knob.

LCOS TEST SWITCH.

The LCOS test switch (10, figure 1-35), located on the optical sight and control panel, is provided to allow an operational check and a fault isolation check to be performed on the LCOS while installed in the airplane without the aid of test equipment. The switch has positions 1 and 2 and is spring-loaded to the center OFF position. Position 1 is used for performing in-flight and ground self tests. Position 2 is used for performing ground fault isolation tests.

AIMING RETICLE CAGE SWITCH.

The aiming reticle cage switch (6, figure 1-35), located on the optical sight and control panel, has positions AZ CAGE, CAGE, and UNCAGE. In the AZ CAGE position, the aiming reticle is mechanically caged in azimuth only. In the CAGE position the aiming reticle is mechanically caged to the armament datum line. In the UNCAGE position, the aiming reticle is free to move in azimuth and in elevation.

AIMING RETICLE CAGE BUTTON.

The aiming reticle cage button (38, figure 1-4), located under the forward contour of the aircraft commander's left throttle, is a push button marked CAGE. Depressing and holding the button will cage the aiming reticle to the armament datum line.

AIMING RETICLE BRIGHTNESS KNOB.

The aiming reticle brightness knob (9, figure 1-35), located on the optical sight and control panel, is provided to adjust the brilliance of the aiming reticle. Rotating the knob full clockwise will provide full brilliancy. Rotating the knob full counterclockwise will turn off the aiming reticle.

LCOS Mode Select Knob Positions Versus Indications

MODE SELECT KNOB POSITIONS	AIMING RETICLE	ANALOG BAR	PITCH, G, AND GLIDE SLOPE DEVIATION INDICATOR	AIRSPEED DEVIATION INDICATOR
OFF	LCOS deenergized			
COMM	Positioned to preset depression angle. Caged in azimuth	Not displayed	Deviation from preset pitch angle	Deviation from preset airspeed
GAR-8	Caged, with pipper on armament datum line	Range envelope of GAR-8 missile	Zero (Inoperative)	Zero (Inoperative)
GUN-AA	Positioned to computed lead angle	Deviation from preset maximum gun range	Zero (Inoperative)	Zero (Inoperative)
GUN-AG	Positioned to preset depression angle. Caged in azimuth	Deviation from preset range	Deviation from preset pitch angle	Deviation from preset airspeed
RKT-AG	Positioned to preset depression angle. Caged in azimuth	Deviation from preset range	Deviation from preset pitch angle	Deviation from preset airspeed
DIV BOMB	Positioned in elevation by manual depression and in azimuth by drift angle	Deviation from preset range	Deviation from preset pitch angle	Deviation from preset airspeed
LOF-BOMB	Positioned in elevation by manual depression and in azimuth by drift angle	Deviation from preset pitch angle	Deviation from 4G	Deviation from preset airspeed
LEV-BOMB	Positioned in elevation by manual depression and in azimuth by drift angle	Not displayed	Zero (Inoperative)	Deviation from preset airspeed
HOM	Positioned by radar homing and warning signals	Not displayed	Deviation from preset pitch angle	Deviation from preset airspeed

Figure 1-36.

COMMAND BAR BRIGHTNESS KNOB.

The command bar brightness knob (7, figure 1-35), located on the optical sight and control panel, is provided to adjust brilliance of the command steering bars. Rotating the knob full clockwise will provide full brilliancy. Rotating the knob full counterclockwise will turn off the command steering bars.

LCOS MODE SELECT KNOB.

The LCOS mode select knob (8, figure 1-35), located on the optical sight and control panel, is used to select the various modes of LCOS operation.

Note

When the LCOS mode select knob is placed to the DIV BOMB, RKT AG or GUN AG, the attack radar and TFR are used in conjunction with the LCOS for air-to-ground ranging. Under this condition the attack radar and TFR ground mapping or situation scope presentations will be unusable and should be ignored.

Refer to figure 1-36 for LCOS indication in the various knob positions.

LCOS OPERATION.

1. LCOS mode select knob - Set to mode compatible with weapon(s) being carried.
2. Aiming reticle and command bar brightness knobs - Adjust for desired brilliance.
3. Aiming reticle cage switch - UNCAGE.
4. Range set knob - Set to required range if operating in GAR-8, Gun, RKT AG, or dive bomb modes.
5. Reticle depression set knob - Set to desired value if operating in GUN-AG, RKT-AG, dive bomb, level bomb, or loft bomb modes.
6. True airspeed set knob - Set to desired airspeed.
7. Glide/dive angle counter - Set to desired angle if operating in GUN-AG, RKT-AG, dive bomb, or loft bomb modes.

Refer to classified Supplement, 1F-111(Y)A-1B, for information pertaining to the following systems:

RADAR HOMING AND WARNING SYSTEM (AN/APS-109).

INFRARED COUNTERMEASURES RECEIVER (AN/ALR-23).

COUNTERMEASURES DISPENSER (AN/ALE-28).

TRACK BREAKER.

COMMUNICATIONS EQUIPMENT.

For a listing and function of communications equipment see figure 1-37.

UHF COMMAND RADIO (AN/ARC-51).

The UHF command radio provides air-to-air and air-to-ground communications and automatic direction finding (ADF) in conjunction with the AN/ARA-50. The radio equipment consists of a receiver-transmitter (RT) unit, a guard receiver, a control panel, an antenna selector, a blade type upper antenna, a conical type lower antenna and a loop ADF antenna. See figure 1-40 for antenna locations. There are 3500 channels available in 50 kilocycle increments in the frequency range from 225.0 to 399.9 mc (megacycles). The RT unit and guard receiver are located in the right forward equipment bay. The receiver section of the RT unit provides ADF bearing signals to the number two pointer of the bearing distance heading indicator (BDHI) and audio to the interphone when the ADF function is selected. The guard receiver monitors the guard frequency of 243.0 mc when guard function is selected. The control panel allows selection of 20 preset channels and manual selection of any frequency in the frequency range of the radio. The upper and lower antenna compliment each other to provide omni-directional antenna coverage. An automatic feature allows the receiver to select the antenna which receives the first usable signal; however, either the upper or lower antenna may be manually selected. The upper antenna also serves as the TACAN antenna. When the ADF function is selected the receiver is connected to the ADF loop antenna. The UHF radio operates on 115 volt ac power from the ac essential bus and 28 volt dc power from the essential dc bus. The ADF operates on 115 volt ac power from the right main ac bus and 28 volt dc power from the main dc bus.

UHF Radio Function Selector Knob.

The UHF radio function selector knob (4, figure 1-38), located on the UHF radio control panel, has four positions marked OFF, T/R, T/R + G and ADF. Rotating the knob to the T/R position activates the RT unit for normal transmission and reception on the channel selected; the guard receiver is inoperative. Rotating the knob to the T/R + G position also activates the RT unit for normal use and in addition activates the guard receiver to allow monitoring guard frequency. In the ADF position the receiver is switched to the ADF loop antenna and bearing information and audio are supplied to the BDHI and interphone respectively. If the microphone switch is held to TRANS while in the ADF position the UHF antennas are switched back into the circuit and the ADF is disabled until the microphone switch is released. The ADF position of the switch is inoperative when operating on emergency electrical power.

UHF Radio Mode Selector Knob.

A three position UHF radio mode selector knob (7, figure 1-38), located on the UHF radio control panel, permits selection of the desired operating mode. The knob is marked PRESET CHAN, MAN, and GD XMIT. The PRESET CHAN position is used when selecting one of the 20 preset frequencies. The MAN position is used when utilizing frequencies that are selected by the manual frequency selector knobs. The GD XMIT position tunes the main RT unit to the guard frequency of 243.0 megacycles.

UHF Radio Preset Channel Selector Knob.

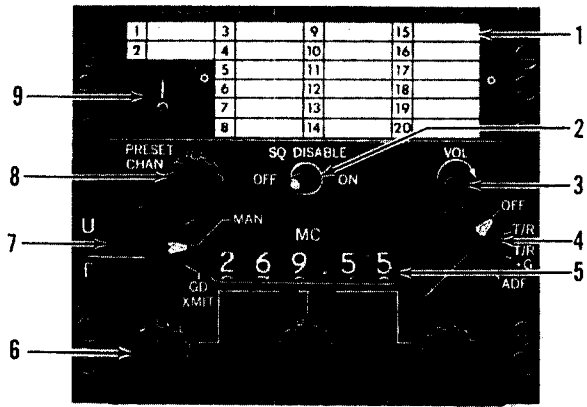
The preset channel selector knob (8, figure 1-38), located on the UHF radio control panel, permits selection of one of twenty preset frequencies. With the mode selector switch at PRESET, movement of the preset channel selector knob changes the frequency to that of the channel selected. There are 20 channels, numbered 1 through 20, that may be individually selected. Frequencies for each channel are written on a channel frequency log on the face of the control panel. Frequencies of the preset channels can be changed during flight.

Communications and Avionics Equipment

TYPE	DESIGNATION	FUNCTION	OPERATOR	RANGE	CONTROL LOCATION
UHF COMMAND RADIO	AN/ARC-51	Air-to-air and air-to-ground voice communication	AC Pilot	Line-of-Sight	Right Main Instrument Panel
UHF ADF	AN/ARA-50	Provides bearing information to selected UHF stations	AC Pilot	Line-of-Sight	Right Main Instrument Panel
HF RADIO	AN/ARC-112	Air-to-air and air-to-ground long range voice communications	AC Pilot	5000 miles	Right Console
INTERPHONE	AN/AIC-18 or AN/AIC-25	Interphone between crew members and monitoring of all communications facilities	AC Pilot		Left & Right Console
IDENTIFICATION RADAR (IFF-SIF)	APX 46 A/G	Provides coded IFF replies to an interrogating ground radar station	Pilot	Line-of-Sight	Right Pedestal
TACAN	AN/ARN-52	Provides bearing and distance information to TACAN stations	AC Pilot	Line-of-Sight up to 300 NM	Right Main Instrument Panel
ILS	AN/ARN-58	Provides visual indications for ILS letdowns	Pilot	Localizer 45 NM Glide Slope 25 NM	Right Pedestal
RADAR ALTIMETER	AN/APN-67	Provides precise altitude measurements from 0 to 5,000 feet	AC	0-5000 feet	Left Main Instrument Panel
TERRAIN FOLLOWING RADAR	AN/APQ-110	Provides all weather, low altitude terrain following, obstacle avoidance and blind letdown capability	Pilot	Line-of-Sight up to 15 miles	Right Main Instrument Panel
ATTACK RADAR	AN/APQ-113	All weather navigation, fixtaking, bombing, and air-to-air attack	Pilot	Line-of-Sight up to 160 miles	Right Main Instrument Panel and Right Console
LEAD COMPUTING OPTICAL SIGHT	AN/ASG-23	Provides air-to-air and air-to-ground attack capability and duplicate information as displayed on ADI for instrument flying	AC	Line-of-Sight	Left Main Instrument Panel

Figure 1-37.

UHF Radio Control Panel



1. CHANNEL FREQUENCY LOG
2. SQUELCH DISABLE SWITCH
3. VOLUME CONTROL KNOB
4. FUNCTION SELECTOR KNOB
5. FREQUENCY INDICATOR WINDOW
6. MANUAL FREQUENCY SELECTOR KNOBS (3)
7. MODE SELECTOR KNOB
8. PRESENT CHANNEL SELECTOR KNOB
9. PRESENT CHANNEL INDICATOR WINDOW

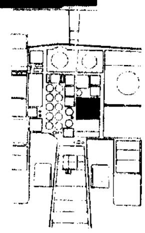


Figure 1-38.

UHF Radio Manual Frequency Selector Knobs.

Three UHF radio manual frequency selector knobs (6, figure 1-38), located on the UHF radio control panel, are provided for manually selecting frequencies. Manual frequency selection can be made in steps of 50 kilocycles from 225 through 399.95 megacycles. The first selector knobwill select the first two digits of the desired frequency with a range of 22 through 39 in multiples of 10. The second knob selects the third digit and has a range of 0 through 9. The third knob selects the last two digits in multiples of 0.05 and has a range of 0.00 through 0.95. The selected frequency is displayed in a window on the face of the UHF radio control panel.

UHF Radio Volume Control Knob.

The volume control knob (3, figure 1-38), located on the UHF radio control panel, increases the volume of the receiver when turned clockwise and decreases it when turned counterclockwise.

Squelch Disable Switch.

The squelch disable switch (2, figure 1-38), located on the UHF radio control panel, is a two-position switch marked ON and OFF. The switch is provided so that the squelch can be selected for compatibility with the strength of the signal being received. Placing the switch to ON disables (turns off) the squelch. Placing the switch to OFF turns the squelch on.

Transmitter Selector Knobs.

Two transmitter selector knobs (3, figure 1-41), marked HF, UHF, and INT, are located on the left and right interphone control panels to select either the HF or UHF radio or interphone as desired for transmission.

Microphone Switches.

A three position spring loaded, sliding type microphone switch, marked TRANS and INPH with an unmarked off center position, is located on each right throttle (19, figure 1-4 and 7, figure 1-17). The switch is moved up to the TRANS position for radio transmissions or down to INPH for interphone operation. It is spring loaded to the center off position.

UHF Radio Antenna Selector Switch.

The three position UHF radio antenna selector switch (1, figure 1-30), located on the antenna select panel, controls the selection of the upper and lower UHF antennas. The switch is marked UPPER, AUTO, and LOWER. Placing the switch to AUTO causes the antenna selector to control the antenna switching relay to select the correct antenna. Placing the switch in the LOWER or UPPER position controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

UHF Radio Frequency Indicator Window.

The UHF radio frequency indicator window (5, figure 1-38), located on the UHF radio control panel, indicates the frequency selected for transmission or receiving. The window has five digits, the first two digits and last two digits are set by frequency selector knobs to the left and right of window. The third or center digit is set by a knob directly below the window.

UHF Radio Operation.

1. Transmitter selector knob - UHF.
2. Antenna selector switch - AUTO.
3. Function selector knob - As required (T/R or T/R + G).
4. Mode selector knob - PRESET or MAN.
   If MAN position is used, the desired frequency must be set with the manual frequency selector knobs.
5. Preset channel selector knob - Desired channel.

6. Volume control knob - Desirable level.

7. Squelch disable switch - As required.

8. Microphone switch - TRANS.

Automatic Direction Finder Operation.

Homing.

1. Function selector knob - ADF.

2. Mode selector knob - PRESET or MAN. Select the desired frequency with the preset channel selector knob or with the manual frequency selector knobs.

3. Turn the airplane until the No. 2 pointer of the BDHI points to the fixed index mark at the top of the instrument.

4. To turn off, move the function selector switch from the ADF position.

Direction Finding.

1. Function selector knob - ADF.

2. Mode selector knob - PRESET or MAN. Select the desired frequency with the preset channel selector knob or with the manual frequency selector knobs.

3. Observe the heading of the No. 2 pointer on the BDHI.

4. To turn off, move the function selector switch from the ADF position.

HF RADIO (AN/ARC-112).

The HF radio provides long range high frequency single side band air-to-air and air-to-ground communications. The radio operates in three modes: USB, upper side band; LSB, lower side band; and AM, upper side band with amplitude-modulated carried for receivers without single side band reception capability. There are 28,000 channels available in 1 kilocycle increments in the frequency range of 2,000 through 29,999 kilocycles. Components of the radio include a receiver-transmitter (RT) unit, amplifier power supply, antenna, antenna coupler, antenna coupler control and control panel. The RT unit, amplifier power supply and antenna coupler control are located in the right forward electronic bay. The antenna coupler is located in the aft fuselage below the antenna which is a part of the vertical stabilizer and dorsal fin. The antenna is impedance matched to the receiver-transmitter. The system incorporates self test features for maintenance troubleshooting. The control panel is located at the pilot's station on the right console. Once the system is placed in operation either crew member can use the equipment. The radio operates on 115 volt ac power from the right main ac bus. Procedures for normal operation of the HF radio are contained within the appropriate portion of Section II.

HF Radio Mode Selector Knob.

The HF radio mode selector knob (5, figure 1-39), located on the HF radio control panel, has six positions marked OFF, USB, LSB and AM with two TEST positions marked RCVR EXC and XMIT. In the OFF position the system is deenergized. Moving the knob from OFF to any other position applies power to the system. Placing the knob to USB or LSB provides upper or lower side band transmission and reception respectively. The AM position provides upper side band transmission with an amplitude-modulated carrier for reception by receivers without single side band capability. The TEST positions provide self test features used for maintenance troubleshooting.

HF Radio Frequency Selector Knobs.

Four HF radio frequency selector knobs (1, figure 1-39), located on the HF radio control panel, provide a means of setting desired frequencies. Each knob has an indicator line drawn to the window(s) it controls on the frequency indicator.

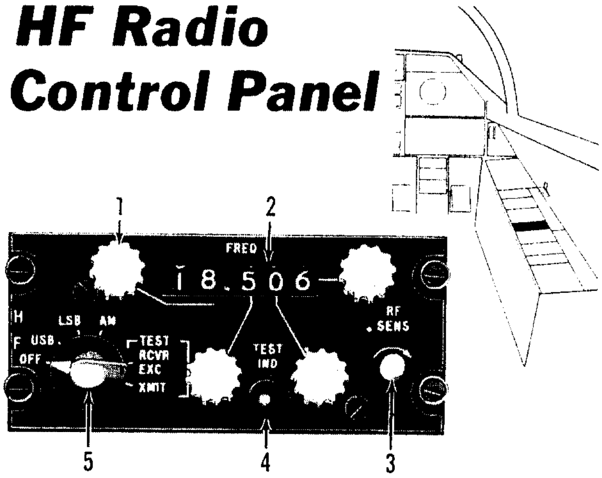
HF Radio Frequency Indicator Window.

The HF radio frequency indicator window (2, figure 1-39), located on the HF radio control panel, has five digits indicating the frequency selected for transmission or receiving. Each window has an indicator line drawn to its corresponding frequency selector knob.

Radio Frequency Sensitivity Knob.

The radio frequency sensitivity knob (3, figure 1-39), located on the HF radio control panel, is labeled RF SENS. The knob provides an adjustment for receiver sensitivity.

HF Radio Control Panel



1. FREQUENCY SELECTOR KNOBS (4)
2. FREQUENCY INDICATOR WINDOW
3. RADIO FREQUENCY SENSITIVITY KNOB
4. TEST INDICATOR LAMP
5. MODE SELECTOR KNOB

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Figure 1-39.

Antenna Locations

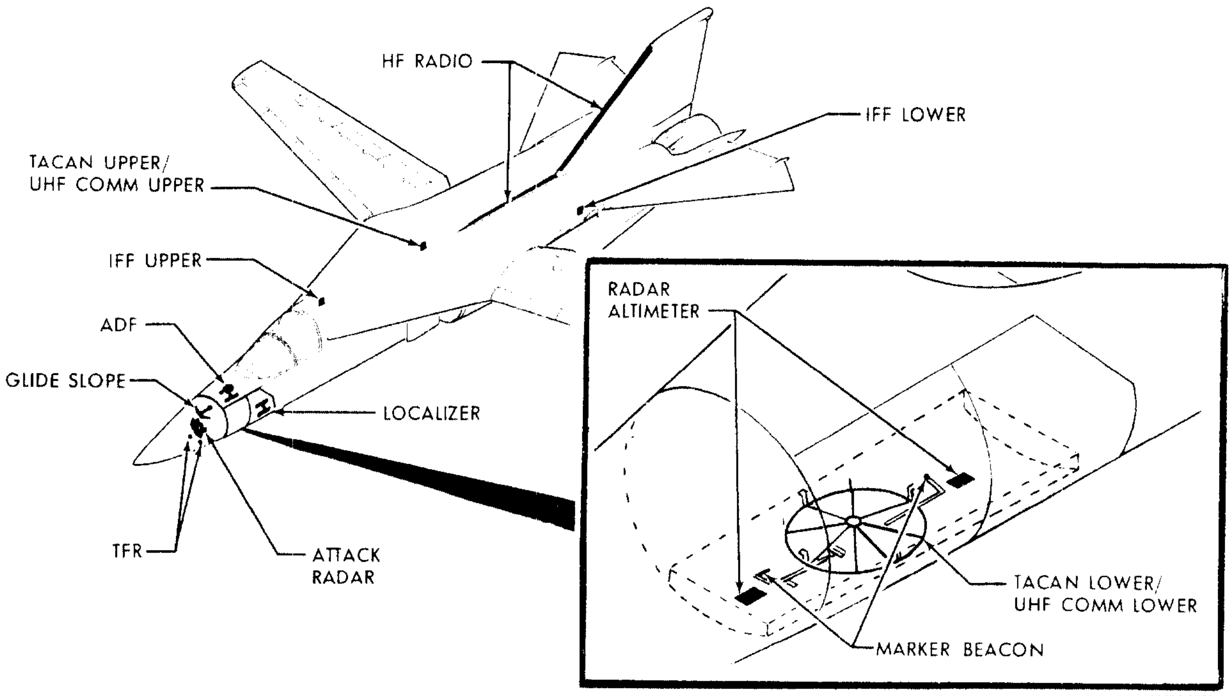

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Figure 1-40.

HF Radio Test Indicator Lamp.

The green test indicator lamp (4, figure 1-39), located on the HF radio control panel, provides both self test and malfunction indications. The lamp is used in conjunction with the mode selector knob test positions for system self test during maintenance trouble shooting. When the mode selector knob is in a normal operating position a flashing lamp will indicate a system malfunction.

HF Communication System Operation.

1. Transmitter selector knob - HF.
2. HF monitor knob - On.
3. Mode selector knob - Desired mode.
4. Desired frequency - Set.
   A mute period will indicate the RT unit is setting to the new frequency. The system should not be keyed during this period. If the frequency was already set when the system was turned on, rotate the 1-kc knob one digit aft frequency and then back to the desired frequency. This will allow the R-T unit to properly tune to the desired frequency.
5. RF sensitivity knob - Adjusted.
   Adjust the RF sensitivity knob to receive signals just above the noise level of the receiver, then adjust the interphone monitor knob for a comfortable listening level. Proper balance is indicated when background noise is just audible and a weak signal is raised to comfortable level.
6. Microphone switch - TRANS.
   After a frequency change, a 1000 cycle tone will be heard when the "mike" switch is first placed to TRANS. This indicates that the RT unit and antenna coupler are tuning. When the tone ceases, the tuning cycle is complete and a side tone will be heard when transmitting. Lack of a side tone indicates the pressure in the amplifier power supply or antenna coupler is below 15.7 PSIA.

INTERPHONE (AN/AIC-18 OR AN/AIC-25).

The interphone provides the following functions: Communications between crew members and between crew members and ground crew; monitoring and volume control of UHF radio, HF radio, TACAN, ILS, marker beacon, RHAW and missile tone reception; and hot mic and call capability. Two identical interphone control panels (figure 1-41) located on the left and right consoles are provided for the aircraft commander and pilot. Interphone stations for ground crew operation are located in the nose wheel well and main landing gear well. The interphone operates on 28 volt dc power from the essential dc bus. Power is applied to the interphone whenever power is on the airplane.

Communications Monitor Knobs.

Eight push-pull communications monitor knobs (1, figure 1-41), located on each interphone control panel, are marked and monitor the functions as follows:

INT	-	Interphone
UHF	-	UHF Command Radio
HF	-	HF Radio
ILS	-	ILS and Localizer
MB	-	Marker Beacon
TACAN	-	TACAN Identification
RHAWS	-	Radar Homing and Warning System
MISSILE	-	Missile Tones

Other signals fed to the interphone panel are a landing gear warning tone and a reduce speed warning tone. The monitor knobs are pulled out to turn on and pushed in to turn off. When pulled out, each knob may be rotated for volume control.

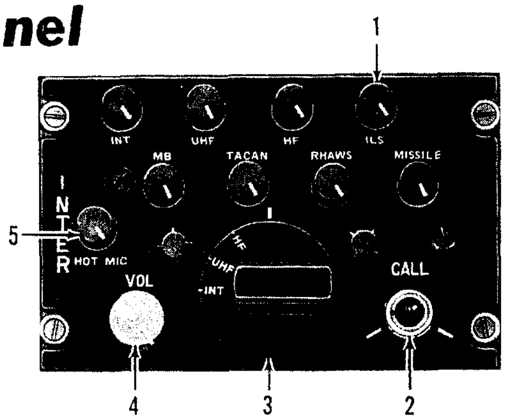
Master Volume Control Knob.

A master volume control knob (4, figure 1-41), located on each interphone control panel, controls the volume of all of inputs to the panel. If a change to an individual input volume is desired, it can be accomplished by rotating the appropriate monitor knob.

Hot Microphone Button.

A push-pull (HOT MIC) hot microphone button (5, figure 1-41), located on each interphone control panel, provides a continually operating microphone when it

Interphone Control Panel



1. MONITOR KNOBS (8)
2. INTERPHONE CALL BUTTON
3. TRANSMITTER SELECTOR KNOB
4. MASTER VOLUME CONTROL KNOB
5. HOT MICROPHONE BUTTON

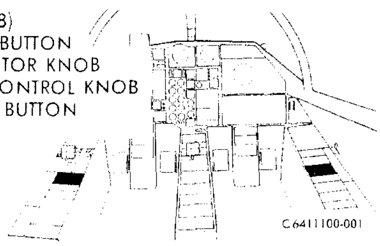


Figure 1-41.

is pulled. When this switch is pulled, the crew member can talk on the intercommunication system without using the microphone switch.

Note

The use of HOT MIC when there is a high background noise level in the cockpit will interfere with UHF communications.

Interphone Call Button.

The interphone call button (2, figure 1-41), located on the interphone control panel, enables the crew member to call the other crew members or the ground crew regardless of the positions of the monitor knobs or master volume control knob. When depressed, the button boosts the interphone volume, reduces the operator's own sidetone level, and allows other sta-

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tions to receive the call signal regardless of the position of the other stations' monitor knobs or transmitter selector knob. The position of the button does not affect other selected audio signals.

Transmitter Selector Knobs.

Two three position transmitter selector knobs (3, figure 1-41), located on each interphone control panel, are provided to select either UHF or HF radio or interphone as desired for transmission. The knobs are marked UHF, HF and INT. In either the HF or UHF positions only the radio transmitter selected will be keyed when the microphone switch is moved up to TRANS position. Regardless of the position of the transmitter selector switch, the interphone may be used by moving the microphone switch on the throttle down to the INPH position. The INT position of the switch has no function other than enabling use of the interphone system by moving the microphone switch either up or down.

Microphone Switch.

A three position sliding type microphone switch, marked TRANS and INPH with an unmarked OFF position, is located on each right throttle (19, figure 1-4 and 7, figure 1-17). The switch is moved up to TRANS position for radio transmissions or down to INPH position for interphone transmission. The pushbutton is spring loaded to the center OFF position. When the transmitter selector switch is in the INT position, moving the switch to either position allows interphone use.

Exterior Interphone Stations.

Exterior interphone stations in the nose wheel well and the main landing gear wheel well have a volume control knob, a call pushbutton, and a receptacle for ground cord plug in. The call pushbutton and volume control knob function the same as these controls on the interphone control panel.

Interphone Operation.

1. Aircraft electrical power - On.
2. Monitor knobs - Pulled as required.
3. Master volume control knob - Adjusted for desired volume.

LIGHTING SYSTEM.

The lighting system is divided into external and internal lights.

EXTERIOR LIGHTING.

The exterior lights include; position lights, formation lights, anti-collision/fuselage lights, air refueling lights, landing lights and a taxi light. The position lights consist of a red left wing tip light, a green right wing tip light, and a white tail light. On airplanes (12)→ an additional position light is mounted in each wing glove. The light in the right glove is red and the light in the left glove is green to correspond to the lights in the right and left wing tip. On these airplanes the wing tip position lights will light when the wing sweep angle is between 16 and 30 degrees. When the wings are swept aft of 30 degrees the wing tip light will go out and the glove light will light. The reverse will occur as the wings are swept forward. The formation lights consist of a set of three lights, located on the upper and lower surfaces of each wing tip (green on the right wing and red on the left wing), and four yellow lights located forward and aft on each side of the fuselage. Two anti-collision/fuselage lights, one located on top and one located on the bottom of the fuselage, serve as white fuselage lights when retracted and flashing red anti-collision lights when extended. Two air refueling lights mounted in the air refueling receptacle are provided for night refueling operations. A limit switch on the air refueling receptacle door provides power to the receptacle light control knob when the door is open. Two landing lights and a taxi light are located on the nose landing gear. A limit switch on the nose gear doors will turn the lights off if they are on when the gear is retracted.

Position Light Switches.

Three position light switches (4, figure 1-42), are located on the lighting control panel. Two switches, labeled WING and TAIL, have three positions, marked BRT (bright), OFF and DIM, for selecting the desired intensity of the position lights. The third switch is a two position switch marked FLASH and STEADY to control the operation of the position lights. Placing the switch to FLASH causes the position lights to flash at a rate of 80 cycles per minute. The switches control 28 volt dc power from the engine start bus.

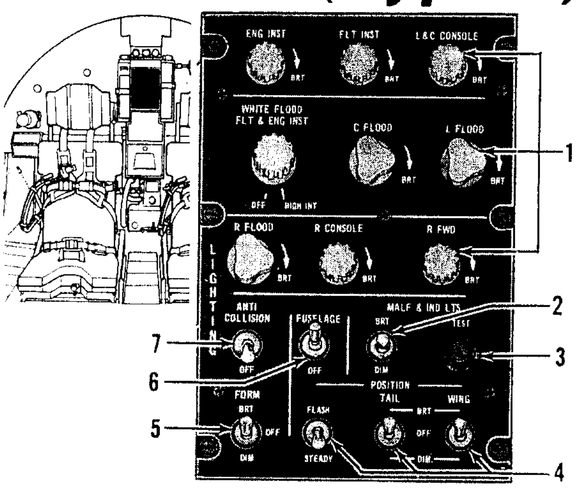
Position Lights/Stores Refuel Battery Power Switch. (12)→

The position lights/stores refuel battery power switch (4, figure 1-16), located on the ground check panel, has three positions marked POS LIGHTS, NORM and STORES REFUEL. Placing the switch to the POS LIGHTS position will supply battery power to the position lights for added safety during ground handling. Placing the switch to NORM deenergizes the circuit. The switch is held in the NORM position when the ground check panel door is closed. For a description of the STORES REFUEL position of the switch refer to the Fuel Supply System this section.

Formation Lights Switch.

The formation lights switch (5, figure 1-42), located on the lighting control panel, provides selection of the desired intensity of the lights. The switch is marked BRT (bright), OFF and DIM and controls 28 volt dc power from the main dc bus.

Lighting Control Panel (Typical)



1. INTERNAL LIGHTING CONTROL KNOBS (9)
2. MALFUNCTION INDICATOR LAMP DIMMING SWITCH
3. MALFUNCTION INDICATOR LAMP TEST BUTTON
4. POSITION LIGHT SWITCHES
5. FORMATION LIGHT SWITCH
6. FUSELAGE LIGHT SWITCH
7. ANTI-COLLISION LIGHT SWITCH

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Figure 1-42.

Anti-Collision Lights Switch.

The anti-collision lights switch (7, figure 1-42), is located on the lighting control panel. The switch is labeled ANTI-COLLISION and has one position marked OFF and an unmarked ON position. Placing the switch to ON causes the anti-collision lights to light, extend and rotate. Placing the switch to OFF causes the lights to retract, go out and stop rotating. The switch controls 115 volt ac power from the main ac bus.

Fuselage Lights Switch.

The fuselage lights switch (6, figure 1-42), is located on the lighting control panel. The switch is labeled FUSELAGE and has a position marked OFF and an unmarked ON position. Placing the switch to ON, lights a white light in the top and bottom of the fuselage.

Air Refueling Receptacle Lights Control Knob. (8) (10) (12)→

The air refueling receptacle lights control knob (36, figure 1-21), is located on the right main instrument panel. The knob is labeled A/R RECP LT. The full counterclockwise position of the knob turns the lights off. As the knob is turned clockwise detent positions at spaced intervals vary the intensity of the lights from off to full brightness. The knob controls 28 volt ac power from the main ac bus.

Landing and Taxi Lights Switch.

The landing and taxi lights switch (13, figure 1-4) is located on the miscellaneous control panel. The switch is marked LANDING, OFF and TAXI. The switch controls 115 volt ac power from the ac essential bus which in turn controls relays to provide 28 volt dc power to the filaments in the lights.

INTERNAL LIGHTING.

The internal lights include; red instrument panel and console lights, red and white flood lights and utility lights. The instrument panel and console lights consist of five circuits, each with an individual control knob, for the flight instruments, engine instruments, left and center console, right console and right main instrument panel. They are powered by 115 volt ac power from the right main ac bus. The flood lights consist of left, center and right red flood lights and high intensity white flood lights at various locations around the cockpit. The red flood lights provide cockpit lighting in the event the instrument panel and console lights fail. Each red flood light has an individual control knob. The white flood lights provide high intensity lighting to prevent temporary blindness from lightning when flying in weather. One control knob adjusts the intensity of all the white flood lights. Both the red and white flood lights receive 115 volt ac power from the ac essential bus. Two utility lights, (1A, figure 1-16 and 8, figure 1-45A) one for each side of the cockpit, are provided for individual work lights. They are normally stowed on the left side of the aft console and on the right side wall but can be moved to various locations about the crew station. The front of each utility light can be rotated to change color from white to red and vice versa. A rheostat on the aft end of each light must be turned clockwise to turn the light on and set the desired intensity. The utility lights are powered by 28 volt dc from the engine start bus.

Internal Lighting Control Knobs.

Nine internal lighting control knobs (1, figure 1-42), located on the lighting control panel, control the various internal lighting circuits. The full counterclockwise position of each knob turns the lights off. As the knobs are turned clockwise, detent positions at spaced intervals vary the intensity of the lights from off to full brightness. Five of the knobs control the red instrument panel and console lighting. Knobs are labeled and control the respective circuits as follows:

FLT INST - Left main instrument panel.

ENG INST - Engine instruments.

L&C CONSOLE - Left and center consoles.

R CONSOLE - Right console.

R FWD - Right main instrument panel.

The red flood lights are controlled by individual knobs marked R FLOOD, C FLOOD and L FLOOD for the right, center and left flood lights respectively. A single knob marked WHITE FLOOD FLT & ENG INST controls all the white flood lights. This knob is similar to the other knobs except it is marked OFF at the full counterclockwise position and HIGH INT (high intensity) near the full clockwise position. Turning the knob past HIGH INT turns all the white flood lights to maximum intensity.

CANOPY.

The canopy consists of left and right clam shell hatches hinged to a center beam assembly. The hatches open to a maximum of 65°. Each hatch has an external and internal canopy latch handle for opening or closing. When the hatches are closed and latched, the internal handle locks in place to prevent inadvertent unlatching of the hatch inflight. Each hatch is manually raised or lowered with the aid of an air/oil counterpoise. The counterpoise will also hold the hatch in any position selected. On airplanes (1)→(11), the canopy center beam, aft canopy extension, and both canopy hatches are jettisoned as a single unit. During the ejection cycle, the canopy is automatically jettisoned by actuating either the face screen ejection handles or the secondary ejection handle. If the canopy fails to jettison during the ejection cycle it is impossible to eject through the canopy. In this event an internal canopy jettison handle, located on the center beam, permits jettison of the canopy from the inside. On all airplanes, an external emergency canopy release handle, located outside on the right side of the fuselage, provides a means of explosively releasing the canopy hatches for emergency entrance to the crew module. On airplanes (17) and (18) the left canopy hatch can be explosively detached along the canopy center beam to allow crew rescue from above by helicopter

INTERNAL CANOPY LATCH HANDLES.

Two canopy latch handles are located on the inside lower horizontal frame member of each canopy hatch (figure 1-2). An over-center spring-loaded canopy latch handle lock tab, in the face of each canopy latch handle, locks the handle in the latched position to prevent inadvertent opening inflight. When the lock tab is flush the canopy latch handle is locked. Pressing in on the forward part of the lock tab will cause the rear part of the tab to snap out, unlocking the canopy latch handle. The handle must then be pulled out and aft to a detent position to unlatch the hatch. Once the hatch is unlatched, pulling the handle further aft past detent engages the counterpoise to aid in opening. When the desired hatch position is attained, the handle will return to detent when released and lock the counterpoise to hold the hatch. Each handle is mechanically linked to a flush external canopy latch handle located outside of each hatch. Inflation of the canopy pressurization seal is automatically operated by closure of the canopy hatch. The actuator mounted on the hatch lower surface depresses a plunger in the canopy sill to inflate the seals and turn off the canopy unlock warning lamp.

CANOPY EXTERNAL LATCH HANDLES.

Two flush mounted canopy external latch handles are located on the lower horizontal frame member of each canopy hatch. Each handle is mechanically linked to its respective internal handle. Pressing in on the forward part of the handle will extend the rear portion of the handle so that it may be grasped to unlatch and raise the hatch. If the internal handle is locked in the closed position the hatch cannot be opened from the outside except by actuation of the canopy external emergency release handle.

CANOPY JETTISON HANDLE. (1)→(11)

A yellow T-shaped canopy jettison handle located forward on the center windshield beam (figure 1-2), provides a means of jettisoning both canopy hatches simultaneously. A button is located on each side of the "T" and either one may be depressed to release the handle. Pulling the handle will cause the center beam to explosively unlatch. The canopy jettison actuator will push the leading edge of the unlatched canopy up. After the leading edge is open 85°, the canopy will leave the aircraft. The two hatches lock together prior to leaving the aircraft to prevent the hatches from folding in and hitting the crew. The hatches cannot be jettisoned individually. A ball lock safety pin assembly is installed in the canopy jettison handle during ground operation to prevent inadvertent jettisoning of the canopy.

WARNING

Do not jettison the canopy unless both hatches are closed and latched. To do so could result in fatal injury from flying debris from the canopy sill.

LEFT CANOPY DETACH HANDLE. (17) (18)

The left canopy hatch can be detached along the center canopy beam by pulling the left canopy detach handle (figure 1-47A), located on the aft bulkhead above the aircraft commander's seat. Pulling the handle fires an initiator which in turn fires an explosive charge to separate the left canopy hatch along the center beam. The handle is marked CANOPY DETACH. A safety pin is inserted in the handle to prevent inadvertent actuation.

CANOPY EXTERNAL EMERGENCY RELEASE HANDLE.

A flush mounted round spring-loaded external emergency canopy release handle on the right side of the fuselage provides a means of gaining access to the crew module in event of an emergency on the ground when the latches are locked from the inside. Pressing in on the center of the handle will release a latch and allow the handle to spring out. This will expose enough of the handle so that it can be grasped. Pulling the handle after approximately six feet of cable uncoils, will fire the external canopy release initiator which in turn fires an explosive strip along the

Air Conditioning and Pressurization System

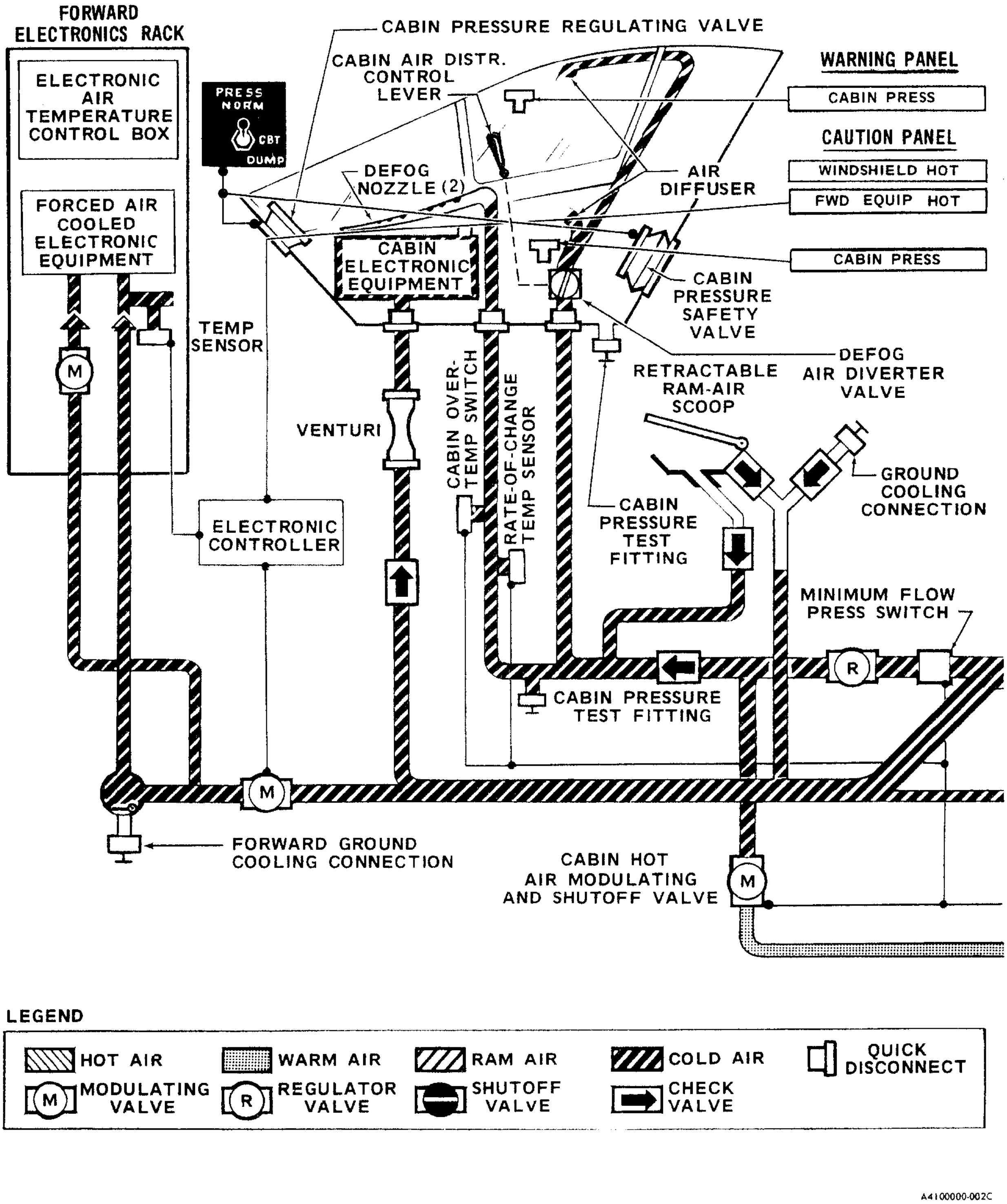


Figure 1-43. (Sheet 1 of 2)

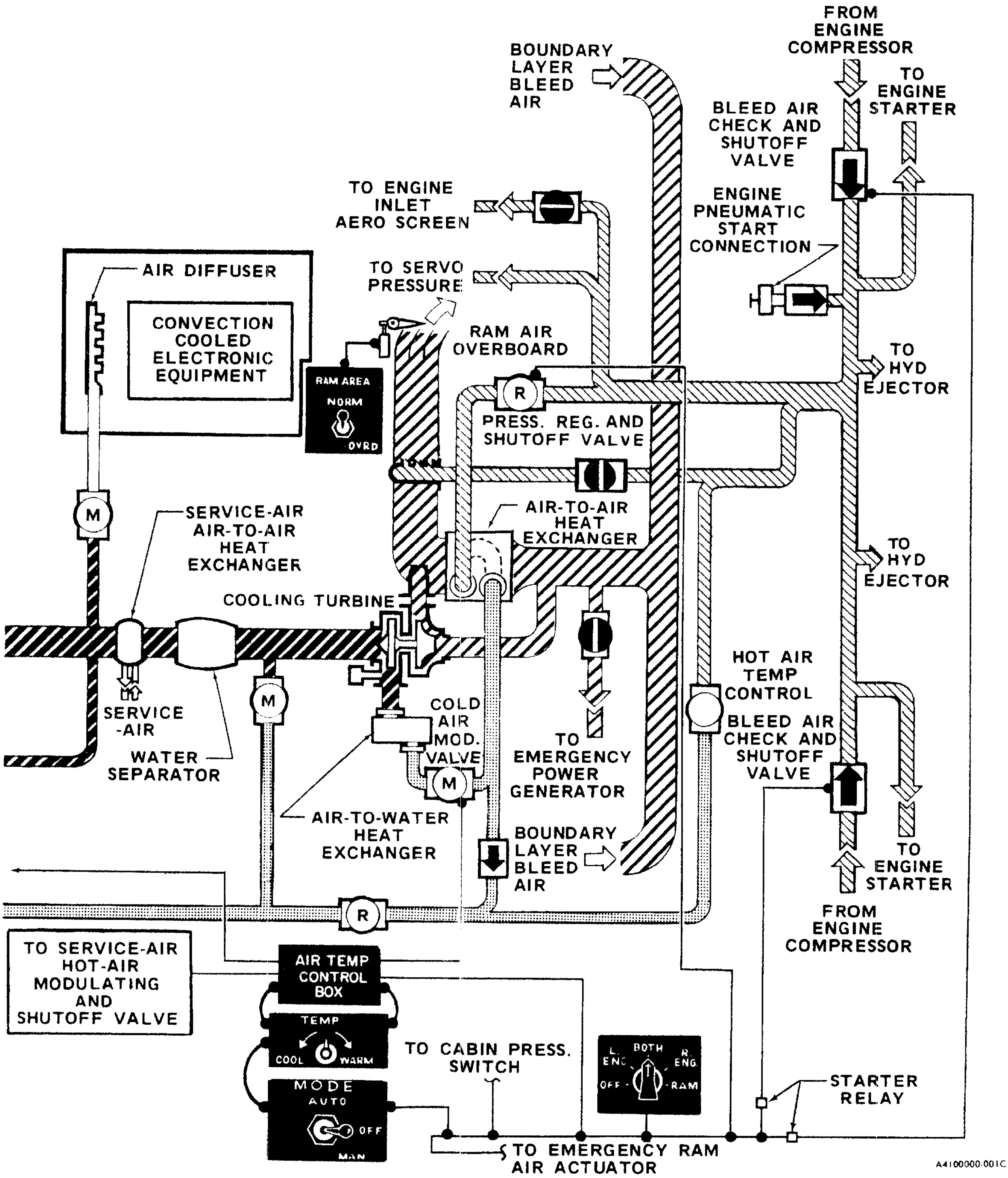


Figure 1-43. (Sheet 2 of 2)

canopy sills. Detonation of the explosive strip will remove the canopy latch hooks to release the hatches. The canopy hatches must then be manually raised to gain access to the cockpit.

WARNING

Do not pull the canopy external emergency release handle unless both hatches are closed and latched. To do so could result in fatal injury to the occupants from flying debris from the canopy sill.

A ball lock safety pin assembly (7, figure 1-45A) is installed in the canopy emergency release initiator, located on the right side wall, to prevent inadvertent actuation.

CANOPY INTERNAL EMERGENCY RELEASE HANDLE. (12)→

The T-shaped canopy internal emergency release handle (figure 1-47A), located on the canopy center beam assembly, is provided to release the canopy hatches in the event that the normal canopy latch handles fail. Depressing a release button on either side of the handle and pulling the handle out, will fire an initiator which in turn fires an explosive strip along the canopy sills. Detonation of the explosive strip will remove the canopy latch hooks to release the hatches. The canopy hatches may then be raised manually.

WARNING

Do not pull the canopy internal emergency release handle unless both hatches are closed and latched. To do so could result in fatal injury to the occupants from debris from the canopy sill.

A ball lock safety pin assembly is installed in the canopy internal emergency release handle during ground operation to prevent inadvertent actuation.

CANOPY UNLOCK WARNING LAMP.

A red canopy unlock warning lamp located on the left warning and caution lamp panel (1, figure 1-5), will light when either hatch is not locked. When lighted the word CANOPY is visible on the face of the lamp.

AIR CONDITIONING AND PRESSURIZATION SYSTEM.

The air conditioning and pressurization systems (figure 1-43), combine to provide temperature-controlled, pressure-regulated air for heating, ventilating, pressurizing the cockpit and inflating the canopy seals. The system also provides air to the forward and aft electronic equipment bays, anti-icing and defog systems, windshield rain removal system and anti-G and pressure suits.

AIR CONDITIONING SYSTEM.

The air conditioning system provides temperature controlled air for the cockpit. The system also provides a temperature controlled flow of cooling air to the electronic equipment that requires a controlled environment for efficient operation. See figure 1-43. High pressure hot air is bled from the sixteenth stage compressor of each engine. This bleed air is directed through a tee fitting to a common duct and is routed through an air-to-air heat exchanger, where it is cooled by ram air that is circulated through the heat exchanger. The air is then routed through an air-to-water heat exchanger where it is further cooled and then enters the cooling turbine. The cooling turbine further cools the air to a temperature suitable for cooling the cockpit and electronic equipment bays. The cold air leaving the turbine passes through a water separator to remove most of the free moisture. A cabin temperature controller is fed signals from temperature sensors and from a pilot operated control panel. The temperature controller controls the setting of the cold air modulating valves. It also controls the setting of the cockpit hot air modulating and shutoff valve which allows hot air to mix with the refrigerated air stream, obtaining air at the selected temperature. This air then enters the cockpit through diffusers. An air connection is located on the lower right side of the fuselage aft of the cockpit and can be connected to a ground cooling cart to provide cooling air to the cockpit and all equipment. In the event the air conditioning system malfunctions, emergency ram air operation is available for ventilation and cooling.

Cabin Air Distribution Control Lever.

A cabin air distribution control lever (3, figure 1-18), located on the right side wall, controls distribution of air flow in the cockpit. The lever is labeled CABIN AIR DISTR and has two positions marked FWD DEFOG and AFT. The normal position of the lever is in the AFT position. In this position air flow into the cockpit is separated between the air diffusers on the rear bulkhead of the cockpit and the windshield defog system with approximately 85 percent directed to the diffusers. Moving the lever toward the FWD DEFOG position will decrease airflow through the air diffusers and increase airflow through the defog system. When the lever is in the full forward position all the airflow will be directed through the defog system. Although the AFT position is considered normal to obtain maximum airflow, desired crew comfort is accomplished by selecting any intermediate position between FWD DEFOG and AFT.

Air Source Selector Knob.

The air source selector knob (2, figure 1-44), located on the air conditioning control panel, has five positions marked OFF, L ENG, BOTH, R ENG, and RAM.

The knob controls bleed air source or allows selection of emergency ram air operation when the normal system is not operating. In the OFF position, both the left and right bleed air check and shutoff valves are closed. In the L ENG position, the left engine is the source of bleed air and the right bleed air check and shutoff valve is closed. In the BOTH position, both the left and the right bleed air check and shutoff valves are open and supplying bleed air to the air conditioning system. In the R ENG position, the right engine is the source of the bleed air and the left bleed air check and shutoff valve is closed. In the RAM position, the normal pressurization system pressure regulating and shut off valve is closed, the ram air door is open, and the cockpit pressure regulating and relief valves are open. This will dump cabin pressure and allow combined ram air flow and regulated engine bleed air to ventilate the cabin. Temperature control of this air is available by using the temperature control knob to control the amount of engine bleed air mixed with ram air.

WARNING

To prevent excessive temperatures when pressure suits are being worn, the air conditioning system mode selector switch must not be placed to the OFF position prior to or while operating in the RAM position.

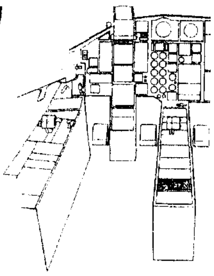
Air Conditioning System Mode Selector Switch.

The mode selector switch (5, figure 1-44), located on the air conditioning control panel, is a three position switch marked AUTO, OFF, and MAN. In the AUTO position, the cockpit temperature is automatically controlled at the temperature selected by the temperature control knob. A signal goes to the controller which opens or closes the modulating valves to maintain the selected temperature. In the MAN position, the cockpit temperature controller is bypassed and control of the modulating temperature control valves is directly from the termperature control knob. In the OFF position all power is removed from the system and the valves in the system, which control cabin temperature, will declutch and go to the full cool position. The valve controlling pressure suit ventilation temperature will remain in the position it was in when power was removed.

Temperature Control Knob.

The temperature control knob (1, figure 1-44), located on the air conditioning control panel, is provided to select cockpit temperature. The knob can be rotated through a 300° arc and has mechanical stops at each end. The extreme counterclockwise end is marked COOL and the clockwise end is marked WARM. With the mode selector switch in AUTO, rotating the knob in either direction sends a signal to

Air Conditioning Control Panel



1. TEMPERATURE CONTROL KNOB
2. AIR SOURCE SELECTOR KNOB
3. RAM AREA CONTROL SWITCH
4. PRESSURIZATION SELECTOR SWITCH
5. MODE SELECTOR SWITCH

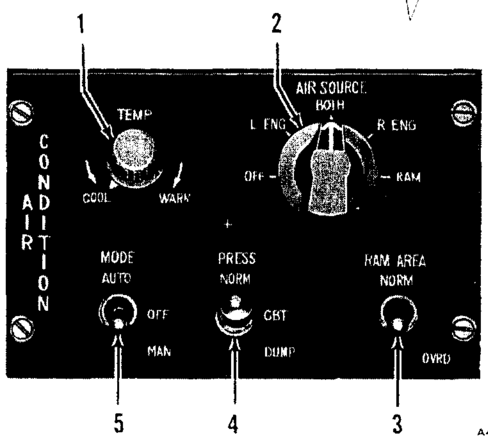


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Figure 1-44.

the cockpit temperature controller which constantly positions the modulating temperature control valves to maintain the selected temperature. When the temperature control knob is positioned at the midpoint between COOL and WARM, the cockpit temperature is maintained at approximately 19°C (67°F).

Note

Operation with the temperature control knob at full COOL in warm weather or full WARM in cool weather with the mode selector knob in AUTO may result in an objectionable noise with the high flow in the cockpit. The amount of airflow can be reduced by backing the knob off the full COOL or WARM position.

With the mode selector switch in MANUAL, the signal goes directly to the modulating temperature control valves, opening or closing them as directed by the signal generated from the temperature control knob. During manual operation the valves will respond only when the knob is held against one of the extreme positions, COOL or WARM. Maximum valve travel time from maximum cold to maximum warm is approximately 45 seconds.

Cabin Pressure Schedule

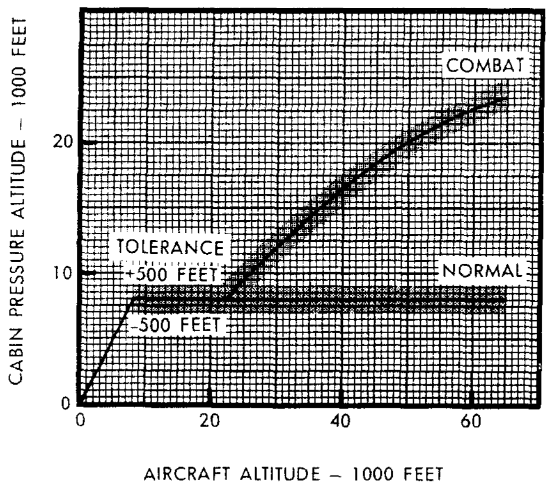


Figure 1-45.

Ram Area Control Switch.

The ram area control switch (3, figure 1-44), located on the air conditioning control panel, is a two position switch marked NORM and OVRD. The switch provides a means of controlling the amount of ram airflow through the air-to-air heat exchanger by opening or closing an exit door in the ram air discharge exit. In the NORM position, the central air data computer automatically controls the position of the door. When the outside air temperature is below 75°F and airspeed is above 225 knots the door will be closed to reduce drag. All other combinations of outside temperature and airspeed will result in automatic door opening. Placing the switch to OVRD will override the automatic functions of the central air data computer and open the door to its full travel. On some airplanes the ram air discharge exit door has been removed, therefore the switch in inoperative.

Equipment Hot Caution Lamp.

The amber equipment hot caution lamp, marked FWD EQUIP HOT, is located on the main caution light panel (14, figure 1-5). The lamp will light if the cooling air flow is insufficient.

Air Conditioning System Operation.

Automatic Mode.

1. Air source selector knob - BOTH.
2. Mode selector switch - AUTO.
3. Temperature control knob - Set for desired temperature.
4. Ram area control switch - NORM.

Manual Mode. In the event of a malfunction of the cabin temperature controller, cabin temperature may be manually controlled as follows:

1. Air source selector knob - BOTH.
2. Mode selector switch - MAN.
3. Temperature control knob - Set for desired temperature.
   The temperature control knob must be held to either full COOL or full WARM position to adjust for desired temperature.

Ram Air Mode. In the event the air conditioning system fails, ram air mode can be used for cockpit and equipment cooling. Refer to "Ram Air Mode Limit Speed," Section V. During ram air mode operation, cockpit temperature can be controlled as follows:

1. Air source selector knob - RAM.

WARNING

To prevent excessive temperatures when pressure suits are being worn, the air conditioning system mode selector switch must not be placed to the OFF position prior to or while operating in the RAM position.

2. Mode selector switch - AUTO or MAN.
3. Temperature control knob - Set for desired temperature.
   If the mode selector switch is positioned to MAN, the temperature control knob must be held to either full COOL or WARM position to adjust for desired temperature.

PRESSURIZATION SYSTEM.

Pressurization of the cockpit, canopy seals, anti-g suits, pressure suits, attack radar, terrain following radar and track breaker is provided by the pressurization system. Pressure in the cockpit is controlled by a pressure regulating valve located in the front of the cockpit. When the airplane is below 8000 feet, the pressure regulating valve automatically maintains an unpressurized condition in the cockpit regardless of the schedule selected. Cockpit ventilation is provided by the regulating valve continually modulating, depending on the volume of input air. A cabin pressure safety valve located at the rear of the cockpit will relieve pressure any time the cockpit pressure exceeds outside pressure by 11.2 psi. An emergency ram air scoop, which can be opened into the airstream, will admit air into the crew and electronic equipment compartments in the event of loss of cooling and pressurization air from the cooling turbine. When combat cabin pressure schedule is selected, the system maintains a maximum pressure differential of 5 psi above ambient pressure at altitudes above 23,000 feet. See figure 1-45 for cockpit pressure schedule for normal and combat conditions.

Pressurization Selector Switch.

The pressurization selector switch (4, figure 1-44), located on the air conditioning control panel, is a three position lever lock switch with positions NORM, CBT, and DUMP. In the NORM position, the cockpit pressure is selected to a schedule that will maintain an 8000 foot cabin altitude from 8000 feet up to the operational ceiling of the airplane. In the CBT (combat) position, the cockpit maintains an 8000 foot cabin altitude from 8000 feet up to 22,500 foot altitude and then maintains a constant 5 psi differential above 8000 feet up to the operational ceiling of the airplane. In DUMP position, the cabin pressure regulator and the cabin pressure safety valve are open and the cockpit is not pressurized.

Cabin Altitude Indicator.

A cabin altitude indicator (6, figure 1-4), located on the left console, is provided to monitor cabin altitude.

Pressurization Caution Lamp.

An amber pressurization caution lamp marked CABIN PRESS is located on the main caution light panel (14, figure 1-5). The lamp will light when the cabin altitude is above 10,000 feet.

Pressurization Warning Lamp.

A red pressurization warning lamp (1, figure 1-5) marked CABIN PRESS is located on the left main instrument panel. The lamp will light when the cabin pressure is above 38,000 feet.

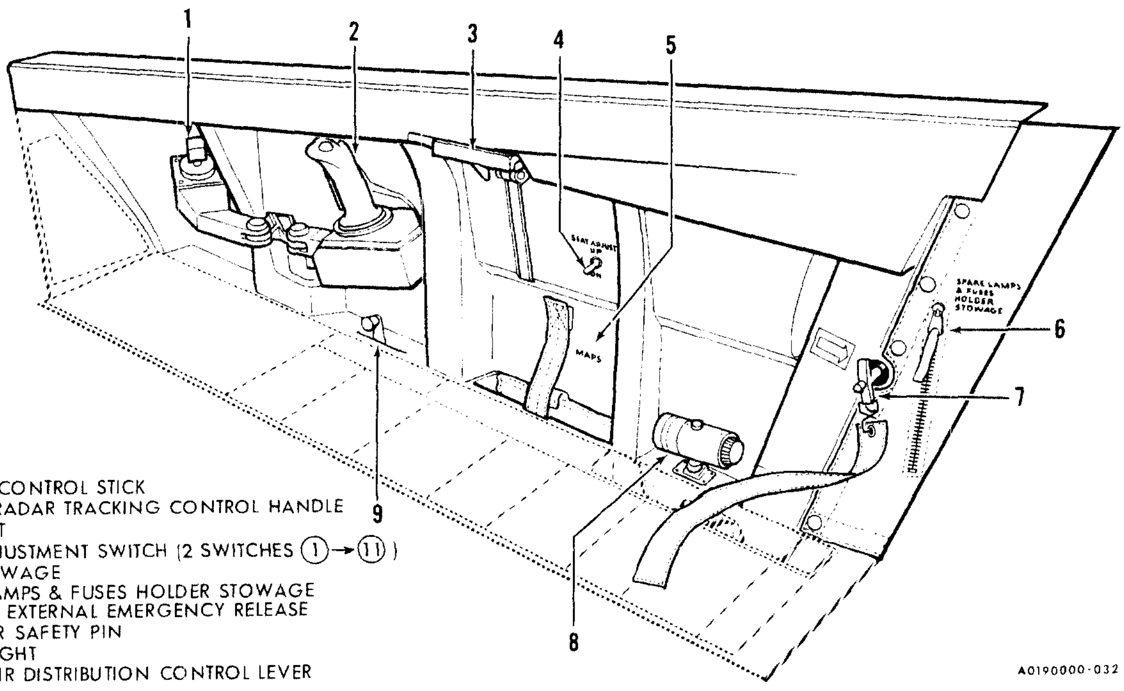
Equipment Low Pressure Caution Lamp.

An amber low equipment pressure caution lamp marked LOW EQUIP PRESS is located on the main caution light panel (14, figure 1-5). The lamp will light when the supply pressure to the pressurized electronic equipment requiring one atmosphere pressure drops below 12.5 (±.5) psia.

Emergency Pressurization System. (12)→

The crew module escape system incorporates an emergency pressurization system. The system operates automatically during ejection to maintain pressurization of the module and canopy hatch seals, or

Right Sidewall (Typical)



1. GAM-83 CONTROL STICK
2. ATTACK RADAR TRACKING CONTROL HANDLE
3. ARM REST
4. SEAT ADJUSTMENT SWITCH (2 SWITCHES (1)→(11))
5. MAP STOWAGE
6. SPARE LAMPS & FUSES HOLDER STOWAGE
7. CANOPY EXTERNAL EMERGENCY RELEASE INITIATOR SAFETY PIN
8. UTILITY LIGHT
9. CABIN AIR DISTRIBUTION CONTROL LEVER

Figure 1-45A.

in event of failure of the automatic feature, the system is manually activated with a ring-shaped handle. Also, during other phases of flight, this system provides an alternate pneumatic supply source for pressurization of the crew module and canopy hatch seals in event of failure of the normal pressurization system. Pressure for the system is contained in a 650 cubic inch storage bottle located behind the seat bulkhead. When activated, an aneroid-operated absolute pressure regulator which senses cabin altitude, will open if cabin altitude is above 24,000 feet. Volume of the storage bottle is sufficient to maintain this cabin altitude for approximately 4 minutes at maximum ejection altitude even with high leakage.

Emergency Pressurization Handle. (12)→ The emergency pressurization handle (20, figure 1-16), located on the upper right corner of the aft console, is provided to manually activate the emergency pressurization system. Pulling the handle out will open the aneroid-operated absolute pressure regulator.

Emergency Pressurization System Gage. (12)→ An emergency pressurization system gage, located on the cockpit aft bulkhead, is provided to indicate the pressure within the emergency pressurization system storage bottle. The gage is calibrated from 0 to 4000 psi in 500 psi increments.

ANTI-ICING AND DEFOG SYSTEMS.

Pitot Static Probe Anti-Icing.

The pitot static probe is equipped with a heating element powered by the 115 volt ac essential bus. When energized, the element will heat the probe, thus preventing the formation of ice.

Pitot Heater Switch. The pitot heater switch (4, figure 1-46), labeled PITOT HEATERS, is a two position switch marked OFF with an unmarked ON position and is located on the windshield wash/anti-icing control panel. When the switch is placed to ON, 115 volt ac power is furnished to heating elements in the pitot-static tube, the angle of attack and, when the landing gear is retracted, to the total temperature probe. To ground check the heater in the total temperature probe, the CADC test switch must be held to the HIGH position in addition to placing the pitot heater switch to ON. When the switch is OFF, the heaters are deenergized. The switch controls 28 volt dc power from the main and essential buses.

Engine Anti-Icing Systems.

The engine anti-icing system prevents formation of ice on the engine inlet guide vanes, and the engine nose cone. The engine anti-icing system uses regulated compressor bleed air. The engine inlet anti-icing system prevents formation of ice on the spike tip, the leading edge of the auxiliary cowl and inside the main engine air duct of the auxiliary cowl. The engine inlet anti-icing system uses air from the air conditioning system hot air manifold in the main landing gear wheel. Idle rpm will provide sufficient hot air for anti-icing. The spike sensing probe anti-icing system prevents formation of ice on the spike local mach probe and spike lip shock probe. The probes are heated by 115 volt ac electrical heaters. Although the engine anti-icing, engine inlet anti-icing, and spike sensing probe anti-icing are three separate systems, they are controlled by a single, three position switch. Both automatic and manual modes of operation are provided. An electronic ice detector is located in the left engine air inlet. When icing conditions exist, a signal is transmitted to the icing caution lamp regardless of the position of the engine/inlet anti-icing switch.

Engine/Inlet Anti-Icing Switch. The engine/inlet anti-icing switch (3, figure 1-46), located on the windshield wash/anti-icing control panel, is a three position switch marked AUTO, MAN and OFF. The lever lock-type switch locks in all three positions. In the AUTO position, the anti-icing circuitry is armed, and when the electronic ice detector senses an icing condition a signal is transmitted to the icing caution lamp. The signal also energizes a relay which turns on the elements in the spike sensing probe heaters and opens the engine anti-icing and engine inlet anti-icing control valves allowing the circulation of hot air through the anti-iced components. Approximately 60 seconds after the icing condition ceases, the hot air valves will close, the spike probe heating elements will be deenergized and the engine icing caution lamp will go out. When the switch is placed to MAN, the engine anti-icing and engine inlet anti-icing valves open and the spike probe heating elements are energized whether or not the ice detector senses an icing condition. Placing the switch to OFF shuts off air to the engine anti-icing and engine inlet anti-icing systems, and turns off the spike probe heating elements; however, the icing caution lamp will still be operational.

Engine Icing Caution Lamp. The engine icing caution lamp, located on the main caution lamp panel (14, figure 1-5), will light when the electronic ice detector senses an icing condition. While the icing condition exists, the caution lamp will remain lighted regardless of the position of the engine/inlet anti-icing switch. The lamp will go out 60 seconds after the icing condition ceases.

Inlet Hot Caution Lamp. The inlet hot caution lamp, located on the main caution lamp panel (14, figure 1-5), provides an indication that the temperature of anti-icing bleed air to the translating cowls has exceeded 450 (±10) degrees fahrenheit. When the lamp lights the words INLET HOT are visible and anti-icing air to the translating cowls is automatically shutoff, then the lamp will go out.

Windshield Defog System.

Air for windshield defogging and cabin air distribution share the same control lever. For description, refer to "Cabin Air Distribution Control Lever", this section.

WINDSHIELD WASH AND RAIN REMOVAL SYSTEM.

The windshield wash and rain removal system is provided to keep both the windshields clear of impinging rain and insects. Compressor bleed air at a temperature of 390°F and a pressure of 45 psi is directed over the outside of the windshields by a fixed area nozzle. This hot air blast will evaporate impinging rain and prevent further accumulation of rain on the windshield. Windshield wash is accomplished by injecting a liquid wash solution into the rain removal nozzle. This serves as a wetting and scrubbing action to remove insects from the windshields. The windshield wash solution is contained in a one gallon tank located on the right side of the nose wheel well. The tank is pressurized to 15 psi by compressor bleed air.

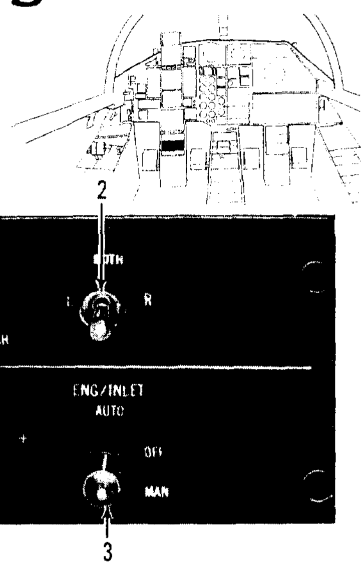
Windshield Wash/Rain Removal Selector Switch.

The windshield wash/rain removal selector switch (1, figure 1-46), located on windshield wash/anti-icing control panel, has three positions marked RAIN REMOVE, WASH, and OFF. The switch is spring loaded from the WASH to the OFF position, and is locked out of the RAIN REMOVE position. The switch must be pulled out to move from OFF to RAIN REMOVE. On those airplanes modified by T.O. 1F-111A-575 the switch is locked out of the RAIN REMOVE position. On these airplanes the switch must be pulled out to move from OFF to RAIN REMOVE. Placing the switch to RAIN REMOVE will open the rain remove shutoff valves, allowing temperature and pressure regulated compressor bleed air to be directed to the windshield(s) selected by the windshield selector switch. When the switch is placed to WASH a time delay relay is energized to open the rain remove shutoff valve and the windshield wash shutoff valve selected by the windshield wash selector switch. While these valves are open, compressor bleed air and liquid windshield wash solution will be directed to the selected windshield(s). Positioning the switch from WASH to OFF will close the valves after a 5-second delay, shutting off the air and windshield wash solution. When the switch is in the OFF position the windshield wash and rain removal system is deenergized.

Windshield Selector Switch.

The windshield selector switch (2, figure 1-46), located on the windshield wash/anti-icing control panel, has three positions marked L (left), R (right), and BOTH. Selection of any of the positions will determine the windshield(s) to be washed or receive rain remove as a function of the position of the windshield wash/rain removal selector switch.

Windshield Wash / Anti-Icing Control Panel



1. WINDSHIELD WASH/RAIN REMOVAL SELECTOR SWITCH
2. WINDSHIELD WASH SELECTOR SWITCH
3. ENGINE/INLET ANTI-ICING SWITCH
4. PITOT HEATERS SWITCH

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Figure 1-46.

Windshield Hot Caution Lamp.

The windshield hot caution lamp, located on the main caution lamp panel (14, figure 1-5) is provided to indicate when the windshield temperature is above limits. An overheat switch, installed in the rain removal air supply duct upstream of the shutoff valve, will close whenever the air temperature is above 450°F. When the overheat switch closes, a circuit is completed to close the rain remove shutoff valves and light the windshield hot caution lamp. After the switch closes the caution lamp will normally go out within 15 seconds.

OXYGEN SYSTEM.

The oxygen system consists of a normal (liquid) system located in the forward fuselage and cockpit and an emergency (gaseous) system located in the seat pan of the ejection seat, airplanes 1→ 11, or behind the cockpit aft bulkhead on 12→.

NORMAL OXYGEN SYSTEM.

The normal oxygen system is designed for use with a pressure demand type oxygen regulator and mask to provide pressure regulated 100 percent oxygen to the crew members. See figure 1-47 for oxygen duration. A converter, located on the left side of the aircraft below the crew module changes the liquid oxygen to gaseous oxygen that is pressure regulated at 70 to 80 psi during normal usage. From the converter, the gaseous oxygen passes through a heat exchanger that warms it for breathing. The oxygen is then directed through a manually operated control valve, a regulator, and into the face mask. Oxygen regulation is accomplished by a demand type mini-regulator mounted on the right side of the torso harness. Maximum duration of the normal oxygen supply is 31.2 hours when both crew members are on oxygen at a cabin altitude greater than 35,000 feet. Maximum normal oxygen duration for two crew members at a cabin altitude of 8000 feet is 7.8 hours. Evaporation loss will com-

Oxygen Duration

*2 crew members (double duration for 1 crew member)

CABIN ALTITUDE	CONSUMPTION cu. ft./hr.	DURATION - HOURS*										
35,000	9.8	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1	DESCEND TO BELOW 10,000' MSL
30,000	13.4	22.8	20.5	18.3	16.0	13.7	11.4	9.1	6.8	4.6	2.3	
28,000	15.0	20.4	18.4	16.3	14.3	12.2	10.2	8.2	6.1	4.1	2.0	
26,000	16.0	19.1	17.2	15.3	13.4	11.5	9.6	7.6	5.7	3.8	1.9	
24,000	18.52	16.5	14.9	13.2	11.6	9.9	8.3	6.6	5.0	3.3	1.6	
22,000	20.76	14.7	13.3	11.8	10.3	8.8	7.4	5.9	4.4	2.9	1.5	
20,000	23.0	13.3	12.0	10.6	9.3	8.0	6.6	5.3	4.0	2.7	1.3	
18,000	25.24	12.1	10.9	9.7	8.5	7.3	6.1	4.8	3.6	2.4	1.2	
16,000	27.48	11.1	10.0	8.9	7.8	6.7	5.6	4.4	3.3	2.2	1.1	
14,000	30.0	10.2	9.2	8.2	7.1	6.1	5.1	4.1	3.1	2.0	1.0	
12,000	32.8	9.3	8.4	7.5	6.5	5.6	4.7	3.7	2.8	1.9	0.9	
10,000	35.6	8.6	7.7	6.9	6.0	5.1	4.3	3.4	2.6	1.7	0.8	
8,000	39.4	7.8	7.0	6.2	5.4	4.7	3.9	3.1	2.3	1.5	0.7	
0	55.6	5.5	4.9	4.4	3.8	3.3	2.7	2.2	1.6	1.1	0.5	
AVAILABLE OXYGEN	LITERS	10	9	8	7	6	5	4	3	2	1	LESS THAN 1
	Cu Ft GAS	306.0	275.4	244.8	214.2	183.6	153.0	122.4	91.8	61.2	30.6	

Figure 1-47.

pletely empty the fully serviced system in approximately 10 days. The system has a ten liter capacity and is serviced through a single point filler valve located within an access door on the left side of the fuselage (figure 1-50). Procedures for normal operation of the normal oxygen system are contained within the appropriate portion of Section II.

Oxygen Control Levers.

Two oxygen control levers are provided to control flow of oxygen from the supply system to the oxygen regulator. Each lever has positions ON and OFF. When in the ON position, oxygen is supplied from the converter to the regulator; when in OFF, oxygen flow is shut off at the control valve in the suit-mask panel. The levers are located on the left and center consoles (28, figure 1-4 and 22, figure 1-17).

Oxygen Quantity Indicator.

An oxygen quantity indicator is located on the left console (7, figure 1-4). The indicator indicates the total quantity of liquid oxygen in the converter. The indicator dial is graduated from zero to 20 liters in increments of one liter. The indicator operates on 115 volt ac power from the essential bus. In the event of power failure, the indicator pointer will drive below zero, a fail safe indication.

Oxygen Quantity Indicator Test Button.

A test button used for checking the oxygen quantity indicator is located on the left console (8, figure 1-4). When the button is held depressed, the indicator pointer will move to the zero liter indication if the indicating system is operating properly. When the button is released, the pointer will move back to the original reading. The oxygen caution light will illuminate during an indicator check when the pointer indicates a quantity of 2 liters or less.

Oxygen Caution Lamp.

An amber caution lamp on the main caution light panel (14, figure 1-5) will light when oxygen quantity indicator indicates 2 liters or less or when oxygen system pressure is less than 42 (±2) psi. When the caution lamp lights, inspection of the oxygen quantity gauge will determine whether the lamp came on because of low quantity or low pressure. When the lamp is lighted, the letters OXY will be visible on the caution light panel, and the master caution lamp will light. The oxygen caution lamp operates on 28 volt dc power from the 28 volt dc essential bus.

EMERGENCY OXYGEN SYSTEM. (1)→(11)

The emergency oxygen system is located in the seat pan of the ejection seat (figure 1-49). The system consists of two high pressure containers, a single point refill valve, a pressure gage, a pressure reducer valve and an emergency oxygen deployment handle (green ring). The oxygen containers consist of a "U" shaped tube which forms the contour of the seat pan and a bottle located under the forward edge of the seat. The two are interconnected to provide a total of 63 cubic inches of oxygen which will last approximately 10 minutes under emergency conditions. The system is refilled through the single point refill valve located on the left side of the seat pan. System pressure is indicated on a gage on the right side of the seat pan. The pressure reducer valve is connected to a lanyard anchored to the floor of the cockpit to turn the system on and regulate oxygen flow when the seat is ejected. Also, during other phases of flight, this system provides an emergency oxygen supply in event of failure of the normal oxygen system.

Emergency Oxygen Gage. (1)→(11)

The emergency oxygen gage (figure 1-49), is located on the forward right side of the seat pan. The gage is marked REFILL in the red region and FULL in the black region with index marks at 1800 and 2500 psi. Minimum pressure is 1800 psi at 70°F.

Emergency Oxygen Handle (Green Ring). (1)→(11)

The emergency oxygen handle (green ring) is located in the seat pan cutout (figure 1-49). Pulling the handle will actuate the pressure reducer valve to turn the system on and regulate oxygen flow to the crew member's mask.

EMERGENCY OXYGEN SYSTEM. (12)→

The crew module is equipped with an emergency oxygen system consisting of two oxygen bottles, a pressure reducer, a pressure gage, and a manual handle. The system is activated automatically during ejection or in event of failure of the automatic feature, it is manually activated by a handle. Also, during other phases of flight, this system provides an emergency oxygen supply in event of failure of the normal oxygen system. When activated either manually or automatically, gaseous oxygen at 1800 to 2100 psi flows to a pressure reducer where it is reduced to 50 to 90 psi. It is then routed into the normal oxygen system upstream of the oxygen control valves. Sufficient emergency oxygen is available for 10 minutes duration at 27,000 feet.

Emergency Oxygen Handle. (12)→

The green emergency oxygen handle (11B, figure 1-16), is located on the aft console. During ejection, this handle is used to manually activate the emergency oxygen system in the event automatic activation fails. Also, in event of failure of the normal oxygen system during other phases of flight this handle is used to provide an emergency oxygen supply. Raising the handle will open the emergency oxygen pressure reducer allowing oxygen to flow to each oxygen control valve.

Emergency Oxygen Pressure Gage. (12)→

The emergency oxygen pressure gage (11C, figure 1-16), located on the aft console, indicates the pressure in the emergency oxygen bottles. The gage is marked REFILL in the red region and FULL in the black region with index marks at 1800 and 2500 psi.

OXYGEN SYSTEM NORMAL OPERATION.

Operation of the oxygen system for either crew member is as follows:

1. Oxygen hose connections - Checked.

2. Oxygen control valve toggle - ON.

OXYGEN SYSTEM ALTERNATE OPERATION.

If the normal oxygen system fails or is depleted, proceed as follows: (See figure 1-47.)

1. Emergency oxygen bottle green ring or T-handle - Pulled.

Note

The emergency oxygen system will provide oxygen for approximately 10 minutes.

CREW MODULE ESCAPE SYSTEM. (12)→

The crew module (figure 1-47A), forms an integral portion of the forward fuselage and encompasses the pressurized cabin and forward portion of the wing glove.Crew entrance to the module is provided through left and right canopy hatches. Refer to Canopies this section. The system provides maximum protection for both crew members throughout the airplane performance envelope including zero altitude and zero speed ejection capability. The system protects the occupants from environmental hazards on either land or water and provides underwater escape capabilities. Also provided, are an emergency oxygen supply system and a self-contained emergency pressurization system. Both of these systems are provided, primarily, for use during ejection. However, either system can be manually activated during normal phases of flight, as a backup to the associated normal system.

WARNING

The removal or addition of components in the crew module or the absence of a crew member will change the center of gravity of the module and adversely affect its stability on ejection.

For additional information, refer to "Oxygen System", and "Air Conditioning and Pressurization System", this section.

CREW MODULE SEATS. (12)→

The crew module seats (figure 1-47B), are electrically adjustable vertically and manually adjustable forward and aft. The seat headrest structure, which is attached to the aft bulkhead, and the seat pan are manually adjustable forward and aft. The forward adjustment of the headrest requires the inertial reel to be unlocked. The seat back is attached by pivot pins to the back of the seat pan and is attached through telescopic structures to pivot pins on the headrest. Each seat is equipped with an upper and a lower torso harness. The upper torso harness consists of adjustable shoulder straps, chest straps, and trunk straps. These straps attach at the center of the crewmember's chest.by means of a quick-release buckle. The chest straps and trunk straps are attached to the seat structure and the shoulder straps are attached to the inertia reel. Wedge shaped plastic blocks are attached on the right chest strap and right trunk strap on each seat harness, to attach the oxygen regulator in either position as convenient for the crew member. The lower torso harness consists of an adjustable lap belt which is attached to each side of the aft seat pan structure. The lower torso harness attaches on both sides by means of quick release buckles. Each seat is equipped with an inertia reel located behind the head rest. When unlocked, the inertia reel will allow the shoulder straps to extend or retract automatically to allow freedom of movement of the crew member. When an excessive g-force is encountered or when manually locked, the inertia reel will prevent further shoulder strap extension and will take up shoulder strap slack as the crew member returns to a normal position. The inertia reel is also equipped with an explosive cartridge in a power retraction device which, during ejection, will retract the shoulder straps and lock the reel. The right seat must be removed to gain access to the survival equipment compartment.

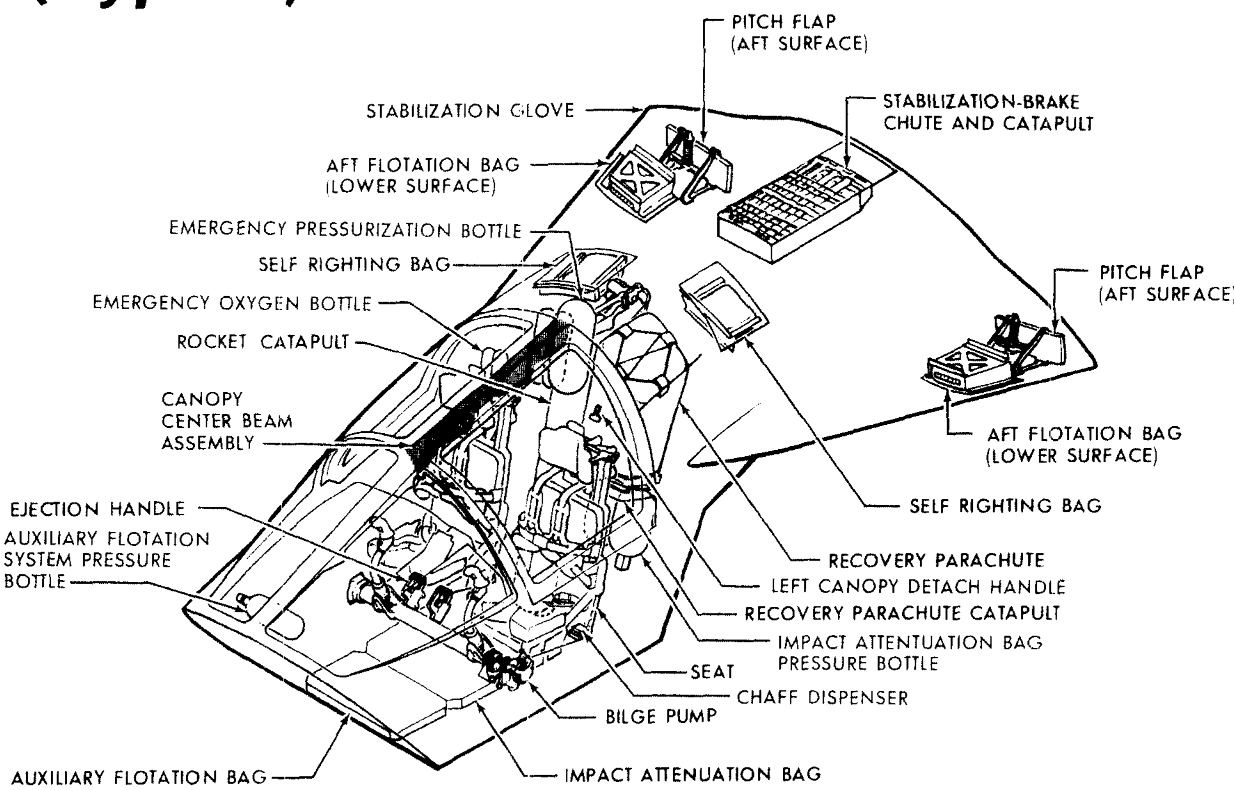
EJECTION EQUIPMENT. (12)→

The ejection equipment consists of the necessary initiators, severance components and the rocket motor. Actuation of either ejection handle initiator provides an explosive impulse sequenced to lock the shoulder harness inertia reels in the retracted position, activate the emergency oxygen and cockpit pressurization system, release the chaff dispenser, activate guillotine cutters, and to ignite the rocket motor. Two pressure initiators which are activated by rocket motor pressure build-up after ignition are provided to activate the severance components and to deploy the stabilization-brake and recovery parachutes and impact attenuation bag. The severance components consist of the flexible linear shaped charges (FLSC) and explosive guillotine cutters. The FLSC is located around the crew module so that detonation will cut the splice plate joining the crew module to the airplane. FLSC is also used to remove the covers over the parachutes and the flotation, self righting, and impact attenuation bags. The explosive guillotine cutters are provided to sever antenna leads, secondary control cables, and an oxygen line. Quick disconnects located in the crew module floor are used for separation of the normal air conditioning and pressurization system ducts, the flight controls, and the electrical wiring. The rocket motor, located between the crew members and behind the seat bulkhead, provides the thrust to propel the crew module up and away from the aircraft.

RECOVERY AND LANDING EQUIPMENT. 12→

The recovery and landing equipment consists of stabilization components, the recovery parachute, landing and flotation components, and underwater escape components. The stabilization components consist of the stabilization glove, stabilization-brake parachute, and the pitch flaps. The stabilization glove which forms the forward portion of the wing glove is an integral part of the crew module. This glove section serves to stabilize the flight of the crew module until deployment of the recovery parachute. The pitch flaps, in the under surface of the glove section, assist in maintaining crew module horizontal stability. The stabilization-brake parachute, which is contained in a compartment in the center of the top aft section of the glove, is used

Crew Module General Arrangement (Typical)



CANOPY CENTER BEAM ASSEMBLY

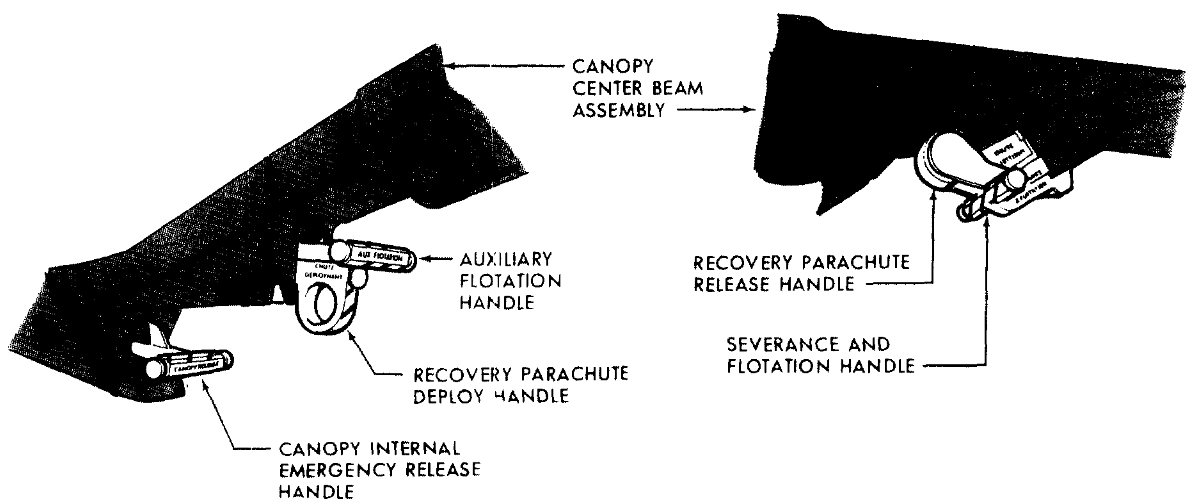


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Figure 1-47A.

Crew Module Seats

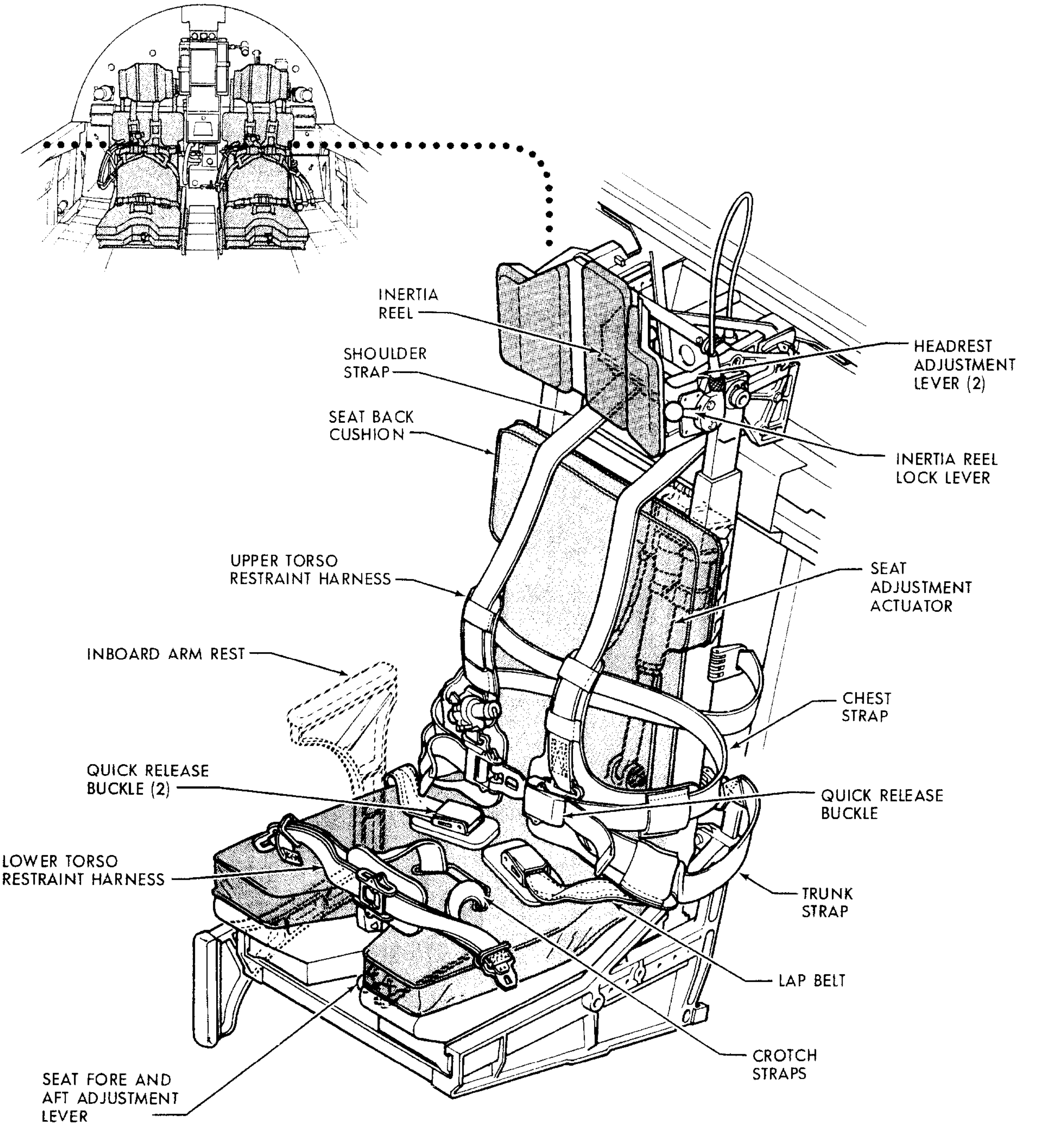

Figure 1-47A.

to decelerate the crew module and assist in maintaining stable flight prior to recovery parachute deployment. The stabilization-brake parachute is a six foot diameter ribbon type parachute attached by two bridles to the outboard aft sections of the glove section. The recovery parachute has a ringsail canopy with a 49 foot deployed diameter. The parachute is attached by two bridles to the crew module so that the module will maintain an upright and level attitude during descent. The parachute is housed in a container located between the seat bulkhead and the aft pressure bulkhead. This container rests on the parachute catapult pan. The catapult forcibly deploys the parachute at a velocity sufficient to ensure proper bag strip-off. A q operated selector monitors airplane speed to select one of three possible time delays in unlocking a barostat initiator. When below 15,000 feet, the barostat initiator, if unlocked will fire and in turn fire the catapult to deploy the recovery parachute. The parachute is initially deployed in a reefed configuration. The parachute is disreefed by three cutters which sever the reefing line shortly after line stretch is reached. The landing and flotation components consist of an inflatable landing impact attenuation bag, flotation bags and self-righting bags. The impact attenuation bag, located in the crew module floor, inflates automatically during descent and serves to cushion the landing impact. Regulated pneumatic pressure for inflation of the bag is contained in two storage bottles in the crew module. Pressure within the bag is maintained at 2 psi. Although the crew module is watertight and will float, additional bouyancy is provided by a flotation bag at each aft corner of the glove section and by an auxiliary flotation bag at the front of the crew module. Inflation of the aft flotation bags is accomplished either manually by use of a T-handle initiator in the cockpit or automatically by action of the underwater severance initiator. Inflation of the auxiliary flotation bag is accomplished manually by a T-handle initiator in the cockpit. The pressure source for inflation of the flotation bags is contained in two storage bottles located in the crew module. The underwater escape components provide either manual or automatic crew module severance, inflation of the aft flotation and self-righting bags, and actuation of the emergency oxygen system. This system is used in the event that an airplane has ditched with the crew module still attached. Manual actuation is accomplished by pulling a T-handle initiator in the cockpit. Automatic actuation is accomplished by an underwater severance initiator. This initiator when submerged in water and sensing a depth of between 10 and 20 feet will actuate to perform the same functions as the T-handle initiator.

SURVIVAL EQUIPMENT. (12)→

The survival equipment consists of locating aids, special equipment, and standard survival equipment. The locating aids consist of a chaff dispenser, an AN/URT-21 radio beacon set, an AN/URC-10 radio set, and a portable distress beacon light. The chaff dispenser, when armed, will activate to dispense chaff automatically during the ejection sequence. A control lever in the cockpit is provided to either arm or disarm the dispenser prior to ejection. The AN/URT-21 radio beacon set (3A, figure 1-18), located in the right console, will emit an intermittent, modulated tone to aid in rescue operations. The manually operated set is connected to the crew module mounted emergency UHF antenna which erects upon ejection. The set may also be used with its own retractable antenna. The AN/URC-10 radio, located in the survival equipment stowage compartment, provides a means of voice communication. The portable distress beacon light, also located in the survival equipment stowage compartment, produces a powerful flashing light to aid in night rescue operations. The special survival equipment consists of two air ventilation masks and a combination bilge/flotation bag inflation pump. The air ventilation masks located in the survival equipment stowage compartment, are provided for use when the canopy hatches must remain closed because of rough seas or inclement weather. The mask hoses may be connected to air mask connector valves located adjacent to the crew seats. An air supply tube leads from each connector valve to an outside opening well above the water line. The combination bilge/flotation bag inflation pump is operated by fore and aft motion of the control stick. This will cause simultaneous pumping of water overboard and inflation of the flotation bags. Over-inflation of the bags is prevented by relief valves. Standard survival equipment is provided for all climatic conditions. This equipment is stored in the survival equipment stowage compartment behind the pilot's seat. Access to the compartment is obtained by rotating the pilot's seat forward and down and disconnecting the seat actuator. The contents of the survival equipment stowage compartment will be determined by the applicable using command.

CREW MODULE EJECTION SEQUENCE. (12)→

When either ejection handle is pulled, the following ejection sequence (figure 1-47C) occurs automatically. Pulling either handle fires an initiator that simultaneously retracts both inertia lock reels, actuates the emergency oxygen and cabin pressurization systems, actuates the chaff dispenser, if armed, fires the explosive guillotines, ignites the rocket motor, and unlocks the manual recovery chute deployment handle. Pressure buildup of the rocket motor fires two additional initiators. The first initiator acts as a backup to actuate the emergency oxygen and cabin pressurization system, the chaff dispenser, and guillotines, and also activates the crew module severance system. The FLSC detonates and severs the crew module from the airplane. The second initiator actuates the stabilization-brake parachute, the thrust reducer, and unlocks the barostat initiator. When the barostat initiator is unlocked and senses an altitude below 15,000 feet, it will fire and ignite the SMDC train to remove the recovery parachute severable cover and fire the parachute catapult. This barostat initiator, will also remove the impact attenuation bag severable cover and fire the explosive valves in the impact attenuation bag air bottles causing the bag to inflate. A third function of the barostat initiator

Crew Module Ejection Sequence

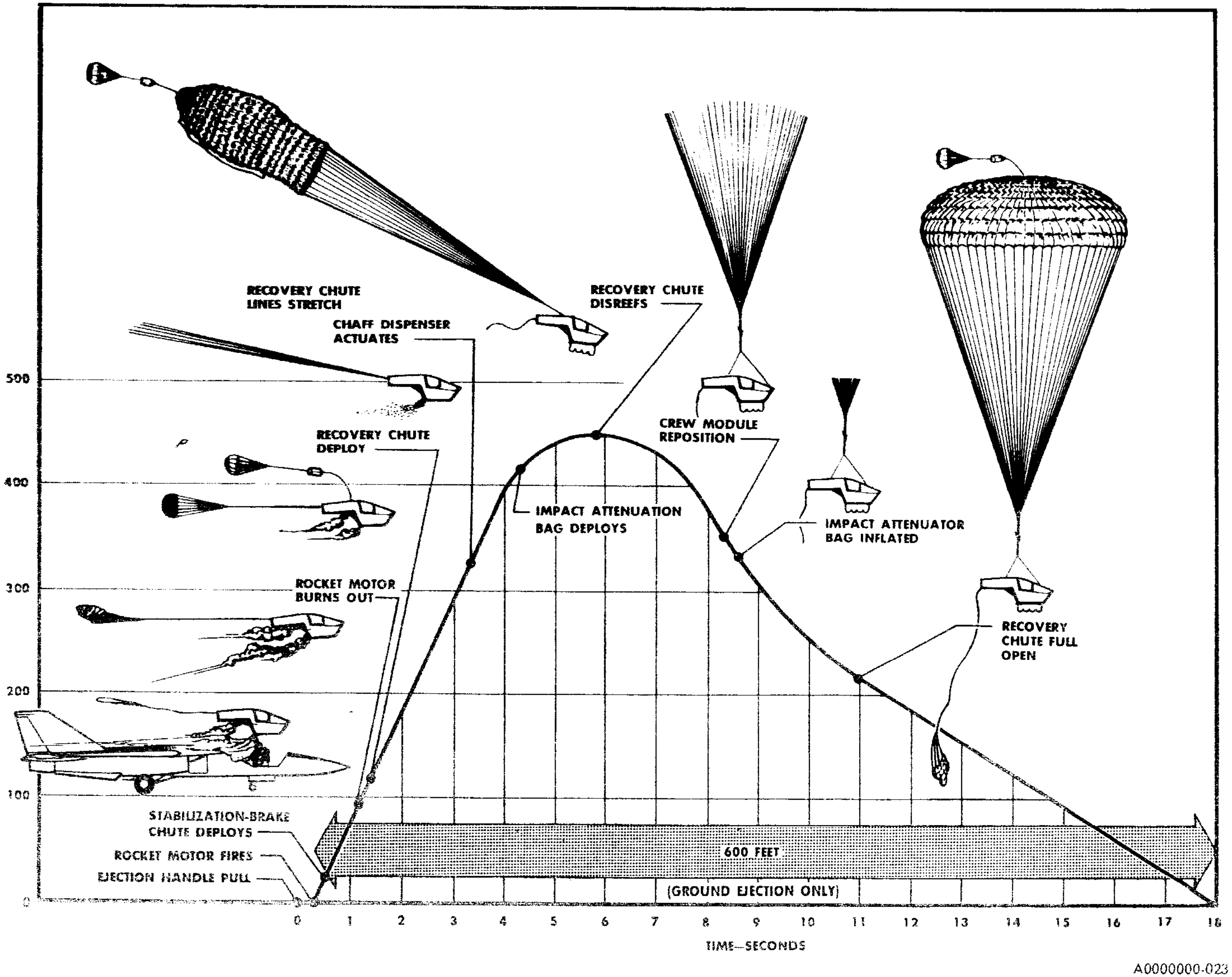

Figure 1-47C.

is, to erect the emergency UHF antenna and to fire the explosive pin retractor, releasing the repositioning bridle cable which allows the crew module to assume the correct touchdown attitude.

DITCHING ESCAPE SEQUENCE. (12)→

If the airplane is ditched, crew module severance and flotation bag deployment may be initiated either manually or automatically. If the severance and flotation handle is pulled or the automatic underwater severance initiator is actuated, the following sequence of events occurs: An initiator is fired to (1) fire the FLSC to separate the crew module from the airplane, (2) remove the severable covers over the aft flotation bags and the self-righting bags, (3) fire the explosive valve in an air storage bottle to inflate the aft flotation bags and the left self-righting bag and (4) fire the explosive valve in an air storage bottle to inflate the right self-righting bag.

Seat Forward and Aft Adjustment Lever. (12)→

The seat forward and aft adjustment lever (figure 1-47B), located in front of the seat pan between the crewmember's legs, is provided to unlock the seat from the carriage to allow forward and aft adjustment. When the handle is pulled up, the seat will unlock to allow a maximum of 3 inches travel from full aft to full forward. Since this lever does not provide headrest adjustment, forward and aft adjustment of the seat will result in a tilting of the seat back.

Seat Adjustment Switches. (12)→

Vertical adjustment of each seat is provided by a switch located on each left and right sidewall (2, figure 1-13A and 4, figure 1-45A) adjacent to the seat. Each switch has positions marked UP and DOWN and is spring-loaded to the center unmarked OFF position. Positioning a switch to either UP or DOWN energizes an electrical actuator to raise or lower the seat as selected. The seat has a maximum vertical travel of 5 inches.

Headrest Adjustment Lever. (12)→

A headrest adjustment lever (figure 1-47B), located on either side of each seat headrest is provided for fore and aft adjustment of the headrest. Depressing either lever will unlock the headrest allowing it to be moved either forward or aft. Releasing the lever will lock the headrest in place. Since the seat back is attached to the headrest, fore and aft movement of the headrest will cause the seat back to tilt.

Inertia Reel Control Handle. (12)→

The inertia reel control handle (figure 1-47B), located on the left side of each seat headrest is provided to lock or unlock the inertia reel.

Ejection Handles. (12)→

Two ejection handles (2A, figure 1-17), one located on either side of the center console adjacent to the crewmember's seat, are provided to initiate the ejection cycle. When the lock release on the top of handle is depressed the handle is released and may be pulled out. Pulling the handle out approximately 1/2 inch will fire the initiator to start the ejection sequence.

Recovery Parachute Deploy Handle. (12)→

The ring-shaped recovery parachute deploy handle (figure 1-47A), located on canopy center beam assembly, is provided as an emergency means of deploying the recovery parachute should the normal method fail. Pulling the handle will fire an initiator to unlock the barostat and will also fire the stabilization brake parachute catapult. The chute deploy handle cannot be pulled until an ejection initiator has been fired.

Recovery Parachute Release Handle. (12)→

The ring-shaped recovery parachute release handle (figure 1-47A), located on the canopy center beam assembly, is provided to release the recovery parachute from the crew module after landing. Pulling the handle fires the parachute release retractors at the bridle attaching points releasing the bridles from the crew module. The recovery parachute release handle cannot be pulled until the severance and flotation handle has been pulled.

Auxiliary Flotation Handle. (12)→

The T-shaped auxiliary flotation handle (figure 1-47A), located on the canopy center beam assembly, is provided to inflate the auxiliary flotation bag on the front of the crew module. Pressing a release button on either side of the handle and pulling the handle out fires an initiator which in turn removes the severable cover over the auxiliary flotation bag and fires an explosive valve in an air storage bottle to inflate the bag.

Severance and Flotation Handle. (12)→

The T-shaped severance and flotation handle (figure 1-47A), located on the canopy center beam assembly, is provided for escape in the event the airplane has ditched. Pressing a release button on either side of the handle and pulling the handle out will fire the FLSC and guillotines, separating the crew module from the airplane, and will inflate the aft flotation bags and the self righting bags. Pulling the handle will also activate the emergency oxygen system.

Control Sticks. (12)→

The control sticks are used after a water landing as a combination bilge/flotation bag inflation pump. After landing, the bilge pump drive connector pin is removed from the pin stowage hole and inserted in the operating hole. This connects the pump to the control stick. A plunger, adjacent to the pin stowage hole, must be pushed in to open the pump air and water outlet valves. Fore and aft motion of the stick will then operate the pump. For description of the control sticks, refer to "Flight Control System," this section.

EJECTION SEATS. (1)—(11)

Each crew station is equipped with a rocket catapult open ejection seat (figure 1-49). The ballistically initiated rocket catapult operates independently of other airplane systems to permit emergency escape during flight. The seat is electrically adjustable vertically and horizontally. The seat headrest is a recessed V-type, covered with foam rubber. A safety latch handle, located in the center of the headrest, rotates downward to safety the seat ejection controls while on the ground. The latch must be stowed to the flush (armed) position before the crew member's head will fit in the head rest. The seat incorporates personnel restraint equipment composed of an inertia reel, shoulder harness and lap belt, a parachute with an automatic barometric time delay actuator, a survival kit, a seat pan cushion containing an emergency oxygen system, two pneumatic seat man separators, and two sets of ejection handles. Ejection is accomplished by either pulling the face screen ejection handles down or by pulling the secondary ejection handle up. (See figure 1-48 for ejection sequence.) The first few inches of travel of either handle will jettison the canopy. As the canopy leaves the airplane a canopy/ejection handle interlock release cam is actuated allowing additional travel of the handles to release the rocket catapult firing pin.

Note

> If the canopy will not jettison with the ejection handles the seat cannot be ejected until the canopy is jettisoned with the canopy jettison T-handle, located on the canopy center beam.

During ejection the crew member is held firmly in the seat by the inertia reel shoulder harness which is attached to the crew member's torso harness across the shoulders and by the lap belt which is attached to the torso harness on each side. As the seat starts to move up the seat rails, the oxygen and communications lines pull free of the airplane systems and the emergency oxygen supply located under the seat pan is activated by a lanyard. As the seat continues to move up the rails, a trip mechanism fires a 3/4 second delay initiator which supplies gas pressure to release the restraints, disconnects the ejection handles, and opens a nitrogen bottle to inflate the seat-man separator bladders and forcibly separate the crew member from the seat at the height of seat trajectory. As the crew member is separated from the seat, the automatic parachute time delay actuator is armed by a lanyard attached to the seat. In the event the seat-man separator fails, a harness release handle located on the right side of the seat must be pulled to disconnect the restraints; then the crew member must manually push free of the seat. Separation in this manner will not arm the automatic parachute opener and the crew member must pull the manual D-ring on the left shoulder strap of the parachute harness to deploy the parachute. For a description of the seat pan and survival kit, refer to "Emergency Equipment", this section.

SURVIVAL EQUIPMENT.

The survival equipment consists of a seat pan and a survival kit which contains survival gear and a one man life raft. The seat pan and survival kit fit into the bottom recessed area of the ejection seat. The seat pan is attached along its rear edge to the bottom of the parachute. The seat pan serves as a seat cushion and contains the emergency oxygen system. An oxygen and communication lead from the airplane systems enters the seat pan at the left rear corner, goes through an attachment point, and exits to provide a point for the crew member to plug in his communications lead and oxygen regulator. The survival kit is attached to the bottom of the seat pan and to the safety belt on each side. A lanyard is provided on the right rear of the kit to attach the kit and life raft to the crew members' torso harness after bailout. The kit is divided into two compartments. These items are contained in an inner container that is connected to the life raft by a lanyard. The rear compartment of the kit contains a one man life raft. A D-ring on a cable extending from the right rear of the kit must be pulled to open the rear of the compartment to allow access to the life raft. A CO_2 bottle release line attached to the raft must be manually pulled to inflate the raft.

PERSONNEL RESTRAINT EQUIPMENT.

Each ejection seat is equipped with personnel equipment restraints consisting of an inertia reel, a shoulder harness, and lap belt. The inertia reel is located in the head rest area of the seat and in controlled by a handle on the left arm of the seat. When unlocked, the inertia reel will allow the shoulder harness to extend and retract automatically to allow freedom of movement of the crew member. When an excessive G force is encountered or when manually locked, the inertia reel will prevent further harness extension and will take up as the crew member returns to a normal sitting position. Fittings on one end of each shoulder harness strap are attached into the inertia reel at the head rest. The other end of each strap is sewn on to each parachute riser line strap and is equipped with a quick release hook fastener which attaches the shoulder harness and parachute risers to buckles on the chest of the torso harness worn by the crew member. The manual parachute deployment D-ring is mounted on the left shoulder harness strap. The lap belt consists of two short straps attached at one end on each side of the seat. The other end of each strap is equipped with quick release hook fasteners which attach to buckles on the leg straps of the torso harness worn by the crew

Ejection Sequence

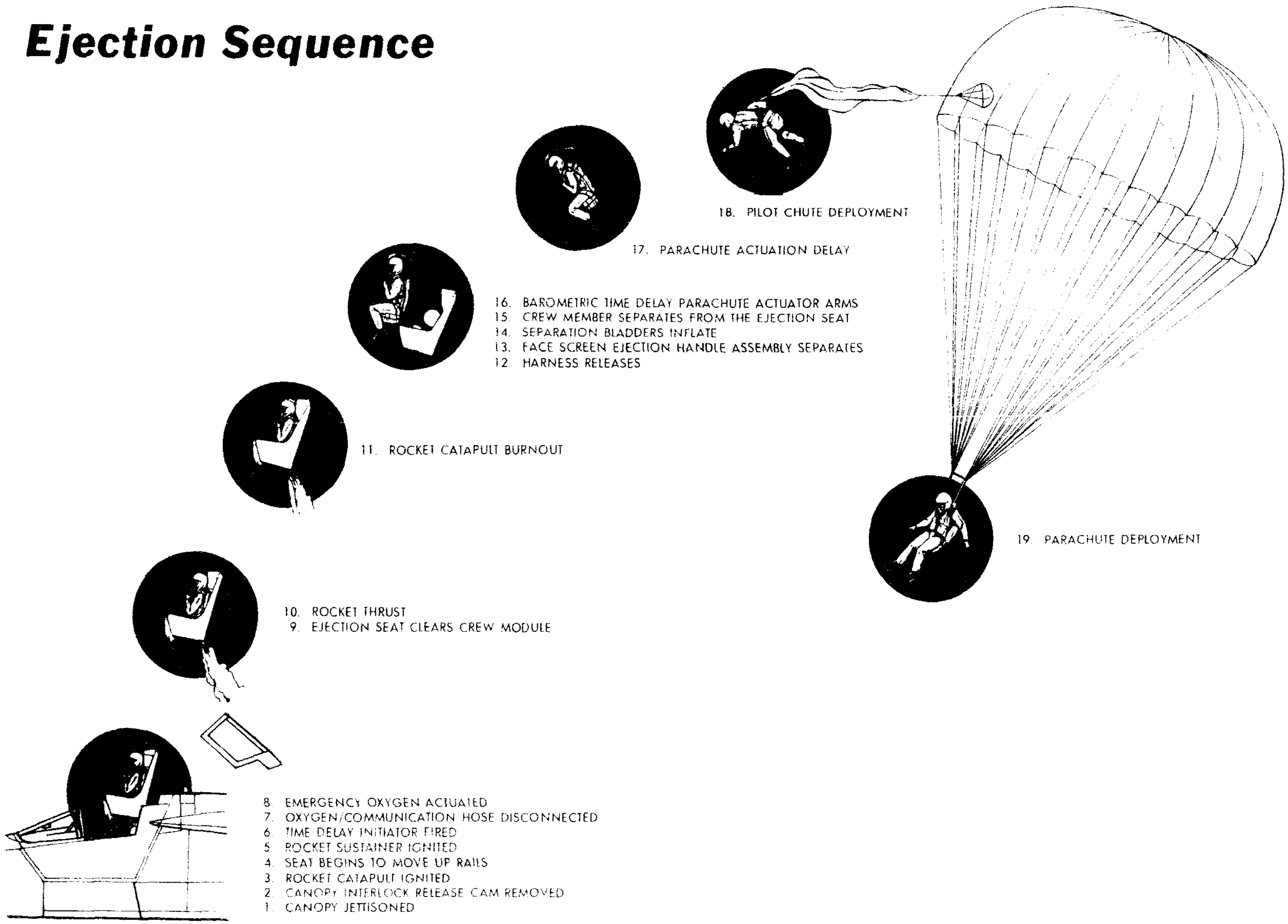

Figure 1-48.

member. Each lap belt strap is adjustable. During the ejection cycle, the lap belt straps are detached from the seat and the shoulder harness straps are released from the inertia reel to allow separation of the crew member from the seat.

SEAT-MAN SEPARATOR. (1)→(11)

The seat-man separator provides positive separation of the occupant and survival kit from the seat at the optimum point in the ejection cycle. Separation is accomplished by the rapid inflation of rubber bladders under the survival kit and behind the parachute. Operation of the seat-man separator is initiated by the firing of a 3/4 second delay initiator as the seat is ejected from the airplane. Gas pressure from the initiator releases the lap belt straps, shoulder harness straps, and ejection handles from the seat, and pierces the diaphragm on a nitrogen bottle to provide pressure to inflate the bladders and force the crew member out of the seat.

AUTOMATIC OPENING PARACHUTE. (1)→(11)

The ejection seat is equipped with an NB-9 automatic opening parachute. The parachute is recessed into the back of the ejection seat and normally serves as a back rest for the crew member. The parachute is equipped for use with a torso harness. The riser line straps from the chute canopy are sewn to one end of the shoulder harness straps. The combined riser line/shoulder harness end of each strap is equipped with a quick release hook fastener and latches into D-rings on the chest of the torso harness. The manual parachute D-ring is mounted on the left shoulder harness strap. The parachute is equipped with an automatic barometric time delay actuator which is armed by a lanyard when the seat-man separator separates the crew member from the seat during the ejection cycle. If the crew member ejects below 15,000 feet, the parachute will automatically deploy after a 2 second delay. If above 15,000 feet the parachute will deploy as the crew member free falls through 15,000 (±500) feet. Should the crew member have to manually separate from the seat after ejection, the automatic parachute time delay actuator will not be armed and the manual D-ring will have to be used to deploy the chute. After landing, the crew member can detach himself from the parachute riser lines by depressing the quick disconnect fasteners on his torso harness.

EJECTION SEAT CONTROLS. (1)→(11)

Face Screen Ejection Handles. (1)→(11)

A pair of flexible loop face screen ejection handles, located directly above the ejection seat on each side of the headrest (figure 1-49), are the primary ejection control for the seat. The handles are grasped by each hand and pulled down in front of the face to initiate the ejection cycle. In this position, the nylon screen between the handles covers the crew member's faces to afford wind blast protection during ejection. Travel of a cable pulled by the handles is in two distinct steps. Initial travel of the handles jettisons the canopy. As the canopy leaves the airplane, a canopy/ejection handle interlock release cam is pulled releasing the ejection handle for additional travel. Continuing to pull the handles initiates seat ejection.

Note

> If the canopy will not jettison with the ejection handles, the seat cannot be ejected until the canopy is jettisoned with the canopy jettison T-handle located on the canopy center beam.

Pulling either face screen ejection handle will eject the seat.

Secondary Ejection Handle. (1)→(11)

A single flexible loop secondary ejection handle is located on the front of the seat pan between the crew member's legs (figure 1-49). This handle provides an alternate method for ejection in case emergency conditions make reaching the face screen ejection handles impossible or impractical. The function of the ejection cycle is the same for the secondary ejection handle as for the face screen ejection handle except, it does not afford face protection.

Ejection Safety Latch Handle. (1)→(11)

The ejection mechanism of the seat is safetied by the ejection safety latch handle located vertically in the center of the headrest (figure 1-49). The handle is hinged at the bottom and spring loaded to the stowed or armed position. To safety the seat, the handle must be rotated down from the top to the horizontal position, where a spring loaded safety lever bar locks it in place. When in this position, the ejection mechanism is mechanically locked and neither of the ejection handles can be actuated. The handle extends outward approximately 6 inches so that crew member's head cannot fit into the headrest while the bar is extended. To arm the seat, the spring loaded safety lever bar must be depressed to its stop and the ejection safety latch handle allowed to move upward into the stowed or armed position. The handle is painted in cross hatched black and yellow.

Inertia Reel Control Handle. (1)→(11)

The inertia reel control handle, located on the left side of the ejection seat (figure 1-49), is marked LOCKED and UNLOCKED. A flexible control cable couples the lever to the inertia reel, permitting manual locking and release of the inertia reel. Normally the inertia reel control handle is in the UNLOCKED (aft) position. When the inertia reel locks due to G forces, it is necessary to cycle the control handle to LOCKED (forward), then to UNLOCKED (aft).

Ejection Seat

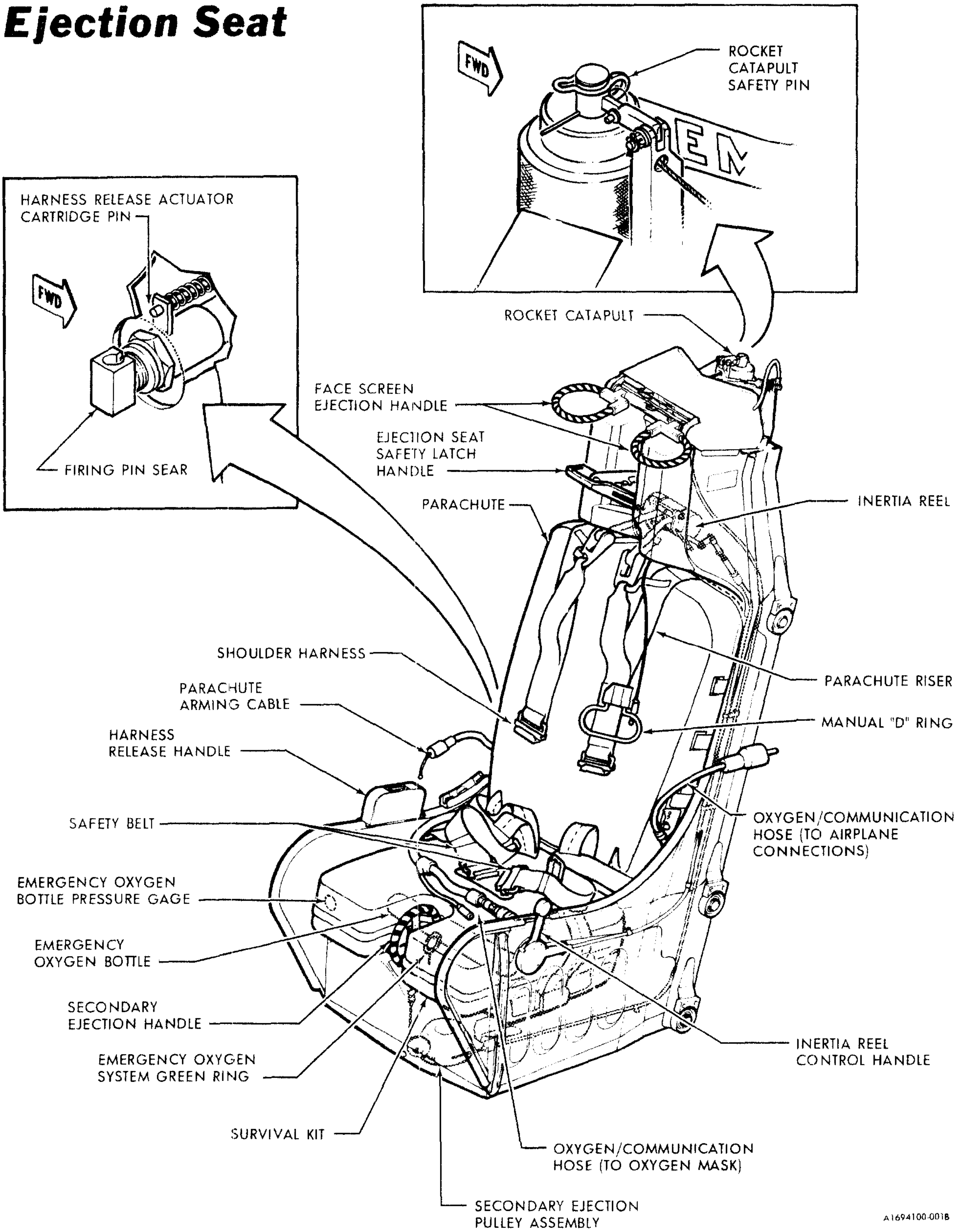

Figure 1-49.

Seat Adjustment Switches. (1)→(11)

Two toggle-type seat adjustment switches, located on each sidewall (2, figure 1-13A) and (4, figure 1-45A) are provided for each ejection seat. One is for vertical adjustment, the other for horizontal adjustment. The seat adjustment switches are three position switches spring loaded to the center unmarked OFF position. By positioning either switch to one of its momentary positions (UP, DOWN, FWD, or AFT), the seat may be adjusted for personnel comfort. The aircraft commander's seat adjustment switches are located on the left side wall. The pilot's seat adjustment switches are located on the right side wall.

Harness Release Handle. (1)→(11)

The harness release handle, located on the right side of the seat (figure 1-49), provides a means of mechanically disconnecting the shoulder harness and lap belt from the seat. Squeezing and pulling the handle actuates a cable which releases the lap belts and shoulder harness. The handle serves a dual purpose, to provide a backup means of separating the crew member from the seat in the event the seatman separator fails during ejection and/or for rapid evacuation from the airplane in the event of an emergency on the ground.

WARNING

Separation from the seat after ejection by use of the handle will render the automatic parachute time delay actuator inoperative; therefore, the crew member will have to pull the manual D-ring to deploy the chute.

The handle may also be used to release the survival kit for removal from the seat for inspection. Once the handle has been actuated, the inertia reel straps and lap belt will be released from their attaching points and must be re-attached. The harness attachment points are reset by pulling a harness reset button on the right rear of the seat.

Ejection Seat Ground Safety Pins. (1)→(11)

Safety pins with red identification streamers are installed in both rocket catapult firing pins. The pins are installed to prevent inadvertent actuation or firing of these devices and must be removed before flight.

WARNING

The absence of a red streamer does not necessarily mean the safety pin has been removed.

MISCELLANEOUS EQUIPMENT.

THERMAL RADIATION PROTECTION. (12)→

Thermal radiation protection for the crew is provided by side curtains on the canopy hatches and a hinged forward panel located between the glare shield and windshield.

Side Curtains. (12)→

Each curtain is mounted along the upper edge of each canopy hatch on either side of the center canopy beam. When stowed the curtain is folded as an accordion in the shape of a fan with the hinge forward. As the curtain is extended it unfolds to form an arc from the top rear to the bottom forward edge of the hatch. The rim of the arc rides in a track to form a light seal. When fully extended the forward edge of the curtain forms a light seal against the forward hatch structure, thus completely covering the canopy hatch glass. The curtain is retained in the stowed position by a spring tension latch. A handle labeled RADIATION CURTAIN is provided on the forward edge of the curtain to extend or retract the curtain. A positive latch on the forward seal locks the curtain in the extended position. A push button labeled CURTAIN RELEASE must be depressed to release the curtain for retraction. A decal located adjacent to the curtain release button contains instructions for extending or stowing the curtain.

Forward Panel. (12)→

The forward panel is constructed in two sections to form a thermal radiation shield across the front of the cockpit between the top of the glare shield and the windshield. The panel is hinged along the aft edge of the glare shield and folds forward to lie on top of the glare shield when not needed. A slide catch on each section secures the panel against the glare shield. A cable lanyard attached to the slide catch is provided to unlatch the catch and erect each section. The right section must be raised first. When erected a friction catch retains the upper edge of each section against the windshield arch to provide a light seal. To stow the panel each crew member disengages the friction catch by pushing forward on his section adjacent to the catch. When disengaged the panel will fall forward on the glare shield. The slide catches on each section should be engaged to retain the panel in the stowed position. A decal located on the forward canopy hatch structure contains instructions for erecting and stowing the panel.

CREW ENTRANCE LADDERS AND STEPS.

Crew entrance ladders and steps, located on each side of the fuselage, provide crew access to the cockpit without the aid of ground support equipment. When not in use both sets of ladders and steps are retracted into the sides of the fuselage. Each left or right ladder and step can be electrically extended from inside the cockpit. Push-button releases are provided on the outside of the fuselage to manually extend the ladders and steps from the ground. The ladders and steps must be manually stowed from the ground.

Entrance Ladder Switch.

The entrance ladder switch (3, figure 1-16), located on the aft console, has three positions marked L (left), R (right) and OFF. Placing the switch to L or R will provide 28 volt d-c power to a solenoid in the respective ladder and step to release the ladder and step for extension. The switch is spring-loaded to the center OFF position.

ANTI-G SUIT.

Each anti-G suit is connected to the airplane pressure source by an anti-G suit hose located on the left console (26, figure 1-4) and aft console (11D, figure 1-16). On airplanes (1)→(11) the pilot's anti-G suit connections are located on the center console. Pressure for the anti-G suit is supplied from the engine compressor section. A test button marked ANTI-G PUSH-TO-TEST, located on the left console (24, figure 1-4) and aft console (11F, figure 1-16), is provided to check operation of the anti-G suit valve. When the button is depressed, the anti-G suit bladders will inflate. When the button is released, the bladders will deflate.

MIRRORS.

Four rear view mirrors, two on each side of the cockpit canopy frame (figure 1-2) are installed to permit the crew rearward vision without moving from their normal sitting position. The mirrors are adjustable in tilt only.

MAP STOWAGE.

The cockpit is furnished with two map cases, located on the left and right sidewalls (9, figure 1-13A) and (5, figure 1-45A). Each map case is installed in an upright position. The aircraft commander's map case is labeled MAPS & ENROUTE SUPPL. The pilot's map case is labeled MAPS. A nylon retaining strap, attached to each map case, extends upward, and attaches to the cockpit sidewall fairing.

DATA STOWAGE CASE.

The cockpit contains a black nylon vinyl coated data case located in the outboard aft end of the right console (7, figure 1-18). The data case consists of the case and a flap with a metal snap fastener to prevent data from inadvertently falling from the case. The case is labeled DATA STOWAGE.

LET DOWN CHART STOWAGE.

Two letdown chart stowage compartments are located on each side of the aft console (12, figure 1-16). The right compartment is labeled LETDOWN CHART HOLDER, the left compartment is labeled LETDOWN CHARTS. Each compartment is provided with a strap and fastener to secure the charts and holder. On airplanes (1)→(11) the compartments are located below the ground check panel on the aft console.

SAFETY PIN STOWAGE.

The safety pins are stowed in a nylon bag (figure 1-2) located on the aft wall above the aircraft commander's air conditioning vent. The metal zipper is placed diagonally across the bag. The bag is labeled EJECTION SYSTEM SAFETY PIN STOWAGE. On airplanes (12)→ the safety pin stowage compartment is located on the left sidewall (10, figure 1-13A).

SPARE LAMP AND FUSE HOLDER STOWAGE.

A nylon bag (figure 1-2) with a metal zipper is provided on the aft wall for spare lamps and fuses. The bag is the same as that provided for the escape system safety pins and is located opposite the escape system stowage bag. The bag is labeled SPARE LAMPS & FUSES HOLDER STOWAGE. On airplanes (12)→ the spare lamp and fuse holder is located outboard of the pilot's relief container compartment (6, figure 1-45A).

CHECKLIST STOWAGE. (12)→

A space for stowing the checklist is provided on the left sidewall (1, figure 1-13A). A nylon strap retains the checklist in place.

FOOD STOWAGE COMPARTMENT. (12)→

A food stowage compartment is provided for the crew on the aft bulkhead at the left of the aircraft commander's seat. The door of the compartment is held closed by a springloaded latch.

CHART BOARD STOWAGE. (12)→

Space is provided for stowage of a chart board on the right side of the pilot's seat. A fabric strap snaps over the chart board when it is stowed to hold it securely in place.

RELIEF CONTAINER STOWAGE. (12)→

Relief containers for each crew member are located in small compartments on the aft bulkhead, outboard of each seat. Each compartment is enclosed by a fabric cover with a zipper opening. The relief containers are plastic bottles with screw caps to prevent leakage. Each bottle holds approximately 3 pints.

HOOD STOWAGE COMPARTMENT. (12)→

A hood stowage compartment, located on the right side of the aft bulkhead just above the pilot's relief container, is provided to store the attack radar scope hood.

CORRECTION CARD HOLDERS.

The cockpit is provided with three correction card holders. One four inch holder is attached underneath the left-aft section of the glare shield. The card holder is labeled EPR SETTING. One four inch and one two inch card holders are attached underneath the right-aft section of the glare shield on the approximate centerline of the airplane. The four inch holder is labeled UHF FREQUENCY CARD. The two inch holder is labeled COMPASS CORRECTION CARD and is to the right of the four inch holder. Each card holder is attached by spring tensioned hinges riveted to the glare shield. The springs allow the card holders to be pulled into position for reading purposes and held secure against lower side of glare shield when released.

LET DOWN CHART HOLDER.

A let down chart holder (figure 1-2) is provided to clearly display letdown charts where they can be easily followed during instrument letdowns. The holder is a rectangular transparent pane the size of a letdown chart and is attached in a swivel socket on the canopy center beam. It can be swivelled to the left or right and latched in place for use by either crew member. The holder has both red and white lighting which can be mixed as desired by control knobs located on the top of holder. The holder is stowed in a receptacle in the aft console when not in use.

ARM RESTS. (12)→

Arm rests (3, figure 1-13A and figure 1-145A) are provided on each sidewall. When not in use they are folded upwards under the canopy sill.

LIQUID CONTAINERS. (12)→

Two insulated liquid containers provide the crew with hot or cold liquids during flight. The containers are stowed in recessed receptacles in the seat bulkhead, outboard of each head rest. A spring loaded latch on the front of each receptacle holds the respective container firmly in place against a coil spring in the bottom of the receptacle when the container is stowed. Each container holds approximately 1 quart.

STARTER CARTRIDGE STOWAGE CONTAINER.

A starter cartridge stowage container, located on the left forward side of the main landing gear wheel well, is provided to carry two spare starter cartridges. The container is made of plastic and has a detachable cover to allow servicing or access to the spare cartridges when needed.

Servicing Diagram

NOTES:

1. Hydraulic Oil—MIL-H-5606
2. Air—MIL-P-5818 or Nitrogen—FS BB-N-411a, Type I, Grade B
3. Oil—MIL-L-7808
4. Demineralized Water MIL-D-4024
5. Fuel—MIL-J-5624 (JP-4)
6. Liquid Oxygen—MIL-O-27210A, Grade A, Type II

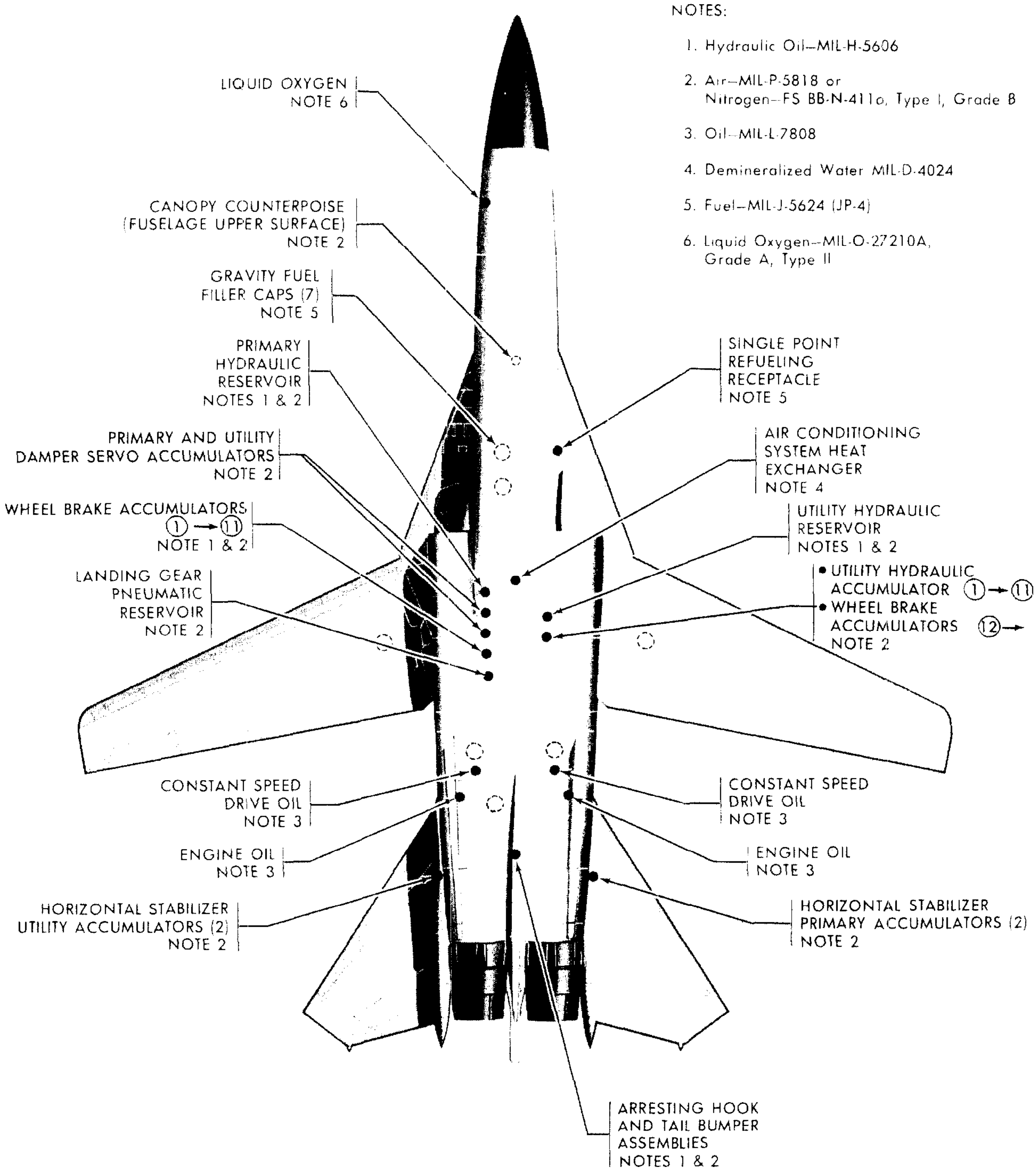

Figure 1-50.

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SECTION II

NORMAL PROCEDURES

TABLE OF CONTENTS.

As indicated in Section I, various systems are designed for automatic operation. However, for different intervals of time during the Flight Test program, it is necessary to limit some systems to manual operation within certain flight envelopes. The procedures in this section reflect the currently recommended modes of operation and will be up-dated in future changes to the manual.

Note

- Items coded (AC-P) are applicable to both the aircraft commander and the pilot. Items coded (P) are applicable to the pilot only, and items not coded are applicable to the aircraft commander only.
- Differences between airplanes are designated by number symbols such as (8), (10), (12)→ (16), etc. For an explanation of airplane code numbers, refer to "Airplane Designation Codes," in the front of the manual.

PREPARATION FOR FLIGHT.

FLIGHT RESTRICTIONS.

Refer to Section V for the operating limitations imposed on the airplane.

FLIGHT PLANNING.

Refer to Appendix I to determine takeoff, cruise control, fuel planning and management, and landing data necessary to complete the mission.

TAKEOFF AND LANDING DATA CARDS.

Refer to Appendix I for information necessary to complete the Takeoff and Landing Data Card in the Flight Crew Checklist. Recheck data just prior to flight to determine the effect of atmospheric, runway, or airplane configuration changes. If required, revise Takeoff and Landing Data Card to reflect latest information.

WEIGHT AND BALANCE.

Refer to Section V for weight limitations and to the Manual of Weight and Balance Data, T.O. 1-1B-40, for airplane and crew module loading information.

Note

(12)→ The crew module should not be considered flyable without its full crew and complement of survival equipment, or the equivalent ballast to maintain center-of-gravity. To assure stability of the module in event of ejection, it must be loaded in accordance with T.O. 1-1B-40.

CHECKLISTS.

This Flight Manual contains only amplified procedures. Abbreviated flight crew checklist T.O.'s 1F-111(Y)A-1CL-1 and -2 are issued as separate documents.

ENTRANCE.

To extend either entrance ladder, proceed as follows:

1. Entrance step - Extend. (AC-P)
   Rotate screw in center of step until step extends.

2. Entrance ladder - Extend. (AC-P)
   Support the entrance ladder door by hand and depress the unlock button on the door. Allow the door to extend.

Note

After entering the airplane, the entrance ladder and step must be manually stowed by the ground crew.

PREFLIGHT CHECK.

BEFORE EXTERIOR INSPECTION.

1. Form 781 - Check for airplane status and release. (AC-P)

EXTERIOR INSPECTION.

The exterior inspection is based upon the fact that maintenance personnel have completed all of the requirements of the Scheduled Inspection and Maintenance Requirements Manual for preflight and post flight: therefore, duplicate inspections and operational checks of systems have been eliminated except for those needed in the interest of flight safety: Following the route shown in figure 2-1, check all surfaces for any type of damage: signs of fuel, oil, hydraulic or other fluid leaks that may have developed since the preflight inspection. Check all access doors and covers for security.

Note

With the airplane parked at a certain angle to the prevailing wind, it is not unusual for the engine fan and compressor to windmill. This rotation will cause no damage.

BEFORE ENTERING COCKPIT.

1. Windshield and canopy glass - Check. (AC-P)
   Check all enclosure glass for condition and cleanliness.

2. Canopy hatches - Check. (AC-P)
   Check the hatch for proper operation, condition and condition of seal.

3. Canopy external emergency release handle - Check. (P)
   Check that plunger of the release handle is sealed and flush with fuselage surface.

4. (1)→(11) Parachute - Check. (AC-P)
   Check for proper installation of the parachute in the seat and condition of parachute pack and harness. Check that the parachute part number is 12K090-3 (containing a 15,000 foot automatic opener).

INTERIOR INSPECTION.

Power Off.

1. (1)→(11) Canopy jettison handle safety pin - Removed. (P)

2. (1)→(11) Ejection seat - Check. (AC-P)

   a. Emergency oxygen green ring - Checked (AC-P)
      Check that the green ring is properly stowed in the pocket on the left forward face of the seat pan.

   b. Emergency oxygen bottle pressure - Checked. (AC-P)
      Check the emergency oxygen bottle gage on the forward right side of the seat pan for minimum pressure of 1800 psi.

   c. Harness release actuator cartridge - Checked. (AC-P)
      Check that the harness release actuator cartridge on the right side of the seat pan is installed. A protruding red pin indicates that a cartridge is not installed. Check the firing pin sear for proper installation.

   d. Rocket catapult safety pin (1) and canopy jettison initiator safety pin (1) - Check removed. (AC-P)

   e. Parachute arming cable - Checked. (AC-P)
      Check that the swaged ball on the end of the parachute automatic opener arming cable is properly retained in the harness release handle.

   f. Harness release handle - Checked. (AC-P)
      Check that the harness release handle on the right arm of the seat is down and secure.

   g. Shoulder harness, lap belt, and inertia reel-Connected and checked. (AC-P)
      Attach the parachute riser-shoulder harness release fittings to the torso harness upper buckles and the lap belt fittings to the lower torso harness fittings. Check operation of the inertia reel in the locked and unlocked positions.

3. (12)→ Crew module - Check. (AC-P)

   a. Ejection handle safety pins (2) - Installed.
      Before entering the cockpit, check that a safety pin is installed in each ejection handle.

Exterior Inspection

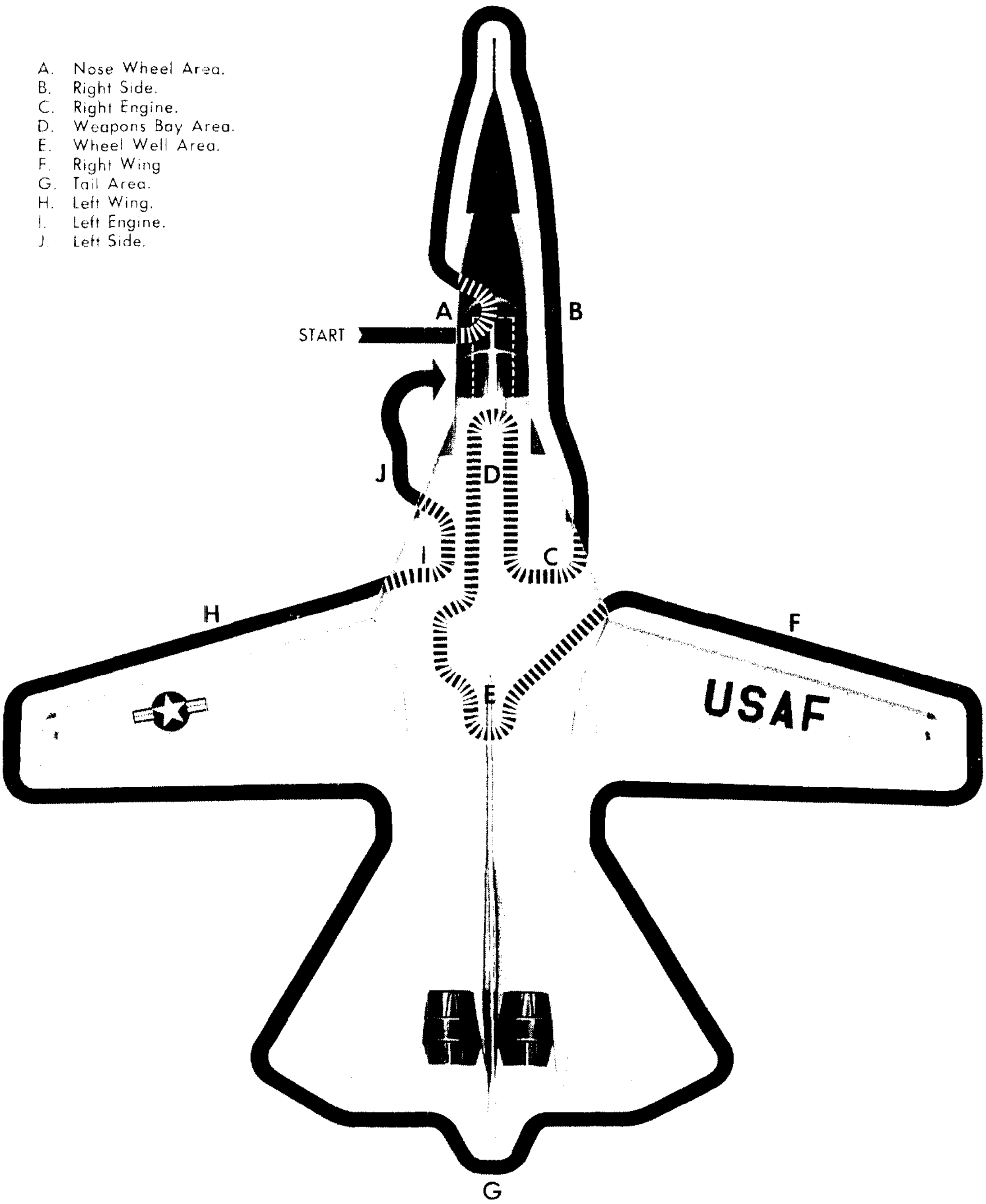


Figure 2-1.

b. Severance and flotation, recovery parachute release, auxiliary flotation and canopy internal emergency release handle safety pins (3) - Removed.
Upon entering the cockpit, check that the above safety pins are removed.

c. Emergency pressurization bottle pressure - 3000 psi minimum at 70°F.

d. Emergency oxygen bottle pressure - 1800 psi minimum at 70°F.

e. Survival equipment compartment - Checked.
Check the survival equipment compartment for security, seals intact.

f. Upper and lower torso restraint harness and inertia reel - Checked. (AC-P)
Check the condition of the restraint harness. Check operation of the inertia reel in the locked and unlocked position.

g. Oxygen regulator - Inserted in torso harness receptacle. (AC-P)

(1) Oxygen mask and communication cord - Connected. (AC-P)

(2) Oxygen lever - ON, flow check. (AC-P)
Check that there is a normal flow of oxygen.

h. Anti-G suit hose - Connect and check. (AC-P)
Check for proper routing and make sure that the hose is not restricted by personal harnesses.

i. Pressure suit vent knob - Full CCW.

4. Canopy external emergency release initiator safety pin (1) - Check removed. (P)

5. All switches, knobs and controls - Off, normal or safe. (AC-P)

6. (12)→ Foreign object damage prevention door switch - RETRACT.

7. Flap and slat handle - Corresponds with surface position.

8. Anti-skid switch - ON.

9. (1) Flight instrument reference select switch - AUX.

10. (2)→ Flight instrument reference select switch - PRI.

11. Landing gear handle - DN.

12. Auxiliary brake handle - Pulled.

13. Utility hydraulic system isolation switch - ON.

14. Instrument coupler mode selector knob - NAV.

15. AFRS compass and latitude knobs - SLAVED and set.

16. Engine inlet anti-icing switch - AUTO. (if installed)

17. (1)→(11) Engine fire pull handles - IN.

CAUTION

If either fire pull handle has been pulled, do not attempt to start that engine. Check with crew chief to insure that the hydraulic shutoff valve has been reset prior to starting the engine. Starting an engine with the hydraulic shutoff valve closed could cause damage to the hydraulic system.

18. Landing gear alternate release handle - In.

19. Fuel tank pressurization selector switch - AUTO.

~~20. Translating cowl switches - AUTO.~~
SEE SS-8

21. Generator switches - ON.

22. Emergency generator switch - AUTO.

23. Air source selector knob - BOTH.

24. Pressurization selector switch - CBT.

25. Air conditioning mode selector switch - AUTO.

26. Aft console - checked.
Check all circuit breakers in and all switches on the ground check panel in proper positions.

27. Publications and flight data - Checked. (AC-P)
Check that all applicable current flight information publications are aboard.

Power On. (Aircraft Commander)

1. Battery switch - ON.
Check the engine turbine inlet temperature indicators. The power-off flag in the indicators will go out of view when the battery is on.

Note

If the engines are to be started using battery power, the following "Power On" checks must be delayed until the engines are running.

2. External power switch - ON (if applicable).
   If external power is to be used, place the external power switch ON and check that the electrical power flow indicator displays TIE.

3. Air conditioning - Check.
   Check that air conditioning is connected and functioning properly to provide equipment cooling.

4. TACAN, radar altimeter, and UHF radio - On.

5. Lighting control panel - Checked. (AC-P)
   Check operation of the interior light rheostats and set for desired intensity. Check operation of bright and dim switch and select desired intensity. Check external lights if night flight is anticipated.

6. Malfunction and indicator lamps - Checked. (AC-P)
   Pilot depresses the malfunction and indicator lamps test button, and aircraft commander and pilot check that all malfunction and indicator lamps light.

Note

With power on the airplane but without engines running the following lamps will normally be lighted:

L and R PRI HYD
L and R UTIL HYD
L and R FUEL PRESS
PITCH, ROLL and YAW DAMPER
PRI ATT/HDG
AUX/ATT
ANTI-SKID
CANOPY
L and R ENG OVERSPEED

If the damper channel caution lamps are lighted, depress damper reset button. If lamps remain lighted, a malfunction is indicated.

7. Interphone panel - Set and checked. (AC-P)
   Pull mixer knobs to ON and adjust volume on those functions that are to be used.

8. Seat and rudder pedals - Adjusted. (AC-P)
   Check operation of the seat and adjust seat and rudder pedals as desired.

9. Wing sweep handle - Corresponds with wing position.

10. (8)→ Oil quantity indicators - Checked.
   Check that indicators show 8 to 16 quarts, depress the oil quantity indicator test button, and check that indicators show decrease. Then, release test button and check that indicators return to original readings.

11. Oxygen quantity - Checked.
   Check that oxygen quantity is adequate for mission. Depress oxygen quantity test button; oxygen quantity indicator should decrease to zero. Note that the oxygen quantity caution lamp lights when indication is 2 liters. Release the test button and note that the caution lamp goes out and that the quantity indication returns to original value.

12. Landing gear position indicator lamps - Checked.

13. Lead computing optical sight - Set as required.

   a. LCOS mode select knob - Set to weapon delivery mode.

   b. Aiming reticle and command bar brightness knobs - Adjust for desired brilliance.

   c. Aiming reticle cage switch - As required.

   d. Range set knob - Set in accordance with weapon.

   e. Reticle depression set knob - Set to desired value.

   f. True airspeed set knob - Set to desired airspeed.

14. Radar altimeter - Checked and set.

Note

This check cannot be accurately performed in an enclosure such as a maintenance dock or hangar.

   a. Minimum altitude index pointer - 50 feet.

   b. Channel selector switch - CHAN 1.

   c. Power off warning flag - Out of view.
      After approximately 120 seconds warmup time, the power-off warning flag should disappear from view. The altitude pointer should read zero and the radar altitude low warning lamp should light.

   d. Radar altimeter control knob - Depressed.
      Observe that the altitude pointer drives to 100 (±10) feet and the radar altitude low warning lamp goes out.

   e. Radar altimeter control knob - Release.
      Observe the altitude pointer returns to zero and the radar altitude low warning lamp lights.

   f. Channel selector switch - CHAN 2.
      Repeat steps d and e.

g. Set the minimum altitude index pointer compatible with mission to be flown.

15. Fire detect circuit - Checked.
Check that both fire warning lamps light while holding the agent discharge/fire detect test switch to the FIRE DETECT TEST position.

16. AFRS synchronization indicator - Nulled.

17. Fuel quantity indicators - Checked.
Check fuel quantity, momentarily depress the fuel quantity indicator test button and check that the indicators show decrease. Release test button and check that indicators return to original readings.

18. Engine fuel feed selector knob - AUTO.
Place the engine feed selector knob to AUTO and check that fuel pump low pressure indicator lamps number 1 through 6 blink.

19. Engine feed selector knob - AUTO, or BOTH.
AUTO if forward and aft tanks are within normal 8200 (±300) pound differential.
BOTH if tanks are not within this differential.

20. Fuel transfer knob - WING (If fuel is in wing tanks).
Fuel pump low pressure indicator lamp 7, 8, 9, and 10 should blink.

21. Fuel transfer knob - OFF.

22. Translating cowl switches - EXTEND.

23. Translating cowl indicators - OPEN.

24. Ground check panel - Check switches in correct positions and close the access door.

25. Request ground crew to prepare to start engines.

Power On. (Pilot)

Refer to figure 2-2, Danger Areas for the extent of electronic equipment radiation hazard areas.

1. (2) TACAN - On.

2. Bomb nav mode selector knob - HEAT.

3. Platform heat indicator lamp - Lighted.

Note

The platform heat indicator lamp may not light if the stabilization platform has been operating within 30 minutes preceding this alignment.

4. Altitude/test selector knob - NORM.

5. Present position latitude counter - Checked.
If latitude is incorrect proceed as follows:

a. Platform alignment control knob - PLATFORM OFF.

b. Bomb nav mode selector knob - ALIGN.

c. Present position latitude counter - Set.

d. Bomb nav mode selector knob - HEAT.

e. Platform alignment control knob - NORMAL.

6. Magnetic variation counter - Check and set to local variation.

7. Platform heat indicator lamp - Out.

8. Bomb nav mode selector knob - ALIGN.

9. Present position longitude counter - Check and set if necessary.

10. Attack radar function selector knob - STBY.
Check that the antenna cage pushbutton indicator lamp is lighted and that the antenna tilt indicator indicates +30. A minimum of 40 seconds is required for warmup.

11. UHF radio - Checked.

12. Platform align indicator lamp - On steady after 1 minute, flashing after additional 4 minutes.
The platform align indicator lamp should light within approximately one minute, then commence flashing within an additional 4 minutes. However, if the airplane is parked in an area where the normal earth's magnetic variation is significantly distorted (i.e. magnetic variation is not accurately known) more time may be required. A flashing platform align indicator lamp indicates the platform is sufficiently aligned to meet specification performance.

13. Magnetic heading synchronization indicator - Nulled and steady (if time permits).
A nulled and steady condition may not occur until after the platform align indicator lamp begins flashing. If the indicator is not nulled and time permits, the best possible alignment of the platform can be obtained by allowing the magnetic heading synchronization indicator to a null. If the airplane is not to be moved immediately, the mode selector knob may be left in the ALIGN position until just before airplane movement. This will prevent any system error buildup during the waiting period.

14. Altitude alignment - Perform.

    a. Fixpoint elevation counter - Set to 3090.

    b. Altitude/test selector knob - ALIGN.

    c. Go lamp - Lighted.
       Rotate the altitude calibration knob until the go lamp lights.

    d. Altitude/test selector knob - NORM.

    e. Go lamp - Out.

15. Fixpoint elevation counter - Set.
    Set the fixpoint elevation counter to the elevation of the anticipated fixpoint.

16. Attack radar function selector knob - ON.
    The function selector knob should remain in the ON position for a minimum of 5 minutes. Check that the antenna cage pushbutton indicator lamp is lighted. If the lamp is out, depress the button to cage the antenna.

17. Bomb nav destination storage - Set.

    a. Fix mode DEST STORAGE 1 button - Depress.
       Computed course and miles to destination will remain at the computed values existing at the time the storage button is depressed. The attack radar cursor will be absent if the attack radar mode selector knob is in the GND AUTO or GND VEL position.
       When counters stop driving, enter number one destination storage coordinates into the destination position counters.

    b. Fix mode DEST STORAGE 2 button - Depress.
       When counters stop driving, enter number two destination storage coordinates into the destination position counters.

    c. Fix mode DEST STORAGE 3 button - Depress.
       When counters stop driving, enter number three destination storage coordinates into the destination position counters.

    d. Fix mode TARGET selector button - Depress
       Course and distance computations resume to the new destination. The attack radar cursors will be on the new destination if in range when the attack radar mode selector knob is in the GND AUTO or GND VEL position.

18. Destination counters - Set to coordinates of destination or first steering point.

19. Attack radar - Set.

    a. Antenna tilt indicator - Zero.

    b. Scope intensity control knob - As desired.

    c. Bezel/range marks - Checked and set.

    d. Range azimuth cursors - Checked and set.

    e. Function selector knob - XMIT.

    f. IF gain knob - Tune for best picture.

    g. Video adjustment knob - Tune for best picture.

    h. Antenna tilt control knob - Detent.

    i. Range selector knob - Desired range.

20. Countermeasures receiver function selector knob - As required.

BEFORE STARTING ENGINES.

Refer to figure 2-2, Danger Areas for the extent of personnel hazard areas.

1. Ground crew report - Ready for engine start.
   Fire guard posted, engine and run area clear, chocks in place, translating cowls open, external starter air available, ready for engine start.

2. (12)→ Foreign object damage prevention door switch - EXTEND.

STARTING ENGINES.

The engines can be started by using air pressure from a ground source or by a pyrotechnic cartridge. Electrical power required for engine starting may be supplied either by the airplane battery or by an external source. The initial start may be the left or right engine. With either engine operating, the remaining engine may be started by pneumatic crossbleed. Starting is normally accomplished by use of external electrical power and external pneumatic pressure source.

WARNING

- If battery power only is used during start, a check of the fire detection system cannot be made until one engine driven generator is supplying power to the ac buses.

- Do not attempt a pneumatic start or fly the airplane with an unfired cartridge in the breech. To do so could result in damage to the cartridge causing it to explode if used for subsequent starts.

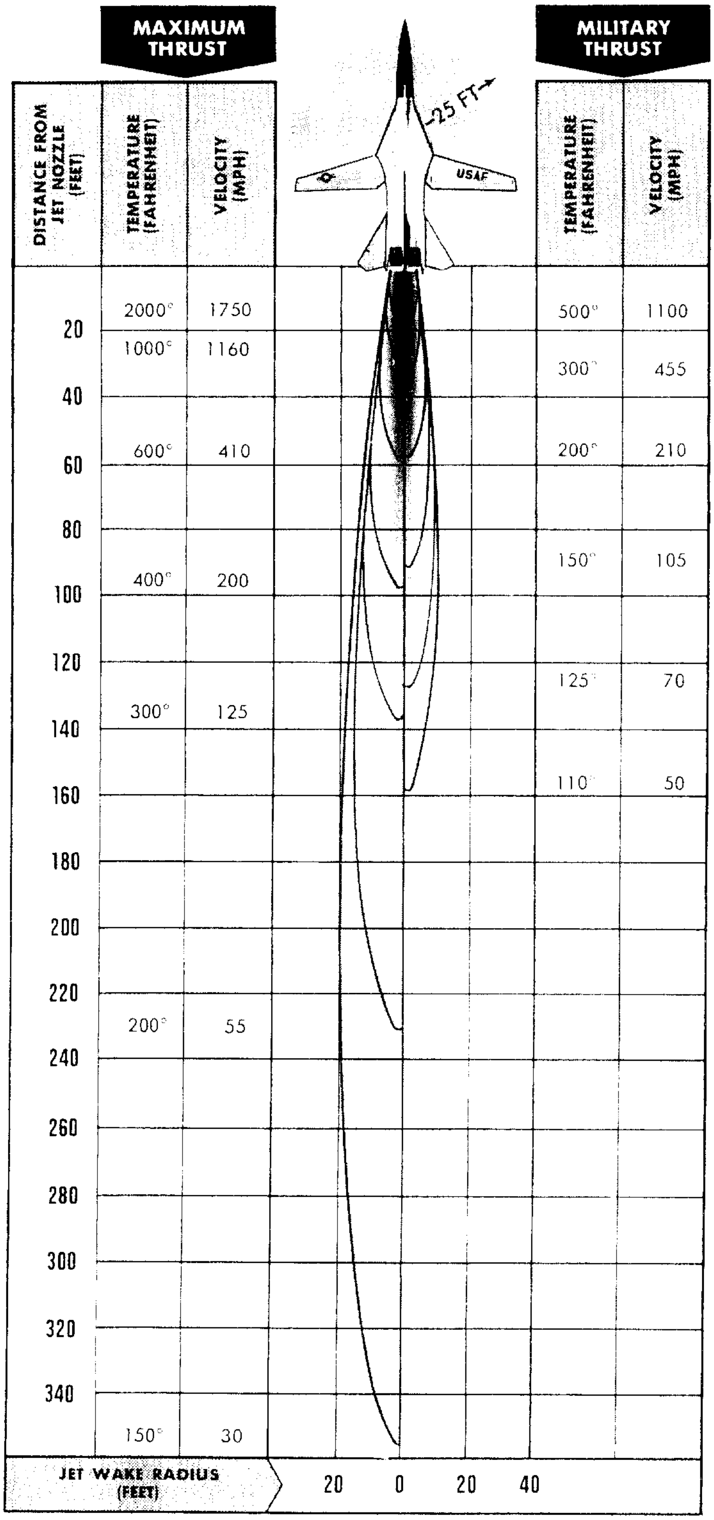
Danger Areas

MAXIMUM THRUST

TEMPERATURE (FAHRENHEIT)	VELOCITY (MPH)
2000°	1750
1000°	1160
600°	410
400°	200
300°	125
200°	55
150°	30

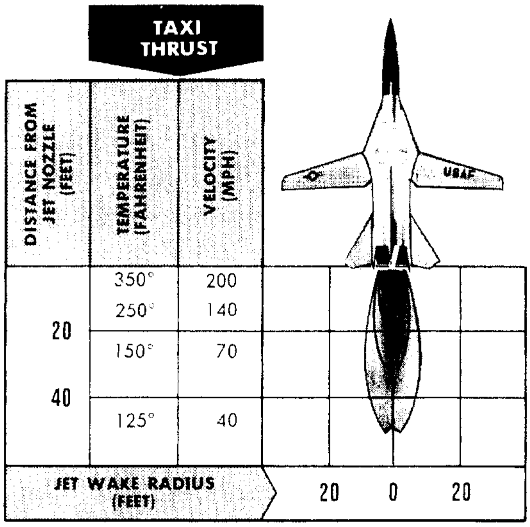
MILITARY THRUST

TEMPERATURE (FAHRENHEIT)	VELOCITY (MPH)
500°	1100
300°	455
200°	210
150°	105
125°	70
110°	50



TAXI THRUST

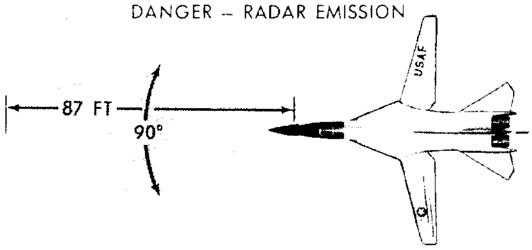
TEMPERATURE (FAHRENHEIT)	VELOCITY (MPH)
350°	200
250°	140
150°	70
125°	40

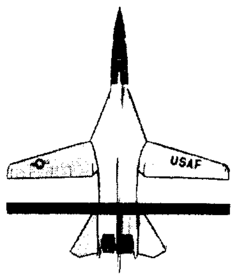


WARNING

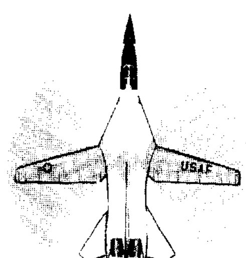
At high thrust settings, the danger area around the intake ducts may extend as far as four feet aft of the duct lip.

With engines operating above idle RPM ear protection should be worn due to high engine noise levels. At idle RPM do not expose unprotected ears to engine noise for periods greater than 5 minutes.





ROTATING PLANES OF ENGINE TURBINES



TIRE AVOIDANCE
Avoid tire area for 45 to 60 minutes after airplane has stopped. If necessary, approach from the front or rear only.

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Figure 2-2.

- Do not initiate a cartridge start with any nacelle door open on the engine being started. To do so could result in possible overheating of adjacent structure and/or ignition of accumulated fuel and oil.

1. Engine ground start switch - PNEU or CARTRIDGE. (As applicable.)
   Place the engine ground start switch to PNEU when starting the engines with external starter air or to CARTRIDGE for a cartridge start.

2. Applicable engine throttle - Start position.

   a. Hydraulic low pressure caution lamps - Out.
      Check that hydraulic low pressure caution lamps go out below 16.5 percent and before 1100 psi.

WARNING

During a cartridge start, if there is no indication of engine rpm rise and no smoke is visible at the starter exhaust port, a misfire has occurred. It will not be possible to start the engine until the starter is reloaded. If there is a delay in obtaining engine rotation, but smoke is visible at the starter exhaust port, a hangfire has occurred. In the event of aborted start due to misfire, hangfire or slow burning cartridge, a mandatory 30 minute waiting period must be observed before opening the breech for cartridge removal.

3. Engine throttle - IDLE.
   On a pneumatic start advance the throttle to IDLE after the engine rpm reaches 16.5 percent. On a cartridge start advance the throttle to IDLE after the first indication of rpm.

4. Engine instruments - Check.

   a. Fuel flow - 1100 pph max.

CAUTION

If fuel flow exceeds 1100 pph during acceleration to idle, shut down the engine to prevent a hot start.

   b. TIT indicator - 705°C max.

   c. Engine oil pressure - Check for indication.

   d. Idle rpm - 58.0 to 64.0 percent.

   e. Idle fuel flow - 800 (±50) pph min.

   f. Idle oil pressure - 30 to 50 psi.

   g. Nozzle position - Open.

5. Generator caution lamp - Out.
   The operating generator caution lamp will go out before the engine reaches idle rpm.

Note

If battery power is being utilized for engine start the emergency generator may come on the line momentarily.

6. Engine overspeed caution lamp - Out.

7. Hydraulic pressure indicators - 2950 to 3250 psi.

8. External air conditioning - Disconnected.

CAUTION

If a period of 3 minutes or more will lapse prior to starting the other engine, the engine ground start switch must be positioned to OFF. This will allow the air conditioning system to provide cooling air for the equipment bays and hydraulic fluid thus preventing equipment damage due to overheating. This procedure will also provide cooling air to the crew compartment.

9. Engine starter air - Disconnected. (If applicable.)
   If a pneumatic start is being made, disconnect the engine start air source.

10. Remaining engine - Started.
    Repeat steps 1 thru 7. If crossbleed is being used for starting second engine, advance the throttle to 85 percent on the operating engine, until second engine reaches 40 percent, then retard throttle to IDLE.

11. Engine ground start switch - OFF.

    a. After a pneumatic start check that the engine ground start switch returns to OFF when the second engine reaches 36 to 39 percent.

    b. After a cartridge start place the ground start switch to OFF when the second engine reaches 40 percent.

12. Generator caution lamp - Out, power flow indicator ISOL.
    The second generator caution lamp will go out and the power flow indicator will go from TIE to ISOL before the engine reaches idle rpm.

13. Oil quantity indicators - 16 quarts.

14. External power switch - OFF, power unit removed. (If applicable.)
    If external power was used for engine start turn the external power switch OFF and remove the ground power unit from the airplane.

15. Emergency generator switch - TEST, then AUTO.
    Place the emergency generator switch to TEST. The emergency generator indicator lamp will light after 3 seconds indicating that the emergency generator is operating within limits. The power flow indicator should display a crosshatch. Check voltages at 115 (±5) volts and frequency at 400 (±8) cps. Place the emergency generator switch to AUTO. Check that indicator lamp goes out and that the power flow indicator displays NORM.

16. Air source selector knob - L.ENG, R.ENG then BOTH.
    Check for air conditioning air flow in L.ENG and R.ENG positions, then place the knob to BOTH.

Note

If battery power was utilized for engine start complete the "power on" checks prior to proceeding to the next checklist.

BEFORE TAXIING.

1. Flight controls - Clear.

2. Flight control check - Airplanes (1)(3)→(7):

   a. Pitch and roll gain selector switches - AUTO.

   b. Auxiliary pitch trim switch - Checked and STICK.
      Move the auxiliary pitch trim switch to NOSE DN, NOSE UP, and then to the STICK position. Control surface travel should correspond to switch position.

   c. Takeoff trim - Set.

   d. Pitch and roll autopilot/damper and yaw damper switches - OFF.

   e. Flight controls - Free.

   f. Pitch and roll autopilot/damper and yaw damper switches - ON.

   g. Damper reset button - Momentarily depress.

   h. Flight control system switch - NORM.

   i. Stability augmentation test switch - SURFACE MOTION.

   j. Flight control master test button - Depress and hold.
      With the master test button depressed check the following:

      (1) Rudder moves to right and then to the left.

      (2) Left horizontal stabilizer trailing edge moves slightly down.

      (3) Right horizontal stabilizer trailing edge moves full down.

      (4) Control system caution lamps do not light.

   k. Stability augmentation test switch - SURFACE MOTION & LIGHTS.

      (1) Rudder initially drives right then returns to neutral.

      (2) Left horizontal stabilizer trailing edge moves slightly down.

      (3) Right horizontal stabilizer trailing edge moves full down.

      (4) Check pitch, roll and yaw damper and channel caution lamps light (6).

   l. Stability augmentation test switch - OFF.
      Check that the control surfaces return to the neutral position.

   m. Flight control master test button - Release.

   n. Damper reset button - Depress and release.
      Check that all channel and damper caution lamps go out.

   o. Flight control system switch - T.O. & LAND.

   p. Trim - Checked and set for takeoff.
      Manually trim nose down, left wing down, and left rudder. Check that control surfaces and stick move to appropriate positions to correspond to trim command. Depress the takeoff trim button and hold until the green takeoff trim lamp lights.

   q. Pitch gain switch and control knob - Set manual, 30 (±10) percent and lock.

3. Flight control check - Airplanes (2) (8)→:

   a. Pitch and roll autopilot/damper and yaw damper switches - OFF.
      Place the pitch and roll autopilot/damper

and yaw damper switches to OFF and check that the pitch, roll and yaw damper caution lamps light.

b. Flight controls - Checked.
Check the flight controls for freedom of movement.

c. Pitch and roll autopilot/damper and yaw damper switches - DAMPER.

d. Damper reset button - Momentarily depressed.
Check that the pitch, roll and yaw damper caution lamps go out.

d-1. Auxiliary pitch trim switch - Checked and STICK:
Move the auxiliary pitch trim switch to NOSE DN, NOSE UP, and then to the STICK position. Control surface travel should correspond to switch position.

d-2. Takeoff trim - Set.

e. Control system switch - NORM.
Check that the control system switch is in the NORM position.

f. Pitch gain selector switch - MANUAL.

g. Pitch gain control knob - Set 100% and lock.
Set for an indicator reading of 100 percent and lock.

h. Roll gain selector switch - AUTO.
Check that the roll gain indicator reads 100 percent.

i. Stability augmentation test switch - SURFACE MOTION.

j. Master test button - Depress and hold.
With the master test button depressed check the following:

(1) Rudder moves to right and then to the left.

(2) Left horizontal stabilizer trailing edge moves slightly down.

(3) Right horizontal stabilizer trailing edge moves full down.

(4) Control system caution lamps do not light.

k. Master test button and stability augmentation test switch - Release.

l. Pitch gain selector switch - AUTO.

m. Stability augmentation test switch - SURFACE MOTION & LIGHTS.

n. Master test button - Depress and hold.
With the master test button depressed check the following:

(1) Rudder initially drives right then returns to neutral.

(2) Left horizontal stabilizer trailing edge moves slightly down.

(3) Right horizontal stabilizer trailing edge moves full down.

(4) Check pitch, roll and yaw damper and channel caution lamps and pitch and roll gain changer caution lamps light (8).

o. Master test button and stability augmentation test switch - Release.

p. Pitch and roll gain selector switches - MANUAL.

q. 2 8 → 11 Pitch and roll gain control knobs - Set pitch 10 percent, roll 50 percent and lock.

r. 12 → Pitch and roll gain control knobs - Set pitch 10%, roll 20% and lock.

s. 2 8 → 11 Pitch gain selector switch - AUTO.

t. 12 → Pitch and roll gain selector switches - AUTO.

u. Control system switch - T.O. & LAND.

v. Damper reset button - Depress momentarily.

w. All caution lamps (8) - Out in 20 seconds.

x. 2 8 → 11 Manual gains indicators - Check pitch gain 30 (±10) percent, roll gain 50 (±5) percent.

y. 12 → Manual gains indicators - Check pitch gain 30 (±10) percent, roll gain 100 (±5) percent.

z. Take off trim button - Depress and hold until the takeoff trim lamp lights.

4. Autopilot - Checked.

a. Pitch and roll autopilot/damper switches - AUTOPILOT.
With yaw damper switch OFF, roll autopilot will not engage. Reference not engaged caution lamp will be lighted. Switch will stay engaged.

b. Yaw damper switch - DAMPER.
Reference not engaged caution lamp will go out.

c. Autopilot release lever - Depressed.
Check that pitch and roll autopilot/damper switches go to the DAMPER position.

d. Pitch and roll autopilot/damper switches - AUTOPILOT.

e. Control stick steering - Checked.
Move control stick and check that reference not engaged caution lamp lights. Lamp will go out when stick is returned to neutral.

f. Altitude hold and constant track switches - Engaged.
Reference not engaged caution lamp lights.

g. Reference engage button - Depressed.
Reference not engaged caution lamp goes out.

h. Move stick, then release.
Reference not engaged caution lamp lights.

i. Reference engage button - Depressed.
Reference not engaged caution lamp goes out.

j. Autopilot release lever - Depressed.
Check that the roll and pitch autopilot/damper switches go to DAMPER and that the altitude hold and constant track switches go to OFF.

5. Flaps - Checked.
Position the flap and slat handle to FLAP DOWN. Check that slats and flaps move to the extended positions. Check flap and slat indicators for proper indication.

6. Ground roll spoilers - Checked.

a. Left and right throttles - Advance approximately 3°.

b. Ground roll spoiler switch - BRAKE.
Check that all spoilers remain down.

c. Left throttle - IDLE.
Check that all spoilers remain down.

d. Right throttle - IDLE.
Check that all spoilers extend.

e. Left throttle - Advance approximately 3°.
Check that all spoilers go down.

f. Right throttle - Advance approximately 3°.
Check that all spoilers remain down.

g. Ground roll spoiler switch - OFF.

7. Heading indicators - Crosschecked. (AC-P)
All heading information should agree within 3 degrees.

8. TACAN - Checked. (AC-P)

9. Altimeters - Set. (AC-P)

10. EPR - Set.

11. Clock - Set. (P)

12. IFF-STBY, code set. (P)

13. Bomb nav mode selector knob - As required. (P)

14. Magnetic heading synchronization indicator - Nulled. (P)
Rotate the magnetic variation counter control knob until a null is obtained on the indicator.

15. Attack radar mode selector knob - GND MAN. (P)

16. Weapon bay doors - Closed. (P)

17. (12)→ Crew module ejection handle safety pins (2) - Removed and stowed.

18. (1)→(11) Ejection seat safety latch handle - ARMED. (AC-P)

19. Ground crew and pilot report - Ready to taxi.
The ground crew will report all ground service disconnected and removed from the airplane, slats and flaps configuration, all ground locks removed and in sight, disconnect ground interphone and remove chocks.

TAXIING.

1. Canopy hatches - As desired. (AC-P)

CAUTION

Do not exceed 60 knots with the hatches open to prevent damage to the hatches.

Taxi speed shall be controlled by throttle adjustments so that dragging the brakes will not be necessary. This will prevent overheating and excessive wear of the brakes.

2. Auxiliary brake handle - In.

3. Brakes - Checked.
Depress brake pedals and check for proper braking action.

Turning Radius and Ground Clearance

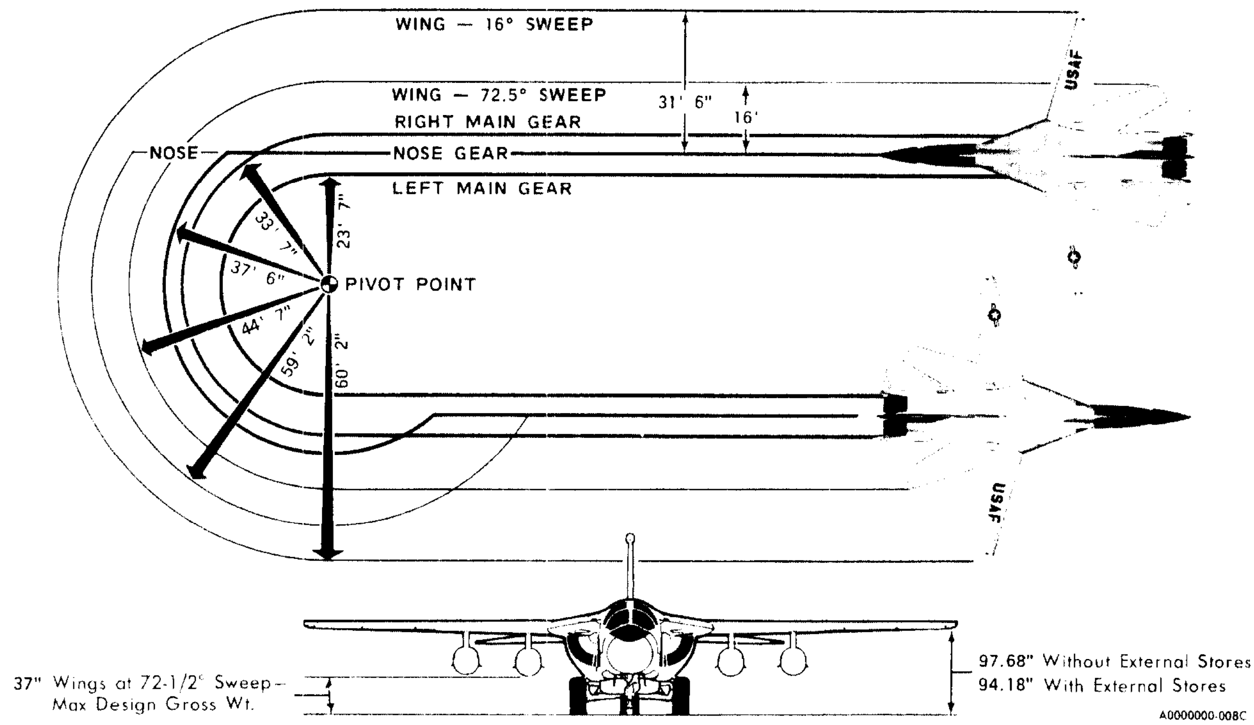

Figure 2-3.

4. Nose wheel steering - Engaged.
   Check that the nose wheel steering indicator lamp is on. Check engagement of nose wheel steering by slight movement of rudder pedals.

5. Flight instruments - Checked. (AC-P)
   Check the flight instruments for proper operation during taxi.

6. Navigation equipment - Set and checked. (AC-P)

7. (3)(5)(8)→ HF radio - Checked and set. (AC-P)

   a. Transmitter selector knob - HF.

   b. HF monitor knob - Pulled.

   c. Mode selector knob - Desired mode.

   d. Desired frequency - Set.
      A mute period will indicate the RT unit is setting to the new frequency. The system should not be keyed during this period. If the frequency was already set when the system was turned on, rotate the 1-kc knob one digit aft frequency and then back to the desired frequency. This will allow the RT unit to properly tune to the desired frequency.

   e. RF sensitivity knob - Adjusted.
      Adjust the RF sensitivity knob to receive signals just above the noise level of the receiver, then adjust the interphone monitor knob for a comfortable listening level. Proper balance is indicated when background noise is just audible and a weak signal is raised to comfortable level.

   f. Microphone switch - TRANS.
      After a frequency change, a 1000 cycle tone will be heard when the "mike" switch is first placed to TRANS. This indicates that the RT unit and antenna coupler are tuning. When the tone ceases, the tuning cycle is complete and a side tone will be heard when transmitting. Lack of a side tone indicates the pressure in the amplifier power supply or antenna coupler is below 15.7 PSIA.

   g. Transmitter selector knob - UHF.

8. TFR control panel - Set as required. (if installed)

9. Hydraulic pressure - Checked.
   Check for 2950 to 3250 psi indication.

10. External fuel tanks - Checked (if installed).
    Momentarily position the fuel transfer knob to OUTBD, CENTER and INBD to purge the tanks transfer valves. Return the fuel transfer knob to AUTO.

Note

If mission requirements are such that all external fuel is required, the fuel transfer knob should be left in the OUTBD, CENTER, and INBD positions long enough to insure that fuel is being transferred to the fuselage tanks.

BEFORE TAKEOFF.

1. Canopy hatches - Closed and latched. (AC-P)

2. Canopy latch handle lock tab - Flush. (AC-P)
   Snap the spring-loaded latch handle lock tab into the locked (flush) position and pull on the latch handle to check that it is locked.

3. Anti-collision light - ON.

4. Wings, flaps, and slats - Set for takeoff.
   Check the surface position indicator for selected wing, flap, and slat settings.

5. Speed brake switch - IN.

6. Control system switch - T.O. & LAND.

7. Takeoff trim - Checked.
   Depress the takeoff trim button and hold until the takeoff trim indicator lamp lights. Release the button and the lamp should go out.

8. Pitch and roll manual gain switches and control knobs - Check per placard. (AC-P)

9. Pitch and roll autopilot/damper and yaw damper switches - DAMPER.

10. Warning and caution lamps - Checked. (AC-P)
    Check that all warning lamps are out and that caution lamps are compatible with mission.

11. (3)(5)→ Pitot heater and engine/inlet anti-icing switches - Climatic.

12. Fuel quantity and fuel distribution - Checked.

13. Engine feed selector knob - As desired.

14. IFF master control knob - NORM. (P)

15. (3)(5)→ Attack radar antenna - Uncage. (P)
    Depress the antenna cage pushbutton indicator lamp to uncage the antenna.

16. Oxygen - Checked. (AC-P)

17. (12)→ Foreign object damage prevention door switch - RETRACT.

18. Takeoff data - Checked. (AC-P)

TAKEOFF.

1. Nose wheel steering button - Disengaged, lamp out.

2. Engines - Set for takeoff and checked.

3. Brakes - Release.

4. Go-no-go speed - Checked.

AFTER TAKEOFF.

1. Landing gear handle - UP.
   When the airplane is definitely airborne, retract the landing gear. Check that the landing gear position indicator lights and the warning light in the landing gear handle go out. The landing gear and landing gear doors should be up and locked before reaching 295 KIAS.

WARNING

Any time it is necessary to depress the landing gear handle lock release button to move the handle to the UP position, the crew member should immediately suspect a malfunction of the landing gear ground safety switch. A failure of this switch, which left it in the closed position, would render ineffective the AUTO position of the fuel tank pressurization switch and cause all spoilers to remain armed even with the landing gear retracted. If a malfunction of the landing gear safety switch is suspected, the fuel tank pressurization switch should be placed to PRESSURIZE and the spoiler switch to OFF.

Note

The fuel tank pressurization caution lamp will momentarily light when the landing gear handle is moved to the up position.

Takeoff (Typical)

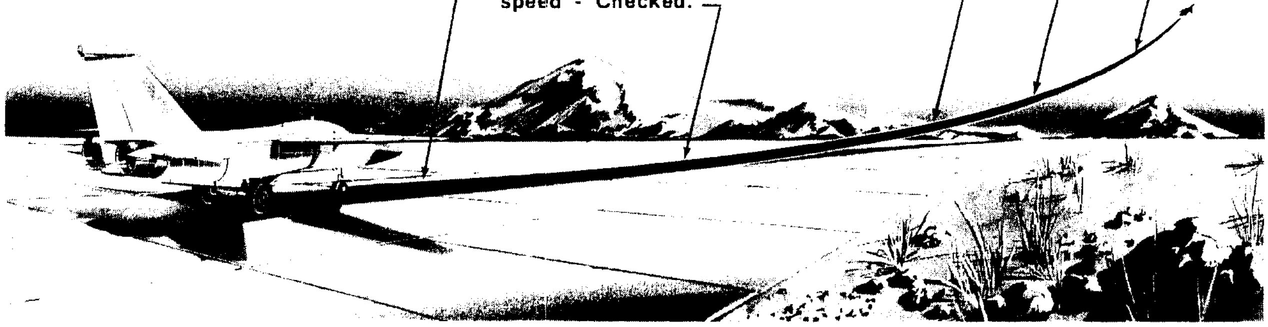

Figure 2-4.

2. Flap and slat handle - UP.
   Raise the flaps prior to reaching flap limit speed (Airplanes (1)→(11) 190 KIAS; Airplanes (12)→ 225 KIAS).
3. Wing sweep handle - As required.
4. Utility hydraulic system isolation switch - ISOLATE.
5. Control system switch - NORM.

Note

The rudder authority caution lamp will light momentarily when the control system switch is placed to NORM.

6. (1)→(11) Pitch and roll gain selector switches and control knobs - Set per placard. (AC)
7. Engine instruments - Checked.
8. Fuel quantity indicators - Checked.
   Check the fuel quantity indicators for normal fuel usage.
9. Fuel transfer knob - As required.
   If wing or external fuel is carried, transfer to the main tanks should be initiated.
10. Translating cowl switches - RETRACT (.5 to .6 mach).
11. Translating cowl indicators - CLOSED.
12. Altimeters - Reset. (AC-P)
13. IFF/SIF - Checked. (P)

SS-8

CLIMB.

The recommended climb speed, as shown in Appendix I, should be followed.

CRUISE.

Refer to Appendix I for cruise operating data. Refer to Section I for fuel system operation.

CAUTION

Brake pedal actuation in flight should be avoided to prevent depletion of the brake accumulator.

For airspeeds below mach 1.2, leave the engine inlet spike control switches in the OFF position. At airspeeds of mach 1.2 and above, position these switches to AUTO.

WARNING

If an engine inlet spike control switch is un-intentionally moved to OVERRIDE, leave it in that position. If a spike control switch is moved to OVERRIDE momentarily, then back to AUTO, return it to the OVERRIDE position immediately to prevent possible loss of the utility hydraulic system.

At airspeeds below 400 KIAS, use AUTO gains. For airspeeds of 400 to 640 KIAS, set manual gains for pitch 30 percent and roll 50 percent. At airspeeds above 640 KIAS, set manual gains for pitch 10 percent and roll 20 percent.

AIR REFUELING.

NORMAL OPERATION.

1. Engine feed selector knob - AUTO or BOTH.
2. Air refueling switch - OPEN.
3. Air refueling receptacle light control knob - Bright.
4. Nose wheel steering/air refueling indicator lamp - Checked.
   Check that the nose wheel steering/air refueling indicator lamp lights indicating that the refueling receptacle is extended.
5. Tanker aircraft boom - Engaged.
   Check that the nose wheel steering/air refueling indicator lamp goes out indicating that the boom is latched in place.
6. Fuel quantity indicators - Checked.
   Check the fuel quantity indicators to insure that fuel is being transferred.
7. Refueling - Completed.
   Check that the nose wheel steering/air refueling indicator lamp lights when all tanks are full, indicating that the boom has disconnected.
8. Air refueling switch - CLOSE.
   Check that the nose wheel steering/air refueling indicator lamp goes out.
9. Engine feed selector knob - As required.

ALTERNATE OPERATION.

If the automatic latching mechanism fails to latch the boom in place, as indicated by the nose wheel steering/air refueling indicator lamp remaining on after boom engagement, proceed as follows:

1. Air refueling switch - EBL.
2. Tanker aircraft boom - Engaged.
   Check that the nose wheel steering/air refueling indicator lamp goes out indicating that the boom is latched in place.
3. Fuel quantity indicators - Checked.
   Check the fuel quantity indicators to insure that fuel is being transferred.
4. Nose wheel steering/air refueling button - Depressed (at completion of refueling).
   When the tanks are full, depress the nose wheel steering/air refueling button to disengage the refueling boom. Check that the nose wheel steering/air refueling indicator lamp lights indicating boom disconnection.
5. Air refueling switch - CLOSE.
   Check that the nose wheel steering/air refueling indicator lamp goes out.
6. Engine feed selector knob - As required.

FLIGHT CHARACTERISTICS.

Refer to Section VI for flight characteristics.

DESCENT.

1. 3 5 → Pitot heater switches and engine/inlet anti-icing switches - Climatic.
2. Cabin air distribution control lever - DEFOG. (P)
3. Fuel quantity and fuel panel - Checked.
4. Hydraulic pressure - Checked.
   Check for 2950 to 3250 psi indication.
5. Wing sweep handle - As required.
6. IFF - As required. (P)
7. Altimeter - Set. (AC-P)

BEFORE LANDING.

1. Translating cowl switches - OPEN.
   Position both cowl switches to OPEN and check that position indicators show OPEN.

SS-8

WARNING

To prevent engine compressor stall and subsequent loss of power during the critical phases of landing, or in event of a go-around, the translating cowls must be OPEN.

6. Control system switch - T.O. & LAND.
   When below 300 KIAS, place the control system switch to T.O. & LAND. The rudder authority caution lamp will light momentarily when this switch position is selected.

7. Fuel quantity and feed - Checked.

8. Landing data - Checked. (AC-P)

9. Speed brake switch - As required.

10. Landing gear handle - DN.
    Extend the landing gear after airspeed is below 295 KIAS. Check that warning light in landing gear handle is out and landing gear position indicators lights are lighted.

CAUTION

At speeds above 250 KIAS the nose gear may not lock in the down position. Should this occur, decelerate to below 250 KIAS and check for nose gear down and locked indication. If a down and locked indication is not obtained, recycle the landing gear up and down.

11. Translating cowl indicators - OPEN.

12. Utility hydraulic system isolation switch - ON.

13. Flap and slat handle - Set 15 degrees.
    When below 290 KIAS (1)→(11) or 297 KIAS (12)→, extend flaps to 15 degrees.

14. Flap and slat handle - DOWN.
    When below 190 KIAS (1)→(11) or 225 KIAS (12)→, position handle to DOWN and check flap and slat position indicator to assure that surfaces have moved to the selected position.

LANDING.

Note

See figure 2-5 for normal landing pattern and airspeeds.

1. Throttles - As required.

2. Touchdown - As computed.

3. Nose wheel - Lower to runway.

4. Ground roll spoiler switch - BRAKE.

5. Brakes - As required.

6. Nose wheel steering - As required.

LANDING ON SLIPPERY RUNWAYS AND/OR MINIMUM RUN LANDINGS.

The technique for a wet runway landing is essentially the same as for a normal landing. Particular attention should be paid to maintaining final approach speed and touching down as close to the end of the runway as safety permits. The ground roll spoiler switch should be ON prior to landing. As with the normal landing technique, power should be reduced to IDLE immediately upon touchdown. If maximum deceleration is desired, maximum antiskid braking should be initiated immediately upon touchdown and held throughout the ground roll. For this purpose, any amount of excess pedal displacement is satisfactory, up to and including full deflection. Full aft stick should be applied at approximately 90 KIAS and held throughout the remainder of the ground roll. Full aft stick at this speed should not lift the nose gear from the runway, under normal landing gross weight and cg configurations. Full aft stick provides additional aerodynamic drag and transfers aircraft weight to the main gear to provide maximum wheel braking potential. Be prepared to lower the arresting hook to engage the runway barrier if the airplane cannot be stopped prior to reaching the end of the runway.

TOUCH AND GO LANDING.

Prior to accomplishing a touch and go landing, perform the normal before landing cockpit check and position the ground roll spoiler switch to OFF. After touchdown, smoothly advance the throttles to MIL power and check engine instruments for normal indication. When airborne, proceed with the normal after takeoff-climb checklist if required. Return the ground roll spoiler switch to BRAKE prior to final landing.

GO-AROUND.

The decision to go around should be made as early as possible. When the decision to go around is made, smoothly advance the throttles as required. Continue the approach because a touchdown may be necessary. As the airplane accelerates, rotate the nose to a climbing attitude. Retract the landing gear and flaps, below 225 KIAS if required. Fly clear of the runway as soon as practicable. (See figure 2-6.)

Landing Pattern (Typical)

BASED ON LANDING GROSS
WEIGHT OF 50,000 POUNDS.

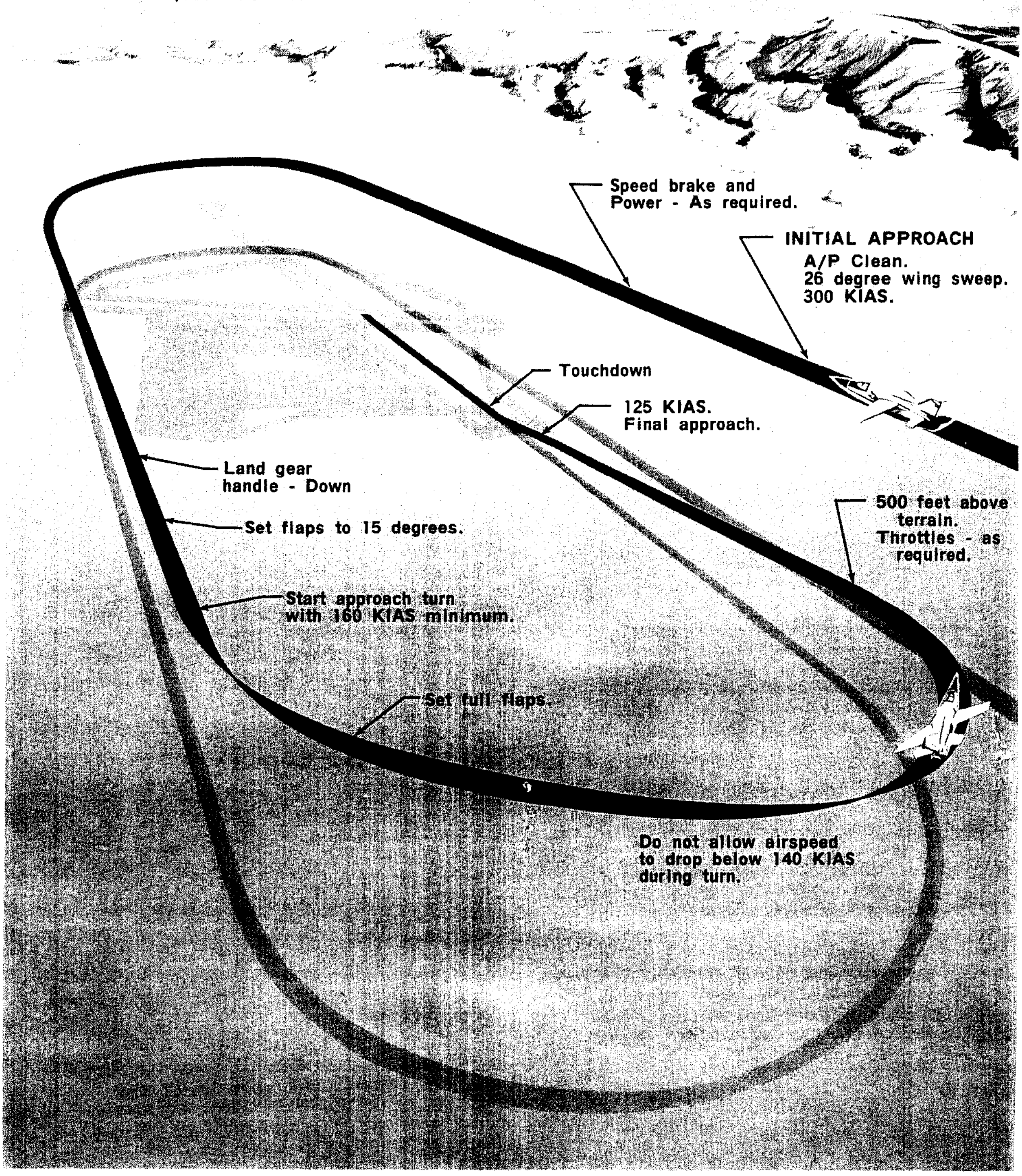

Figure 2-5.

Go-Around (Typical)

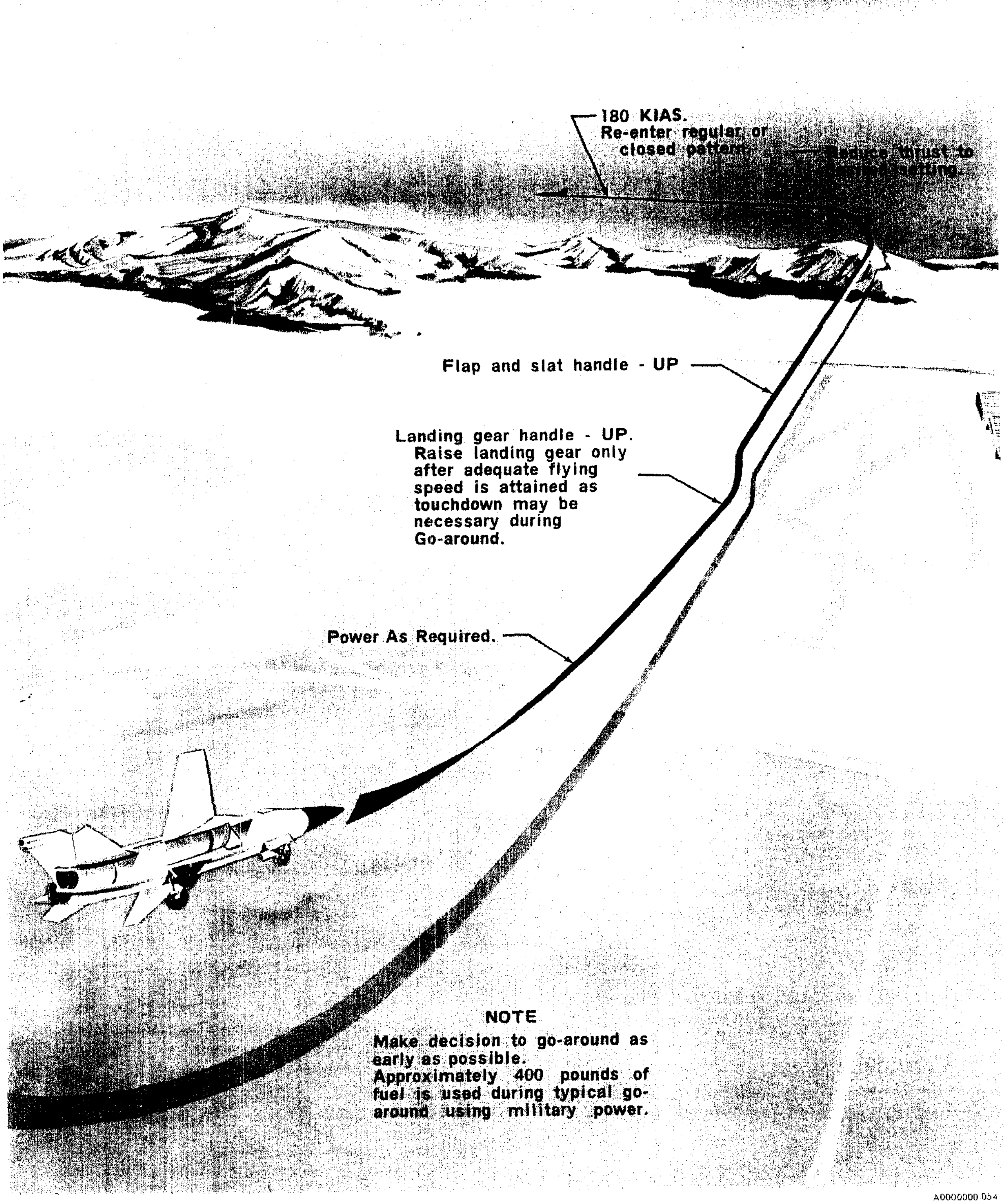

Figure 2-6.

AFTER LANDING.

1. (12)→ Foreign object damage prevention door switch - EXTEND.
2. Ground roll spoiler switch - OFF.
3. Flap and slat handle - As required (normally extended).
4. IFF master control knob - OFF. (P)
5. (3)(5)→ Pitot heater and engine/inlet anti-icing switches - OFF.
6. Air refuel switch - OPEN (if required).
   If air refueling operations were accomplished during the flight, position the air refuel switch to OPEN.

ENGINE SHUTDOWN.

CAUTION

To prevent possible damage to the brakes from overheating, do not pull the auxiliary brake handle.

1. Wheels - Chocked.
2. Wing sweep handle - As required.
3. Flap and slat handle - As required (normally extended).
4. Bomb nav mode selector knob - OFF. (P)
5. (3)(5)→ Attack radar function selector knob - OFF. (P)
6. Applicable throttle - OFF.
   Place throttle of the first engine started to OFF.
7. Hydraulic pressure - Checked.
   Check for 2950 to 3250 psi indication.
8. Remaining throttle - OFF.

CAUTION

To prevent possible engine damage due to overtemperature, do not attempt to restart the engine for at least five minutes after shutdown.

9. Emergency generator - Checked.
   Check that the emergency generator indicator lamp lights when the generators disconnect from the ac buses.
10. (12)→ Foreign object damage prevention door switch - RETRACT.

BEFORE LEAVING AIRPLANE.

1. All switches and controls - Off, normal or safe. (AC-P)
   Starting on the left side of the crew compartment, position all switches and controls off, normal or safe.
2. (1)→(11) Canopy jettison handle safety pin (1) - Installed. (P)
3. (1)→(11) Ejection seat safety latch handle - SAFE. (AC-P)
   Move the ejection seat safety latch handle from the armed position to the safe position on both seats.
4. (12)→ Crew module ejection handle safety pins (2) - Installed.
5. (12)→ Crew module severance and flotation, recovery parachute release, auxiliary flotation and canopy internal emergency release handle safety pins (3) - Installed.

STRANGE FIELD.

If it is necessary to land at an airfield where normal ground support equipment or personnel is not available, the air crew will be responsible for performing or closely supervising the required airplane servicing. There are several items which must be performed after engine shutdown, and additional items of servicing and inspection are required prior to takeoff. It is recommended that the air crew become familiar with the servicing procedures for all items listed on the Servicing Diagram, Section I. Engine starting is normally accomplished with gas turbine generator set A/M32A-60. The unit supplies engine starting air and ac power for the airplane electrical systems. Alternate engine starting equipment consists of an MA-1A gas turbine as a source of air pressure with MD-3A, or MD-4, or the airplane battery as a source of electrical power for ignition. Electrical power requirement for ground refueling, if power is deemed necessary consists of an A/M32A-60, or either MD-3A or MD-4 as a substitute. The following check list supplements the normal operating procedures and includes items that would normally be accomplished by the ground crew.

AFTER LANDING.

1. Engine oil level - Checked.
   Check oil level indication on dipstick and determine quantity required to bring oil level to the 20 quart level or FULL MARK. Service with oil MIL-L-7808.

Note

The engine oil system must be checked and serviced within 15 minutes after shutdown in order to determine accurate consumption as variable amounts of oil can leak from tank into the gearbox over longer periods.

2. Hydraulic reservoirs - Checked.
   Check the utility and primary hydraulic reservoirs for specified accumulator preload and fluid level in accordance with placard.

3. Refueling - Accomplish. (as required)

   a. Single Point Refueling.

Note

External electrical power may be connected during refueling if desired for monitoring instruments: otherwise, external power is not necessary.

(1) Airplane and refueling equipment - Grounded.
    Insure that the airplane and all refueling equipment are statically grounded.

(2) Nose gear chocks - Removed.
    Remove all work stands and equipment under the aircraft which might cause damage when the landing gear shock struts compress due to increased fuel load.

(3) Precheck selector valves - REFUEL.

(4) Position lights/stores refuel battery power switch - STORES REFUEL.
    If external tanks are installed, place the position lights/stores refuel battery power switch to STORES REFUEL.

(5) Fueling hose ground cable - Connected.
    Connect the grounding cable from the fueling hose to the airplane.

(6) Ground refueling receptacle cap - Removed.

(7) Fuel nozzle - Connected to refueling receptacle.

(8) Start fuel servicing unit and open fuel nozzle.

(9) Precheck selector valves - PRI or CK. (as applicable)
    Within a few seconds after fuel flow is indicated, position all precheck selector valves to PRI or CK as applicable. The fuel flow should drop to less than 10 gpm indicating that all primary valves have closed.

CAUTION

Do not allow fuel flow to the aft tank or wing tanks for more than a few seconds when the forward tank quantity is below 7500 pounds. To do so may cause a longitudinal unbalance and cause the airplane to tip up.

Note

If fuel flow drops to 5 gpm or less, proceed to step 10. If fuel flow does not drop, determine which refuel valve has malfunctioned as follows. Select the aft tank valve to SEC and observe the flowmeter for 30 seconds, then select PRI. If flow did not drop below 5 gpm when SEC position was selected, repeat the test for the forward tank. If flow is not stopped when SEC position is selected for the forward tank, repeat the test for each wing by changing positions for the wing precheck selector valve located on the lower surface of each wing. The defective valve will be indicated by a drop of flow.

(10) Fuselage tank precheck selector valves - REFUEL then SEC.
     Individually rotate the fuselage tank precheck selector valves to REFUEL and then to SEC while observing the flowmeter. Flow should rise at least 100 gpm while in the REFUEL position, indicating that the selected refuel valve has opened. The valve should then close when the SEC position is selected.

(11) Precheck selector valves - REFUEL.
     Continue refueling operations.

(12) Tank pressure gage - Monitor.
     If pressure exceeds 3 psi, discontinue refueling operation and determine the cause. The tanks should be depressurized and air should flow from the vent during fueling.

Note

Fuel tanks are full and valves are closed when the flowmeter on the fuel truck falls to zero.

(13) Fuel nozzle - Closed.
     At completion of refueling, close the fuel nozzle and stop the refueling truck pump.

(14) Fuel nozzle and grounding cable - Disconnected.

(15) Refueling receptacle cap - Installed.

(16) Single point refueling control access doors - Closed and latched.

(17) Position lights/stores refuel battery power switch - NORM (if external tanks were fueled).

Note

Failure to return position lights /stores refuel battery power switch to NORM will produce drain on the battery when external electrical power is not connected.

b. Gravity Refueling.

(1) Connect external power.

Note

External power is not required; however, a full reservoir tank will not be assured until after engine start unless engine feed is selected and fuel pumps operated for approximately 2 minutes with the forward tank at 4000 pounds or more.

(2) Airplane and refueling equipment - Grounded.
Insure that the airplane and all refueling equipment are statically grounded.

(3) Nose gear chocks - Removed.
Remove all work stands and equipment under the aircraft which might cause damage when the landing gear shock struts compress due to the increased fuel load.

(4) Fuel Tank Pressurization.
If tanks are pressurized, place the tank pressurization switch to AUTO to relieve pressure.

CAUTION

The vent tank is within the vertical stabilizer and extends near the top; therefore, if fuel has entered the vent tank a head pressure will exist. Extreme care must be exercised when removing the gravity refuel caps from any fuel tank. Loosen the cap slightly watching for signs of fuel flow prior to removing the cap.

(5) If the forward tank quantity is 4000 pounds or greater, place forward tank selection switch to ENG FEED and allow fuel pumps to operate for approximately 2 minutes to assure a full reservoir tank.

(6) Bay F-1 and F-2 - Refueled.

Note

Remove filler cap from bay F-1 and then bay F-2. If fuel seeps out as bay F-2 filler cap is loosened, do not continue removing cap as bay F-2 is full. Fill bay F-1 only. Otherwise, fill bay F-2 and then bay F-1.

If forward tank initially had less then 4000 pounds, perform step 4 after the forward tank has been filled above 4000 pounds, and then continue filling.

(7) Gravity refuel the remaining tanks in the following order:

(a) Bay A-1

(b) Bay A-2

(c) Wing Tanks

Note

If a partial fuel load is required, the forward tank should contain 8200 pounds more fuel than the aft tank. Any fuel added to the wings shall be distributed equally between the wing tanks.

(8) Fuel Filler Caps - Secure.

POSTFLIGHT.

1. Exterior inspection - Complete.
Follow route shown in figure 2-1. Make necessary entries in the Form 781.

Note

While performing the strange field postflight, and preflight, exterior inspections check for the following:

Cuts, scratches, loose rivets and fuel leaks.

All drain plugs for leakage.

That all access doors and panels are secure.

Reservoirs and accumulators for proper servicing. Refer to figure 2-7.

Ground area around airplane for cleanliness.

Airplane is now ready for relaunch; however, if flight is terminated or takeoff substantially delayed, accomplish the following:

a. Canopies - Closed.

b. Ground locks - Installed (if available).

c. Pitot cover - Installed (if available).

DELAYED TAKEOFF.

If takeoff has been delayed for an extended time (over 12 hours), a normal exterior preflight should be accomplished following route shown in figure 2-1. The following systems should be checked and serviced as required. Upon completion, follow normal procedures Section II. Complete required Form 781 entries prior to takeoff.

1. Liquid oxygen - Checked.
   Service with liquid oxygen MIL-O-27210, Grade A, Type II.

2. Pneumatic pressure - Checked.
   The following accumulators or reservoir pneumatic pressures should be checked for required pressure range specified for the ambient temperature. Service with Air: MIL-P-5518 or Nitrogen: FS BB-N-411, Type I, Grade B.

Pneumatic Servicing Requirements Table

System or Component	Pressure (PSIG) (At 70°F, 21°C)
Landing gear pneumatic reservoir	3000
Alternate trapeze system (2)	3000
FOD door system	3000
Alternate spike (air inlet) control (2)	3000
Primary/utility hydraulic accumulator	500
Primary and utility damper servos	1400
Wheel brake accumulators (2)	800
Horizontal stabilizer accumulators, utility (2)	1400
Horizontal stabilizer accumulators, primary (2)	1400
Overwing fairing	1800
Canopy Counterpoise (2)	860 (Hatch Open) 1063 (Hatch Closed)

Figure 2-7.

3. Constant speed drive - Checked.
   Check outboard sight gage on both left and right drive units. If oil is in the green band, no servicing is required. If servicing is required, proceed as follows. Service with oil MIL-L-7808.

   a. Refill very slowly until oil level reaches the bottom of the green band. Shut off oil supply to avoid overfilling, and allow oil level to equalize. As much as 5 minutes may be required.

   b. Repeat preceding step until oil level is stabilized in the green band.

4. Utility and primary hydraulic reservoirs - Checked.
   Check the utility and primary hydraulic reservoirs for specified accumulator preload and fluid level in accordance with instruction placard. If servicing is required, proceed as follows. Service with oil: MIL-H-5606.

   a. Check the hydraulic reservoir pneumatic pressurization system for proper service.

   b. Position aircraft hydraulic hand pump selector valve to BRAKE and pump brake accumulators to 3100 psi pressure prior to servicing the utility reservoir.

   c. Fill reservoir slowly until quantity gage indicates proper fluid level as shown on reservoir service placard.

   d. Open reservoir air bleed valve (lower aft end of reservoir) sufficiently to bleed trapped air from the reservoir fluid chamber.

   e. Check the reservoir quantity indicator for proper fluid level.

   f. Repeat steps 4 through 5 until the reservoir is fully serviced and free of air.

5. Landing gear shock struts - Checked.
   Check nose landing gear shock strut and main landing gear shock struts inflated in accordance with strut instruction placard. Service with Air: MIL-P-5518 or Nitrogen: FS BB-N-411, Type I, Grade B.

6. Tires - Checked.

   a. Main landing gear tires - 140 ±10 psi.

   b. Nose landing gear tire - 170 ±10 psi.

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SECTION III

EMERGENCY PROCEDURES

TABLE OF CONTENTS.

This section contains procedures to be followed to correct an emergency condition. These procedures will insure maximum safety for the crew and/or aircraft until a safe landing or other appropriate action is accomplished. Multiple emergencies, adverse weather, and other peculiar conditions may require modification of these procedures. The CRITICAL items (ALL CAPITAL BOLD FACE LETTERS) contained in the various emergency procedures are those steps which must be performed immediately without reference to written checklists. These critical steps shall be committed to memory. All other steps, wherein there is time available to consult a checklist, are considered NON-CRITICAL. The nature and severity of the encountered emergency will dictate the necessity for complying with all or part of the steps in a particular procedure. It is essential, therefore, that aircrews determine the correct course of action by use of sound judgment. As soon as possible, the aircraft commander should notify the pilot and flight leader of any existing emergency and of the intended action. When an emergency occurs, three basic rules are established which apply to airborne emergencies. They should be thoroughly understood by all aircrews.

1. Maintain aircraft control.
2. Analyze the situation and take proper action.
3. Land as soon as practicable.

Note

The canopy hatches should remain closed during all emergencies that could result in a crash or fire such as crash landings, aborted takeoffs, and barrier engagements. The protection the canopies afford the crew during these emergencies far outweighs the isolated risk of entrapment due to a canopy malfunction or overturn.

GROUND OPERATION EMERGENCIES

ENGINE FIRE DURING START.

1. **THROTTLE(S) — OFF.**
2. Engine start switch - Activate for 20 seconds (if external air is available).

If Fire Persists:

3. Fire pull handle or push button - Actuate. Actuate the fire pull handle or pushbutton for affected engine.
4. Agent discharge control - Actuate.

WARNING

The fire extinguishing agent is available for one engine only. Selection of the engine to which the agent is to be directed is made by pulling the appropriate fire pull handle or depressing the appropriate fire push-button.

5. All switches - OFF. If time and conditions permit, turn all switches off.
6. Abandon the airplane.

WHEEL BRAKE SYSTEM EMERGENCY OPERATION.

In the event of utility hydraulic system failure, normal braking technique should be used until no longer effective. If airplane is not stopped, pull the auxiliary brake handle.

CAUTION

With the auxiliary brake handle pulled, the brakes are locked.

EMERGENCY ENTRANCE.

Emergency entrances are shown in Figure 3-6.

WARNING

With the hatches closed and locked from the inside, the hatches cannot be opened with the normal exterior canopy latch handle.

ABANDONING THE AIRPLANE ON THE GROUND.

In an emergency requiring ground abandonment, the primary concern should be to leave the immediate area of the airplane as soon as possible. Salvaging emergency and survival equipment should not be considered. To abandon the airplane, disconnect personal leads and harness, open canopy hatches by the normal method if possible; if not possible, make sure the canopy latch handle is returned to the closed and latched position, then pull the canopy internal emergency release handle, or on airplanes 1 → 11 pull the canopy jettison handle.

WARNING

- On airplanes 12 → do not actuate the canopy internal emergency release handles unless both canopy hatches are closed and latched. To do so could result in fatal injury from debris flying from the canopy sill.
- On airplanes 1 → 11 the oxygen mask hose must be disconnected from the oxygen regulator and the regulator must be detached from the torso harness before the crewmember can abandon the airplane.
- The restraint harness should remain fastened and the inertia reel locked until the airplane is stopped.

TAKEOFF EMERGENCIES

During all takeoff emergencies when takeoff is aborted, the canopy will not be jettisoned.

ABORT.

1. **THROTTLE(S)—IDLE (OFF FOR FIRE).**
2. **GROUND ROLL SPOILER SWITCH—BRAKE.**
3. **EXTERNAL LOAD—JETTISON (IF NECESSARY).**
4. **ARRESTING HOOK—EXTEND (IF REQUIRED).**
5. Shoulder harness - Locked.

ENGINE FAILURE DURING TAKEOFF.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort" procedures, this section.

If Takeoff Is Continued:

1. **THROTTLE—MAXIMUM (NORMAL OPERATING ENGINE).**
2. **EXTERNAL LOAD—JETTISON (IF NECESSARY).**
3. Landing gear handle - UP (when airborne).
4. Flap and slat handle - As required.
5. Throttle of failed engine - OFF.
6. Attempt airstart if failure was non-mechanical and engine appears normal.
7. Fuel - Dump. (As required).
8. Land as soon as practicable.

ENGINE FIRE DURING TAKEOFF.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort" procedure, this section.

If Takeoff Is Continued:

1. **THROTTLE—MAXIMUM (NORMAL OPERATING ENGINE).**
2. **THROTTLE—OFF (ENGINE INDICATING FIRE).**
3. **EXTERNAL LOAD—JETTISON (IF NECESSARY).**
4. **FIRE PULL HANDLE OR PUSHBUTTON—ACTUATE.**
5. **AGENT DISCHARGE CONTROL—ACTUATE.**
6. **IF FIRE CONTINUES—EJECT.**
7. Landing gear handle - UP (When airborne).
8. Flap and slat handle - As required.
9. If fire goes out - Land as soon as practicable.

AFTERBUNER FAILURE DURING TAKEOFF.

If an afterburner fails during takeoff, the resulting loss of power is significant. Takeoff need not be aborted if takeoff speed and distance are compatible with computed takeoff minimums.

TIRE FAILURE DURING TAKEOFF.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort" procedures, this section.
2. **ANTI-SKID—OFF.**

If Takeoff Is Continued:

1. **EXTERNAL LOAD—JETTISON (IF NECESSARY).**
2. **DO NOT RETRACT GEAR.**

Note

If it can be determined that the blown tire has been torn from the wheel, the gear may be retracted, depending upon mission urgency.

3. Instruments - Check.
   Monitor hydraulic pressures and fuel quantities.
4. Dump fuel as necessary and land as soon as practicable.

INFLIGHT EMERGENCIES

CAUTION LAMP ANALYSIS.

See Figure 3-5 for analysis and suggested corrective action to be taken whenever a caution lamp is lighted.

EMERGENCY WING SWEEP OPERATION.

The necessity for emergency wing sweep operation may arise from either of two conditions; one engine inoperative, or one hydraulic system inoperative. In either condition, normal wing sweep commands can result in a severe drop in available hydraulic pressure thus degrading flight control response. While sweeping the wings under the above conditions, maintain as high an rpm on the engine(s) as practicable, and maintain 1 g straight and level flight. When operating with one engine out, if the operating engine rpm is allowed to drop below 90 percent and the windmilling engine is below 40 percent rpm, it will be necessary to sweep the wings in increments of 1/4 to 1/2 inch, of wing sweep handle movement.

If Supersonic:

1. Wing sweep handle - 45 degrees.
   If an engine failure or hydraulic system failure occurs at supersonic speed, sweep the wings to 45 degrees as soon as practicable, while still supersonic. Refer to "Primary or Utility Hydraulic System Failure", this section.

Subsonic:

1. Wing sweep handle - 26 degrees.
   When subsonic, sweep the wings to 26 degrees while maintaining straight and level 1 g flight.

SINGLE ENGINE FAILURE DURING FLIGHT.

NONMECHANICAL FAILURE.

1. Attempt airstart.
   If the engine failure is attributed to something other than a mechanical failure, an airstart may be attempted. Follow "Airstart" procedures, this section.

MECHANICAL FAILURE.

1. Throttle of affected engine - OFF.
2. Land as soon as practicable.

AIRSTART.

See figure 3-1 for Airstart Envelope.

Note

The engine is equipped with auto ignition and will normally restart automatically. If the engine has flamed out because of other problems such as fuel starvation, the following procedure is recommended for airstarting.

1. **AIRSTART IGNITION BUTTON—DEPRESS.**
   Depress the airstart ignition button immediately upon indication of a flameout.
2. **FUEL PANEL—CHECKED.**
   Check fuel feed selection and fuel quantities to assure that fuel is available to the engine.

If restart is not accomplished within 50 seconds:

3. Throttle of affected engine - OFF.
4. Airstart ignition button - Depress momentarily.
5. Throttle of affected engine - IDLE.
   Check for relight.

If airstart has not been accomplished by the time engine rpm is below 16 percent:

6. Throttle of affected engine - OFF.
7. Engine ground start switch - PNEU.
8. Throttle of affected engine - START.
9. Throttle of affected engine (at 16.5 percent) - IDLE.
   Check for relight.

DOUBLE ENGINE FAILURE DURING FLIGHT.

WARNING

Should a double engine failure occur and an airstart of at least one engine cannot be affected, the flight control system will become inoperative when hydraulic pressure is lost and flight cannot be continued.

1. **EJECT.**

Airstart Envelope

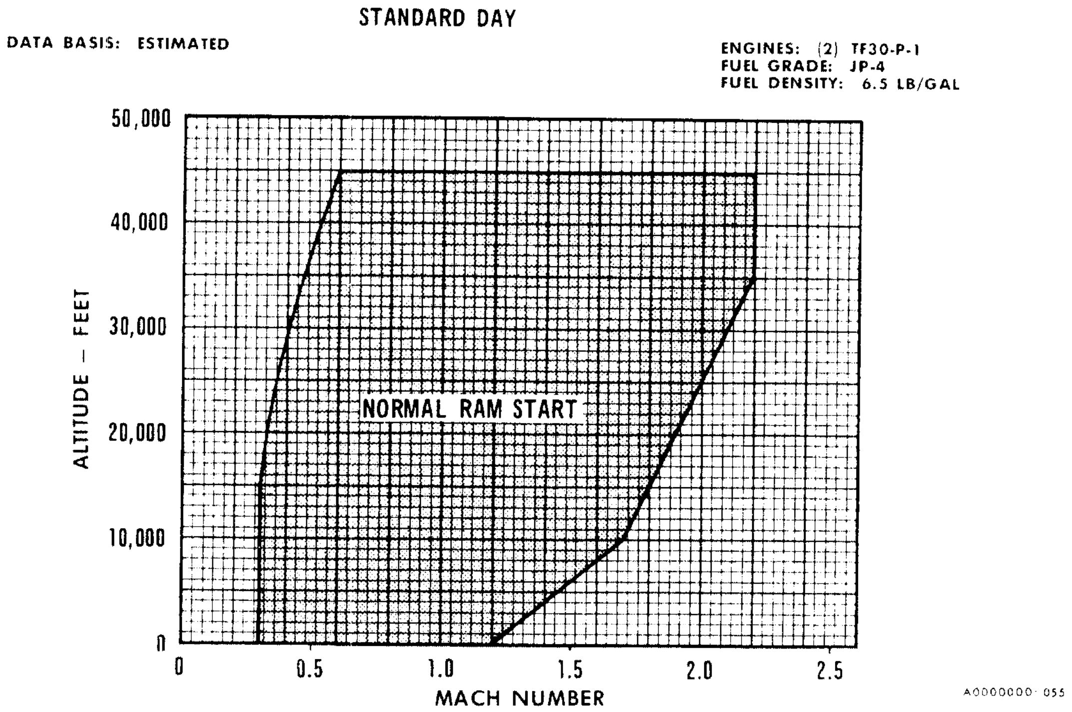


Figure 3-1.

COMPRESSOR STALL.

A compressor stall is an aerodynamic disruption of the airflow through the compressor and is caused by subjecting the compressor to a pressure ratio above its capabilities at the existing conditions. Compressor stalls may be induced by engine or inlet control malfunction, excessive angle of attack or yaw causing poor inlet air distribution, or rapid throttle reversal (high power to low power and return). Compressor stalls may be self clearing, may cause flameout, or may result in a steady state fully developed stall. In the first case no immediate action is required. In some cases the engine will stall and immediately recover with only an evidence of a stall being a light to moderate "bang". In the second case the automatic restart circuit in the engine will furnish ignition and the engine may be recovered by moving the throttle to idle to gain a restart and then reapplying power. The third case requires recognition and corrective action to restore power and prevent damage to the engine from over temperature. A compressor stall may be recognized by a pulsation felt through the airframe, an audible noise which may vary from a faint muffled thud to a very loud "bang", a loss of thrust indicated on the engine instruments, no EPR response to throttle movement and as a general rule, a rise in turbine inlet temperature. In the event of compressor stall on one or both engines proceed as follows:

1. Throttle of affected engine(s) - IDLE.
   Move the throttle of the affected engine to IDLE and check for recovery. If the engine recovers, attempt gradual application of power. If supersonic, advance throttle to MIL or above.

If Stall Does Not Immediately Clear:

2. Decelerate to mach 0.6.
3. Translating cowl switch(es) - EXTEND.
   Decelerate to below Mach 0.6, extend the translating cowl.
4. Repeat step 1.

Note

In the event that a compressor stall and/or afterburner blowout occurs in afterburner operation, but a fully stalled engine condition does not follow, an afterburner relight from military power may be attempted immediately at any flight condition.

ENGINE FIRE DURING FLIGHT.

1. **THROTTLE—OFF (ENGINE INDICATING FIRE).**
2. **FIRE PULL HANDLE OR PUSHBUTTON OF AFFECTED ENGINE—ACTUATE.**
3. **AGENT DISCHARGE CONTROL—ACTUATE.**
4. **IF FIRE CONTINUES—EJECT.**
5. If fire ceases - Land as soon as practicable.

CAUTION

Do not attempt to restart the failed engine. If the fire ceases, and a landing is to be accomplished, make a single engine landing.

SMOKE AND FUME ELIMINATION.

1. Oxygen mask and fittings - Checked.
   Check oxygen mask and oxygen hose fittings for security.
2. Air source selector knob - L. ENG, R. ENG.

Note

Attempt to determine if the engines are the source of smoke by selecting L. ENG and R. ENG positions. If source of smoke cannot be isolated to an engine, proceed as follows:

3. Air source selector knob - OFF.
4. Airspeed and altitude - As required.
   Descent to between 25,000 and 15,000 feet and/or decelerate to between 1.0 and 0.5 mach depending on altitude and airspeed, and refer to "Ram Air Mode Limits," Section V.

Attempt to isolate source of smoke or fumes as follows:

5. Electrical equipment - OFF.
   Turn off all electrical equipment not considered essential for flight.
6. Electrical equipment - ON, as required.
   Turn on electrical equipment, one system at a time, and check for smoke until source is determined.
7. Air source selector knob - RAM. (if smoke or fumes persist).

WARNING

To prevent excessive temperatures when pressure suits are being worn, the air conditioning system mode selector switch must not be placed to the OFF position prior to or while operating in the Ram position.

Note

- Moving the air source selector knob from OFF to RAM should be accomplished without pausing in the intermediate positions, to prevent the possible introduction of more smoke from one or both of the engines.
- Selecting RAM position will open the ram air scoop, dump cabin pressure, and close the pressure regulating and shutoff valve.

UNLOCKED CANOPY INDICATION DURING FLIGHT.

1. Visors - Down.
2. Oxygen mask and fittings - Checked.
   Check oxygen mask and oxygen hose fittings for security.

WARNING

When the cabin pressure schedule is changed from normal to combat, monitor the cabin pressure altimeter for a rapid increase in cabin altitude. If the cabin altitude does not increase, immediately position the pressurization selector switch to DUMP.

3. Canopy latch handle - Check locked.
4. Pressurization selector switch - COMBAT (when above 30,000 feet), DUMP (when below 30,000 feet).
5. Decelerate and descend.
   Maintain low subsonic airspeeds to minimize possibility of loss of a canopy hatch.
6. Land as soon as practicable if caution lamp remains lighted.

INFLIGHT GLASS PANEL CRACKS OR FAILURE.

1. Visors - Down.
2. Oxygen mask and fitting - Checked.
   Check oxygen mask and oxygen hose fittings for security.
3. Pressurization selector switch - COMBAT (when above 30,000 feet), DUMP (when below 30,000 feet).

WARNING

When the cabin pressure schedule is changed from normal to combat, monitor the cabin pressure altimeter for a rapid increase in cabin altitude. If the cabin altitude does not increase, immediately position the pressurization selector switch to DUMP.

4. Decelerate and descend.
5. Pressurization selector switch - DUMP (when subsonic).
6. Land as soon as practicable.

EJECTION.

Every emergency in which ejection is considered will have its particular set of circumstances, involving such factors as speed, attitude and control, and altitude. Under level flight conditions eject at least 2000 feet above the terrain whenever possible.

WARNING

Do not delay ejection below 2000 feet above the terrain in futile attempts to start the engines or for other reasons that may commit you to marginal conditions for safe ejection. Accident statistics emphatically show a progressive decrease in successful ejections as altitude decreases below 2000 feet above the terrain.

Under spin or dive conditions, eject at least 15,000 feet above the terrain whenever possible. If the airplane is controllable, attempt to decelerate as much as practical prior to ejection by zooming the airplane, thus trading airspeed for altitude. If the airplane is not controllable, ejection must be accomplished at whatever speed exists, as this offers the only opportunity for survival. An ejection at low altitudes is facilitated by pulling the nose of the aircraft above the horizon ("zoom-up maneuver"). This maneuver affects the trajectory of the crew module or ejection seat, providing a greater increase in altitude than if ejection is performed in a level flight attitude. Provided a positive rate of climb is maintained, this gain in altitude will increase the time available for complete actuation of the ejection equipment. To ensure survival during extremely low-altitude ejections, the automatic features of the equipment must be used and depended upon. Refer to "Crew Module Escape System" and "Ejection Seats", Section I, for sequence of events after ejection.

EJECTION (CREW MODULE ONLY). (12)→

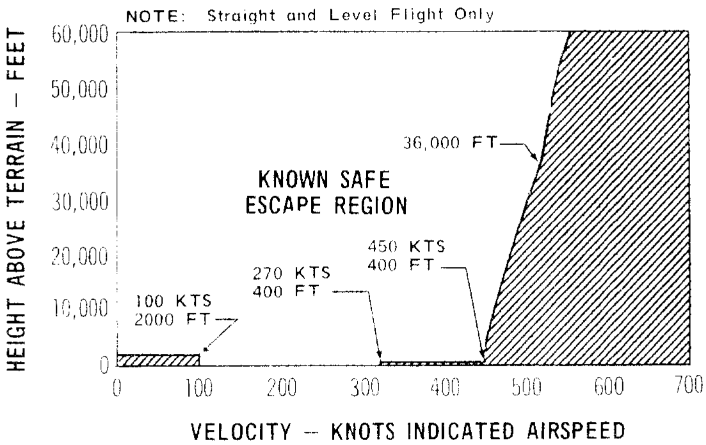
The crew module escape system provides maximum protection for crewmembers throughout the aircraft performance envelope including zero altitude and zero speed ejection capability. However, as with all aircraft ejection systems, safe ejection is enhanced by establishing the best conditions possible prior to ejection. The crew module ejection procedures and envelope is shown in figure 3-2. The envelope shown, reflects only the best or safest conditions; the decision as to when to eject or not eject in an emergency should not be rigidly determined by the fact that the aircraft is in or out of the "Safe" envelope. For example, figure 3-2 shows that the safest low speed/low altitude minimums for ejection are 100 KIAS and 2000 feet respectively. However, this in no way denies the possible need to eject under less desirable conditions.

EJECTION (EJECTION SEATS ONLY). (1)→(11)

Ejection with the open ejection seat demands special consideration of factors such as airspeed and altitude. Airspeed should be decreased as much as practicable for any inflight ejection, but is especially important for low altitude ejection due to wind forces. At sea level, wind blast will exert minor forces on the body at airspeeds up to approximately 475 KIAS, appreciable forces between 475 and approximately 600 KIAS, and excessive forces above 600 KIAS. The ejection procedures and envelope for the open seat is shown in figure 3-3. The envelope shown, reflects only the best or safest conditions; the decision as to when to eject or not eject in an emergency should not be rigidly determined by the fact that the aircraft is in or out of the "safe" envelope. For example, figure 3-3 shows that the safest low speed/low altitude minimums for ejection are 100 KIAS and 2000 feet respectively. However, this in no way denies the possible need to eject under less desirable conditions. The normal time required for seat separation and parachute deployment cannot be shortened. The harness release handle must not be pulled before ejection, regardless of altitude. If the harness release handle is pulled, the automatic opening feature of the parachute is eliminated and seat separation may be too rapid at high speeds.

Ejection Procedures (Crew Module Only)

BEFORE EJECTION



PREPARATION FOR EJECTION (IF TIME PERMITS)

1. Reduce airspeed (as practicable).
2. Advise crewmember of situation.
3. Transmit MAYDAY (give position).
4. IFF master control knob — EMERGENCY (P).
5. Inertia reel control handle — LOCKED (AC-P).

WARNING

- Minimum ejection altitudes presented in this chart were determined through sled tests and are based on distance above terrain on initiation of ejection (i.e., time module is fired). These figures do not provide any safety factor for such matters as equipment malfunction, etc. These figures are quoted only to show the minimum altitude you must attain in the event of such low altitude emergencies as fire on takeoff. These minimum altitudes are much higher when the aircraft is losing altitude.
- Under spin or dive conditions, ejection should be accomplished above 15,000 feet above the terrain.

EJECTION

1. EJECTION HANDLE — SQUEEZE AND PULL (AC or P)

DURING DESCENT

1. Emergency oxygen handle — Pull (If required).
2. Emergency pressurization ring — Pull (If required).
3. Parachute deploy handle (when below 15,000 feet) — Pull (If required).

AFTER LANDING

1. Severance and flotation handle — Pull.
2. Parachute release handle — Pull.

If landing is made in water, proceed as follows:

3. Auxiliary flotation handle — Pull (If required).
4. Bilge pump — Engage and operate (If required).
5. Air ventilation masks — Connect (If required).

If the crew module lands inverted:

6. Inertia reel control handle — Cycle to relieve tension.
7. Upper torso harness chest buckle — Release.
8. Left and right lap belt buckles — Release.

Using one arm to provide support against the canopy, release the left and right lap belt buckles; then, using both arms lower yourself out of the seat.

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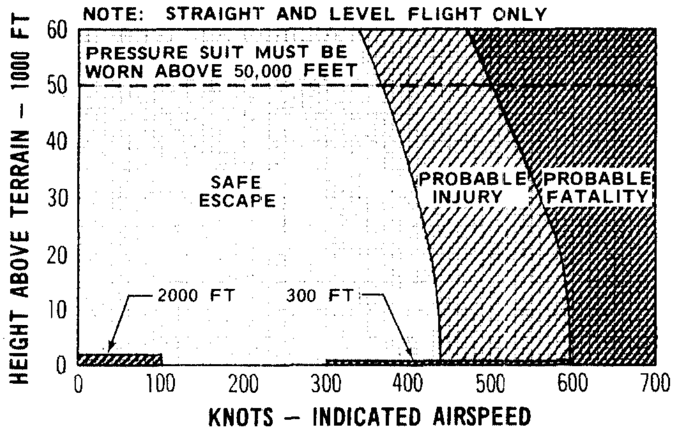
Figure 3-2.

Ejection Procedures (Ejection Seats Only)



WARNING

Minimum recovery height presented in this chart was determined through tests and is based on distance above terrain on initiation of ejection (i.e., time seat is fired). These figures do not provide any safety factor for such matters as equipment malfunction, etc. These figures are quoted only to show the minimum altitude you must attain in the event of such low altitude emergencies as fire on takeoff. These minimum altitudes are much higher when the aircraft is losing altitude. Under spin or dive conditions, ejection should be accomplished above 15,000 feet above the terrain.

PREPARATION FOR EJECTION

IF TIME PERMITS

1. Visors down.
2. Retard throttles to OFF and slow airplane as much as possible.
3. Place IFF selector switch to EMERGENCY.
4. Transmit Mayday and give position report to nearest radio facility.
5. Aim airplane toward uninhabited area.
6. Pressurization selector switch – DUMP.
7. Shoulder harness – LOCKED.

EJECTION

1. **SIT ERECT WITH SPINE STRAIGHT, HEAD FIRMLY AGAINST HEADREST, FEET ON RUDDER PEDALS.**
2. **ADVISE CREW MEMBER OF NECESSITY FOR EJECTION.**
3. **GRASP FACE SCREEN EJECTION HANDLE WITH BOTH HANDS AND PULL OUT AND DOWN UNTIL SEAT CATAPULT FIRES.**

ALTERNATE EJECTION PROCEDURE

NOTE

If circumstances prevent using the face screen ejection handle, ejection can be accomplished using the secondary ejection handle located in front of the seat pan between the crew member's legs.

1. Sit erect with spine straight, head firmly against headrest, feet on rudder pedals.
2. Advise crew member of necessity for ejection.
3. Grasp secondary ejection handle with both hands and pull up until seat catapult fires.

IF CANOPY FAILS TO JETTISON

WARNING

The seat catapult cannot be fired until the canopy has jettisoned, therefore, it is impossible to eject through the canopy. The following method of jettisoning the canopy should be used only when the ejection handle fails to jettison the canopy. Do not allow the face screen ejection handle to blow back and become inaccessible.

1. Hold the ejection handle with one hand but do not pull any further.
2. Press release button and pull the canopy jettison handle with free hand.
3. After the canopy has jettisoned. Grasp the ejection handle with both hands and pull until seat catapult fires.

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Figure 3-3. (Sheet 1)

AFTER EJECTION

WARNING

If the emergency oxygen system fails to automatically activate, the crew member must manually pull the green-ring located in front of the seat pan between the crew member's legs. If a pressure suit is not worn, breathing will be cut off when the emergency oxygen bottle is depleted. The crew member must remove the oxygen mask from his face to resume breathing.

If the automatic harness separation mechanism fails to operate, the crewmember must manually separate from the seat as follows:

1. Harness release handle – Pull.
   Squeeze the harness release handle and pull.

Note

When the harness release handle is pulled, the seat separation bladders will be inoperable. The crewmember must forcibly push the seat away.

2. Seat – Separated.
   Forcibly push the seat away from body.
3. Parachute D-ring (if below 15,000 feet) – Pull.

WARNING

The parachute actuator lanyard will not arm the parachute actuator automatically if the harness release handle is pulled. The parachute must be manually deployed by pulling the D-ring. If ejection occurs above 15,000 feet, free-fall to 15,000 feet before deploying the parachute.

After the parachute has been opened, either automatically or manually, proceed as follows:

4. Oxygen mask – Remove.
5. Oxygen supply line – Disconnect from regulator.

Note

The oxygen mask may be replaced loosely prior to touchdown to provide protection for the face during landing.

6. Left lap belt to survival kit connector – Release.
   Allow the survival kit to suspend from the right side.
7. Raft retainer lanyard – Secure to torso harness.
   Loop the raft retainer lanyard around the torso harness and secure with snap.
8. Right lap belt to survival kit connector (over land only) – Release.
   Allow the survival kit and uninflated raft to suspend directly below.
9. Inflate life preserver – If required.
   If a water landing is to be made, inflate life preserver prior to entering water.

AFTER LANDING

1. Parachute risers – Release.
2. Open life raft compartment and inflate life raft.
3. Right lap belt to survival kit connector – Release.
4. Board life raft.
5. Retrieve survival equipment tied to large end of raft and discard seat pan and parachute pack.

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Figure 3-3. (Sheet 2)

HYDRAULIC SYSTEM FAILURE.

Failure of either hydraulic system will cause the pitch, roll, and yaw damper caution lamps and the associated hydraulic low pressure caution lamp to light. The damper servo actuators will operate as non-redundant servos. For operating characteristics peculiar only to the primary or utility system, refer to the applicable system failure.

PRIMARY HYDRAULIC SYSTEM FAILURE.

As primary hydraulic pressure drops, forces will be felt in the control stick and with "hands off," will be very noticeable.

Supersonic.

1. Throttles - Retard.
   Reduce airspeed to subsonic.
2. Pitch and roll gain selector switches - MAN.
3. Pitch and roll manual gain control knobs - 10% and 20% respectively.
4. Do not attempt to reset damper caution lamps or turn off roll or yaw dampers. It may be desirable to minimize stick talk-back by turning the pitch damper OFF.
5. Wing sweep handle - 45 degrees.
   If wings are aft of 45 degrees, sweep forward to 45 degrees.

CAUTION

Maintain 1 g flight while changing wing position. Change wing sweep position at least 50 percent slower than normal rate to avoid depleting pressure of utility hydraulic system.

Subsonic.

1. Wing sweep handle - EXTEND.
   Maintain wing sweep position compatible with airspeed and sweep wings to 26 degrees when at appropriate airspeed. Minimize flight control movement during wing sweep and speed brake operation.
2. Pitch and roll manual gain control knobs - As desired (30% maximum).
   Pitch and roll gains may be increased to 30% maximum if desired for airspeeds less than 500 KIAS.
3. Maintain airspeed within the damper off operating limits.
4. Land as soon as practicable.

UTILITY HYDRAULIC SYSTEM FAILURE.

The damper servo actuators will operate as non-redundant servos but the flight control system performance will appear normal in other respects. The control stick feel will be normal.

Supersonic.

1. Throttles - Retard
   Reduce airspeed to subsonic.
2. Do not attempt to reset damper caution lamps or turn off dampers.
3. Wing sweep handle - 45 degrees.
   If wings are aft of 45 degrees, sweep forward to 45 degrees.

CAUTION

Maintain 1 g flight while changing wing position. Change wing sweep position at least 50 percent slower than normal rate to avoid depleting pressure in the primary hydraulic system.

Subsonic.

1. Wing sweep handle - EXTEND.
   Maintain wing sweep position compatible with airspeed and sweep wings to 26 degrees when as appropriate airspeed. Minimize flight control movement during wing sweep and speed brake operation.
2. Maintain airspeed within the damper off operating limits.
3. Land as soon as practicable.

COMPLETE HYDRAULIC SYSTEM FAILURE.

WARNING

If both hydraulic systems fail during flight the flight control system will be inoperative and flight cannot be continued.

1. EJECT.
   When hydraulic pressure is no longer available to control the airplane - eject.

FLIGHT CONTROL SYSTEM MALFUNCTIONS.

Various flight control system malfunctions are indicated by the lighting of an associated caution lamp. All system malfunctions, however, do not constitute a potential emergency, even though the associated caution lamp is lighted. Therefore, only those malfunctions which may develop into an emergency are covered here. Refer to Figure 3-5 for analysis of all caution lamps.

ROLL GAIN CHANGER CAUTION LAMP LIGHTED.

An error in one of the redundant gain changers will cause the roll gain changer caution lamp to light.

1. Damper reset button - Depress momentarily.
   If depressing the damper reset button causes the lamp to go out, continue normal operation. If lamp does not reset, select manual roll gain at 20 percent. Again depress damper reset button. If lamp resets, continue operation. (Do not exceed 50 percent gain.) If lamp does not reset, decrease speed to less than 320 KIAS and 0.8 mach. Utilize 50 percent manual gain for remainder of flight.

PITCH GAIN CHANGER CAUTION LAMP LIGHTED.

An error in one of the redundant gain changers will cause the pitch gain changer caution lamp to light.

1. Damper reset button - Depress momentarily.
   If depressing the damper reset button causes the lamp to go out, continue normal operation. If lamp does not reset, select manual pitch gain at 10 percent. Again depress damper reset button. If lamp resets, continue operation. (Do not exceed 30 percent gain.) If lamp does not reset, decrease speed to less than 320 KIAS and 0.8 mach. Utilize 30 percent gain for remainder of flight.

PITCH, ROLL, OR YAW CHANNEL CAUTION LAMP LIGHTED.

Failure of one of the redundant electrical signal paths causes the appropriate channel caution lamp to light. The failed signal will be electronically rejected and aircraft damping will be unaffected. If failure was a zero command, the appropriate channel lamp will come on during a maneuver. Depressing the damper reset button will cause the caution lamp to reset for this type of failure. Normal operation can be continued as long as the channel lamp can be reset since any subsequent failure will cause either no effect, or zero stability augmentation in the affected channel. If the failure is a hardover signal, the system electronically rejects the failed signal and the channel lamp will immediately light and will not reset. For this condition, normal damping is present, however, a secondary failure could cause either no effect, or a hardover damper servo. For this reason, airspeed should be reduced to the "Dampers off" flight condition, the affected damper turned off, and the aircraft returned for landing when practical.

1. Damper Reset Button - Depress.
   If depressing the damper reset button causes the channel lamp to go out, continue normal operation. The lamp may come on during subsequent maneuvers, however, normal operation can be continued as long as lamp will reset. If depressing the damper reset button does not reset the caution lamp, one branch has failed hardover. Observe the stability augmentation off operating limit, turn off the affected damper, and land as soon as practicable.

PITCH, ROLL, OR YAW DAMPER CAUTION LAMP LIGHTED.

A lighted damper lamp indicates that the three signals to the damper servo do not agree. If the lamp remains out after the damper reset button is momentarily depressed, one of the three signals has failed to a zero or null command. Any subsequent failure in that axis will result in either normal operation or zero damping. If the damper lamp remains lighted after the damper reset button is momentarily depressed, one of the three signals has failed to a hardover command and has been voted out. A subsequent failure in that axis could cause the damper to go hardover. Certain power failures to the flight control computers have the effect of causing one damper command to fail to a zero or null command. In some of these cases the roll or pitch gain meter will drop suddenly to less than 10 percent. These cases should be treated the same as a damper lamp that will reset. If a damper lamp lights or gain meters suddenly drop to less than 10 percent, proceed as follows:

1. Reduce speed to the applicable "Stability Augmentation Off Limits," Section V.
2. Damper reset button - Depress momentarily.
   a. If lamp does not reset, turn affected damper OFF.
   b. If lamp does reset, leave damper on.
   c. If gain meters drop suddenly to less than 10 percent, leave dampers on.
3. Land as soon as practical.

RUDDER AUTHORITY CAUTION LAMP LIGHTED.

If rudder authority differs from that programmed by the control system switch, the rudder authority caution lamp will light.

1. Rudder authority switch - Check.
   Check that the rudder authority switch is in AUTO. If lamp remains lighted, the rudder authority may be unscheduled. At high speeds, exercise caution in the use of rudder pedals. For landing, if lamp remains lighted, place the rudder authority switch to FULL. If the lamp still remains on, rudder and nose wheel steering authority may be limited.

TF FLY-UP OFF CAUTION LAMP LIGHTED.

During TF operation, the TF fly-up off caution lamp will light if any one of the following conditions exist:

a. TFR system on but Auto TF switch is OFF.

b. (Deleted)

c. Landing gear is down.

d. Control system switch in T.O. & LAND.

1. Auto TF Switch - FLY UP ONLY or AUTO TF.
   If caution lamp remains lighted, the automatic fly-up capability is not available.

SINGLE GENERATOR FAILURE.

Failure of one generator will be noted by the lighting of the applicable caution lamp. One generator in normal operation is sufficient to support the entire electrical load or demand. Should a generator caution lamp light, proceed as follows:

1. Electrical control panel - Check.
   Check electrical control panel for TIE indication in power flow indicator.

2. Generator switch - OFF, then GEN.

3. Generator caution lamp - Check.
   If the generator fault has been corrected, the generator will be reconnected to the system and the caution lamp will go out. If the generator caution lamp remains lighted, proceed as follows:

4. Generator switch - OFF then TEST.
   If the caution lamp goes out with the switch in TEST position, it indicates that the generator is operating normally and the malfunction is associated with the contactor circuit or the caution lamp circuit. The generator switch should then be returned to OFF and left there. If the caution lamp remains lighted in TEST, proceed as follows:

5. Generator switch - OFF.

6. Generator decouple button - Depress.

DOUBLE GENERATOR FAILURE.

Double generator failure will not result in a total loss of electrical power for more than the maximum of 3 seconds required for the emergency generator to provide power for the essential AC and DC buses. For listing of equipment powered by the essential buses, refer to "Electrical Power Supply System, Section I".

Note

In the event that the emergency generator does not come on within 3 seconds, place the emergency generator switch to ON.

1. Electrical control panel - Check.
   Check electrical control panel for EMER indication in power flow indicator.

2. Generator switches - OFF then GENERATOR.

3. Reduce electrical load to minimum necessary to sustain safe flight.

4. Fuel panel - Check.
   With only the emergency generator providing electrical power, only fuel boost pumps 4 & 5 will be operable. Refer to "Fuel System Operation on Emergency Electrical Power," this section.

5. Land as soon as practicable.

FUEL SYSTEM OPERATION ON EMERGENCY ELECTRICAL POWER.

When operating on the emergency generator, the electrical power provided will operate only one fuel booster pump at a time (number 4 pump in the forward tank or number 5 pump in the aft tank) or the two outboard wing transfer pumps. The transfer pumps cannot be operated while one of the fuselage booster pumps is operating. When the engine feed selector knob is in FWD, only the number 4 pump in the forward tank will be operating and will supply fuel to both engines. When the engine feed selector knob is in AFT or BOTH, only the number 5 pump in the aft tank will be operating and will supply fuel to both engines. When the engine feed selector knob is in AUTO, either pump 4 or pump 5 will operate depending on fuel distribution. If the fuel differential is greater than 8500 pounds, number 4 pump will supply fuel to the engines. If the fuel differential is less than 7900 pounds, number 5 pump will supply fuel to the engines. If, when the AUTO position is initially selected, the fuel differential is less than 7900 pounds, the number 5 pump will transfer fuel to the forward tank until the proper fuel differential is established. From this point on, either pump 4 or 5 will be automatically selected to supply fuel directly to the engines. During the period that pump 5 is transferring fuel forward, the engines will be operating on suction feed. In order to transfer fuel from the wing tanks, the engine feed selector knob must be turned OFF and the fuel transfer knob placed to WING. This will result in the engines being fed by suction from the forward tank. Fuselage tank fuel quantities must be closely monitored to maintain the proper distribution during wing transfer. If distribution gets out of tolerance, it can be corrected by positioning the engine feed selector knob to AUTO. During suction feed, the fuel manifold low pressure caution lamps may come on. This should cause no concern since sufficient fuel flow should be available to operate in maximum afterburner up to 6000 feet or military power up to 30,000 feet.

ENGINE FEED.

1. Engine feed selector knob - AUTO.
   Closely monitor fuel quantity in the fuselage tanks to maintain 8200 (±300) pounds fuel differential.

2. Fuel tank pressurization selector switch - PRESSURIZE.

FUEL TRANSFER.

1. Fuel transfer knob - WING.

2. Engine feed selector knob - OFF.
   Monitor fuel quantity in the fuselage tanks to maintain 8200 (±300) pounds fuel differential.

Note

When the wings are swept aft, a larger amount of fuel will be trapped in the wing tanks. To transfer all available fuel from the wing tanks, the wings must be in the extended positions. Gravity transfer of fuel is not possible.

FUEL MANIFOLD LOW PRESSURE CAUTION LAMP INDICATION.

ONE FUEL MANIFOLD LOW PRESSURE CAUTION LAMP.

1. Flowmeter - Checked.
   Check appropriate fuel flowmeter to determine if fuel flow is excessive.

2. If flow is excessive, shut down the affected engine, pull the fire pull handle or depress the fire pushbutton, and extend the speed brakes.

3. If fuel flow is normal, check totalizer to determine if there is an excessive loss of fuel.

4. If fuel loss is excessive, shut down the affected engine, pull the fire pull handle or depress the fire pushbutton, and extend the speed brakes.

5. If fuel consumption is normal, check the fuel booster pump low pressure indicator lamps.

6. If fuel booster pump low pressure indicator lamps are on, indicating booster pump failure, retard appropriate throttle until manifold low pressure caution lamp goes out.

Note

With all booster pumps inoperative and normal tank pressurization available, the engines will operate in maximum afterburner up to 6000 feet or military power up to 30,000 feet.

7. If fuel booster pump low pressure indicator lamps are normal, descend to 30,000 feet and land as soon as practicable.

TWO FUEL MANIFOLD LOW PRESSURE CAUTION LAMPS.

1. Engine fuel feed selector knob - Checked.
   Check engine fuel feed selection to insure that fuel is available to the engines.

2. Flowmeters - Checked.
   Check fuel flowmeters to determine if fuel flow to either engine is excessive.

3. If either flowmeter indicates excessive fuel flow, shut down that engine, pull the fire pull handle or depress the fire pushbutton and extend the speed brakes.

4. If fuel flow is normal, check the totalizer to determine if there is an excessive loss of fuel.

5. If fuel loss is excessive, retard throttles until one caution lamp goes out. Then shut down the engine with the caution lamp still on. Pull the fire pull handle or depress the fire pushbutton and extend the speed brakes.

6. If fuel consumption is normal, check the fuel booster pump low pressure indicator lamps.

7. If fuel booster pump low pressure indicator lamps are on, indicating pump failure, retard throttles until the manifold low pressure caution lamps go out.

Note

With all booster pumps inoperative and normal tank pressurization available, the engines will operate in maximum afterburner up to 6000 feet or military power up to 30,000 feet.

8. If fuel booster pump low pressure lamps are normal, descend to 30,000 feet or below and land as soon as practicable.

OIL SYSTEM FAILURE.

An oil system malfunction on either engine is recognized by a change in oil pressure or a complete loss of oil pressure. In general, it is advisable to shut the engine down as soon as possible after a drop in oil pressure is indicated, to minimize the possibility of damage to the engine. However, if thrust is critical, the engine may be utilized as long as it continues to produce power.

OIL PRESSURE BETWEEN 30 AND 40 PSI (EXCEPT AT IDLE).

1. Throttle of affected engine - IDLE.

2. Monitor oil pressure.

OIL PRESSURE BELOW 30 PSI.

1. Throttle of affected engine - OFF. (If flight conditions permit).

CAUTION

If oil pressure goes to below 30 psi and it is necessary to keep the engine operating to sustain flight, engine seizure can be expected.

OIL PRESSURE ABOVE 50 PSI.

1. Throttle of affected engine - Retard.
   Reduce thrust on affected engine. If oil pressure can be maintained in 40 to 50 psi range continue to operate engine at the reduced power setting. If oil pressure cannot be reduced to the 40 to 50 psi range, shut the engine down.

SPIKE SYSTEM FAILURE.

Since there is no positive means of determining spike position, a spike system failure or spike mispositioning can be recognized only by a reduction in engine or engine inlet performance. The evidence of a spike system failure will differ according to airspeed at the time of failure. Failure of the spike system at mach numbers above 1.5 will most probably be evidenced by inlet buzz and/or compressor stall. Failure of the spike in the lower speed range may result in an engine compressor stall.

1. Airspeed - Reduce as necessary.
   If above mach 1.5 and inlet buzz and/or compressor stall is present, decelerate to mach 1.5 or until buzz or compressor stall disappear.

LANDING.

1. Engine spike caution lamps - OUT.
   When at mach 0.3 (approximately 200 KIAS at sea level or 175 KIAS at 6000 ft), check that engine spike caution lamps are out. If either lamp is lighted, proceed with next step.

2. Applicable spike control switch - Move to AUTO position.

   a. If lamp goes out, return switch to OFF.

   b. If lamp remains lighted, place switch to OVERRIDE.

WARNING

After a spike control switch has been positioned to OVERRIDE, do not change its position for remainder of flight. To do so could result in loss of utility hydraulic system.

AIR CONDITIONING AND PRESSURIZATION SYSTEM FAILURE.

UNCONTROLLED CABIN OVERHEAT.

If hazardous cabin overheat occurs, place the air source selector knob to OFF and turn off all non-essential electronic equipment. Descend to an altitude where cockpit pressurization and/or heating is not required, and land as soon as practicable.

LANDING EMERGENCIES

The canopy will be retained during all landing emergencies.

LANDING WITH PRIMARY OR UTILITY HYDRAULIC SYSTEM FAILURE.

Fly an extended downwind leg sufficiently long to provide time for lowering the landing gear and flaps by the alternate method. After touchdown, normal braking and anti-skid will be available until the brake accumulator pressure has been reduced to 1100 ±100 psi (after approximately 10-14 full brake applications). Differential braking must be used to maintain directional control during landing roll. To minimize consumption of brake accumulator hydraulic fluid, braking should be accomplished by as few brake applications as possible. A single moderate and steadily increasing brake application is recommended. If the number of brake applications utilized or the amount of anti-skid cycling is great enough to reduce accumulator pressure to less than 1100 ±100 psi, normal braking will not be available and it will be necessary to pull the auxiliary brake handle to stop the airplane.

1. Control system switch - T.O. & LAND.

2. Landing Gear - Extend.
   Extend the landing gear using the "Landing Gear Emergency Extension" procedures, this section.

3. Spike control switches - As required.
   Prior to landing with either hydraulic system inoperative, decelerate to mach 0.3 (approximately 200 KIAS at sea level or 175 KIAS at 6000 ft), and check that engine spike caution lamps are out. If either lamp is lighted, place the affected spike control switch to OVERRIDE.

WARNING

After a spike control switch has been positioned to OVERRIDE, do not change its position for remainder of flight. To do so could result in loss of utility hydraulic system.

4. Flaps and slats - Extend.
   Extend the flaps and slats using the "Emergency Extension of Flaps and Slats" procedures, this section.

Note

Lateral response rates will be reduced due to one pair of spoilers being inoperable.

5. Maintain directional control after touchdown by differential braking.

Note

- If only the utility hydraulic system is operative, only the outboard spoilers will be available. If only the primary hydraulic system is operative, the inboard spoilers only will be available.

MAIN LANDING GEAR FAILURE TO EXTEND AND LOCK AFTER RELEASING FROM UPLOCK.

1. Landing gear handle - Recycle.
   Recycle the landing gear handle from DN to UP then back to DN. Check for gear down indication.

2. Impose a g load on the airplane and check for gear down indication.

3. Aircraft commanders speed brake switch - OFF.

4. Utility hydraulic system isolation switch - ISOLATE.
   The switch must be held in ISOLATE until completion of step 5.

5. Landing gear handle - UP.
   Check for gear down indication.

6. Move the speed brake a small distance in both directions and check for gear down indication.

CAUTION

Do not move speed brake far enough aft so as to interfere with the main landing gear.

7. Landing gear handle - DOWN.

8. If gear still does not extend, follow "Emergency Landing Gear Extension" procedure, this section.

LANDING GEAR EMERGENCY EXTENSION. See SS-7 dated 21 April 67

If the landing gear cannot be extended using the normal procedures, proceed as follows:

1. Landing gear handle - ~~DN, below 160 KIAS~~ UP.
2. Landing gear EMERGENCY ~~(alternate)~~ release handle - Pull. *
   Check landing gear wheel position lights for green indication.

* At recommended minimum flying speed

3. Landing gear indicator lamps - ~~Out.~~ Lighted
4. Landing gear Handle - DOWN
5. Landing gear handle warning lamp - Out.

Note

After the landing gear alternate release handle is pulled, nose wheel steering will be inoperative and the nose wheel will be cocked to one side. The nose wheel being cocked will present no directional control difficulty on touchdown however since the hydraulic pressure holding it cocked is slight. During landing roll, the nose wheel should be held off the runway as long as possible.

If the speed brake fails to retract to the trail position, as indicated by the landing gear handle warning lamp remaining on after the gear has extended and locked:

6. Landing gear alternate release handle - IN.

CAUTION

With the landing gear alternate release handle in, reduced air loads during landing will allow the speed brake to extend and drag the landing surface.

If landing gear fails to extend or lock down after the landing gear alternate release handle is pulled, follow "Landing With Gear Up or Unlocked" procedures, this section.

After Landing.

6. Landing gear (alternate) release handle - In
   After the airplane is parked and prior to engine shutdown, ensure that the landing gear alternate release handle is pushed in.

CAUTION 1→11

If the landing gear alternate release handle is not pushed in after engine shutdown, when electrical power is removed from the buses hydraulic pressure will drive the speed brake full down and cause damage from ground contact.

LANDING WITH UNSAFE GEAR INDICATION.

1. Slow aircraft to appropriate limit speed and extend flaps to 15 degrees.

2. Landing gear circuit breakers - Check.
   Check the landing gear control, landing gear warning circuit breakers.

3. Landing gear handle - Recycle.

If landing gear is still unsafe:

4. Landing gear alternate release handle - Pull.

Note

After the landing gear alternate release handle is pulled nose wheel steering will be inoperative.

If landing gear is still unsafe:

5. Obtain a visual gear check from another airplane and/or the control tower if possible.

If landing gear is still unsafe follow "Landing With Landing Gear Up or Unlocked" procedures, this section.

LANDING WITH MAIN GEAR UP OR UNLOCKED AND NOSE GEAR DOWN OR LANDING WITH ALL LANDING GEAR UP OR UNLOCKED.

1. Landing gear handle - DN.

2. Landing gear alternate release handle - Pull.

3. External load - Jettison.

4. Dump or burn excess fuel.
   Lighten the aircraft as much as possible by dumping or burning excess fuel.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the airplane. This should cause no concern however since the fire will remain behind the airplane. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

5. External tanks - Retain if empty.

6. Battery switch - OFF.

7. Shoulder harness - LOCKED.

8. Fly a normal landing pattern and make a normal landing.

Note

Attempt to touchdown at normal landing attitude. Do not try to hold the aircraft off the runway by increasing angle of attack. Lower the nose to the runway while elevator control is still available.

9. Throttles - OFF.
   Immediately after touchdown, shut down the engines.

10. Fire pull handles or pushbuttons - Actuate.

11. Abandon the airplane.

LANDING WITH NOSE GEAR UP OR UNLOCKED, MAIN GEAR DOWN.

1. Landing gear handle - DN.

2. External load - Jettison.

3. Dump or burn excess fuel.
   Lighten the aircraft as much as possible by dumping or burning excess fuel.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the airplane. This should cause no concern however since the fire will remain behind the airplane. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

4. External tanks - Retain if empty.
5. Battery switch - OFF.
6. Antiskid switch - ON.
7. Shoulder harness - LOCKED.
8. Fly a normal landing pattern and make a normal landing.

Note

Attempt to touch down at normal landing attitude. Hold the nose off the runway as long as practicable. Aerodynamic braking is very effective in the nose high attitude, howev^r, if braking is required, use light braking so as not to override elevator control. Lower the nose gently to the runway before elevator control is lost. Do not use ground roll spoiler brakes.

9. Throttles - OFF.
   After the nose is on the runway shut down the engines.
10. Fire pull handles or pushbuttons - Actuate.
11. Abandon the airplane.

LANDING WITH BLOWN TIRE.

MAIN GEAR TIRE.

1. Fly a normal landing pattern.
2. Arresting hook - Extend.
3. Antiskid switch - OFF.
   With the antiskid switch OFF, and using differential braking, the wheel with the blown tire can be locked.
4. Touch down on side of runway opposite the blown tire.
5. Ground roll spoiler switch - BRAKE.
6. Lower nose and use nosewheel steering and brakes as required to keep airplane on runway.
7. Brakes - Lock brake with blown tire. Use other brake as required.

NOSE GEAR TIRE.

Use same procedure as with a main tire except land in the center of the runway and hold the nose off the runway as long as possible. Antiskid should be left on when landing with a blown nose wheel tire. Do not lock either brake on landing.

NOSE WHEEL STEERING MALFUNCTION.

If nose wheel steering system malfunction is indicated by hard-over nose wheel steering, or loss of directional control, disengage nose steering by pulling the landing gear alternate release handle. Maintain directional control with rudder and differential braking.

EMERGENCY EXTENSION OF FLAPS AND SLATS.

1. Airspeed - As required.
   Reduce airspeed to applicable flap limit speed.
2. Flap and slat switch - EMER.
3. 1→11 Emergency slat switch - EXTEND.
   Hold emergency slat switch in EXTEND until slats are full down.
4. 1→11 Emergency flap switch - EXTEND.
   Hold emergency flap switch in extend until flaps are in the desired position.
5. 12→ Emergency flap and slat switch - EXTEND.
   Hold the emergency flap and slat switch in EXTEND until the slats are down and the flaps are in the desired position.

SINGLE ENGINE LANDING AND GO-AROUND.

During single engine operation, utility and primary hydraulic system flow is reduced by almost 50 percent. Because of this, the landing gear system, speed brake system, and air inlet control system will each absorb total flow of the utility hydraulic system when actuated. Avoid operation of more than one utility hydraulic system function at a time. Since the flight control system utilizes both utility and primary pressure, operation of necessary utility hydraulic system functions should be accomplished while in level flight. Wing sweep changes may require as much as 20 seconds for completion. During wing sweep operation, no other demands should be placed on the utility system such as speed brakes, air inlet control or flaps.

Changes in wing sweep should be accomplished in straight and level flight. During the landing approach, keep rpm on the operating engine as high as practicable until touchdown. The possibility of a go-around should be recognized early. If a go-around is necessary, advance the throttle of the operating engine and continue approach until go-around airspeed is reached. Refer to Appendix I for single engine climb speeds. When landing in a gusty crosswind, final approach airspeed should be increased by ten knots.

LANDING WITH ONE TRANSLATING COWL CLOSED.

The following procedure is recommended for the prevention of, or operating with, an engine compressor stall occurring in the lower speed range (below mach 0.3) as a result of a translating cowl failing in the closed position. An imminent engine compressor stall may be evidenced by vibration and/or fluctuating engine instruments. An actual compressor stall is evidenced by an explosive like disturbance which is heard and felt.

1. Burn or dump excess fuel.
2. Accomplish normal pattern and landing procedures.
3. Affected engine power - Set.
   Set power on affected engine to approximately 75-78 percent rpm and use the other engine for all subsequent power requirements.
4. Reduce airspeed with flaps and landing gear as necessary.

WARNING

Inducing a nose high attitude to reduce speed should be avoided to prevent possible excessive sink rate.

SS-8

Note

~~The translating cowl caution lamp will light at approximately 135 KIAS.~~

LANDING WITH BOTH TRANSLATING COWLS CLOSED.

Follow same procedures as those listed for "Landing With One Translating Cowl Closed" except, establish landing pattern as necessary to avoid power settings that exceed those listed.

SIMULATED SINGLE ENGINE LANDING.

Simulated single engine landing should be flown with one engine at idle rpm, following the "Single Engine Landing" procedure, this section.

LANDING WITH WINGS AT 26 DEGREES SWEEP OR GREATER AND NO FLAPS.

Landing with wings and flaps in other than normal landing configuration will necessitate a long, shallow, straight-in approach. Avoid abrupt maneuvers or flight in excess of 1 g.

1. Burn or dump excess fuel.
   Because of the high approach and touchdown airspeeds involved during landing with wings swept past 26 degrees, burn or dump as much fuel as practicable prior to entering traffic pattern.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the airplane. This should cause no concern however since the fire will remain behind the airplane. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

2. Landing cockpit check - Completed.
   Perform normal landing cockpit check.
3. Touchdown airspeed - As required.
   During transition from 50 feet altitude to touchdown, the rate of descent should be approximately 480 FPM. The angle of attack should be approximately 13° for wing sweep or 50° or greater and 10° for wing sweep of 35° or less. Touch down as near to the approach end of the runway as possible.

Note

The following braking technique is based on the assumption that sufficient runway is available. If less than the required runway is available, maximum braking should be initiated as soon as possible.

4. Throttles - IDLE.

Note

Ground roll spoilers will not be available at wing sweep angles of 35° or greater.

Emergency Landing Airspeeds and Ground Roll Distances

50,000 POUNDS GROSS WEIGHT

Wing Sweep (Degrees)	Pattern Speed (Knots)	Final Approach Speed (Knots)	Touchdown Speed (Knots)	Ground Roll Distance (Feet)
26	220	175	165	5000 4100*
50	250	184	174	6000
60	265	193	183	8300
72.5	280	205	195	9800

55,000 POUNDS GROSS WEIGHT

Wing Sweep (Degrees)	Pattern Speed (Knots)	Final Approach Speed (Knots)	Touchdown Speed (Knots)	Ground Roll Distance (Feet)
26	220	187	177	8700 5600*
50	250	196	186	10,600
60	265	206	196	12,600
72.5	280	218	208	14,000

WARNING

Minimum allowable gross weight at 16 degrees wing sweep is 56,000 pounds.

16 DEGREE WING SWEEP (56,000 POUNDS)

	Pattern Speed (Knots)	Final Approach Speed (Knots)	Touchdown Speed (Knots)	Ground Roll Distance (Feet)
SLATS AND FLAPS RETRACTED	215	180	170	5000* 6600
SLATS AND FLAPS EXTENDED	140	135	125	2400*

*Ground roll spoilers extended at touchdown.

Figure 3-4.

5. After touchdown, hold the nose wheel off the runway (approximately 10° angle of attack). At 173 KIAS, apply as much braking pressure as possible while still maintaining a 10° angle of attack.

6. At 135 KIAS, smoothly lower the nose wheel to the runway and apply maximum braking. Hold the control stick full back to utilize the maximum drag of the horizontal tail.

CAUTION

If excessive braking is used to high speeds, the wheel blowout plugs may relieve tire pressure within 3 to 15 minutes after stop. Provisions should be made to cope with wheel fires which may start shortly after the blowout plugs relieve.

7. Arresting hook - As required.

Note

Figure 3-4 provides airspeeds and ground roll distances for specific gross weights. For gross weights in excess of these, refer to Appendix I.

CAUTION

Call the fire department after an emergency landing which results in hot wheels or brakes or tail hook. Do not shut the engines down until after the fire trucks arrive. Fuel venting from the engines after shut down may be ignited by the affected hot part.

Caution Lamp Analysis

Indicator	Cause	Corrective Action
ROLL GAIN CHANGER	One of the redundant gain changers is in error	Depress damper reset button momentarily. If lamp resets, continue operation. If lamp does not reset, select manual roll gain at 20%. Again depress damper reset button. If lamp resets, continue operation. (Do not exceed 50% gain.) If lamp does not reset, decrease speed to less than 320 KIAS and 0.8 mach. Utilize 50% manual gain for remainder of flight
PITCH GAIN CHANGER	One of the redundant gain changers is in error	Depress damper reset button momentarily. If lamp resets, continue operation. If lamp does not reset, select manual pitch gain at 10%. Again depress damper reset button. If lamp resets, continue operation. (Do not exceed 30% gain.) If lamp does not reset, decrease speed to less than 320 KIAS and 0.8 mach. Utilize 30% gain for remainder of flight
ROLL CHANNEL OR PITCH CHANNEL OR YAW CHANNEL	One of the triple redundant channels is in error	Depress the damper reset button each time a channel lamp lights. Continue normal operation as long as lamps will reset. If a lamp will not reset, change speed to a damper off region and turn the malfunctioned damper off. Land as soon as practicable
ROLL DAMPER OR PITCH DAMPER OR YAW DAMPER	One of the triple redundant commands to a damper servo is in error	Reduce speed to the applicable stability augmentation system off operating limits, Section V. Depress Damper Reset Button. If light will not reset, turn affected damper off. Land as soon as practical

Figure 3-5. (Sheet 1)

Caution Lamp Analysis

Indicator	Cause	Corrective Action
L PRI HYD R PRI HYD	Pressure output of the indicated primary hydraulic pump is below 400 to 600 PSI	Monitor primary hydraulic pressure. If normal pressure - continue flight. If abnormal pressure - follow PRIMARY HYDRAULIC SYSTEM FAILURE, this section
L UTIL HYD R UTIL HYD	Pressure output of the indicated primary hydraulic pump is below 400 to 600 PSI	Monitor utility hydraulic pressure. If normal pressure - continue flight. If abnormal pressure - follow UTILITY HYDRAULIC SYSTEM FAILURE, this section
ROLL, PITCH, AND YAW DAMPER WITH PRIM HYD CAUTION LIGHTS	Primary hydraulic pressure is low	Reduce speed to subsonic. Monitor primary hydraulic pressure, reset dampers if pressure is greater than 2000 psi. For lower pressures, set 10% pitch and 20% roll manual gains. Turn pitch damper off if stick talk back is excessive. Wing sweep to 50 degrees or less at 1/2 normal rate. Land as soon as practical
ROLL, PITCH, AND YAW DAMPER WITH UTIL CAUTION LIGHTS	Utility pressure is low	Reduce speed to subsonic. Depress Damper Reset button if pressure is greater than 2000 psi. Do not reset for lower pressures. Follow normal flight control operating procedures, Section II. Wing sweep to 45 degrees or less at 1/2 normal rate. Land as soon as practical
L FUEL PRESS R FUEL PRESS	Affected fuel manifold pressure is less than 15.5 PSIA	Check fuel feed selector switch and fuel pump pressure lamps. Check fuel flow. If fuel pressure continues to drop, consult emergency procedures
L ENG SPIKE R ENG SPIKE	Airspeed is 0.3 mach or below and the affected spike has not extended or has not collapsed	Position appropriate spike control switch(es) to OVERRIDE. Do not attempt to return to AUTO position after the spike control switch has been placed to OVERRIDE
L ENG OIL HOT R ENG OIL HOT	Oil temperature of affected engine exceeds 245°F (118°C)	Retard throttle of affected engine to IDLE and monitor oil pressure. If oil pressure drops below 30 PSI, shut down the affected engine
L ENG OVERSPEED R ENG OVERSPEED	Excessive N1 RPM	Retard throttle of affected engine. Lamp should go out at reduced power. If lamp remains on, operate engine at reduced power
L GEN R GEN	Indicated generator has malfunctioned and has disconnected from its ac bus	Check power flow indicator displays TIE. The operating generator will automatically connect to the inoperative bus

Figure 3-5. (Sheet 2)

Caution Lamp Analysis

Indicator	Cause	Corrective Action
ANTISKID	Loss of power control box, failure of one or more skid detectors, or loss of hydraulic pressure	Check utility hydraulic pressure. Modulate brakes during landing. If hydraulic pressure is normal, a lighted lamp will indicate total or partial antiskid loss
NUCLEAR		Not operable
SPOILER	One pair of spoilers has been voted out and locked down	Maintain positive control of aircraft attitude and decelerate to safe speed. Attempt to reset spoiler one time only but expect a rapid roll transient if spoiler is still failed. A spoiler that was voted out because of an active failure will not likely reset. The roll rate capability during landing will be reduced by approximately 50 percent
FOD	FOD prevention doors extended	Close doors prior to take off
CADC	One of CADC monitors indicates malfunction	Cross check flight instruments to determine if any are inoperative. Use standby instruments in lieu of malfunctioning primary instruments
PRI HOT UTIL HOT	Indicated hydraulic system fluid temperature is above 230°F (110°C)	Reduce speed. Monitor hydraulic pressure. Reduce demand on hydraulic system. At temperatures above 230°F, the hydraulic system may aerate and develop those characteristics associated with air in hydraulic system
FUEL FLOW	Useable fuel in fuselage reservoir tank is 2115 to 2585 pounds or less	Transfer any available fuel into forward fuselage tank. If no other fuel is available, land as soon as possible. Fuel conditions may vary when this lamp comes on. Evaluete the condition and take necessary action
COWL	SC-3 One or both cowls not fully closed [illegible] mach 0.5	[illegible] check cowl switch [illegible] CLOSE and pos. indicators show CLOSE. If lamp remains on, do not [illegible] [illegible] open limit speed
OIL LOW	Oil level in either engine down to 4 quarts	Check oil quantity indicators. Shutdown affected engine if not needed. If engine needed shutdown when oil pressure starts to drop
INLET HOT	Anti-icing air temperature excessive	Shut off engine inlet anti-icing. Lamp should go out. If not, slow airplane
OXY	Total liquid oxygen remaining is two liters or less or pressure is 42 psi or less	Descend to a safe altitude. Refer to "Oxygen Duration Table," Section I

Figure 3-5. (Sheet 3)

Caution Lamp Analysis

Indicator	Cause	Corrective Action
HOOK DOWN	Arresting hook is not up and locked	Land past the approach end barrier. Arresting hook cannot be retracted in flight
REF NOT ENGAGED	Selected autopilot reference is not engaged	If caused by control stick steering, return stick to neutral position. If caused by malfunction, depress autopilot disengage lever. If any mode other than attitude stabilization is selected, REF ENGAGE button must be depressed to re-engage autopilot
AUX ATT	AFRS altitude information unreliable	Place flight instrument reference select switch to PRI. The standby attitude indicator will be unreliable
PRI ATT/HDG	Failure of inertial reference unit or computer display unit	Position flight instrument reference select switch to AUX. Autopilot switches will go to DAMPER in the AUX mode. Caution lamp will remain lighted whenever switch is not in PRI position
TF FLY-UP OFF	1. TFR mode switch is OFF while operating in TFR 2. TF data bad signal or landing gear down or Control System switch in T.O. & Land	Position TFR mode switch to FLY UP ONLY or AUTO TF. If caution lamp remains on, the automatic fly-up capability is not available
RUDDER AUTHORITY	Rudder authority differs from that programmed by the control system switch	Check rudder authority switch in AUTO. If lamp remains on, the rudder authority may be unscheduled. At high speeds, exercise caution in the use of rudder pedals. For landing, if lamp remains on, place the rudder authority switch to FULL. If the lamp still remains on, rudder and nose wheel steering authority may be limited
FUEL DISTRIB	The automatic fuel distribution has failed. Fuel distribution is out of limits	Select FWD or AFT tank feed until proper fuel distribution is regained. Monitor fuel quantities and fuel distribution
TANK PRESS	Fuel tank pressurization is not compatible with aircraft configuration	Place fuel tank pressurization selector switch to appropriate position to cause the lamp to go out. Monitor fuel quantities and assure that pressure loss has not affected fuel quantity or distribution
CABIN PRESS	Cabin altitude above 10,000 feet	Check oxygen equipment. Assure oxygen is on. Check that pressurization selector switch is in NORM

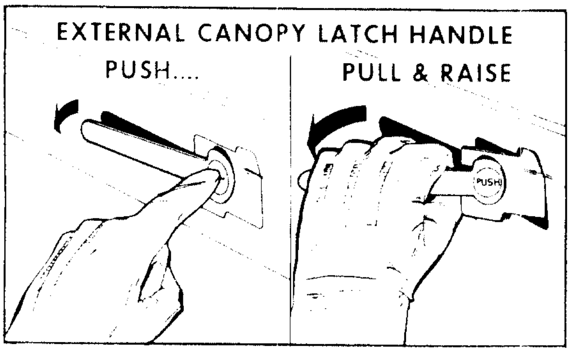
Figure 3-5. (Sheet 4)

Caution Lamp Analysis

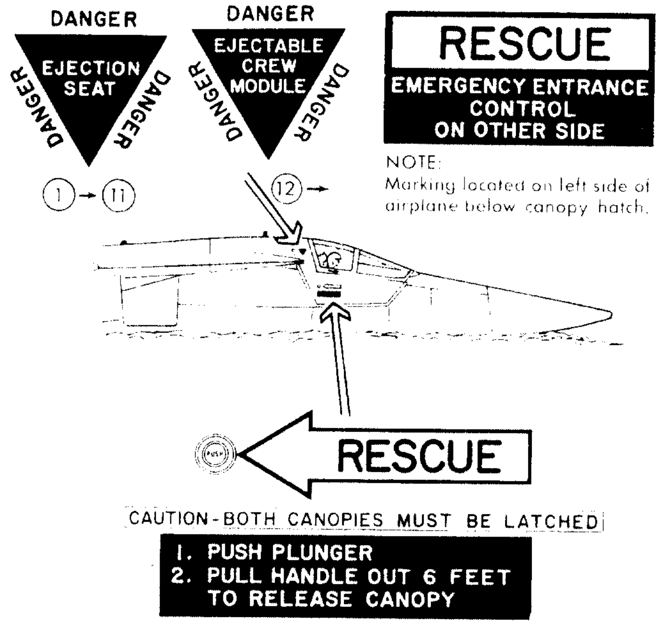
Indicator	Cause	Corrective Action
LOW EQUIP PRESS	Pressure to forward equipment bay pressurized components is less than 12.5 (±0.5) psi	Turn off TFR, Attack Radar, CMRS, and Track-breaker. This equipment requires one-atmosphere pressurization for proper operation
ICING	Icing condition sensed by ice detector	Place pitot heater anti-icing switch to ON. Check engine inlet anti-icing system is operating. If not, go to MANUAL. Lamp will remain on until 60 seconds after icing condition ceases
FWD EQUIP HOT	Low airflow and/or high temperature airflow supplied for equipment cooling	Switch to manual operation and rotate temperature control knob to COOL, or when practicable, increase engine rpm. If lamp remains on, turn off all nonessential equipment until lamp goes out
AFT EQUIP HOT		Not operable
WINDSHIELD HOT	Supply air exceeds 450°F	Place rain removal switch to OFF

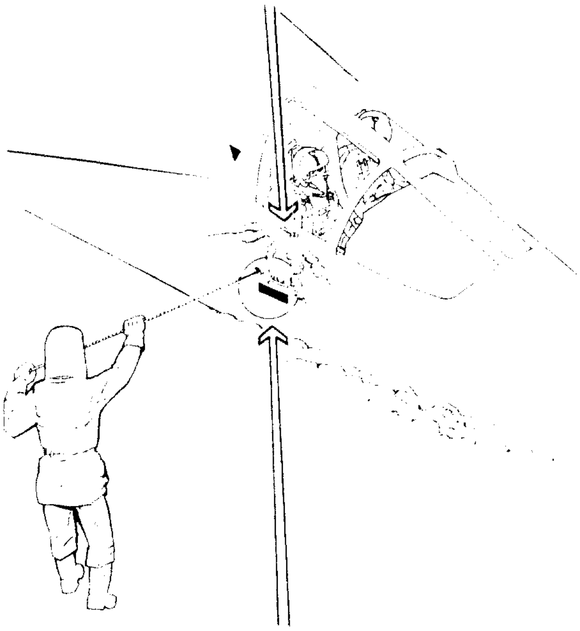
Figure 3-5. (Sheet 5)

Emergency Entrance



NOTE:
The external canopy latch handle may be used only when the internal canopy latch handle is unlocked.





Canopy emergency opening provisions are installed on the right side of the exterior of the airplane below the canopy hatch.

To release both canopy hatch latches:

1. Push plunger to release spring-loaded handle.
2. Grasp handle which is attached to cable and pull out approximately six feet.

WARNING

Do not pull the canopy external emergency release handle unless both canopy hatches are closed and latched. To do so may result in fatal injury to the occupants from debris flying from the canopy sill.

CAUTION

When latch handle is rotated, sill hook which has been severed from cabin sill may drop off when canopy is raised.

NOTE

Actuating emergency release handle will unlock hatches only. Hatches must then be raised manually.

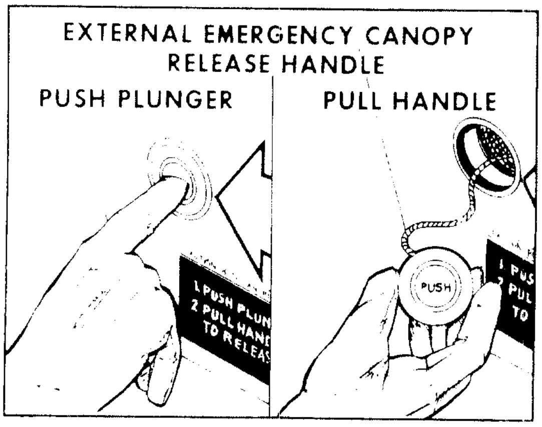


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Figure 3-6.

This is the last page of Section III.

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Section IV
CREW DUTIES

Not applicable to this airplane.

SECTION V

OPERATING LIMITATIONS

TABLE OF CONTENTS.

INTRODUCTION.

This section includes limitations that must be observed for safe and efficient operation of the engines and the airplane. Special attention should be given to the instrument marking illustration (figure 5-1), since these limitations are not necessarily repeated under their respective sections. When necessary, an additional explanation of instrument markings is covered under appropriate headings.

CAUTION

The flight crew will make all necessary entries in Form 781 to indicate when any limitations have been exceeded. Entries shall include the time interval, where applicable, as well as the actual instrument reading value for the limitation that was exceeded.

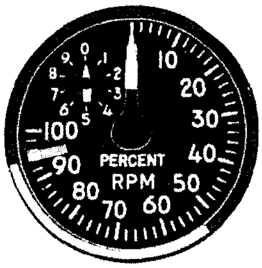
The limitations contained herein, other than those associated with engine ground operation, are applicable for operations within 80% of airplane design limits. Limitations which are more restrictive than those included in this section, and therefore, must be considered for mission planning, are contained in Temporary Flight Limitations Report (FZM-12-922). This document is available through contractors Flight Personnel assigned to the Flight Test Base of Operations.

MINIMUM CREW REQUIREMENTS.

The minimum crew for normal flight is two. The minimum crew for mission completion is two. Certain test missions may be flown with one pilot.

Instrument Markings

BASED ON JP-4 FUEL



NOTE
Tachometer and Turbine Inlet Temperature markings are depicted for TF30-P-1 Engine.

TACHOMETER

YTF30-P-1 ENGINE (with 16th stage compressor modification):

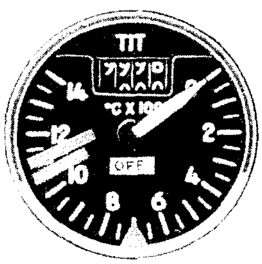
(GREEN) 54 to 93.6 percent – Normal operating range.

(RED) 93.6 percent – Maximum operating speed.

TF30-P-1 ENGINE

(GREEN) 54 to 95.7 percent – Normal operating range.

(RED) 95.7 percent – Maximum operating speed.



TURBINE INLET TEMPERATURE

YTF30-P-1 ENGINE

(GREEN) 300 to 920°C – Normal operating range.

(RED) 540°C – Starting (momentary).

(RED) 1080°C – Maximum Military operation.

(RED) 1100°C – During acceleration (2 minutes) (Engine transients).

TF30-P-1 ENGINE

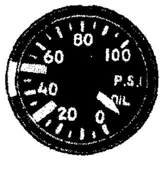
(GREEN) 300 to 1110°C – Normal operating range.

(RED) 705°C – Starting (momentary).

(RED) 1110°C – Maximum Military operation.

(RED) 1130°C – During acceleration (2 minutes) (Engine transients).

OIL PRESSURE

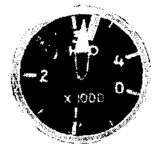


(GREEN) 40 to 50 psi – Normal range.

(RED) 30 psi – Maximum during idle.

(RED) 50 psi – Maximum.

HYDRAULIC PRESSURE



(GREEN) 2950 to 3250 psi – Normal range.

(RED) 3250 psi – Maximum.

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Figure 5-1.

ENGINE LIMITATIONS.

GROUND OPERATION.

Maximum IDLE time is unlimited.

Maximum time at MIL power - 45 minutes.

Maximum time at any A/B power - 3 minutes.

CAUTION

With the engine inlet screens installed, the engines shall not be operated above 85% rpm.

After operating in A/B power for 3 minutes, retard the throttle to 70% rpm for 3 minutes minimum before advancing to any higher power.

YTF30-P-1 Engine - Avoid steady state engine operation between 74.8 percent and 76.5 percent.

Note

With all of the engine doors open or removed, Maximum A/B operating time can be extended to 5 minutes. After 5 minutes of operation, the cooling period is for a minimum of 3 minutes with the throttle retarded to 70% rpm before advancing to a higher power again.

INFLIGHT OPERATION.

Maximum continuous afterburner operation.

YTF30-P-1	15 minutes.
TF30-P-1	45 minutes.

CAUTION

YTF30-P-1 Engine - Avoid steady state engine operation between 74.8 percent and 76.5 percent.

ENGINE ACCELERATION LIMITS.

Refer to figure 5-1.

ENGINE OVERSPEED LIMIT.

Refer to figure 5-1.

ZERO "G" TIME LIMIT.

Zero "G" time limit - 10 seconds.

NEGATIVE "G" TIME LIMIT.

Negative "g" time limit is 10 seconds. Do not initiate a zero or negative "g" maneuver when the fuel low caution lamp is lighted.

Note

The fuel low caution lamp may light during a negative "g" maneuver.

ALTERNATE FUEL.

This information will be supplied when available.

OIL TEMPERATURE LIMITATIONS.

Maximum temperature is 120°C (248°F). ENG OIL HOT caution lamp will light at 121°C (250°F).

STARTER LIMITATIONS.

The starter is limited to 2 cartridge starts in a 15 minute period or 5 consecutive pneumatic starts after which a 1 hour cooling period must be observed. The starter is limited to the following periods of continuous operation after which a 15 minute cooling period must be observed:

Left Starter	10 minutes
Right Starter 1→9, 11→18 .	10 minutes
10 .	2 minutes

AIRSPEED LIMITATIONS.

FLUTTER LIMITATIONS.

Refer to T.O. 1F-111(Y)A-1A.

AIRSPEED AND ALTITUDE OPERATIONAL LIMIT ENVELOPES.

FLAP LIMIT SPEEDS.

Airplanes 1→11

Zero to 15 degrees flap deflection	290 KCAS/mach 0.62 whichever is less
Greater than 15 degrees flap deflection .	190 KCAS/mach 0.47 whichever is less

Airplanes 12, 13, 18

Zero to 30 degrees flap deflection	297 KCAS/mach 0.62 whichever is less

Greater than 30 degrees flap deflection .	225 KCAS/mach 0.48 whichever is less

Airplanes (14)→(17)

Zero to 15 degrees flap deflection	297 KCAS/mach 0.62 whichever is less
15 to 30 degrees flap deflection	250 KCAS/mach 0.62 whichever is less
Greater than 30 degrees flap deflection .	225 KCAS/mach 0.48 whichever is less.

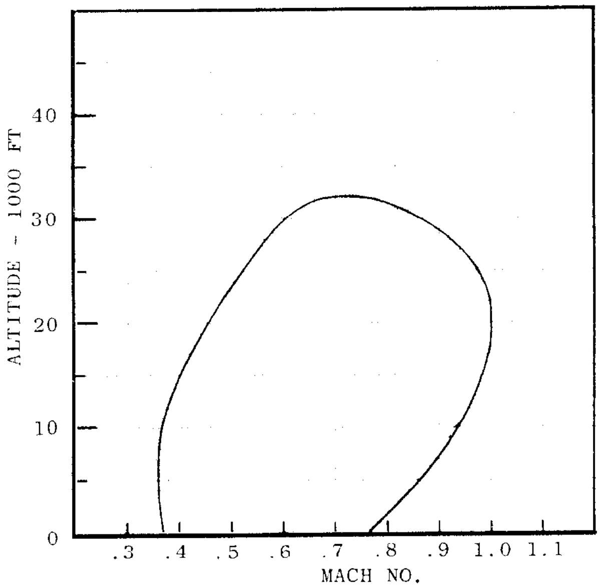
LANDING GEAR OPERATION LIMIT.

The maximum speeds at which the landing gear may be operated are:

Retraction	295 KIAS
Extension	295 KIAS
With landing gear extended . . .	295 KIAS
Emergency extension	160 KIAS

Ram Air Mode Limits

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966 STANDARD DAY



NOTE:
Flight outside the continuous envelope is not recommended.

A0000000-128

Figure 5-2.

TRANSLATING COWL.

Do not exceed 417 KIAS or mach .90, whichever is less, with the translating cowl in any position other than fully closed.

RAM AIR MODE LIMIT SPEED.

Maximum speeds for ram air operation at various altitudes are shown in figure 5-2. Operation outside the continuous envelope is not recommended because there is inadequate air flow for cooling at the lower airspeeds and cabin temperature will be too warm at the higher airspeeds.

WARNING

Do not operate RAM Air Mode above 330 KIAS. Structural failure of the ground cooling service air door may occur.

TIRE LIMIT SPEED (ZERO WIND).

Maximum tire takeoff speed . . .	157 KIAS
Emergency landing maximum tire speed	200 KIAS

Note

Tail wind component must be subtracted from the zero wind tire limit speed. The headwind component must be added to the zero wind tire limit speed.

TAXI SPEED.

Maximum taxi speed . . . . 25 knots

TAIL HOOK ENGAGEMENT SPEED.

For maximum tailhook engaging speed, see figure 5-3.

MINIMUM FLYING SPEEDS.

The low mach number, minimum flying speeds are presented in figure 5-4 as a function of gross weight for lg level flight. Flight at speeds below the recommended minimum flying speeds is prohibited until flight tests determine the level of stall and/or buffet warning, and spin entry and recovery techniques have been completed. It is strongly recommended that an intermediate flap setting of approximately 15 degrees be used between 225 and 250 KIAS. This will allow the clean airplane to attain an airspeed which will assure an angle of attack well below stall when the flaps are fully retracted.

Maximum Arresting Hook Engaging Speed For Bak 12 Barrier

DATA BASIS: ESTIMATED

DATE: 11 NOVEMBER 1966

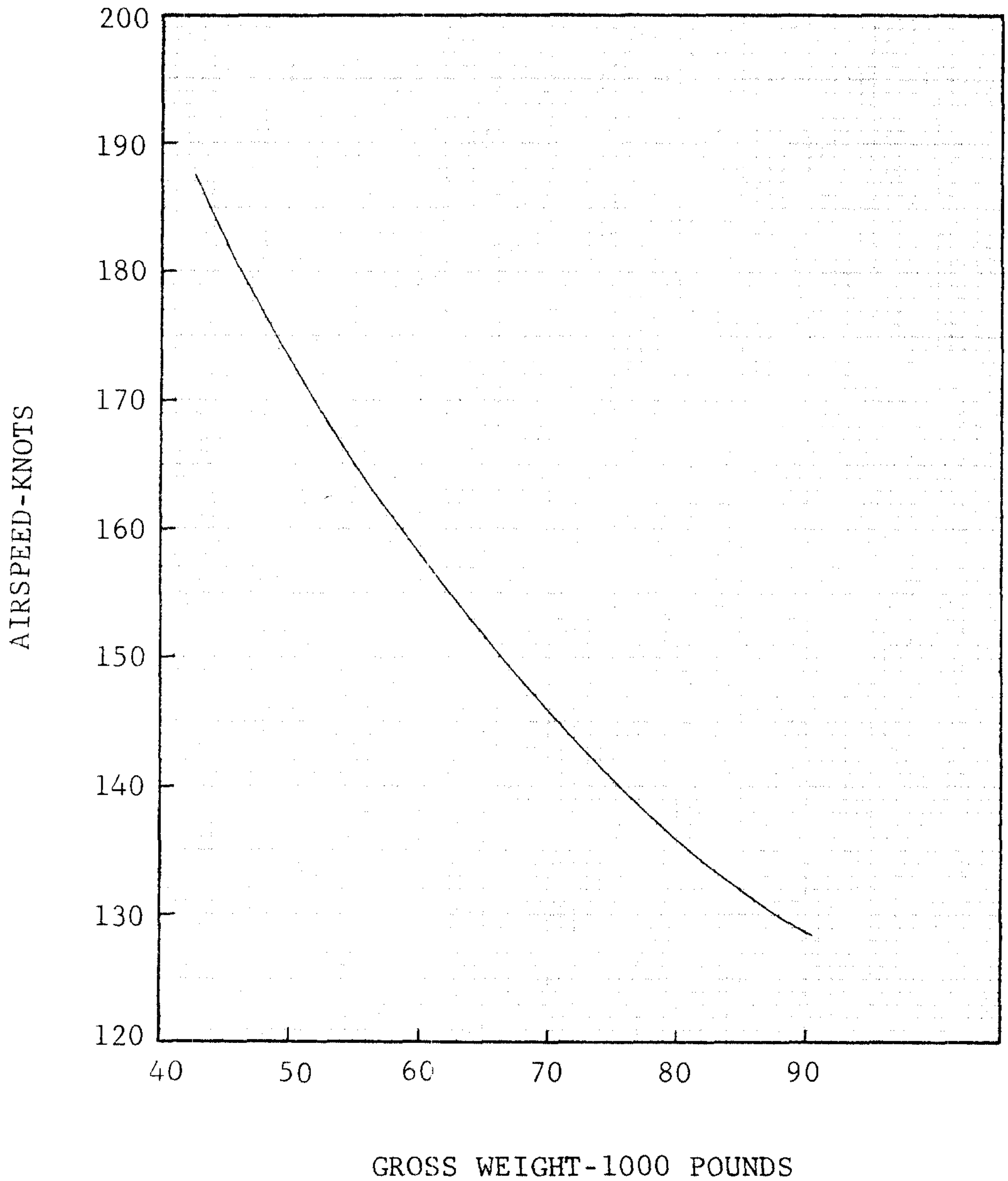


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Figure 5-3.

Minimum Flying Speed-Takeoff and Landing

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966

MACH = 0 to 0.3
WING SWEEP = 16°
FLAPS = 35° - 37.5°

FUEL GRADE: JP-4
ENGINES: Y & TF 30-P-1

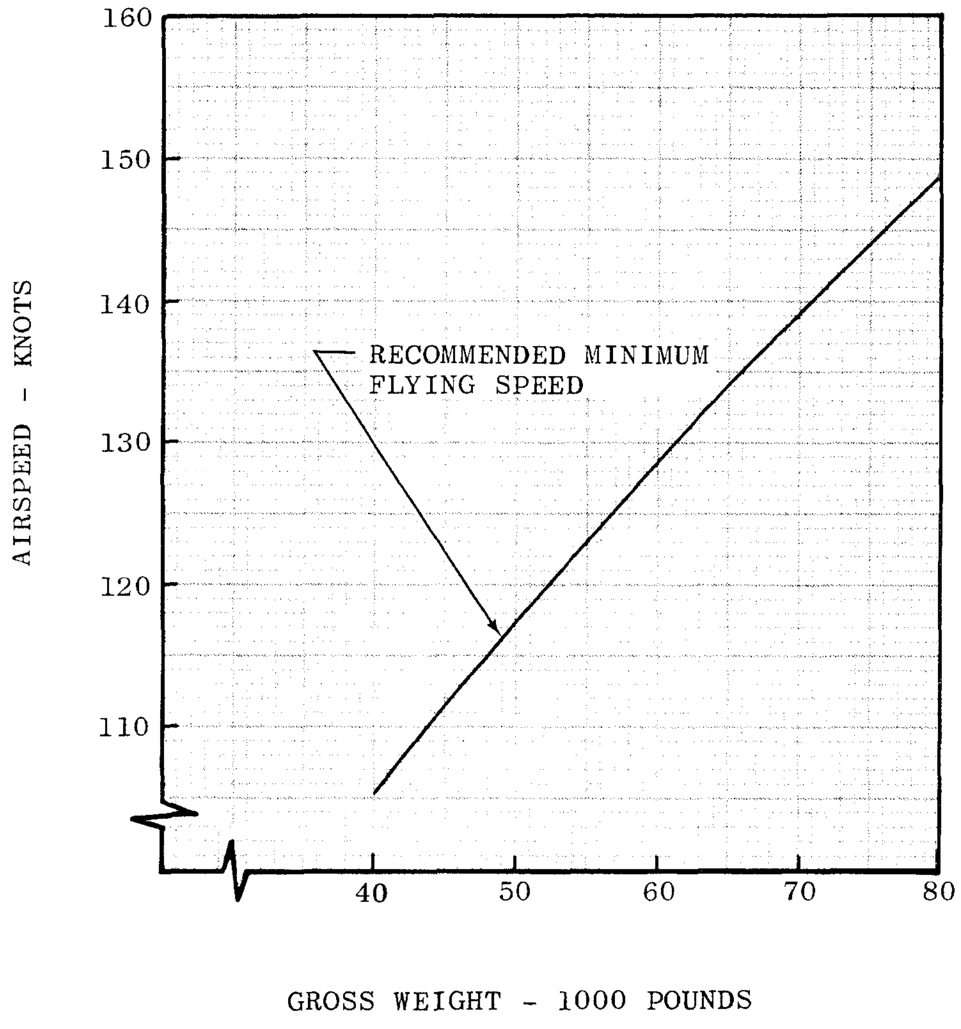

Figure 5-4. 1 → 11 *(Sheet 1)*

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Minimum Flying Speed-Takeoff And Landing Configuration

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966

MACH = 0 TO 0.3
WING SWEEP = 26°
FLAPS = 35° - 37.5°

FUEL GRADE: JP-4
ENGINES: Y & TF 30-P-1

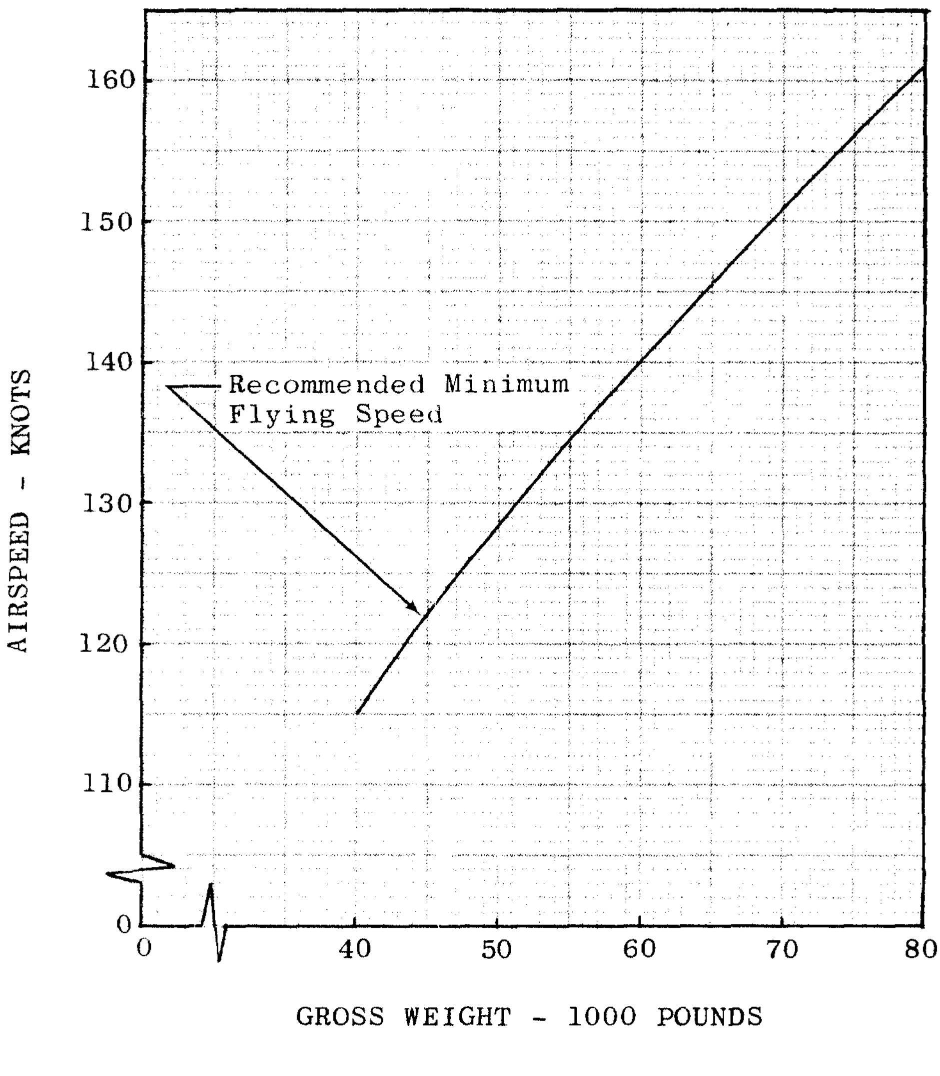


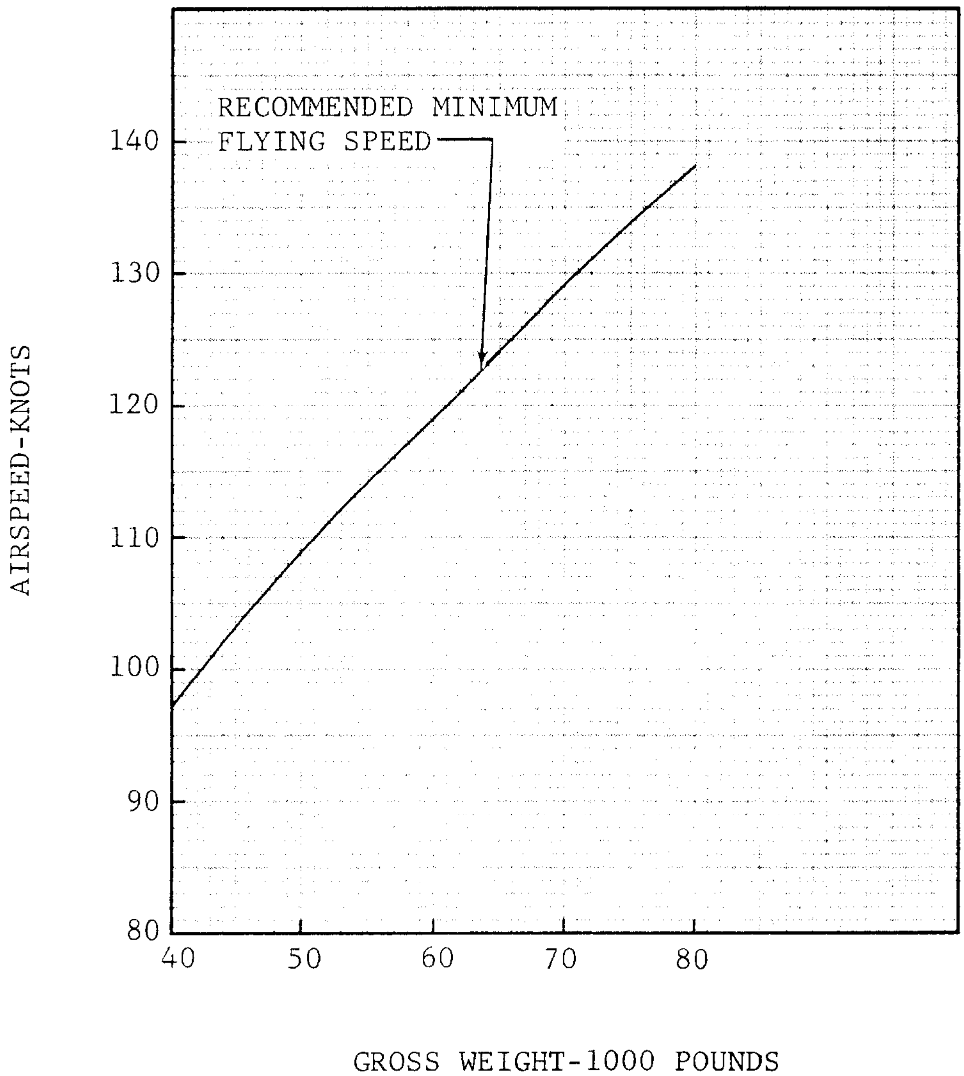
Figure 5-4. ①→⑪ *(Sheet 2)*

Minimum Flying Speed-Takeoff And Landing

MACH=0 TO 0.3

DATA BASIS: ESTIMATED | WING SWEEP=16° | FUEL GRADE: JP-4

DATE: 11 NOVEMBER 1966 | FLAPS=35° TO 37.5° | ENGINES:Y&TF30-P-1



A0000000-033

Figure 5-4. (Sheet 3)

Minimum Flying Speed-Takeoff and Landing

DATA BASIS: ESTIMATED

DATE: 11 NOVEMBER 1966

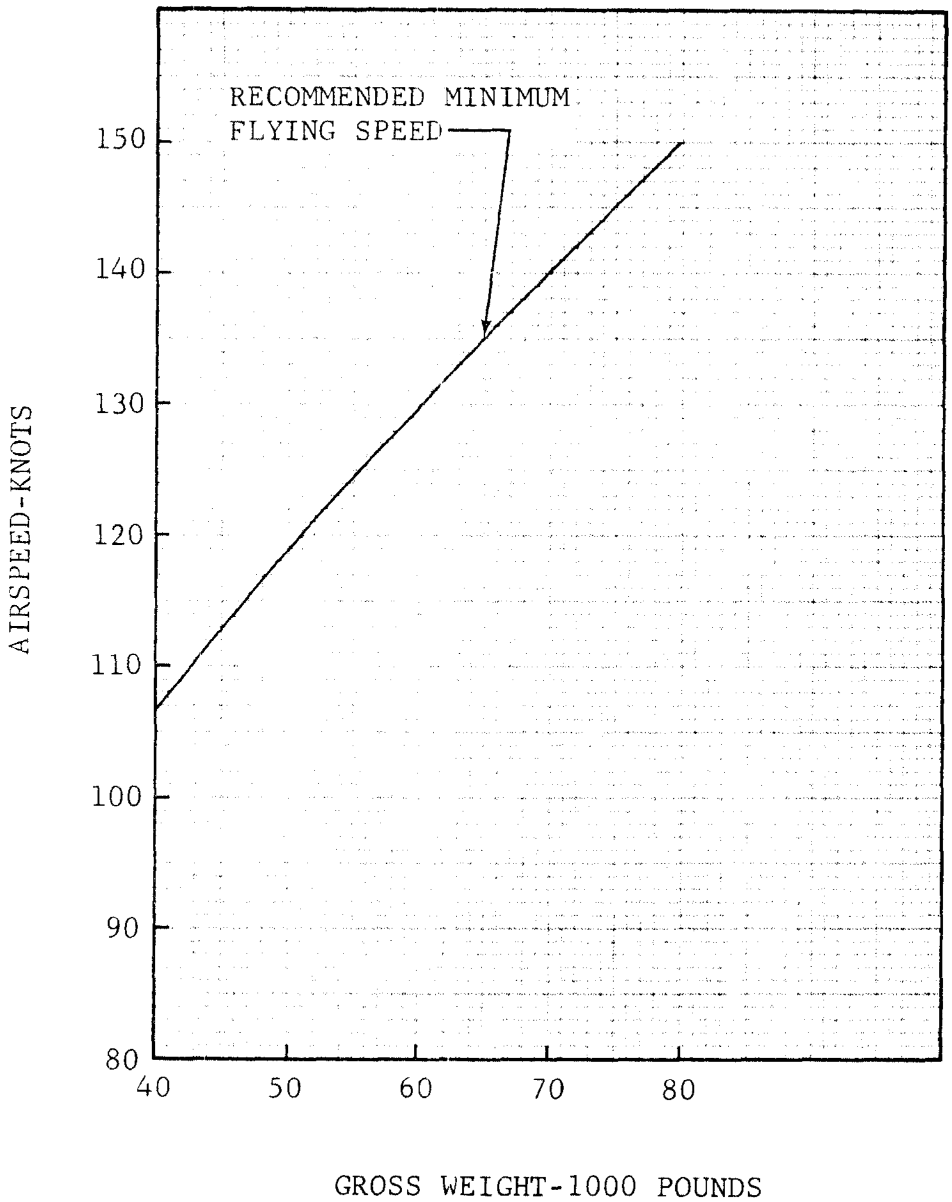
MACH=0 TO 0.3

WING SWEEP=26°

FLAPS=35° TO 37.5°

FUEL GRADE: JP-4

ENGINES:Y&TF30-P-1



A0000000-034

Figure 5-4. (Sheet 4)

Minimum Flying Speed-Takeoff And Landing

CLEAN AIRPLANE

DATA BASIS: ESTIMATED

MACH=0 TO 0.3

FUEL GRADE: JP-4

DATE: 11 NOVEMBER 1966

WING SWEEP=16°

ENGINES:Y&TF30-P-1

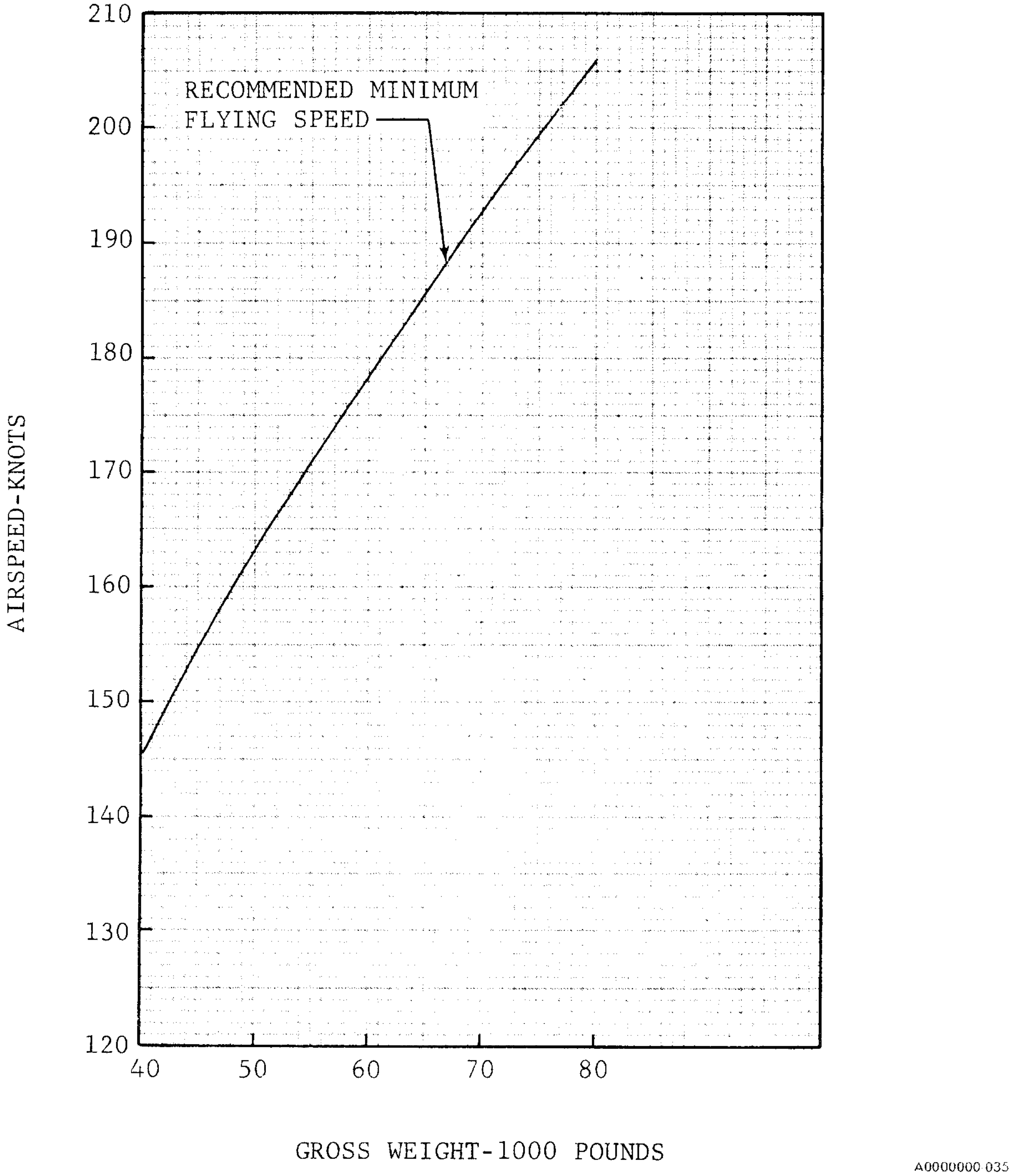


Figure 5-4. (Sheet 5)

Minimum Flying Speed-Takeoff And Landing

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966

CLEAN AIRPLANE
MACH=0 TO 0.3
WING SWEEP=26°

FUEL GRADE: JP-4
ENGINES: Y&TF30-P-1

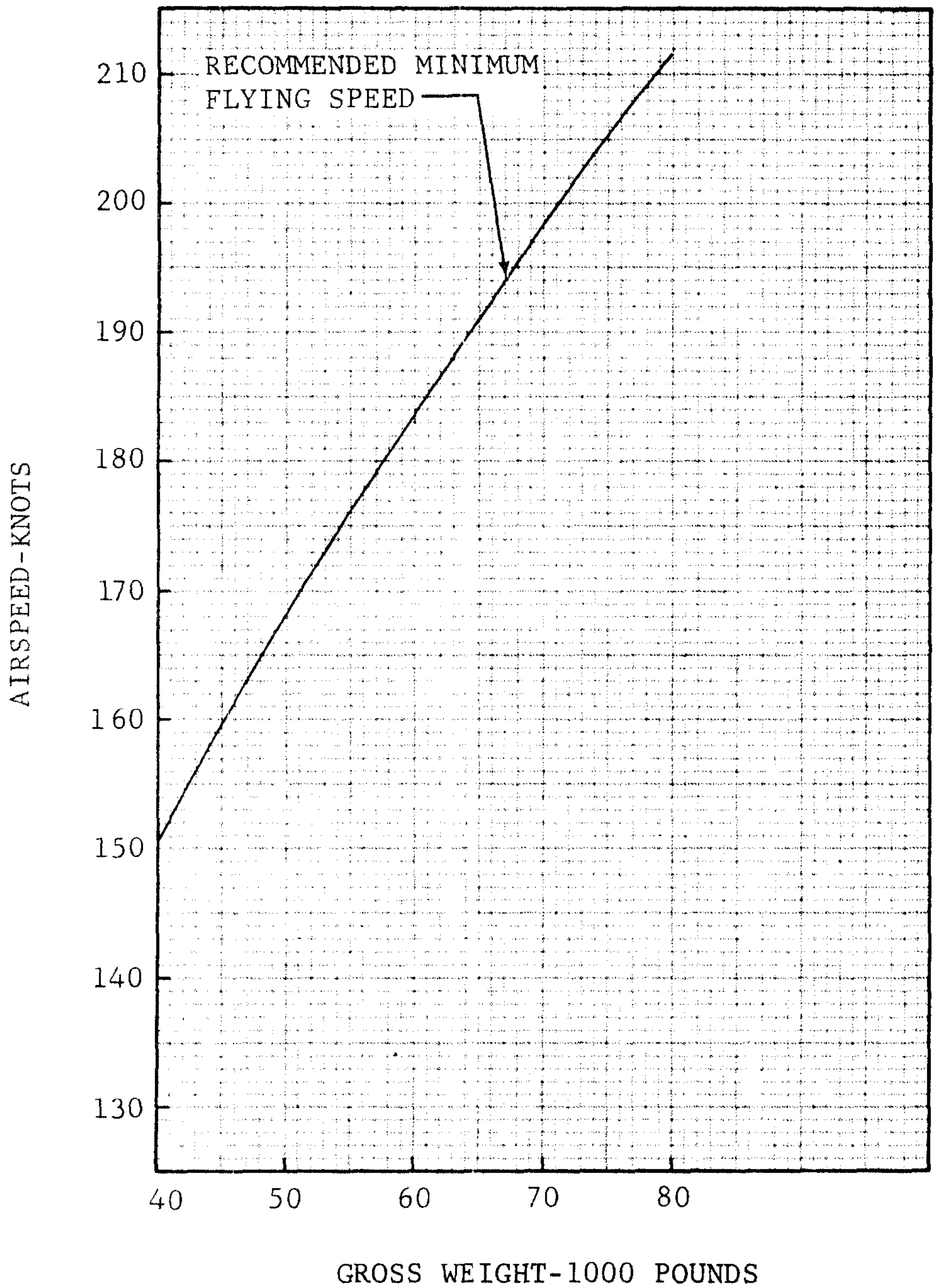


Figure 5-4. (Sheet 6)

Minimum Flying Speed-Takeoff And Landing

CLEAN AIRPLANE

DATA BASIS: ESTIMATED
MACH=0 TO 0.3
FUEL GRADE: JP-4

DATE: 11 NOVEMBER 1966
WING SWEEP=35°
ENGINES: Y&TF30-P-1

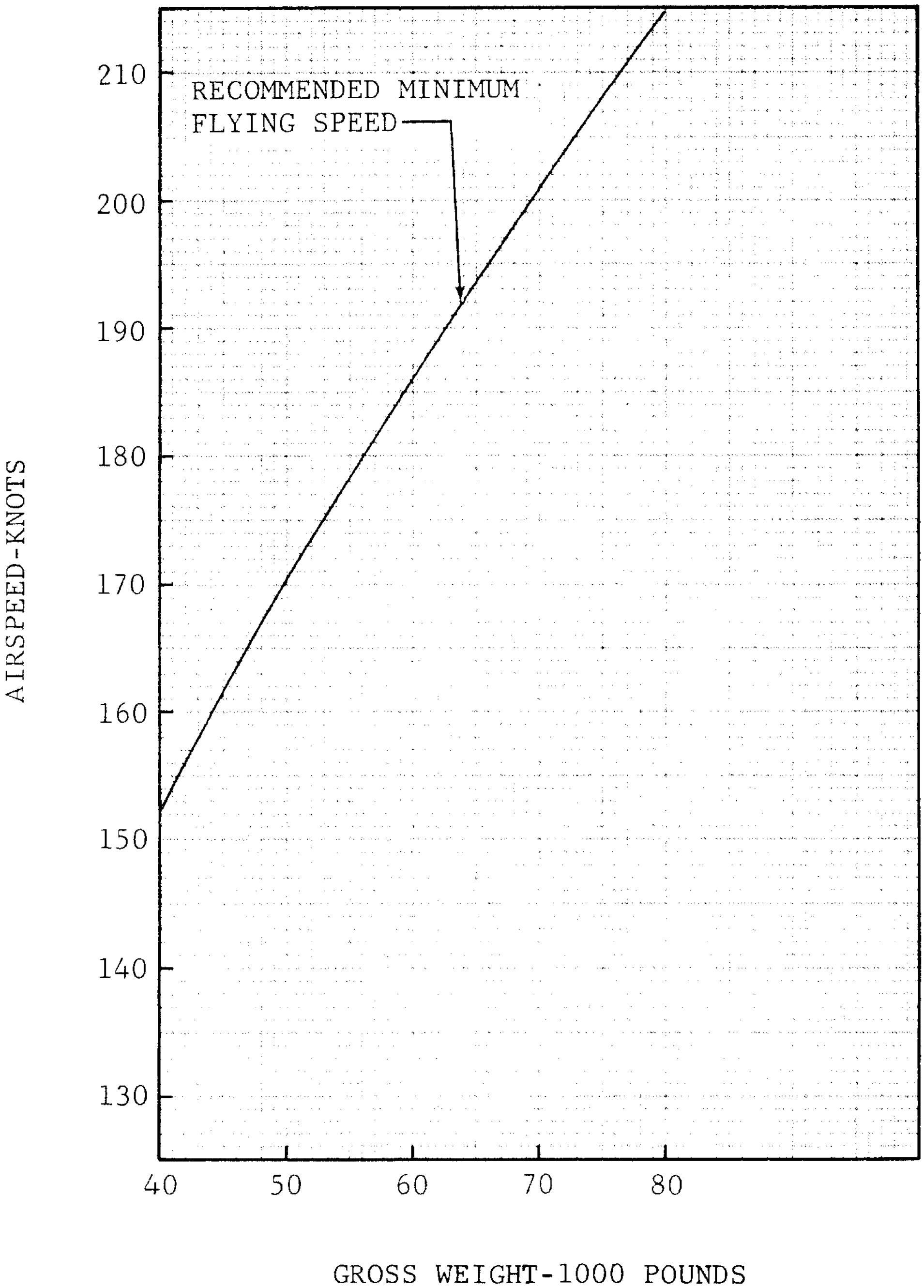


Figure 5-4. (Sheet 7)

Minimum Flying Speed-Takeoff And Landing

CLEAN AIRPLANE

DATA BASIS ESTIMATED | MACH=0 TO 0.4 | FUEL GRADE: JP-4

DATE: 11 NOVEMBER 1966 | WING SWEEP=50^{O} | ENGINES: Y&TF30-P-1

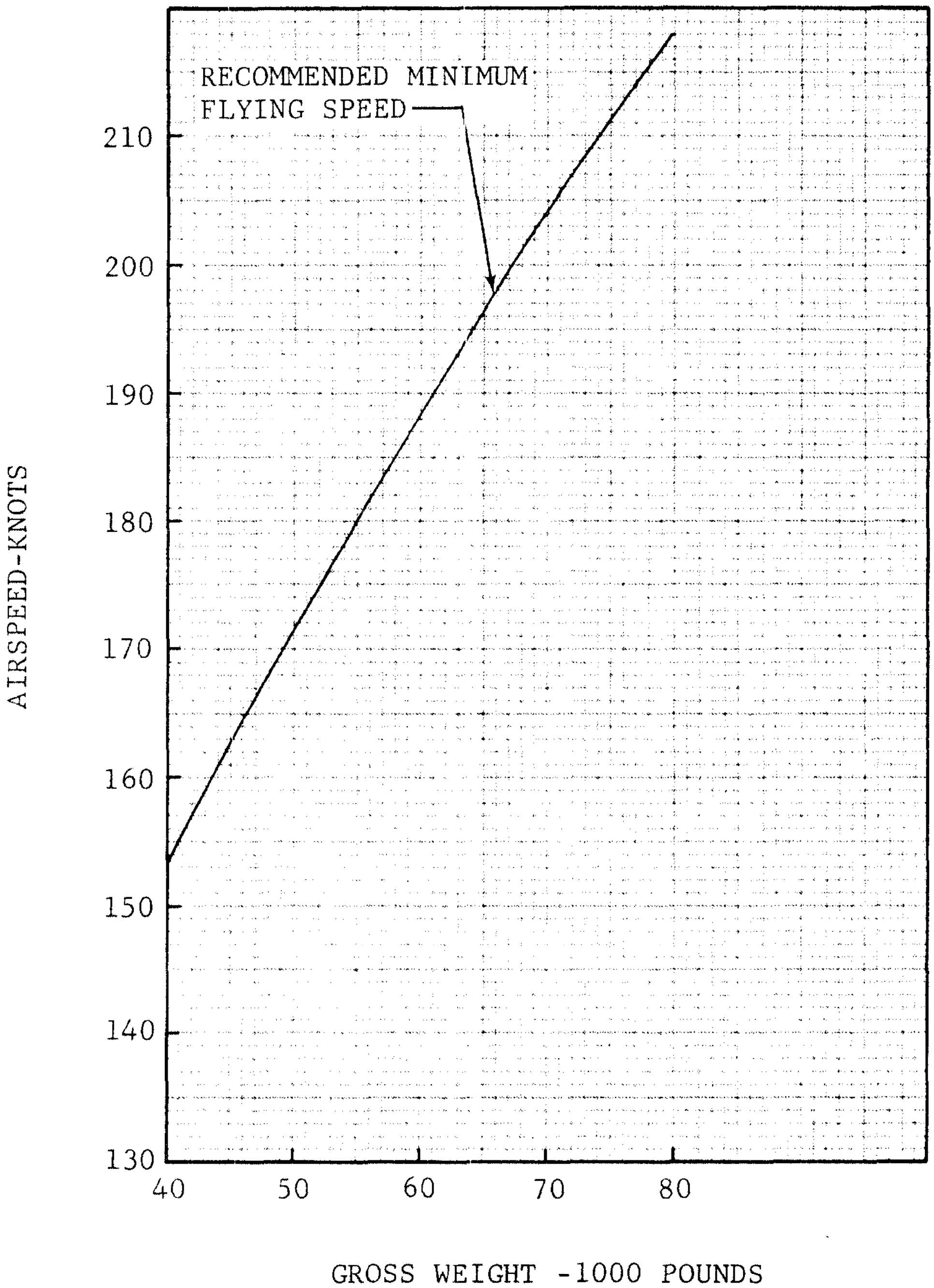


Figure 5-4. (Sheet 8)

Minimum Flying Speed-Takeoff And Landing

CLEAN AIRPLANE

DATA BASIS: ESTIMATED

MACH=0 TO 0.4

FUEL GRADE: JP-4

DATE: 11 NOVEMBER 1966

WING SWEEP=72.5°

ENGINES: Y&TF30-P-1

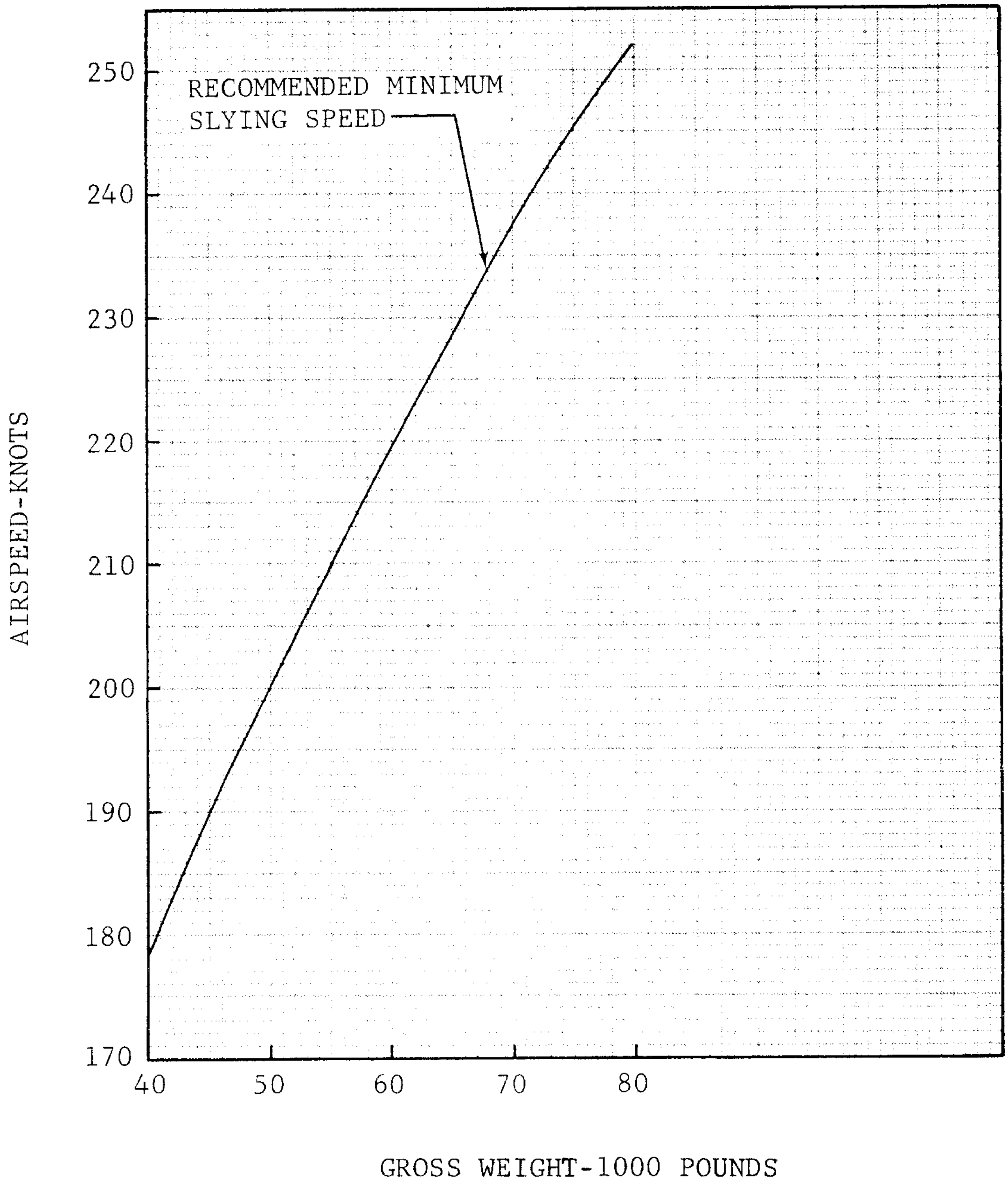


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Figure 5-4. (Sheet 9)

MINIMUM FLYING SPEEDS FOR GENERATOR CONSTANT SPEED DRIVE OIL COOLING.

Minimum airspeeds/altitudes for constant speed drive oil cooling for continuous single alternator operation are as follows:

250 KIAS - 20,000 feet and above.

200 KIAS - below 20,000 feet.

Note

Flight below minimum speeds is permitted for time not to exceed five minutes to accomplish required maneuvers.

MANEUVERABILITY LIMITATIONS.

LONGITUDINAL LIMITATIONS.

Refer to T.O. 1F-111(Y)A-1A.

ROLL LIMITATIONS.

The following roll limitations are based on 80 percent limit strength values. It should be noted that full normal lateral stick deflection corresponds to a ±2 differential horizontal stabilizer command which is indicated by a force detent.

At Wing Sweep Angles Where Spoilers Are Operational. (Sweep Angles of 45 Degrees or Less).

Do not exceed 1/2 normal lateral stick deflection at speeds greater than 450 KIAS at any altitude.

With fuel in wings, do not exceed 1/2 normal lateral stick deflection at any mach number at any altitude.

Limitations with external stores will be incorporated when available.

At Wing Sweep Angles Where Spoilers Are Not Operational. (Sweep Angles Greater Than 45 Degrees).

At altitudes less than 25,000 feet, do not exceed 1/2 normal lateral stick deflection at speeds greater than 525 KIAS.

At All Wing Sweep Angles.

Do not exceed full normal lateral stick deflection at any mach number at any altitude except under emergency conditions requiring more than normal lateral control.

YAW LIMITATIONS.

The following yaw limitations are based on 80 percent limit strength values.

Yaw Limitations in the Takeoff and Landing Configuration.

In the takeoff and landing configuration, do not exceed the allowable rudder commands shown in figure 5-5.

Yaw Limitations in Other Than Takeoff and Landing Configuration.

Do not exceed ±7.5 degrees rudder command at any mach number at any altitude except under emergency conditions requiring more than normal directional control.

ACCELERATION LIMITATIONS.

LIMIT MANEUVER LOAD FACTORS.

Refer to T.O. 1F-111(Y)A-1A.

CENTER-OF-GRAVITY LIMITATIONS.

Center-of-gravity limits are presented as a function of wing sweep angle and are applicable for all gross weights. Limits for the takeoff and landing configuration are shown in figure 5-6. The aft limit shown was set to maintain a static margin of at least one percent. Forward limits are based on nose gear unstick for a full trailing edge up horizontal tail deflection at airplane lift off speed. Since thrust has a large effect on nose gear unstick, limits are presented for military and maximum A/B thrust. The above limits are applicable to flap limit speeds. The low speed limits for the clean airplane (flaps and slats retracted) are also shown in figure 5-6. Aft limit shown was set to maintain a static margin of at least one percent. Limits are applicable up to flap limits speeds. High speed limits for the clean airplane are presented in figure 5-6. Aft limit shown for wing sweep angles of 16 degrees through 35 degrees is based on maintaining a one percent static margin at .8 mach number at sea level. Aft limit for wing sweep angles of 50 through 72.5 degrees is based on maintaining a minimum level of directional stability. The aft center of gravity limits at which the airplane will tip over at brake release with maximum afterburner thrust are presented in figure 5-7 as a function of gross weight and center of gravity location. These limits are well aft of the aft center of gravity location. These limits are well aft of the aft center of gravity for normal loadings. However, abnormal loadings could result in an aft center of gravity which would cause the airplane to tip over at brake release with afterburner thrust.

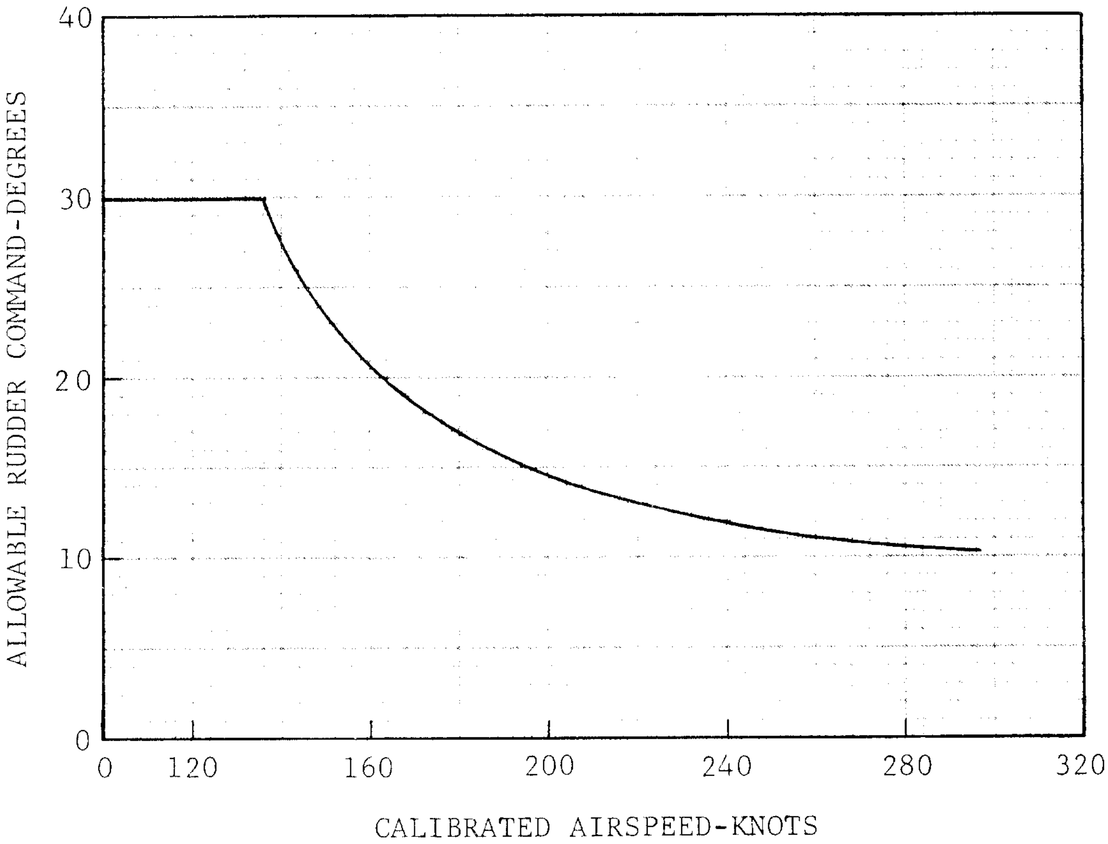
Allowable Rudder Command Takeoff And Landing Configuration

DATA BASIS: ESTIMATED

DATE: 11 NOVEMBER 1966

FUEL GRADE: JP-4

ENGINES: Y&TF30-P-1



A00000UU-045

Figure 5-5.

Center Of Gravity Restrictions

DATA BASIS: ESTIMATED
DATE: 20 NOVEMBER 1964

FLAP AND GEAR: DOWN
NO EXTERNAL STORES

FUEL GRADE: JP4
ENGINES: Y & TF 30-P-1

APPLICABLE TO FLAP LIMIT SPEED

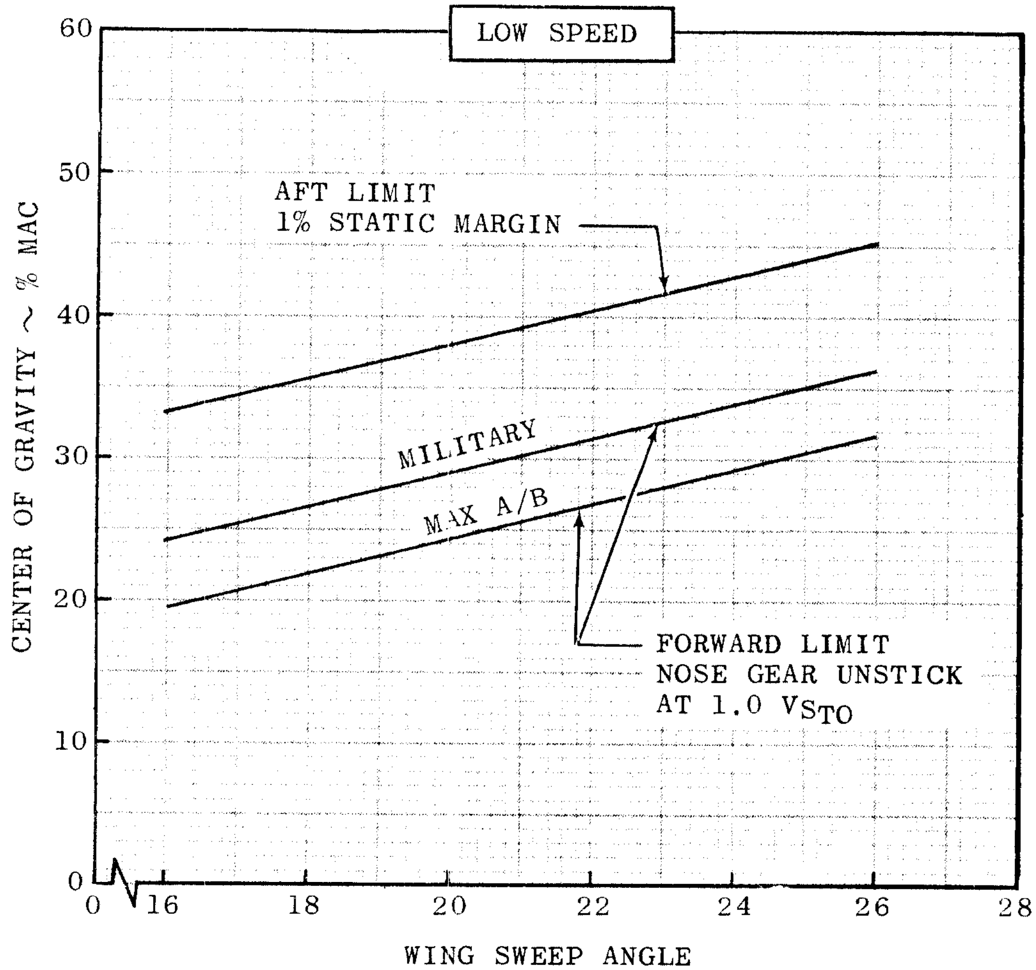


Figure 5-6. (1)→(11) *(Sheet 1)*

A0000000-040

Center Of Gravity Restrictions

FLAPS AND GEAR: DOWN

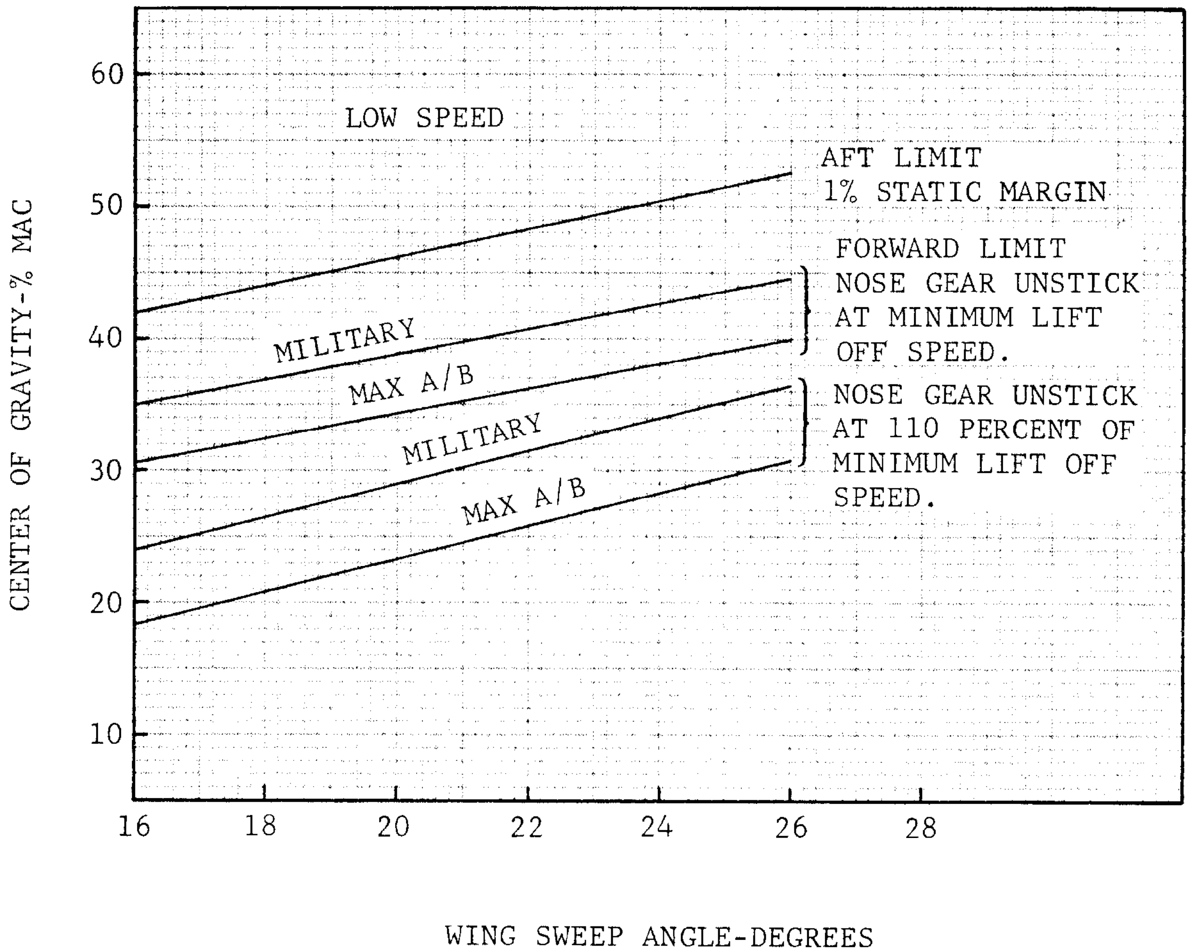
NO EXTERNAL STORES

DATA BASIS: ESTIMATED

DATE: 11 NOVEMBER 1966

FUEL GRADE: JP-4

ENGINES: Y&TF30-P-1



A0000000-041

Figure 5-6. 12 → 18 *(Sheet 2)*

Center Of Gravity Restrictions

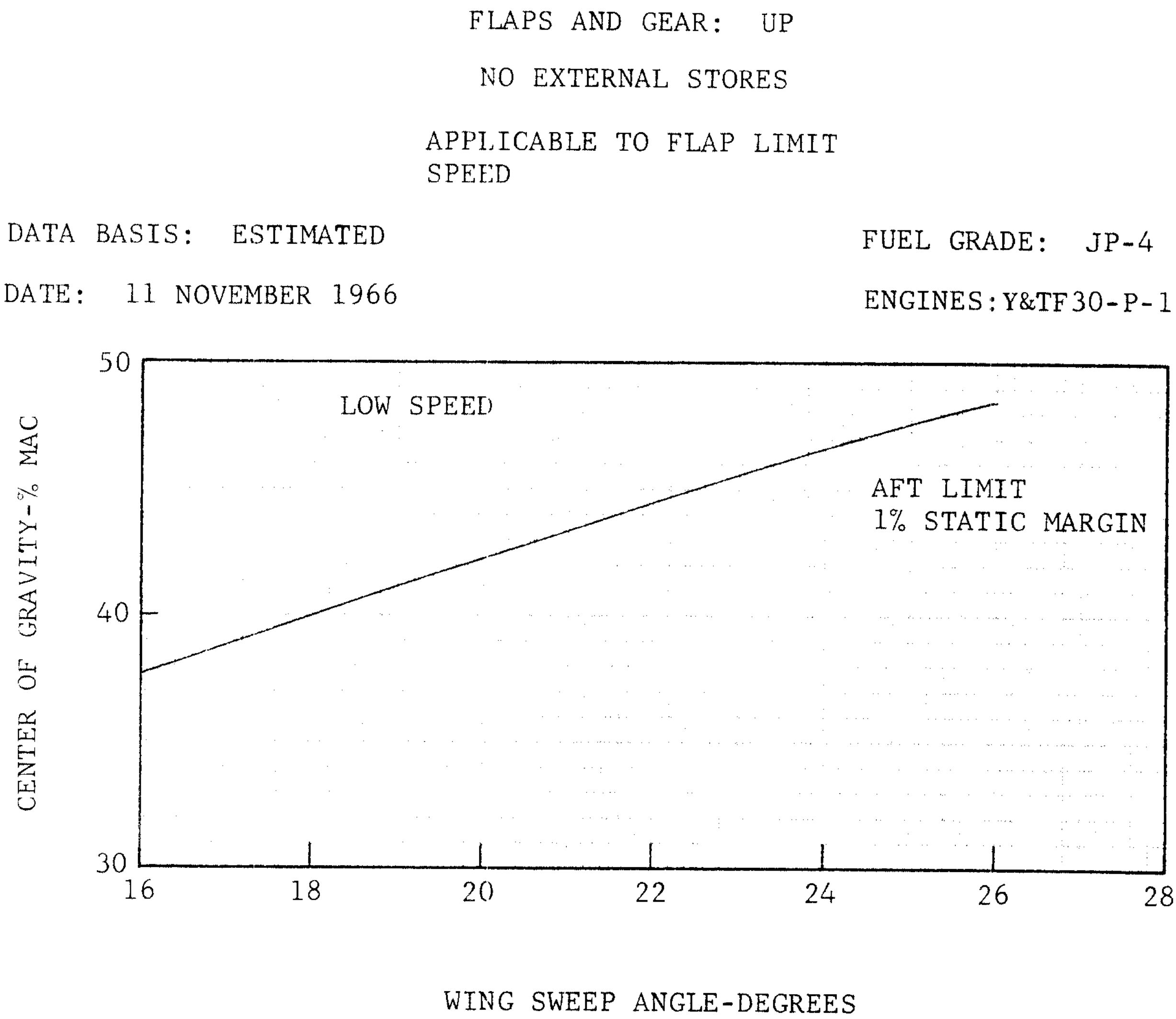


Figure 5-6. (Sheet 3)

A0000000-042

Center Of Gravity Restrictions

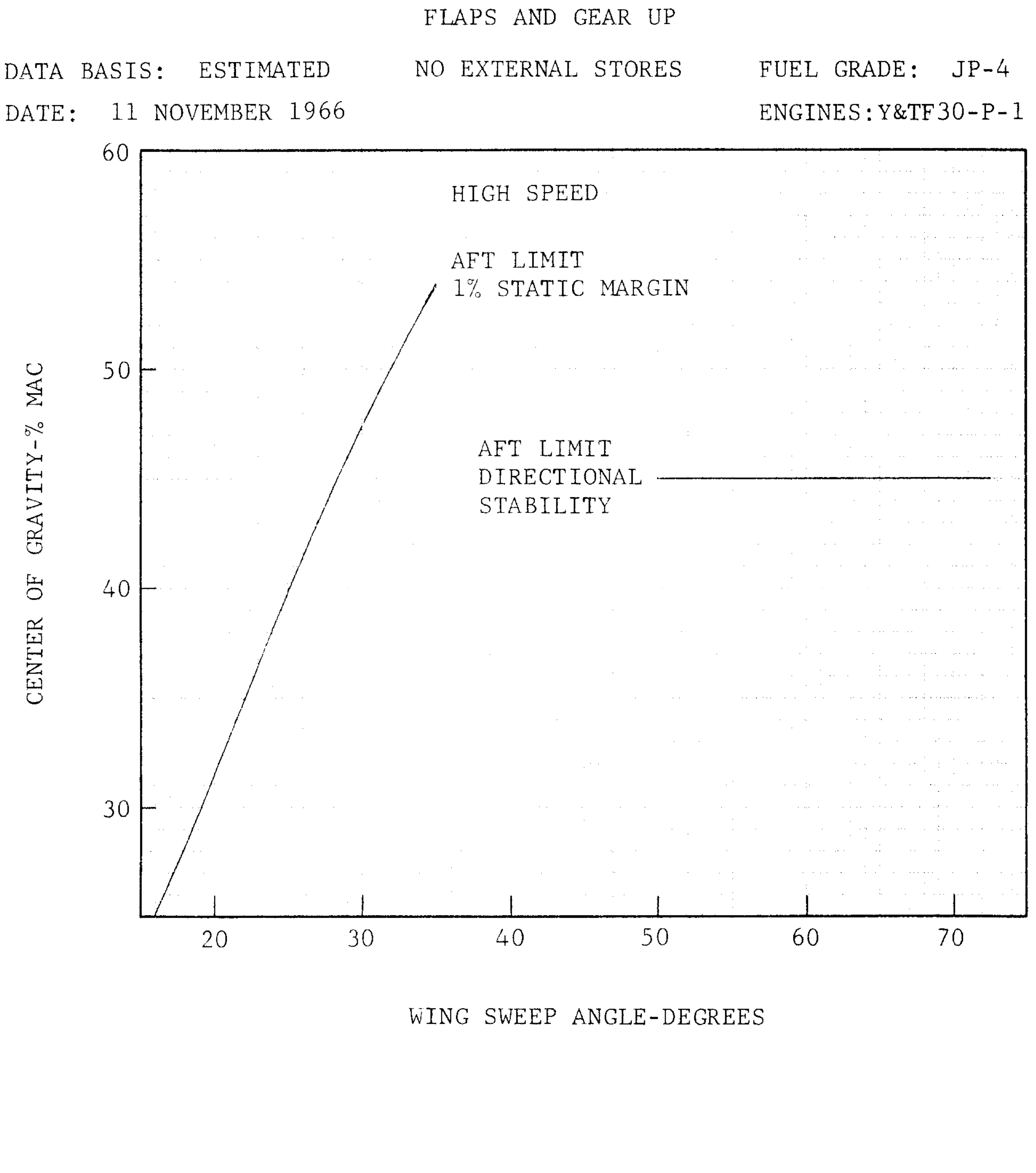


Figure 5-6. (Sheet 4)

Airplane Upset Limit At Brake Release

MAXIMUM A/B THRUST

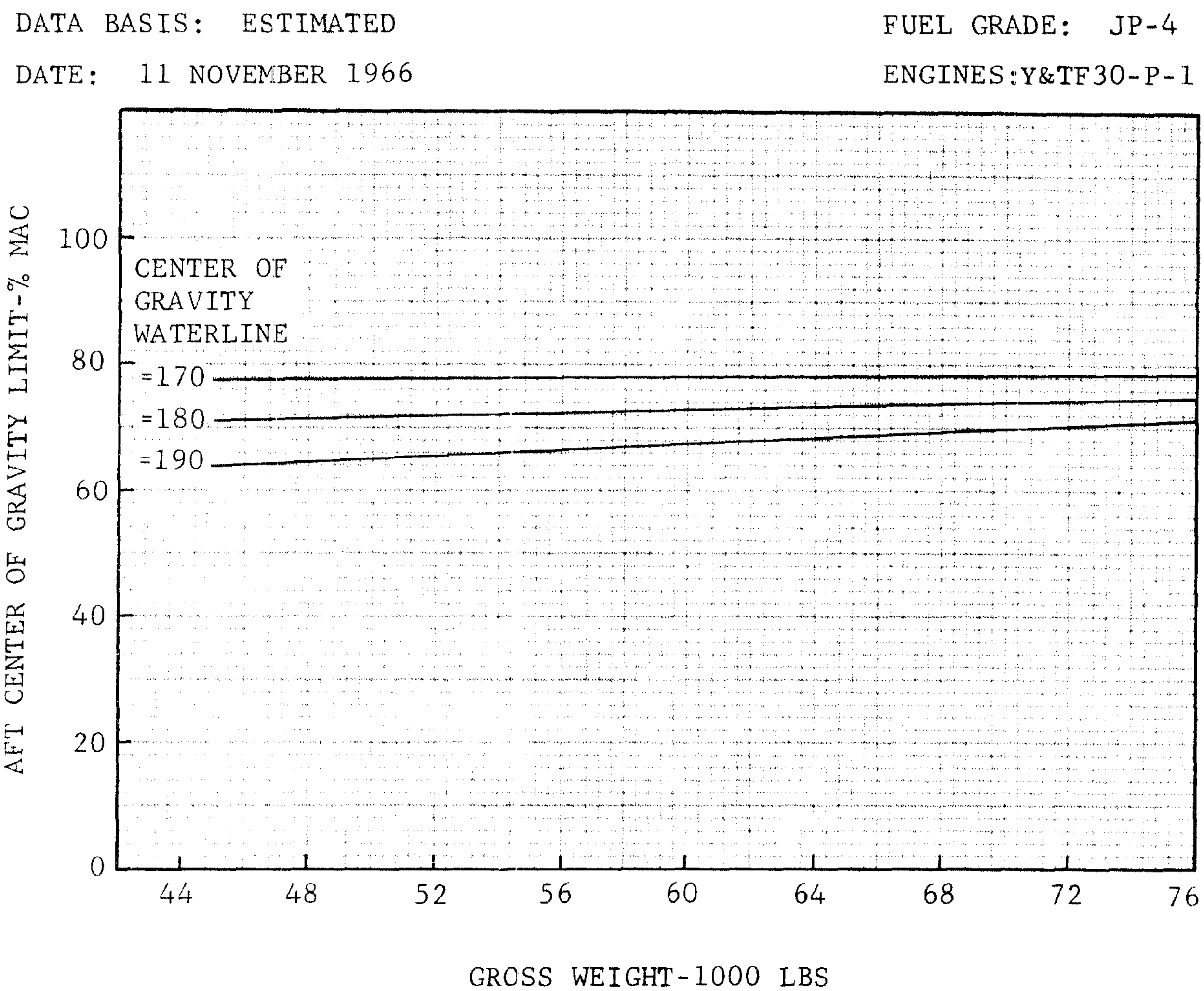


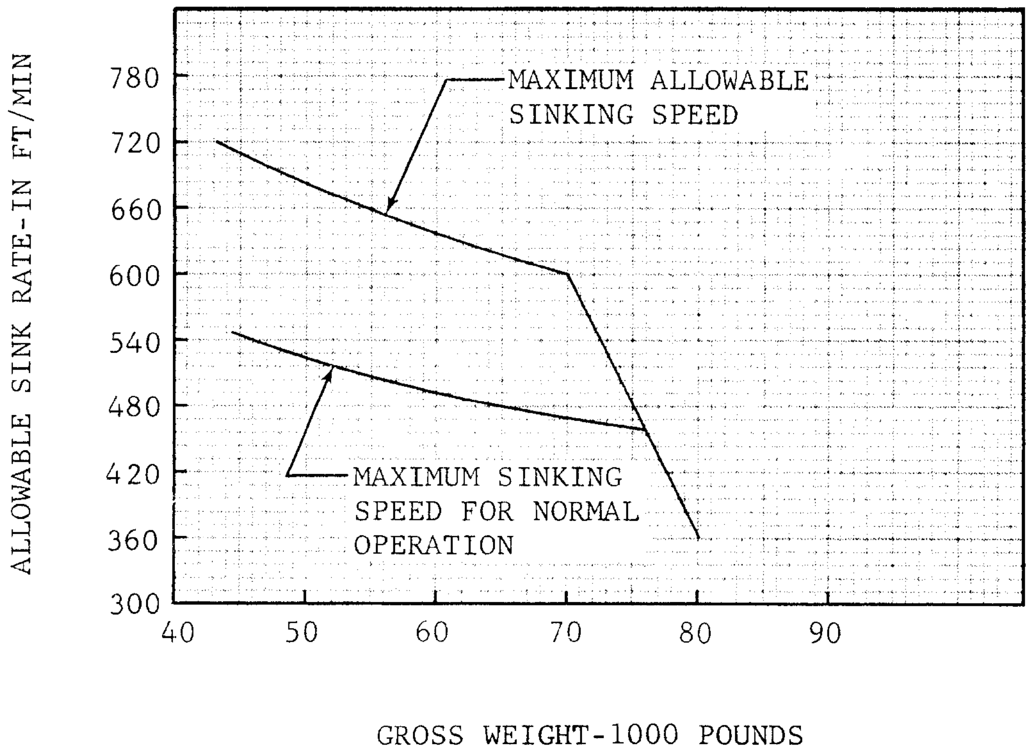
Figure 5-7.

A0000000-044

Allowable Sink Rate At Touchdown

DATA BASIS: ESTIMATED

DATE: 11 NOVEMBER 1966



NOTE:
For airplanes 1 through 11, the maximum allowable sinking speed will be 360 Ft/Min for landings in which the outboard fixed pylon store weight is in excess of 1500 lbs.

A0000000-030

Figure 5-8.

GROSS WEIGHT – CENTER-OF-GRAVITY LIMITATIONS FOR TAXI AND LANDING.

Refer to T.O. 1F-111(Y)A-1A.

CENTER-OF-GRAVITY LOADING AND CAPABILITIES.

Refer to T.O. 1F-111(Y)A-1A.

GROSS WEIGHT LIMITATIONS.

For gross weight limitations other than those listed here, refer to T.O. 1F-111(Y)A-1A.

TAXI AND TAKEOFF GROSS WEIGHT.

Taxi and takeoff operations at weights above 72,000 pounds shall be confined to well prepared runways until completion of structural certification tests.

MAXIMUM LANDING GROSS WEIGHT.

The allowable sink rate at touchdown as a function of gross weight is shown in figure 5-8.

FLIGHT CONTROL SYSTEM LIMITATIONS.

STABILITY AUGMENTATION SYSTEM OFF OPERATING LIMITS.

Figure 5-9 presents a partial listing of stability augmentation system off operating limits provided as a ready rapid reference. For a complete discussion and detailed charts presenting the entire stability augmentation system off operating envelopes, refer to "Flight Control System Limitations", Section V, T.O. 1F-111(Y)A-1A.

BRAKE LIMITATIONS.

BRAKE APPLICATION SPEED LIMIT.

Brake energy limits are presented in figure 5-10. The example lines on figure 5-10 explain how to determine the amount of energy absorbed by the brakes during a stop.

Example: Full Stop Landing.

Given: Gross weight = 60,000 pounds.
Airspeed when brakes applied = 100 knots IAS.
Pressure Altitude = 1000 feet.
Outside air temperature = 80 degrees F.

Find: Brake Energy Absorbed.

Solution:

Following example lines on figure 5-10 the brake energy absorbed is 13.4 million foot-pounds per brake.

Stability Augmentation System Off Operating Limits

DO NOT EXCEED THE FOLLOWING AIRSPEEDS/ALTITUDES:

PITCH DAMPER OFF		
Wing Sweep	Altitude	Airspeed
26 Degrees	Sea Level to 20,000 feet 20,000 to 35,000 feet Above 35,000 feet	M.57 M.70 M.80
35 Degrees	Sea Level to 10,000 feet 10,000 to 15,000 feet Above 15,000 feet	M.60 M.80 Limit Speed
50 Degrees	Sea Level to 20,000 feet Above 20,000 feet	M.70 Limit Speed
72.5 Degrees	Sea Level to 17,500 feet Above 17,500 feet	M.70 Limit Speed
ROLL OR YAW DAMPER OFF		
Wing Sweep	Altitude	Airspeed
26 and 35 Degrees	None	None
50 and 72.5 Degrees	None	M1.5

Figure 5-9.

CAUTION

If maximum braking capacity is utilized (danger zone), wheel blowout plugs will relieve tire pressure within 3 to 15 minutes after the stop. Provisions should be made to cope with possible wheel fires which may start shortly after blowout plug release.

Note

When figure 5-10 is used to determine values of brake energy absorbed, it is assumed that the airplane is brought to a complete stop with a single continuous application of the brakes.

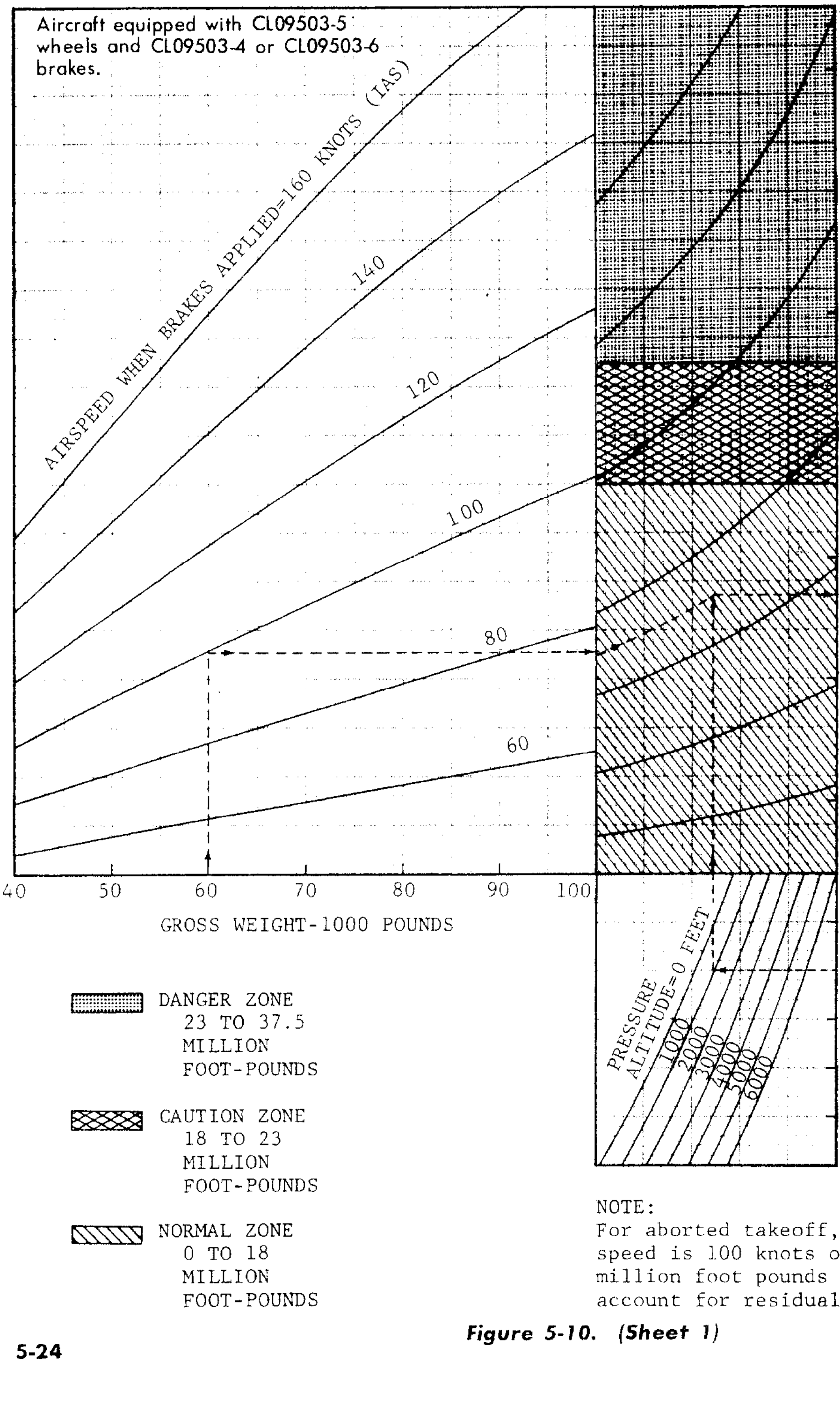
Brake Energy Limits

SLATS-FLAPS-SPOILERS EXTENDED

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966

FUEL GRADE: JP4
ENGINES: Y&TF30-P-1

The following information explains action to be taken when a stop in the DANGER, CAUTION, or NORMAL ZONES is performed:



NOTE:
For aborted takeoff, if brake application speed is 100 knots or above, add four million foot pounds to charted value to account for residual engine thrust.

DANGER ZONE

1. Use moderate braking below 25 knots until taxi speed of 5–10 knots is obtained. Release brakes, if possible, and maintain forward motion.

CAUTION

Applying maximum brake pressure below 25 knots may cause brake rotors and stators to fuse together.

2. Proceed to the nearest parking area clear of other airplanes and personnel without stopping and as quickly as possible. Do not set parking brakes.

3. Request fire fighting equipment. Hydraulic fluid fire is imminent. Approach main landing gear from front or rear for fire fighting purposes only.

4. Extinguishing agents shall be applied as a fog or foam on the tires and directly to the brakes. Do not spray liquid directly on the wheels.

CAUTION ZONE

1. The area in the vicinity of the main landing gear within 50 ft. of any brake should be regarded as unsafe during the first one hour and 15 minutes after the stop, unless the thermal release plugs have blown allowing the tires to be deflated.

2. Do not set parking brakes. Request fire fighting equipment. Hydraulic fluid fire is possible.

3. Do not attempt takeoff until the brake housings and tires are cool to the bare hand to prevent possible tire failure during takeoff or in flight.

NORMAL ZONE

1. Parking brakes may be set.

2. If stop does not exceed 14 million foot pounds per brake, subsequent takeoff may be performed immediately. However, brake application is restricted to speeds and gross weights in the CAUTION ZONE or below in the event a subsequent takeoff is aborted.

3. Unrestricted subsequent takeoff may be performed only after brake housings and tires are cool to the bare hand.

4. If stop exceeds 14 million foot pounds per brake, subsequent takeoff may be performed only after brake housing and tires are cool to the bare hand.

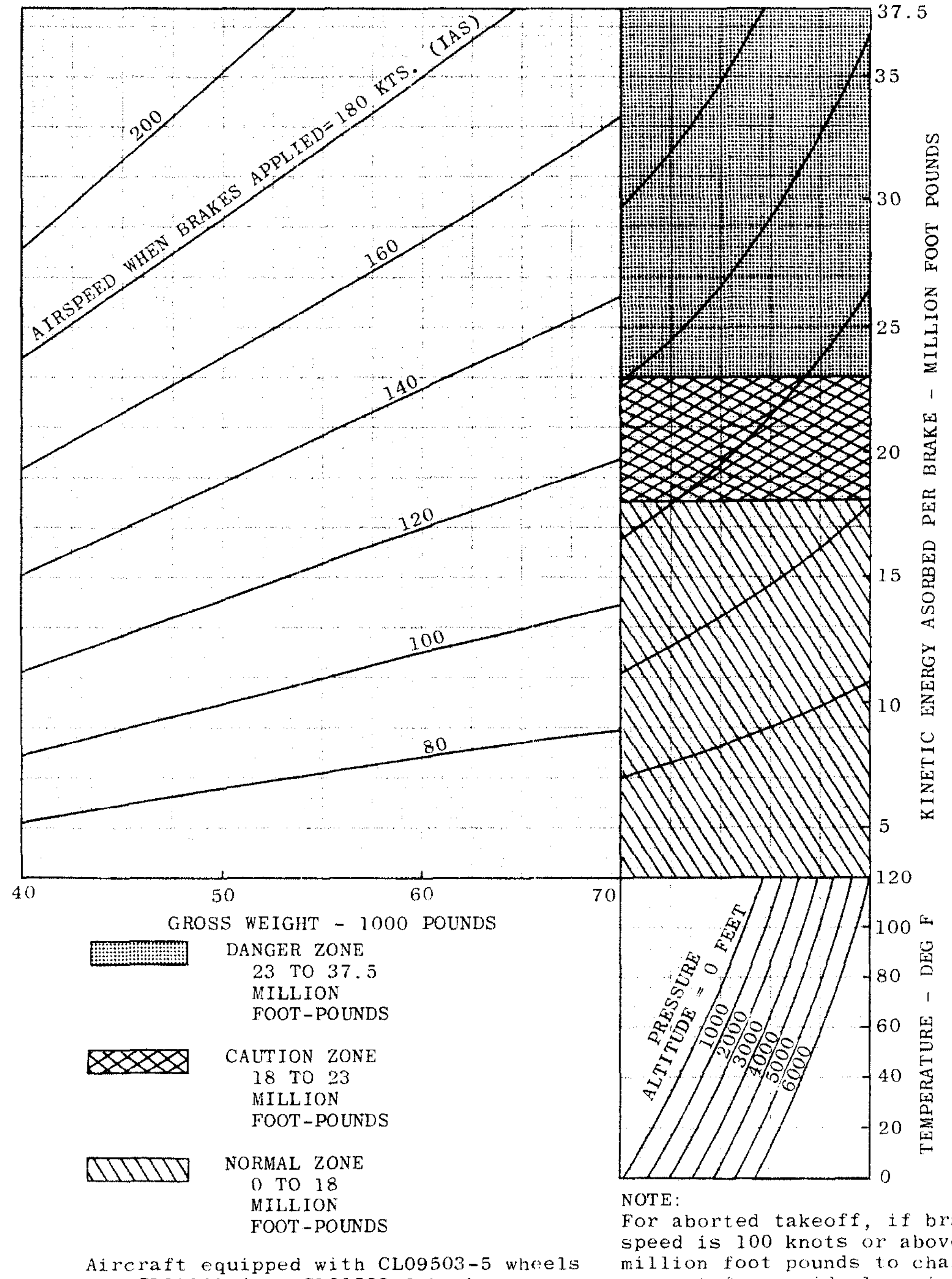
A0000000-046

Figure 5-10. (Sheet 1)

Brake Energy Limits

DATE BASIS: ESTIMATED FLAPS - SLATS - SPOILERS RETRACTED FUEL GRADE: JP4
DATE: 11 NOVEMBER 1966 ENGINES: Y&TF30-P-1

The following information explains action to be taken when a stop in the DANGER, CAUTION, or NORMAL ZONES is performed:



Aircraft equipped with CL09503-5 wheels and CL09503-4 or CL09503-6 brakes.

NOTE:
For aborted takeoff, if brake application speed is 100 knots or above, add four million foot pounds to charted value to account for residual engine thrust.

DANGER ZONE

1. Use moderate braking below 25 knots until taxi speed of 5—10 knots is obtained. Release brakes, if possible, and maintain forward motion.

CAUTION

Applying maximum brake pressure below 25 knots may cause brake rotors and stators to fuse together.

2. Proceed to the nearest parking area clear of other airplanes and personnel without stopping and as quickly as possible. Do not set parking brakes.

3. Request firefighting equipment. Hydraulic fluid fire is imminent. Approach main landing gear from front or rear for fire fighting purposes only.

4. Extinguishing agents shall be applied as a fog or foam on the tires and directly to the brakes. Do not spray liquid directly on the wheels.

CAUTION ZONE

1. The area in the vicinity of the main landing gear within 50 ft. of any brake should be regarded as unsafe during the first one hour and 15 minutes after the stop, unless the thermal release plugs have blown allowing the tires to be deflated.

2. Do not set parking brakes. Request fire fighting equipment. Hydraulic fluid fire is possible.

3. Do not attempt takeoff until the brake housings and tires are cool to the bare hand to prevent possible tire failure during takeoff or in flight.

NORMAL ZONE

1. Parking brakes may be set.

2. If stop does not exceed 14 million foot pounds per brake, subsequent takeoff may be performed immediately. However, brake application is restricted to speeds and gross weight in the CAUTION ZONE or below in the event a subsequent takeoff is aborted.

3. Unrestricted subsequent takeoff may be performed only after brake housings and tires are cool to the bare hand.

4. If stop exceeds 14 million foot pounds per brake, subsequent takeoff may be performed only after brake housing and tires are cool to the bare hand.

A0000000-047

Figure 5-10. (Sheet 2)

Tail Bumper Contact Limits

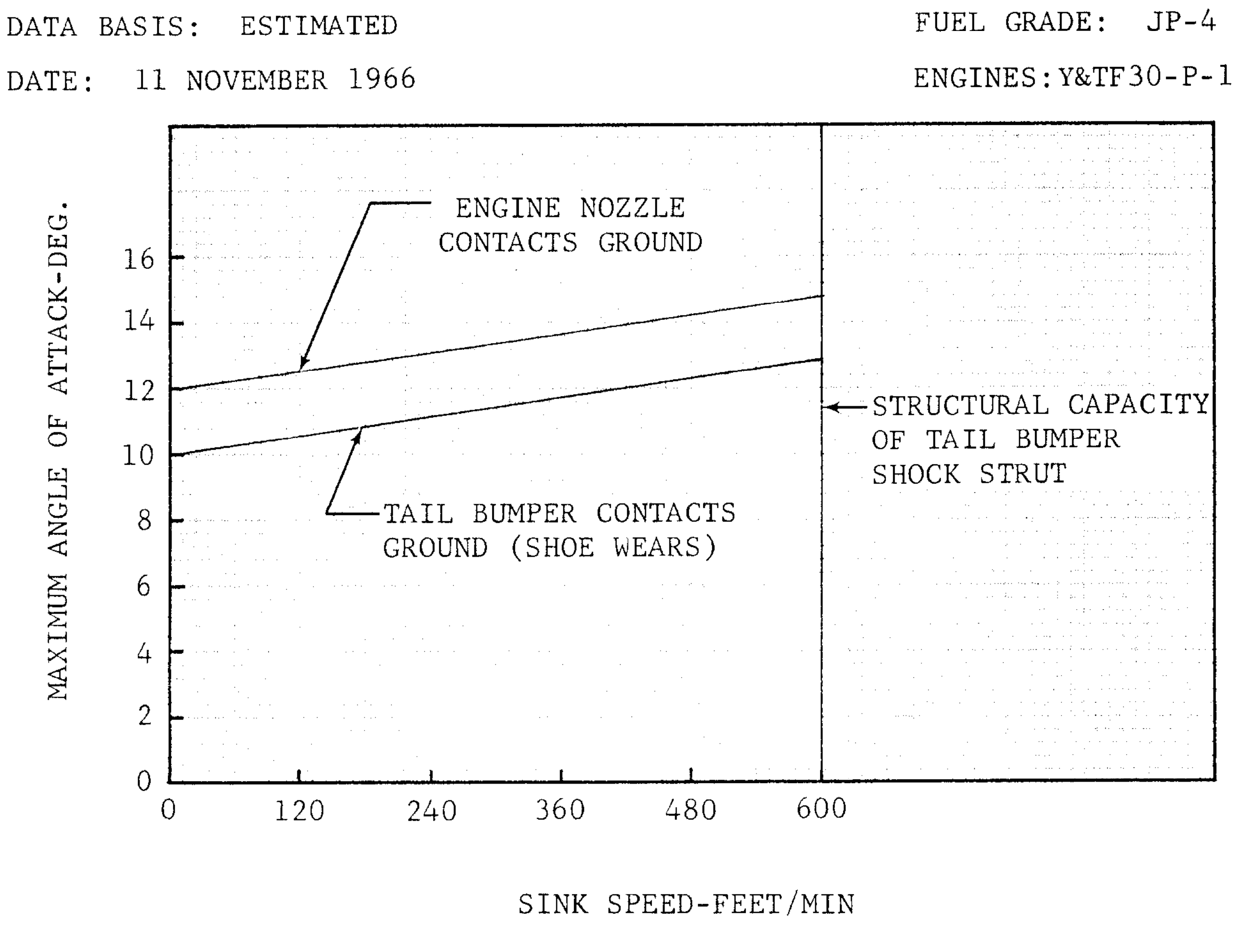


Figure 5-11.

A0000000-048

For aircraft not equipped with CL09503-5 wheels and CL09503-4 or CL09503-6 brakes, refer to Temporary Flight Limitations Report (FZM-12-922).

MINIMUM ANTI-SKID CONTROL SPEED.

Minimum anti-skid control speed - 20 knots.

Employ only light to medium braking below this speed.

MISCELLANEOUS OPERATIONAL LIMITATIONS.

ANGLE OF ATTACK FOR BUFFET ONSET.

Refer to T.O. 1F-111(Y)A-1A.

SPEED BRAKE LIMIT.

For speed brake limitations, refer to Temporary Flight Limitations Report (FZM-12-922).

TAIL BUMPER CONTACT LIMITS.

See figure 5-11 for maximum angle of attack versus sink rate for tail bumper contact limits.

CANOPY HATCH OPERATING SPEED.

Taxi speed limit with canopy hatches open - 60 KIAS.

The canopy hatches are not designed to be opened in flight.

CAUTION

The pilot should consider gusts of severe surface winds as a contributing factor to the 60 knot restriction.

HYDRAULIC SYSTEM OPERATING LIMITS.

The normal hydraulic operating cycle is limited to 5 complete flap and slat operation cycles (extension and retraction) in a 5 minute period. To avoid excessive hydraulic temperatures, a delay of 10 minutes is required before repeating.

EMERGENCY FLAP EXTENSION LIMITS.

The emergency (electrical) operating cycle limit is 1 cycle, then a delay of 10 minutes before repeating.

INCLEMENT WEATHER OPERATING LIMITATIONS.

Do not fly in inclement weather where icing conditions are likely to exist. Some airplanes are not equipped with windshield wash, rain remove, or complete anti-icing provisions.

This is the last page of Section V

SECTION VI

FLIGHT CHARACTERISTICS

TABLE OF CONTENTS.

INTRODUCTION.

The flight characteristics information presented in this section is based primarily on estimations supplemented by flight experience. Detailed stability and control flight testing has not been completed. Utilization of the variable sweep concept has not resulted in unusual flight characteristics. The main features of the flight control system (self adaptive gain changing and command augmentation) significantly minimizes variations in stability and control characteristics over the large mach-altitude operating spectrum of the aircraft. The low friction and breakout forces associated with the flight control system enhance ease of handling and manueverability. Wing sweep transition will not be reflected to the pilot in the form of a trim change due to the series trim feature of the flight control system which acts as an automatic trim system. At a fixed mach-altitude condition, wing sweep transition will be noticed only by the increase in aircraft angle of attack and attitude for an aft movement of the wing. For a forward movement of the wing, a decrease in angle of attack and attitude will occur.

FLIGHT CONTROL SYSTEM.

A detailed description of the flight control system appears in Section I. A short summary is presented herein of those features pertinent to flight characteristics.

LONGITUDINAL CONTROL.

Longitudinal control is accomplished by means of symmetrical (elevator) motion of the horizontal stabilizers. A direct mechanical linkage is provided between the control sticks and the horizontal stabilizer servo actuators. Artificial feel is provided by a fixed spring which reflects a force of 10.8 pounds per inch of stick deflection, measured at the stick grip. The elevator deflection per inch of stick is varied throughout the flight envelope by the command augmentation feature to maintain a relatively constant stick force per g. The command augmentation feature compares commanded normal acceleration and pitch rate with actual aircraft normal acceleration and pitch rate, and generates a pitch damper command proportional to the error. The pitch damper signal is summed mechanically with the stick input, and if the actual aircraft response is less than commanded, the damper will produce an elevator deflection in a direction to aid the stick input. Conversely, if the actual aircraft response is greater than commanded, the pitch damper input will oppose the stick input. This action tends to maintain relatively constant aircraft response throughout the flight envelope.

Stability Augmentation.

Stability augmentation system gains in the pitch and roll channel are automatically set by the self-adaptive gain system. The stability augmentation system is provided automatically whenever aircraft power is supplied to the flight control system. The stability augmentation system is redundant from the sensors to the damper servos. Caution lamps are provided on the caution panel to indicate a failure in one of the redundant branches. The procedures to be followed when any flight control system caution lamp is lighted are discussed in Section III.

Manual Trim.

Manual trim inputs are commanded from the trim button on either of the control stick grips when the auxiliary trim switch is in the stick position. Manual trim inputs can also be commanded by the switch on the auxiliary flight control panel in case of a stick trim button failure. During normal operation the manual trim commands the pitch parallel trim actuator. Manual trim commands move the stick and elevator linkage unless restrained by the pilot. If the pitch damper is turned off, the parallel trim actuator will be driven to stick neutral, and the trim buttons will then drive the series trim actuator. Under normal operations in high speed flight when the pitch damper is on, the pitch series trim actuator relieves the steady state load carried by the pitch damper servo. This action relieves the damper from carrying one g trim and minimizes damper disengage transients. In low speed flight with the flaps and slats extended the series trim actuator is locked during normal operation. The series trim commands are added in series with the stick inputs and as such, the manual trim commands using the pitch series trim are not reflected to the pilot's stick.

LATERAL CONTROL.

When the wings are forward of 45 degrees, lateral control of the aircraft is provided by two spoiler segments on each wing panel in addition to the asymmetrical (aileron) motion of the horizontal stabilizers. Lateral control of the airplane with the wings aft of 45 degrees is achieved using only asymmetrical motion of the horizontal stabilizers. A direct mechanical linkage from the control stick to the horizontal stabilizer servo actuators is provided. Mixing of the aileron and elevator control to the actuators is also accomplished mechanically. The spoilers are connected to the control stick by redundant electrical circuits. A force detent occurs at approximately one-half maximum lateral stick deflection. With the roll damper on, full normal control is achieved when the stick is moved to the detent. The detent should not be exceeded for normal operation. With the roll damper off, full spoiler deflection (for wing sweeps less than 45 degrees) and one-fourth asymmetrical horizontal stabilizer deflection are provided when the stick is moved to the detent. Full asymmetrical horizontal stabilizer deflection can be obtained, for emergency operation only, by overpowering the force detent and displacing the stick to the physical stops.

Stability Augmentation.

Roll rate command augmentation is provided when the roll damper is on. The command augmentation feature compares commanded roll rate with actual aircraft roll rate and generates a roll damper command proportional to the error. If the actual airplane response is greater than commanded, the roll damper will reduce the commanded roll rate and if it is less it will increase the commanded roll rate. This feature tends to maintain a relatively constant roll rate per pound of force throughout the flight envelope. Roll damping is also provided through the roll damper and is available only when the damper is on.

Manual Trim.

Manual trim inputs are commanded through the roll damper from the trim button on either of the control stick grips when the auxiliary trim switch is in the stick position. When the roll damper is off or the auxiliary trim switch is out of the stick position, no roll trim is provided and the aircraft should be trimmed laterally with the rudder.

DIRECTION CONTROL.

Directional control is achieved using direct mechanical linkage between conventional rudder pedals and the servo actuator of the rudder. Yaw damping and automatic turn coordination are provided when the yaw damper is engaged. Rudder manual trim is actuated by displacing the rudder trim switch on the flight control test panel to the right or left. Rudder trim is added in series with the rudder pedal inputs and is not reflected to the pedals.

FLIGHT WITH STABILITY AUGMENTATION SYSTEM OFF.

A partial tabulation of stability augmentation system off operating limits is provided in Section V for aircrew convenience. For a complete discussion and detailed charts presenting the entire stability augmentation system off operating envelope, refer to "Flight Control System Limitations", Section V, T.O. 1F-111(Y)A-1A.

LEVEL FLIGHT CHARACTERISTICS.

TAKEOFF.

Takeoff is normally accomplished with the wings in the 16 degree wing sweep position with full flaps. Adequate longitudinal control is available to lift the nosegear and obtain lift-off attitude prior to attaining lift-off velocity. During normal performance takeoffs, aft stick forces of approximately 15 to 20 pounds will be required for nosegear lift-off. Immediately after lift-off a forward stick movement is required to arrest the rotation of the aircraft. During climbout, trim changes associated with retraction of the gear and slats and flaps are small. A noticeable increase in angle of attack will occur as the slats and flaps are retracted. This is a result of the loss in lift as the slats and flaps are retracted. Slats and flaps should not be fully retracted until such time that the minimum recommended flying speeds for the clean airplane are attained. (Refer to "Minimum Flying Speed", Section V.) Upon completion of slat and flap retraction, the wing should be swept to 26 degrees.

SUBSONIC FLIGHT.

Operation of the aircraft at subsonic speeds up to mach 0.8 should normally be accomplished with a wing sweep of 26 degrees. Generally, response and damping about all axes in this speed range is considered excellent based on flight experience to date. Flight at wing sweeps aft of 45 degrees for subsonic flight is not recommended due to the fact that the spoilers are locked out and as a result roll control is significantly reduced. For wing sweeps aft of 45 degrees, rolling maneuvers should not exceed 60 degrees of bank to prevent excessive sideslip angles from being developed. However, all other characteristics of the aircraft are considered good at the aft sweep angles.

TRANSONIC FLIGHT.

During operation of the aircraft at transonic mach numbers (mach 0.9 to 1.1) wing sweep angles of 45 to 72.5 degrees should be utilized. At 20,000 feet and above, sweep angles of 45 degrees are recommended to keep the aircraft angle of attack low which will result in better acceleration characteristics. At the lower altitudes, more aft sweep angles are recommended to optimize acceleration. Although the spoilers will be locked out with the more aft sweeps, roll performance will be improved due to the lower angle of attack. During transonic flight above 25,000 feet a relatively small directional trim change occurs just prior to achieving supersonic flight. As altitude is decreased in this speed regime the trim change is more noticeable, and below 10,000 feet may be exhibited as a small Dutch roll transient accompanied by mild buffet. No trim changes occur longitudinally or laterally. The exact cause of this characteristic is not known at this time, but will be investigated further. Predicted longitudinal short period characteristics for a wing sweep of 50 degrees at low altitude-high transonic and low supersonic speeds indicate that the short period will have a frequency of approximately 1.0 to 1.2 cycles per second. Operation in these flight conditions should be avoided through the utilization of sweep angles at or near the maximum sweep angle until flight evaluation of this high longitudinal short period characteristic has been completed.

SUPERSONIC FLIGHT.

Flight in the supersonic flight spectrum (mach number 1.2 and above) should normally be accomplished with the wing swept fully aft. Flight can be made in this speed range with sweep angles as low as 50 degrees. Throughout the supersonic flight spectrum covered to date, response and damping characteristics are good.

LANDING.

Landing should normally be accomplished with the wing at 26 degrees. This sweep is compatible with that associated for subsonic flight and as such the slats and flaps can be extended without further wing sweep operation. Speed limits for this wing sweep allow a wide range of operation and maneuvering during approach to the landing. The aircraft generally exhibits good handling characteristics in the traffic pattern. Turn coordination is less than optimum and the aircraft exhibits a tendency to roll out when establishing a bank angle as the lateral control is returned to neutral. With the slats and flaps extended, adverse sideslip during banks becomes very noticeable and as such rudder control is required. Although crosswind landing experience has been rather limited to date, landings have been accomplished up to 25 knots effective crosswind conditions and have required no unusual techniques.

MANEUVERING FLIGHT CHARACTERISTICS.

Wing sweep angles for maneuvering flight are compatible with those previously described for level flight characteristics. During flight with the slats and flaps extended, longitudinal maneuvering should not be allowed to exceed an angle of attack of 20 degrees to preclude the entrance to a stall. Large rolling maneuvers should also be avoided to preclude buildup of excessive adverse sideslip. During pull-ups or turns at high speeds (slats and flaps retracted), the stick force per g is relatively independent of wing sweep and altitude. A mild variation with mach number, however, does exist. Stick deflection per g also exhibits the same basic characteristics. (See figure 6-1). Flight test results obtained to date essentially confirm the basic levels and trends of the predicted maneuver gradients. With the pitch damper off, much larger variations in the stick force per g characteristics are evidenced since the command augmentation will be inoperative. Flight in given portions of the operating spectrum is restricted in such case due to low stick force per g (less than 3 pounds per g). Refer to T.O. 1F-111(Y)A-1A for operating limitations. Buffet of the aircraft in high speed flight is generally encountered at angles of attack above eight to ten degrees. The buffet onset is characterized by an amplitude of ±.05 g's of constant frequency. Further increase in angle of attack results in this amplitude increasing to approximately ±.2 to .3 g's. Flight into heavy buffet (±.5 g's) should be avoided until completion of stall and spin testing of the aircraft. During supersonic flight at altitudes above 30,000 feet with aft sweeps, full back longitudinal control maneuvers may result in some stick "talkback" being detected. This characteristic is a result of the pitch damper attaining its full deflection. Loss of pitch damping in one direction will result but may be restored by relieving the back force being held. This same characteristic is exhibited at negative load factors for the aft sweep throughout its operational flight spectrum. Throughout the operational envelope of the aircraft, rolling maneuvers should be accomplished with the wing sweeps recommended under "Level Flight Characteristics", this section. Rolling maneuvers with wing sweeps of more than 45 degrees at subsonic flight should be avoided due to the decreased roll control when the spoilers are locked out. At high speeds during maximum rolling maneuvers, abrupt forward stick motion should be avoided to preclude rapid buildup in roll rate.

DIVE RECOVERY.

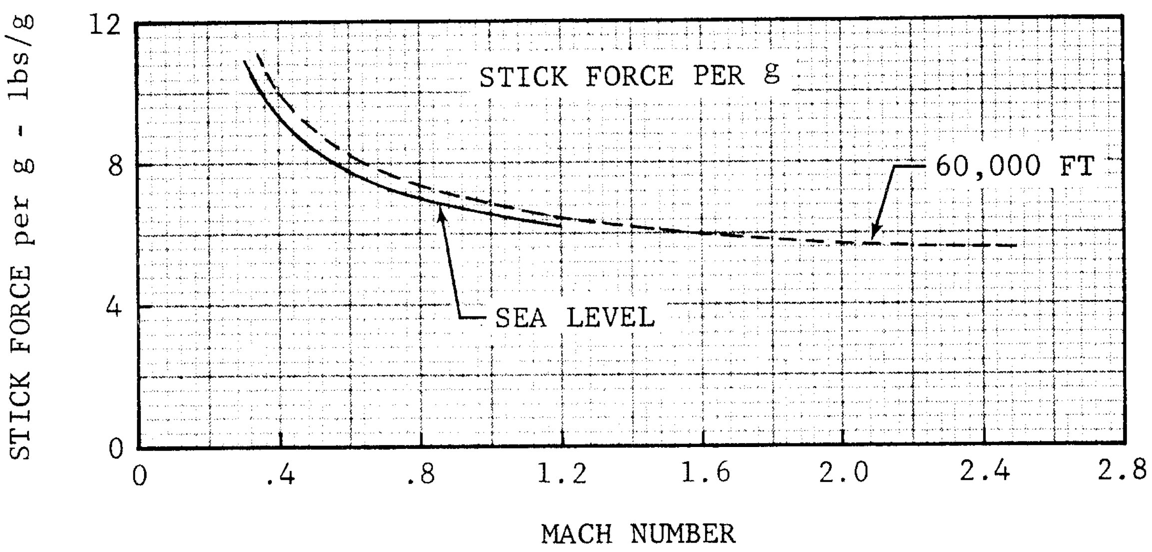
Refer to T.O. 1F-111(Y)A-1A.

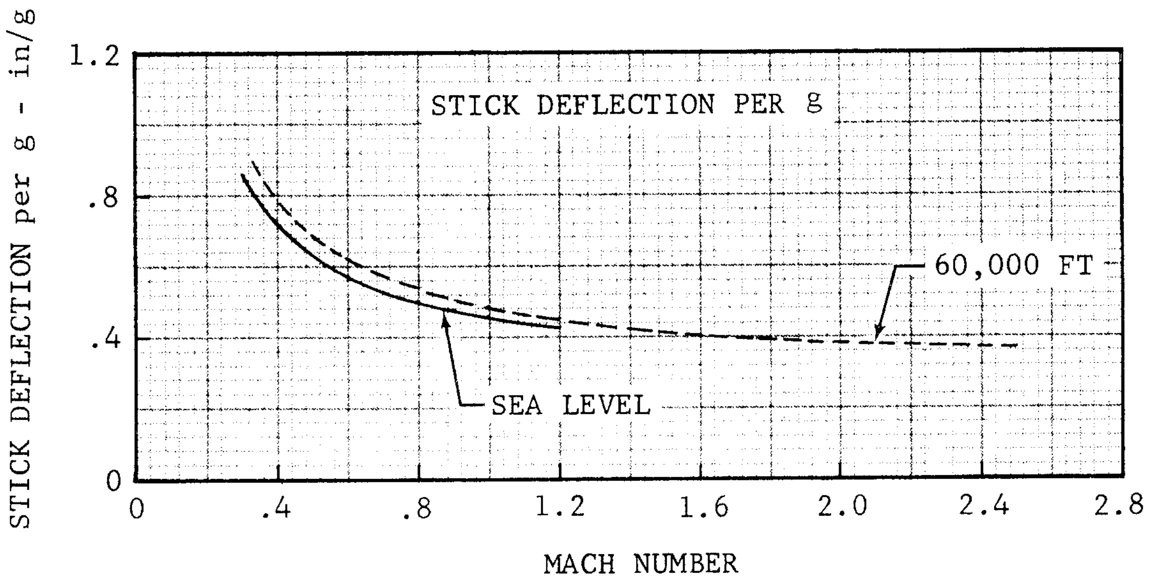
Longitudinal Maneuver Gradients

WING SWEEPS: 26° TO 72.5°

DATA BASIS: ESTIMATED
DATE: 11 NOVEMBER 1966

FUEL GRADE: JP-4
ENGINES: Y&TF30-P-1





1. NO EXTERNAL STORES
2. STABILITY AUGMENTATION SYSTEM ON

A0000000-049

Figure 6-1.

STALLS.

LOW SPEED (SLATS AND FLAPS EXTENDED).

Stalling characteristics are expected to be conventional in all configurations and normal techniques. Maintaining wings level while applying forward stick, should allow adequate recovery from all stalled conditions. Warning of impending stall is not present. In order to preclude entering a stall, an angle of attack of 18 degrees should not be exceeded. The characteristics of the airplane at stall should be as follows: No increase in lift with increased angle of attack, no apparent change in pitching moment, possible divergency in yaw and sharp reduction in roll power due to reduction in spoiler effectiveness. Refer to Section V for minimum flying speeds.

WARNING

Flights into heavy buffet or intentional stalls are prohibited.

HIGH SPEED (SLATS AND FLAPS RETRACTED).

Accelerated stalls are expected to be preceded by adequate stall warning in the form of airframe buffet. For wing sweeps of 26 through 45 degrees, stall characteristics should be similar to those described above. For wing sweeps of 45 through 72.5 degrees, no stall, in the usual sense of the word, is encountered. A stall for these wing sweeps will be defined by degradation of aircraft lateral or directional control at angles of attack above twenty degrees. For most conditions, there will be a tendency to diverge in yaw. To preclude entering these conditions, an angle of attack of 18 degrees should not be exceeded. Normal recovery techniques (maintaining wings level while applying forward stick) are expected to provide adequate recovery from all stall conditions.

WARNING

Flights into heavy buffet or intentional stalls are prohibited.

SPINS.

Spin tests have not been conducted on the airplane and consequently, its exact spin characteristics are unknown. However, spin investigations conducted during wind tunnel testing have indicated a tendency for the airplane to enter a spin in any configuration and wing sweep. Spin entry from 1G stall with symmetrical controls has been possible in some instances. However, when asymmetric rudder and/or lateral control is supplied at stall, spin entries should be expected for all configurations. Crossed controls at or before accelerated stalls should likewise produce abrupt entries into erect or inverted spins. It should be possible to enter spins at all altitudes and at mach numbers up to supersonic speeds. Engine stall or flameout should occur prior to or at spin entry.

WARNING

Intentional spins and flight into heavy buffet and/or stall are prohibited.

UPRIGHT SPINS.

Upright spins are expected to be primarily oscillatory in nature with excursions up to ±25 degrees in pitch and ±35 degrees in roll. The spin rates can vary from 10 to 4 seconds per turn. A flat spin mode has also been noted for wing sweeps of 50 degrees and less. This mode is relatively free from oscillations about any axes and spins at 4 to 2.3 seconds per turn. The latter could produce pilot loads of up to 5 g's forward. This mode should rarely be encountered and would develop only after a spin has been permitted to build up for several turns.

UPRIGHT SPIN RECOVERY.

Lateral control is the primary spin recovery system and it is imperative that the stick be held into the spin until recovery is obtained. Under no circumstances should an attempt be made to oscillate the airplane out of spin. Immediately upon recognition of the direction of rotation (turn indicator should be used to verify direction) apply the following procedure:

1. Throttles - IDLE.
2. Opposite rudder (switch to full rudder authority when in clean configuration).
3. Roll into spin using full stick throw (longitudinally keep stick neutral).

WARNING

Do not move stick around as this will deplete your hydraulic supply and will reduce possibility of recovery.

4. Do not make any configuration changes to flaps, gear, speed brakes or wing sweep.
5. Neutralize controls upon recovery.
6. Throttles - As required.
7. Switch back to normal rudder authority.

The airplane is expected to recover from spins in a near vertical attitude. Unless a pullup is initiated immediately, ground clearance can become critical.

WARNING

If the airplane has not recovered by the time 15,000 feet has been reached - EJECT.

INVERTED SPINS.

An inverted spin is similar to erect oscillatory spin and relatively easy to terminate.

INVERTED SPIN RECOVERY.

Immediately upon recognition of inverted spins (turn indicator should be used to verify direction), use the following procedures:

1. Throttles - IDLE.
2. Apply opposite rudder (switch to full rudder authority when in clean configuration).
3. Neutralize stick.
4. Neutralize rudder upon recovery and initiate immediate pullup.
5. Throttles - As required.
6. Switch back to limited rudder authority.

WARNING

If the airplane has not recovered by the time 15,000 feet is reached - EJECT.

FLIGHT WITH EXTERNAL STORES.

Testing with external stores has not been completed. It is not anticipated, however, that the addition of external stores will result in any unusual flight characteristics.

FLIGHT WITH SPEED BRAKE EXTENDED.

Extension of the speed brake will result in airplane buffet and a random pulsating motion in the lateral-directional axes.

This is the last page of Section VI.

SECTION VII

ALL WEATHER OPERATION

TABLE OF CONTENTS.

Note

In general, this section consists of procedures and information which differ from, or are supplementary to, the normal operating procedures in Section II. In some cases, however, repetition has been necessary for emphasis, clarity, or continuity of thought.

INSTRUMENT FLIGHT PROCEDURES

This airplane is designed to perform operational missions in all extremes of weather. Handling characteristics are such that supersonic flight may be safely accomplished. On instrument flights, delays in departure and descent, and low climb rates to altitude are often required in high density control areas. These factors may increase fuel consumption, reduce flight endurance and dictate that all flight under instrument conditions be carefully planned and that due consideration be given to the additional time and fuel which may be required.

BEFORE TAKEOFF.

1. Line up visually with center line of runway.
2. Heading indicator - Set heading marker under top index.
3. Attitude indicator - Adjust to indicate 2.5° nose down position.
4. Pitot heat - Climatic.

INSTRUMENT TAKEOFF.

Make normal takeoff.

INSTRUMENT CLIMB.

After lift off, maintain 10 degrees pitch attitude until reaching climb speed and then establish climb configuration and climb schedule. The optimum thrust climb schedule recommended in Appendix I is suitable for instrument flight.

INSTRUMENT CRUISING FLIGHT.

Thrust settings and configuration for optimum cruise schedule recommended in Appendix I are satisfactory while using standard instrument techniques. Maximum bank angle of 30 degrees is normally used.

HOLDING.

Holding should be accomplished at 280 KIAS. Maximum bank angle of 30 degrees is normally used.

JET PENETRATION.

Prior to beginning penetration ascertain the weather conditions and the availability of radar or ILS. If ceiling or visibility is below published minimums, make the decision to proceed to an alternate while still at altitude. Refer to the descent charts in Appendix I for appropriate airspeeds and airplane configuration. For maximum range an idle power descent at 225 KIAS, with 26 degree wing sweep and speed brakes retracted is recommended. For minimum time in descent, mach .80 or 300 KIAS, whichever is less, at a 26 degree wing sweep and speed brake extended is recommended.

PAR/ASR APPROACH.

Refer to descent charts in Appendix I for appropriate airspeeds.

ILS APPROACH.

Refer to descent charts in Appendix I for appropriate airspeeds. Figure 7-1 illustrates a typical Radar/ILS approach.

1. ILS power switch - POWER.
2. Frequency selector knob - Set.
   Set the frequency selector knob to the frequency of the localizer facility to be used.
3. Volume control - Adjusted.
   Adjust the volume to a comfortable level while identifying the station.
4. Instrument system coupler mode selector knob - ILS.
5. HSI course selector window - Set.
   Set the inbound localizer course in the HSI course selector window.
6. Radar altimeter - Set.
   Set radar altimeter index pointer to the minimum altitude for the approach. Closely monitor the radar altimeter on final approach to assure safe terrain clearance.
7. Monitor ADI, LCOS, and HSI for proper indications.

AIRBORNE INSTRUMENT LOW APPROACH (AILA).

The bomb-nav system, in conjunction with the attack radar, can be used for making letdowns and approaches to runways not equipped with ground based letdown facilities and as a backup for monitoring TACAN, ILS and Radar letdowns. Airplane altitude should be calibrated and the position should be updated by the most accurate means available immediately prior to starting the letdown.

1. Altitude calibration - Completed.
   Calibrate altitude over a point of known elevation, preferably the landing runway.

WARNING

Altitude calibration is critical. AILA glide slope angle to desired touchdown point is computed on the basis of ground range and altitude. Altitude calibration errors will cause the glide slope to be computed to a point short of or over the desired touchdown point.

2. Airplane position - Updated.
   Update the airplane position using the most accurate means available. Use the attack radar to periodically update the position during letdown.
3. Bomb nav mode selector knob - SHORT RANGE.
   Place the bomb nav mode selector knob to the SHORT RANGE normal navigation or auxiliary navigation position as applicable.
4. Destination position counters - Set.
   Set the destination position counters to the coordinates of the touchdown point on the approach runway.
5. Target fix mode selector button - Depressed.
6. Fixpoint elevation counter - Set.
   Set the fixpoint elevation counter to the elevation of the touchdown point on the runway.
7. Glide/dive angle counter - Set.
   Set the glide/dive angle counter to the desired glide slope angle of the approach runway.
8. Magnetic heading synchronization indicator - Nulled.
9. Radar altimeter - Set.
   Set the radar altimeter index pointer to the minimum altitude for the approach. Closely monitor the radar altimeter on final approach to assure safe terrain clearance.
10. Instrument system mode selector knob - AILA.
11. Horizontal situation indicator - Set.
    Set the inbound heading of the runway in the course indicator window and set the heading marker to the runway heading on the compass card.
12. Instrument system coupler pitch steering switch - ALT REF.
    The altitude reference position may be used during AILA until glide slope is intercepted.

MISSED APPROACH PROCEDURE.

Refer to Go-Around, Section II, for missed approach procedure.

ICE AND RAIN

Do not fly in inclement weather where icing conditions are likely to exist. Some airplanes are not equipped with windshield wash, rain remove, or complete anti-icing provisions. There are no provisions for surface anti-icing. The performance capabilities of the airplane should be utilized to avoid extreme icing conditions. When icing is encountered, a change in altitude, course, or airspeed should be made.

WARNING

In the event of the pitot tube icing, the airspeed and mach indicators may drop to zero or remain fixed. All systems that receive intelligence based upon pitot pressure through the central air data computer, and the standby airspeed instruments, will be affected. The loss of airspeed indication during climb or descent is an extremely dangerous safety of flight hazard. The attitude indicator, vertical velocity indicator, altimeter and power setting can be used during the emergency flight condition.

Substantial ice buildups can necessitate increased power setting for maintaining airspeed and could cause distortions in the shape of air foil surfaces, thus affecting the lift and handling characteristics of the airplane. Either of these conditions tends to reduce total range. Rain has little or no appreciable effect on the flight characteristics.

OPERATION IN RAIN OR ICING CONDITIONS.

GROUND OPERATION.

Operate the airplane and systems as indicated in the "Cold Weather Procedures" in this section. Rain removal should be used when needed to improve visibility.

TAKEOFF AND INITIAL CLIMB.

Accomplish takeoff in the normal manner. Refusal speed will be considerably lower and the emergency stopping distance greater on wet or icy runways.

CRUISE.

Operate the airplane as necessary to avoid icing conditions. When ice is encountered, pitot heat and engine anti-icing should be used. Do not operate in rain, sleet, or hail longer than absolutely necessary. If it becomes necessary to fly in these conditions, constantly check the aircraft leading edges, including radome, for indications of peeling or other structural deterioration of the airplane surfaces. In the event deterioration of surfaces is observed, maintain airspeed as low as practicable and land at the nearest suitable airfield as soon as possible. If heavy precipitation conditions of the above type are encountered at and speed or light to moderate conditions exist at high airspeeds, an entry must be made in Form 781.

WARNING

To minimize impact damage from rain, sleet, or hail, do not exceed 450 KTAS.

TURBULENCE AND THUNDERSTORMS

WARNING

Flight through thunderstorm acitivity or known severe turbulence is not recommended and should be avoided if at all possible. Careful judgment must be exercised by the pilot in determining capability to safely enter or circumnavigate areas of such weather activity. The appropriate corrective action to be taken if moderate or greater turbulence is forecast will be planned with assistance of the weather forecaster during the weather briefing.

The use of attack radar, ground mode, provides an excellent means of navigating between or around storm cells and the airplane is capable of climbing over the top of most developed thunderstorms. If circumstances should force the flight into a zone of severe turbulence, establish throttle setting and pitch attitude. Recommended thunderstorm penetration airspeed is 275-300 KIAS.

Note

When using terrain following radar, the back scatter from drizzle or rain and other forms of precipitation will often be visible on the scope. It should be quite apparent to the operator that if the precipitation is so heavy that he cannot determine visually where the terrain ends and the precipitation begins, the automatic signal detection circuitry will also be incapable of this descrimination and a climb command will result.

Note

The following factors, singly or in combination, may cause engine flameout:

Flight in cumulus buildups containing a high moisture content.

Engine icing of inlet guide vanes.

Turbulence associated with penetration of thunderstorms can result in excessively high angles of attack with resultant marginal engine performance.

NIGHT FLYING

Night flight necessitates a high degree of instrument proficiency and more dependence on flight instruments than would be expected for normal day VFR operations. Otherwise, techniques used in night flying do not differ appreciably from those used in daylight operation. Cockpit lighting has been designed to enhance night flying capability.

WARNING

The anti-collision lights should be turned OFF during flight through actual instrument conditions to avoid spatial disorientation resulting from the rotating reflections on the clouds. The navigation lights may be set to flash unless this becomes distracting in clouds.

COLD WEATHER PROCEDURES

Most cold weather operating difficulties are encountered on the ground. The following instructions are to be used in conjunction with the normal procedures given in Section II when cold weather operation is necessary.

BEFORE ENTERING AIRPLANE.

WARNING

All accumulated ice and snow must be removed from the airplane before flight is attempted. Takeoff distance and climb out performance can be adversely affected by ice and snow accumulations. The degree of roughness and distribution of these accumulations can vary stall speeds and alter flight characteristics to a degree hazardous to safe flight.

Ensure that water does not accumulate on control hinge areas or other critical areas where refreezing may cause damage or binding.

CAUTION

To avoid damage to airplane surfaces, do not permit ice to be chipped or scraped away.

Remove all protective covers and duct plugs; check to see that all surfaces, ducts, struts, drains and vents are free of snow, ice and frost. Ice and encrusted snow may be removed by using de-icing fluid or by direct air flow from a portable ground heater. Inspect the airplane carefully for fuel and hydraulic leaks caused by the contraction of fittings or by shrinkage of packings. Inspect areas behind the airplane to ensure that water or snow will not be blown onto personnel and equipment during engine start.

STARTING ENGINES.

Use normal procedures for starting engines. The throttles may be advanced to allowable power settings as long as engine instruments register within the engine operating limits. Refer to "Engine Operating Limits", Section V.

BEFORE TAXIING.

Check all instruments for normal operation. Check flight controls, flaps and slats for proper operation. Cycle flight controls to circulate warm fluid throughout the systems and check control reaction and operation.

TAXIING.

WARNING

Nose wheel steering may not be completely effective when taxiing on ice or hard packed snow. A combination of nose wheel steering and braking is recommended. Exercise care and taxi at reduced speed while operating on these surfaces. Increase the normal interval between airplanes to insure safe stopping distance and to prevent icing of airplane surfaces by melted snow and ice in the jet blast of preceding airplanes.

CAUTION

Painted areas on runways, taxiways and ramps are significantly more slippery than on unpainted areas, particularly when wet. In addition, painted areas sometimes serve as condensation surfaces and it is possible to have wet, frosty or even icy conditions on those areas when the overall weather condition is dry.

TAKEOFF.

Insure that takeoff data accounts for reduced braking capability due to ice and snow on runway in event of an abort. Make normal takeoff. Care should be exercised to avoid exceeding climb schedule speeds due to additional thrust available at low temperatures.

AFTER TAKEOFF.

After takeoff from a wet snow-covered or slush-covered field, operate the landing gear through several complete cycles to prevent gear freezing in the retracted position.

DESCENT.

Follow normal procedures.

LANDING.

Make normal approach and landing.

ENGINE SHUTDOWN.

Shut down the engine in the normal manner.

BEFORE LEAVING THE AIRPLANE.

Leave the canopy partly open; this will allow circulation within the cockpit to reduce windshield and canopy frosting.

HOT WEATHER AND DESERT OPERATION

Hot weather and desert operation requires that added precautions be taken against damage from dust, sand, and high temperatures. Particular attention should be given to those components and systems (engine, fuel, oil, hydraulic, pitot-static, etc.) which are susceptible to contamination, malfunction, or damage from sand and dust. All the filters on the airplane should be checked frequently. Components containing plastic or rubber parts should be protected as much as possible from blowing sand and extreme temperatures. During conditions of blowing sand and dust, the canopies should be closed and sealed and all protective covers installed when the airplane is not is use.

CAUTION

Do not attempt takeoff or engine operation in a sand storm or dust storm if avoidable. Park airplane crosswind and shut down engine to prevent damage from sand or dust.

EXTERIOR INSPECTION.

Inspect the exposed areas of the shock strut and actuator pistons on the landing gear and have them cleaned as required. Check engine inlet ducts for sand accumulation. Check tires for signs of blistering, and check for overinflation of tires and struts due to high ambient temperatures. Check for fuel or hydraulic leakage due to thermal expansion of sealing materials. Inspect the area aft of the airplane to make sure that engine exhaust will not cause sand or dust to be blown onto personnel or equipment when engines are started.

INTERIOR INSPECTION.

Inspect the crew compartments for excessive dust accumulation.

ENGINE START.

Follow normal procedures.

BEFORE TAXIING.

Ground testing should be complete but accomplished as expeditiously as possible.

TAXIING.

Follow normal procedures.

TAKEOFF.

Allow for longer takeoff distances in hot weather. Refer to Appendix I for recommended takeoff speeds and required takeoff distances.

CAUTION

It is imperative that takeoff not be made at lower than recommended speeds. When outside air temperature is high, do not rotate too soon, as more than usual takeoff distance will be required to obtain takeoff speed.

APPROACH AND LANDING.

Maintain recommended approach and landing speeds as shown in Appendix I. Allow for longer landing rolls resulting from increased true airspeeds.

CAUTION

Hot weather operation requires the pilot to be cautious of gusts and wind shifts near the ground.

ENGINE SHUTDOWN.

Follow normal procedures.

BEFORE LEAVING AIRPLANE.

Follow normal procedures.

This is the last page of Section VII.

Appendix I

PERFORMANCE DATA

Performance Data normally contained in Appendix I is temporarily provided in Classified Supplement T.O. 1F-111A(Y)A-1A.

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