Flight Manual
F-111E

USAF SERIES AIRCRAFT
SERIAL NO. AF 67-115 THRU AF 68-084

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THIS PUBLICATION IS INCOMPLETE WITHOUT T.O. 1F-111E-1-1 AND T.O. 1F-111E-1-2

THIS PUBLICATION REPLACES T.O. 1F-111E-1 DATED 4 FEBRUARY 1972, AND SAFETY SUPPLEMENTS T.O. 1F-111E-1SS-40, -42, -43, -45, -48, -52, -53, AND -54; AND OPERATIONAL SUPPLEMENTS T.O. 1F-111E-1S-22, -23, -25, -28, -29, AND -31.

See Numerical Index and Requirement Table for Current Status of Safety and Operational Supplements.

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

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CURRENT
FLIGHT CREW CHECKLIST
1F-111E-1CL-1
30 JUNE 1973

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SCOPE.

This manual contains the necessary information for safe and efficient operation of the aircraft. These instructions provide you with a general knowledge of the aircraft, 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 aircraft. 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. 0-1-1-4 and/or the Flyleaf of each Formal Safety and Operational Supplement for a listing of all current Flight Manuals, Checklists, and Safety and Operational Supplements. The Supplement Flyleaf provides a complete history of all Safety and Operational Supplements that have been issued and is the normal method used to notify flight crews of Supplement rescission.

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 1F-111E-1-1.

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 procedures. An Abbreviated Checklist is issued as a separate document. (See the back of the title page for the date of your latest Checklist.) Line items in the Flight Manual and the Checklist are identical with respect to arrangement and line item 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 crew member is entitled to personal copies of the Flight Manual, Safety Supplements, Operational Supplements, and Check Lists. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your supply personnel—

it is their job to fulfill your Technical Order requests. Basically, you must order the required quantities on the appropriate Numerical Index and Requirement Table (NIRT). Technical Orders 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver these publications to the flight crews immediately upon receipt.

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 on AF Form 847 through your Command Headquarters to: AFPRO GD/CAD (QAE) ATTN: CTOCU, P. O. Box 371, Fort Worth, Texas 76101.

AIRCRAFT DESIGNATION CODES.

Major differences between aircraft 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:

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▸"through" or "and on."

GLOSSARY

AAA	Anti-Aircraft Artillery
AA GUN	Air-to-Air Gunnery
AA RKT	Air-to-Air Rocketry
ABC	Automatic Ballistic Computer
ADL	Armament Datum Line
AFC	Automatic Frequency Control
AG GUN	Air-to-Ground Gunnery
AI	Airborne Intercept
AILA	Airborne Instrument Low Approach
AIPD	Airborne Intercept Pulse Doppler
AIR FF	Air Freefall
AIR RET	Air Retard
ALTM	Altimeter
ALT REF	Altitude Reference
ARS	Attack Radar System
ATF	Automatic Terrain Following
ATT GYRO	Attitude Gyro
AUD / PO	Audio/Push Off
AUX ATT	Auxiliary Attitude
AUX NAV	Auxiliary Navigation
AVVI	Altitude-Vertical Velocity Indicator
AYC	Adverse Yaw Compensation
B/C	Biological/Chemical
BCU	Ballistic Computer Unit
BNDTI	Bomb Nav Distance Time Indicator
CADC	Central Air Data Computer
CCIP	Continuous Computed Impact Point
CCM	Counter-Counter Measures
CIR	Circular
CKT	Circuit
CMDS	Countermeasures Dispenser Set
CMRS	Countermeasures Receiver Set
COM	Common
COMP	Compass
CPU	Central Program Unit
CRS SEL NAV	Course Select Navigation
CRT	Cathode Ray Tube
DBT	Dual Bombing Timer
DEST	Destination
DISP	Dispenser
FL Dual	Flare Dual
FDC	Flight Director Computer
FTC	Fast Time Constant
FW	Forward Warning
GND/GRD	Ground
GND MAN	Ground Manual
GND VEL	Ground Velocity
GRD FF	Ground Freefall
GRD RET	Ground Retard
HOM	Homing
IF	Intermediate Frequency
IFIS	Integrated Flight Instrument System
I/P	Identification of Position
IR	Infrared Radiation
IRRS	Infrared Receiver Set
IRT	Infrared Track
IRS	Infrared Search
IRU	Inertial Reference Unit
ISC	Instrument System Coupler
JETT	Jettison
LARA	Low Altitude Radar Altimeter
LCOS	Lead Computing Optical Sight
LNCH	Launch
LO	Local Oscillator
LOF BOMB	Loft Bombing
LSB	Lower Side Band
MAN CRS	Manual Course
MAN FIX	Manual Fix
MAN HDG	Manual Heading
MFC	Manual Frequency Control
MI/DIA	Miles/Diameter
MLD	Missile Launch Detect
MLR	Missile Launch Response
MSMA	Maximum Safe Mach Assembly
MR	Milli-radian
MRT	Modulator-Receiver Transmitter
MUL SEC	Multiple Sector
NC	Navigation Computer
NWS/AR	Nosewheel Steering/Air Refueling
OAP	Offset Aimpoint
OMO	Omni Warning Open
OMS	Off, Monitor, Safe
OMT	Omni Warning Threat
PFN	Pulse Forming Network
PP/PRES POS	Present Position
PPI	Plan Position Indicator
PR	Program
RAD	Radiation
RBS	Radar Bomb Scoring
REC	Receive
RKT	Rocket
RPT	Repeat
Rs	Slant Range

SAM	Surface Air Missile
SEC	Sector
SIF	Selective Identification Frequency
S/L	Search/Lock
SLC	Side Lobe Cancellation
SP	Stabilized Platform
SPC	Special Purpose Chaff
STAB AUG	Stability Augmentation
STC	Sensitivity Time Control
TBC	Trackbreaker Chaff
TIT	Turbine Inlet Temperature
TR-1	Infrared Source Detected
TR-M	Multiple Infrared Source Detected
TTI	Total Temperature Indicator
TVG	Time Varying Gain
TWS	Track While Scan
USB	Upper Sideband

DEFINITIONS:

ASYMMETRICAL LOADING

Weapon/tank load on any pylon is not identical to the corresponding pylon on opposite wing.

SYMMETRIC MANEUVER

A maneuver which imposes a symmetrical aerodynamic load on the aircraft such as a pull-up or push-over or steady bank.

ASYMMETRIC MANEUVER

A maneuver which imposes an asymmetrical rolling pull-out or intentional side-slip.

HERTZ (Hz)

A new term for cycles per second (cps) when used in conjunction with electrical or electronic language.

LIMIT CYCLE

A sustained oscillation which builds up to and maintains a constant amplitude.

The terms AC, WSO, BOTH, and COMMAND RESPONSE, used in Section II are defined as follows:

AC — Tasks accomplished by the left seat crew member.

WSO — Tasks accomplished by the right seat crew member.

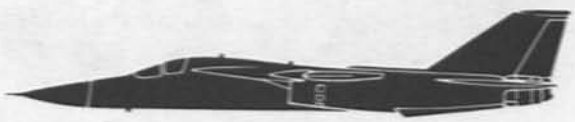
BOTH — Joint tasks accomplished concurrently by both crew members.

COMMAND RESPONSE — Those procedures in which the right seat crew member will state the first portion of the checklist statement and the left seat crew member will respond with the second portion of the checklist.

GO — Ground observer action is required.

All procedures in Section II that are not followed by one of the above terms are accomplished by the left seat crew member.

The F-111E



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AIRPLANE RETROFIT TECHNICAL ORDER INFORMATION

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. 0-1-1-4) for the complete listing of TCTO's which are applicable to this aircraft.

T.O. No.	*Short Title*	*System/Equipment Affected*
1F-111-550	Modify Cabin Pressure Regulator To Prevent Water Leak	Crew Module, Section I
1F-111-563	Install Improved AN/APS-109/ALR-41 RHAWS	Penetration Aids, Section I, T.O. 1F-111E-1-2
1F-111-572	Install Wheel Well Overheat Detection System	Wheel Well Overheat Detection System, Section I
1F-111-583	Install Hydraulic Fuse	Hydraulic System, Section I
1F-111-599	Modify Hydraulic Isolate Circuit to Nose Gear	Landing Gear System, Section I
1F-111-612	Roll Limitations	Maneuverability Limitations, Section V
1F-111-613	Automatic Activation of AN/URT-27 Radio Beacon Set	Communications Equipment, Section I
1F-111-617	Redesign Spoiler Actuator	Maneuverability Limitations, Section V
1F-111-645	Modify Weapon Bay Gun Cooling System	Armament System, Section I
1F-111-670	Provide Fire Detection System in Weapons Bay	Fire and Overheat Detection Systems, Section I
1F-111-673	Provide Redundant Fuel Distribution	Fuel System, Section I
1F-111-677	Add New Ground Interphone Receptacle	Communications Equipment, Section I
1F-111-687	Air Flow Control Switch	Air Conditioning and Pressurization System, Section I
1F-111-760	Add Strap to Restraint Harness	Crew Module, Section I
1F-111-766	Provide 25 Degree Flap Handle Detent	Wing Flaps and Slats, Section I
1F-111-780	Relocate Attack Radar Scope Controls	Attack Radar, Section I
1F-111-798	Weapon Bay Gun	Stores Limitations, Section V
1F-111-824	Add New Full Flap Stop	Wing Flaps and Slats, Section I
1F-111-833	Install Modified CMRS	Penetrations Aids 1F-111E-1-2
1F-111-856	Remove Strike Camera	Strike Camera, Section I
1F-111-863	Install AN/ARC-123 HF Radio	Communications Equipment, Section I

T.O. No.	*Short Title*	*System/Equipment Affected*
1F-111-891	Provides Stall Warning Lamp and Audio Signal	Flight Control System, Section I
1F-111E-505	Provide QRC Pods	Penetration Aids, 1F-111E-1-2
1F-111E-506	Provide PAL Controller	Armament System, Section I
1F-111E-507	Provide Auto Radar Scope Photography	Attack Radar, Section I

SECTION I

DESCRIPTION & OPERATION

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THE AIRCRAFT.

The F-111E is a two place (side-by-side) long range fighter bomber built by General Dynamics, Fort Worth Division. The aircraft 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 turbofan 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 aircraft 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 and gravity refueling capability through filler caps in the top of the wing and fuselage. See figure 1-1 for aircraft general arrangement and figure 1-2 for crew station general arrangement.

General Arrangement Diagram

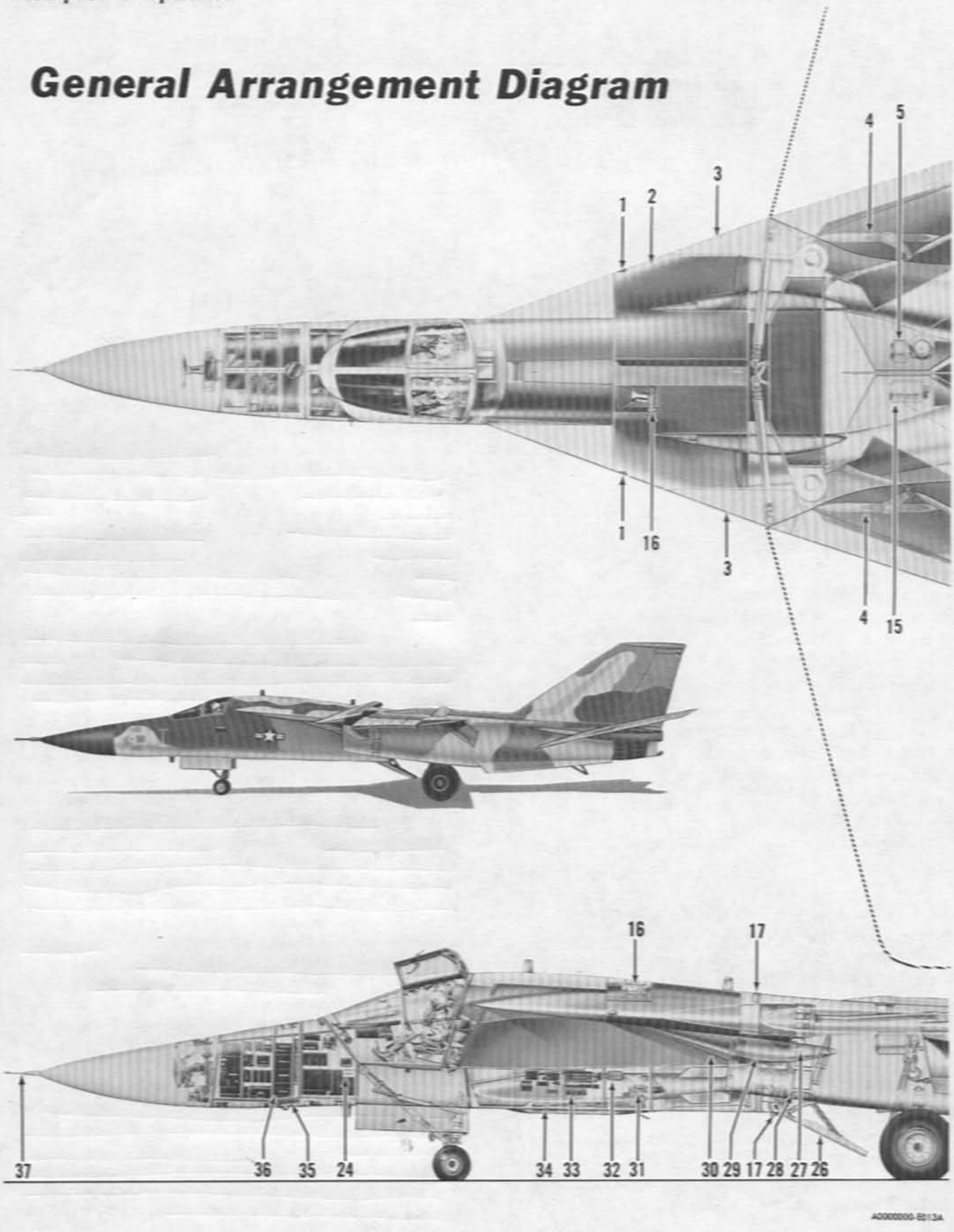
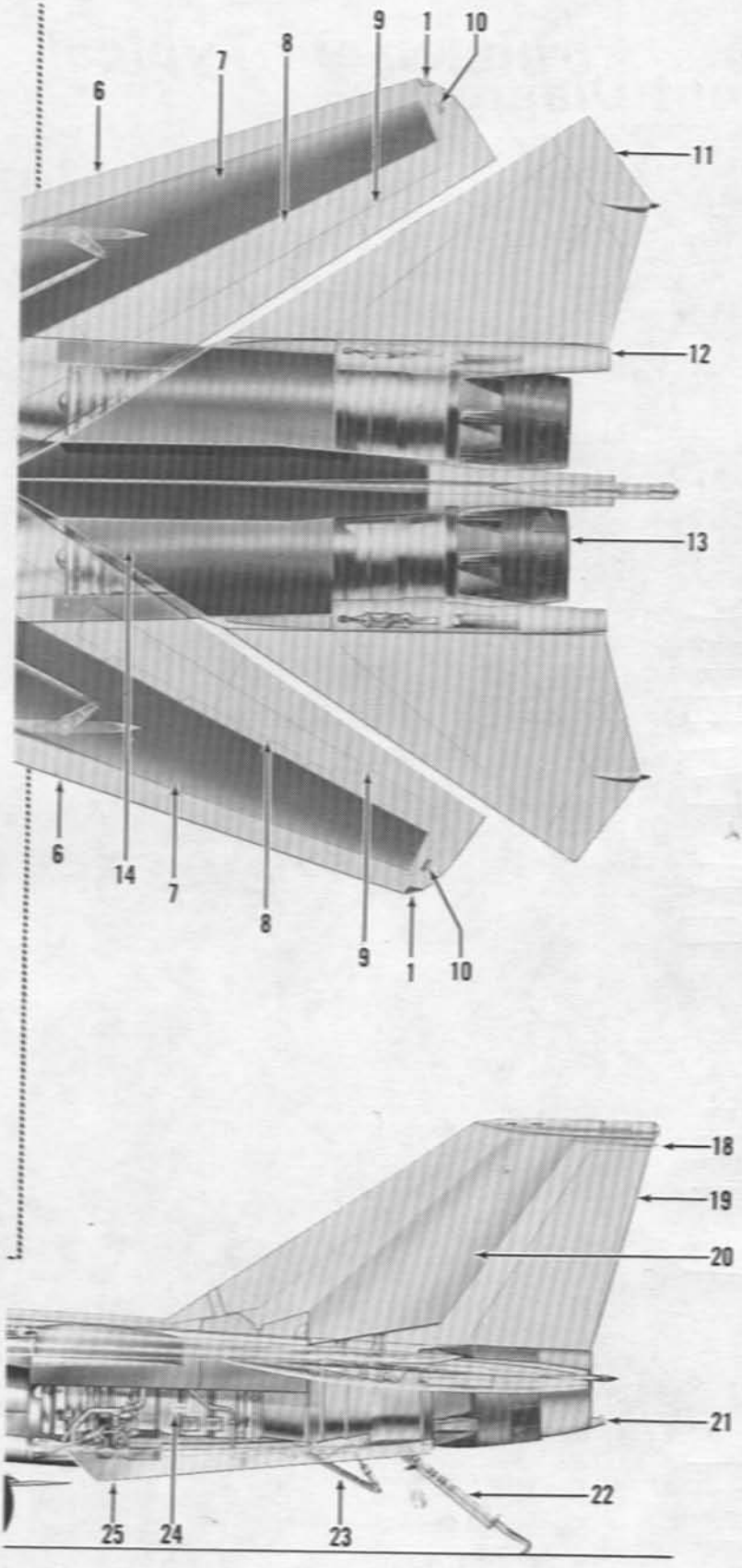


Figure 1-1. (Sheet 1)



1. Wing Position Lights.
2. Forward Fuel Tank.
3. Rotating Glove.
4. Pivoting Pylons.
5. Primary Hydraulic System Reservoir.
6. Slats.
7. Wing Fuel Tanks.
8. Spoilers.
9. Wing Flaps.
10. Wing Formation Lights (Upr & Lwr).
11. Horizontal Stabilizer.
12. Speed Bumps.
13. Engines.
14. Aft Fuel Tank.
15. Utility Hydraulic System Reservoir.
16. Air Refueling Receptacle.
17. Anti-Collision Lights (Upper & Lower).
18. Tail Position Light.
19. Rudder.
20. Fuel Vent Tank.
21. Fuel Dump/Vent Outlet.
22. Arresting Hook.
23. Tail Bumper.
24. Fuselage Formation Lights (4).
25. Strake (2).
26. Forward Landing Gear Door/Speed Brake.
27. Air Conditioning System Cooling Air Intake.
28. Blow-in Doors.
29. Spike.
30. Splitter Vane.
31. Fuel System Precheck Selector Panel.
32. Single Point Refueling Adapter Receptacle.
33. Aft Electronic Equipment Bay.
34. Weapons Bay.
35. Strike Camera.
36. Forward Electronic Equipment Bay.
37. Pitot Static Probe.

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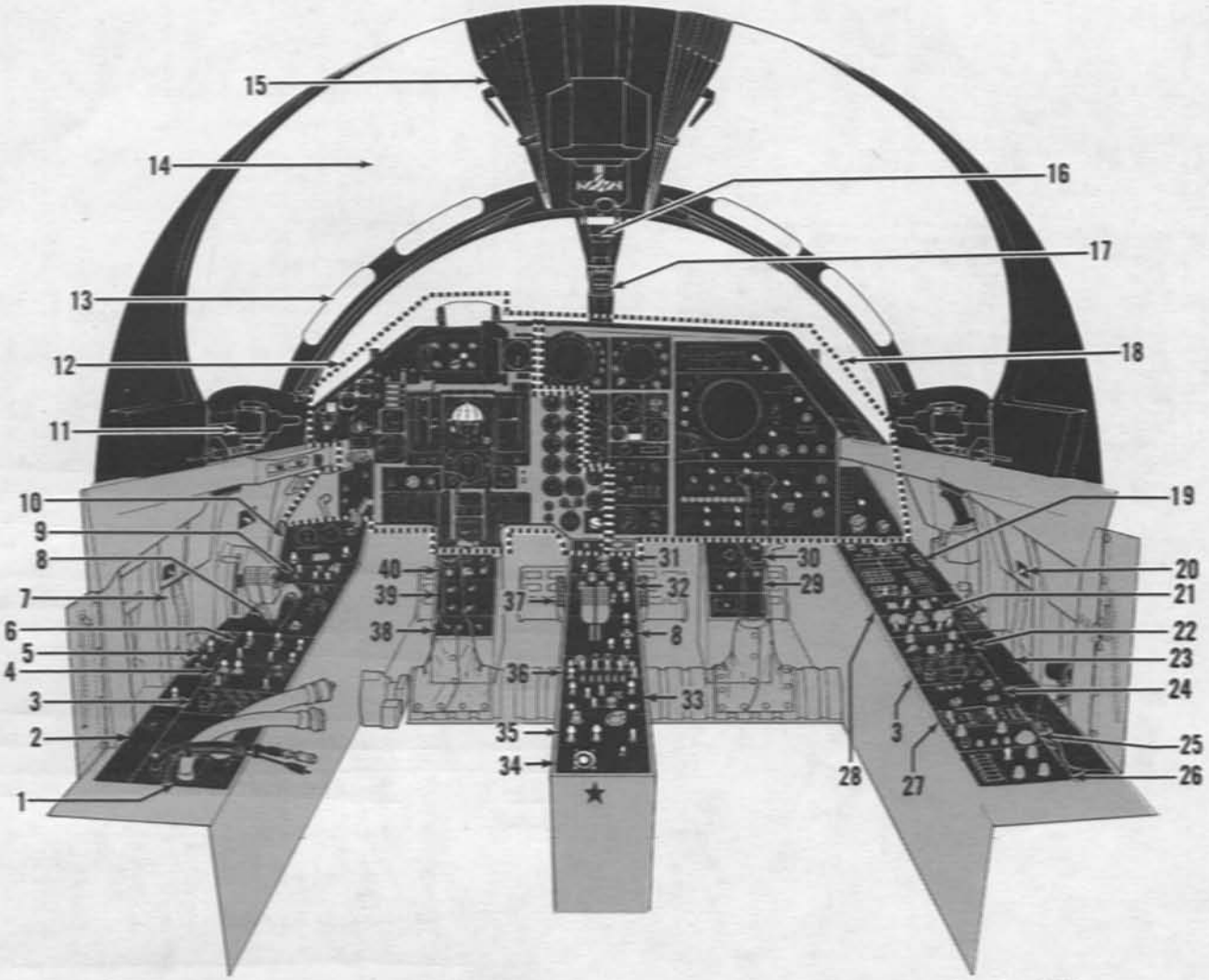
Figure 1-1. (Sheet 2)

Crew Station General Arrangement (Typical)



1. Left Station Oxygen-Suit Control Panel (See fig 1-45).
2. Oxygen Gage Panel (See fig. 1-46).
3. Interphone Panel (2) (See fig. 1-53). ★
4. Auxiliary Flight Control Panel (See fig. 1-25).
5. Flight Control Switch Panel (See fig. 1-26).
6. Autopilot Damper Panel (See fig. 1-27).
7. Left Sidewall (See fig. 1-18).
8. Throttle Panel (2) (See fig. 1-4).
9. Miscellaneous Switch Panel (See fig. 1-56). ★
10. Auxiliary Gage Panel (See fig. 1-14).
11. Internal Canopy Latch Handles (2).
12. Left Main Instrument Panel (See fig. 1-5).
13. Mirrors (4).
14. Canopy.
15. Thermal Curtain (2).
16. Canopy Center Beam Assembly.
17. Magnetic Compass.
18. Right Main Instrument Panel (See fig. 1-32).
19. Weapons Control Panel (See fig. 1-62). ★
20. Right Sidewall (See fig. 1-42).
21. Attack Radar Control Panel (See fig. 1-70).
22. ★ HF Radio Control Panel (See fig. 1-52).
23. Strike Camera Control Panel (See fig. 1-66).
24. * ECM Control Panel.
25. * CMRS Control Panel.
26. * CMDS Control Panel.
27. ★ * ECM Pod Control Panel.
28. * ECM Destruct Control Panel.
29. Burst Control and Target Elevation Panels (See fig. 1-60).
30. ILS Control Panel (See fig. 1-55).
31. Fuel Control Panel (See fig. 1-8).
32. TFR Control Panel (See fig. 1-77)
33. Electrical Control Panel (See fig. 1-10).
34. ★ Scope Camera Control Panel (See fig. 1-68)
35. Air Conditioning Control Panel (See fig. 1-40).
36. IFF Control Panel (See fig. 1-57).
37. Ejection Handles (2).
38. ★ Antenna Select Panel (See fig. 1-58).
39. Windshield Wash/Anti-Icing Control Panel (See fig. 1-43).
40. Compass Control Panel (See fig. 1-30).

*See T.O. 1F-111E-1-2

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Figure 1-2. (Sheet 1)

41. Hood Stowage Compartment.
42. Air Diffusers (2).
43. Liquid Containers (2).
44. Circuit Breaker Panel (See fig. 1-12).
45. Letdown Chart Stowage Compartment.
46. Internal Lighting Fuse Panel (2).
47. Lighting Control Panel (See fig. 1-38).
48. Utility Light.
49. Quick Rescue Kit.
50. Letdown Chart Holder Stowage Compartment.
51. Ground Check Panel (See Fig. 1-29)
52. Food Stowage Compartment.
53. Relief Container Stowage Compartment (2)
54. Right Station Oxygen Suit Control Panel (See fig. 1-45).

*

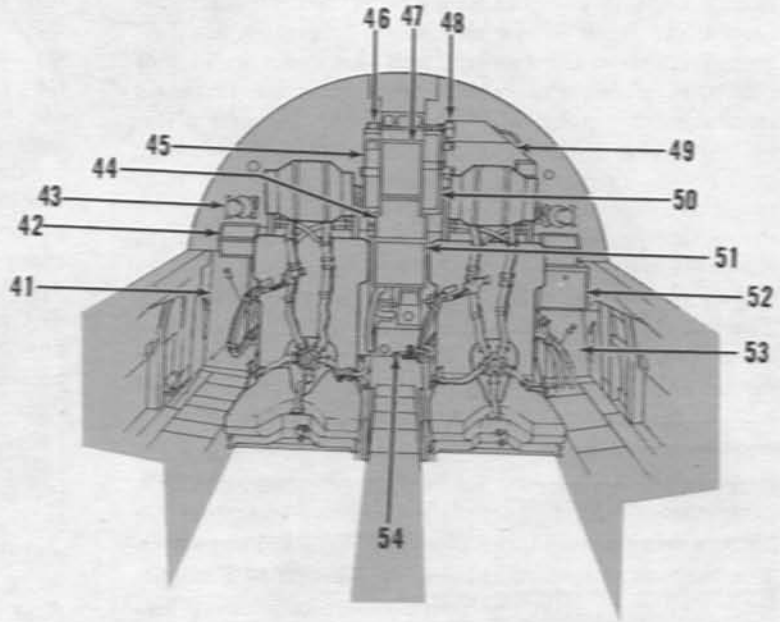

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Figure 1-2. (Sheet 2)

AIRCRAFT DIMENSIONS.

Length (overall including pitot static boom) — 75 feet, 6.5 inches

Wing span (wings swept) — 32 feet

Wing span (wings extended)—63 feet

Height (to top of vertical tail) — 17 feet, 1.4 inches

Refer to Section II for turning radius.

AIRCRAFT WEIGHT.

The aircraft operating weight is approximately 48,700 pounds. This weight includes two crewmembers (430 lbs), engine oil (60 lbs), unusable fuel (292 lbs), cooling water (229 lbs), oxygen (25 lbs), weapon bay gun (1100 lbs), and nose wheel well trim ballast (720 lbs).

Note

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 an aircraft commander (AC) and a weapons system officer (WSO), seated side-by-side.

ENGINES.

The aircraft is powered by two Pratt and Whitney TF-30 sixteen-stage axial flow turbofan engines equipped with afterburners. The engines are mounted side by side in the fuselage and are interchangeable. The sea level, standard day uninstalled thrust rating of the engine is in the 10,000 pound class in military power and in the 18,000 pound class in afterburner. Provisions are made for starting the engines with an external pneumatic ground starter cart. Also the left engine has the capability of being started by means of a pyrotechnic cartridge. 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 blow-in doors located in the outboard side of each nacelle. Boundary layer diverter ducts 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. Engine compressor bleed air from the sixteenth stage is utilized for the following functions:

- Cockpit, weapon bay and electronic equipment bay air conditioning and pressurization.
- Fuel tank and hydraulic reservoir pressurization.
- Engine inlet anti-icing.
- Engine inlet vortex destroyers.
- Windshield wash and rain removal.
- Throttle boost.
- Canopy and wing seals.
- Starting opposite engine.
- Engine nacelle ventilation and hydraulic oil, engine oil and constant-speed drive oil cooling on the ground.

Twelfth stage compressor bleed air is used for engine inlet guide vane anti-icing. Seventh stage compressor bleeds open under certain conditions to prevent compressor stall.

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 unnecessary 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 the 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 mach number which is provided from the central air data computer.

Note

Malfunctions of the CADC are normally indicated by the CADC caution lamp. However, failures can occur which result in incorrect mach data from the CADC to the fuel control unit without an accompanying CADC caution lamp. The effect of a CADC mach failure on the fuel control unit can occur only when the landing gear handle is in the UP position and will manifest itself with a sudden reduction in engine thrust. This malfunction will also result in an abnormally high mach indication on the AMI.

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 springloaded 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 springloaded 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 (AB) 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 springloaded blow-in doors, located near the aft end of the afterburner are provided to allow outside air into the engine to increase total engine thrust under certain flight conditions. The doors will remain open until inside engine pressure is greater than outside pressure plus the spring tension of the doors. 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 afterburner with modulated fuel control in each zone can be selected to provide fully variable throttle settings between minimum and maximum AB. When the throttle is advanced for afterburner initiation and when high pressure compressor speed exceeds the afterburner arming speed (79-84 percent N_2), 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 to initiate afterburner operation. With the advancement of the throttle into AB, the afterburner igniter valve releases an auxiliary squirt of fuel which is injected just aft of the fourth stage turbine; then zone 1 fuel flow begins. After zone 1 flow begins, initial afterburner ignition is provided by a hot streak ignition system. The igniter valve injects a slug (main squirt) of fuel into number 4 combustion chamber creating a local overrich mixture. This fuel is ignited by the combustion chamber fire and the rich mixture results in a longer flame that burns past the turbines to provide hot streak ignition for the auxiliary squirt, which in turn ignites zone 1. Completion of the main squirt into number 4 combustion chamber provides a signal for cessation of the auxiliary squirt. If afterburner operation is not achieved, the throttle must be retarded to MIL or below and readvanced into AB to repeat the above series of events required for afterburner ignition.

ENGINE INLET SPIKES.

Engine inlet air flow is regulated throughout the entire aircraft speed range in order to maintain maximum engine performance. This regulation of the air flow 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 local mach and diffuser exit mach pressure ratios are sensed by 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 one-shot pneumatic override system is provided to position and lock the spike full forward and fully contracted.

Engine Fuel Control System

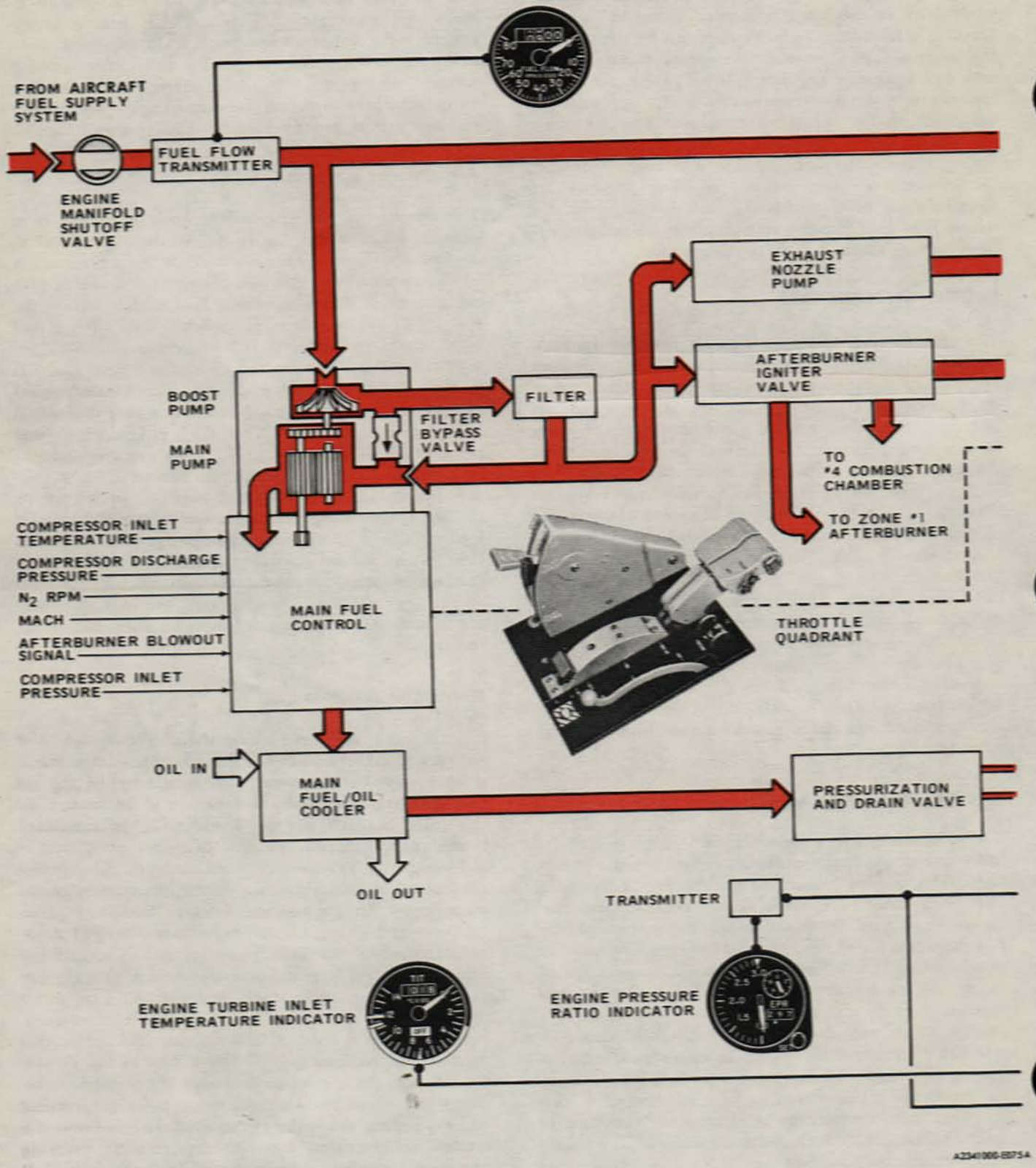


Figure 1-3. (Sheet 1)

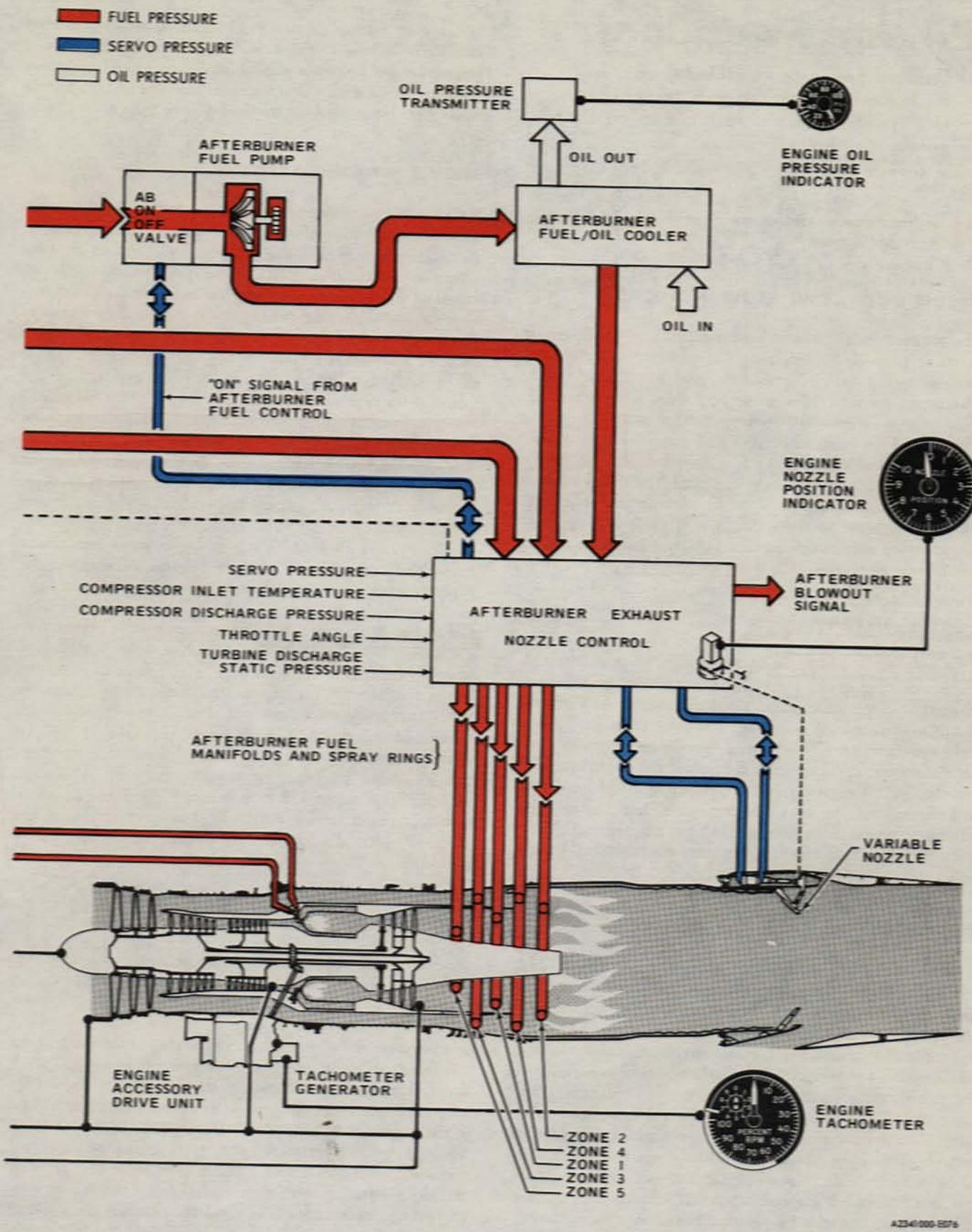


Figure 1-3. (Sheet 2)

Note

Once the pneumatic override has been used, the hydraulic shuttle valves in the spike control system must be repositioned on the ground with hydraulic pressure off.

An electronic anti-icing system prevents ice formation on the sensors. Refer to "Anti-Icing and Defog Systems," this section.

ENGINE INLET COWL BLOW-IN DOORS.

The engine inlet cowls are equipped with blow-in doors to provide an opening for additional air to the engine during ground operation and low speed flight. The doors are free floating so that they will automatically assume a position corresponding to the pressure differential between inlet duct and outside air pressure. On the ground and during low speed flight the doors will assume a position between full open and closed as this differential pressure varies with engine demand. As speed is increased the doors will close and remain closed when ram effect increases inlet pressure to above outside air pressure.

VORTEX DESTROYERS.

The possibility of ingestion of foreign objects into the engine during ground operation is reduced by an aerodynamic screen of engine bleed air, which is directed down and outboard beneath each inlet through vortex destroyer air jets. The vortex destroyers 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. When the weight of the airplane is on the landing gear a ground safety switch, located on the landing gear automatically activates the vortex prevention air screen.

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 non-afterburner operation except for ground engine idle at which time it is positioned fully open for minimum thrust. The nozzle closes when the corresponding throttle is advanced 3 degrees above IDLE. If afterburner blowout occurs, the blowout signal 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 throttle must first be moved to the mil-power range.

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. 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 over the OFF ramp 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 opens on the last engine to be started. Ignition is also cut off when the throttle is retarded over the OFF ramp. An automatic circuit energizes in the event of a combustion chamber flameout by sensing the rate of change of burner pressure. This is accomplished by an automatic ignition switch which will remain activated for 15 to 60 seconds depending on compressor discharge pressure.

ENGINE STARTING SYSTEM.

Several means are provided for starting the engines. The left engine can be started by pyrotechnic cartridge, both engines can be started by external pneumatic pressure, and once either engine is running the remaining engine can be started by pneumatic crossbleed from the operating engine. The left engine is equipped with a cartridge-pneumatic starter to provide the flexibility of operation without ground support equipment. The right engine is equipped with a pneumatic starter only. Electrical power required for starting can be obtained from either an external ground source or the aircraft battery. When starting the left engine with the cartridge, the cartridges is ignited by placing the ground start switch to CARTRIDGE and lifting the left throttle out of the OFF position. When starting the engines with a pneumatic source, either external or crossbleed,

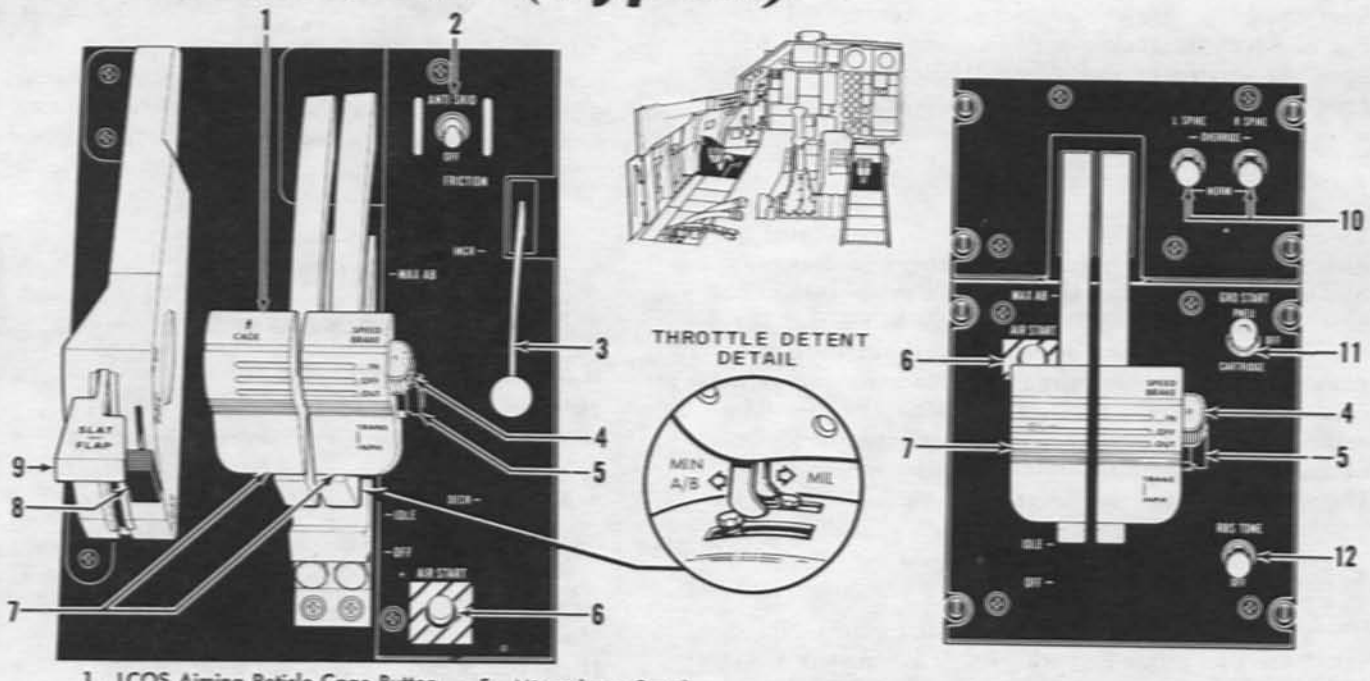
placing the ground start switch to PNEU and lifting the left or right throttle out of the OFF position opens the starter pressure shutoff valve, on the engine being started, and allows pneumatic pressure to drive the respective starter. After a pneumatic start the ground start switch will return to OFF when the centrifugal cutout switch in the starter on the second engine started opens. This will occur at 38 to 41 percent on the left engine and at 45 to 48 percent on the right engine. This breaks the starter control circuit and allows the starter pressure shutoff valve to close, shutting off the pneumatic pressure. In the event an engine start was not initially attained, the starter may be energized again when engine rpm falls below 20 percent. Two spare cartridges can be carried in the main landing gear wheel well.

ENGINE CONTROLS AND INDICATORS.

Throttles.

A set of throttles (7, figure 1-4), is provided for each crew member. 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. Pneumatic power boost, from the cabin pressurization system, is provided to assist throttle movement. The force required to move the throttles varies from 2 to 30 pounds, with pneumatic boost, depending on the position of the friction lever. In the event pneumatic boost is lost, the force required to move the throttles is from 10 to 40 pounds depending on the friction lever position. Both sets of throttles have positions marked OFF, IDLE, MIL, and MAX AB, respectively. Only the left set of throttles can be raised to go into or from the OFF position. The right set of throttles cannot be used for engine starting or shutdown. When the left set of throttles are lifted to move them out of the OFF position, the throttle starter switches are actuated. If the ground start switch is in the CARTRIDGE position, lifting the left throttle of the left set will automatically fire the left engine starter cartridge. If the ground start switch is in the PNEU position, lifting either throttle of the left set will open the starter pneumatic pressure shutoff valve on the respective engine to allow starting by pneumatic pressure. Movement of the throttle over the OFF ramp activates the engine ignition system. An

Throttle Panels (Typical)



1. LCOS Aiming Reticle Cage Button.
2. Anti-Skid Control Switch.
3. Throttle Friction Lever.
4. Speed Brake Switch.
5. Microphone Switch.
6. Airstart Button.
7. Throttles.
8. Flap/Slat Handle Gate Release.
9. Flap/Slat Handle.
10. Spike Control Switches.
11. Engine Ground Start Switch.
12. RBS Tone Switch.

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Figure 1-4.

adjustable detent at the MIL position provides a means of readily selecting this position. A detent is also provided at the minimum AB position. To attain the minimum AB detent position the throttle must first be advanced into the afterburner range and then retarded until the detent is felt. Refer to figure 1-4 for a detail of both the MIL and minimum AB throttle positions. The left throttle of the left set includes a cage-gyro switch for caging the LCOS gyros. The right throttle of each set includes a microphone switch and a speed brake switch.

Engine Ground Start Switch.

The engine ground start switch (11, figure 1-4), located on the center throttle panel, is a three position switch marked PNEU, OFF and CARTRIDGE. The switch is solenoid held in the PNEU and CARTRIDGE positions and is spring-loaded to and locked in the OFF position. The switch toggle must be pulled out before it can be moved to either PNEU or CARTRIDGE. Placing the switch to either the PNEU or CARTRIDGE position supplies power to arm the throttle start switches. With the switch in the PNEU position lifting either throttle out of the OFF position allows electrical power from the respective throttle start switch to open the starter pressure shutoff valve on the engine being started. With the switch in the CARTRIDGE position lifting the left throttle out of the off position allows electrical power from the throttle start switch to fire the cartridge. A centrifugal cutout switch in the starter of the last engine started will open the circuit to the solenoid holding the engine ground start switch and it will return to OFF.

Airstart Buttons.

Two airstart pushbuttons (6, figure 1-4), one located on each throttle panel, respectively, provide a means of obtaining ignition for air starting the engines. The buttons are marked AIR START. When either airstart button is momentarily 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 a minimum of 55 seconds after the airstart button is released, thereby providing ignition for this length of time.

Ground Ignition Cutoff Switch.

The ground ignition cutoff switch (10, figure 1-29), located on the ground check panel, is labeled GRD IGNITION and has two positions marked NORM and OFF. When the switch is in OFF, a relay is energized, which deactivates the engine electrical ignition system for both engines by grounding the ignition alternator output. When the switch is in the NORM position, the relay is deactivated and the ignition circuits are not grounded through this relay.

Mach Trim Test Switch.

The mach trim test switch (3, figure 1-29) located on the ground check panel, has two positions marked NORM and TEST. The switch is provided for maintenance ground check operation of the engine mach lever on the fuel control unit.

Compressor Bleed Valve Control Switches.

Two engine compressor bleed valve control switches (4, figure 1-56), located on the miscellaneous switch panel, provide automatic or manual control of the engine compressor bleed valves to aid in the prevention of compressor stalls. The switches are labeled COMPRESSOR BLEED with an L and R for the respective engine. Each switch has three positions marked AUTO, OPEN and CLOSE. The switches are locked in each position and must be pulled out before they can be moved. With the switches in AUTO the compressor bleed valve on each engine will automatically open when one or more of the following conditions occur:

- Throttle settings are below MIL when the aircraft is on the ground.
- Speed is greater than mach 0.44 and angle of attack is greater than 14 degrees.
- Throttle settings are below MIL and speed is greater than mach 1.1.
- Speed is greater than mach 1.75 $\pm$0.10.

The CLOSE position of the switches is provided to override the automatic open signal for ground checkout of the system. The OPEN position of the switches is provided to open the valves in the event the automatic feature fails.

WARNING

If a compressor bleed valve fails in the open position a thrust loss on the engine involved of 7.5 percent at military power to 17 percent at max AB can be expected under standard day conditions at speeds between 0 to 0.3 mach.

Note

If the compressor bleed valve position indicator fails to indicate bleed open for the above conditions the bleed switch should be placed in the OPEN position. If the indicator still fails to indicate OPEN then continued operation at the above conditions will probably result in compressor stall.

Spike Control Switches.

Two spike control switches (10, figure 1-4), located on the center throttle panel, are labeled L SPIKE and R SPIKE respectively. The switches are lever lock type switches with two positions marked OVERRIDE and NORM. In the NORM position the spikes are automatically controlled to maintain maximum engine performance. When either switch is positioned to OVERRIDE, pneumatic pressure is applied to the spike actuator to move the spike to the full forward and fully contracted position. The switch must be pulled out of the lock before it can be moved from either position.

Spike Test Buttons.

Two spike test buttons (2, figure 1-29), located on 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 positions.

Engine Tachometers.

Two engine tachometers (31, figure 1-5), located on the left main instrument panel, indicate the percent of RPM of the high pressure compressor (N_2) 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.

Compressor Bleed Valve Position Indicator.

A flip-flop type compressor bleed valve position indicator (5, figure 1-56), located on the miscellaneous switch panel, is provided to indicate the commanded positions of each engine compressor bleed valve. When power is off the indicator shows crosshatched. When power is on the following indications are provided:

- NONE—Neither valve open
- BOTH—Both valves open
- LEFT—Left valve open (right valve closed)
- RIGHT—Right valve open (left valve closed).

Engine Fuel Flow Indicators.

Two engine fuel flow indicators (33, 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 100 pph.

Note

Fuel flow indications may fluctuate as much as ± 300 pph for all flow rates. Fluctuation in excess of this amount must be investigated.

Engine Nozzle Position Indicators.

Two engine nozzle position indicators (34, 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 left main ac bus.

Note

The nozzle position also represents an approximate percent of the available AB thrust.

Engine Oil Pressure Indicators.

Two engine oil pressure indicators (36, 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.

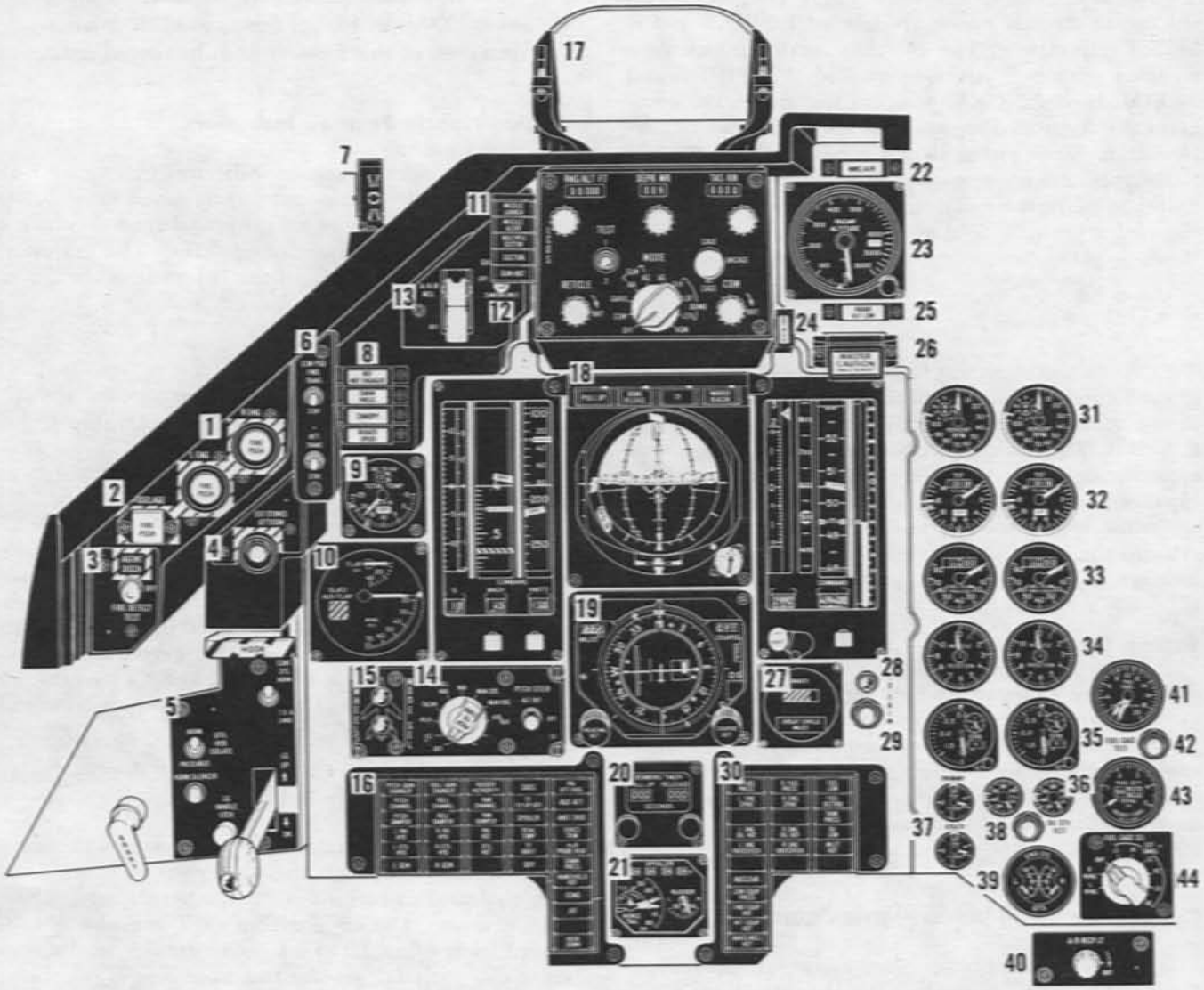
Engine Pressure Ratio Indicators.

Two engine pressure ratio (EPR) indicators (35, 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. The indicators are supplied 115 volt ac power from the left main ac bus.

Engine Turbine Inlet Temperature Indicators.

Two engine turbine inlet temperature (TIT) indicators (32, 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. Power to the TIT indicators is supplied from the 28 volt dc engine start bus. Consequently the indicators will operate with the bat-

Left Main Instrument Panel (Typical)



1. Engine Fire Pushbutton Warning Lamps.
2. Fuselage Fire Pushbutton Warning Lamp.
3. Agent Discharge/Fire Detect Test Switch.
4. External Stores Jettison Button.
5. Landing Gear Control Panel (See fig. 1-15).
6. Pilot's ECM Pod Control Panel ★ (See T.O. 1F-111E-1-2)
7. Angle-of-Attack Indexer.
8. Left Warning and Caution Panel.
9. Total Temperature Indicator.
10. Wing Sweep Flap/Slat Position Indicator.
11. Upper Warning and Caution Panel.
12. Gun/Camera Control Switch.
13. Air/Air IR Missile Switch.
14. Instrument System Coupler Control Panel.
15. Landing Gear Position Indicator Lamps.
16. Left Main Caution Lamp Panel.
★ 17. Lead Computing Optical Sight and Control Panel, (See fig. 1-81)
18. Upper Warning and Caution Lamp Panel.
19. Integrated Flight Instruments, (See fig. 1-33).
★ 20. Dual Bombing Timer, (See fig. 1-65)
21. Control Surface Position Indicator.
22. Nose Wheel Steering/Air Refueling Indicator Lamp.
23. Radar Altimeter Indicator.
24. Stall Warning Lamp.
25. Radar Altitude Low Warning Lamp.
26. Master Caution Lamp.
27. Bomb Nav Distance-Time Indicator.
28. Takeoff Trim Indicator Lamp.
29. Takeoff Trim Button
30. Right Main Caution Lamp Panel.
31. Engine Tachometers.
32. Engine Turbine Inlet Temperature Indicators.
33. Engine Fuel Flow Indicators.
34. Engine Nozzle Position Indicators.
35. Engine Pressure Ratio Indicators.
36. Engine Oil Pressure Indicators.
37. Hydraulic Pressure Indicators.
38. Oil Quantity Indicator Test Button.
39. Oil Quantity Indicator.
40. Air Refueling Receptacle Lights Control Knob.
41. Fuselage Fuel Quantity Indicator.
42. Fuel Quantity Indicator Test Button.
43. Total/Select Fuel Quantity Indicator.
44. Fuel Quantity Indicator Selector Knob.

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Figure 1-5.

tery switch ON. 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 1-37). When lighted, the letters L ENG SPIKE and R ENG SPIKE, respectively are visible. A spike caution lamp lights when the aircraft mach number is less than 0.35 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.

Engine Oil Hot Caution Lamps.

The two engine oil hot caution lamps are located on the main caution lamp panel (figure 1-37). When the oil temperature of either engine exceeds 250 degrees F, the associated lamp will light. When lighted, the letters L ENG OIL HOT; and R ENG OIL HOT will be visible in the respective lamp.

Engine Overspeed Caution Lamps.

Two amber engine overspeed lamps, one for each engine, are located on the main caution lamp panel (figure 1-37). When lighted the letters L ENG OVERSPEED and R ENG OVERSPEED are visible. An engine overspeed lamp will light at N1 compressor speeds of approximately 10,500 rpm and above. In addition, the lamps will light prior to engine start, provided there is electrical power on the airplane, and will go out prior to reaching idle rpm. The lamps operate on 28 volt dc electrical power from the essential dc bus.

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 Pushbutton Warning Lamps.

Two fire pushbutton warning lamps (1, figure 1-5), labeled 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 switch to the affected engine. Depressing the button again will open the fuel shutoff valve and disarm the fire extinguisher agent discharge valve; however, the hydraulic shutoff valves will remain closed. The buttons are covered by frangible covers to provide a visual indication when the buttons have been actuated.

WARNING

Caution must be exercised to prevent inadvertently depressing the wrong push button and shutting down the good engine since the hydraulic shutoff valves cannot be reopened in-flight.

Agent Discharge/Fire Detect Test Switch.

The agent discharge/fire detection test switch (3, figure 1-5), located on the left main instrument panel, is a three position lock lever switch marked AGENT DISCH, OFF and FIRE DETECT TEST. The switch is spring loaded to the OFF position and is locked out of the AGENT DISCH position to prevent inadvertent actuation. To move the switch to AGENT DISCH it must be pulled out of the lock. Momentarily positioning 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. For other functions of this switch, refer to "Fuselage Fire and Wheel Well Overheat Systems," this section.

Fire Detection System Ground Test Switches.

Two fire detection system ground test switches (1, figure 1-29), located on the ground check panel, are labeled R ENG and L ENG. The switches have three positions marked CONTROL BOX, NORM and ELEMENT and are used to ground check the system circuitry during maintenance or troubleshooting. Only one circuit can be checked at a time. The switches are springloaded to the NORM position.

ENGINE OPERATION.

Engine Acceleration.

Engine acceleration time is severely affected by the amount of compressor discharge air being bled from the engine and by outside temperature. The engine may require 15 seconds to accelerate from idle to military when air conditioning is taken from that engine during ground operation. In flight this effect is minimized but during final approach for landing, engine acceleration may require as much as 10 seconds to increase thrust from idle to military.

Engine Reset Operation.

During afterburner operation under certain combinations of high ram pressures and temperature, the main engine fuel control unit will automatically schedule higher engine parameters of fuel flow, N_2 rpm and turbine inlet temperature. This is normal operation known as engine reset. Under these conditions, the engine rpm and turbine inlet temperature will increase. This may occur at airspeeds in excess of 660 KIAS and is more pronounced at low altitude.

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). 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 specification and servicing location, refer to figure 1-84.

ENGINE OIL QUANTITY INDICATOR.

The engine oil quantity indicator (39, figure 1-5), located on the left main instrument panel, 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 ac power from the left main ac bus and 28 volt dc power from the main dc bus.

Note

- The indicated oil quantity exhibits variations during normal operations. When a cold engine is started, the oil quantity indication may drop as much as five quarts at idle power settings. After an engine has warmed up, the oil quantity indications may vary as much as three quarts (increase or decrease) at various power settings from idle thru military.
- If the oil quantity indicating system for either engine malfunctions, that indicator will drive to below zero and the oil low caution lamp will be inoperative for that engine. The oil low caution lamp will, however, continue to monitor the oil quantity for the other engine. To confirm that the malfunction is in the oil quantity indicating system rather than an actual oil low condition, the oil low caution lamp may be checked by depressing the malfunction and indicator lamp test button, located on the lighting control panel.

ENGINE OIL QUANTITY INDICATOR TEST BUTTON.

The engine oil quantity indicator test button (38, figure 1-5), located on the left main instrument panel 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. Also, the oil low caution lamp will light when the oil quantity test button is depressed, provided the oil quantity measuring system is operating properly. When the button is released the pointers will return to their previous indications and the oil low caution lamp will go out.

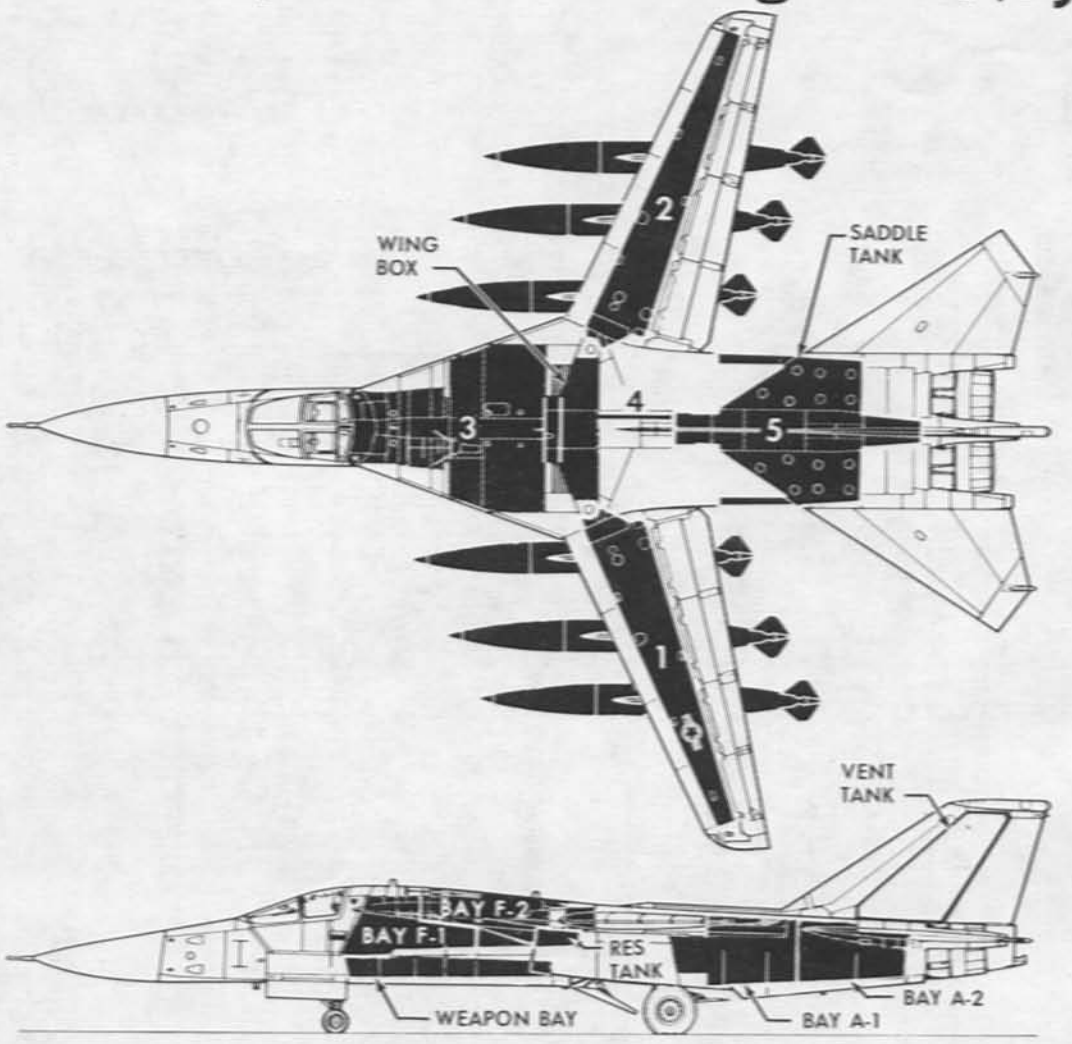
OIL LOW CAUTION LAMP.

An oil low caution lamp (figure 1-37), located on the main caution lamp panel, lights any time the oil level in either left or right engine oil supply tank drops to four (4) quarts usable oil remaining. Also, the lamp lights when the OIL QTY TEST button is depressed. When the lamp is lighted the words OIL LOW are visible.

FUEL SUPPLY SYSTEM.

The fuel supply system (figure 1-7) 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. The fuel system employs twelve fuel pumps, of which six deliver fuel to the engine and six are used to transfer fuel from the wing tanks and weapons bay tanks to the fuselage

Fuel Quantity and Tank Arrangement (Typical)



	LOCATION	QUANTITY			
		USABLE FUEL		FULLY SERVICED	
		GALLONS	POUNDS	GALLONS	POUNDS
1	LEFT WING INTERNAL TANK	389.2	2,530	390.7	2,540
2	RIGHT WING INTERNAL TANK	389.2	2,530	390.7	2,540
3	FORWARD FUSELAGE TANK	2,808.3	18,254	2,825.2	18,364
4	FUEL LINES	37.1	241	53.4	347
5	AFT FUSELAGE TANK	1,428.8	9,287	1,430.9	9,301
	TOTAL	5,052.6	32,812	5,090.9	33,092

NOTES:

1. These are average figures based on single point refueling at normal ramp altitude. Weights based on JP-4 fuel at 6.5 pounds per gallon. (Std. Day Only).

2. Each external tank, when carried, will have the following capacities.

USABLE FUEL		FULLY SERVICED	
GALLONS	POUNDS	GALLONS	POUNDS
	PIVOT PYLONS		
601.2	3,908	603.4	3,922
	FIXED PYLONS		
603.2	3,921	605.4	3,935

3. Weapons bay tanks, when carried, will have the following capacities.

USABLE FUEL		FULLY SERVICED	
GALLONS	POUNDS	GALLONS	POUNDS
	LEFT WEAPONS BAY TANKS		
269.2	1,750	272.3	1,770
	RIGHT WEAPONS BAY TANKS		
290	1,885	293.1	1,905

A0000000-E0179

Figure 1-6.

Fuel Supply System

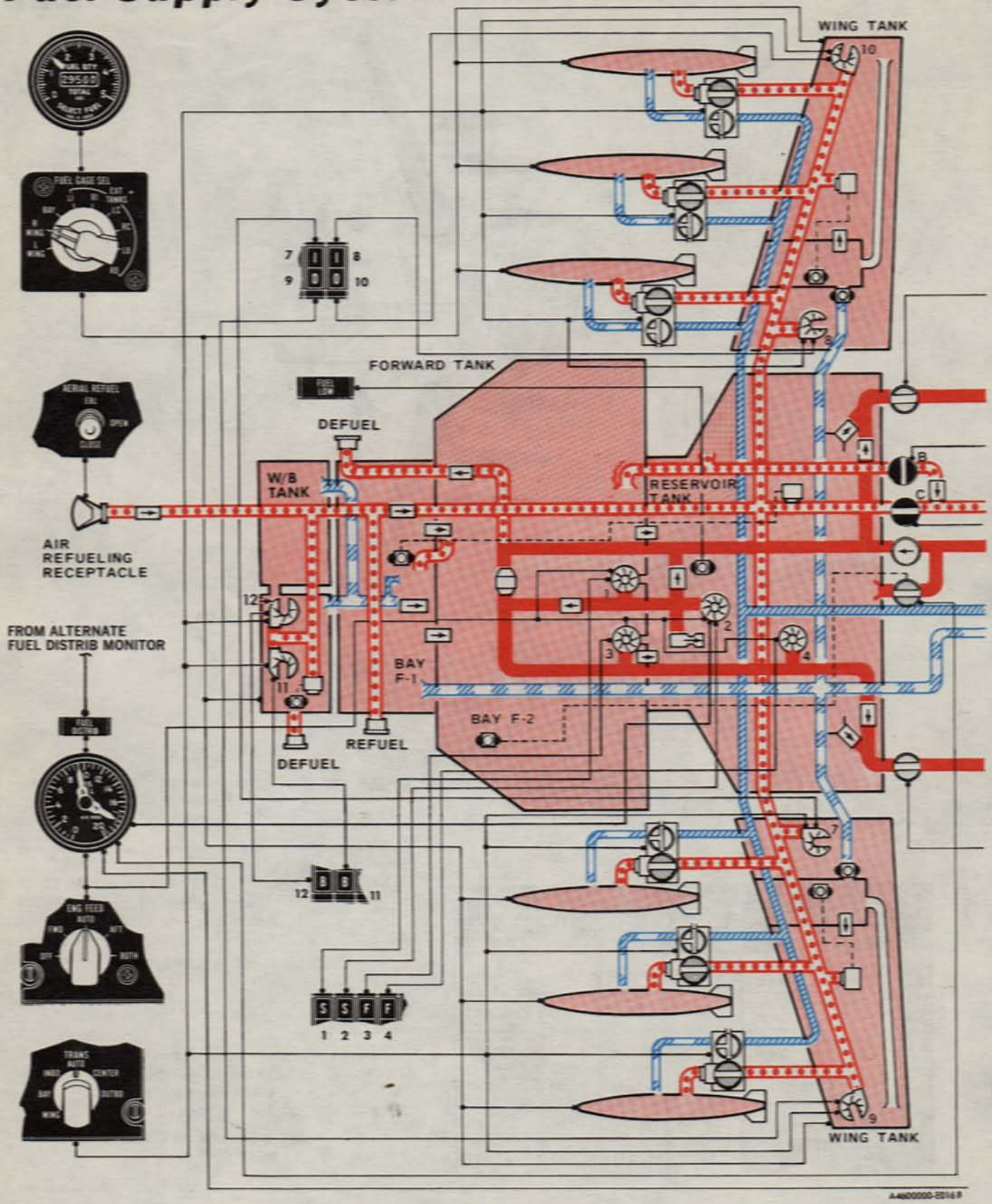


Figure 1-7. (Sheet 1)

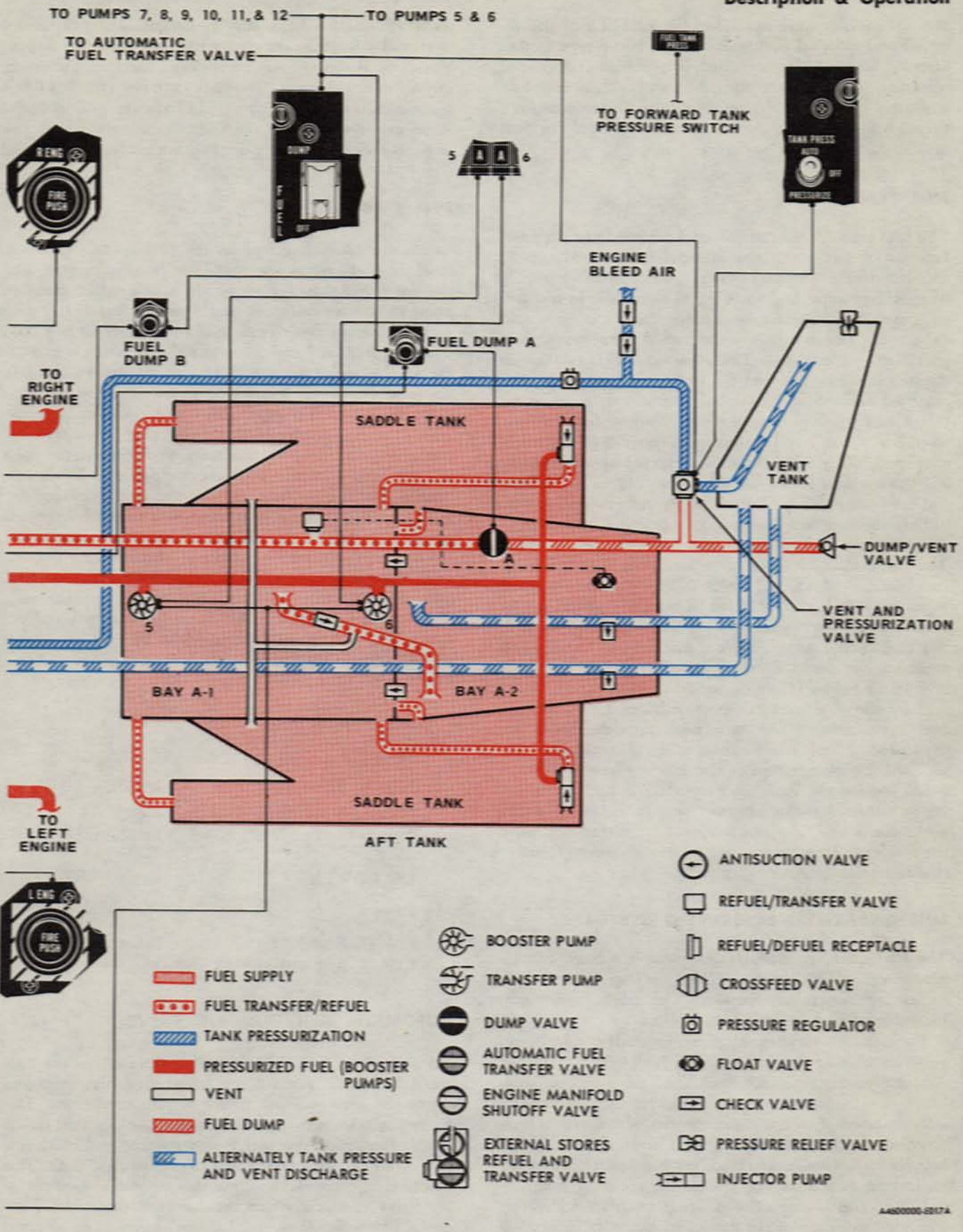


Figure 1-7. (Sheet 2)

tanks. Provisions are made for air refueling of the internal and external fuel tanks from a boom-type tanker aircraft. Single-point refueling is provided for ground servicing. All tanks are equipped with refuel automatic shutoff valves. Gravity refueling can be accomplished through filler caps in the wings and fuselage. For fuel tank capacities, refer to figure 1-6.

FUEL TANKS.

The fuel tanks consist of internal forward and aft fuselage tanks, left and right internal wing tanks, up to six detachable external wing tanks, two removable weapon bay tanks and an integral vent tank in the vertical stabilizer. See figure 1-6 for tank locations and capacities. The fuselage tanks are divided into compartments called bays. The forward fuselage tank is divided into bays F-1 and F-2 and a reservoir tank. The reservoir tank includes the fuel contained in the wing carry through box. Flapper valves allow fuel to flow from bay F-1 to bay F-2 and from bay F-2 to the reservoir tank. The reservoir tank reserves approximately 2500 pounds of fuel after all other fuel in the system has been used. A float switch in the reservoir tank provides a caution lamp indication when the fuel level in the reservoir tank drops below 2300 (±235) pounds. The aft tank is divided into bay A-1, incorporating two "saddle" tanks, and bay A-2. Interconnecting stand pipes provide fuel flow between bays A-1 and A-2, and ejector pumps transfer saddle tank fuel into bay A-1. All fuel in the wing external and internal tanks and weapons bay tanks must be transferred into the fuselage tanks before it can be used. All tanks are pressurized by cooled engine compressor bleed air to prevent fuel vaporization. The vent tank provides space for expansion of fuel in the system when all tanks are fully serviced. Booster pumps in the fuselage tanks are provided for engine feed, and transfer of fuel from the aft to forward tank. Transfer pumps in the internal wing tanks and weapon bay tanks transfer fuel into the fuselage. A pressurization system is used to transfer fuel from the external wing tanks.

FUEL QUANTITY MEASUREMENT SYSTEM.

The fuel quantity measuring system is a basic capacitance sensing type system. There are four independent indicating functions: forward, aft, select, and total. Each function consists of the following components: Capacitance sensors (tank units), located in each tank, provide a value proportional to fuel height/volume. An intermediate device measures the sum of tank units. A signal positions the indicator until a balance is established. The indicator consists of a servo, which responds to the amplifier. A gear train drives the rebalance wiper and a rebalance potentiometer. The four indicating functions are housed in the fuel quantity indicator and the total select fuel quantity indicator. The independence of the total fuel quantity indication is achieved by use of dual sensor tank units. Each indicating function has a density compensation capacitor that will always be covered with fuel until the respective tank is empty. An exception occurs in the select circuit in the aft tank. The fuel gage test circuit substitutes a fixed value which should indicate 2000 pounds. A normal response by the indicators verifies that the indicator circuit is functioning normally.

FUEL PUMPS.

There are 12 fuel pumps in the fuel system. The six fuselage fuel pumps are dual inlet booster pumps, and the four wing fuel pumps are single inlet transfer pumps. Booster 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, 8 and 10 are in the right wing and 11 and 12 are in the left weapons bay tank. 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 supply, the fuel provided by pump 1 is circulated into the reservoir tank through a pressure relief valve. Number 2 pump is energized by the pressure sensing switch when on AFT feed, BOTH or when on AUTO feed and the fuselage fuel quantity indicator indicates less than approximately 8500 pounds differential between the forward and aft tanks. All pumps are controlled by 28 volt dc power and are energized by 115 volt ac power from the following electrical busses:

AC POWER

Pumps	*Bus*
1, 3, 6, 9, 10 and 11	R. Main
4, 5, 7, 8 and 12	Essential
2 only	L. Main

DC POWER

Pumps	*Bus*
4, 5, 7, 8 and 12	Essential
1, 2, 3, 6, 9, 10 and 11	Main

AUTOMATIC FUEL TRANSFER VALVE.

An automatic fuel transfer valve, located in the forward fuel tank, permits the transfer of fuel from the aft to the forward tank under certain conditions. The valve is electrically operated by the fuselage fuel quantity indicator or the fuel dump switch, and mechanically operated by a float valve in the forward tank. The mechanical float valve allows the automatic fuel transfer valve to open when the forward tank fuel level drops below 9,000 pounds. If the engine feed selector is placed to a position that will cause the aft tank

pumps to operate, the fuel will be transferred forward to maintain the 9,000 pound level until the aft tank is empty. Electrical operation of the automatic fuel transfer valve is described under "Automatic Fuel Distribution (Primary)" and "Fuel Dump System," this section.

ENGINE FUEL SUPPLY SYSTEM.

The engine fuel supply system functions in five modes to provide fuel flow to the engines and control the fuel distribution between the fuselage tanks. The five modes as selected with the engine feed selector knob are: AUTO, BOTH, FWD (forward), AFT and OFF. In the AUTO mode the fuselage fuel quantity indicator automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure an operational airplane center-of-gravity. Refer to "Automatic Fuel Distribution (Primary)," this section, for description of operation in the AUTO mode. In the OFF mode the engines are supplied with fuel by gravity (suction) from the forward tank. In the BOTH mode of operation, the left engine is fed from the forward tank and the right engine is fed from the aft tank. In this mode there is no automatic fuel distribution control and forward and aft tank fuel differential must be controlled by monitoring the fuselage fuel quantity indicator and manually selecting either FWD, AFT, or BOTH feed. In the event the fuselage fuel quantity indicator is inoperative or malfunctions, refer to "Abnormal Fuel Distribution/Indication," Section III. During FWD or AFT mode operation, both engines are fed from the forward or aft tank respectively. When on AFT feed, under conditions of high fuel flow, the forward standby pumps will assist in meeting the high demand on an aft tank. In the event of loss of electrical power to the fuel system the engines will gravity (suction) feed from the forward tank.

AUTOMATIC FUEL DISTRIBUTION (PRIMARY).

In the AUTO mode, the fuselage fuel quantity is controlled by the F (forward) and A (aft) pointers on the instrument that automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure aircraft center-of-gravity. The engines are supplied fuel from the forward tank or both tanks depending upon the position of the switches in the indicator. If the differential in the forward tank is greater than approximately 8500 pounds, as is the case when the tanks are fully serviced and until all wing and external fuel has been transferred into the forward tank, the indicator will turn the aft tank pumps off, and feed both engines from the forward tank. As the differential between the tanks is decreased to approximately 8200 pounds, the indicator will detect the proper fuel distribution and feed the left engine from the forward tank and the right engine from the aft tank. When the differential between the tanks decreases to approximately 7900 pounds, the indicator will open an automatic transfer valve, to transfer fuel forward and regain the proper fuel distribution. With the engine feed selector in AUTO, when the differential between the forward and aft tank pointers becomes less than 7,600 pounds or greater than 10,000 pounds, switches in the indicator will cause the fuel distribution caution lamp to light.

WARNING

The use of AUTO engine feed when the fuselage fuel quantity indicator is malfunctioning or inoperative could result in exceeding the center-of-gravity limits and loss of control of the aircraft.

ALTERNATE FUEL DISTRIBUTION MONITORING SYSTEM.

On aircraft 41 ♦ and those modified by T.O. 1F-111-673, an alternate fuel distribution monitoring system provides a means to monitor fuel distribution between the forward and aft tanks independent of the fuel quantity indication system. The system includes four fuel level sensing units and a control unit. Two of the sensors are installed in the forward tank and two in the aft tank. One sensor in the forward tank is located at a fuel level of approximately 12,000 pounds, the other at about 9000 pounds. Likewise the two in the aft tank are located at approximately 5300 pounds and 2500 pounds. When operating in OFF, FWD, AFT or BOTH and the forward tank fuel level drops below the 12,000 pound sensor, a signal will be provided to turn the fuel distribution caution lamp on if the 5300 pound sensor in the aft tank is covered. Likewise when the fuel level in the forward tank decreases to a point below the 9000 pound level, the fuel distribution caution lamp will light if the aft tank level is above the 2500 pound sensor. A 12 second time delay is provided to eliminate fuel distribution signals due to fuel sloshing. When operating in AUTO, the alternate fuel distribution monitoring system is a back-up to the normal system. If a malfunction occurs in the automatic fuel distribution control system that allows the actual fuel distribution to reach the above conditions, the alternate monitoring system will light the fuel distribution caution lamp and turn on the aft tank pumps if they were not operating.

FUEL TRANSFER.

In order to use the fuel in the external tanks, internal wing tanks, or weapons bay tanks, it must first be transferred to the fuselage tanks. Normally the tanks are emptied in the order of external, weapons bay, and then internal wing tanks. Fuel transfer is controlled by the transfer knob. The fuel level in the fuse-

lage is maintained by float valves which open or close refuel valves to allow transfer into the fuselage tanks any time they are not full. The refuel valves cannot be controlled from the cockpit. Transfer from any pair of external tanks, weapons bay tanks, or internal wing tanks can be manually selected. (Refer to "Fuel System Operation," this section.) When automatic transfer is selected, the transfer of fuel is automatically sequenced from the outboard, center, and inboard external tanks, weapons bay tanks, and then the internal wing tanks, in that order.

Note

Both external tanks in a pair must be empty before transfer will commence from the next pair.

When the weapons bay tank runs dry, a one minute delay will occur to assure complete scavenging of the tank before the wing tanks will transfer. Transfer from the weapons bay and internal wing tanks is effected by transfer pumps. Transfer from the external wing tank is accomplished by pressurizing the selected tanks with cooled engine compressor bleed air at 36 to 41 psi. When transferring from the weapons bay tanks, or 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. The exact fuel quantity where the individual wing pump lamps light cannot be established accurately because it depends upon a large number of variables; attitude, wing sweep, roll angle, load factors, fuel temperature and density, weapon loading, wing deflection, etc. However, for level flight with the wing sweep forward, the outboard pumps normally run out of fuel and cause the outboard pump low pressure lamp to light before the inboard pump lights. If the wings are swept AFT, the reverse is true.

FUEL PRESSURIZATION AND VENT SYSTEM.

Fuel system pressurization is provided to prevent loss of fuel from vaporization during flight. All tanks are pressurized by this system except the external tanks. Pressurization is provided by cooled engine compressor bleed air. The system functions in two modes: automatic and manual, as controlled by a fuel tank pressurization selector switch. In the automatic mode the tanks are pressurized when the landing gear is retracted. In this mode the system is automatically depressurized when the refueling receptacle door is opened or when the gear is extended. The tanks can also be pressurized by manually placing the fuel tank pressurization switch to PRESSURIZE in the event the automatic feature fails or if it is desired to pressurize the tanks with the air refueling receptacle door open or when the landing gear is down. The system maintains a pressure differential of 5 to 6 psi by means of a fuel tank vent and pressurization control valve. Should the pressure exceed 6 psi the valve will open and vent the excess air overboard through the dump/vent outlet at the rear of the fuselage. Except when transferring fuel from them, the external tanks are vented to atmospheric pressure through a vent port at the trailing edge of each pylon. Engine compressor bleed air at 36 to 41 psi is used to pressurize the external tanks for fuel transfer. The fuel transfer knob controls servo air to pressurize the external tanks. This is independent of the fuel tank pressurization system; however, engine bleed air must be available as selected by the air source selector.

FUEL DUMP SYSTEM.

The fuel dump system provides the capability of jettisoning fuel at a rate of 2300 pounds per minute. Fuel tank pressurization provides the force to jettison the fuel from the forward tank into the dump manifold and overboard through the vent/dump outlet at the aft end of the fuselage. This flow is controlled by motor operated dump valves A and B which receive power through circuit breakers located in the crew compartment. These two valves provide redundant shutoff capability for the dump system and valves are normally closed except during dumping operation. Dump valve B normally prevents fuel loss from the forward tank in the event of a broken refuel/dump line. Dump valve A normally prevents refuel and transfer flow from going overboard through the vent/dump outlet. In addition to dump valves A and B, dump valve C is provided. This valve is normally open but closes during dumping operation to prevent tank pressurization from flowing overboard through the dump line from the wings when the wing tanks are empty. Dump valve C receives power from dump A circuit breaker. The fuel dump system also utilizes the fuel transfer system to transfer fuel from the aft, bay and wing tanks to the forward tank. This is accomplished by relays which also receive power from dump B circuit breaker through the dump switch. When DUMP is selected, fuel immediately starts to transfer from the aft and wing tanks. Weapons bay tanks (if installed) will transfer before the wing tanks.

AIR REFUELING SYSTEM.

The air refueling system is 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 offset to the left and 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 slipway door is 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/transfer fuel manifold lines into the fuel tanks at a rate of 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. When the last tank refuel valve closes an increase in the refuel line pressure is sensed by a pressure switch which automatically provides a signal to unlatch the boom from the receptacle. 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. An emergency boom latch capability is provided to latch the boom in place in the event the boom will not latch in the receptacle during normal operation. The emergency boom latch function also provides pneumatic power to open the doors and extend the receptacle in the event utility hydraulic pressure is lost. Sufficient pneumatic pressure is available to operate the receptacle through two cycles (open and close) with 4 hook-ups during each cycle.

SINGLE POINT REFUELING SYSTEM.

The single point refueling system enables all aircraft 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.

Gravity refueling is accomplished through six filler caps and one vent cap 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 four filler caps in the fuselage: one each for F-1, F-2, A-1 and A-2 tanks. In addition to the filler cap located above the right saddle tank in bay A-1, a vent cap is provided above the left saddle tank. This cap must be removed to allow air to escape while the tank is being filled from the right side. To service the reservoir (trap) tank during gravity refueling, the booster pumps in the forward tank must be operated at least 2 minutes. The external tanks can be refilled by transferring fuel from the wing tanks. For detailed gravity refueling procedures refer to "Strange Field," Section II.

FUEL SYSTEM CONTROLS AND INDICATORS.

Engine Feed Selector Knob.

The engine feed selector knob (5, figure 1-8), located on the fuel control panel, has five-positions marked OFF, FWD, AUTO, AFT, and BOTH. When the knob is rotated to OFF, all fuel boost pumps are de-energized. Each knob position will energize the following pumps or place them on standby as indicated:

FWD —1, 2, 3 and 4 energized.

AUTO—Fwd & Aft fuselage fuel quantity indicator differential approximately 8200; 1, 3, 4, 5 & 6 energized; 2 on standby.

Fwd & Aft fuselage fuel quantity indicator differential approximately 7900 or less; 1, 3, 4, 5 & 6 energized, 2 on standby.

Fwd & Aft fuselage fuel quantity indicator differential approximately 8500 or greater; 1, 2, 3, & 4 energized.

AFT —1, 5 & 6 energized, 2 on standby.

BOTH—1, 3, 4, 5 & 6 energized, 2 on standby.

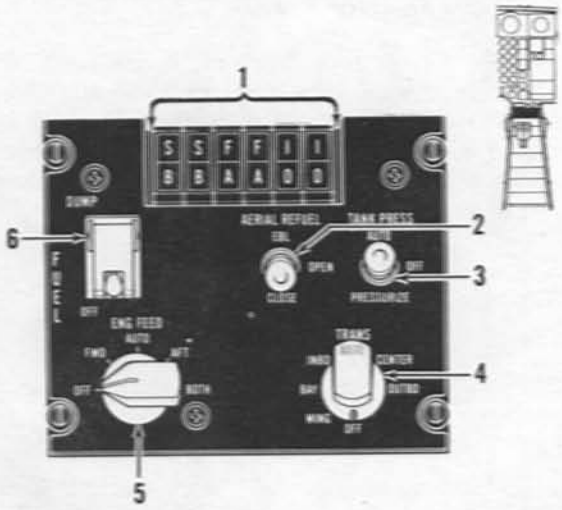
When the knob is placed to FWD both engines are fed from the forward tank. When the knob is placed to the AUTO position the fuselage fuel quantity indicator, controlled by the F (forward) and A (aft) pointers on the instrument, automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure an operational aircraft center-of-gravity. The engines are supplied fuel from the forward tank or both tanks depending upon the position of the pointers. When the knob is placed to AFT, both engines are fed from the aft tank. However, when on AFT feed under conditions of high fuel flow, the forward standby pumps will assist in meeting the high demand on the aft tank. The standby pumps will also feed the engines from the forward tank should the aft tank run dry when on AFT feed.

WARNING

Do not use AFT feed selection when negative g operation is anticipated. Under negative g conditions only number 2 standby pump will be feeding the engines and engine flameout could result at MIL power or above.

When the knob is placed to the BOTH position the left engine is fed from the forward tank and the right engine is fed from the aft tank. In this position there is no automatic fuel distribution control and forward and aft tank fuel differential must be controlled by monitoring the fuselage fuel quantity indicator and manually selecting either FWD, AFT, or BOTH feed.

Fuel Control Panel



1. Fuel Pump Low Pressure Indicator Lamps.
2. Air Refueling Switch.
3. Fuel Tank Pressurization Selector Switch.
4. Fuel Transfer Knob.
★ 5. Engine Feed Selector Knob.
6. Fuel Dump Switch.

A4681000-E021A

Figure 1-8.

Note

The knob must be in either the AUTO or BOTH position to enable the functions of the air refueling switch and the BOTH position should be selected to insure taking on a full load of fuel during air refueling.

Fuel Transfer Knob.

The fuel transfer knob (4, figure 1-8), located on the fuel control panel, has seven positions marked WING, BAY, 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. When the knob is positioned to BAY, two transfer pumps in the left weapons bay tank are energized to transfer fuel to the fuselage tank. 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, weapons bay 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.

Note

When all fuel has been transferred, as indicated by the fuel quantity indicator and fuel pump low pressure indicator lamps, the knob should be turned to OFF. This will prevent excessive fuel transfer pump wear, conserve electric power and turn off the fuel pump indicator lamps. The fuel transfer knob must be in the OFF position to allow refueling the wing and weapons bay tanks. Common fuel manifold lines are utilized for both fuel transfer and refueling, therefore if the transfer system is maintaining pressure in the manifold the refueling valves in these tanks cannot open to allow refueling.

Fuel Dump Switch.

The fuel dump switch (6, figure 1-8), located on the fuel control panel, is a two-position switch marked DUMP and OFF. A guard holds the switch in the OFF position to prevent inadvertent actuation. The functions of this switch are explained under "Fuel Dump System," this section.

Fuel Tank Pressurization Selector Switch.

The fuel tank pressurization selector switch (3, figure 1-8), located on the fuel control panel, is a three-position, lever-lock 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 air 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. Pressurization of the fuel tanks will not be provided when the air source selector knob is in the OFF or EMER position.

Air Refueling Switch.

The air refueling switch (2, figure 1-8), located on the fuel control panel, is a three position lever lock toggle switch marked EBL, OPEN and CLOSE. Refer to "Air Refueling," this section, for operation of the air refueling switch.

Nose Wheel Steering/Air Refuel Buttons.

The nose wheel steering/air refuel buttons (4, figure 1-24), located one on each control stick grip, are labeled NWS and A/R DISC. The air refueling functions of the buttons are activated when the aircraft is airborne to provide 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.

Position Lights/Stores Refuel Battery Power Switch.

The position lights/stores refuel battery power switch (5, figure 1-29), located on the ground check panel, has three positions marked POS LIGHTS, NORM and STORES REFUEL. Placing the switch to the STORES REFUEL position will supply battery power directly from the battery bus to the external fuel tank fuel valves for single point ground refueling regardless of the position of the battery switch. A float switch in the external tank will break the circuit and shut off the flow of fuel when the tanks are full. Placing the switch to NORM de-energizes 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.

Fuel Quantity Indicator Test Button.

The fuel quantity indicator test button (42, figure 1-5), located on the left main instrument panel, is provided to self test the fuel quantity indicators. On aircraft ◀ and those modified by T.O. 1F-111-673 the button has the additional function of self testing the alternate fuel distribution monitoring system. When the button is depressed, the fuel quantity indicators will simultaneously drive to the following indications:

1. Forward and aft tank pointers 2000 ± 400
2. Select tank pointer 2000 ± 100
3. Total fuel digital counter 2000 ± 1250

The fuel distribution caution lamp will light 10 to 12 seconds after the test button has been held depressed, to indicate that the alternate fuel distribution monitoring system is operative. When the button is released, the fuel quantity indicators will return to their original readings and the fuel distribution caution lamp will go out in less than 15 seconds for all engine feed selector knob selections except AUTO. In the AUTO position, the lamp will remain on until the fuselage fuel quantity indicator pointers show a differential between the forward and aft tank greater than 7600 pounds. During the short period of time that the pointers show an abnormal fuel distribution (while they are returning to their original readings), the automatic fuel distribution control system will open the automatic fuel transfer valve and allow a small amount of fuel to be transferred from the aft to the forward tank.

Note

If fuel tank expansion space has been reduced due to fuel overfill or thermal expansion, some fuel venting may occur with the engine feed selector in AUTO, while the fuselage fuel quantity indicators are returning from the test indications. Fuel venting must cease prior to taxiing.

Fuel Quantity Indicator Selector Knob.

The fuel quantity indicator selector knob (44, figure 1-5), located on the left main instrument panel, has nine positions marked L WING, R WING, BAY, 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.

Note

Placing the fuel quantity indicator selector knob to an external tank position when there is no tank installed will provide a below zero indication.

Total/Select Fuel Quantity Indicator.

The total/select fuel quantity indicator (43, 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 compensator sensor, located in the aft tank, to correct for variations in fuel densities. 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. The actual error will depend on the amount of fuel remaining in other tanks, however, a maximum error of 1000 pounds could exist.

Fuselage Fuel Quantity Indicator.

The fuselage fuel quantity indicator (41, figure 1-5), located on the left main instrument panel, provides indications of the amount of fuel in the forward and aft fuselage tanks. In addition, when operating in automatic engine feed the indicator, through a series of internal switches controlled by the F (forward) and A (aft) pointers on the instrument, automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure aircraft center-of-gravity. Refer to "Automatic Fuel Distribution (Primary)," this section, for a description of this function of the fuel quantity indicator.

WARNING

The use of auto engine feed when the fuselage fuel quantity indicator is malfunctioning or inoperative could result in exceeding the center-of-gravity limits and loss of control of the aircraft.

The indicator is graduated from 0 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. One dot indicates the point at which aft to forward transfer will occur, and the other indicates the point at which the aft tank pumps are shut off to maintain the 8200 pound differential. Two bar indices outboard of the dots indicate the point at which the fuel distribution caution lamp will light to indicate that 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 Manifold Low Pressure Caution Lamps.

Two amber fuel manifold low pressure caution lamps (figure 1-37), 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 psi.

Fuel Low Caution Lamp.

The amber fuel low caution lamp (figure 1-37) located on the main caution panel is controlled by a float switch in the reservoir tank. When the lamp is lighted, the letters FUEL LOW are visible indicating that the fuel level in the reservoir tank is less than 2300 (±235) pounds. Due to the gaging system tolerances, the forward fuel quantity indication will be between 1700 and 3000 pounds.

WARNING

Negative g operation must be avoided whenever the fuel low caution lamp is lighted. The fuel system can supply fuel to the engines during negative g operation for 10 seconds if the reservoir tank is initially full. There may be no negative g capability if the fuel low caution lamp is on, indicating that the reservoir tank is not full.

Note

If boost pump 1 fails to provide fuel circulation through the reservoir from bay F-2, the small amount of air trapped in the top of the wing carry through box may expand, lowering the fuel level and causing the fuel low caution lamp to light. Engine fuel supply, other than for negative g, will not be jeopardized. During climb, with afterburners operating the fuel low caution lamp may occasionally light. This is caused by air from the fuel that collects at the top of the reservoir tank, allowing the fuel low float switch to actuate. This does not indicate a malfunction or constitute a hazardous condition for positive g flight. The lamp should go out after engine flow from the forward tank is reduced to less than 40,000 pounds per hour.

Fuel Pump Low Pressure Indicator Lamps.

Twelve fuel pump low pressure indicator lamps (1, figure 1-8), one for each fuel pump, are located on the fuel control panel. When a fuel pump is energized, whether by automatic or manual tank selection, and the pump is not generating at least 3.5 (±0.5) psi, the lamp corresponding to the pump will light. The lamps are arranged in a double row, and the face of the lamps are marked in pairs to correspond to each pump as follows:

S — Standby pumps 1 and 2

F — Forward fuselage tank pumps 3 and 4

I — Wing tank inboard transfer pumps 7 and 8

B — Weapons bay tank transfer pumps 11 and 12

A — Aft fuselage tank pumps 5 and 6

O — Wing tank outboard transfer pumps 9 and 10.

Fuel Tank Pressurization Caution Lamp.

The fuel tank pressurization caution lamp (figure 1-37), 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.

Note

During descent, with the engines at idle, engine bleed air pressure is reduced resulting in a lower air flow to the fuel tanks. At descent rates greater than 6,000 feet per minute it is possible for the fuel tank pressure to drop below 3.5 psi causing the lamp to light. This is not an indication of a malfunction or hazardous condition.

Fuel Distribution Caution Lamp.

The fuel distribution caution lamp (figure 1-37), located on the main caution panel, is provided to indicate an abnormal fuel distribution between the forward and aft tanks. The lamp has two signal input sources: (1) With the engine feed selector in AUTO, the automatic fuel distribution control system will light the lamp if the differential between the F and A pointers becomes less than 7600 pounds or greater than 10,000 pounds. (2) With the engine feed selector in any position, including OFF, the alternate fuel distribution monitoring system will light the lamp for abnormal aft center-of-gravity conditions only.

Note

On aircraft prior to T.O. 1F-111-673, the fuel distribution caution lamp will function only in AUTO mode.

Nose Wheel Steering/Air Refueling Indicator Lamp.

The nose wheel steering/air refueling indicator lamp (22, figure 1-5), located on the left main instrument panel, is labeled NWS/AR. For air refueling, the lamp indicates 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 light again. When the air refueling switch is in the EBL position, the lamp indications are the same as when in normal operation, except the lamp will go out if the NWS/AR button is depressed. The lamp will come on when the NWS/AR button is released if a disconnect has occurred. 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 on the left side of the fuselage 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.

The fuel system can be operated in either an automatic or manual mode. The automatic mode is normally used since is requires a minimum amount of crew monitoring. Manual mode serves primarily as a backup in the event automatic operation malfunctions.

Normal (Automatic) Operation.

Normal system operation is accomplished with both the engine feed selector and fuel transfer knobs in AUTO. In this configuration the following functions are automatically performed:

- As fuselage fuel is used, fuel is transferred into the fuselage tanks from the external tanks, weapon bay tanks and internal wing tanks, in that sequence.
- If all tanks were fully serviced at takeoff both engines will be fed from the forward fuselage tank until external and internal wing tanks and weapon bay tanks are expended and the fuel level in the forward tank is burned down to approximately 8200 pounds of fuel more than the aft tank. At this point the system will automatically switch to a split feed condition (feeding the right engine from the aft tank and the left engine from the forward tank) to maintain the differential thereby keeping the aircraft center-of-gravity within operational limits.
- If the forward tank is burned down to approximately 7900 differential the automatic transfer valve will open to allow fuel to be transferred from the aft tank to the forward tank. This will re-establish the 8200 pound differential.
- If the aft tank is burned down to 8500 pounds differential, the aft tank pumps are turned off and both engines are fed from the forward tank until the 8200 pound differential is re-established.

Manual Operation.

In the event that either automatic engine feed or automatic fuel transfer become inoperative, manual backup is available. During manual engine feed the forward tank must be maintained at least 8000 pounds more than the aft tank by manual selection of either FWD or AFT feed to establish the proper differential. Once

the differential has been established BOTH should be selected to maintain the differential. During manual transfer the fuel transfer knob is positioned progressively to OUTBD, CENTER, INBD, BAY, and WING to empty the external wing, bay, and internal wing tanks, in that sequence. As each tank is selected for transfer, the corresponding fuel quantity indicator selector knob position should be selected to monitor the fuel level in the tank being emptied. It will be necessary to frequently switch the knob between the left and right external and internal wing tanks to monitor fuel transfer from these tanks.

Note

There should be a delay of approximately one minute after each tank(s) (external, bay or internal) indicates empty to insure all fuel is transferred before selecting the next tank.

During fuel transfer, fuel will be transferred to the forward and aft tank. If the fuel level in the fuselage tanks is lowered before all fuel is transferred, auto engine feed should be used to achieve the proper differential between the forward and aft tanks, and the forward and aft tank fuel quantities should be monitored during the fuel transfer operation. However, when operating in auto engine feed, distribution is corrected by burning fuel from the forward tank; and during low fuel consumption rates, the fuel distribution caution lamp may light indicating excessive fuel in the forward tank.

Gravity Fuel Feed.

The engine driven fuel pumps will gravity (suction) feed the engines in the event of an electrical malfunction which prevents booster pump operation. In this condition fuel will be used from the forward tank only. An anti-suction valve between the forward and aft tanks prevents suction feed from the aft tank to prevent the suction of air into the engine feed line in the event the aft tank is empty.

Fuel Dumping.

With the fuel dump switch in the OFF position, dump valves A and B are closed and C is open. When the switch is positioned to DUMP, the following events occur:

1. Dump valves A and B open and C closes
2. The automatic transfer valve opens
3. The fuel tanks pressurize (with the air source selector knob in any position other than OFF or EMER)
4. Booster pumps 5 and 6 in the aft tank transfer fuel to the forward tank (If in AUTO with more than 8500 pound differential, 6 only)
5. Transfer pumps 11 and 12 in the weapons bay tank, if installed, transfer fuel to the forward tank
6. When the weapons bay tanks are empty, pumps 7, 8, 9 and 10 transfer fuel from the wing tanks to the forward tank.

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 handle, or the air refueling door, provided the air source selector knob is in a position other than OFF or EMER. Sufficient air is available to obtain the normal dump rate of 2300 pounds per minute when either engine RPM exceeds 85 percent. Tank pressurization forces fuel from the forward fuselage tank into the dump manifold and overboard through the vent/dump valve located on the aft centerbody. Fuel will be transferred from aft to forward tank at approximately 1750 pounds per minute if both aft tank pumps are operating or at 1100 pounds per minute if only one pump is operating. If external tank transfer is selected during a dumping operation, the rate of transfer from the external tanks is relatively slow; therefore, if required by operational considerations, these tanks should be jettisoned. All fuel except that in the reservoir tank (approximately 2500 pounds) can be dumped.

WARNING

To avoid the possibility of dumped fuel reentering the aircraft and causing a fire hazard, fuel dumping should be accomplished in straight and level flight at airspeeds no greater than 350 KIAS or mach 0.75, whichever is less.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the aircraft. 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 happen automatically when the landing gear is extended for landing.) During fuel dumping operations it should be noted that the automatic center-of-gravity control will not operate normally. If the engine feed selector knob 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 and the 8200 pound differential may not be maintained causing the fuel distribution caution lamp to light. 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. Although fuel is normally forced overboard by tank pressurization during fuel dumping, some dump capability exists when tanks are not pressurized (air source selector knob in the OFF or EMER position). The fuel that is transferred to the forward tank will flow overboard, through the dump/vent outlet, at approximately the transfer rate, if the forward tank is nearly full. If the forward tank is not initially full, a portion of the fuel being transferred may partially fill the forward tank. After the tanks from which fuel is being transferred are empty, a portion of the fuel in the forward tank will flow overboard by gravity. The fuel flow rate from the forward tank will be approximately 500 pounds per minute when the tank is full, and will gradually decrease to zero. The quantity of the fuel that can be dumped from the forward tank depends on the attitude of the aircraft, the higher the nose of the aircraft, the more fuel dumped. At level flight, the dump flow from the forward tank will cease at a fuel quantity in the forward tank of approximately 13,000 pounds. In order to obtain maximum fuel dump rate, without tank pressurization, the engine feed selector switch should be positioned to BOTH to prevent the automatic fuel distribution system from turning off number 5 booster pump.

Air Refueling.

In order to open the receptacle the engine fuel feed selector must be selected to AUTO or BOTH, and the air refueling switch must be selected to OPEN or EBL. When the receptacle is open, the NWS A/R lamp will light to indicate the receptacle is open and the system is ready to accept the refueling probe.

Note

During ground operation when the air refueling door is open, the nose wheel steering/air refueling indicator lamp will light to indicate door position and nose wheel steering cannot be monitored.

When the tanker/refueling probe is inserted into the receptacle, it is automatically latched in place and the NWS/AR lamp will go out to indicate when the latches have closed. Refueling is accomplished with the refuel switch selected to OPEN. In this position a disconnect signal can be provided from the tanker or from either crew member by use of the NWS/AR, disconnect button. In addition, when all tanks are full, fuel flow is interrupted by automatic closing of the refuel valves. A pressure switch will sense a rise in pressure in the refuel manifold and automatically provide a disconnect signal. Three seconds after the disconnect has occurred, the refuel system will automatically "reset" itself and light the NWS/AR lamp to indicate the system is again ready to receive a probe or that the receptacle should be closed. In addition, if the air refueling amplifier malfunctions, the EBL position on the air refuel switch will permit refueling. The procedure for EBL refueling is the same as the automatic procedure described above except a disconnect signal cannot be provided from any source other than the crew using the NWS/AR button. When the button is depressed, the NWS/AR lamp will remain out until it is released. The NWS/AR lamp will light when the probe is out of the receptacle. If a malfunction of the hydraulic control solenoid has occurred that prevents opening of the receptacle, opening can then be accomplished by selecting EBL. This mode uses a separate solenoid to open the receptacle. Certain failures may require the air refuel circuit breaker to be reset after EBL is selected. In the event utility hydraulic power is not available, a back-up pneumatic system is provided. This system is energized by selecting EBL. Once in EBL, the OPEN position may be selected. This will allow the system to operate as it does in the OPEN position. Pneumatic power to operate the system will remain on until 5 seconds after the air refueling switch is placed to CLOSE. Sufficient pneumatic power is available to operate the receptacle through two cycles (open and close) with four hook-ups during each cycle. In the event of a failure that prevents a normal disconnect, a pressure relief valve is provided in the receptacle hydraulic latch actuator that will allow the probe to be pulled out by brute force if the boom tension exceeds 5000 pounds. Normal operating boom loads do not exceed 2300 pounds. Design loads for the receptacle and the tanker boom exceed 16,000 pounds. Lights are provided to illuminate the receptacle. The lights are turned on by a switch when the slipway door is open. The intensity of the lights is controlled by the air refueling receptacle control knob. The knob should normally be at the mid-point of its control range when not in use. This assures that the lights are on at the beginning of night refueling but does not waste the service life of the bulbs during day refueling. Refer to T.O. 1-1C-1-1 for general air refueling procedures and T.O. 1-1C-1-18 for specific air refueling procedures for the F-111.

Single Point Refueling.

For single point refueling procedures, refer to "Strange Field," Section II.

AC Electrical Power Supply System (Typ)

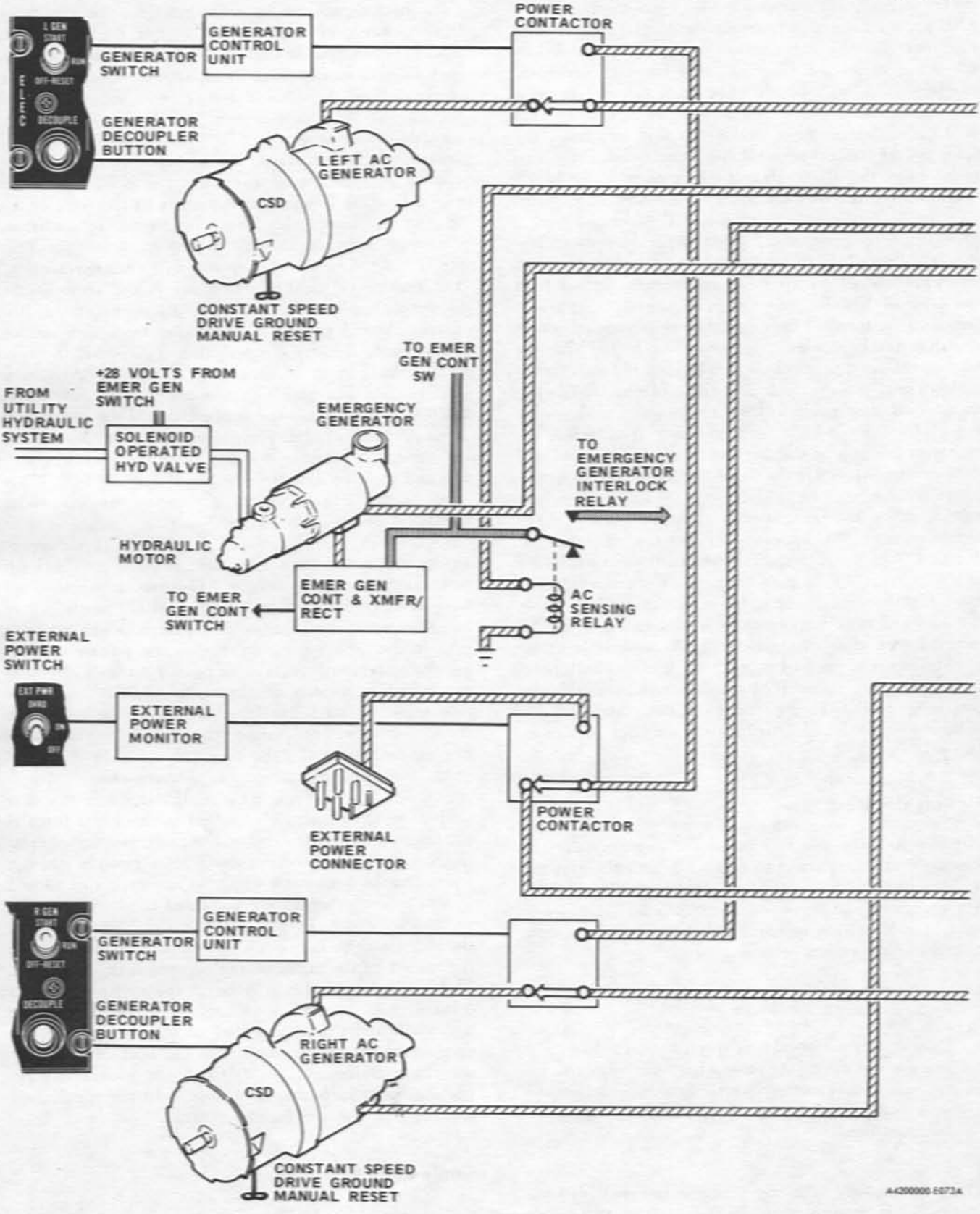


Figure 1-9. (Sheet 1)

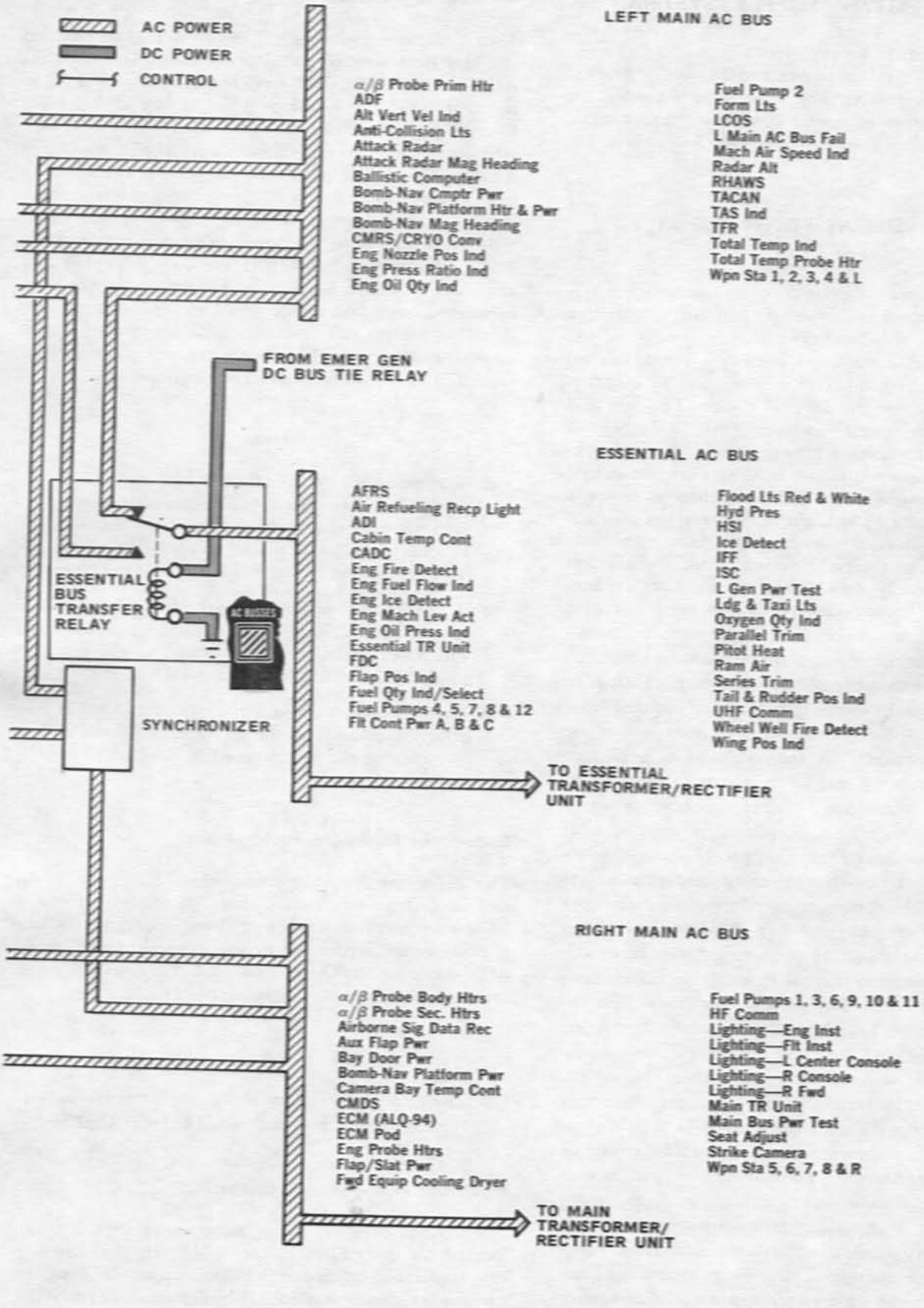


Figure 1-9. (Sheet 2)

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 figures 1-9 and 1-11.)

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. A cooling system is provided to cool constant-speed drive oil with ram air and by circulating the oil through the fuel-oil cooler. The left and right generators operate independently and there is no phase synchronization between them. 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 to 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 generator but disconnects it from the bus. If the malfunction is corrected, the generator may be reconnected to the bus by properly positioning the generator switch. If a malfunction causing an excessive amount of heat occurs in the constant-speed drive 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 de-energized, 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.

Two generator switches (1, figure 1-10), located on the electrical control panel are marked OFF-RESET, RUN and START. The switches are lever locked in the OFF-RESET position and are spring-loaded from START to RUN. Placing either switch to OFF-RESET will reset the generator-control unit of the respective generator, but will not excite the generator. Holding the switch to START will excite the generator automatically and connect it to its respective bus. This will be indicated by the power flow indicator and the generator caution lamp will go out. Allowing the switch to return to RUN, after the generator has been connected to its bus, establishes a switching configuration that assures safe operation in the event of a subsequent malfunction. The switch must be positioned to OFF-RESET to allow generator de-coupling.

Note

- If a generator is deexcited while connected to the bus, it will not automatically reset, even though the fault condition is cleared. The switch must be placed to OFF-RESET to reset the system.
- If the engine is started with the generator switch in the RUN position the generator will be excited but not connected to the bus until the switch is placed to START.
- There should be a definite pause (approximately 1 second) in the START position before placing the switch to RUN to allow time for power transfer.

Generator Decouple Pushbuttons.

The generator decouple pushbuttons (5, 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.

Note

The generators cannot be decoupled until the respective switch is positioned to OFF-RESET.

Electrical Power Flow Indicator.

The electrical power flow indicator (4, 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 NORM, indicating that the buses are iso-

lated 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 the ac essential bus, the indicator will display EMER. When ground power is connected to the aircraft and supplying power to the ac buses, the indicator will display TIE until the right engine is started and its generator comes on the line, then it will indicate NORM. The indicator will display a crosshatched surface if there is no ac power applied to the aircraft while the emergency generator switch is in TEST.

Emergency Generator Switch.

The emergency generator switch (7, 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.

WARNING

If emergency generator switch is placed to ON or TEST, check power flow indicator. If flow indicator reads EMER, an ac sensing relay failure is indicated. If in flight, place battery switch to OFF prior to placing emergency generator switch to AUTO, and leave off for remainder of flight. Failure to do so will cause power interruptions and possible severe damper transients.

When the switch is in the TEST position, the emergency generator operates, but is not connected 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 transformer/rectifiers. Opening of the dc bus contactor is indicated by a crosshatch display in the electrical power flow indicator. When the emergency generator switch is placed to TEST, the essential transformer/rectifier unit has failed if the landing gear position lamps are out and the main transformer/rectifier has failed if the total temperature indicator off flag comes into view.

Electrical Control Panel

1. Generator Switches.
2. Battery Switch.
3. External Power Switch.
4. Electrical Power Flow Indicator.
5. Generator Decouple Pushbuttons.
6. Emergency Generator Indicator/Cutoff Pushbutton.
7. Emergency Generator Switch.

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Figure 1-10.

Emergency Generator Indicator/Cutoff Push Button.

The emergency generator indicator/cutoff push button (6, figure 1-10), located on the electrical control panel, provides a means of de-exciting the emergency generator. The push button is marked OPR (operate) and PULL OFF. When the button is depressed the emergency generator will come on the line and supply power to the aircraft systems whenever both engine driven generators fail. Should this occur a green indicator lamp in the button will light. When the emergency generator is supplying power pulling the button out will de-excite the emergency generator and shut off its power output. The button is normally safety wired in the OPR position.

External Power Switch.

The external power switch (3, 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 aircraft ac buses. In the ON position with neither engine operating, external power supplies total aircraft power. With the left engine operating, the left main ac generator will supply total aircraft electrical load, and

DC Electrical Power Supply System (Typical)

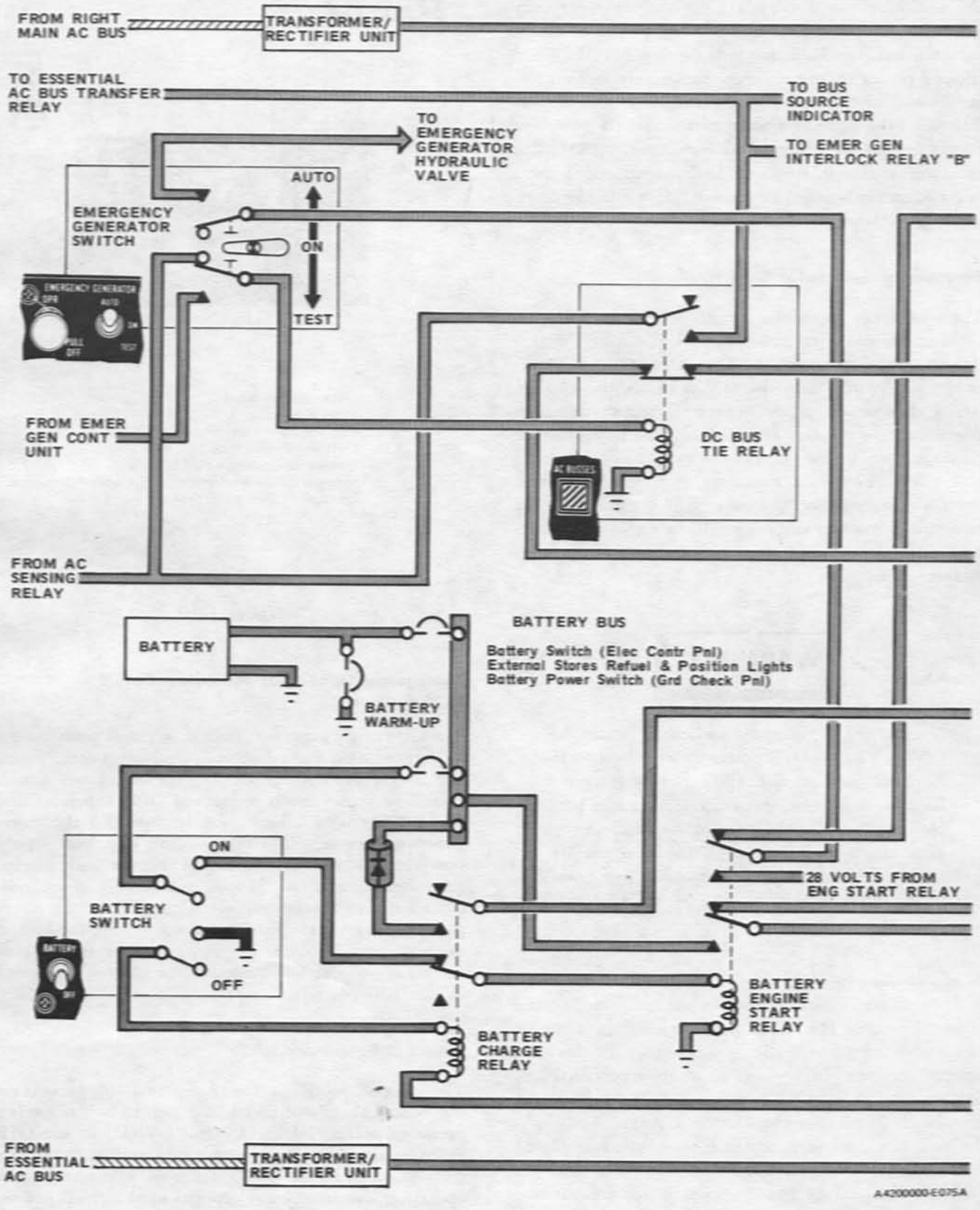


Figure 1-11. (Sheet 1)

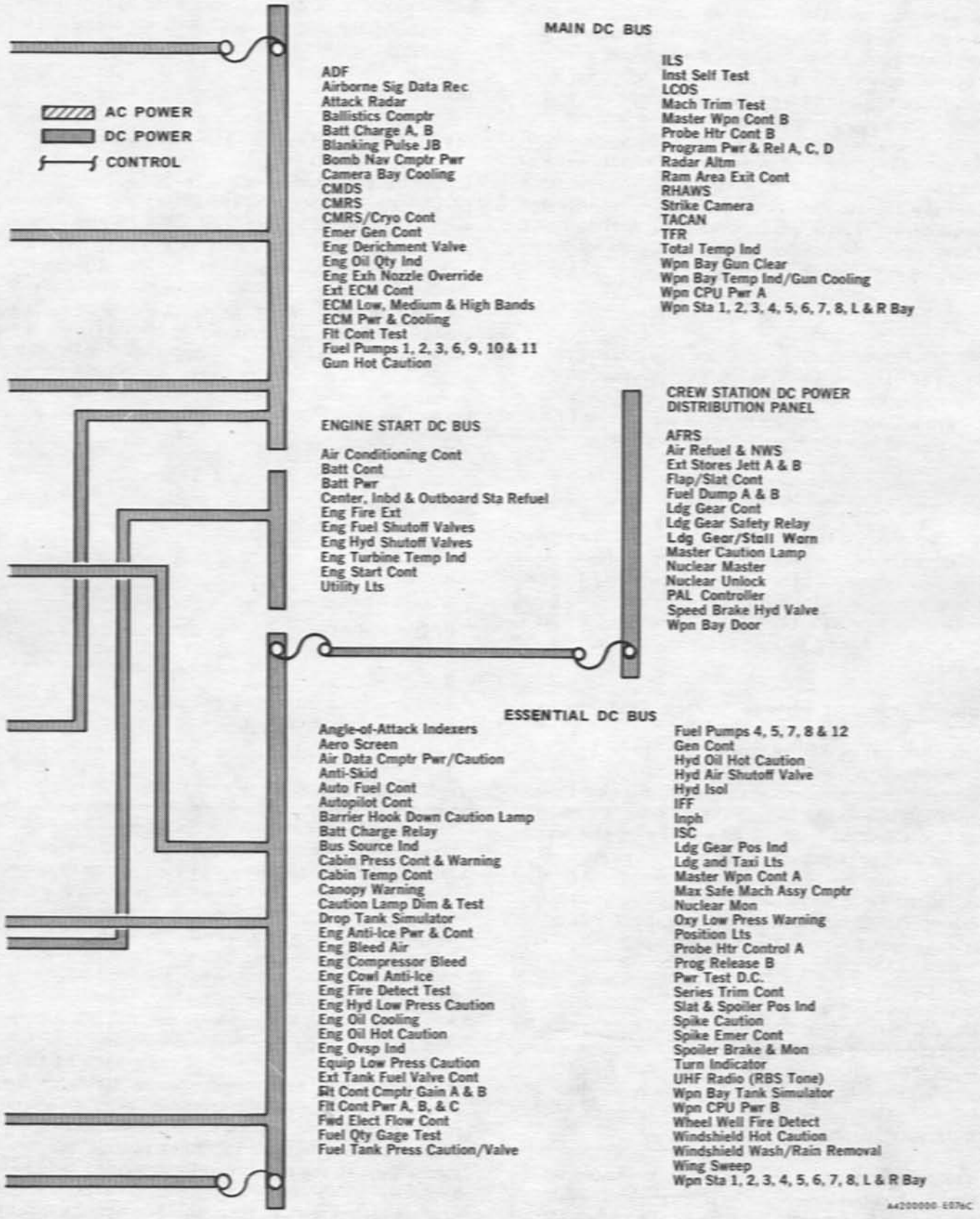


Figure 1-11. (Sheet 2)

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 aircraft buses. The override position does not override external power with improper phase sequence.

Note

If electromagnetic radiation is experienced while the switch is in the ON position, the power monitor may be affected and reject external power. Power to the aircraft may be regained by placing the switch to OVRD.

Generator Caution Lamps.

Two generator caution lamps (figure 1-37), are located on the main caution lamp panel. Either lamp will light when its respective generator is disconnected from its bus and remain lighted until the generator switch has been placed to START. When lighted, the letters L GEN are visible in the left lamp and R GEN in the right lamp.

DIRECT CURRENT POWER SUPPLY SYSTEM.

DC electrical power is provided by two 28 volt dc transformer-rectifier units (converters) and a 24 volt battery. There are three dc buses: a main dc bus, an essential dc bus, and an engine start bus. The essential dc bus is divided into two separate buses, one located in the forward equipment bay and one located in the crew module on the aft console (figure 1-12). The essential buses are electrically connected. During normal operation, the main dc bus section receives power from the main transformer-rectifier unit which is connected to the right main ac bus. The essential dc bus and the engine start bus receive power from the essential transformer-rectifier unit which is connected to the essential ac bus. A bus-tie contactor connects the essential dc bus to the main 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 (2, 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, and the engine start bus is connected to the essential dc bus. When the battery switch is positioned to OFF, the battery charger circuit is disconnected from the main dc bus.

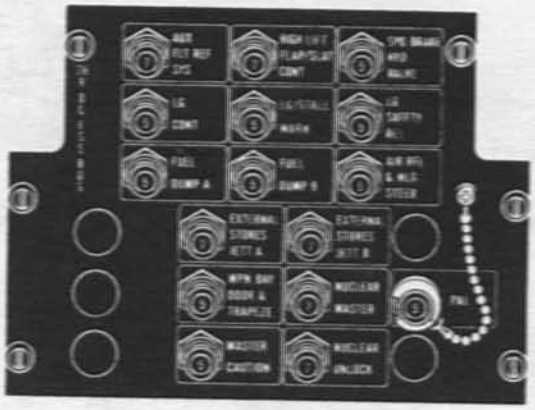
HYDRAULIC POWER SUPPLY SYSTEM.

Hydraulic power is supplied by two independent, parallel hydraulic systems designated as the primary and utility systems (figure 1-13). 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 reduced power for wing sweep and flight control operation. The primary hydraulic system supplies hydraulic power solely for operation of the wing sweep and flight control systems. In addition to supplying wing sweep and flight control hydraulic power, the utility system also supplies power for operation of:

- Nose wheel steering
- Landing gear
- Wheel brakes
- Speed brake
- Flaps/slats
- Rotating glove
- Weapons bay doors
- Rudder authority
- Weapons bay gun
- Tail bumper
- Emergency electrical generator
- Air inlet (spike) control
- Air refueling system

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 main instrument panel. Low pressure caution lights for each of the four pumps are displayed on the caution lamp panel. An isolation unit incorporated into the system reserves utility pressure for flight control and wing sweep only, in the event of primary system failure. It also performs a second function of isolating hydraulic pressure after takeoff from those systems normally only associated with takeoff and landing.

Circuit Breaker Panel (Typical)



CIRCUIT BREAKER	FUNCTION
AFRS	Provides power to the primary att/hdg caution lamp, the aux/att caution lamp and used as the AFRS good signal.
HIGH LIFT FLAP/SLAT CONT	Provides power to the flap/slat asymmetry system and power to the flap/slat emergency motor.
SPEED BRAKE HYD VALVE	Provides power to the speed brake hydraulic valves.
LG CONT	Provides power to the extend and retract solenoids on the landing gear hydraulic valve.
LG/STALL WARN	Provides power to the landing gear handle warning lamp, gear up and lock indicator lamps, stall warning lamp, and to the warning tone generator.
LG SAFETY RELAY	Provides power to the ground safety switches.
FUEL DUMP A	Provides power to fuel dump valve A and C
FUEL DUMP B	Provides power to fuel dump valve B, dump relays A and B, and auto transfer solenoid valve

CIRCUIT BREAKER	FUNCTION
AIR RFL & NLG STEER	Provides power to the air refueling receptacle or nose landing gear steering.
EXT STORES JET A	Provides power for external stores and fixed pylons jettison.
EXT STORES JET B	Same as Jet A
WPN BAY DOOR & TRAPEZE	Provides power for operation of the weapon bay door and trapeze.
NUCLEAR MASTER	Provides source of power and control for the aircraft monitor and control system (AMAC).
MASTER CAUTION	Provides power to the master caution lamp.
NUCLEAR UNLOCK	Provides power to monitor the status of the inflight lock on the MAU rack.
PAL	Provides power to PAL for enabling Nuclear Bombs for pre-arming.

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Figure 1-12.

Hydraulic Power Supply System (Utility)

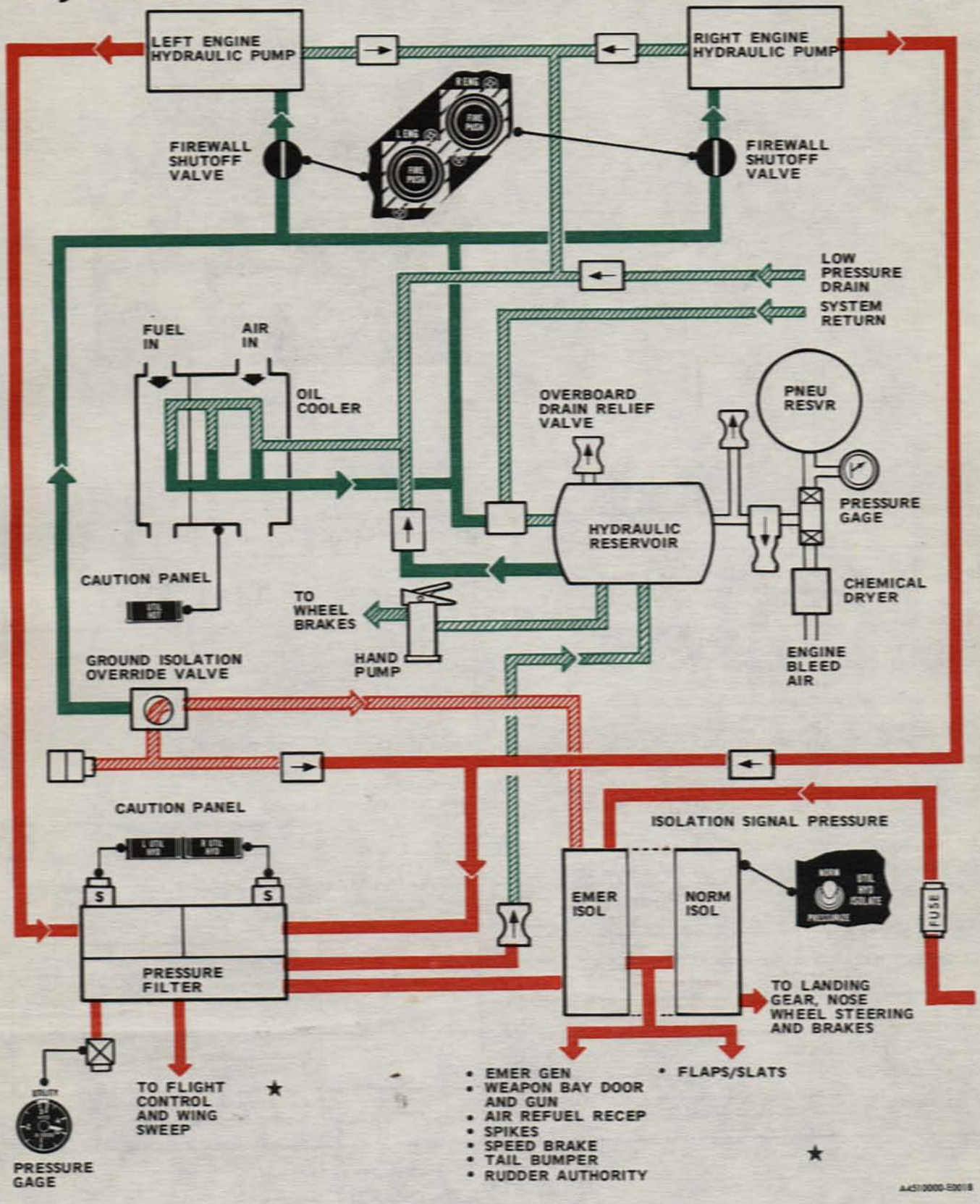


Figure 1-13. (Sheet 1)

Hydraulic Power Supply System (Primary)

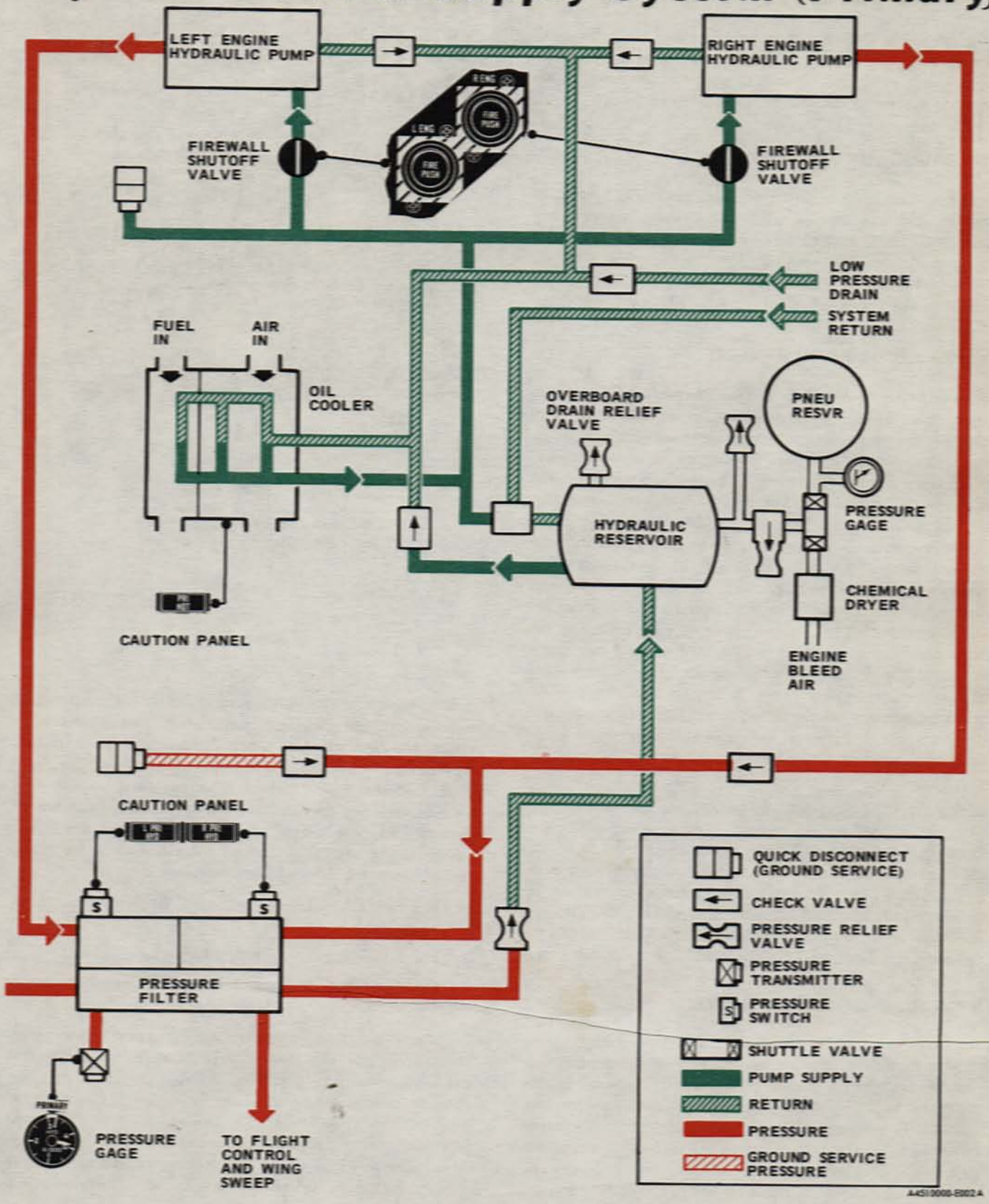


Figure 1-13. (Sheet 2)

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.

HYDRAULIC ACCUMULATORS.

Eight accumulators, three in the primary hydraulic system and five in the utility hydraulic system, are provided. Each system has two accumulators for the horizontal stabilizer and one for the autopilot damper servos. The utility system has two accumulators for the wheel brake system. See figure 1-84 for servicing data.

HYDRAULIC FLUID RESERVOIRS.

Both primary and utility hydraulic reservoirs are floating piston, air-oil separated type using nitrogen (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 bypass type filter is located upstream of the reservoir. The reservoir also acts as a surge damper for return line impulse pressures. See figure 1-84 for servicing data.

HYDRAULIC COOLING SYSTEM.

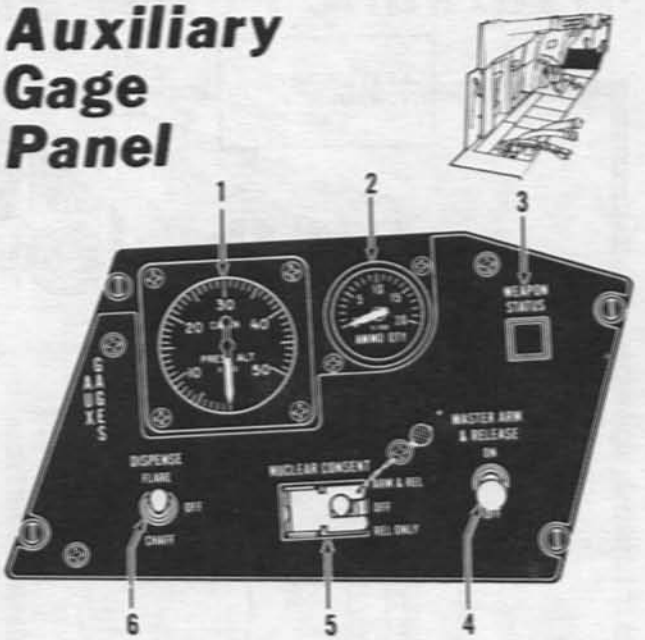
Cooling is provided by an air-to-hydraulic heat exchanger and fuel-to-hydraulic heat exchanger in each hydraulic system. The controls function according to total temperature and are arranged so that the cooling medium is air only at low speeds, fuel and air at intermediate speeds, and fuel only at high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

HYDRAULIC ISOLATION VALVE.

An isolation valve is incorporated in the utility system to automatically provide emergency and normal isolation of certain functions of the utility system. In the event of loss of pressure in the primary system, the valve will automatically go into emergency isolation at approximately 500 psi, and cut off all systems except flight controls and wing sweep. The primary system pressure must increase to approximately 1200 psi to bring the systems out of isolation. The normal isolation function of the valve is electrically controlled and will isolate the landing gear, wheel brake and nose wheel steering systems when the aircraft is in flight. A separate electrically controlled shutoff valve is included in the flap/slat hydraulic pressure line to provide flap/slat system isolation. The landing gear, wheel brakes and nose wheel steering isolation takes place immediately after the last of all the following three controlling conditions are satisfied:

1. The utility hydraulic isolation override switch is in NORM.
2. The landing gear is up and locked.
3. The flap/slat handle is UP and the flaps and slats are retracted.

Flap/slat isolation is controlled by these same three conditions but the hydraulic shutoff valve is electrically sequenced to provide isolation 30 seconds after the last controlling condition is satisfied. On aircraft modified by T.O. 1F-111-599 the isolation of the landing gear, wheel brakes and nose wheel steering systems is dependent upon controlling conditions 1 and 2 only and flap/slat position will not affect isolation of these systems.



1. Cabin Altitude Indicator.
2. Rounds Counter.
3. Weapon Status Indicator.
4. Master Arm and Release Switch.
5. Nuclear Consent Switch.
6. Flares/Chaff Dispense Switch.

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Figure 1-14.

UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH.

The utility hydraulic system isolation switch (8, figure 1-15), with positions marked NORM and PRESSURIZE, is located on the landing gear control panel. The NORM position functions in conjunction with the landing gear, flaps and slats allowing the following systems to be isolated from the utility system:

- Landing gear
- Nosewheel steering
- Brakes
- Flaps/slats.

During normal operation positioning the switch to PRESSURIZE supplies utility hydraulic pressure to these systems. On aircraft modified by T.O. 1F-111-599, the utility hydraulic system isolation switch will not affect flap/slat isolation.

HYDRAULIC FUSE.

On aircraft modified by T.O. 1F-111-583, a hydraulic fuse is installed on the primary system (sensing) pressure line to system isolation valve. The fuse prevents loss of primary system fluid in event of line rupture by shutting off flow at the downstream port of the fuse.

HYDRAULIC PRESSURE INDICATORS.

Two 0-4000 psi pressure indicators (37, figure 1-5), one each for the utility and primary systems, are located on the left main instrument panel.

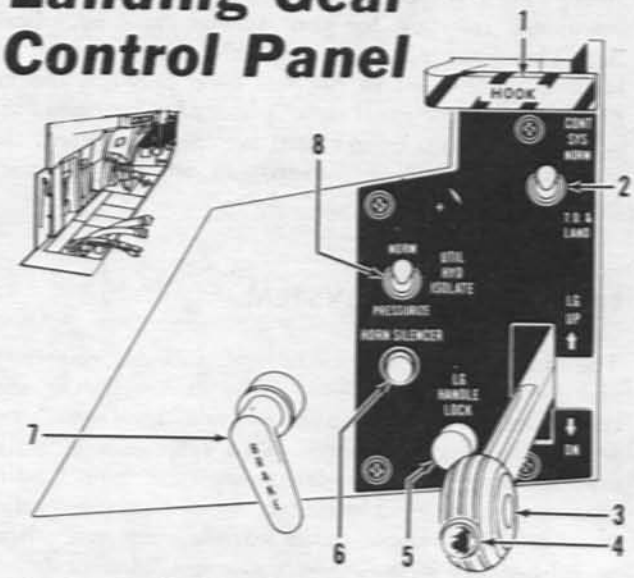
LOW PRESSURE CAUTION LAMPS.

Four low pressure caution lamps, energized by pressure switches in each pump pressure line, are located on the main caution lamp panel (figure 1-37). These lamps light before system pressure descends below 400 psi. When lighted, the letters L PRI HYD, L UTIL HYD, R PRI HYD, and R UTIL HYD will be visible in the respective lamps.

HYDRAULIC FLUID OVERHEAT CAUTION LAMPS.

Two hydraulic fluid overheat caution lamps, one for each system, are located on the main caution lamp panel (figure 1-37). A lamp lights when the hydraulic fluid temperature of the associated system exceeds 240±10 degrees F (115±6 degrees C). When lighted, the following letters will be visible in the respective lamps: PRI HOT; and UTIL HOT.

Landing Gear Control Panel



1. Arresting Hook Handle.
2. Flight Control System Switch.
3. Landing Gear Handle.
4. Landing Gear Warning Lamp.
5. Landing Gear Handle Lock Release Button.
6. Landing Gear Warning Horn Silencer Button.
7. Auxiliary Brake Handle.
8. Utility Hydraulic System Isolation Switch.

AI300000-E001

Figure 1-15.

PNEUMATIC POWER SUPPLY SYSTEMS.

There are eight independent pneumatic power supply systems as follows:

1. Hydraulic reservoir pressurization system
2. Landing gear emergency extend system
3. Inflight refueling emergency system
4. Spike emergency extend system
5. Overwing fairing system.
6. Wheel brakes parking/emergency system.
7. Arresting hook extend system.
8. Tail bumper system.

Two pneumatic reservoirs, one for each hydraulic system reservoir, provide pneumatic pressure for hydraulic system operation. Pressure for emergency extension of the landing gear is provided by a pneumatic reservoir located in the main landing gear wheel well. Pneumatic pressure for emergency operation of the air refueling doors, receptacle, and boom latch 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. The overwing fairing actuators are provided pneumatic pressure to hold the fairing firmly

against the airplane structure regardless of the wing sweep. Each wheel brake circuit is provided with a pneumatic reservoir for parking and emergency braking. The arresting hook and tail bumper pneumatic systems are located in the tail cone between the engines. For a functional description of each pneumatic system, refer to the associated system description, this section. For servicing information on the pneumatic systems, see figure 1-84.

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 over heated brakes. The nose landing gear has dual-mounted wheels. The landing gear system is normally powered by the utility hydraulic system. A pneumatic system is provided as an alternate (emergency) means of extending the gear in the event the normal system fails.

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. Ground safety (squat) switches provide an electrical signal that is used for a number of purposes such as to prevent the landing gear handle from being positioned to UP while the aircraft is on the ground, for antiskid system touchdown feature operation, steering system operation and other functions not related to the landing gear system.

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.

Ground Safety (Squat) Switches.

The main landing gear safety (squat) switches are located on the main landing gear lateral beams as shown on figure 1-16 and provide an electrical signal that will affect the operation of certain aircraft systems. While the weight of the aircraft is on the wheels the squat switches prevent activation of the following in-flight systems.

- Weapons firing.
- Cowl anti-icing.
- Adverse yaw compensation.
- Artificial stall warning.
- Operation of landing gear handle from the down position.
- Secondary alpha/beta probe heater.
- Touchdown skid control.

The switches will permit the activation of the following systems when the aircraft is on the ground and de-activate them when the aircraft becomes airborne.

- Vortex Destroyers.
- Nose wheel steering.
- Ground roll spoilers.
- Ground to pilot interphone.
- Ejector air hydraulic cooling.
- Ejector air engine oil cooling.
- Flight control ground test power.
- Engine nozzle ground idle open position.
- Engine bleed opening with throttle below MIL.

Landing Gear Handle.

The landing gear handle (3, figure 1-15), located on the landing gear control panel, has two positions marked UP and DN. A red landing gear position indicator lamp is located in the end of the landing gear handle. 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 landing gear control valve, sending hydraulic pressure to the nose gear downlock actuator, nose gear retract actuator, nose gear uplock door actuator, speed brake door actuator, and brake control valve. 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. As the nose wheel doors close, snubbers mounted on the doors engage the tires to stop nose wheel rotation. The main gear forward door (speed brake) actuator extends the door. Hydraulic pressure at approximately

750 psi is metered to one brake circuit to stop main gear wheel rotation. 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 uplock actuator, and main gear retract 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.

Note

On aircraft prior to modification by T.O. 1F-111-599, the nose landing gear may not be fully retracted when landing gear isolation occurs. In this event the warning lamp on the landing gear handle will remain lighted indicating that the landing gear is not fully up and locked. Positioning the utility hydraulic system isolation switch to PRESSURIZE will resupply hydraulic pressure to the landing gear system permitting the retraction to continue. When the warning lamp goes out, indicating the retraction cycle is complete, positioning the utility hydraulic system isolation switch to NORM will isolate the landing gear system.

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 both UP and DN positions by a spring actuated lock. An electrical solenoid operated by the landing gear safety (squat) switches unlocks the handle in the DN position when aircraft weight is not on the landing gear. Aircraft weight on gear positions the ground safety switches such that the solenoid is de-energized, thereby permitting the spring actuated lock to secure the handle in the DN position and prevent inadvertent gear retraction on the ground.

Landing Gear Handle Lock Release Button.

The landing gear handle lock release button, located on the landing gear control panel (5, figure 1-15), unlocks the spring actuated handle lock. The button must be depressed to release the landing gear handle from the UP position for landing gear extension. Normally, it is not necessary to depress the button when retracting the gear because the spring actuated lock is unlocked by the ground safety switch operated solenoid. Should the solenoid, safety switch or associated electrical circuit malfunction, depressing the button will release the handle to allow gear retraction.

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 (squat) switches. If a malfunction of the landing gear ground safety (squat) switches is suspected, refer to Section III.

Landing Gear Emergency Release Handle.

The landing gear emergency release handle (10, figure 1-32), 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 emergency method it cannot be retracted. 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

If the handle is pushed in, the weight of the door and the lack of air load as the aircraft slows after landing will cause the door to extend and drag the ground.

Ground Safety Locks and Safety Pins

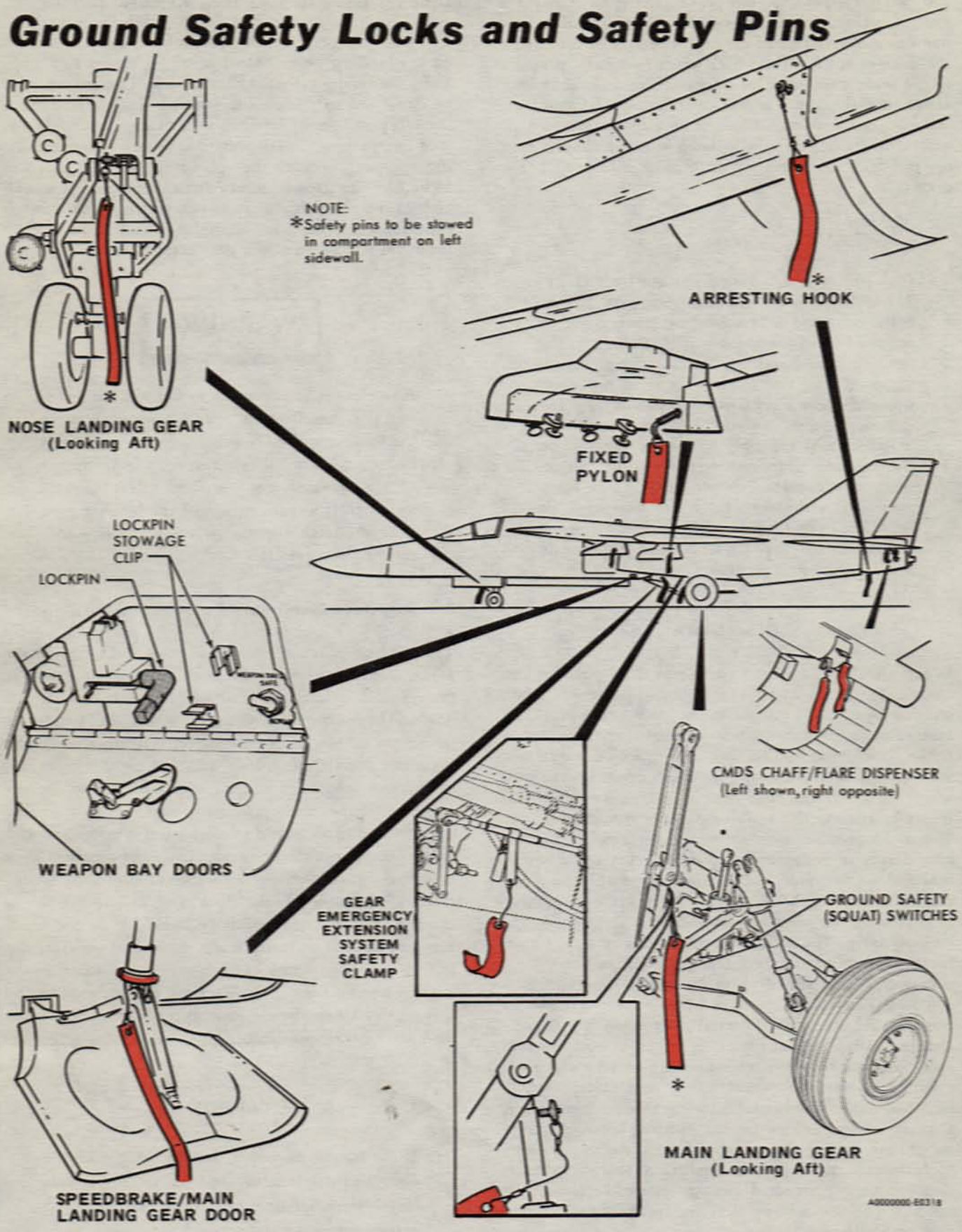


Figure 1-16. (Sheet 1)

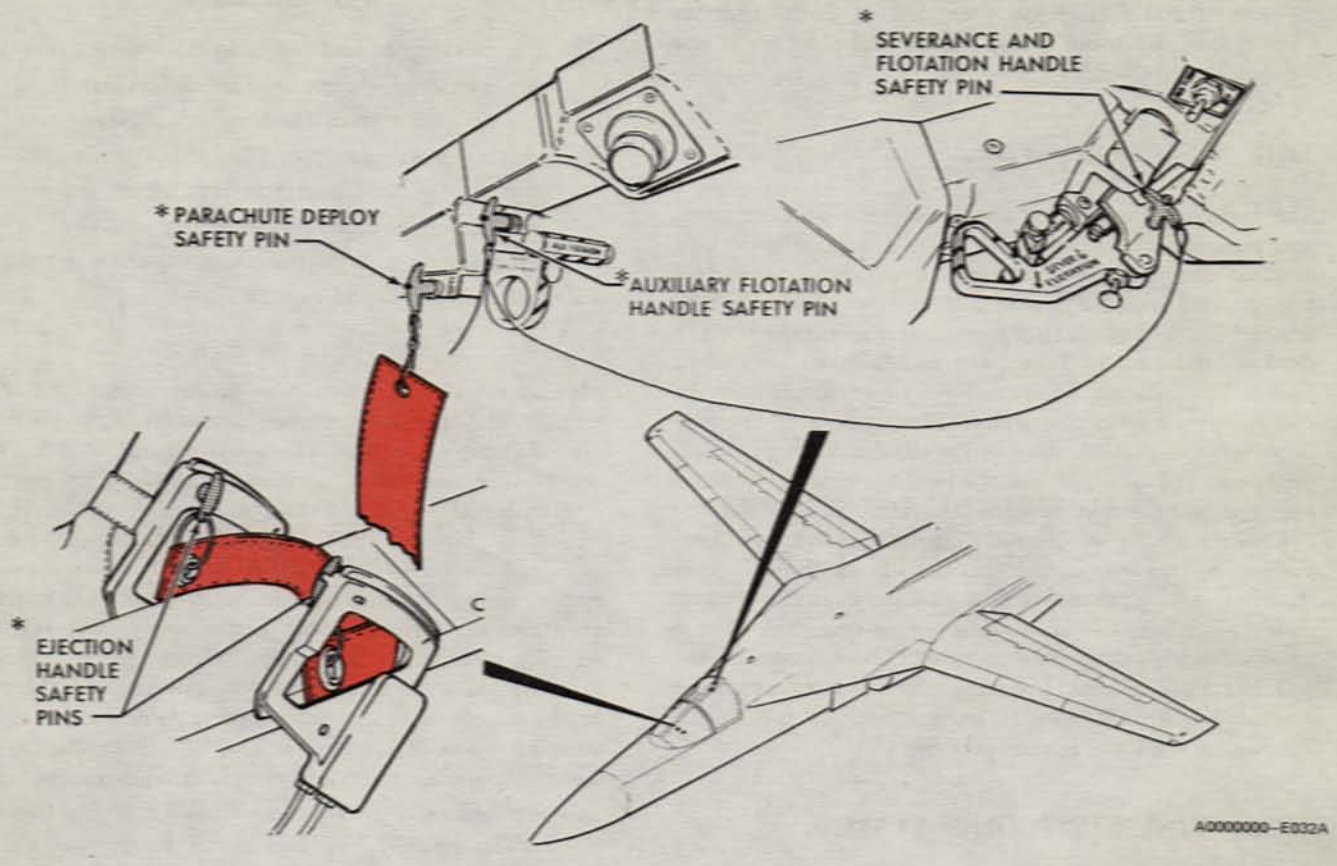


Figure 1-16. (Sheet 2)

Landing Gear Warning Horn.

The landing gear warning horn provides an intermittent audible signal in the crew members' headsets when an unsafe landing gear condition exists. If the nose or main landing gear is not down and locked and/or speed brake is not in trail, the horn sounds when all of the following conditions exist:

1. Indicated airspeed is below 160 (±12) knots.
2. The aircraft is below an altitude of 10,000 (±350) feet.
3. One or both throttles are set below minimum cruise setting. (Approximately 90 percent engine RPM)

On aircraft modified by T.O. 1F-111-891, the warning horn is also used as a stall warning indication. Refer to "Artificial Stall Warning System," this section.

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 (6, figure 1-15).

Note

On aircraft modified by T.O. 1F-111-891, the stall warning lamp will flash and the rudder pedal shaker will operate as long as the horn silencer button is held depressed.

Landing Gear Position Indicator Lamps.

A planform silhouette of the aircraft having two green indicator lamps is located on the left main instrument panel (15, figure 1-5). The lamps are positioned to represent the nose and main landing gear. When the landing gear is down and locked, the green indicator lamps are lighted. A safe up-and-locked landing gear condition is indicated when both the green indicator lamps and the red warning lamp in the landing gear handle are off. The red warning lamp, when lighted, indicates one of the following:

- Disagreement between the speed brake position and that commanded by the landing gear handle.
- Landing gear is in-transit or has failed to lock in selected position.

- When accompanied by the warning horn, denoting that airspeed is below 160 ±12 knots, the aircraft is below 10,000 feet ±350 feet, one or both throttles are below minimum cruise setting and the gear is not down and locked.

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.

NOSE WHEEL STEERING SYSTEM.

Nose wheel steering provides aircraft directional control during taxiing, takeoff and landing. The system is electrically engaged, hydraulically powered and controlled by the rudder pedals. The nose wheels are positioned by a linear hydraulic actuator controlled by a mechanical rotary servo valve. Rudder pedal movement at either crew station is transmitted to the steering valve by mechanical linkage which includes a cam device on the valve input shaft. The cam device provides a gradually increasing ratio between steering angle and rudder pedal displacement. A relatively larger pedal displacement is required to obtain an increment of steering angle near the neutral rudder pedal position than is required near the full rudder pedal position. Utility hydraulic system pressure supplied to the steering servo valve is controlled by a solenoid operated shutoff valve and a pressure regulator. When steering is engaged, the energized solenoid valve applies full system pressure to achieve a high level steering torque. When steering is disengaged, the pressure regulator supplies approximately 10 percent system pressure for a low level steering torque used to center the nose wheels during retraction. Steering input linkage motion occurring during nose gear retraction automatically centers the nose wheels with up to 50 percent rudder pedal displacement.

CAUTION

If a misaligned/malfunctioning steering system is evident, do not take off unless required, and do not retract landing gear. Nose gear steering adjustment can be checked by disengaging nose wheel steering while taxiing on a level surface. If a steering transient is observed on reengagement, a misalignment/malfunction is indicated.

Maximum rudder pedal deflection steers the nose wheels 40 degrees either side of center with resultant aircraft turning radius as shown on figure 2-2. Nose wheel shimmy damping is accomplished by restricting hydraulic flow within the steering valve. If the flaps and slats are retracted, 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 allow sufficient rudder pedal movement for full steering authority. The nose wheel steering system is equipped with a limit switch mounted on the nose landing gear shock strut adapter. When nose wheel steering exceeds approximately 40° from center, the switch opens an electrical circuit to the control valve and automatically prevents controlled steering through the rudder pedals. The NWS/AR lamp will go out whenever the controlled steering range is exceeded. When the nose wheels are returned to the normal steering range, (0° to 40°), controlled steering automatically reengages. Power for engaging steering is furnished from the essential dc bus.

Note

Nose wheel steering will not be available if the landing gear is extended using the landing gear emergency release handle.

NOSE WHEEL STEERING/AIR REFUEL BUTTONS.

A nose wheel steering/air refuel button (4, figure 1-24), is located on each control stick grip. The buttons are labeled NWS and A/R DISC. With the weight of the aircraft on the main landing 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. The nose wheel steering will engage and remain engaged as long as the NWS button is held depressed.

Note

- When the nose wheel steering button is depressed and released to disengage the system, a three second time delay is initiated. The system may be re-engaged during the three second period by depressing the button, but the holding relay will not be energized during this period.
- If for any reason the air refueling door is open the function of the button is unchanged.

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 main instrument panel (22, figure 1-5). The lamp will light when the nose wheel steering system is energized. For a description of the air refueling disconnect function of the lamp, refer to "Fuel Supply System," this section. The lamp receives power from the 28 volt dc essential bus.

Note

During ground operation when the air refueling door is open the nose wheel steering/air refueling indicator lamp will light to indicate door position and nose wheel steering cannot be monitored.

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 for power off braking. 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. Two hydraulic accumulators are provided in the system to supply brake system pressure for normal power off braking or failure of either hydraulic system. If the primary system fails, the hydraulic isolation valve will isolate the brake accumulators. If the utility system should fail, the brake accumulators are then isolated by non-return valves. 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. At this pressure all normal braking will be lost and the pedals will be fully depressed.

CAUTION

Do not actuate the brake pedals in-flight. 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, only emergency braking will be available by pulling the auxiliary brake handle.

After the priority valves close, the remaining fluid can be utilized only by pulling the auxiliary brake handle; however, this will lock the brakes under some conditions such as wet or icy runways. Normally on a dry runway the brakes will not lock.

AUTOMATIC BRAKING SYSTEM.

The automatic braking system functions to stop wheel rotation after takeoff and prior to gear retraction. When the aircraft has become airborne, moving the gear handle to the UP position will provide hydraulic pressure through the brake control valve at approximately 750 psi to one circuit (one-half) of the pressure pistons in each wheel. Automatic brake pressure is

applied simultaneously with main landing gear forward door (speed brake) opening pressure so that wheel rotation is stopped before the door is fully open, and the landing gear unlocks for retraction. The pressure is relieved after gear and flaps/slats are retracted as the brakes are isolated from the hydraulic system.

ANTI-SKID SYSTEM.

The anti-skid control system provides the following functions:

- Touchdown skid control.
- Proportional skid control.
- Lock wheel skid control.
- Anti-skid failure detection.

Touchdown skid control prevents the brakes from being applied when the weight of the aircraft is off the landing gear and the speed of both wheels is below 20 MPH. Proportional skid control operates throughout the aircraft ground speed range by utilizing wheel deceleration to reduce brake pressure in proportion to a skid tendency. Locked wheel skid control is activated above 20 MPH and causes either brake to be fully released if proportional skid control does not prevent a skid from occurring. Locked wheel skid control would function, for example, should a brake seize or if a wheel is unable to spin-up due to hydroplaning. The failure detection circuit will automatically return the brake system to manual control in the event of an anti-skid malfunction.

Anti-Skid Control Switch.

The anti-skid control switch (2, figure 1-4) is located on the left crew member'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 displacement.

Anti-Skid Caution Lamp.

An amber caution lamp labeled ANTI-SKID is located on the main caution lamp panel (figure 1-37). The lamp will light when the anti-skid switch is in the ON position and a malfunction has caused the anti-skid system to become de-energized. The anti-skid caution lamp will light anytime the landing gear is down and the anti-skid switch is not in the ON position.

Note

When the anti-skid caution lamp is lighted, anti-skid control is not available and braking will be in direct response to the pedal displacement.

AUXILIARY BRAKE HANDLE.

The auxiliary brake handle (7, figure 1-15), labeled 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 aircraft 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 (up to 3150 psi) is ported directly to the brake cylinders, bypassing the metering valves and the anti-skid valves. It should be noted that 1750 psi is the normal maximum pressure available from brake pedal actuation. Therefore, the auxiliary brake handle should not be pulled while the aircraft is in motion except when braking cannot be achieved by pedal actuation.

CAUTION

Pulling the auxiliary brake handle while the aircraft is moving may cause the wheels to lock, if normal brake accumulator pressure is available, and result in tire skidding or blowout, and may result in fire.

BRAKE HYDRAULIC HAND PUMP.

A hydraulic hand pump (figure 1-13), located in the main landing gear wheel well, is provided to replenish brake accumulator pressure during ground handling operation.

AIRCRAFT ARRESTING SYSTEM.

The arresting system provides for emergency arrestment of the aircraft. 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 1-15), located on the landing gear control panel, 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 pulling the handle 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 (figure 1-37). 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. On aircraft prior to modification by T.O. 1F-111-599, the slats must be retracted. Also when the speed brake is extended and the slat handle is moved out of the UP position, the speed brake will retract. For operation of the speed brake as a landing gear door refer to "Landing Gear System," this section.

Note

- The utility hydraulic system isolate switch must be in the NORM position for speed brake operation.
- Some smoke may enter the cockpit immediately following speed brake extension due to minute oil leakage from engine compressor bearings during deceleration.

Speed Brake Switches.

A three-position speed brake switch (4, figure 1-4), 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 left crew member's switch is detented in all positions. The right crew member's switch is spring loaded to OFF from both the IN and OUT positions and will override the left crew member's switch. When the right crew member's switch is released to OFF, the speed brake will move to the position selected by the left crew member's switch. When both switches are positioned to OFF, the speed brake is hydraulically locked in its present position. To maintain a constant load on the door, and to insure minimum drag the left crew member's switch must remain in the IN position, except during speed brake operation. The speed brake switches are activated to allow operation of the speed brake by a switch on the landing gear uplock when the landing gear is retracted.

GROUND ROLL SPOILERS.

Deceleration during ground roll is aided by symmetrical extension of the flight control spoilers which reduces aerodynamic lift and allows maximum effectiveness of the wheel brakes.

Ground Roll Spoiler Switch.

The ground roll spoiler switch (8, figure 1-18), located on the crew module left sidewall, has positions BRAKE and OFF. If the weight of the aircraft is on the landing gear and both throttles are in IDLE, positioning this switch to BRAKE will cause the flight control spoilers to extend. Under the same conditions placing the switch to OFF will retract the spoilers. With the spoiler switch positioned to BRAKE, if the aircraft weight is removed from the landing gear or if either throttle is advanced out of IDLE, the spoilers will automatically retract.

WING FLAPS AND SLATS.

WING FLAPS.

The wing flaps are full span multisection Fowler-type flaps. Integral with each flap section is a mechanically controlled vane. As the flap extends downward the vane is positioned by a mechanical linkage to provide the proper airflow through the space between the flap leading edge and the spoiler trailing edge. Each wing flap is divided into four sections that are mechanically connected and operate as one unit. The flaps are powered from the utility hydraulic system by a single hydraulic motor that is connected to a main drive actuator assembly. The flaps are extended by 6 mechanical actuators (3 for each flap) that are driven by the hydraulic motor through a system of associated gear boxes and torque shafts. An electric motor connected to the main drive actuator assembly provides an emergency mode of operation in the event of either hydraulic system failure. The main drive actuator is mechanically interlocked with the wing sweep control to prevent the wings from being swept aft of the 26 degree position when the flaps are not in the UP position. In the same manner the flap and slat handle is locked in the UP position whenever either the wing or wing sweep handle is positioned at greater than 26½ degrees.

Note

In the event slats/flaps do not extend with the wing sweep handle at the 26 degree detent, move the handle slightly forward of 26 degrees and reattempt extension.

Flap asymmetry monitoring devices are provided to detect a broken torque shaft. When the asymmetry device senses asymmetric flap position, a signal is sent to close the flap/slat drive shutoff valve and to engage brakes attached to the flap/slat torque shafts. Once the torque shaft brakes are engaged by this method, the flaps and slats cannot be extended or retracted by either the normal or emergency mode.

WING SLATS.

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 flaps and are connected to the flap drive assembly by flexible drive shafts. On the extend cycle, the slats will extend to the full down position before the flaps start to extend. On the retract cycle, the flaps will fully retract before the slats start to retract.

Note

If an asymmetrical slat condition occurs, the aircraft will enter a roll in the direction of the extended slat. The initial movement of the flaps will cause the slats and flaps to lock up.

ROTATING GLOVES.

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 (3, figure 1-1). 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/Slat Handle.

The flap/slat handle (9, figure 1-4), located on the left throttle panel has three positions labeled UP, SLAT DOWN and FLAP DOWN. A manually operated gate (8, figure 1-4) is provided between the SLAT DOWN and FLAP DOWN areas to permit slat operation without flap operation when the flaps are up and to permit ease of flap retraction without affecting slat position. The gate must be released when the handle is 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/slat drive control valve, directing utility hydraulic pressure to the flap/slat main drive actuator. The main drive actuator rotates the slat flexible shafts positioning the rotating gloves and extending the slats to a position that corresponds to the handle position. Moving the handle down to the gate will cause the slats to fully extend. Extension of the slats beyond 70 percent automatically places the flight control system in the takeoff and landing configuration if the flight control system switch is in NORM. (Refer to "Flight Control System Switch" in this section.) When the gate is released and the handle is moved into the flap down area, the main drive actuator will rotate the flexible shafts connected to the flap linear actuators, extending the flaps to a position corresponding to the handle position. The flaps can be set at an infinite number of positions between up and full down. A detent position is provided in the flap/slat control mechanism to aid in selecting the 15-degree flap position. On aircraft modified by T.O. 1F-111-766, a 25-degree detent is also provided as an aid in selecting the 25-degree flap position. Full down position of the flap/slat handle will provide 37.5 degrees of flap deflection. On aircraft modified by T.O. 1F-111-824, the full down position of the flaps is 34 degrees. The flap and slat drive mechanism is so designed that it will not extend the flaps until the slats are fully down and will not retract the slats until the flaps are up. Therefore moving the handle from the FLAP DOWN position to UP position will first cause the flaps to retract and then the slats to retract. Normal time for retraction or extension of the flaps and slats is approximately 12 seconds.

Note

The handle is inoperative when the flap and slat switch is in the EMER position.

Flap and Slat Switch.

The flap and slat switch (1, figure 1-25), located on the auxiliary flight control panel, has two positions marked EMER and NORM. 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 hydraulic system failure. When the switch is in EMER, the flap control shutoff valve is closed, disabling the flap drive motor, and the flaps and slats may be extended and retracted electrically by use of the emergency flap and slat switch.

Emergency Flap and Slat Switch.

The emergency flap and slat switch (2, figure 1-25), located on the auxiliary flight control panel, 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 operat-

Wing Sweep and Pylon System

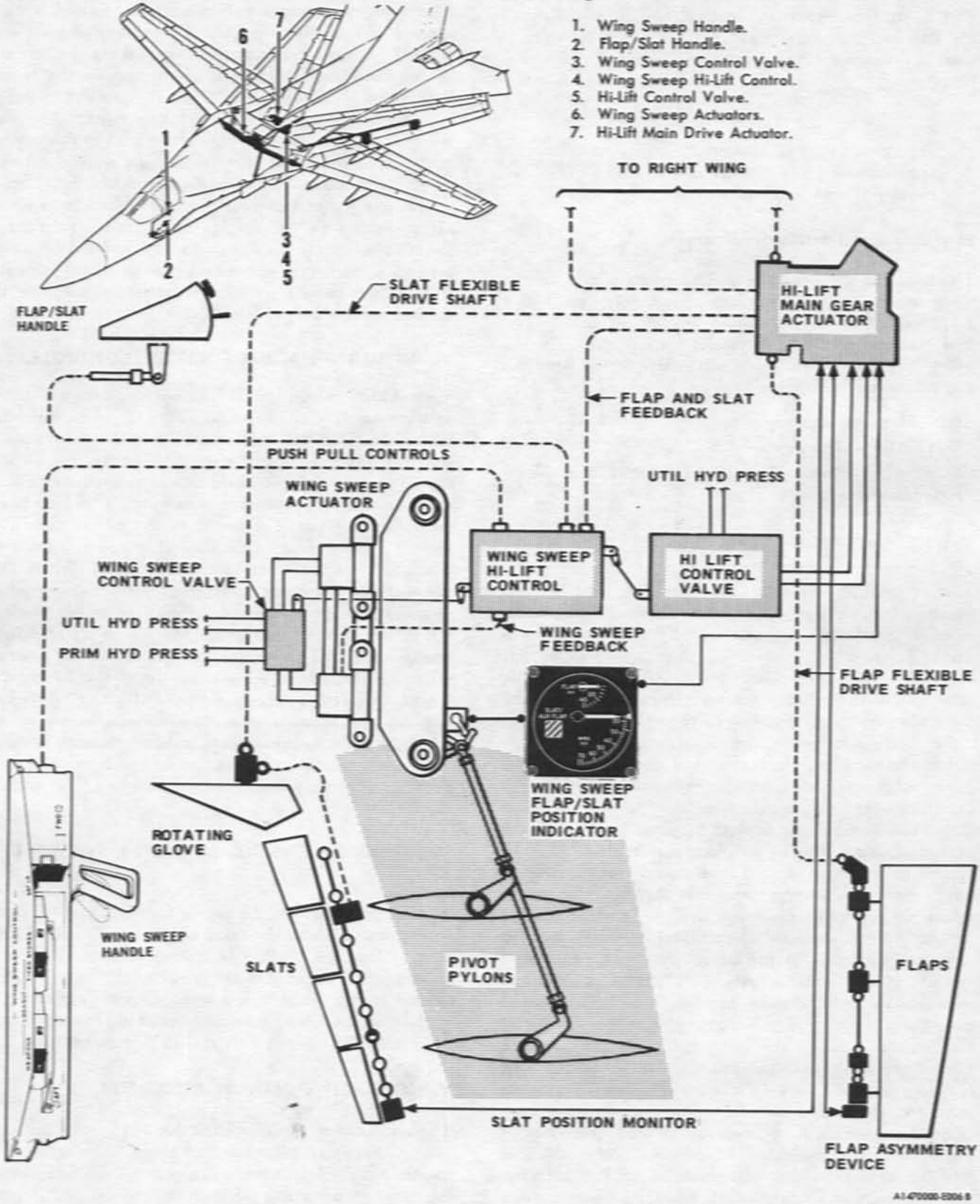


Figure 1-17.

ing the main flaps and slats in the event of either 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.

Note

Emergency flap extension or retraction takes approximately 60 seconds at 180 KIAS. This time will vary with airspeed.

Flap and Slat Position Indicators.

The flap and slat position indicators are a part of the wing sweep, flap/slat position indicators (10, figure 1-5), located on the left main instrument panel. The flap position indicator provides flap position in degrees. The slat indication is a window which provides the following indications:

- UP—Slats retracted
- SLAT DN—Slats down
- Crosshatch—Power off or slats in transit or at an intermediate position.

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 1-17). The left actuator is furnished power by the primary hydraulic system, and the right 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 nonreversing threads in the actuator. Also, a mechanical interlock prevents the wing sweep handle from being moved past the 26½ degree position when either the flap/slat handle is out of the UP position or the flaps are out of the fully retracted position. A 16 degree lock is also installed. This lock was intended to prevent sweeping the wings aft of 16 degrees when the auxiliary flaps, which have been deactivated, were out of the zero position. A loss of power to the solenoid which operates the lock will result in locking the wing sweep handle at 16 degrees, if moved to this position. However, this does not create a problem since all critical phases of flight, i.e., takeoff, landing, etc., can be accomplished with the wings in this position.

WING SWEEP CONTROL HANDLE.

The wing sweep control handle (5, figure 1-18) is shaped like a pistol grip and is spring loaded to a stowed position under the canopy sill on the left side 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. The handle is pulled aft to sweep the wings aft and pushed forward to sweep the wings forward. As the handle is moved an index mark on the wing sweep position indicator follows the handle position to assist in selecting the desired wing sweep position.

WING SWEEP HANDLE LOCKOUT CONTROLS.

Two wing sweep handle lockout controls (6, figure 1-18), 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 54 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 between 72.5 and 26 degrees.

WING SWEEP HANDLE 26 DEGREE FORWARD GATE.

A wing sweep handle 26 degree forward gate (4, figure 1-18), 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 is a part of the wing sweep, flap/slat position indicator (10, figure 1-5), 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. An index mark at 26 degrees provides a reference for selecting this position. A movable command index on

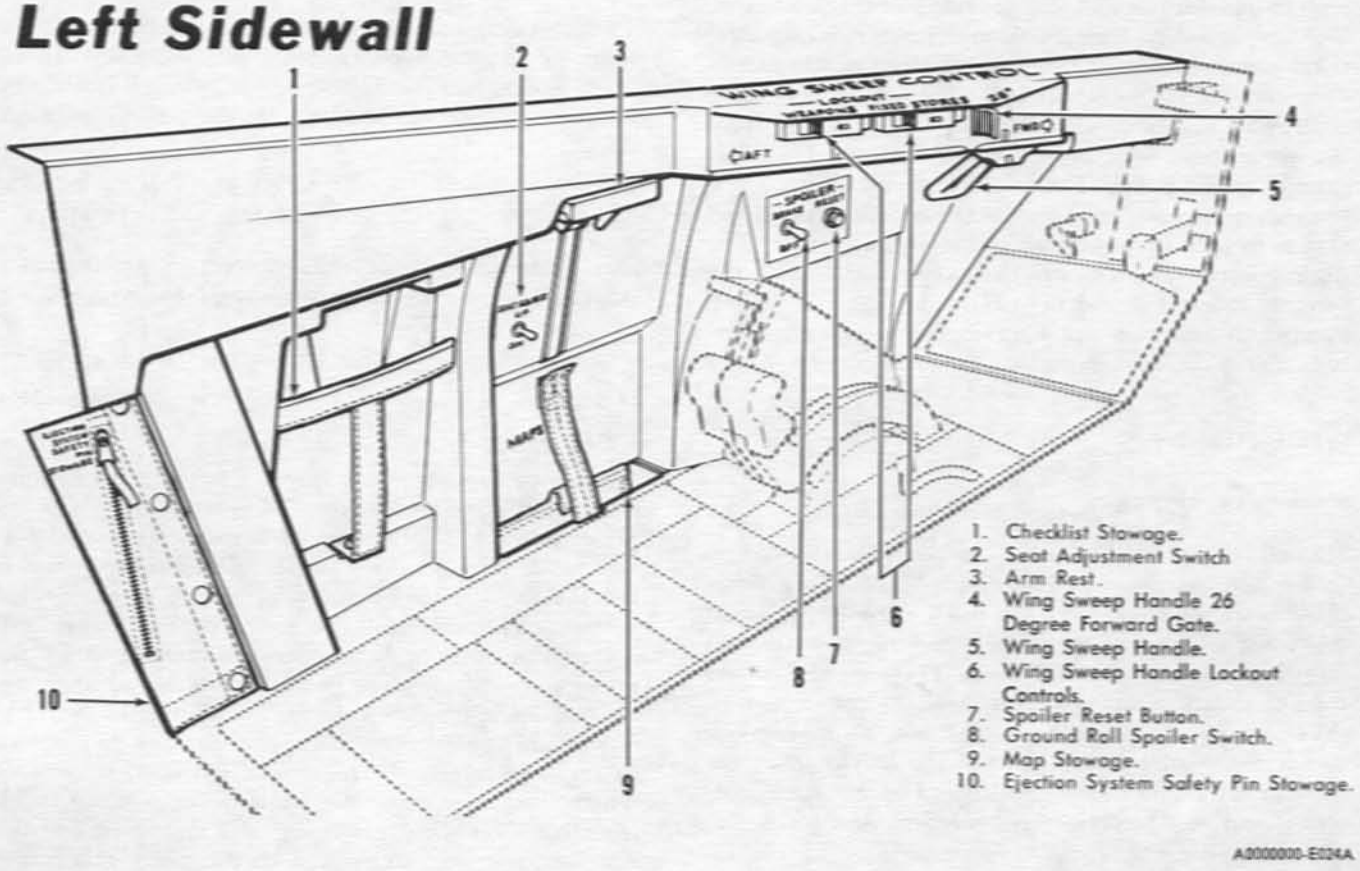


Figure 1-18.

the outside of the scale is provided to assist in setting the wing sweep handle to the desired position. 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 pointer.

FLIGHT CONTROL SYSTEM.

The primary flight control system provides control of the aircraft by movement of the primary control surfaces. The primary control surfaces consist of a rudder, spoilers on each wing and movable horizontal stabilizers. Pitch attitude of the aircraft is controlled by symmetrical deflection of the horizontal stabilizer surfaces. Roll attitude is controlled by asymmetrical deflection of the horizontal stabilizer surfaces; and when the wing sweep angle is less than 45 degrees, roll control is aided by action of two spoilers on top of each wing. Yaw control of the aircraft is accomplished by deflection of a rudder surface located on the trailing edge of the vertical stabilizer. Hydraulic servo actuators are used to produce control surface movement. The control stick at each crew station is mechanically and electrically interconnected with the flight control system. The right stick may be removed for various mission requirements. This must be accomplished while the aircraft is on the ground. When the right stick has been removed, an electrical plug is inserted in place of the stick to maintain electrical continuity. The two sets of rudder pedals are mechanically linked together. A system of push-pull tubes, bell cranks, and pulleys are used to connect the cockpit controls with the rudder and horizontal stabilizer hydraulic actuators. The linkage connections are secured with self-retaining bolts, which use self-locking cotter keyed nuts. Loss of the cotter key and self-locking nut will not cause separation of the connection. The stability augmentation system employs redundant sensors, electronic circuitry and electro-hydraulic dampers. The three damper actuators, the horizontal stabilizer actuators, and the rudder actuator are supplied by both primary and utility hydraulic systems and can operate on either system should one system fail. The pitch and roll damper response (gain) is varied by a self-adaptive system as flight conditions change. Command augmentation, through the pitch and roll dampers, augments the pilot inputs to provide a near constant relationship between control force and aircraft response throughout the operational en-

velope. Automatic failure detection and rejection, as well as self-test features, are provided in the pitch, roll, and yaw stability augmentation systems. Should electrical power be absent from one or more of the redundant computers, the applicable channel caution lamp will light. Power to all computers is controlled from the three computer power switches located on the ground check panel. The pitch and roll damper systems accept inputs from the CADC and the navigation system to provide pitch and roll autopilot modes. The flight control system functions in conjunction with the terrain following radar (TFR) through the pitch damper to maintain the aircraft at a preselected altitude above the terrain.

PITCH CHANNEL.

Mechanical Linkage.

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. Figure 1-19 is a simplified schematic of the mechanical flight control linkage. Control stick centering and feel forces are provided by the pitch feel spring.

Parallel Trim Actuator.

The parallel trim actuator will cause displacement of the horizontal stabilizers and will also cause displacement of the control stick unless the stick is manually restrained.

Series Trim Actuator.

The series trim actuator output is added downstream of the pitch feel spring. Displacement of this actuator will cause the horizontal stabilizers to displace, but it will not normally cause displacement of the control stick. Likewise, displacement of the pitch damper servo will cause the horizontal stabilizers to displace but will not normally cause displacement of the control stick.

Control Stick Movement.

Control stick displacement can occur due to series trim actuator or pitch damper displacements if, for some reason, their action can not cause displacement of the pitch command output rod shown in figure 1-19. (Refer to "Stick Talk Back," this section.) Control stick stops are provided in the crew module to limit the available stick motion.

Pitch Command Limits.

The following table summarizes the stop limits and the limits of the other components just described. In all cases, the values listed are in degrees of horizontal stabilizer deflection assuming all the other inputs are zero.

INPUT	NOSE UP LIMIT	NOSE DOWN LIMIT
Control stick	22 degrees	14 degrees
Pitch mixer	25 degrees	15 degrees
Parallel trim actuator	10 degrees	8 degrees
Series trim actuator	10 degrees	4 degrees
Pitch damper servo	13 degrees	13 degrees

With both trim actuators at zero and the pitch damper OFF, total stick travel from neutral to full aft is approximately 6½ inches, and from neutral to full forward is approximately 4 inches. When takeoff trim is set, the parallel trim actuator drives to a zero degree surface command, and the series trim actuator will drive to position both horizontal stabilizers to 3.8 degrees trailing edge up. With the pitch damper on, control stick displacement or parallel trim actuator displacement will cause the pitch damper to displace in response to signals from the pitch stick transducer which usually limits the amount of control stick available. Combined pitch and roll movements of the control stick are transmitted to the pitch/roll mixer assembly where they are converted into left and right horizontal stabilizer actuator command signals. (See figure 1-19.) Deflection of the left and right horizontal stabilizers is limited by the horizontal actuator ram stroke. The nominal limits are 30 degrees trailing edge up and 15 degrees trailing edge down.

Power Supply.

Power to the pitch damper servo, series trim actuator, parallel trim actuator, and stick transducer is controlled from the three computer power switches on the ground check panel. When these switches are turned OFF, the pitch damper actuator is hydraulically driven to its zero position and the series and parallel trim actuators will stop. If hydraulic pressure is not available, the position of the pitch damper servo is indeterminate.

Pitch/Roll Mixing.

Combined pitch and roll movements of the control stick are transmitted by the linkage of their respective channel to the pitch/roll mixer assembly. In this assembly the pitch/roll commands are summed mechanically and converted into left and right horizontal stabilizer actuator command signals. (See figure 1-19.)

Pitch and Roll Mechanical Schematic (Typical)

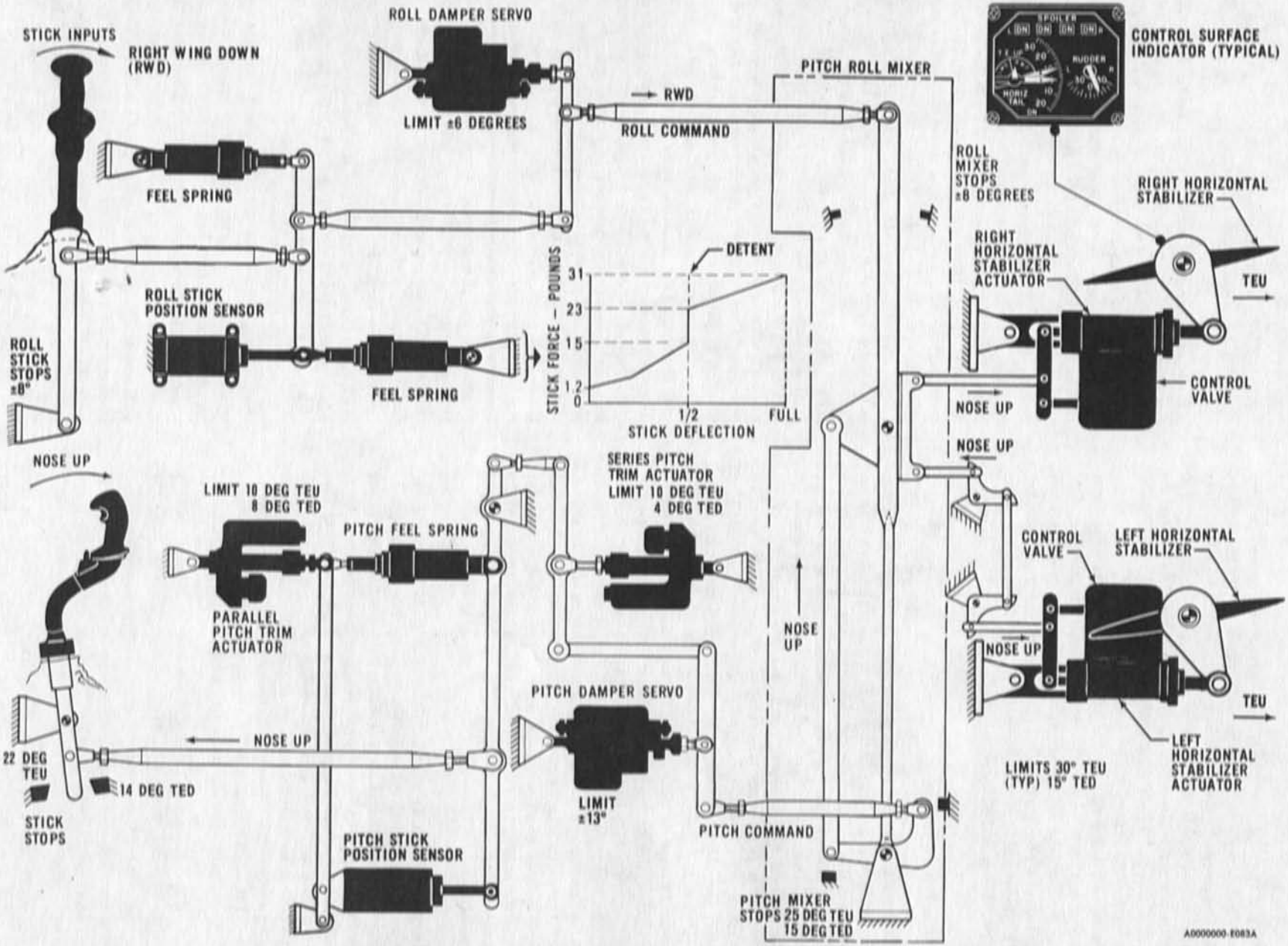


Figure 1-19.

Mixer Outputs.

The output of the pitch roll mixer is transmitted by two push-pull tubes. The left push-pull tube causes displacement of the control valve located on the left horizontal stabilizer actuator. When the control valve is displaced from neutral, hydraulic fluid is ported to the left-hand horizontal stabilizer actuator, which results in horizontal stabilizer displacement. The right push-pull tube moves the control valve on the right horizontal stabilizer actuator in a similar fashion. Deflection of the left and right horizontal stabilizers is limited by the horizontal stabilizer actuator ram stroke. The nominal nose up limit is 30 degrees and the nose down limit is 15 degrees.

Pitch Trim System.

Pitch trim can be effected from one of three inputs. These inputs are parallel pitch trim, series pitch trim, and auxiliary pitch trim through the pitch damper servo. Parallel pitch trim allows the pilot to change the stick neutral point and is the prime trim mode in the takeoff and landing configuration. Also it may be used as a supplemental trim when the flight control system is in normal. Series pitch trim is provided to carry the elevator required for trim and serves to minimize steady state errors during autopilot modes and auto TF operation. An auxiliary pitch trim system, utilizing the series trim actuator and pitch computer, is provided as a backup trim system and to be used if a trim failure occurs. The operation of each trim input is determined by (a) the auxiliary trim switch position, (b) the pitch damper switch position, and (c) the control system configuration as determined by the slat position and the control system switch. Figure 1-20 summarizes the trim system as a function of these variables.

Parallel Pitch Trim. Parallel pitch trim is the prime trim mode in the takeoff and landing configuration. It may also be used as supplemental trim when the flight control system is in normal. During normal operation, parallel trim will be at neutral for one "g" flight. The parallel pitch trim actuator is driven from the control stick(s) trim button only if the pitch damper switch is in DAMPER and the auxiliary pitch trim switch is in STICK. The parallel pitch trim actuator can also be commanded to drive by one of the following means:

- By depressing the takeoff trim button which centers the parallel trim actuators at the takeoff position (zero degrees).
- By turning the pitch damper switch to OFF which centers and locks parallel trim at takeoff position (zero degrees).
- By moving the auxiliary pitch trim switch out of the STICK position which centers and locks parallel trim at takeoff position (zero degrees).
- By placing the .pitch autopilot/damper switch to AUTOPILOT or placing the auto TF switch to AUTO TF, which centers and locks parallel trim at takeoff position (zero degrees).

The parallel trim actuator, when trimmed, will cause the control stick, pitch damper, and horizontal stabilizers to displace. The control stick will be centered and the pitch trim function of the stick trim button is disabled when pitch autopilot is selected, auto TF is selected, and during TFR fail safe flyup maneuvers.

Series Pitch Trim (Autotrim). The series pitch trim actuator drives the horizontal stabilizers but does not normally drive the control stick. The actuator is locked at its present position when the control system is switched to the takeoff and land configuration. The takeoff and landing configuration is established by either placing the control system switch to T.O. & LAND or by extension of the slats. The series trim is driven to its 3.8-degrees trailing edge up position when the takeoff trim button is depressed. While the control system is in its normal flight configuration the series pitch trim will act to maintain the aircraft's normal acceleration at a value proportional to stick position. This is accomplished by the series trim driving in response to a pitch damper position signal. When the pitch damper is at zero the series trim will stop driving. This is called the null mode of operation. The null mode will be in effect unless the pitch damper is turned off or takeoff and land configuration is established. If the pitch damper is turned off, series trim will lock at its present position, and it can then be driven from the control stick trim button(s). The authority of the series pitch trim actuator is 10 degrees trailing edge up and 4 degrees trailing edge down. The actuator is rate limited at 1.4 degrees per second. Displacement of the control stick by force or by trim will command an increase or decrease in normal acceleration through the command augmentation feature. Changes in the elevator required to hold the aircraft in the one "g" flight condition while the stick is at neutral are provided by the pitch series trim. The pitch series trim is driven from the pitch damper position transducer. If the pitch damper inputs are zero, i.e., no pitch rate and one "g" normal acceleration, and the stick is at neutral, then the pitch damper will be at zero, and the series trim actuator will stop. If the elevator required to hold the aircraft at this condition varies due to power or wing sweep changes, then the damper will displace to oppose aircraft rotation. This will cause the series trim to drive until the damper inputs again become zero. Thus the series trim system provides the steady state elevator required to maintain the aircraft in trim. Because of this action, the stick will be at the same position for one "g" flight regardless of speed, unless the control system is in the T.O. & LAND configuration.

Flight Control System Configuration vs Pitch Trim Operation

PITCH TRIM DURING MANUAL CONTROL MODES

FLIGHT CONTROL SYSTEM CONFIGURATION			
Auxiliary Pitch Trim Switch	*Autopilot Damper Switch* **	**Flight Configuration*	*PITCH TRIM OPERATION*
STICK	DAMPER	Clean	1. Stick trim drives parallel pitch trim 2. Series trim actuator in damper null mode
STICK	DAMPER	Takeoff & Land	1. Stick trim drives parallel pitch trim 2. Series trim locked at existing position
STICK	OFF	Clean or Take-off & Land	1. Parallel pitch trim actuator centers and locks at zero 2. Stick trim drives series pitch trim actuator
OFF	OFF	Clean or Take-off & Land	1. Parallel pitch trim actuator centers and locks at zero 2. Series trim actuator may be driven from the stick trim switch or the auxiliary pitch trim switch
OFF	DAMPER	Clean	1. Parallel pitch trim actuator centers and locks at zero 2. Series pitch trim actuator in damper null mode 3. Auxiliary pitch trim switch drives damper
OFF	DAMPER	Takeoff & Land	1. Parallel pitch trim actuator centers and locks at zero 2. Auxiliary pitch trim switch drives damper and also drives the series pitch trim actuator

*Flight Configuration:

Clean: Flight control system switch—NORM and slats-up

T.O. & L: (1) Flight control system switch—NORM and slats-down or
(2) Flight control system switch—T.O. & LAND

PITCH TRIM DURING AUTOMATIC CONTROL MODES

MODE	*PITCH TRIM OPERATION*
AUTO TF	1. Stick trim deactivated and parallel pitch trim actuator centers 2. Auxiliary pitch trim switch must not be used in this mode 3. Series pitch trim actuator is driven by Auto TF climb/dive error
PITCH AUTOPILOT MODES	1. Stick trim deactivated and parallel pitch trim actuator centers 2. Auxiliary pitch trim switch must not be used in these modes 3. Series pitch trim is driven by pitch autopilot error signal

Figure 1-20.

Note

During ground operation, series trim is not able to null the damper since there is no aircraft response. This will result in trim drift either nose up or nose down while the slats are retracted and the control system switch is in NORMAL.

Auxiliary Pitch Trim. The auxiliary pitch trim system is armed when the auxiliary pitch trim switch is placed to the center (OFF) position. When the switch is placed to the NOSE UP or NOSE DOWN position, pitch trim is provided by one or both of the following:

- Positioning an auxiliary pitch trim integrator in the feel and trim assembly, which sends command signals to the pitch damper.
- By directly driving the pitch series trim actuator.

The net effect is a change to the aircraft pitch trim. Figure 1-20 defines whether (1), (2), or (3) is used as a function of the flight control systems configuration.

WARNING

Improper operation will result if the auxiliary pitch trim switch is operated during auto TF or pitch autopilot modes.

The authority of the auxiliary pitch trim integrator command to the pitch computer will be a function of the pitch computer gain.

Stability Augmentation.

Pitch stability augmentation is provided through the redundant pitch damper system to provide aircraft damping and to improve the handling characteristics of the aircraft. (Refer to "Pitch Channel and Pitch Damper Redundancy," this section.) Figure 1-21 shows the basic mechanical system and the pitch damper system. The pitch computer provides damping signals to the pitch damper servo in response to normal acceleration and pitch rate feed back signals from the accelerometers and gyros. The stick position transducer signals are zero when the control stick is at neutral, thus providing a zero command signal to the pitch computer. Series trim will provide the steady state elevator to maintain the aircraft in trim; the normal accelerometer and pitch rate gyro signals will provide damping, and the aircraft will be in one "g" flight.

Command Augmentation.

The effectiveness of any control surface varies with the flight conditions. At low speed and high altitude several degrees of elevator are required to command a one "g" maneuver while at high speed and low altitude it may take less than a degree. With the pitch damper off, stick force and surface movement are directly related to stick motion; thus heavy stick forces will be required at low speed and light forces will be required during high speed, low altitude flight. With the pitch damper on, the stability augmentation feature, sensing pitch rate and normal acceleration, may oppose the initial aircraft response. However, when the stick is moved to command "g's" the stick input to the elevator is augmented by the pitch damper so that for practically any flight condition (high speed or low speed, at all altitudes) "g" response will be about the same for a given stick force. The command augmentation feature accomplishes this by adding elevator, if it is needed, to get the aircraft started on its pitch change, then as it approaches the commanded "g's," the damper and series trim will move the elevator so that aircraft response zeros out on the commanded "g." At low airspeed where control effectiveness is low, the damper will work to give the higher elevator deflection needed to reach commanded "g's." As airspeed increases, less deflection is required and the system works the damper to result in less deflection for the same stick movement. Under steady state conditions, the series trim is driven by the out-of-neutral damper signal. This has two simultaneous effects: the series trim action provides the final input to achieve the commanded response and also nulls the damper, allowing full authority to be restored. Basically the command augmentation system changes the elevator deflection so that the airplane response to stick inputs is the same regardless of airspeed or altitude.

WARNING

The command augmentation feature of the flight control system will attempt to maintain the commanded level of pitch rate and "g" force regardless of variations in airspeed. During flight conditions where airspeed is decreasing, the pitch control surface will be commanded to increase angle-of-attack, without additional pilot input, in an attempt to maintain the commanded level of pitch rate and "g" force. Failure to monitor and control angle-of-attack within limits can result in inadvertent, rapid entry into a stalled condition.

Automatic Pitch Control.

The pitch computer can also accept inputs from either the autopilot pitch submodes or the TFR computer to provide automatic pitch control through the pitch damper. Refer to figure 1-21. Interlocks are provided to prevent incompatible mode selection. Refer to "Autopilot System" and "Terrain Following Radar (AN/APQ-110)," this section.

Pitch Channel Electrical Schematic (Typical)

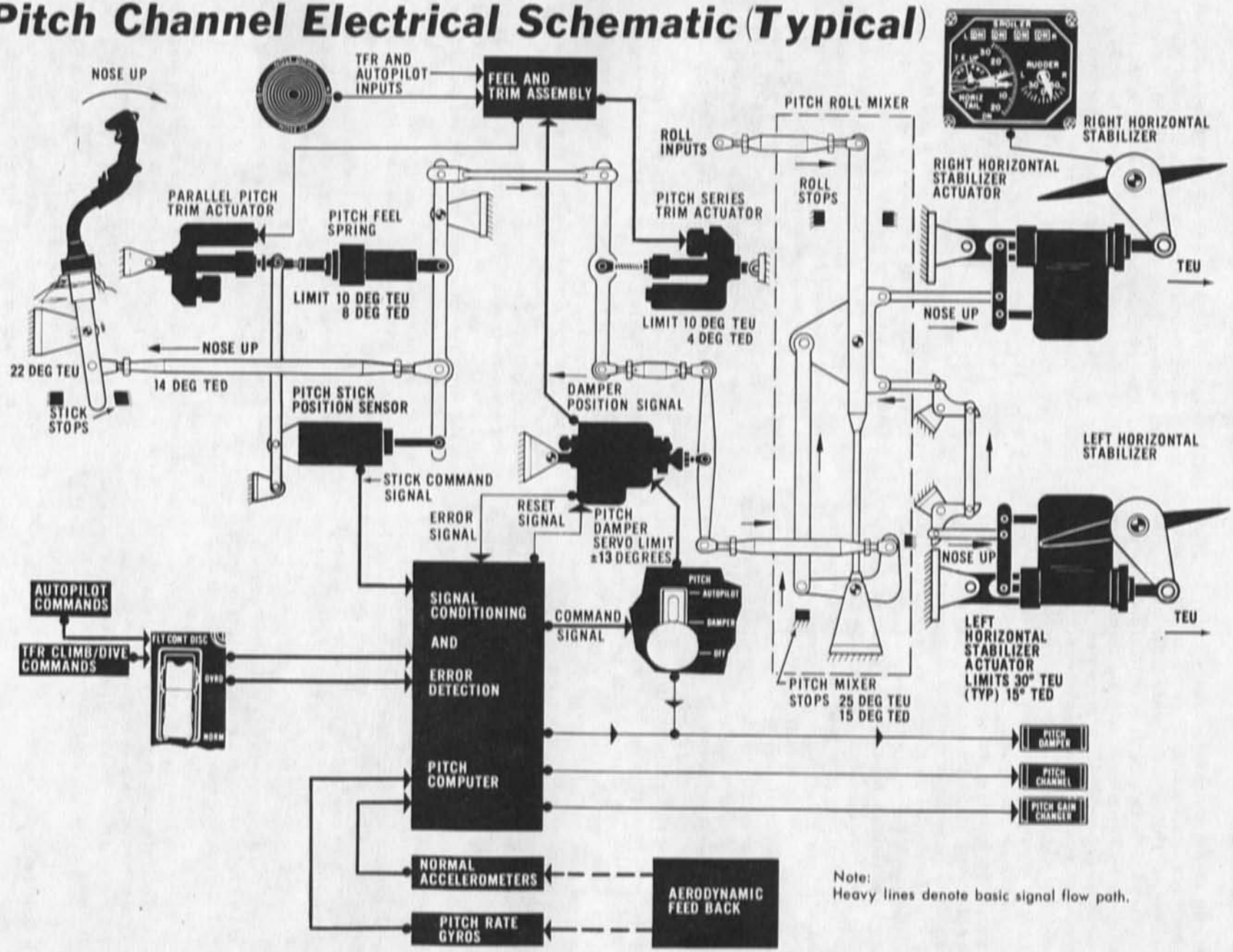


Figure 1-21.

Pitch Gain Control.

The gain of the command signal sent to the pitch damper servo is automatically varied as flight conditions change. This is accomplished by continuously monitoring the pitch rate gyro and normal acceleration signals to determine if the gain should be either increased or decreased. Since the system modifies its gain as a function of its own performance, it is called a self-adaptive gain system. In general the required gain varies inversely with dynamic pressure. Elevator effectiveness is affected by many variables such as mach number, altitude, wing sweep angle, gross weight, center-of-gravity, and external stores. The damper contributions in response to command augmentation inputs is a function of gain, therefore, a gain increase will compensate for reduced elevator effectiveness by giving more elevator for the same command, thereby holding the aircraft response and damping nearly constant. If the available gain is too low, the aircraft response will appear sluggish to stick commands. If the gain becomes too high, a small amplitude pitch oscillation may exist for a few cycles until the gain control circuit, which senses this oscillatory condition, reduces the gain to the proper value. The frequency of this oscillation (the adaptive frequency) will be between 1.4 to 3.0 cycles per second; the gain changer is designed to either increase or decrease the gain for this range of input frequencies depending on their amplitude and persistence. Also, the gain changer will increase the gain for inputs less than 1.4 cps. Rapidly changing flight conditions can result in the computed gain lagging the optimum gain for a short period. Aircraft motion due to turbulence or aircraft vibration, such as experienced with speed brake operation, will cause the adaptive gains to decrease. When the pitch damper switch is turned OFF, the pitch gain is driven to its minimum of 12 percent, and the pitch damper servo centers. If the gain becomes high enough due to a malfunction to cause the adaptive frequency to persist, the gain can be reset to its minimum value by momentarily cycling the pitch damper switch OFF and then back to the DAMPER position. This should stop the oscillation.

Note

When the damper switch is positioned to the DAMPER position, the gains may require up to 2 minutes to increase to the optimum value. During this time the aircraft response and damping may be noticeably reduced. Consequently if the damper switch is cycled during TFR operation, aircraft response may be degraded up to one minute after the switch is returned to the damper position.

Pitch Gain Lock.

When the flight control system is in the takeoff and land configuration, the pitch gain is locked.

Artificial Stall Warning System.

On aircraft modified by T.O. 1F-111-891, the artificial stall warning system consists of a rudder pedal shaker, a stall warning lamp, and an audible stall warning signal. The system is automatically armed by the landing gear squat switch when the aircraft becomes airborne. The pedal shaker, lamp, and audible signal all occur simultaneously when either of the following conditions exists:

1. When the wings are swept forward of 50 (±2) degrees and the true wing angle-of-attack exceeds 14 (+0.25, −0.75) degrees.
2. When the wings are swept aft of 50 (±2) degrees and the true wing angle-of-attack is greater than 14 degrees, the stall warning system will be activated when the probe angle-of-attack, in degrees, (independent of the CADC) plus the pitch rate, in degrees per second, total 18 (±1). Since the angle-of-attack presented on the AMI is compensated as a function of mach number, the AMI reading for stall warning activation will vary as mach number changes. The AMI reading at which stall warning will occur for zero pitch rate is as follows:

Less than mach 0.30	18 (±1.6) degrees
Greater than mach 0.45 but less than mach 1.25	19.7 (±1.6) degrees
Greater than mach 1.40	18.8 (±1.6) degrees

In addition to the above conditions, when the flight control disconnect switch is in the OVRD position and the true wing angle-of-attack is greater than 14 degrees the stall warning lamp and audible signal will occur regardless of the wing sweep position, however the pedal shaker will be inoperative. The stall warning lamp (24, figure 1-5) is a flashing red lamp located on the left main instrument panel. When lighted the word STALL appears on the face of the lamp. Lamp intensity is controlled by the malfunction and indicator lamp dimming switch when the flight instrument lighting control knob is on. The audible stall warning signal is a continuous tone applied to the headsets of both crew members. The stall warning audible signal may be silenced by depressing the landing gear horn silencer button. Silencing of either the landing gear warning horn or the stall warning signal will not prevent subsequent audible tone warning from the other circuit. Operation of the horn silencer will not deactivate the stall warning lamp or the rudder pedal shaker. This system may be ground checked through use of the malfunction and indicator lamp test button.

Note

During ground checks, if the flight control system is in the takeoff and landing configuration, the rudder may deflect due to an adverse yaw compensation input when the malfunction and indicator lamp test button is depressed.

Aircraft not modified by T.O. 1F-111-891, are equipped with an artificial stall warning device in the form of a mild rudder pedal shaking to provide additional stall warning. The system is automatically armed by the landing gear squat switch when the aircraft becomes airborne. The system is activated when the probe angle-of-attack, in degrees, plus pitch rate in degrees per second, total 18 (±1.0). The probe angle-of-attack signal is fed directly into the stall warning circuit, and is completely independent of the central air data computer. When angle-of-attack is increasing rapidly, the pitch rate will cause the pedal shaker to be energized before the angle-of-attack limits are exceeded. With the flaps and slats extended, the stall warning is activated for zero pitch rate at 18.0 (±1.0) degrees true wing angle-of-attack. With the flaps and slats retracted, the stall warning device will be activated for zero pitch rate at 19.7 (±1.0) degrees true wing angle-of-attack up to mach 1.3 and 18.8 (±1.0) from mach greater than 1.3. Since the angle-of-attack as presented on the airspeed mach indicator (AMI) is compensated as a function of mach number, the AMI reading for pedal shaker operation will vary as mach number changes. The AMI reading at which stall warning will be activated for zero pitch rate is shown below:

Mach No.	AMI Reading
• Less than mach 0.30	18 (±1.6) degrees
• Greater than mach 0.45 but less than mach 1.25	19.7 (±1.6) degrees
• Greater than mach 1.40	18.8 (±1.6) degrees

For every degree per second of pitch rate, the AMI readings will be one degree less.

Note

During maneuvers where high pitch rates are generated such as dive recoveries and in pitch-outs, the stall warning will be activated at relatively low angles-of-attack (approximately 12.0 degrees). This should cause no concern as long as angle-of-attack is monitored to assure that the limits are not exceeded.

Pitch Channel Automatic Switching.

Certain changes occur within the pitch channel in the flight control system during the takeoff and landing phase. Automatic switching is accomplished by the slat position and the weight on gear signals. The control system switch is usually in the NORM position, but when positioned to the T.O. & LAND position, it will override the slat position signal and place the control system in the takeoff and landing mode. The following describes the configuration for four different conditions:

1. Taxi—Weight on gear, slats down, pitch damper switch—DAMPER.
   a. Series trim is locked at its existing position but may be set to 3.8 degrees trailing edge up by the takeoff trim button.
   b. Pitch gains are locked at 30 percent.
   c. The stall warning device is not armed.
   d. Normal accelerometer signals are locked out.
2. Lift off—Slats down, weight off gear, pitch damper switch — DAMPER, auxiliary trim switch — STICK. Takeoff trim set prior to takeoff.
   a. Series trim is locked at 3.8 degrees trailing edge up.
   b. Parallel trim can be used to trim the aircraft as required.
   c. The stall warning device is armed.
   d. Normal accelerometer signals are locked out.
3. Cruise—Slats retracted, gear up, pitch damper switch—DAMPER.
   a. Series trim actuator is in the null mode.
   b. Pitch gain is in its adaptive mode and automatically increases or decreases the gain of the pitch damper command signal as flight conditions change.
   c. Parallel trim should be used to trim the aircraft to 1-g flight. The stick will be at neutral when this is accomplished.
   d. The stall warning device is armed.
   e. Normal accelometer signals are active.
4. Slat extension prior to landing.
   a. Series trim actuator locks at its existing position.
   b. Pitch rate gain is locked.
   c. Parallel trim can be used.
   d. The stall warning device is armed.
   e. Normal accelerometer signals are locked out.

In the event that the flight control system is not in the takeoff and land configuration, the pitch gain changer and roll gain changer caution lamps will both light in flight when the landing gear handle is in the DN position. If the proper switching subsequently takes place by extension of the slats or by placing the control systems switch to T.O. & LAND, both caution lamps will go out, indicating the proper configuration has been established.

Pitch Channel and Pitch Damper Redundancy.

All stability augmentation signals enter the pitch computer where they are converted into three command signals. Computer inputs are supplied from three separate sources to three identical pitch computer boards. Each board separately computes a pitch gain and a pitch command signal. The three separately computed command signals are inputs to signal selectors 1, 2, and 3. Each signal selector rejects the highest signal and the lowest signal. The output of each selector is the remaining signal, or the middle value signal. Each

board has a signal comparator circuit, which compares that board's input signal with the selector's output signal. Should these signals be significantly different, the pitch channel caution lamp will light. For an initial failure, which causes a pitch channel caution lamp to light, the operation of the pitch damper system will be unaffected. However, a subsequent failure of another branch could result in (a) no change in operation, (b) zero pitch damper commands, or (c) a hard over pitch damper. The operation depends on the nature of the first, and subsequent failure(s). If the first failure resulted in a zero command from the affected board, normal operation can be continued and the pitch channel caution lamp can be reset. If the lamp cannot be reset, this means that the first failure is a hard over failure, and a second failure in the same direction will cause a hard over damper. For this reason, it is recommended that flight conditions be changed to observe the damper off envelope and then turn the pitch damper OFF. When the pitch damper is turned OFF, the pitch damper caution lamp will light.

Signal Selection.

The output of each signal selector will be the same for a single malfunction. Downstream of the signal selectors are three servo amplifiers, each of which receives its signal selector's output and in-turn sends a command signal to its separate servo valve within the pitch damper servo. When the pitch damper switch is turned OFF, the amplifier currents to the damper servo go to zero and the damper is hydraulically centered.

Damper Hydraulic Logic.

The pitch damper has two active valves and one model valve. The average of the command signals from the two active valves hydraulically control the movement of the damper output rod. A third servo valve controls a model servo and does not control the damper rod output. The position of the damper output rod and the position of the model servo are compared to detect malfunctions. Should a malfunction exist, the damper output rod position will not agree with the model servo position and the pitch damper caution lamp will light. Hydraulic logic within the pitch damper servo will identify the discrepant servo valve command. If the failure is due to one of the two active valves controlling the damper output rod, a vote will occur and the discrepant valve is hydraulically shut off. Control of the output rod is then dependent upon the command from the remaining active servo valve. A transient will be felt in the aircraft when this vote occurs. The discrepant valve can be placed into operation by depressing the damper reset button. Should the valve still have a discrepant output command, another vote will occur and the valve will again be shut off. Should the discrepant valve be the model valve, the pitch damper caution lamp will light, but the pitch damper rod will not be affected, and a transient will not be felt.

Hydraulic Servos.

The two active servo valves which control the damper output rod are supplied from separate hydraulic systems. In the event of a single hydraulic system failure, the valve controlled by the failed hydraulic system will be hydraulically shut off, and the pitch damper caution lamp will light. Normal damper operation will continue by using the remaining active servo valve and hydraulic system. In this case the damper reset button has no effect unless hydraulic pressure returns to normal.

Pitch Gain Changer Redundancy.

Within the pitch computer, three separate circuits compute the required gain. These three outputs are processed through their gain selectors in the same manner described for the signal selectors. Should a discrepancy exist between the three separate gain calculations, a pitch gain changer caution lamp will light, indicating failure of one of the three circuits. Operation will be unaffected until a second failure occurs. If the gain changer light can not be reset, a second failure could cause the total gain to go to minimum, maximum, or be unaffected. For this reason, decrease in airspeed is recommended if the lamp can not be reset.

ROLL CHANNEL.

Mechanical Linkage.

Lateral movement of the control stick is transmitted to the pitch roll mixer assembly by a system of push-pull rods and bellcranks. The pitch roll mixer adds the roll commands to the pitch commands and sends summed commands to the left and right horizontal stabilizer control valves and actuators. Figure 1-19 shows a simplified mechanical schematic of the roll channel linkage and damper system. Stick centering and stick feel forces are provided by the roll feel assembly. In this assembly, two feel springs are provided. When compressed, the low gradient feel spring provides nonlinear stick forces until its limit is reached at one-half stick travel. One-half stick displacement commands a 2-degree roll displacement through the pitch/roll mixer to each of the horizontal stabilizers with a stick force detent being encountered at this point. Approximately eight additional pounds must be exerted on the control stick before additional travel can be achieved. The high gradient feel spring breakout creates this force detent and provides the stick force gradient until maximum stick deflection is achieved. Maximum stick deflection commands 8 degrees of roll displacement through the pitch/roll mixer to each of the horizontal stabilizers. With the roll damper off the available stick deflection is limited by stick stops located within the cockpit to ±8 degrees of mechanical command. Figure 1-19 shows the gearing, and approximate stick

force provided by the roll feel assembly. Stick breakout force is approximately 1.2 pounds. A 15-pound force is required to reach the detent; 31 pounds will give maximum stick deflection. The roll damper servo also provides roll control inputs to the pitch roll mixer. The roll damper is a redundant electrohydraulic servo actuator with an authority of ±6 degrees. The roll damper servo is used to provide roll trim, stability augmentation, and command augmentation to the horizontal stabilizers. The roll damper servo is identical to the pitch and yaw damper servos.

Power Supply.

The electrical power to the roll damper system is provided through the three computer power switches located on the ground check panel. Electrical power to the spoiler system is provided through the computer power switches, numbers 1 and 2. When these switches are OFF, spoiler operation for roll control is not available. When the roll damper is off, the roll damper servo actuator is hydraulically driven to its neutral position. When hydraulic pressure is not available, the position of the roll damper servo is indeterminate. A stick position transducer, located in the roll feel assembly sends nonlinear command signals to the roll damper and to the spoiler actuators.

Spoiler Operation.

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 has extension pressure supplied by the utility hydraulic system and has lock down pressure supplied by the primary hydraulic system. The inboard pair of spoiler actuators receives extension pressure from the primary hydraulic system and lock down pressure from the utility hydraulic system. Lateral movement of the control stick causes the stick position transducers to generate electrical command signals which are sent through the feel and trim and the wing sweep sensor assembly to the spoiler actuators. There is no mechanical linkage between the stick and the spoiler. Both commanded spoilers extend to a maximum of 45 degrees at the stick force detent. The spoiler extension versus stick displacement is nonlinear.

Spoiler Lockout.

When the wing sweep angle is at 45 degrees, the electrical commands to the inboard spoiler actuators are switched out by the wing sweep sensor, causing the inboard spoilers to retract and lock down. At 47 degrees wing sweep, primary hydraulic pressure is removed from all spoiler actuators, and the electrical command signal to the outboard spoilers is switched out by the wing sweep sensor, causing them to retract and lock down. When the wing sweep reaches 49 degrees, the utility hydraulic pressure is removed from all spoiler actuators.

Spoiler Monitor.

When the wings are forward of 45 degrees, the spoiler monitor will lock the inboard pair or the outboard spoiler pair down should both left and right wing spoilers extend above 15 degrees due to a 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. Extension of spoilers on the opposite wing will cause the monitor, through a voting process, to cut off hydraulic pressure to the malfunctioning spoiler and its mate. This action will retract and lock the pair of spoilers in the down position and cause the spoiler caution lamp to light and result in reduced roll power. 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 due to the limited roll authority of the autopilot. Roll autopilot does not move the control stick, and the pilot's 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.

Stability Augmentation.

Roll stability augmentation is provided by redundant roll rate gyros and electronic computers used in conjunction with a redundant electrohydraulic roll damper servo to provide aircraft roll damping signals. (See figure 1-22.) The roll damper servo also responds to stick command signals from a nonlinear stick position transducer located in the control linkage. This same transducer supplies separate electrical commands to the inboard and outboard spoiler actuators.

Command Augmentation.

The effectiveness of any control surface varies with the flight conditions. At low speed and high altitude several degrees of differential horizontal stabilizer may be required to achieve a given roll rate; while at high speed and low altitude, it may take less than a degree. Also, the roll rate may be significantly increased when spoilers are operable. With the roll damper off,

Roll Channel Electrical Schematic (Typical)

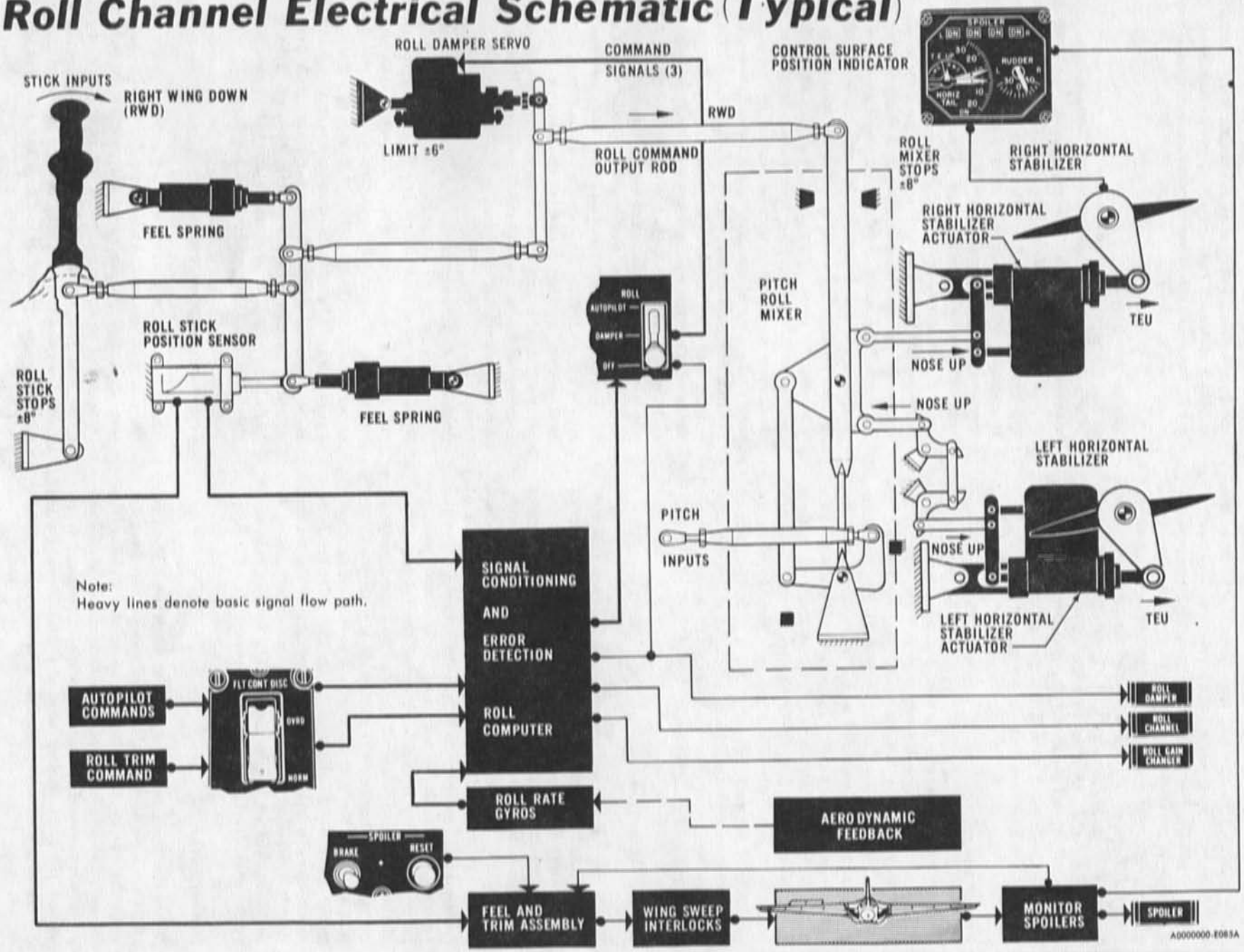


Figure 1-22.

horizontal stabilizer displacement is proportional to stick displacement. With the roll damper on, the stability augmentation input causes roll damper commands to oppose aircraft roll rate. However, when the control stick is displaced, the roll damper also receives an input command signal from the stick position transducer through a lag circuit. This signal represents the commanded aircraft response and reduces the roll damper opposition to pilot initiated maneuvers and augments the pilot's stick input. The steady state roll damper displacement will be proportional to the difference between the commanded response and the actual aircraft response and is directly proportional to the roll adaptive gain. The horizontal stabilizer surface displacement due to control stick inputs will then vary with flight conditions so that variations in the resulting aircraft response will be minimized.

Roll Commands.

The control stick transducer output reaches the maximum at the stick force detent and represents a roll rate command of 160 degrees per second. The roll damper authority is ±6 degrees of differential horizontal stabilizer deflection, i.e., for a left roll the left surface displaces 6 degrees up and the right surface displaces 6 degrees down; the opposite occurs for a right roll. The actual damper deflection will depend on the commanded roll rate, the actual roll rate, and the roll adaptive gain. Refer to "Roll Gain Control," this section. If the stick deflection exceeds the detent, the total command (mechanical plus damper) may exceed the roll command limit. If this occurs, the excess roll command from the damper will cause stick talkback which may appear as pitch or roll stick movements.

Damper Off Operation.

When the roll damper is OFF, full stick deflection requires a force of 31 pounds. This will command the maximum of ±8 degrees of differential horizontal stabilizer deflection. When the roll damper is turned off, horizontal stabilizer control reverts to the direct mechanical linkage command to the horizontal stabilizer actuators.

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. These relays supply 26 volts ac to the roll trim integrator which is a motor driving an electrical transducer. This electrical transducer supplies a roll rate command signal to the roll computer which causes the roll damper servo to position the horizontal stabilizer. 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. Roll trim is controlled by trim buttons located on each control stick. Approximately eight seconds is required to insert the maximum roll trim command. Roll trim is inoperative when the flight control disconnect switch is placed to OVRD or the roll damper switch is positioned to OFF. Under these conditions any previous roll trim inputs will be removed. The pilot can either hold stick force, or hold the wings level with yaw trim and accept the accompanying side slip when roll trim is not available.

Roll Gain Control.

The gain of the command signal sent to the roll damper servo is automatically varied as flight conditions change. This is accomplished by continuously monitoring the roll rate gyro signals to determine if the gain should be either increased or decreased. Since the system modifies its gain as a function of its own performance, it is called a self-adaptive gain system. In general the required gain varies inversely with dynamic pressure. A gain increase will compensate for reduced horizontal stabilizer effectiveness. If the gain is too low, the aircraft will appear sluggish to lateral stick commands. If the gain becomes too high, a small amplitude roll oscillation may exist for a few cycles until the gain control circuit, which senses this oscillatory condition, reduces the gain to the proper value. The frequency of this oscillation (the adaptive frequency) will be between 1.4 to 3.0 cycles per second. The gain changer is designed to either increase or decrease the gain for this range of frequencies depending on the amplitude and persistence. The gain will be increased for frequencies of less than 1.4 cps. Rapidly changing flight conditions can result in the computed gain lagging the optimum gain for a short period. Aircraft motion due to turbulence or aircraft vibration, such as experienced with speed brake operation, will cause the adaptive gains to decrease. When the roll damper switch is turned OFF, the roll gain is driven to its minimum value of 20 percent, and the roll damper servo centers. If the gain becomes high enough, due to a malfunction, to cause the adaptive frequency to persist, resulting in a small amplitude roll oscillation, the gain can be reset to its minimum value by momentarily cycling the roll damper switch OFF and then back to the DAMPER position. This should stop the oscillation. When the damper is turned on, the gains may require up to 2 minutes to increase to the optimum value. During this time the aircraft response and damping may be noticeably reduced. When in the takeoff and land configuration, the roll gain is locked at maximum.

Roll Channel and Roll Damper Redundancy.

The stick position transducer, roll rate gyros, electronic computers, gain control circuits, and the roll damper servo are redundant to the same extent as described under "Pitch Channel and Pitch Damper Redundancy," this section. An electronic malfunction causes

either the roll channel or roll gain changer caution lamps to light. A damper malfunction will cause the roll damper caution lamp to light. The roll damper servo actuator is identical to the pitch and yaw damper servo actuator. Refer to "Spoiler Operation," this section, for the redundancy features incorporated into the spoiler system.

YAW CHANNEL.

Rudder pedal movement is mechanically transmitted to the rudder control valve. This control valve controls the flow of hydraulic fluid from both the primary and utility hydraulic systems to the rudder actuator. Figure 1-23 is a simplified yaw channel mechanical and electrical schematic. A pedal shaker is attached to the pilot's left rudder pedal to provide stall warning. Refer to "Artificial Stall Warning System," this section. The yaw variable feel actuator provides two rudder authorities: full pedal travel of approximately 2.5 inches and ±30 degrees of rudder deflection or limited authority of approximately 1-inch pedal travel and ± 11.25 degrees of rudder deflection. The available authority depends on the slat position, the control system switch position, and the rudder authority switch position. A rudder authority caution lamp will light when available authority does not agree with the authority programmed by the slat position and the control system switch position. Rudder pedal breakout force is 12 pounds. Regardless of rudder authority, 80 pounds is required for maximum pedal deflection. Directional damping is provided by the yaw damper servo. Lateral accelerometer and yaw rate sensors are used as inputs to the fixed gain yaw damper system. Additional signals (side slip, and roll rate), are used in the takeoff and landing phase to improve turn coordination. (See figure 1-23). These signals are inputs to the adverse yaw compensator which supplies AYC signals to the yaw damper. The roll rate gain is increased for high angles of attack. Adverse yaw compensation (AYC) is activated when the slats are extended 70 percent or more when weight is not on the gear. Adverse yaw compensation engage or disengage transients may be experienced when the slats are either extended or retracted while maneuvering. Yaw transients experienced during slat extension or retraction from a wings level attitude is a discrepancy. The authority of the yaw damper is ± 15 degrees. The yaw damper is controlled from the yaw damper switch located on the autopilot/damper panel. When the yaw damper switch is turned to OFF, the yaw damper servo will hydraulically center, and the yaw damper caution lamp will light. When the yaw damper switch is returned to the DAMPER position, the yaw damper caution lamp will go out, and yaw damping will be immediately provided. The lateral accelerometers, yaw rate gyros, adverse yaw compensation circuits, yaw computer and yaw damper servo are redundant to the same extent as described under "Pitch Channel and Pitch Damper Redundancy," this section.

Yaw Channel Automatic Switching.

The yaw damper system uses fixed gains; however, certain switching configurations occur as shown below for a typical takeoff and land cycle.

1. Pre-taxi—Control system switch—NORM, slats are retracted, weight on gear, yaw damper switch—DAMPER.

Note

If the aircraft is not on level ground, the rudder may be displaced slightly due to the lateral accelerometer input into the yaw damper system.

   a. Adverse yaw compensation is inactive.
   b. Rudder authority is limited, and full nose wheel steering is not available.

2. Taxi—Control system switch—NORM, slats extended, weight on gear, yaw damper switch — DAMPER.
   a. Adverse yaw compensation is inactive.
   b. Rudder authority is full, and full nose wheel steering is available.

3. Lift off—Same as taxi, except weight is not on the gear.
   a. Adverse yaw compensation inputs are accepted by the yaw damper system to improve turn coordination (adverse yaw compensation inputs are side slip and adjusted roll rate inputs).
   b. Rudder authority is full.

4. Slat retraction—Gear up, yaw damper switch — DAMPER, control system switch—NORM.
   a. Adverse yaw compensation signals are switched out when slat extension is less than 70 percent.
   b. Rudder authority becomes limited when slat extension is less than 2 percent.

Adverse Yaw Compensation (AYC).

AYC is provided to improve turn coordination in the takeoff and landing configurations. This function is provided automatically whenever the slats are extended 70% or more. Turn coordination is improved by augmenting the directional stability with a sideslip to rudder feedback. This latter signal produces rudder in the direction of the roll and proportional to the roll rate. The roll rate feedback is gain adjusted as a function of angle-of-attack. AYC may also be obtained by placing the control system switch to the T.O. & LAND position. This should only be done below 300 KIAS or mach 0.45, whichever is less. The main gear squat switch deactivates AYC when weight is on the main gear. AYC can be switched out by placing the flight control disconnect switch in the OVRD position.

Yaw Channel Schematic (Typical)

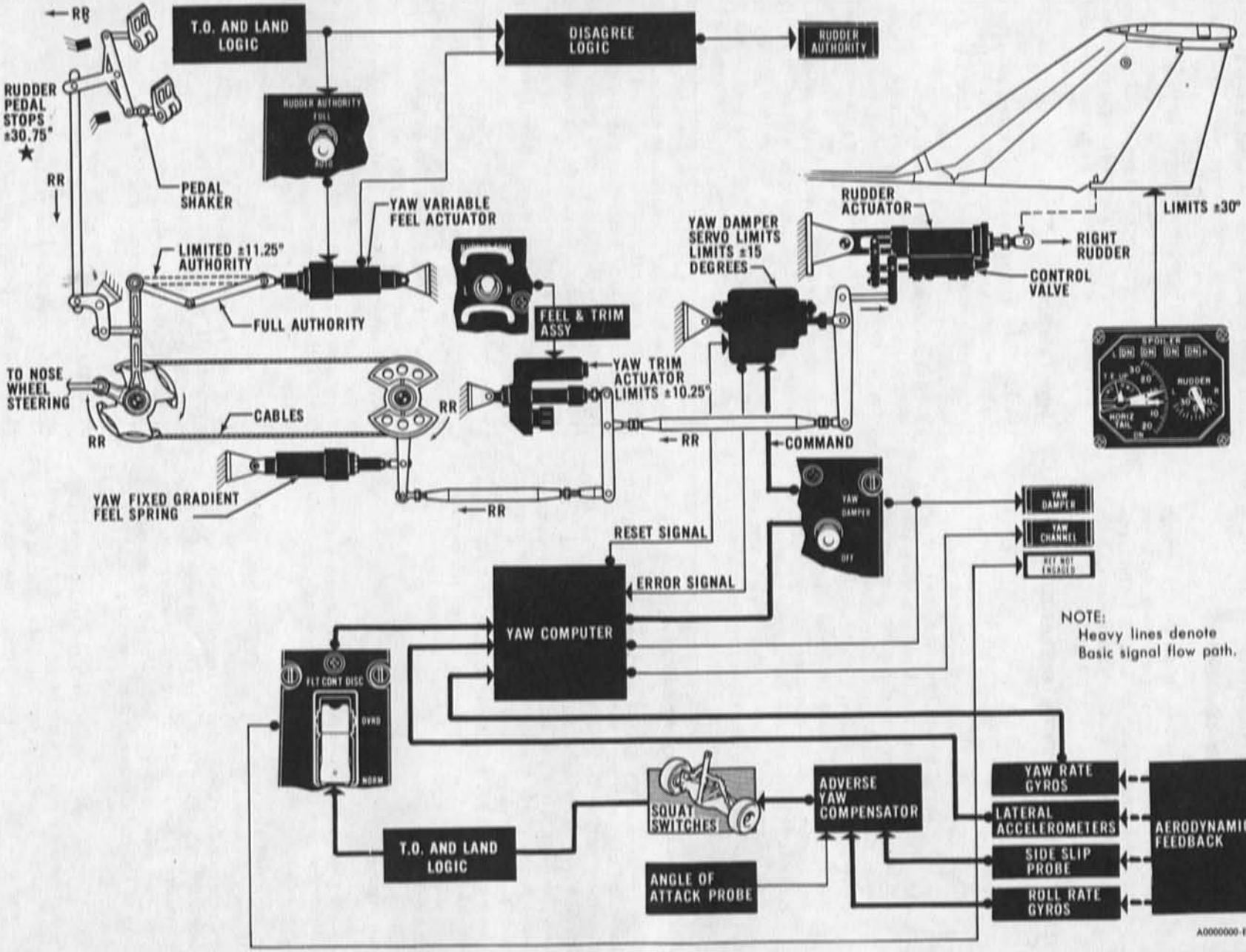


Figure 1-23.

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system switch in T.O. & LAND, can result in loss of control of the aircraft.

FLIGHT CONTROL SYSTEM OPERATIONAL CHARACTERISTICS.

Note

The flight control system computers operate on 115 volt ac power from the essential ac bus. The essential ac bus is, in turn, normally fed by the left generator. An interruption of power to the essential ac bus, such as loss or shutting down of the left generator or switching from left generator to external power, will cause a mild transient of the flight controls. This may also be accompanied by stick movements. Usually this will be felt as a mild airframe disturbance and should not be cause for concern.

Pitch Trimming During Maneuvers.

When parallel pitch trim is used to trim in a pitch command of one degree, the output of the pitch stick transducer will be only 30 percent of the value that would have been present had the stick been held in the same position. Thus the magnitude of the command signal to the pitch damper will be less when trim is used than when force is held for a given stick position. This means that for a constant altitude bank, the control stick must be trimmed to a further aft position than would be required if force is used to hold the same maneuver.

Roll Inputs.

During ground operation lateral stick inputs with the roll damper on will exhibit the following characteristics which are normal.

- The control stick cannot be held past the force detent. When the control stick is returned to neutral after a hard over deflection, the horizontal stabilizers will immediately change from 8 degrees of roll to 6 degrees, and after a small delay it will return to zero roll. This delay is normal and is caused by the saturation of the roll damper system from the stick transducer.
- Rapid lateral stick motions with the roll damper on may result in a small bump being felt in the control stick and a momentary reduction in the affected spoiler position. This effect will be more pronounced at low engine rpm and is caused by rate limiting of the system.
- When roll trim is applied, the roll damper responds and moves the horizontal stabilizers. If the trim command switch is held several seconds, it will saturate the roll damper. When roll trim is then driven in the opposite direction, by either of the stick trim switches or by depressing takeoff trim, the stabilizers may take several seconds to begin to respond. This delay is normal, and represents the time required to bring the trim input below the roll damper saturation limits.

Stick Talk Back.

A condition known as stick talk back may be experienced whenever the pitch/roll mixer output to both or either horizontal stabilizer actuators is unable to respond to or lags the pitch/roll mixer input from damper servo or trim commands. Refer to figure 1-19. When the pitch and roll dampers are on, the stick plus damper command may exceed the mixer limit as set by the mixer linkage stops. The damper response may also lag the stick command for large/rapid control stick inputs. Once the mixer authority is reached, additional damper or trim input will result in feedback to the control stick. This can be felt during ground operation by making large/rapid nose down control stick inputs. Control stick talk back can also be experienced whenever the rate of pitch or roll command into the mixer assembly exceeds the rate at which the horizontal stabilizer actuators can respond. The output rods from the pitch/roll mixer assembly control the left- and right-hand horizontal stabilizer actuators (refer to figure 1-19). Each actuator has its own control valve and feedback bellcrank. A nose up command displaces the control valve input rod aft; this ports fluid from both primary and utility hydraulic systems to each actuator and drives the horizontal stabilizers. When the feedback bellcrank has been repositioned by the actuator to a point where the control valve input is again at a null, the horizontal stabilizer actuator will stop. The stroke of each control valve is limited within the valve housing. This limit may be reached whenever the rate demand of the horizontal stabilizer actuator is greater than the maximum capability of the actuator. Should the limit of the valve stroke be reached, the mixer output rod will momentarily stop and will not respond to further pitch or roll inputs until the commanded rate can be satisfied by the actuator. During this period, a momentary control stick pulse will be felt. This may be noticed during ground operations ("Surface Motion Check") with programmed step inputs to the pitch and roll dampers or with rapid lateral stick inputs. The horizontal stabilizer actuators are rate limited to 36 degrees per second.

PRE-TAXI CHECKS.

Control System Movement Check.

The control stick and rudder pedals are checked for freedom of movement while all dampers are off. During this check the pitch, roll, and yaw damper caution

lamps should be on. The pitch, roll, and yaw channel lamps should not come on during this check. The mechanical linkage is checked for freedom and authority. If takeoff trim is first set, full stick deflection in pitch should cause a horizontal stabilizer indication of 10 degrees trailing edge down and 25 degrees trailing edge up. Roll stick displacement should cause a difference between left and right stabilizers of 4 degrees at detent and 16 degrees at full lateral stick displacement. Rudder control will be either full or limited, depending on the slat position and rudder authority switch position. Fore and aft motion of the control stick with the pitch damper on will result in stick talk back, which is normal. Lateral stick motion with the roll damper on may result in stick talk back, dependent upon the rate of application. The control stick cannot be held past the detent for this configuration. If lateral command inputs are made while maintaining large pitch inputs on the stick, stick talk back can be expected in the pitch direction.

Stability Augmentation Test Operation.

The purpose of the surface motion check is to ensure that all three damper systems will respond to three equal input signals by operating the appropriate damper servo without any malfunction. The purpose of the surface motion and lights check is to ensure that the error detecting system will detect the loss of one of the redundant branches in each channel and light the appropriate caution lamps. With all dampers on, all caution lamps out, flight control configuration normal, after takeoff trim is set, the stick is trimmed nose up momentarily to establish a nose up command to the pitch series trim actuator. Pitch series trim action is checked by verifying that both stabilizers drive trailing edge up until the trim authority limit is reached. After the stabilizers stop driving, the auto TF switch is placed to the AUTO TF position. This action causes the control stick to center, and the reference not engaged caution lamp and TF flyup off caution lamp to light. This switch action also effectively prevents the series trim actuator from driving while the surface motion test is in progress; however, stabilizer drifting may occur when this test is not in progress.

Surface Motion Check.

The master test button is depressed and the stability augmentation switch is held in the SURFACE MOTION position. This action causes all gyros in the pitch, roll, and yaw channels together with the accelerometers in the pitch and yaw channels to displace and send equal signals to the three computers. Unless a malfunction is present, all flight control system caution lamps will remain out; and the pitch, roll, and yaw dampers will displace as required in "After Engine Start (AC)," Section II. When the test switches are released, no caution lamps should light.

Surface Motion and Lights Check.

- When the master test switch is held and the stability augmentation switch is placed to the SURFACE MOTION and LIGHTS position, two out of three gyros in the pitch, roll, and yaw channels are displaced but no accelerometers are torqued. This sends only two rate signals to each computer. The absence of the third branch is detected and all three channel caution lamps light.
- In addition, one branch of each servo is also failed and the three damper caution lamps will light. Further, in the pitch and roll axes, one branch of the gain changer is failed to minimum and this error is detected, causing the pitch gain changer and roll gain changer caution lamps to light. Thus eight lamps light and all three dampers displace as required in "After Engine Start (AC)," Section II. The damper reset button is used to reset all lamps.

Stick Movement During Ground Checks.

During the stability augmentation checks a rapid displacement of the control stick will occur. This is a normal condition. The stick should not be manually constrained during these checks since this will impose unnecessary loads on the stick-damper mechanical linkage.

FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS.

Control Sticks.

The two control sticks (figure 1-24) are mechanically and electrically interconnected. Each stick grip contains a trim button, weapon release button, reference engage button, air refuel/nose wheel steering button, a gun trigger and an autopilot release lever. The control sticks also serve as a means of actuating the crew module bilge/flotation bag inflation pump. Refer to "Crew Module Escape System," this section.

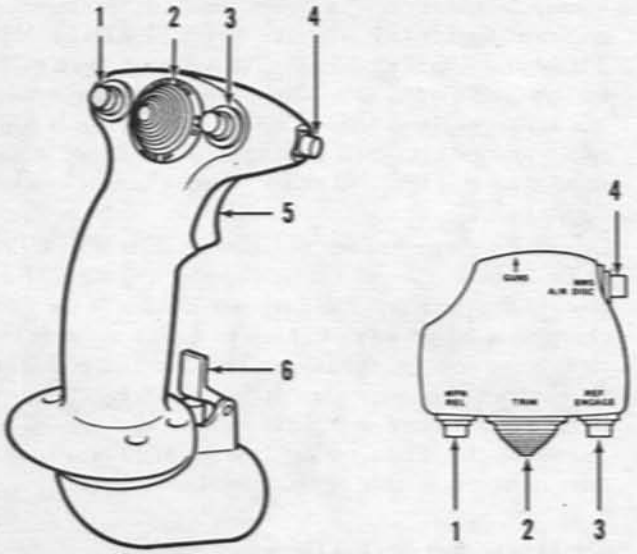
Rudder Pedals.

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 Buttons.

A trim button (2, figure 1-24), 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 horizontal stabilizer actuators to position the horizontal stabilizer surfaces symmetrically with trailing edge either

Control Sticks



1. Weapon Release Button.
2. Trim Button.
3. Reference Engage Button.
4. Nose Wheel Steering/ Air Refuel Button.
5. Gun Trigger.
6. Autopilot Release Lever.

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Figure 1-24.

up or down as selected. Refer to "Pitch Channel", this section, for various trim authorities. Moving the button to LWD or RWD causes the roll damper servo to position the horizontal stabilizer surfaces asymmetrically The left trim button can always override the right trim button control.

Auxiliary Pitch Trim Switch.

An auxiliary pitch trim switch (4, figure 1-25), with positions marked STICK, NOSE DN, NOSE UP, and OFF is located on the auxiliary flight control panel. When the switch is in the STICK position, pitch trim can be commanded by the trim buttons on the control sticks. When the auxiliary pitch trim switch is placed to the center (OFF) position auxiliary pitch trim system is armed. When the switch is placed to the NOSE UP or NOSE DN position, pitch trim is provided as outlined in figure 1-20.

Rudder Trim Switches.

A rudder trim switch (3, figure 1-25), located on the auxiliary flight control panel, is provided for rudder trim control. The switch has two positions marked L and R and is spring-loaded to an unmarked OFF position. Holding the switch to the L or R position 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 10.25 degrees is obtained.

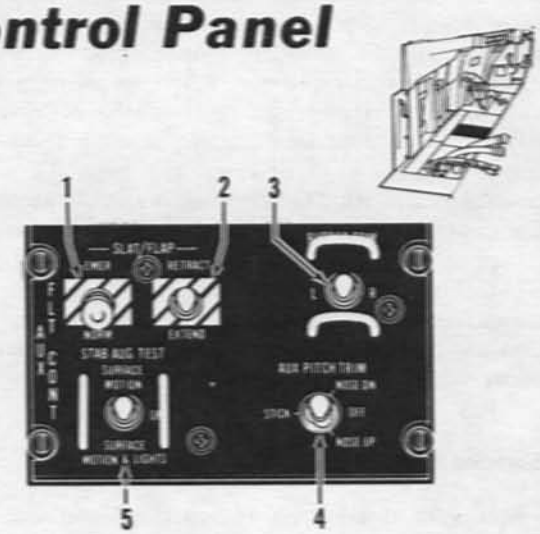
Takeoff Trim Button.

The takeoff trim button (29, figure 1-5) is located on the left main instrument panel. When the button is depressed, the parallel pitch 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 drives the horizontal stabilizers to 3.8 degrees trailing edge up. The button also functions during normal airborne operation.

Autopilot/Damper Switches.

Three switches, one each for the pitch, roll, and yaw channels, are located on the autopilot/damper panel. The pitch and roll damper switches (1, figure 1-27) are three-position switches marked AUTOPILOT, DAMPER and OFF. These switches are solenoid held in the AUTOPILOT position and are springloaded to the DAMPER position. The yaw damper switch (2, figure

Auxiliary Flight Control Panel



1. Flap and Slat Switch.
2. Emergency Flap and Slat Switch.
3. Rudder Trim Switch.
4. Auxiliary Pitch Trim Switch.
5. Stability Augmentation Test Switch.

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Figure 1-25.

Flight Control Switch Panel

1. Flight Control Disconnect Switch.
2. Rudder Authority Switch.
3. Damper Reset Button.
4. Flight Control Master Test Button.
5. Spoiler Test Switch.

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Figure 1-26.

1-27) is a two position switch marked DAMPER and OFF. The three switches are lever locked in the OFF position. Also the switch toggle of the pitch autopilot/ damper switch has been enlarged so that it can be readily identified by feel. Placing any of the switches to DAMPER turns the respective damper on. 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. Refer to "Autopilot System", this section.

Damper Reset Button.

The damper reset button (3, figure 1-26), located on the flight control switch panel, is a momentary push-button 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. Also 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. The button may also be used to reset the pitch and roll gain changer lamps.

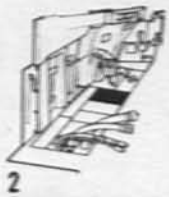
Rudder Authority Switch.

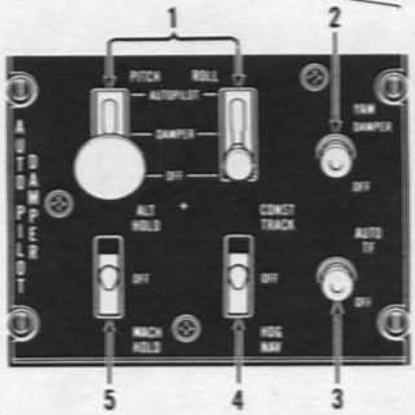
The rudder authority switch (2, figure 1-26), located on the flight control switch panel, has positions marked FULL and AUTO. When the rudder authority switch is in the AUTO position, rudder authority is automatically controlled by slat position. When the rudder authority switch is in the FULL position, full rudder authority is available.

Flight Control System Switch.

The flight control system switch (2, figure 1-15), located on the landing gear control panel, is a two position switch marked T.O & LAND and NORM. The T.O. & LAND position is used in the event of a system malfunction to override the slat position signal and place the control system in T.O. & LAND. The NORM position is used at all other times and the position of the slats determines the configuration of the flight control system. When the slats are extended beyond 70 percent, the following takeoff and landing functions are automatically provided: The yaw variable feel actuator moves to the full authority position, the TFR signals are locked out, the pitch and roll computer gains are locked, and the pitch series trim actu-

Autopilot/ Damper Panel





1. Pitch and Roll Autopilot Damper Switches.
2. Yaw Damper Switch.
3. Auto Terrain Following Switch.
4. Constant Track/Heading Nav Mode Selector Switch.
5. Altitude Hold/Mach Hold Selector Switch.

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Figure 1-27.

ator is locked at its present position. If weight is not on the gear, AYC is activated. When the slats are retracted, the following inflight (clean configuration) functions are provided: the pitch damper can respond to TFR signals (if present), the pitch and roll gains are automatically determined by the flight control computers as flight conditions change, the pitch series trim actuator is unlocked and operates in the null mode, and the yaw variable feel actuator moves to the limited authority position. A warning system is provided to indicate that the control system is not in the takeoff and landing configuration while airborne when the landing gear handle is in the DN position. In this event, both the pitch gain changer and roll gain changer lamps will light. Extension of the slats, or placing the control system switch to T.O. & LAND will place the control system in the takeoff and land configuration. The lamps will go out unless a malfunction is present or unless the flight control disconnect switch is in OVRD. Failure of the two lamps to go out is caused by the override action of the flight control disconnect switch on the adverse yaw compensation and pedal shaker systems. Use of the control system switch in this case will not cause the lamps to go out even though all other switching has taken place.

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system in the takeoff and land configuration can result in loss of control of the aircraft.

Flight Control Disconnect Switch.

The flight control disconnect switch (1, figure 1-26), located on the flight control switch panel, has two positions marked NORM and OVRD (override). A guard covers the switch in the NORM position to prevent inadvertent actuation. Placing the switch to OVRD removes the following:

1. Pitch and roll autopilot commands
2. Roll trim commands
3. Pitch damper trim inputs
4. TFR climb/dive commands
5. Adverse yaw compensation, and pedal shaker commands.

Also, the reference not engaged caution lamp will light when the switch is placed to OVRD. On aircraft modified by T.O. 1F-111-891, placing the flight control disconnect switch in the OVRD position will cause the stall warning lamp and audible stall warning signal to be activated any time the true wing angle of attack exceeds 14 degrees, regardless of wing sweep position. Functions not affected by the flight control disconnect switch are:

1. Stability augmentation
2. Series trim while the auxiliary pitch trim is in STICK
3. TFR fly-up manuever.

Flight Control Master Test Button.

The flight control master test button (4, figure 1-26), located on the flight control switch panel, provides power to the following:

1. Flight control test switches and buttons on the ground check panel,
2. CADC test switch on the ground check panel,
3. Stability augmentation test switch on the auxiliary flight control panel, and the
4. Spoiler test switch on the flight control switch panel.

Depressing the button applies power to the test switches and buttons. In addition, series trim is unlocked when the button is depressed. When the button is released, these switches and buttons are inoperable.

Stability Augmentation Test Switch.

The stability augmentation test switch (5, figure 1-25), located on the auxiliary flight control panel, is a three position switch marked SURFACE MOTION, SURFACE MOTION & LIGHTS with an unmarked center OFF position. With the flight control system in the normal configuration this switch, when used in conjunction with the flight control master test button, provides a means of ground checking the stability augmentation system.

Accelerometer Test Button.

The accelerometer test button (12, figure 1-29) is located on the ground check panel. When depressed, in conjunction with the flight control master test button, all of the accelerometers in the pitch and yaw axes are torqued causing the rudder and horizontal stabilizers to displace. During this check, the rudder moves left and the horizontal stabilizers move trailing edge up due to the action of the yaw and pitch dampers. The yaw and pitch damper and channel caution lamps (4) should not light for this check.

Spoiler Reset Button.

The spoiler reset button (7, figure 1-18), located on the crew module left sidewall, is a momentary pushbutton 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.

Spoiler Test Switch.

The spoiler test switch (5, figure 1-26), located on the flight control switch panel, is a three position switch marked OUTBD, OFF and INBD. The switch is spring-loaded to OFF. The switch is used in conjunction with the flight control master test button to ground check the operation of the spoilers. With the switch in OUTBD, depressing the master test button will cause the outboard pair of spoilers to extend momentarily and then lock down and the spoiler caution lamp will light. Depressing the spoiler reset button will return the spoilers to operation and the spoiler caution lamp will go out. The INBD position of the switch is used to make the same check of the inboard spoilers. If the spoiler switch is moved from OUTBD to INBD (or vice versa) before the reset button is depressed the first pair of spoilers locked down will be returned to operation; however, the caution lamp will remain on. This will invalidate the spoiler check.

Computer Power Switches.

The computer power switches (16, figure 1-29) labeled NO 1, NO 2, and NO 3 are located on the ground check panel. Each power switch controls the power to one branch of the pitch, roll and yaw computers and to other electrical flight control components. AC electrical power is supplied through the three computer power switches from the essential bus. When the switches are ON, and the electrical power is on the aircraft, 400 cycle electrical power is available to the entire flight control system. When these switches are OFF the various trim functions, dampers, and spoilers can no longer be operated and only the basic mechanical hydraulic system is available to operate the flight control surfaces. The switches must be in the ON position in order for the door on the ground check panel to be closed.

Damper Servo Button.

The damper servo button (15, figure 1-29), labeled DMPR SERVO, is located on the ground check panel. 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. Electrical command signals from each computer cause each damper servo to vote hydraulically. This causes the pitch, roll, and yaw damper and channel caution lamps (6) to light.

Rate Gyro Test Buttons.

The rate gyro test buttons (CHAN A, CHAN B, and CHAN C) (14, figure 1-29) are located on the ground check panel. 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. Depressing the CHAN A button causes the "A" gyros to be torqued in the pitch, roll, and yaw channels. Depressing the CHAN B and CHAN C buttons causes their respective gyros to be torqued. When a single CHAN button and the master test button are depressed, all three CHANNEL lamps will light. The control surface motion, if any, will be less than 2 degrees.

Control Surface Position Indicator.

The control surface position indicator (21, figure 1-5), located on the left main instrument panel, is composed of three separate sets of indicators which provide indications of the positions of the spoilers, rudder and horizontal tail (horizontal stabilizer). The position of the spoilers is indicated on four flip-flop type indicators, two for the left and two for the right spoilers. When the spoilers are retracted the letters DN appear in each indicator. As the spoilers extend the indicators become blank. Rudder position is provided by a pointer on a scale, 30 degrees (L) left or (R) right of zero. The scale is graduated in 5 degree increments. The position of the horizontal stabilizers is indicated by two pointers, marked L and R, on a scale, 30 degrees up and 20 degrees down and is graduated in 2 degree increments. An index mark mounted on the axis of the left pointer provides indications of left or right wing down (LWD or RWD) against a scale mounted on the axis of the right pointer. In this manner asymmetric stabilizer position indications also provide left or right wing down indications.

Takeoff Trim Indicator Lamp.

A takeoff trim indicator lamp (28, 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 auxiliary pitch trim integrator is zeroed. When the takeoff trim button is depressed and all trims 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 caution lamps, one each for the pitch, roll, and yaw channels, are located on the main caution lamp panel (figure 1-37). Lighting of any one of the lamps indicates that a malfunction has been sensed in the computer of the respective pitch, roll 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; however, system redundancy has been lost. A power supply failure in the yaw computer will cause

a nonresettable yaw channel light. Such a failure can cause loss of response to the TF climb/dive signals, loss of the TF fly up capability, loss of the reference not engaged and TF fly up off caution lamps and loss of pitch or roll autopilot capability. For these reasons TF operation is not recommended and autopilot operation should be monitored closely if a non-resettable yaw channel lamp is lighted.

Roll, Pitch, and Yaw Damper Caution Lamps.

Three caution lamps, one each for the pitch, roll, and yaw dampers, are located on the main caution lamp panel (figure 1-37). 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; however, system redundancy has been lost.

Spoiler Caution Lamp.

The spoiler caution lamp, located on the main caution lamp panel (figure 1-37), 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. The lamp is also used in conjunction with the spoiler test switch when ground checking spoiler operation. Refer to "Spoiler Test Switch," this section.

Roll and Pitch Gain Changer Caution Lamps.

Two gain changer caution lamps, one each for the roll and pitch gain changer, are located on the main caution lamp panel (figure 1-37). 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; however, system redundancy is lost. Lighting of both lamps simultaneously will normally indicate a disagreement between the position of the control system switch, the position of the slats and the configuration of the flight control system. This will occur when the landing gear handle is in the DN position and the flight control system is not in the takeoff and land configuration or when the slats are retracted and the flight control system is in the takeoff and land configuration.

Rudder Authority Caution Lamp.

A rudder authority caution lamp (figure 1-37) is located on the main caution lamp panel. Lighting of the lamp indicates the rudder authority actuator is not in the position commanded by slat position and/or the control system switch. With the control system switch in the NORM position the lamp will light when the slats are extended if the rudder does not switch to full authority or when the slats are retracted if the rudder does not switch to limited authority.

AUTOPILOT SYSTEM.

The autopilot system consists of electronic circuitry that, in conjunction with the primary flight control system, controls the aircraft during the various modes of autopilot flight. The autopilot system receives input signals from other systems (see figure 1-28) and computes command signals to the pitch and roll dampers to control the aircraft. In addition, the autopilot computes command signals to the series pitch trim actuator and the roll trim integrator units. The autopilot modes are pitch attitude stabilization, roll attitude stabilization, mach hold, altitude hold, heading nav, and constant track. The aircraft may be manually maneuvered at any time by use of control stick steering. Autopilot commands do not cause stick movement.

INPUT FAILURES.

The autopilot system is essentially a nonredundant system. In addition, the signals which are supplied to the autopilot are nonredundant. However, the central air data computer has a failure indication capability which will light the CADS caution lamp should a malfunction occur in the central air data computer. In like manner, the bomb nav system has a failure malfunction detection system which will light the primary attitude/heading caution lamp when certain failures occur. Portions of the autopilot flight control system are redundant and will operate under single failure conditions, and in addition will give a warning lamp indication of the failure.

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 light force in the forward or aft direction, to the top of the control stick. This mode is activated in the roll channel by applying a light force laterally to the control stick. When force is applied in either or both channels, the reference or references are disengaged. The reference not engaged lamp will light, and the pilot can maneuver the aircraft to a new reference. When the force to the control stick is removed attitude stabilization will automatically reengage in the affected channel or channels provided the attitude limits are not exceeded. The reference not

Autopilot-Subsystems Tie-In

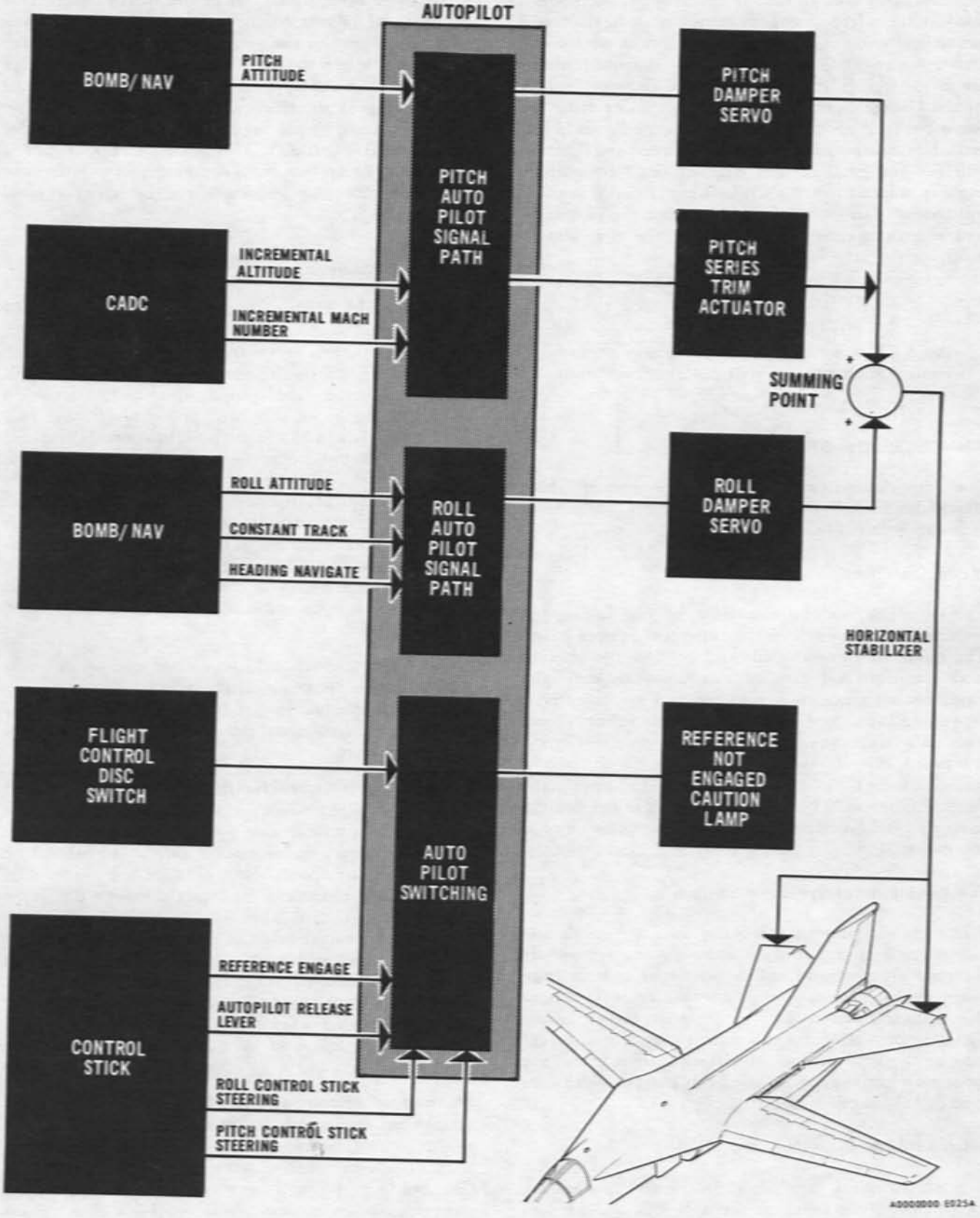


Figure 1-28.

engaged lamp will go out. If an autopilot submode is selected, the reference engage button must be depressed momentarily to engage submodes and turn off the reference not engaged lamp. The nominal attitude limits are ±30 degrees in pitch and ±60 degrees in roll. Actual engage limits will usually be lower than the limits outlined above. Should the engage limits be exceeded in one or both channels, attitude stabilization will not reengage in that channel until its attitude angle is reduced to less than its limit. In addition, the roll channel cannot be engaged if either the pitch attitude is greater than ±30 degrees or the yaw damper is off.

Note

Auto TF is not a mode of the autopilot and control stick steering will not disengage auto TF.

PRINCIPLES OF OPERATION.

The autopilot system is a pilot relief system which allows the aircraft to automatically carry out certain manually selected modes of operation.

Mode Selection.

The autopilot modes are selected by positioning the respective mode switch on the autopilot/damper panel. The mode switches are solenoid held to the selected mode position and may be turned off by manually repositioning the switches to off or by momentarily depressing the autopilot release lever on either control stick. If a mode other than basic attitude stabilization is selected, the reference engage button on either control stick must be momentarily depressed to engage the selected mode(s). The reference not engaged caution lamp will light whenever a selected autopilot mode is not controlling.

Pre-Engagement Synchronization.

When the autopilot is not being used, the pitch and roll attitude signals from the bomb nav system are continuously synchronized in the flight control computer so that at the time of engagement of pitch or roll attitude stabilization, the respective synchronized signal is zero. If for some reason this signal is not zero, at the time attitude stabilization is selected, the mode will not engage. Also, if the attitude limits are exceeded the mode will not engage.

Pitch Attitude Stabilization.

If pitch autopilot (attitude stabilization) is engaged, the pitch attitude synchronizer will lock and any deviation in pitch attitude about the engage attitude will result in an error signal out of the synchronizer. This error signal is sent to the pitch flight control computer and hence to the pitch damper servo. The error signal is also sent to the feel and trim assembly where it is amplified and applied to the series trim actuator which serves as an error integrator. Pitch stabilization may be engaged by placing the pitch autopilot/damper switch to AUTOPILOT. The mode will not engage if pitch attitude exceeds ±30 degrees. A new pitch attitude reference may be established by using control stick steering.

Pitch Sub-Modes.

Altitude Hold. If altitude hold is engaged, an engage signal is sent to the central air data computer which engages a synchro at the reference pressure altitude. Should the pressure altitude change, the synchro will develop an error signal proportional to the deviation from the engage or reference pressure altitude. This error signal is sent to the pitch flight control computer and to the feel and trim assembly. The error signal is summed in the pitch flight control computer with a washed out pitch attitude signal. The purpose of the washed out pitch attitude signal is to provide aircraft damping. This summed signal is sent to the pitch damper servo. The error signal which is sent to the feel and trim assembly is used to drive the series trim actuator as an error integrator which reduces standoff altitude errors.

Mach Hold. If mach hold is selected, the operation is essentially the same as altitude hold except that a delta mach synchro in the central air data computer develops the error signal rather than the delta pressure altitude synchro.

Altitude Hold Characteristics. If the altitude hold mode is engaged while at a stabilized altitude, the flight control system will hold the reference altitude within ±60 feet unless engine power is changed or wing sweep or speed brake changes are made. The altitude hold mode can be engaged up to 2000 fpm rate of climb or dive. The autopilot will cause the aircraft to appear to stabilize at an altitude slightly above or below the reference altitude. The aircraft will then slowly return to within ±60 feet of the reference. Changes in engine power, wing sweep or speed brake while this mode is engaged, will initially cause an altitude standoff, followed by a slow return toward the altitude reference.

Mach Hold Characteristics. If the mach hold mode is engaged while at a stabilized flight condition, the flight control system will hold the reference mach number within ±0.01 mach unless power is changed. Changes in power will initially cause a corresponding change in mach number, followed by a slow return toward the reference mach number.

Note

- Pitch autopilot operation must not be attempted while the slats are extended or if the control system switch is in T.O. & LAND. To do so may cause pitch transients when the autopilot is disengaged.
- Do not use the autopilot in the mach or altitude hold mode during operation in the transonic flight region between 0.90 and 1.10 mach.

The control stick will be centered and the pitch functions of the stick trim button will be inoperative when pitch autopilot/damper switch is placed to AUTOPILOT.

Roll Attitude Stabilization.

When the roll autopilot is engaged and the roll attitude is less than 3.5 (nominal) degrees, the roll synchronizer signal will cause the aircraft to roll to wings level; if the roll angle is greater than 3.5 (nominal) degrees the synchronizer will lock and the aircraft will maintain the roll attitude existing at the time of engagement. The error signal from the roll attitude synchronizer is supplied to the roll flight control computer and to the feel and trim assembly. In the feel and trim assembly this error signal is integrated and then sent to the roll flight control computer where it is summed with the error signal and used to position the roll damper servo.

Roll Sub-Modes.

Heading Navigate. If heading navigate is selected the aircraft will steer to the heading marker. For a discussion of the symbology and displays in each of the steering modes refer to "Instruments," this section, and figure 1-34, this section. In the yaw computer the error signal from the bomb nav system is summed with the absolute roll attitude. The summed signal is bank angle rate limited to ±8 degrees per second and the bank angle command is limited to a nominal ±30 degrees unless a greater bank angle was established at the time of engagement. The signal is then sent to the flight control roll computer and to the feel and trim assembly where it is integrated. The integrated signal from the feel and trim assembly is sent to the roll computer where it is summed with signal from the yaw computer. The resulting signal is then sent to the roll damper servo.

Constant Track. The constant track mode uses a signal from the bomb nav system to maintain the ground track existing at the time the mode was engaged.

Note

The roll autopilot is a pilot assist mode and should not be used for bomb runs for the following reasons:

- Roll rate capability is less than 8 degrees per second, which is too low for last minute heading corrections to be made.
- The autopilot gains and limits were not designed for optimum capture of heading changes.
- Bank angle overshoots will occur when heading corrections are attempted.

PROCEDURES.

Normal Procedures.

Roll Stabilization. Roll autopilot (roll attitude stabilization) may be engaged by selecting AUTOPILOT with the roll autopilot damper switch. The mode will not engage if pitch attitude exceeds ±30 degrees or roll attitude exceeds ±60 degrees. To establish a new reference, without disturbing the switch position, use control stick steering.

Roll Submodes. If a submode of the roll autopilot is desired, it may be selected by positioning the roll autopilot/damper switch to AUTOPILOT and by positioning the constant track/heading nav selector switch to either position. If constant track is selected, the aircraft should be flown until the desired ground track is reached, and then the autopilot engaged by depressing the reference engage button on either stick. The aircraft will then capture and hold the ground track existing at the time of engagement. If heading navigate is selected, depress the reference engage button on either control stick, and the aircraft will fly a computed course direct to destination except if MAN CRS is selected on the ISC. In this case the aircraft will turn to intercept the course selected on the HSI. When stabilized, the autopilot will hold the aircraft course within ±1 degree of the steering error received from the bomb nav system. Depressing the autopilot release lever on either control stick will return the constant track/heading nav selector switch to OFF, and the roll autopilot switch to DAMPER.

Pitch Stabilization. Pitch autopilot (attitude hold) may be engaged by placing the pitch autopilot/damper switch to AUTOPILOT. The mode will not engage if pitch attitude exceeds ±30 degrees. A new pitch attitude reference may be established by using control stick steering. The mode may be disengaged in the same manner as roll autopilot.

Engaging the Autopilot.

1. Flight instrument reference select switch—PRI.
2. ADI—Check for normal indications.

3. Roll and pitch autopilot/damper switches—AUTOPILOT.
4. Reference not engaged caution lamp—Out.
   Check that the reference not engaged caution lamp goes out with no force applied to the control stick.

Selecting the Autopilot Submodes. 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 or heading nav).

Note

The autopilot will stabilize on the desired reference mach or altitude more rapidly when the initial conditions of power and attitude are established prior to engaging the respective mode. When engaged in stabilized flight conditions the autopilot should hold altitude within ±60 feet or speed within ±0.01 mach.

The following procedures are for selecting each control mode after attitude stabilization has been engaged.

Selecting Mach Hold, Altitude Hold or Constant Track Modes. Manually maneuver the aircraft to the desired mach, altitude or heading.

1. Appropriate mode selector switch—Select desired mode.
2. Reference engage button—Depress.
   If it is desired to change the reference speed, attitude or heading, control stick steering may be used to manually fly to the new reference. The new reference may then be established by depressing the reference engage button.

Selecting the Heading Nav Mode.

1. Bomb nav destination counters—Set.
2. Fix mode TARGET or OFFSET selector button—Depress.
3. Instrument system coupler mode selector knob—NAV.
4. Constant track/heading nav mode selector switch—HDG NAV.
5. Reference engage button—Momentarily depress.
   If a steering error is present when the reference engage button is depressed this will result in a heading correction.

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 submode switches will move to OFF.

CONTROL AND INDICATORS.

Constant Track/Heading Nav Mode Selector Switch.

The constant track/heading nav mode selector switch (4, figure 1-27), located on the autopilot/damper panel, is a three position switch marked CONST TRACK, OFF and HDG NAV. The switch is solenoid held by 28 volt dc power to CONST TRACK or HDG NAV and is spring-loaded to OFF.

Altitude Hold/Mach Hold Selector Switch.

The altitude hold/mach hold selector switch (5, figure 1-27), located on the autopilot/damper panel, 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.

Reference Engage Buttons.

A reference engage button (3, figure 1-24), 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 momentarily depressed before the mode will engage. Either button may be used to engage the autopilot.

Autopilot Release Lever.

The autopilot release lever (6, figure 1-24), located at the base of the stick grip, permits either crew member 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 aircraft under pilot control. Refer to "Terrain Following Radar (AN/APQ-110)" in this section for functions of the lever related to that system.

Reference Not Engaged Caution Lamp.

The reference not engaged caution lamp (8, figure 1-5), located on the left main instrument panel, will light under the following conditions:

- The autopilot/damper switches are in the AUTOPILOT position and control stick steering is being used.
- Any autopilot mode (altitude hold, mach hold, constant track, or heading nav) is selected, and the reference engage button has not been depressed.

Note

The use of control stick steering in the axis of the autopilot mode that has been engaged will result in the mode being disengaged. The lamp will light and remain on until the reference engage button is depressed again.

- Either TFR channel mode selector knob is in the TF position and the auto TF switch is OFF.
- The auto TF switch is in AUTO TF and neither TFR channel mode selector knob is in the TF position.
- The flight control disconnect switch is placed to the OVRD (override) position.

The letters REF NOT ENGAGED are visible in the face of the lamp when lighted.

CENTRAL AIR DATA COMPUTER SYSTEM (CADC).

The aircraft 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: static pressure, pitot pressure, total temperature, and indicated angle-of-attack. When this data reaches the computer, it is transformed into electrical signal outputs. The central air data computer is equipped with a failure monitoring system which monitors the computing functions. Should a computing function fail, a 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 terrain following radar, and the flight director computer system repectively. The computer requires 115 volt ac and 28 volt dc power. Listed below are the various aircraft 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 wing angle-of-attack).
3. Maximum safe mach assembly (pressure altitude, mach number, true air temperature).
4. Flight control system (mach number and static pressure).
5. IFF (pressure altitude).
6. Engine fuel control unit (mach number).
7. Engine interstage bleed (mach number, and true wing angle-of-attack).
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 wing angle-of-attack, true airspeed, and angle-of-attack failure monitor).
13. Angle-of-attack indexer (true wing angle-of-attack).
14. Environmental control (indicated airspeed, true air temperature).
15. Flight director (pressure altitude and pressure altitude failure monitor).
16. Landing gear warning (indicated airspeed, pressure altitude).
17. Marker beacon (pressure altitude).
18. On aircraft after T.O. 1F-111-891, stall warning system (true wing angle-of-attack).

CADC POWER SWITCH.

The CADC power switch (11, figure 1-29), with positions POWER and OFF, is located on the ground check panel. When the switch is in the OFF position, no aircraft power is supplied to the CADC or the maximum safe mach assembly. Also the 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 ac power is supplied to the CADC and the maximum safe mach assembly.

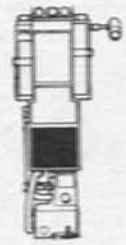
CADC TEST SWITCH.

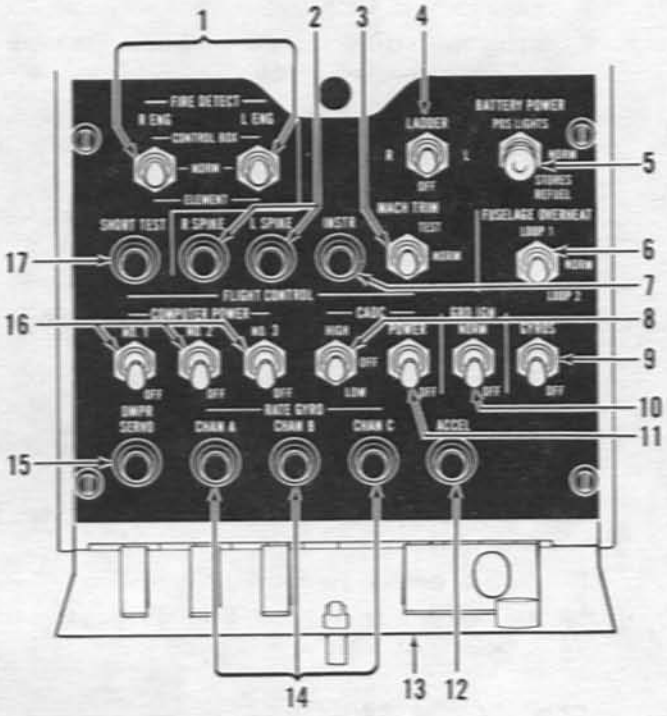
The CADC test switch (8, figure 1-29), with positions HIGH, OFF, and LOW, is located on the ground check panel. The switch is spring-loaded to the OFF position. The switch, when used in conjunction with the flight control master test button, activates a self test system in the CADC. The normal system 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 probe heater switch to ground check the total temperature probe heaters.

Note

When the CADC test switch is used, the bomb nav mode selector knob must be in OFF, HEAT or ALIGN. Otherwise, an alignment error will be injected and a re-alignment will be required.

Ground Check Panel





1. Engine Fire Detection System Ground Test Switches (2).
2. Spike Test Buttons (2).
3. Mach Trim Test Switch.
4. Entrance Ladder Switch (Deactivated).
5. Position Lights/Stores Refuel Battery Power Switch.
6. Fuselage Overheat Test Switch.
7. Instrument Test Button.
8. CADC Test Switch.
9. AFRS Power Switch.
10. Ground Ignition Cutoff Switch.
11. CADC Power Switch.
12. Accelerometer Test Button.
13. Ground Check Panel Access Door.
14. Rate Gyro Test Buttons (3).
15. Damper Servo Button.
16. Computer Power Switches (3).
17. Engine Fire Detection System Short Test Button.

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Figure 1-29.

CADC CAUTION LAMP.

The CADC caution lamp (figure 1-37), located on the main caution lamp panel, will light to indicate malfunctions in the central air data computer or a power supply failure within the maximum safe mach assembly. The lamp will also light when the CADC power switch is in the OFF position when power is on the aircraft. When lighted the letters CADS are visible.

MAXIMUM SAFE MACH ASSEMBLY.

The maximum safe mach assembly (MSMA) receives mach number, pressure altitude, and true free-stream air temperature signals from the central air data computer (CADC) and wing sweep position from the wing sweep sensor. The MSMA commands the maximum safe mach (MSM) bar on the airspeed-mach indicator (AMI) and the reduce speed warning lamp. The MSMA computes the maximum continuous safe mach (design speed) of the aircraft based upon pressure altitude, wing sweep, and temperature. The lower of the two computed limits: (1) design speed as a function of pressure altitude and wing sweep, or (2) design speed as a function of the aircraft skin temperature limit (418°F) is displayed. The MSMA also computes the actual aircraft mach number to the maximum safe mach, computed as a function of pressure altitude and wing sweep, and provides a signal to light the reduce speed warning lamp when the aircraft reaches this allowable design speed. The MSMA utilizes 115 volt ac power from the essential ac bus through the CADC power switch and 28 volt dc power from the essential dc bus. A power failure to the MSMA will cause the CADS caution lamp to light.

CAUTION

The MSMA computes maximum safe mach (design speed) based upon functions of pressure altitude and wing sweep or as a function of the aircraft skin temperature limit (418°F). Any flight speed or temperature restrictions which are more restrictive than the clean aircraft design values are not considered in the MSMA and consequently, are not displayed on the AMI. This is true for temporary or permanent limitations.

AUXILIARY FLIGHT REFERENCE SYSTEM (AFRS).

The auxiliary flight reference system (AFRS) provides standby or backup attitude and directional information. The system consists of a directional and vertical gyro platform, compass control panel, remote compass transmitter (flux valve), and a control amplifier. Changes in aircraft attitude are detected by the vertical gyro and electrically transmitted to the standby attitude indicator at all times and to the attitude director indicator (ADI) whenever the flight instrument reference select switch is in the AUX position or in event of malfunction of the bomb nav system. The directional gyro and compass transmitter provide heading information to the bearing-distance-heading indicator (BDHI) at all times and to the ADI and horizontal situation indicator (HSI) whenever the flight instrument reference select switch is in the AUX position or in event of malfunction of the bomb nav system. The vertical gyro is unlimited in roll but is limited to ±82 degrees in pitch. The directional gyro is attitude stabilized by the vertical gyro. The AFRS compass provides three modes of operation: SLAVED, DG (directional gyro), and COMP (compass). The slaved mode provides gyro stabilized magnetic heading from the remote compass transmitter. This mode is designed for use at latitudes up to 70 degrees. At higher latitudes the horizontal component of the earth's magnetic field becomes too weak to provide reliable heading information and the directional gyro mode should be used. In the directional gyro mode the remote compass transmitter is disconnected from the system and the directional gyro operates as a free gyro to provide directional reference. Free gyro drift of the directional gyro will not exceed ±1 degree per hour. In the directional gyro mode, apparent drift of the directional gyro due to earth's rotation is corrected. The compass mode provides magnetic heading directly from the remote compass transmitter without gyro stabilization. This mode of operation should only be used when the AFRS gyros are suspected to be unreliable. AFRS attitude unreliable and gyro fast erection is indicated by the auxiliary attitude (AUX ATT) caution lamp, the OFF flag on the standby attitude indicator, and the OFF flag on the ADI if it is receiving attitude information from the AFRS.

WARNING

Momentary power interruptions, such as electrical bus transfer, may cause the AFRS gyro to revert to automatic fast erection. If this occurs, gyro fast erection will be indicated as described above for the duration of the two-minute fast erection cycle.

The AFRS operates on 115 volt ac power from the ac essential bus and 28 volt dc power from the dc essential bus.

CONTROLS AND INDICATORS.

Flight Instrument Reference Select Switch.

The flight instrument reference select switch (2, figure 1-56), located on the miscellaneous switch panel, has two positions marked PRI and AUX. 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 and the attack radar which gets roll information only and will also light the primary attitude caution lamp. If the bomb nav system is operating normally, selecting AUX will cause the attack radar antenna to cage and fail the TFR.

Note

If there is a difference between primary and auxiliary headings, verify NC magnetic variation. If the magnetic variation is incorrect, the primary heading displayed to the pilot is in error. If the magnetic variation is correct, the auxiliary heading is in error. When the auxiliary heading is in error and it is selected, the TACAN magnetic bearing on the HSI/BDHI is correct but the relative bearing is in error. With an auxiliary heading error when primary heading is selected, the TACAN magnetic bearing and the relative bearing are incorrect. The CDI, bank steering bar, and autopilot steering corrections are usually valid with primary magnetic heading error on the ADI and HSI.

Auxiliary Flight Reference System Power Switch.

The auxiliary flight reference system power switch (9, figure 1-29), located on the ground check panel, has positions GYROS and OFF. Placing the switch to GYROS supplies power to the AFRS, the BDHI, the standby attitude indicator, and the turn and slip indicator on the ADI. Placing the switch to OFF de-energizes these components.

AFRS Gyro Fast Erect Button.

The auxiliary flight reference system gyro fast erect button (1, figure 1-56), located on the miscellaneous switch panel, provides a means for fast erection of the AFRS. The button is labeled ATT GYRO FAST ERECT. If re-erection of the AFRS gyro is required due to the gyro erecting to a false vertical or the pitch 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. Whenever the fast erect button is depressed, the displacement gyroscope erects at a rate of approximately twelve (12) degrees per minute.

Note

The button should be used only when the aircraft is in level flight to prevent inducing errors into the system. The off flag is visible when the button is depressed.

Compass Mode Selector Knob.

The compass mode selector knob (4, figure 1-30), 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 (compass), and DG (directional gyro). When the slaved mode is selected, gyro-stabilized magnetic heading from the remote compass transmitter is provided. In the directional gyro 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 compass 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 attitude 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 and will not return until the heading set knob is depressed and held to null the synchronization indicator.

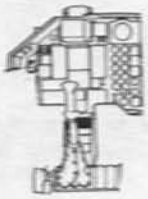
Latitude Correction Knob.

The latitude correction knob, located on the compass control panel (3, figure 1-30), 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 directional gyro mode and improves heading accuracy when operating in slaved mode for latitudes up to 65 degrees.

Heading Set Knob.

The heading set knob (5, figure 1-30), located on the compass control panel, provides a means of rapidly synchronizing the AFRS directional 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 directional gyro 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 directional gyro 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 compass mode, the system continuously tracks the remote compass transmitter and it is not necessary to use the knob.

Compass Control Panel



1. Synchronization Indicator.
2. Hemisphere Selector Switch.
3. Latitude Correction Knob.
4. Compass Mode Selector Knob.
5. Heading Set Knob.

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Figure 1-30.

Hemisphere Selector Switch.

The hemisphere selector switch (2, figure 1-30), 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 aircraft is operating to provide the proper polarity of the gyro drift correction for the earth's rotation.

Synchronization Indicator.

The synchronization indicator, located on the compass control panel (1, figure 1-30), indicates whether or not the AFRS gyro and remote compass are synchronized, when operating in the slaved mode. 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. The heading set knob must be depressed and held until the pointer is centered to synchronize the system. The indicator is de-activated when operating in the directional gyro or compass mode.

Auxiliary Attitude (AUX ATT) Caution Lamp.

The auxiliary attitude caution lamp (figure 1-37), located on the main caution lamp panel, will light if attitude information from the auxiliary flight reference system 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.

Note

If an electrical power interruption causes the AFRS gyro to revert to fast erection, or if the fast erect button is depressed, the auxiliary attitude caution lamp will light and the OFF flag on the standby attitude indicator will appear. In this condition, the aircraft must be maintained in unaccelerated straight and level flight during the fast erection period (normally 2 minutes) to prevent erection of the AFRS gyro to a false vertical.

AUXILIARY FLIGHT REFERENCE SYSTEM OPERATION.

Placing the auxiliary flight reference system power switch to GYROS applies power to the system. The gyro will initially erect at the fast erection rate of 12 degrees per minute minimum for the first two minutes of operation. After the initial erection cycle, the gyro erection rate is reduced to 5 ($\pm$1) degrees per minute. If subsequent re-erection of the gyro is required, manual fast erection may be accomplished by depressing the attitude gyro fast erect button. The gyro pitch erection rate is reduced to one-fourth of the normal erection rate whenever the aircraft fore or aft acceleration exceeds 0.065 "g." Since pitch erection is not completely removed, pitch errors may develop in the AFRS after prolonged aircraft longitudinal acceleration. Therefore, the vertical velocity indicator, altimeter, and angle-of-attack indicator should be crosschecked during aircraft accelerations to insure proper aircraft pitch attitude. The gyro roll erection rate is reduced to one-fourth of the normal erection rate whenever the aircraft bank angle exceeds 8.5 degrees.

Note

When operating on AFRS as the primary source of attitude information, roll into turns at roll rates greater than one degree per second and maintain at least 10 degrees bank angle in turns to insure that the gyro roll erection rate is reduced. If the roll erection rate is not reduced during turns, the gyro will erect to a false vertical and erroneous aircraft roll attitude will be displayed.

Whenever the ADI is receiving attitude information from the bomb nav system, AFRS roll attitude errors resulting from erroneous gyro roll erection will result in disagreement between roll indications on the ADI and standby attitude indicator. If both the ADI and standby attitude indicator are receiving AFRS attitude information, the turn rate pointer on the ADI and the HSI compass card should be monitored in order that erroneous roll indications may be detected. If the compass is operating in the slaved mode, the remote compass transmitter magnetic heading signals are disconnected whenever gyro roll or pitch erection is reduced to prevent the system from synchronizing to erroneous headings caused by an unlevel heading sensor. Disengagement of the remote compass transmitter is indicated by an inactive synchronization indicator on the compass control panel. Initial synchronization of gyro heading when operating in the slaved compass mode, or positioning of the heading indicators to a known heading when operating in the directional gyro mode, is rapidly accomplished by the heading set knob on the compass control panel. Subsequent resynchronization may be required if the pitch limits of the gyro are exceeded. In the slaved mode, the system will re-synchronize automatically at 1.5 ($\pm$0.5) degrees per minute if manual rapid synchronization is not accomplished.

PITOT-STATIC SYSTEM.

A single pitot-static system provides pitot and static pressures required for operation of standby instruments, the central air data computer (CADC), and the crew module q sensor. Connections of both pitot and static pressures are made at the CADC unit and 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. For pitot-static system instrument error and difference between primary and standby instruments, see figure 1-31. It will be noted that a relatively large difference exists between the primary and secondary instrument readings on this figure. This is because the standby instruments are provided with uncorrected data from the pitot-static system while the primary instruments are provided with data from the CADC which compensates for pitot-static system errors.

INSTRUMENTS.

The instruments consist of the total temperature indicator, true airspeed, standby instruments and the integrated flight instrument system.

TOTAL TEMPERATURE INDICATOR.

The total temperature indicator (9, figure 1-5), located on the left main instrument panel, provides indications of aerodynamic heating. The temperature sensing probe is equipped with a heating element for anti-icing. Refer to "Anti-Icing and Defog Systems," 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 degrees and a maximum temperature index mark at 214 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 and 214 degrees. The indicator functions in conjunction with the total temperature caution lamp and the reduce speed warning lamp to provide the following indications:

(1) When the critical temperature of 153 is reached the counter will start to drive down from 300 seconds toward zero and the total temperature caution lamp will light.

(2) The counter will continue to drive until it reaches zero or the temperature is reduced below 153 degrees.

(3) When the maximum temperature index is reached or when the counter drives to zero the reduce speed warning lamp will light and the total temperature caution lamp will go out.

(4) The counter will reverse and drive back to 300 seconds any time the temperature falls below 153 degrees.

(a) If the reduce speed warning lamp was on when the counter reversed it will go out.

(b) If the total temperature caution lamp was out when the counter reversed it will light and remain on until 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 or when the amplifier output signal varies from the temperature probe input signal by 10 to 12 degrees C. The indicator operates on 115 volt ac power from the essential ac bus.

Total Temperature Caution Lamp.

The total temperature caution lamp, located on the left main caution panel (figure 1-37), functions in conjunction with the total temperature indicator and reduce speed warning lamp to provide an indication that the aircraft is being operated in the critical temperature range between 153 and 214 degrees C. Refer to "Total Temperature Indicator," this section, for a description of lamp indications. When lighted, the words TOTAL TEMP are visible in the face of the lamp.

Reduce Speed Warning Lamp.

The reduce speed warning lamp (8, figure 1-5), located on the left main instrument panel, functions in conjunction with the total temperature indicator to indicate that the aircraft has flown for at least 300 seconds in the critical temperature range of from 153 to 214 degrees centigrade or that the maximum temperature index of 214 degrees has been reached or exceeded. Refer to "Total Temperature Indicator," this section, for a description of the lamp functions in conjunction with the indicator. When lighted the words REDUCE SPEED are visible in red on the face of the lamp. 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 1-32), 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 0-1750 knots. The indicator is operated by electrical signals from the CADC. The true airspeed indicator is not reliable when the CADS caution lamp is lighted.

Allowable Differences Between Primary and Standby Instruments

AIRSPEED DIFFERENCE TOLERANCES

DATE: 24 APRIL 1970

Airspeed— Knots	*Altitude — Feet*				
	Sea Level	*10,000*	*20,000*	*30,000*	*40,000*
200	+ 11 — 11	+ 12 — 10	+ 12 — 10	+ 12 — 10	+ 13 — 9
300	+ 17 — 15	+ 18 — 14	+ 19 — 13	+ 20 — 12	+ 19 — 13
400	+ 24 — 18	+ 25 — 17	+ 26 — 16	*	+ 23 — 19
500	+ 31 — 21	+ 33 — 19	*	+ 28 — 24	+ 26 — 26
600	+ 34 — 18	*	+ 28 — 24	+ 26 — 26	+ 26 — 26

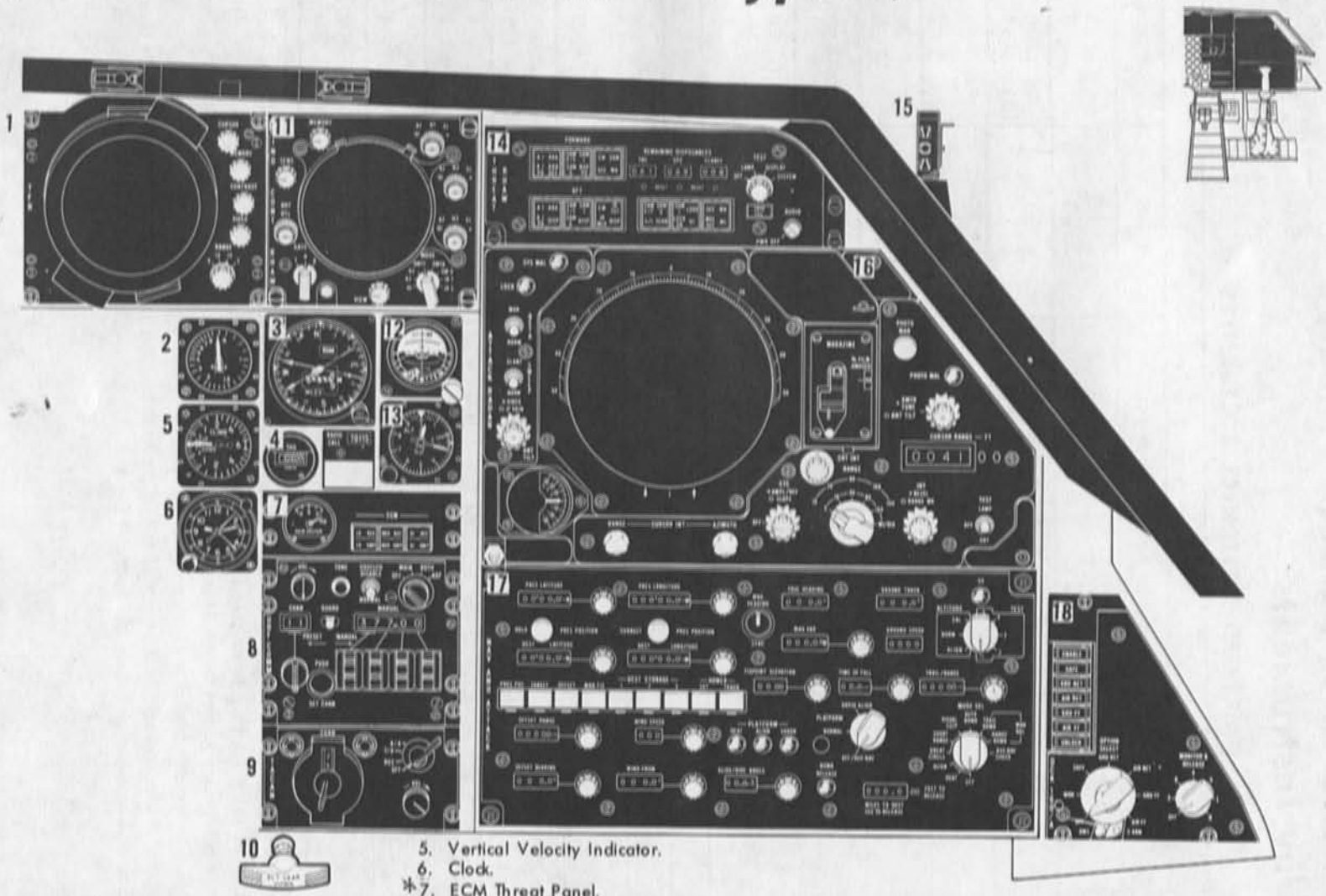
ALTITUDE DIFFERENCE TOLERANCES

Airspeed— Knots	*Altitude — Feet*				
	Sea Level	*10,000*	*20,000*	*30,000*	*40,000*
200	+ 105 — 95	+ 200 — 160	+ 310 — 240	+ 460 — 330	+ 730 — 450
300	+ 140 — 60	+ 255 — 105	+ 420 — 130	+ 685 — 105	+ 985 — 195
400	+ 220 + 20	+ 405 + 45	+ 690 + 140	*	+ 855 — 325
500	+ 405 + 205	+ 700 + 340	*	+ 655 — 135	+ 590 — 590
600	+ 690 + 490	*	+ 565 + 15	+ 395 — 395	+ 590 — 590

NOTES:
1. Enter tables at primary instrument readings to obtain upper and lower limits on differences.
2. Subtract primary reading from standby reading to obtain difference (may be negative).
3. Do not interpolate across heavy lines.
4. * Check not recommended at this condition.
5. Difference limits include indicator and CADC tolerances, and standby instrument position error.
6. Primary and standby altimeter set to 29.92 for the checks.

Figure 1-31.

Right Main Instrument Panel (Typical)



1. Terrain Following Radar Scope Panel. (See fig. 1-77) ★
2. Standby Airspeed Indicator.
3. Bearing-Distance-Heading Indicator.
4. True Airspeed Indicator.
5. Vertical Velocity Indicator.
6. Clock.
*7. ECM Threat Panel.
★ 8. UHF Radio Control Panel.
9. TACAN Control Panel. (See fig. 1-54)
10. Landing Gear Emergency Release Handle.
*11. Radar Homing and Warning Scope Panel.
12. Standby Attitude Indicator.
13. Standby Altimeter.
★*14. Radar Homing and Warning Threat Panel.
15. Angle-of-Attack Indexer.
16. Attack Radar Scope Panel. (See fig. 1-72)
★17. Bomb Nav Control Panel (See fig. 1-59)
18. Nuclear Weapons Control Panel. (See fig. 1-63)

* Refer to T.O. 1F-111E-1-2

A0000000-E018C

Figure 1-32.

STANDBY INSTRUMENTS.

The standby instruments include the airspeed indicator, altimeter, vertical velocity indicator, magnetic compass, attitude indicator and bearing distance heading indicator. These instruments provide back-up indications in the event of failure of the integrated flight instrument system. Position error must be applied to the airspeed and altimeter reading to obtain correct readings. Refer to Appendix I.

Airspeed Indicator.

The airspeed indicator (2, figure 1-32), 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 8.5 times 100 knots.

Altimeter.

The altimeter (13, figure 1-32), 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.

WARNING

Do not push in on the set knob when setting barometric pressure as disengagement of the gear train between the indicating pointers and the barometric scale may occur, resulting in erroneous altimeter readings.

Vertical Velocity Indicator.

The vertical velocity indicator (5, figure 1-32), 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 (17, 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 (12, figure 1-32), 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.

WARNING

The attitude warning flag will not appear with a slight electrical power reduction or 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.

Bearing-Distance-Heading Indicator.

The bearing-distance-heading indicator (3, figure 1-32), is located on the right main instrument panel. The instrument is a remote type heading indicator with a rotating compass card. UHF 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 aircraft 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.

Note

During TACAN operation with the ADF system off, the BDHI (number 1) and ADF (number 2) pointers both indicate TACAN bearing. During TACAN search operation, with the ADF system off, the two pointers may not track together due to differences in pointer slew speeds. However, the two pointers will be together whenever TACAN lock on is accomplished. When the pointers are together it appears that number 2 pointer is providing TACAN information.

The pointer designated as number 2 is also servo driven and displays bearing to a selected UHF transmitter when the ADF position is selected on the UHF radio control panel. The indicator receives heading information from the auxiliary flight reference system. 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 indicator 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
- Auxiliary flight reference system
- Instrument landing system
- Tactical air navigation system
- Terrain following radar
- Radar homing and warning system
- Bomb nav system
- Attack radar system
- Radar altimeter
- Lead computing optical sight system
- Dual bombing timer

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 aircraft. 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 ISC serves as a coupler between the FDC, the instruments and the various aircraft systems which supply information for presentation on the integrated flight instruments. The FDC accepts information from the various aircraft systems listed above, processes this information and returns it to the ISC and on to the instruments. The system incorporates self test features to check reliability and isolate malfunctions. The AMI and AVVI receive power from the left main ac bus. The other components of the system operate on 115 volt and 26 volt ac power from the essential ac bus and 28 volt dc power from the essential dc bus.

CADC Angle-of-Attack Correction.

The angle-of-attack indicating system is composed of a conical probe transmitter located on the left side of the aircraft fuselage, an angle-of-attack correction cam in the central air data computer (CADC), two angle-of-attack indexers and a vertical scale read-out on the airspeed mach indicator. In flight, the angle-of-attack probe generates an indicated angle-of-attack signal which is sent to the CADC and corrected to true angle-of-attack. This correction is necessary since the indicated angle-of-attack contains position errors due to aircraft configuration as well as speed. The position error correction made in the CADC is accomplished as a function of mach number only, even though the actual position error is also a function of flap and slat configuration. Since flap and slat information is not supplied to the CADC, the correction cam is mechanized such that between mach 0.45 and 0.30 the position error correction is changed linearly from the flaps and slats retracted value to the flap and slat extended value. Between these two mach numbers the flaps and slats would be extended and the takeoff and landing configuration fully selected by 0.3 mach. If the aircraft is decelerated below mach 0.45 without extending the flaps and slats or accelerated past 0.3 with the flaps and slats extended, the position error correction applied by the CADC will be in error and the angle-of-attack indicator will no longer read the true aircraft angle-of-attack. For the flaps and slats retracted case, the angle of attack indicator will read lower than true angle of attack below mach 0.45. The error will increase linearly from 0 error at mach 0.45 (approximately 300 KIAS at S.L.) to 1.7 degrees at 0.30 (approximately 200 KIAS at S.L.). Below mach 0.3 the error remains constant at 1.7 degrees. For the flaps and slats extended case, the angle-of-attack indicator will read higher than true angle-of-attack above mach 0.30. The error will increase linearly from 0 error at mach 0.30 to 1.7 degrees at 0.45. Above mach 0.45, the error remains constant at 1.7 degrees. Since the angle-of-attack indexers are commanded by the same signal from the CADC as the indicator, the on-speed lamp will be lighted when the tape reads 10 degrees even though the true angle-of-attack may not be 10 degrees.

Anti-icing is provided to the angle-of-attack probe. The heating elements receive power from the main ac bus and are controlled by the pitot/probe heat switch and the squat switch on the main gear. In flight when a primary heater failure occurs, a failure monitor system automatically energizes the secondary heater and provides the crew member with a failure indication on the α/β probe heat caution lamp. Positioning the pitot/probe heat switch to the OFF/SEC position and observing that the caution lamp goes out will verify that the secondary heater is operating.

Note

Since the angle-of-attack indicator and indexers are commanded by the CADC, these instruments will be inoperative if the CADC is not operating.

Angle-of-Attack Indexer. An angle-of-attack indexer (7, figure 1-5, and 15, figure 1-32) is located on each side of the glare shield. Each indexer consists of 3 color coded symbols arranged vertically. The red low speed (top V-shaped) symbol lights when the angle-of-attack exceeds 10.5 degrees. The green on-speed (center donut-shaped) symbol lights between 9.0 and 11.0 degrees. The amber high speed (bottom inverted V-shaped) symbol will light when the angle-of-attack is less than 9.5 degrees. The indexer lamps function only when the landing gear is in the down position. The indexer lamps may be tested by depressing the malfunction and indicator lamp test button on the lighting control panel. The test should be performed in the bright 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. On aircraft modified by T.O. 1F-111-891, the angle-of-attack indexers receive 28-volt dc power from the essential dc bus.

Airspeed Mach Indicator.

The airspeed mach indicator (AMI) (figure 1-33), located on the left main instrument panel, provides remote reading vertical presentations of true wing angle-of-attack, "g" acceleration, mach airspeed and maximum safe mach 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 (CADC), maximum safe mach assembly and remote accelerometer. In the event of power failure, OFF warning flags will appear across the mach number and airspeed scales. The (OFF) airspeed warning flag will appear in the event of a malfunction or failure in the airspeed section of the AMI or CADC. The circuit breaker for the airspeed mach indicator is located on the left main ac bus.

Note

The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and therefore is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual since it is read directly from the instruments.

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

Angle-of-Attack Indicator. The angle-of-attack indicator, located on the airspeed-mach indicator, indicates in degrees the angular position of the wing chord in relation to the aircraft flight path. The vertical moving tape displays angle-of-attack from minus 10 degrees to plus 25 degrees. The angle-of-attack indicator is operated by the central air data computer.

WARNING

Erroneous readings can occur on the AMI while the aircraft angle-of-attack is above the indicator limit (25 degrees). At aircraft angles-of-attack of approximately 32 to 35 degrees, the angle-of-attack indication may decrease to a value below its limits, but will function normally as soon as the aircraft angle-of-attack is reduced below 25 degrees. If a flight condition is encountered in which the angle-of-attack indicator is near its limits (25 degrees) and then starts to decrease, monitor airspeed and assure that as the angle-of-attack is decreasing the airspeed is steadily increasing. If the airspeed is not steadily increasing, the aircraft is very likely entering a stall/post stall gyration. (Refer to "Emergency Procedures," Section III, for stall recovery.)

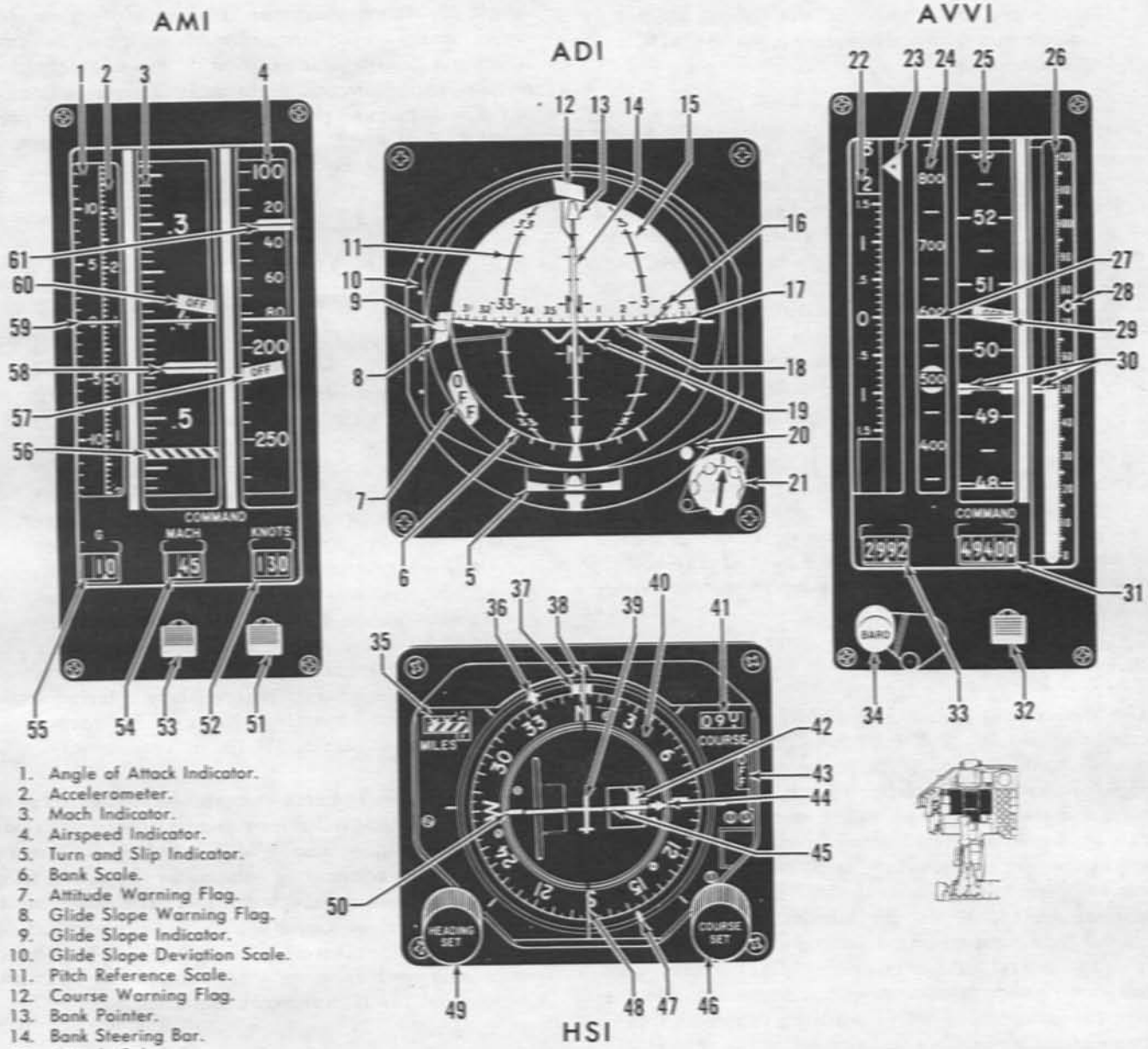
Accelerometer. The accelerometer located adjacent to the angle-of-attack indicator provides normal "g" (load factor) information. The "g"-forces being sustained by the aircraft are continuously shown by the acceleration scale read against a fixed index line. The tape scale is graduated from −4 to +10 "g's". 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.

Note

During abrupt pitching maneuvers, the aircraft rate of onset may exceed the 2 "g" per second maximum speed of the accelerometer tape. If this occurs, the accelerometer indicator readings will be less than actual aircraft acceleration levels.

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

Integrated Flight Instruments



1. Angle of Attack Indicator.
2. Accelerometer.
3. Mach Indicator.
4. Airspeed Indicator.
5. Turn and Slip Indicator.
6. Bank Scale.
7. Attitude Warning Flag.
8. Glide Slope Warning Flag.
9. Glide Slope Indicator.
10. Glide Slope Deviation Scale.
11. Pitch Reference Scale.
12. Course Warning Flag.
13. Bank Pointer.
14. Bank Steering Bar.
15. Attitude Sphere.
16. Heading Reference Scale.
17. Horizon Bar.
18. Pitch Steering Bar.
19. Miniature Aircraft.
20. Pitch Trim Index.
21. Pitch Trim Knob.
22. Vertical Velocity Indicator.
23. Vertical Velocity Index.
24. Vernier Altimeter.
25. Coarse Altimeter.
26. Gross Altimeter.
27. Fixed Index Line.
28. Target Altitude Marker.
29. Altitude Warning Flag.
30. Command Marker.
31. Command Altitude Readout Window.
32. Command Altitude Slewing Switch.
33. Barometric Pressure Readout Window.
34. Barometric Pressure Set Knob.
35. Range Indicator and Warning Flag.
36. Bearing Pointer.
37. Heading Marker.
38. Upper Lubber Line.
39. Aircraft Symbol.
40. Compass Card.
41. Course Selector Window.
42. Course Warning Flag.
43. Power Off Warning Flag.
44. Course Arrow Head.
45. Course Deviation Indicator.
46. Course Set Knob.
47. Bearing Pointer Tail.
48. Lower Lubber Line.
49. Heading Set Knob.
50. Course Arrow Tail.
51. Command Airspeed Slewing Switch.
52. Command Airspeed Readout Window.
53. Command Mach Slewing Switch.
54. Command Mach Readout Window.
55. Acceleration Readout Window.
56. Maximum Allowable Mach Marker.
57. Airspeed Warning Flag.
58. Command Mach Marker.
59. Fixed Index Lines.
60. AMI Power Warning Flag.
61. Command Airspeed Marker.

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Figure 1-33.

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 command 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 aircraft wing sweep position, pressure altitude, and true temperature. The maximum allowable mach marker is operated by an electrical signal from the maximum safe mach assembly.

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 a detected instrument failure or airspeed signal failure within 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. If the slewing switch is moved to the detented position on the right, the commanded airspeed marker will align with the fixed index and continuous digital presentation of the airspeed will then be displayed in the readout window.

Altitude-Vertical Velocity Indicator.

The altitude-vertical velocity indicator (AVVI) (figure 1-33), located on the left 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.

WARNING

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 using the position error shown in Appendix I. The radar altimeter also may be used since it provides an absolute indication of distance above the terrain at altitudes below 5000 feet.

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 altitude scale 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 marker 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) (figure 1-33), located on the left main instrument panel, is a remote indicating instrument which displays attitude, heading, turn and slip, glide slope deviation, altitude 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. The pitch reference may be adjusted with the pitch trim knob. The turn and slip indicator, located in the bottom of the ADI, 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 "Instrument System Coupler Pitch Steering Mode Switch," this section, for ADI indications during various modes of operation). An OFF warning flag indicates loss of power to the ADI when the ADI is receiving attitude information from the bomb nav system. Attitude data to the ADI is received directly from either the bomb nav system stabilized platform (SP) or the auxiliary flight reference system (AFRS) depending on the position of the flight instrument reference select switch. Normal operation of the ADI is with this switch in the PRI position which provides the instrument with signals from the SP. An off warning flag indicates loss of power to the ADI when data is being supplied from the bomb nav system. When data is being supplied from the AFRS, the warning flag indicates loss of power to the ADI or that the data is unreliable. It is possible to have failures within the ADI, AFRS or SP that can result in erroneous or complete loss of attitude reference without the presence of a warning flag or caution lamp indication. Other indications such as unrealistic or rapid changes in winds, ground speeds, or position, excessive radar cursor drift or unusual radar video uniformity, sudden attitude changes while on autopilot or frequent fly-ups while on TF may also indicate a possible erroneous attitude reference. When abnormal disagreement between the ADI and standby attitude indicator is encountered without a warning flag or caution lamp indication, the aircraft should be returned to level flight using basic flight instruments. Do not assume either indicator is reliable until the aircraft is straight and level and one of the indicators is determined to be accurate. If the above checks have determined a malfunctioning SP, the ADI source should be switched to the AFRS. A continuing attitude discrepancy indicates an ADI malfunction, therefore, the standby attitude indicator should be used.

WARNING

Frequent cross checks between the ADI, the standby attitude indicator and other basic flight instruments should be made to detect possible malfunctions. Failure to detect a malfunction and take corrective action could result in a flight attitude from which the aircraft cannot be recovered.

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

Horizontal Situation Indicator.

The horizontal situation indicator (HSI) (figure 1-33), 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. Aircraft heading or its reciprocal are read under an upper and lower lubber line. The aircraft symbol is fixed and is oriented to the nose of the aircraft. A heading set knob is provided to set a heading marker to the desired heading in the manual heading mode. 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. 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 on 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 (14, 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 and NAV position, the switch will return to OFF when a pull-up signal is generated by the armament system.

Note

Altitude reference submode limits are ±500 feet from the reference pressure altitude. If the set limits are exceeded, the reference altitude will change by the amount that the altitude limits are exceeded.

The switch will not hold in the ALT REF position if either TFR channel is in the TF mode and is operating normally.

Instrument System Coupler Mode Selector Knob.

The instrument system coupler mode selector knob, located on the instrument system coupler control panel (14, figure 1-5), has eleven positions. Nine positions of the knob are activated and are marked OFF, ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS, MAN HDG, and AIR/AIR. Two 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: For instrument system coupler modes versus instrument indications refer to figure 1-34. Refer to figure 1-35 for ADI and HSI instrument warning flag analysis. For HSI and ADI steering indication limits refer to figure 1-36.

- In the OFF position, the steering bars and OFF flags are biased out-of-view on the ADI and LCOS leaving attitude and heading displays.
- The ILS (instrument landing system) position provides the capability of flying ILS 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 of the horizontal situation indicator (HSI). Glide slope deviation is displayed on the glide slope deviation indicator on the ADI. Pitch steering commands are 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.

Note

- Once the glide slope is intercepted, a glide slope deviation of more than two dots as measured on the glide slope deviation scale will cause the pitch steering bar on the ADI and LCOS to drive out of view and remain out of view until a correction is made to bring the glide slope indicator back to the previous beam intercept point on the deviation scale.
- Also at glide slope intercept the bank steering bar reference is switched from normal (25 degrees) to approach reference (15 degrees). Refer to figure 1-36.
- If a localizer deviation of more than two dots on the course deviation indicator occurs when the pitch steering bars on the ADI and LCOS are in view, they will drive out of view until a correction is made to bring the localizer deviation back within the two dot limit.

I S C Mode Selector Knob Positions vs Indications

HORIZONTAL SITUATION INDICATION

I S C MODE SELECTOR KNOB POSITIONS

CONTROL / INDICATION	OFF	ILS	AILA	TACAN	CRS SEL NAV	NAV	MAN CRS	MAN HDG	AIR/AIR
Course Set Knob	Not Used	Used to set ldg approach crs in crs set window		Used to set TACAN course	Used to set desired course	Not used	Used to set course	Not used	
Heading Set Knob	Not used							Used to set desired mag hdg	Not used
Compass Card	Heading fr B/N sys when sel, or mag hdg fr the AFRS when pri/att/hdg caution light is ON. In TACAN or AIR/AIR modes it uses AFRS mag hdg only. (Note 1)								
Course Select Window & Course Arrow	Cur grd trk from B/N	Manually selected course				Cur grd trk from B/N	Man sel course	Current ground track from B/N.	
Course Deviation Indicator	Grd trk dev	Loc dev fr loc rec	Loc dev fr B/N	Dev from selected course	Nav dev fr B/N	Grd trk dev	Nav dev from B/N	Not used—Centered	
Heading Marker	Indicates computed course to B/N destination.							Mag hdg man set	Bearing to target from ARS
Power Off Warning Flag	Normal Condition—Out of view. Abnormal Condition—When in view, disregard HSI. Therefore, all manual set course/heading command steering information will be unusable on the ADI or LCOS.								
Range Ind and Warning Flag	Indicates distance to TACAN station.								
Bearing Pointer (TACAN)	Indicates bearing to TACAN station.								
To—From Indicator (TACAN)	Out of view Not used			To or fr TACAN sta	Out of view—Not used				

Note 1: Primary mag heading display will be true heading plus or minus handset mag variation.

Figure 1-34. (Sheet 1)

I S C Mode Selector Knob Positions vs Indications

ATTITUDE DIRECTOR INDICATOR

I S C MODE SELECTOR KNOB POSITIONS

CONTROL / INDICATION	*OFF*	*ILS*	*AILA*	*TACAN*	*CRS SEL NAV*	*NAV*	*MAN CRS*	*MAN HDG*	*AIR/AIR*
Bank Steering Bar	Out of View	Steer to Localizer	Com steer from B/N to set course	Steer to TACAN course	Com steer from B/N to set course	Com steer to selected destination	Com steer to selected course	Hdg from HSI and FDC	Target steer from attack radar
Pitch Steering Bar	Out of View Note 4	In view in alt ref and when G/S beam intercepted	Out of view unless TF or ALT REF is selected						Target elev from attack radar
Heading Reference Scale	Heading from the B/N system when selected, or mag hdg from the AFRS when pri/att/hdg caution lamp is ON. In TACAN or AIR/AIR modes it uses AFRS mag hdg only. (Note 1)								
Attitude Sphere	Pitch and roll from B/N system when selected, or from the AFRS when pri/att/hdg caution lamp is ON.								
Glide Slope Indicator	Out of View	G/S from G/S rec	G/S from B/N set	—Out of View—					
Course Warning Flag	Out of View	Out of view when loc adeq sig stren	Out of view loc valid from B/N	Out of view when TACAN adeq sig stren	Out of view when nav course steering is valid from B/N			Out of view when man hdg valid	Out of view
Glide Slope Warning Flag	Out of View	Out of view when G/S adeq sig stren	Out of view when G/S from B/N valid	—Out of View—					
Attitude Warning Flag	Normal condition—Out of view. Abnormal condition—In view, disregard ADI and use standby attitude indicator or LCOS for attitude and glide slope.								
I S C Pitch Steer Sw	TF or ALT REF	ALT REF prior to G/S (Note 2)	(Note 3) TF or ALT REF						Not useable
Flt Inst Ref Sel Sw	Primary is normal; if pri/att/hdg caution lamp comes on, select AUX position.								

Note 1: Prim mag hdg display is computed true heading plus or minus handset magnetic variation.

Note 2: ALT REF is useable prior to intercepting G/S. When G/S is intercepted the ALT REF is automatically disengaged by the FDC.

Note 3: TF or ALT REF is automatically disengaged by the pull up command from the armament system in CRS SEL NAV & NAV. The pull up signal is applied to the pitch steering bar of the ADI and LCOS.

Note 4: Unless TF or ALT REF is selected.

Figure 1-34. (Sheet 2)

Instrument Warning Flag Analysis

INSTRUMENT	WARNING FLAG	FLAG CONDITION	DISPLAY VALIDITY	RECOMMENDED ACTION
Airspeed Mach Indicator	Power warning flag (Mach tape)	In View	All displays not reliable.	Use standby airspeed indicator.
	Airspeed warning flag (Airspeed tape)	In View (CADS) caution lamp out	Airspeed display not reliable.	Use standby airspeed indicator.
		In View (CADS) caution lamp on	Only acceleration display reliable.	Use standby airspeed indicator.
		Out of View (CADS caution lamp on)	Only airspeed and normal acceleration displays reliable.	Use airspeed and altitude for mach number determination.
Altitude Vertical Velocity Indicator	Altitude Warning Flag	In View	All displays not reliable.	Use standby altimeter and standby vertical velocity indicator.
Attitude Director Indicator	Attitude (OFF) Warning Flag	In View	Only turn and slip reliable.	Use standby attitude indicator.
	Course Warning	In View	Bank steering bar not reliable.	Use HSI course deviation indicator.
	Glide slope warning flag	In View	Glide slope indicator not reliable. If TF submode engaged, pitch steering bar not reliable.	Use other landing mode or system.
Horizontal Situation Indicator	Power (OFF) Warning Flag	In View	All displays not reliable. Also, ADI bank steering bar not reliable if selected ISC mode requires manually setting HSI heading or course.	Use ADI heading and BDHI TACAN bearing and distance.
	Range Warning Flag	In View	Range indicator not reliable.	Use bearing-distance-heading indicator.
	Course Warning Flag	In View	Course deviation indicator not reliable.	Use ADI bank steering bar.
Bearing-Distance-Heading-Indicator	Power (OFF) Warning Flag	In View	All displays not reliable.	Use HSI.
	Range Warning Flag	In View	Range indicator not reliable.	Use HSI.
Standby Attitude Indicator	Power (OFF) Warning Flag	In View	Displays not reliable.	Use ADI.

Figure 1-35.

HSI and ADI Steering

	ILS		AILA		TACAN	CRS SEL NAV	NAV	MAN CRS	MAN HDG	AIR/ AIR	ALT REF Sub Mode
	Norm.	Appr.	Norm.	Appr.							
2 Dot Displ of (HSI) CRS DEV IND	2.5°	2.5°	2.5°	2.5°	10°	2.5°	19.0°	27.6°	Not Used		
2 Dot Displ of ADI G/S DEV IND	0.7°	0.7°	0.7°	0.7°		Not Used					
ADI Bank Steering Bar	25°	15°	25°	15°	40°	Approx 30°			25°	60°	Not Used
ADI Pitch Steering Bar	20°	20°	20°	20°	Not Used					20°	20°

Figure 1-36.

With the radar altimeter operating and set for a minimum altitude penetration, the pitch steering bars on the ADI and LCOS will indicate a fly-up command and the radar altitude low warning lamp will light when the aircraft penetrates the set altitude. If a pull-up is then initiated the fly-up command will be terminated and the warning lamp will go out when the aircraft is above the minimum penetration altitude setting. The pitch steering bar commands will be regained once the glide slope indicator is re-centered or by placing the pitch steering mode switch to ALT REF when level-off altitude is reached. In the event an ILS approach or AILA is begun from above 5000 feet absolute altitude the radar altitude low warning lamp will momentarily light and the pitch steering bars on the ADI and LCOS will momentarily indicate a fly-up command when the aircraft descends through 5000 feet.

- The AILA (airborne instrument low approach) position provides the capability of making instrument letdowns and approaches to runways not equipped with ground based letdown systems. This is an airborne radar approach. The bomb/nav system in conjunction with the attack radar is used to correct the present position longitude and latitude and will furnish simulated localizer and glide slope information to provide the same indications on the ADI, LCOS, and HSI as when using the ILS position. For AILA procedures refer to Section VII.

Note

During AILA approaches, when the bomb nav system is furnishing simulated localizer, the pitch steering bar will remain in view even when the aircraft deviates more than ±2 dots from the glide slope.

- The TACAN (tactical air navigation) position provides the capability of making instrument approaches and flying a selected course to or from a TACAN station. The course arrow and the course selector window are set to the desired course to be flown using the course set knob. 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 magnetic bearing to the station.
- The CRS SEL NAV (course select navigation) position provides the capability of approaching a selected destination along a selected course other than the most direct route. This provides the capability of avoiding weather, obstacles, and enemy areas.

To commence the course select navigation procedure, set destination counters, select CRS SEL NAV on the instrument system coupler and set the selected course in the HSL course selector window. This establishes a course signal to the bomb nav system where a course

error signal is developed. The bomb nav system supplies the flight director computer with two signals. These signals are: (1) the difference between selected courses and existing computed direct course to destination and (2) the difference between existing ground track and computed course to destination. These two signals are combined with aircraft roll to provide steering commands to the bank steering bars on the ADI and LCOS, and course deviation indicator on the HSI. When the selected course is set in the HSI, a right or left steering signal is generated for the ADI and LCOS bank steering bars. This signal depends on (1) aircraft position in relation to selected course, and (2) aircraft ground track in relation to the ground track required to make good on approach flight path. HSI course deviation bar will be displaced two dots until within 2.5 degrees of the selected course. To intercept the selected course at a predetermined position it is necessary to maintain the bank steering bars centered. When the aircraft is on the selected course the heading marker and course arrowhead will be aligned.

- The NAV (basic navigation) position provides computed 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, TRAIL BOMB or RANGE BOMB positions, computed 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. The course set knob and the heading set knob are not used in NAV mode. The course arrow and course selector window display current magnetic ground track from the bomb nav system. This mode is also used in conjunction with the heading navigation mode of autopilot operation. For further information refer to "Autopilot Systems," this section.
- 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 desired course is set in the course selector windows of the HSI. 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. Turn the aircraft to center the bank steering bars on the ADI and LCOS. 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 numerical setting of the heading and align the course arrow with the heading marker to reduce the possibility of heading confusion.

Note

With the instrument system coupler mode selector knob in the OFF, ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS or MAN HDG positions and with the instrument system coupler pitch steering mode switch at OFF, a fly-up command will be displayed on the pitch steering bars on the ADI and LCOS and the radar altitude low warning lamp will light when the airplane descends below the altitude index setting of the radar altimeter. The fly-up command will be terminated and the radar altitude low warning lamp will go out when the airplane climbs back through the altitude index setting. If a descent is begun from above 5000 feet above the ground with the instrument system coupler mode selector knob and pitch steering mode switch in the above positions the radar altitude low warning lamp will momentarily light and the pitch steering bars on the ADI and LCOS will momentarily indicate a fly-up command when the aircraft descends through 5000 feet.

- 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 aircraft to the target. The pitch steering bars on the ADI and LCOS will be activated and indicate the necessary pitch steering correction (aircraft angle-of-attack plus radar antenna tilt angle) to be on target.

Instrument Test Button.

The instrument test button (7, figure 1-29), located on the ground check panel, is provided for ground checking and trouble shooting of the integrated flight instruments, the instrument system coupler, and the total temperature indicator. Depressing and holding the button will provide a set of predetermined indications on the above instruments. 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. Tests selected with the button are completely independent of the CADC.

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 location of the lamps throughout the cockpit see figure 1-37.

MASTER CAUTION LAMP.

The master caution lamp (26, 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 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 (5, figure 1-38), 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 with the following exceptions:

1. The engine and fuselage fire pushbutton warning lamps are not dimmable and are tested by the fire detect test switch.
2. The following lamps are shutter type and are non-dimmable. They can be checked by the malfunction and indicator lamp test button.
   a. Master Caution Lamp.
   b. Radar Altitude Low Warning Lamp.
   c. Nose Wheel Steering/Air Refueling Indicator Lamp.
   d. Reference Not Engaged Caution Lamp.
   e. Cabin Pressure Warning Lamp.
   f. Canopy Unlock Warning Lamp.
   g. Reduce Speed Warning Lamp.
3. The following lamps are individually tested and dimmed.
   a. TFR R & L Channel Fail Indicator Lamps.
   b. IFF Panel Reply & Test Caution Lamps.
   c. CMRS Indicator Lamp.
   d. ECM Pod Panel Lamps.
   e. Angle-of-Attack Indexer Lamps. (Dimming Only)

All lamps are automatically set to bright when the internal lighting control knob (FLT INST) is off or when aircraft power is turned off.

Malfunction and Indicator Lamp Test Button.

The malfunction and indicator lamp test button (6, figure 1-38), located on the lighting control panel, is provided to check the landing gear warning horn and to check warning, caution, and indicator lamps in the cockpit for burned out bulbs. The following lamps are not checked by the test button: (1) the engine and fuselage fire pushbutton warning lamps, (2) the TFR R and L channel fail caution lamps, (3) the IFF control panel reply and test lamps, (4) the CMRS ready/test indicator lamp, (5) the threat pushbutton indicator lamps on the RHAW and ECM threat panels, and (6) the attack radar system malfunction, photo malfunction, and range lock indicator lamps. On aircraft modified by T.O. 1F-111-891, the malfunction and indicator lamp test button is also used to ground check the stall warning system.

Note

On aircraft after T.O. 1F-111-891, during ground checks, with the flight control system switch in T.O. & LAND or with the slats down, depressing the malfunction and indicator lamp test button may cause the rudder to deflect due to an adverse yaw compensation input.

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 (Refer to figure 1-1). The position lights consist of green lights in the right glove and wing tip, red lights in the left glove and wing tip and a white tail light. 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 two lights, located on the upper and lower surfaces of each wing tip, and four lights located forward and aft of each side of the fuselage. The lights in the wing tips correspond to the color of the left and right position lights. The fuselage lights are amber. Two anti-collision/fuselage lights, one located on top

Warning, Caution and Indicator Lamps (Typ)

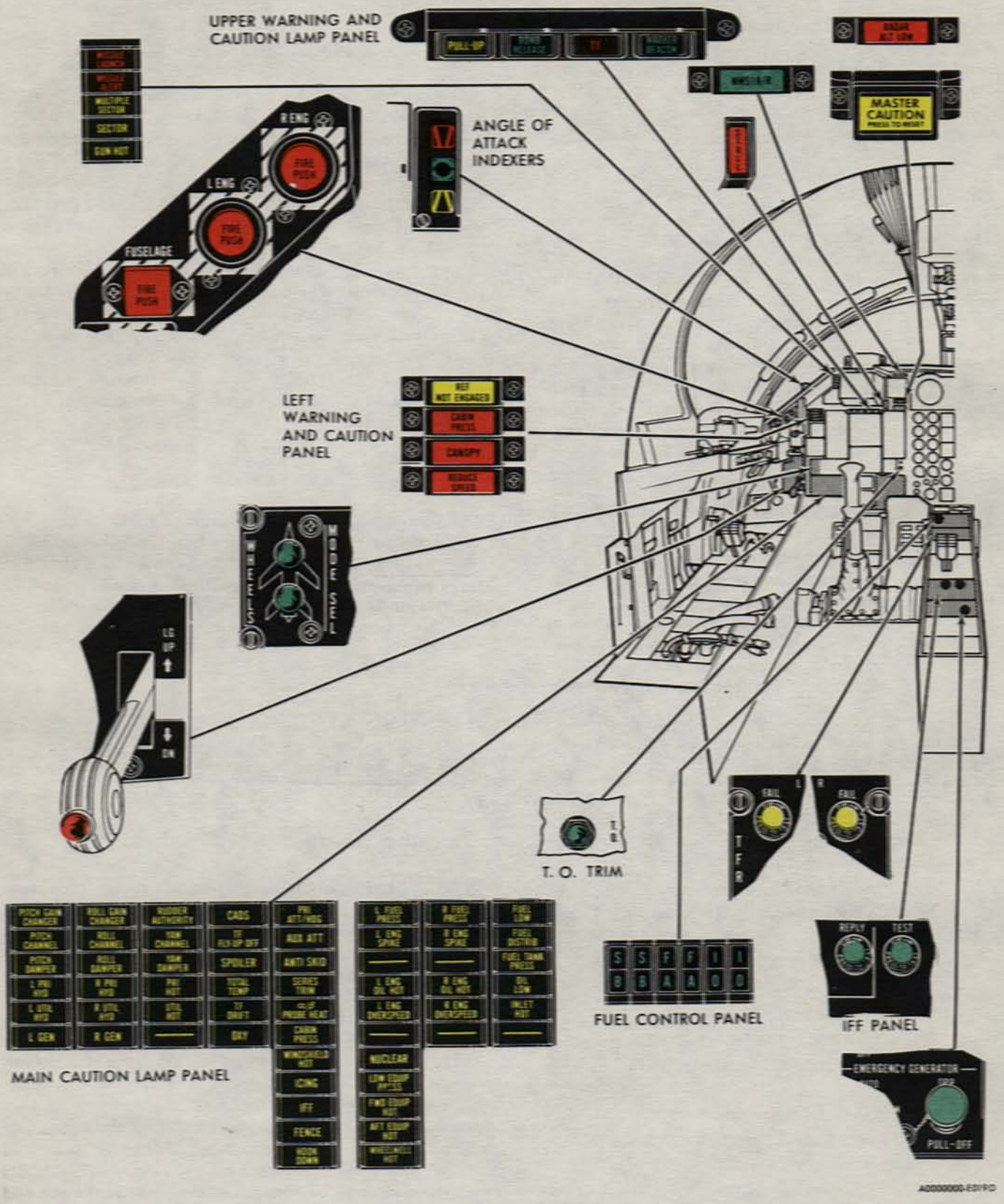

Figure 1-37. (Sheet 1)

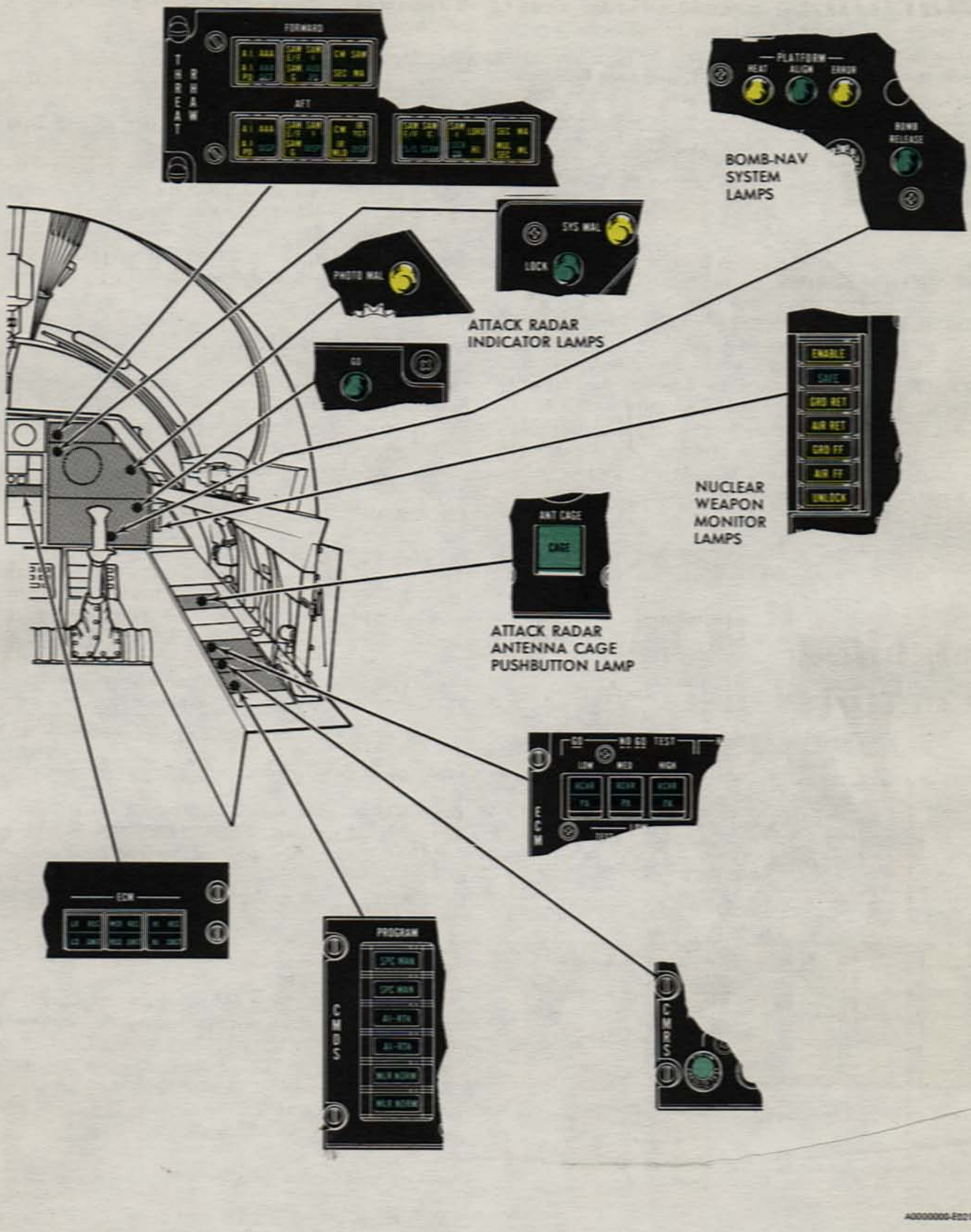

Figure 1-37. (Sheet 2)

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 switch on the nose gear down lock will turn the lights off if they are on when the gear is retracted.

Position Light Switches.

Three position light switches (7, figure 1-38) 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.

Lighting Control Panel

1. Formation Lights Switch.
2. Fuselage Lights Switch.
3. Anti-Collision Lights Switch.
4. Internal Lighting Control Knobs (9).
5. Malfunction and Indicator Lamp Dimming Switch.
6. Malfunction and Indicator Lamp Test Button.
7. Position Lights Switches.

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Figure 1-38.

Position Lights/Stores Refuel Battery Power Switch.

The position lights/stores refuel battery power switch (5, figure 1-29), 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 connects these circuits to the essential dc bus. 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 (1, figure 1-38), 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 115 volt ac power from the left main ac bus.

Anti-Collision Lights Switch.

The anti-collision lights switch (3, figure 1-38) 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 (2, figure 1-38) 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.

The air refueling receptacle lights control knob (40, figure 1-5) is located on the left main instrument panel. The knob is labeled A/R RECP LT. The full counterclockwise position of the knob turns the lights off. Turning the knob clockwise varies the intensity of the lights from off to full brightness. The knob controls 115 volt ac power from the essential ac bus.

Landing and Taxi Lights Switch.

The landing and taxi lights switch (3, figure 1-56) is located on the miscellaneous switch panel. The switch is marked LANDING, OFF and TAXI. If the switch is left in either the LANDING or TAXI positions on

takeoff, a switch on the nose gear down lock will turn them off when the gear is retracted. The switch controls 28 volt dc power from the essential bus which in turn controls relays to provide 115 volt ac power to a transformer which in turn provides 28 volt ac 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 set of red flood lights 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 volts ac power from the ac essential bus. Two utility lights (48, figure 1-2 and 6, figure 1-42), 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 volts dc from the engine start bus.

Internal Lighting Control Knobs.

Nine internal lighting control knobs (4, figure 1-38), 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 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. This will also turn on additional white flood lights. Once these lights are on their intensity may be decreased by turning the knob counterclockwise. All of the white flood lights will be turned off when the knobs is rotated to the OFF position.

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 degrees. 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 in flight. 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.

Note

After descending below 8000 feet in an ejected crew module it is possible that atmospheric pressure differential on the canopy will prevent them from being opened. To eliminate this pressure differential remove the caps from the snorkel ventilation ports and push in on the exposed tube. This should be done on the ground after the module has come to rest.

INTERNAL CANOPY LATCH HANDLES.

Two canopy latch handles are located on the inside lower horizontal frame member of each canopy hatch (11, 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 in flight. 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 must be returned to the detent position to lock the counterpoise and hold the hatch. Each handle is me-

chanically 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, a flush mounted pushbutton plunger located adjacent to the external canopy handle is provided to unlock the internal canopy handle from the outside.

CANOPY UNLOCK WARNING LAMP.

A red canopy unlock warning lamp located on the left warning and caution lamp panel (figure 1-37) 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 SYSTEMS.

The air conditioning and pressurization systems (figure 1-39) 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, anti-"g" and pressure suits, and pneumatic pressure for throttle boost and fuel tank pressurization.

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 weapons bay, and to the electronic equipment in the forward electronics bay which requires a controlled environment for efficient operation. See figure 1-39. 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 cockpit 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.

Note

- During ground operation at high power settings, steam may be discharged from the air-to-water heat exchanger vent located on the lower side of the fuselage between the weapons bay doors and the main landing gear door. This is a normal condition and should be no cause for concern.
- During operations under humid conditions, rapid increases in power may induce flash freezing in the water separator causing a loss of cooling air to the cabin and electronic equipment. Normally flash freezing is of short duration (2-3 minutes), and is self-correcting. This condition may be diagnosed by observing loss of cabin airflow followed by lighting of the forward equipment hot caution lamp 120 seconds later. The duration of this condition may be reduced by placing the engine/inlet anti-icing switch to MAN and return to AUTO when the forward equipment hot caution lamp goes out. Refer to "Caution Lamp Analysis," Section III.
- During operation at low or idle power settings the air conditioning system control valves will modulate toward an open position. When engine power is advanced high airflow will occur until the cabin temperature control valves return pressure to normal. During high cabin airflow cabin pressure may vary ±500 feet.

Cabin Air Distribution Control Lever.

A cabin air distribution control lever (7, figure 1-42), located on the right side wall, controls distribution of airflow 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 the AFT position. In this position air-flow into the cockpit is separated between the rear bulkhead diffusers and the windshield defog system with approximately 85 percent directed to the diffusers. Moving the lever towards 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. During sustained flight at high altitude and high airspeed, optimum crew comfort is obtained when the lever is in full forward position. 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.

Note

- During operation at high power settings, and when airflow to the cabin is high, moisture may be sprayed on the windshield and in the crew members' face. Moving the lever toward the aft position will decrease the possibility of the condition.
- Prior to conducting a rapid descent in a tropical environment, the cabin air distribution control lever should be placed in the FWD DEFOG position. This action will divert the cabin ventilation airflow across the windshield to prevent the inner surface temperature from falling below the local dewpoint. In the event of fog formation, the cabin temperature control knob should be rotated toward the WARM position.

Air Source Selector Knob.

The air source selector knob (2, figure 1-40), located on the air conditioning control panel, has six positions marked OFF, L ENG, BOTH, R ENG, RAM, and EMER. The knob controls bleed air source or allows selection of emergency ram air operation when the normal system is not operating. The knob controls a series of valves which operate as follows in the different knob positions:

- In the OFF position, the left and right engine bleed air check and shutoff valves are closed. The cold air modulating valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating (no flow). The pressure regulating and shutoff valve is closed. Pressure will not be available for pressurization or air conditioning functions, including throttle boost, fuel tanks, electronic equipment or cabin pressurization. Caution lamps associated with these systems may come on shortly after selection of the OFF position.
- In the L ENG position, the left engine is the source of bleed air, and the right engine bleed air check and shutoff valve is closed. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating shutoff valve will be regulating.
- In the BOTH position, the left and right engine bleed air check and shutoff valves are open. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve will be regulating.
- In the R ENG position, the right engine is the source of bleed air and the left engine bleed air check and shutoff valve is closed. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve will be regulating.
- In the RAM position, the engine bleed air check and shutoff valves are open. The cold air modulating valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve is closed. The ram air door is open. The RAM position 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. In the RAM position bleed air pressure is available to the wing seals, fuel tank pressurization system, electronic equipment, windshield wash and rain removal, throttle boost, anti-"g" suits, pressure suits, and canopy seals.

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.

- On aircraft [80] ♦ and those modified by T.O. 1F-111-572, an EMER position is provided. In the EMER position the left and right engine bleed air check and shutoff valves are closed. The cold air modulating valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating (no flow). The pressure regulating and shutoff valve is closed. The emer-

Air Conditioning and Pressurization System (Typ)

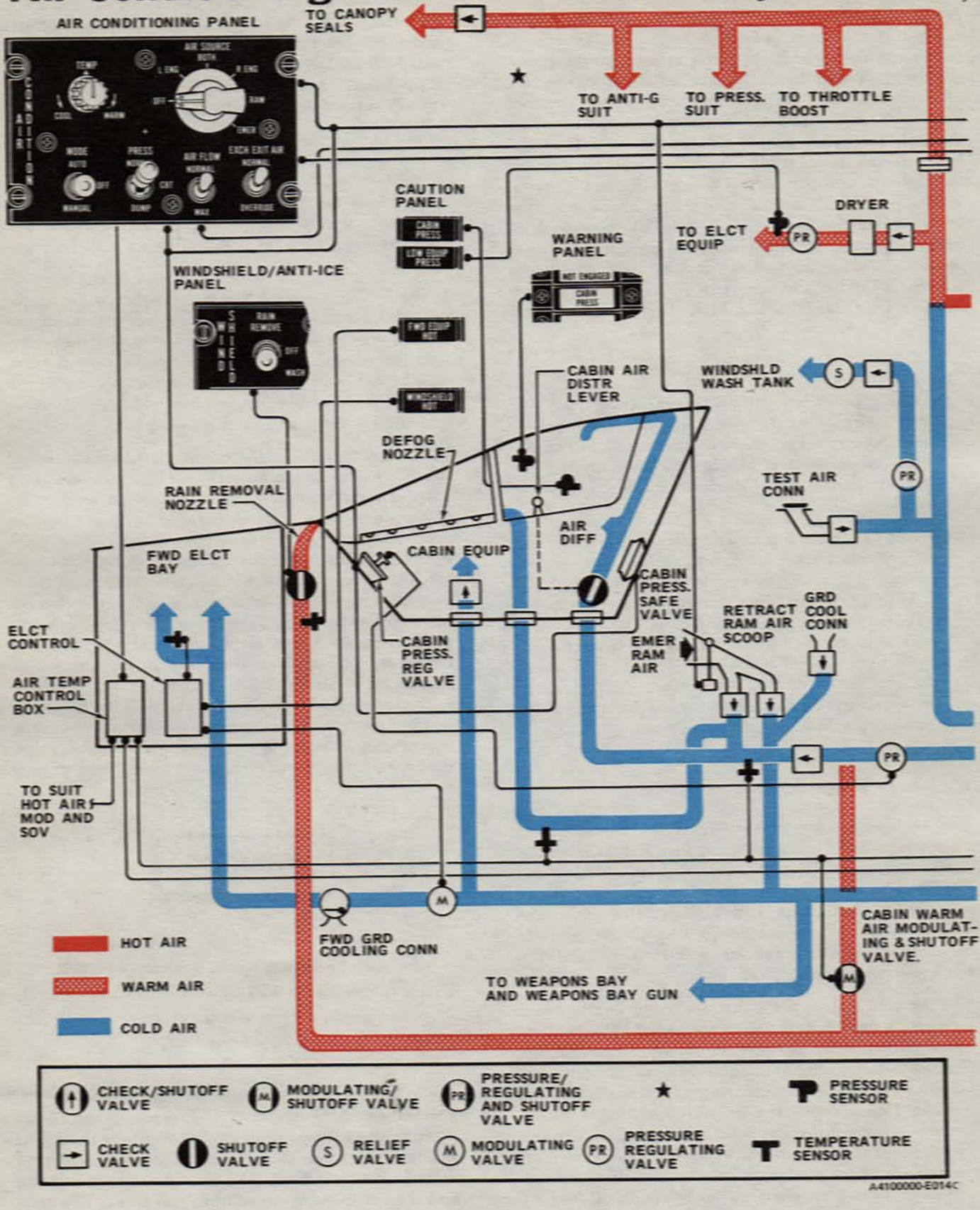


Figure 1-39. (Sheet 1)

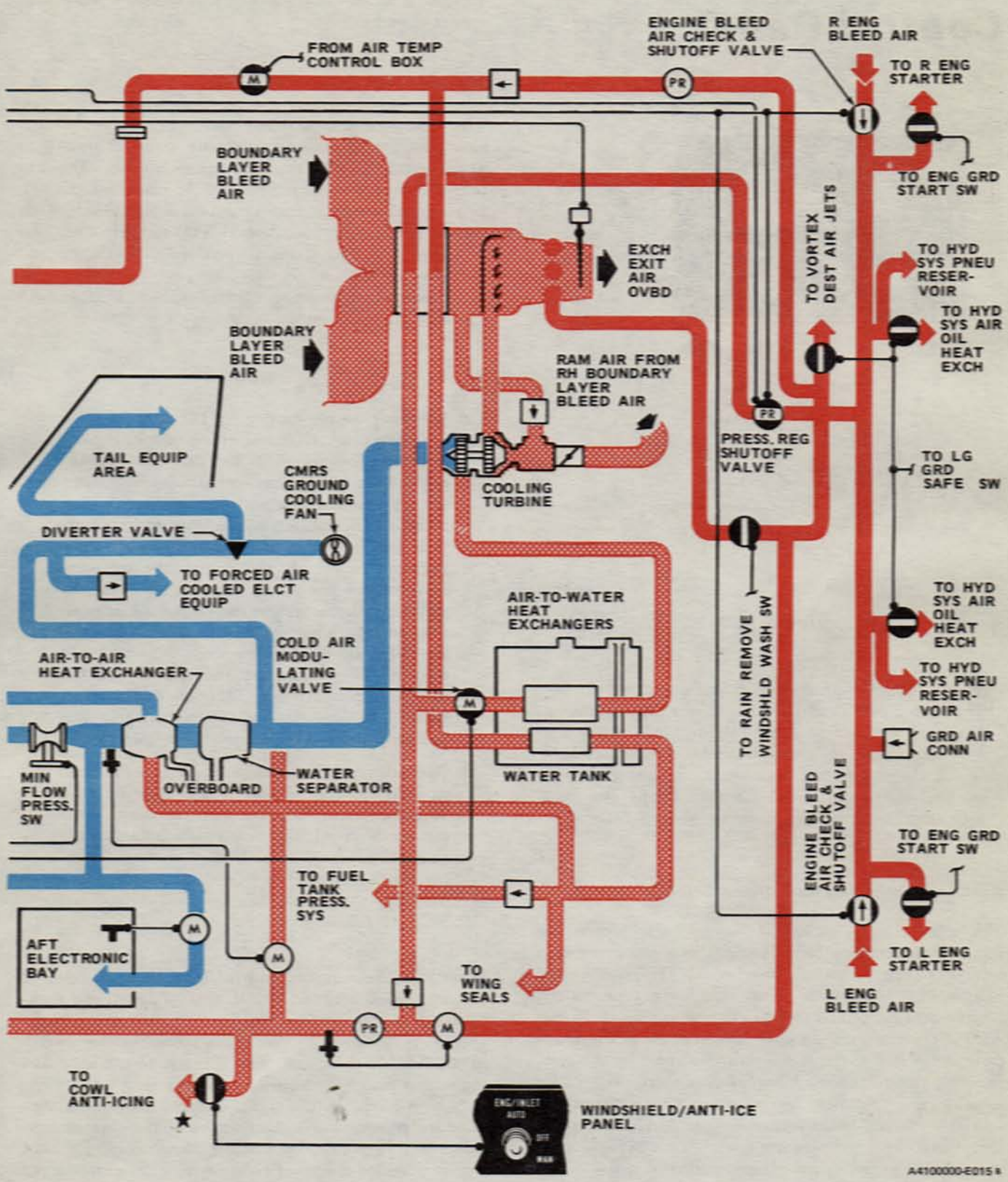
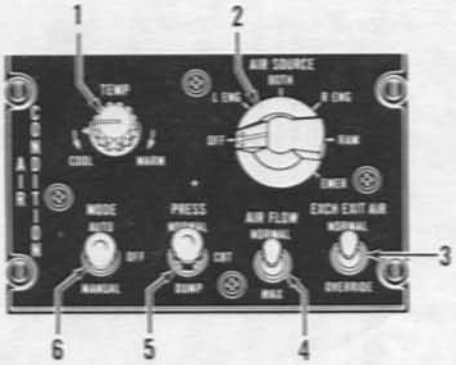


Figure 1-39. (Sheet 2)

Air Conditioning Control Panel



1. Temperature Control Knob.
2. Air Source Selector Knob.
3. Exchange Exit Air Control Switch.
4. Air Flow Selector Switch.
5. Pressurization Selector Switch.
6. Mode Selector Switch.

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Figure 1-40.

gency ram air door will open and provide sufficient air flow for cabin ventilation and equipment cooling. Bleed air will not be available for pressurization of the wing seals, fuel tanks, windshield wash, rain removal, throttle boost, pressure suits, anti-g suits or for electronic equipment. Caution lamps associated with these systems may come on shortly after selection of the EMER position.

Air Conditioning System Mode Selector Switch.

The mode selector switch (6, figure 1-40), located on the air conditioning control panel, is a three position lever-lock toggle switch marked AUTO, OFF, and MANUAL. The switch is a lock lever type switch which must be pulled out to change positions. 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 MANUAL position, the cockpit temperature controller is bypassed and control of the modulating temperature control valves is directly from the temperature 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 1-40), located on the air conditioning control panel, is provided to select cockpit temperature. 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 the cockpit temperature controller which constantly positions the modulating temperature control valves to maintain the selected temperature.

Note

- When operating in the AUTO mode, during extreme climatic conditions (cockpit temperatures below 45°F or above 95°F) the cabin temperature control system will operate in the maximum warming or cooling mode until the sensed cockpit temperature falls within the above range. While the temperature is outside this range, the temperature control knob will have no effect on the airflow and the system may have the appearance of supplying uncontrolled hot (or cold) air. This condition may exist for up to 15 minutes, depending on the severity of the temperature extreme. During this period, control of the airflow can be achieved by utilizing the MANUAL mode of operation.
- 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.

When the temperature control knob is positioned at the mid-point between COOL and WARM, the cockpit temperature is maintained at approximately 19 degrees C (67 degrees F). 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 a spring loaded detent at either one of the extreme positions, COOL or WARM. Maximum valve travel time from maximum cold to maximum warm is approximately 45 seconds.

Exchange (Exch) Exit Air Control Switch.

The exchange exit air control switch (3, figure 1-40), located on the air conditioning control panel, is a two position switch marked NORM and OVERRIDE. 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 degrees 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 OVERRIDE will override the automatic functions of the central air data computer and open the door to its full travel.

Note

The air conditioning system water supply is required for supersonic flight conditions. When the water tank becomes empty at high speed flight, the system will cycle off. If this condition occurs, the switch should be placed to the OVERRIDE position to obtain maximum cooling. Normal system operation can be expected below 65 degrees C total temperature.

Air Flow Selector Switch.

The air flow selector switch (4, figure 1-40), installed on aircraft 40 ↓ and those modified by T.O. 1F-111-687, is a two position switch marked NORMAL and MAX. The switch provides a means of controlling the amount of airflow in the cabin. The NORMAL position provides cabin airflow for normal usage. The MAX position may be used for low level, high speed operation on a hot day. The MAX position may also be used during rain removal, windshield wash or for defogging.

Equipment Hot Caution Lamp.

The amber equipment hot caution lamp, marked FWD EQUIP HOT, is located on the main caution light panel (figure 1-37). The lamp will light if the cooling air flow is insufficient. The following equipment is listed in the order of heat generation. The listing should be used as a guide for equipment shutdown, depending on flight requirements. Shutdown may be required to prevent degraded performance and/or damage from overheating.

- ECM
- Attack Radar
- TFR
- HF Radio (transmit)
- IRRS
- RHAW
- UHF Radio (transmit)
- Bomb Nav
- TACAN
- Radar Altimeter
- HF Radio (receive)
- IFF
- UHF Radio (receive)

PRESSURIZATION SYSTEM.

Pressurization of the cockpit, throttle boost, wing seals, fuel tanks, rain removal, canopy seals, anti-g suits, and electronic equipment 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 aircraft 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 22,500 feet. See figure 1-41 for cockpit pressure schedule for normal and combat conditions.

Cabin Pressure Schedule

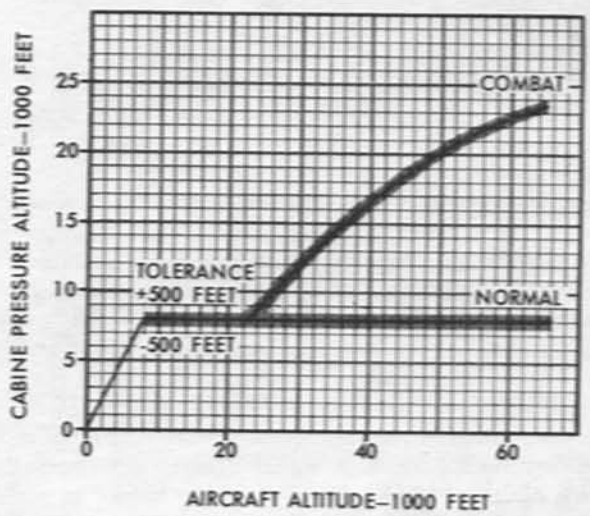
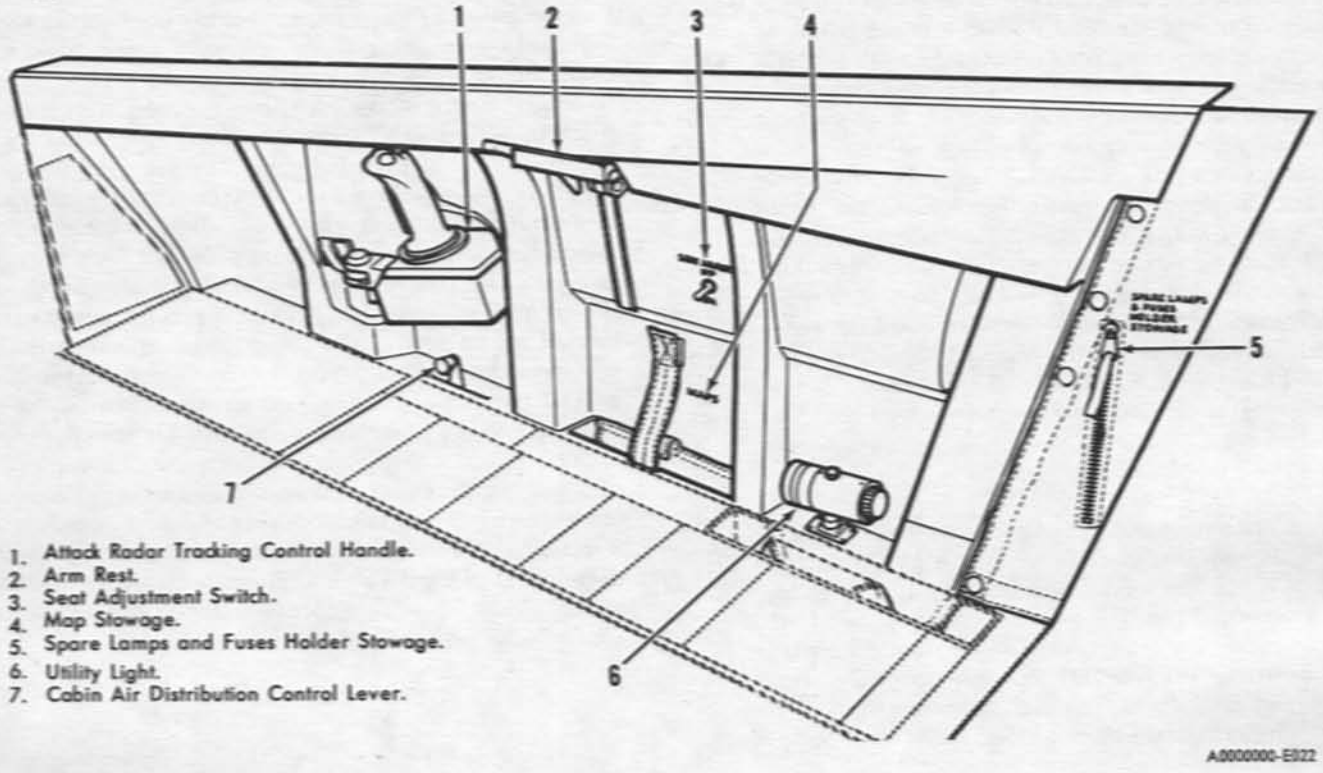


Figure 1-41.

Right Sidewall



1. Attack Radar Tracking Control Handle.
2. Arm Rest.
3. Seat Adjustment Switch.
4. Map Stowage.
5. Spare Lamps and Fuses Holder Stowage.
6. Utility Light.
7. Cabin Air Distribution Control Lever.

Figure 1-42.

CAUTION

A priority valve is incorporated into the system which insures that electronic equipment conditioning takes precedence over the cockpit. Under conditions of high cockpit pressure differential and low air flow, such as an idle power descent from altitude, a slow depletion of cabin pressure may occur to the extent that the cabin pressurization caution lamp may light. Both temperature control and defog functions will be ineffective until engine power is increased. Failure of this valve may cause forward equipment hot caution lamp to light, loss of cabin cooling airflow, unstable cabin pressurization and temperature control.

Pressurization Selector Switch.

The pressurization selector switch (5, figure 1-40), located on the air conditioning control panel, is a three position lever lock type 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 aircraft. 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 ambient pressure. 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 (1, figure 1-14), located on the auxiliary gage panel, 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 (figure 1-37). The lamp will light when the cabin altitude is above 10,000 feet. When operating the cabin pressurization system in COMBAT, the caution lamp will be lighted when aircraft altitude is above 26,000 feet.

Pressurization Warning Lamp.

A red pressurization warning lamp (8, 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 equipment low pressure caution lamp marked LOW EQUIP PRESS is located on the main

caution light panel (figure 1-37). The lamp will light when the supply pressure to the pressurized electronic equipment (TFR and ECM equipment) requiring one atmosphere pressure drops below 12.5 (±0.5) psi.

ANTI-ICING AND DEFOG SYSTEMS.

The anti-icing systems are provided to prevent formation of ice on the engine nose cone, engine inlet guide vanes, spike lip, fixed cowl, auxiliary cowl, pitot static, total temperature, angle-of-attack, and side slip angle probes. The engine and engine inlet anti-icing systems use regulated compressor bleed air from each engine. 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 to activate the system. 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 Anti-Icing System.

The engine anti-icing system prevents formation of ice on the engine inlet guide vanes, and engine nose cone. The engine anti-icing system uses regulated compressor bleed air from the 12th stage compressor of each engine. Both automatic and manual modes of operation are provided. Power for the engine anti-icing system is controlled by the engine/inlet anti-icing switch located on the windshield wash/anti-icing control panel.

Engine/Inlet Anti-Icing.

The engine/inlet anti-icing system prevents formation of ice on the spike tip, leading edge of the fixed cowl, leading edge of the auxiliary cowl, and vortex generators. The engine inlet system uses regulated compressor bleed air from the 16th stage of each engine regulated to 390 (±10) degrees F at 45 psi. 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.

Engine/Inlet Anti-Icing Switch.

The engine/inlet anti-icing switch (3, figure 1-43), 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.

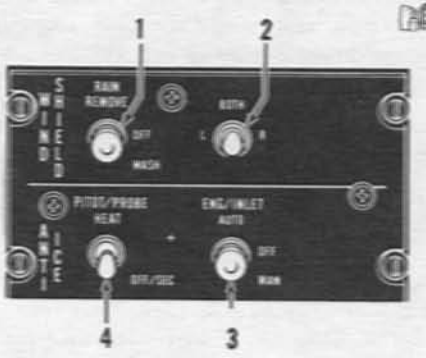
Engine Icing Caution Lamp.

The engine icing caution lamp, located on the main caution lamp panel (figure 1-37), 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 conditions ceases.

Inlet Hot Caution Lamp.

The inlet hot caution lamp, located on the main caution lamp panel (figure 1-37), provides an indication

Windshield Wash/ Anti-Icing Control Panel



1. Windshield Wash/Rain Removal Selector Switch.
2. Windshield Selector Switch.
3. Engine/Inlet Anti-Icing Switch.
4. Pitot/Probe Heater Switch.

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Figure 1-43.

that the temperature of anti-icing bleed air to the auxiliary cowls has exceeded 420 (±10) degrees F. When the lamp lights the words INLET HOT are visible and anti-icing air to the auxiliary cowls is automatically shut off, then the lamp will go out.

Probe Anti-Icing.

Heating elements powered by 115 volt ac are provided on the pitot-static, total temperature, angle-of-attack, and side slip angle probes for anti-icing. Power for the total temperature and primary alpha/beta probe heaters is furnished from the left main ac bus. The primary pitot system heater receives power from the essential ac bus and the secondary pitot system heater receives power from the right main ac bus. The alpha/beta probe secondary and alpha/beta body heaters are furnished power from the right main ac bus. Power to the probe heater is controlled by a pitot/probe heater switch located on the windshield wash/anti-icing control panel. An alpha/beta probe caution lamp is provided to monitor the function of the heaters in the angle-of-attack and side slip angle probes.

Pitot/Probe Heater Switch.

The pitot/probe heater switch (4, figure 1-43), located on the windshield wash/anti-icing control panel, has two positions marked HEAT and OFF/SEC (secondary). The switch performs the following functions:

On the ground.

- The OFF/SEC position turns off power to the probe heaters, and causes the alpha/beta probe heat caution lamp to light.
- The HEAT position furnishes power to primary alpha/beta heaters, and to the pitot-static probe.

Inflight.

- The OFF/SEC position provides power to the secondary alpha/beta heaters and the pitot-static heater. The total temperature heater is off.
- In the HEAT position, power is provided to the primary alpha/beta heaters, pitot-static heater and the total temperature heater.

On takeoff the squat switch on the landing gear activates to arm the secondary heater circuits in the angle-of-attack and sideslip angle probes. During flight if the primary heaters in either the angle of attack or angle of sideslip probes malfunctions, the alpha/beta probe heat caution lamp will light and the secondary heater in the failed probe will be automatically energized. Momentarily placing the switch to OFF/SEC should extinguish the caution lamp while the switch remains in that position, thereby verifying that the secondary heaters are functioning properly. The total temperature probe heater does not receive power while the switch remains in the OFF/SEC position.

CAUTION

To prevent overheating the probes, the switch should not be positioned to HEAT until just prior to takeoff.

The HEAT position may be used to ground check the heater in the pitot-static probe and the primary heaters in the angle-of-attack and sideslip angle probes. Proper operation of the primary heaters will be indicated by the alpha/beta probe heat caution lamp going out immediately after the switch is positioned to HEAT. The total temp probe heater may also be checked by placing the switch to HEAT and depressing the flight control master test switch and holding the CADC test switch to HIGH. Operation of the total temperature heater can be confirmed by observing an increasing temperature on the total temperature indicator.

Note

If the total temperature exceeds 50 degrees C during ground operation the CADS caution lamp may light and remain lighted until total temperature drops below 50 degrees C.

The switch controls 115 volt ac power to the heating elements in the pitot-static probe and total temperature probe, when the aircraft is airborne, and to the primary or secondary heating elements in the angle-of-attack (alpha) probe and side slip angle (beta) probes.

Alpha/Beta Probe Heat Caution Lamp.

The alpha/beta probe heat caution lamp, located on the main caution lamp panel (figure 1-37), provides indications that the angle-of-attack and/or side slip angle probe heaters are not functioning properly as follows:

On the ground.

- Indicates the pitot/probe heater switch is in the OFF/SEC position.
- With the pitot/probe heater switch in the HEAT position, indicates the primary heater element(s) in either or both probes are malfunctioning or have overheated and have been deenergized by the thermostats.

Inflight.

- With the pitot/probe heater switch in the HEAT position, indicates the primary heater element(s) in either or both probes are not functioning.
- With the pitot/probe heater switch in the OFF/SEC position, indicates the secondary heater elements in either or both probes are not functioning.

Note

The alpha/beta probe heat caution lamp is disabled at speeds above mach 1.10.

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 ice, impinging rain, and insects. Compressor bleed air at a temperature of 400 (±10) degrees 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 ice, impinging rain and prevent further accumulation of ice and 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. There is sufficient solution to provide five applications for both windshields or 10 applications for one windshield. 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 1-43), 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. 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). Allowing the switch to return 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.

WARNING

During windshield wash operation, forward vision is effectively obscured. Do not use windshield wash anytime forward vision is essential.

Windshield Selector Switch.

The windshield selector switch (2, figure 1-43), 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 removal air as a function of the position of the windshield wash/rain removal selector switch. For optimum performance of rain removal system, operate left side only. Selection of BOTH position will decrease airflow on each windshield.

Windshield Hot Caution Lamp.

The windshield hot caution lamp, located on the main caution lamp panel (figure 1-37) indicates windshield high temperature. An overheat switch, installed in the rain removal air supply duct upstream of the shutoff valve, will close when the air temperature is above 450 degrees 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.

FUSELAGE FIRE AND WHEEL WELL OVERHEAT SYSTEMS.

Two separate systems are provided for fuselage fire detection and extinguishing, and wheel well overheat detection. The fuselage fire and extinguishing system provides protection for the weapons bay, cheek areas, and crew module stabilization glove area. The wheel well overheat detection system provides an indication of overheat in the wheel well area. Both systems consist of sensing elements, warning or caution lamps and test circuits. The fuselage fire detection system also includes an extinguishing agent similar to the engine fire detect system.

FUSELAGE FIRE DETECTION AND EXTINGUISHING SYSTEM.

On aircraft modified by T.O. 1F-111-670, a fire detection and extinguishing system is provided to detect fire in the weapons bay, cheek areas, and crew module stabilization glove area. When a sufficient rise in temperature is detected by sensors in these areas a fuselage fire warning lamp will light. When a fire is indicated an extinguishing agent may be discharged

into the protected areas simultaneously from a single container located in the nose wheel well area. The system is supplied ac power from the ac essential bus, and dc power from the dc essential bus.

Fuselage Fire Pushbutton Warning Lamp.

The fuselage fire pushbutton warning lamp (2, figure 1-5), located on the left main instrument panel, is labeled FUSELAGE. When a fire is detected the warning lamp will light displaying the words FIRE PUSH. Depressing the button will arm the extinguishing agent discharge/fire detect test switch. The agent discharge/fire detect switch must be placed to the AGENT DISCH position to discharge the extinguishing agent. Redepressing the fuselage fire pushbutton warning lamp will disarm the fire extinguisher agent discharge switch. The button is covered by a frangible cover to prevent inadvertent actuation and provide a visual indication the button has been actuated.

Note

The fuselage fire warning lamp may not go out immediately after discharging the fire extinguishing agent if aircraft structure or equipment adjacent to the sensing element was heated to a temperature above the element setting.

Agent Discharge/Fire Detect Test Switch.

Positioning the agent discharge/fire detect test switch (3, figure 1-5) to the lever-locked AGENT DISCH position causes fire extinguishing agent to be discharged into the affected fuselage area, provided the fuselage fire pushbutton warning lamp has been depressed. Holding the switch to the spring-loaded FIRE DETECT TEST position will light the fuselage fire pushbutton warning lamp if the fuselage fire detection system is operational. For further information on this switch, refer to "Engine Fire Detection and Extinguishing System" and "Wheel Well Overheat Detection System," this section.

Fuselage Overheat Test Switch.

The fuselage overheat test switch (6, figure 1-29), located on the ground check panel, is a three position switch marked LOOP 1, NORM, and LOOP 2. The switch is spring loaded to the NORM position. Positioning the switch to LOOP 1 puts an artificial signal (short to ground) on loop 2 and will cause the WHEEL WELL HOT lamp and the fuselage FIRE PUSH lamp to light if loop 1 is shorted to ground. Loop 2 is checked in the same manner when the switch is positioned to LOOP 2. Both the wheel well overheat and the fuselage fire detect systems are checked simultaneously. When the switch is in the LOOP 1 or LOOP 2 position, a short in a single loop of either the wheel well overheat or the fuselage fire detect system will light both lamps.

Note

During normal operation a signal must be present on both loops of either system to light its associated lamp; a short on a single loop will not prevent detection of an overheat condition.

WHEEL WELL OVERHEAT DETECTION SYSTEM.

On aircraft (30) ♦ and those modified by T.O. 1F-111-572, a wheel well overheat detection system provides a visual indication of an overheat condition in the main wheel area in event of a rupture in the engine bleed air lines. The function of the system is similar to the engine and fuselage fire detection systems. Sensing elements, located in the main wheel well and plumbing crossover areas aft of the main landing gear bulkhead, detect a rise in temperature and light the wheel well hot caution lamp when a predetermined temperature is reached. When an overheat condition is indicated, refer to Section III.

Agent Discharge/Fire Detect Test Switch.

Holding the agent discharge/fire detect test switch (3, figure 1-5) to the spring-loaded FIRE DETECT TEST position will light the wheel well hot caution lamp if the wheel well overheat detection system is operational. The AGENT DISCH position of the switch serves no function with this system. For further information on the switch, refer to "Engine Fire Detection and Extinguishing System" and "Fuselage Fire Detection and Extinguishing System," this section.

Wheel Well Hot Caution Lamp.

A wheel well hot caution lamp (figure 1-37), located on the main caution lamp panel, provides an indication of an overheat condition in the main wheel wells and plumbing crossover areas aft of the landing gear bulkhead, in event of a rupture in the engine bleed air lines. The words WHEELWELL HOT are visible when the lamp is lighted.

OXYGEN SYSTEM.

The oxygen system consists of a normal (liquid) system located in the forward fuselage and cockpit and an emergency (gaseous) system located behind the cockpit aft bulkhead.

NORMAL OXYGEN SYSTEM.

The normal oxygen system consists of a 10 liter (15 liters on aircraft modified by T.O. 1F-111-563) liquid oxygen converter, which converts the liquid oxygen to a gas; a heat exchanger, which heats the gas to a temperature suitable for breathing, an on-off control valve at each station and a diluter demand mini-regulator attached to each crew member's torso harness. A three

Oxygen Duration

DATE: 24 APRIL 1970

With regulator at 100 percent or EMER position.

CABIN ALTITUDE	CONSUMPTION 2 MEN cu. ft./hr.	DURATION — HOURS*														
35,000	9.8	46.8	43.7	40.6	37.4	34.3	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1
30,000	13.4	34.3	32.0	29.7	27.4	25.1	22.8	20.5	18.3	16.0	13.7	11.4	9.1	6.8	4.6	2.3
28,000	15.0	30.6	28.5	26.5	24.4	22.4	20.4	18.4	16.3	14.3	12.2	10.2	8.2	6.1	4.1	2.0
26,000	16.0	28.6	26.7	24.8	22.9	21.0	19.1	17.2	15.3	13.4	11.5	9.6	7.6	5.7	3.8	1.9
24,000	18.52	24.7	23.1	21.4	19.8	18.1	16.5	14.9	13.2	11.6	9.9	8.3	6.6	5.0	3.3	1.6
22,000	20.76	22.1	20.7	19.2	17.7	16.2	14.7	13.3	11.8	10.3	8.8	7.4	5.9	4.4	2.9	1.5
20,000	23.0	20.0	18.6	17.2	15.9	14.6	13.3	12.0	10.6	9.3	8.0	6.6	5.3	4.0	2.7	1.3
18,000	25.24	18.1	16.9	15.7	14.5	13.3	12.1	10.9	9.7	8.5	7.3	6.1	4.8	3.6	2.4	1.2
16,000	27.48	16.8	15.6	14.5	13.4	12.3	11.1	10.0	8.9	7.8	6.7	5.6	4.4	3.3	2.2	1.1
14,000	30.0	15.6	14.5	13.5	12.4	11.4	10.2	9.2	8.2	7.1	6.1	5.1	4.1	3.1	2.0	1.0
12,000	32.8	13.9	13.0	12.1	11.1	10.2	9.3	8.4	7.5	6.5	5.6	4.7	3.7	2.8	1.9	0.9
10,000	35.6	12.9	12.0	11.1	10.3	9.5	8.6	7.7	6.9	6.0	5.1	4.3	3.4	2.6	1.7	0.8
8,000	39.4	11.6	10.8	10.1	9.3	8.5	7.8	7.0	6.2	5.4	4.7	3.9	3.1	2.3	1.5	0.7
0	55.6	8.2	7.7	7.1	6.6	6.0	5.5	4.9	4.4	3.8	3.3	2.7	2.2	1.6	1.1	0.5
AVAILABLE OXYGEN	LITERS (LIQUID)	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
	Cu. Ft. GAS	459.0	428.4	397.8	367.2	336.6	306.0	275.4	244.8	214.2	183.6	153.0	122.4	91.8	61.2	30.6

← Modified Aircraft →

With regulator in NORM position.

← Non-Modified Aircraft →

CABIN ALTITUDE	CONSUMPTION 2 MEN cu. ft./hr.	DURATION — HOURS*														
35,000	9.8	46.7	43.6	40.5	37.4	34.3	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1
30,000	13.2	34.7	32.4	30.1	27.8	25.5	23.2	20.9	18.6	16.2	13.9	11.6	9.3	7.0	4.6	2.3
25,000	14.0	32.8	30.6	28.4	26.2	24.0	21.9	19.7	17.5	15.3	13.1	10.9	8.7	6.6	4.4	2.2
20,000	12.4	37.0	34.6	32.1	29.6	27.2	24.7	22.2	19.8	17.3	14.8	12.3	9.9	7.4	4.9	2.5
15,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
10,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
8,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
5,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
Sea Level	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
AVAILABLE OXYGEN	LITERS (LIQUID)	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
	Cu. Ft. GAS	459.0	428.4	397.8	367.2	336.6	306.0	275.4	244.8	214.2	183.6	153.0	122.4	91.8	61.2	30.6

- *2 crew members (double duration for 1 crew member).
- When available oxygen is less than 1 liter descend to below 10,000 feet MSL.

Figure 1-44.

Oxygen-Suit Control Panel and Diluter Demand Oxygen Regulator

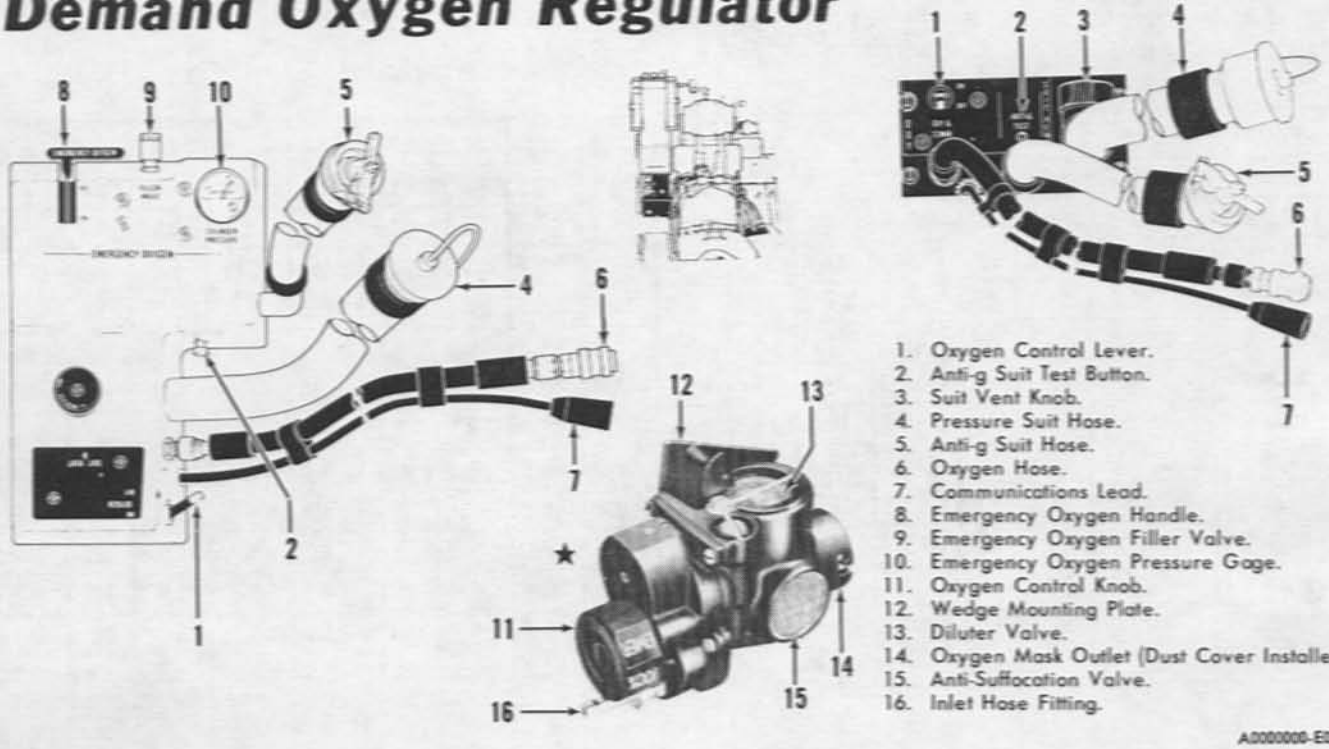


Figure 1-45.

position control knob on the regulator provides for selection of normal diluted oxygen, 100 percent oxygen and, in the event of an emergency, 100 percent oxygen under pressure.

CAUTION

Use care when connecting the oxygen regulator to the restraint harness and oxygen supply hose or when connecting the mask hose to the regulator. The valve port screens on the regulator may be easily damaged by careless or improper handling.

A quick disconnect is provided between the regulator and oxygen mask hose to expedite abandoning the aircraft on the ground. For the duration of the oxygen supply refer to figure 1-44. Refer to figure 1-84 for oxygen system servicing.

Oxygen Control Knob.

The oxygen control knob, located on the oxygen regulator (11, figure 1-45), has three positions marked NORM, 100%, and EMER (emergency). When the knob is positioned to NORM, oxygen, proportionally diluted with air, is supplied to the mask from a 0 to 35,000 altitude. With the knob in the 100% position, 100 percent oxygen is supplied to the mask. Selecting the EMER position provides 100 percent oxygen under pressure to the mask.

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 oxygen-suit control panel. The levers are located on each oxygen-suit control panel (1, figure 1-45).

WARNING

With oxygen control lever in the OFF position, it is possible to breathe ambient air through the oxygen regulator. However, breathing under this condition should be noticeably harder.

Oxygen Gage Panel

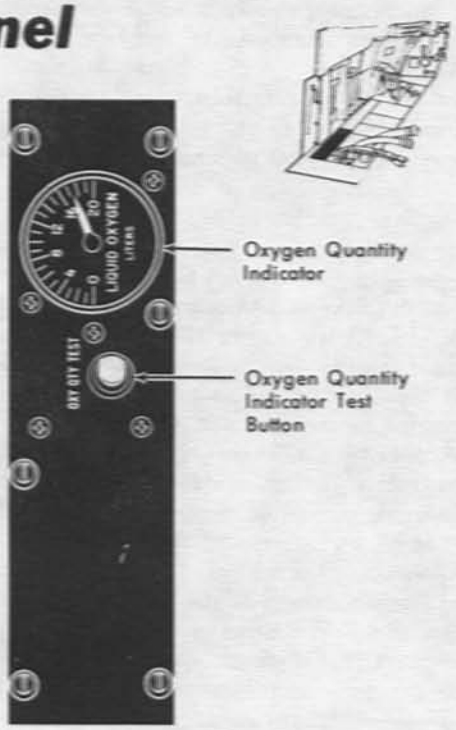


Figure 1-46.

CAUTION

To prevent damage to the regulator do not actuate the lever to ON with the regulator dust covers installed.

Oxygen Quantity Indicator.

An oxygen quantity indicator is located on the oxygen gage panel (figure 1-46), at the left crew station. The indicator is graduated from 0 to 20 liters in increments of one liter to provide the quantity of liquid oxygen in the converter. When fully serviced the indicator should indicate 10 liters. On aircraft modified by T.O. 1F-111-563 the indicator will read 15 liters when fully serviced. 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 oxygen gage panel (figure 1-46). 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 lamp will light 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 lamp panel (figure 1-37) 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 gage 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 lamp 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.

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 auomatically 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, high pressure gaseous oxygen 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 and is controlled in the normal manner. Sufficient emergency oxygen is available for 10 minutes duration at 27,000 feet cabin altitude.

Emergency Oxygen Handle.

The green emergency oxygen handle (8, figure 1-45) is located on the center oxygen-suit panel. 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. Lowering the handle will close the emergency oxygen pressure reducer, except after ejection.

Emergency Oxygen Pressure Gage.

The emergency oxygen pressure gage (10, figure 1-45), located on the center oxygen-suit control panel, 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 1400 and 2500 psi.

CREW MODULE ESCAPE SYSTEM.

The crew module (figure 1-47) 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 "Canopy" this section. The system protects the occupants from environmental hazards on either land or water and provides manual underwater escape capabilities. An emergency oxygen supply system is provided, primarily, for use during ejection. However, the system can be manually activated during normal phases of flight, as a backup to the normal oxygen system. For additional information, refer to "Oxygen System," this section.

CREW MODULE SEATS.

The crew module seats (figure 1-48) 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 a restraint harness to protect the crew member during flight in turbulence and during ejection. The harness consists of shoulder straps and lap straps and a single point harness release attached to the front of the seat by an anchor strap. The shoulder straps are attached to the inertia reel and to the seat structure to prevent them from sliding off the crew member's shoulders. The lap straps are attached to each side of the aft seat pan. The ends of the shoulder and lap straps snap into the single point release. The single point release must be rotated 90 degrees, in either direction, to release the straps. A detent at the 90 degree point will hold the release in that position. The release is spring-loaded to the center (locked) position when it is not in the detent. A wedge shaped plastic block is mounted on the right side of both types of harness to attach the oxygen regulator. 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 moved to gain access to the survival equipment compartment. Instructions for repositioning the seat are located on the back of the seat under the back cushion.

EJECTION EQUIPMENT.

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 system, release the chaff dispenser, activate guillotine cutters, ignite the rocket motor, 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 and explosive guillotine cutters. Flexible linear shaped charges are located around the crew module so that detonation will cut the splice plate joining the crew module to the aircraft. Flexible linear shaped charges are also used to remove the covers over the parachutes and 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.

Rocket Motor.

The rocket motor has both a lower and an upper nozzle. The lower nozzle provides a normal thrust of 27,000 pounds below 300 knots. To avoid excessive "g" forces to the crew members, the rocket motor is provided with two concentric upper nozzles. The small auxiliary nozzle in the center of the upper nozzle fires simultaneously with the lower nozzle to provide about 500 pounds of thrust to counteract slow speed crew module pitchup tendencies. Maximum upper nozzle thrust is achieved at aircraft speeds above 300 knots by severance of the rocket upper nozzle diaphragm. The increase in exhaust area results in reduced lower nozzle thrust (about 9000 pounds at the lower nozzle and 7000 pounds at the upper nozzle) and extended operating time. The rocket motor is located between the seat bulkhead and aft pressure bulkhead of the crew module.

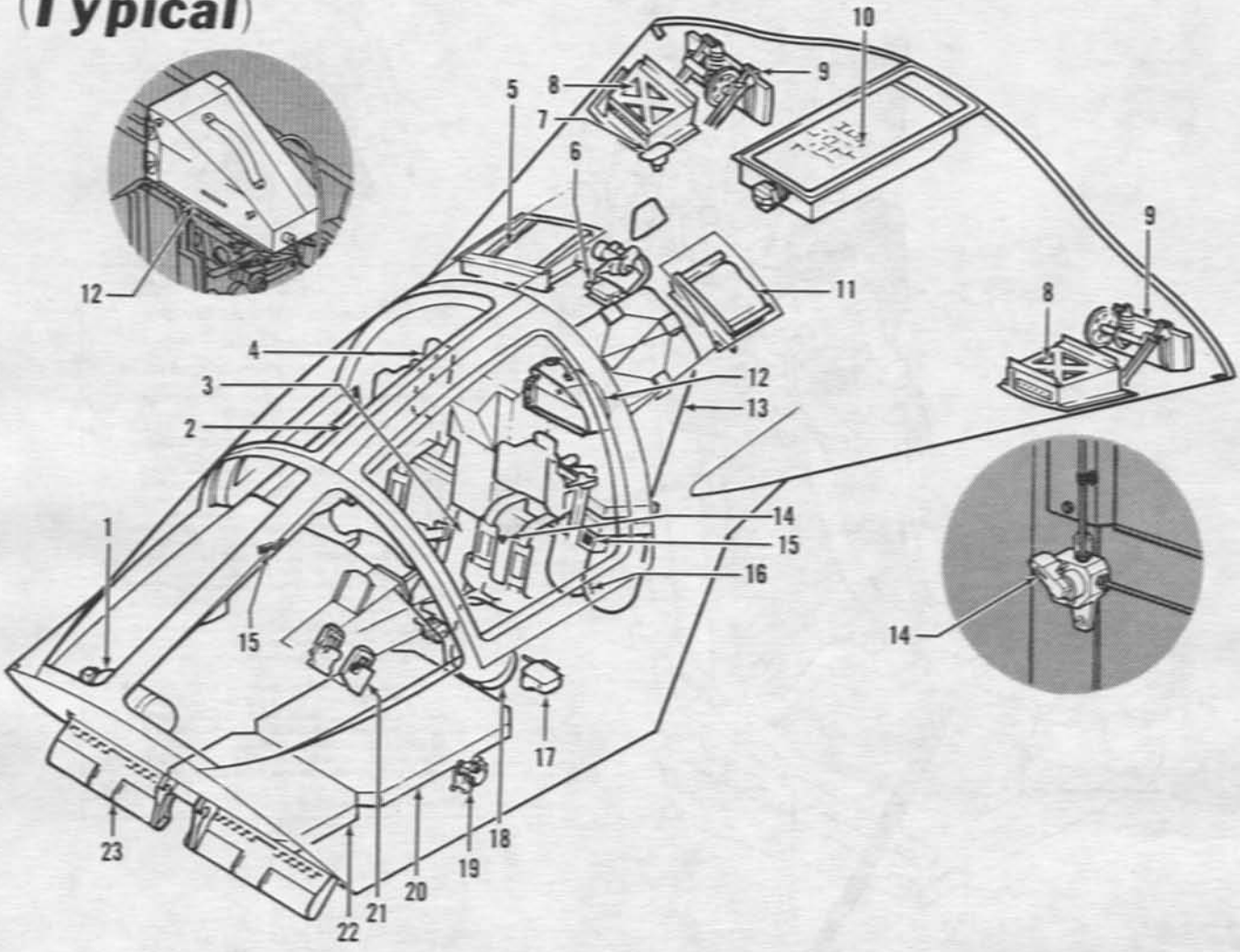
Note

After ground soak at minus 50°F or below, it is necessary to direct warm air (between 80° to 120°F) into the rocket motor compartment for 30 minutes prior to takeoff.

Dual-Mode Q-Actuated Selector.

The dual-mode "q"-actuated selector continuously senses aircraft speed and selects the appropriate time delay. The symbol "q" denotes dynamic (pitot or ram) pressure. The "q"-actuated selector allows activation of

Crew Module General Arrangement (Typical)

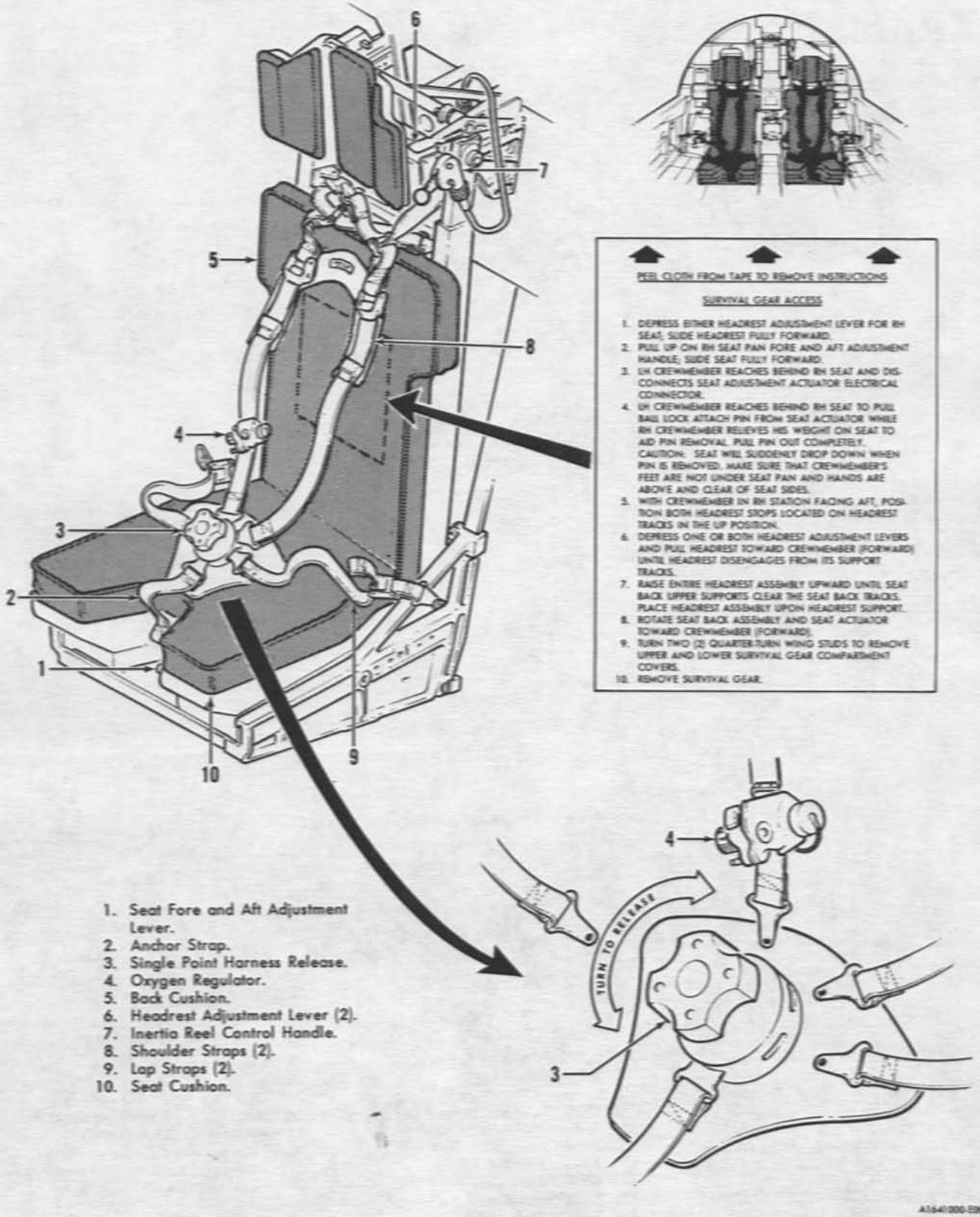


1. Auxiliary Flotation Bag Pressure Bottle.
2. Canopy Center Beam Assembly.
3. Survival Gear.
4. Emergency Oxygen Bottles (2).
5. Right Self-Righting Bag.
6. Barostat Lock Initiator.
7. Emergency UHF Antenna.
8. Aft Flotation Bag (2).
9. Pitch Flap (2).
10. Stabilization Brake Parachute.
11. Left Self-Righting Bag.
12. Quick Rescue Kit.
13. Recovery Parachute.
14. Chaff Dispenser Control Lever.
15. Radio Beacon Set.
16. Impact Attenuation Bag Pressure Bottles (2).
17. Chaff Dispenser.
18. Rocket Motor.
19. Bilge Pump.
20. Impact Attenuation Bag.
21. Ejection Handles (2).
22. Auxiliary Flotation Bag.
23. Chin Flap (2).
24. Recovery Parachute Deploy Handle.
25. Auxiliary Flotation Handle.
26. Severance and Flotation Handle.
27. Recovery Parachute Release Handle.
28. Flood Light Switch.

A0190000-E03D-F

Figure 1-47.

Crew Module Seats (Typical)



1. Seat Fore and Aft Adjustment Lever.
2. Anchor Strap.
3. Single Point Harness Release.
4. Oxygen Regulator.
5. Back Cushion.
6. Headrest Adjustment Lever (2).
7. Inertia Reel Control Handle.
8. Shoulder Straps (2).
9. Lap Straps (2).
10. Seat Cushion.

Figure 1-48.

a 1-second time delay initiator and blocks propagation to the rocket motor upper nozzle when aircraft speed is less than 300 knots. When aircraft speed is greater than 300 knots, the "q"-actuated selector blocks propagation of the 1-second time delay initiator and allows activation of shielded mild detonating cord to the rocket motor upper nozzle.

G-Sensor Initiation.

The "g"-sensor initiator consists of two operating trains. A rotating weighted arm, designated the rotor, in each explosive train is normally locked to prevent firing of the train. Firing of shielded mild detonating cord into the "g"-sensor inlet ports retracts the lock piston. Forward decelerative forces initially hold the rotors against stops. As the forces drop off, the spring-loaded rotors swing aft until they release dual firing pin. This initiates shielded mild detonating cord at the outlet ports and continues the detonation sequence to deploy the recovery parachute. The "g"-sensor initiator is located above the survival equipment compartment. A time delay fires the "g"-sensor initiator 1.6 seconds after rocket motor ignition. After the forward component of acceleration decreases to 2.2 "g's," the "g"-sensor initiator fires and activates the barostat lock initiator.

Barostat Lock Initiator.

The barostat lock initiator consists of two operating trains. An aneroid bellows of each explosive train in this initiator is normally locked to prevent firing of the train, constant cycling, and wearout. Firing of shielded mild detonating cord into the barostat inlet ports initiates an explosive charge that retracts the pins which normally lock the bellows. The aneroid bellows prevents the firing of the explosive train above approximately 15,000 feet. Below this altitude, atmospheric pressure compresses the bellows sufficiently to release the firing pins that initiate booster caps and continue the detonation sequence to remove the recovery parachute and blade antenna severable cover and fire the recovery parachute catapult. The barostat lock initiator is located on the explosive component support bracket in the rocket motor compartment.

RECOVERY AND LANDING EQUIPMENT.

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, pitch flaps and chin 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, and chin flaps under the forward 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 to decelerate the crew module and assist in maintaining stable flight prior to recovery parachute deployment. The stabilization-brake parachute is a six foot flat 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 70 foot flat 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 dynamic pressure "q" actuated selector monitors aircraft speed to select one of three possible time delays. One time delay train includes an acceleration "g" sensor initiator that actuates when crew module longitudinal acceleration drops to 2.2 "g's." Continuation of the firing train unlocks the barostat lock 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 buoyancy 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 and auxiliary flotation bag is accomplished by manually pulling the initiator handles located on the canopy center beam. The pressure source for inflation of these bags is contained in storage bottles located in the crew module. The auxiliary flotation bag is provided for use only in event of cabin flooding and, in that event, to gain additional freeboard to open canopy hatches. It should be reserved and not used unnecessarily. Also, its deployment will cause the crew module to ride higher in the water, thereby becoming more affected by wave action to the detriment of crew comfort. In the event the aircraft is ditched, the crew module can be separated from the aircraft by pulling the severance and flotation initiator handle located on the canopy center beam. Pulling this handle will sever the crew module from the aircraft, inflate the aft flotation and self-righting bags and turn on the emergency oxygen supply.

SURVIVAL EQUIPMENT.

The survival equipment consists of locating aids, a combination bilge/flotation pump, and standard survival equipment. The locating aids consist of a chaff dispenser, radio beacon set, a survival radio set, and various flares, beacons and signal mirrors. 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 radio beacon set will emit an intermittent, modulated tone to aid in rescue operations. Refer to "Communication Equipment," this section, for description of the radio beacon set. The survival radio, located in the survival equipment stowage compartment, provides a means of two way voice communication. 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 right seat. Instructions on how to gain access to the survival equipment compartment are contained on a detachable instruction plate mounted on the back of the right seat behind the back seat cushion (figure 1-48).

WARNING

When gaining access to the survival compartment, crew members should make sure that feet are not under the seat pan and hands are clear and above seat sides. Th seat will suddenly drop down when the ball lock attach pin is removed.

The contents of the survival equipment stowage compartment will be determined by the applicable using command. A small quick rescue kit containing survival equipment provided to facilitate early detection by rescue teams in hostile territory is mounted on the rear bulkhead above the left crew member head rest. The kit is readily accessible to the crew in the event situation demands that the crew abandon the module immediately after landing.

WARNING

Crew members should not carry equipment in their upper torso area. This could interfere with the operation of the restraint harness and cause injury during power retraction of the inertia reel. Also the increased loads on the crew member could result in spinal injury during ejection and landing impact. This does not preclude flying with a properly fitted life preserver.

CREW MODULE EJECTION SEQUENCE.

WARNING

Under certain conditions of crew module weight and/or tail wind, zero altitude and zero airspeed ejection capability may not be available. Because of the variables involved, ejection should not be attempted at zero altitude with less than 50 KIAS.

The ejection sequence is initiated by squeezing and pulling either ejection handle located on the center console. All succeeding functions through landing are automatically actuated by dual explosive firing trains. Emergency oxygen system actuation is automatic; however, a manually actuated backup handle is available if required. The recommended ejection posture if time permits is: head against headrest, feet on rudder pedals, and hands in lap.

WARNING

Seat prepositioning is not necessary, however, back injury, due to the shoulders contacting the headrest, may occur if the seat is not fully down and aft at the time of ejection.

After a delay of 0.35 second to allow for powered repositioning of crew members, the crew module is severed and the rocket motor is ignited. The noise of ejection will be loud but of short duration. Full thrust is sustained for approximately 1.0 second. Crew members may expect moderate ejection accelerations at low speeds becoming more severe at very high speeds. During the first six inches of crew module separation, pitch flaps and stabilization (chin) flaps rotate down into deployed position. Their function is to control crew module trim angle-of-attack and resulting spinal accelerations. The stabilization-brake parachute is deployed 0.15 second after crew module severance. This parachute provides necessary acceleration control and stabilization at speeds above 450 knots. Crew module pitch control at high speeds is provided by firing the rocket motor upper nozzle. Recovery parachute deployment is timed by a sequencing system which senses speed, acceleration, and altitude upon ejection. This system consists of three time delays (1.0, 1.6, and 4.4 seconds), a "q"-actuated selector, a "g"-sensor initiator, and a barostat lock initiator. The

4.4 second delay serves as a safety backup to the other sequencing components. At speeds below 300 knots and altitudes below 15,000 feet, the recovery parachute is deployed after a 1.0 second delay. At speeds above 300 knots and altitudes below 15,000 feet, recovery parachute deployment is controlled by a g-sensor initiator and is, thereby, delayed until crew module longitudinal (fore and aft) deceleration drops to a 2.2 g's. This allows the crew module to decelerate to below the design limit airspeed of the recovery parachute. At altitudes above 15,000 feet, deployment is constrained by a barostat lock initiator. The barostat lock initiator is armed by one of three explosive trains, whichever fires first, these are from the "q"-actuated selector, the "g"-sensor initiator, or a 4.4 second time delay initiator. After the barostat lock initiator is armed and after the crew module falls through 15,000 feet, ambient pressure compresses the aneroid bellows causing the initiator to fire. The recovery parachute is then deployed upward at 45 feet per second. Manual deployment capability, which is operable at the crew member's discretion by means of the parachute deploy handle, is provided as a backup to the automatic barostat system. The parachute deploy handle bypasses the barostat lock initiator. It should not be actuated above 15,000 feet, as read on the standby altimeter, otherwise failure of the recovery parachute may result. The initial recovery parachute inflation and the associated opening shock loads are controlled by a reefing line that holds the parachute canopy opening to about 8 feet diameter. Parachute disreefing to full inflation occurs 2.5 seconds after suspension line stretch. Whereas free-fall from maximum altitude to 15,000 feet occurs in 85 seconds, the remaining descent time after recovery parachute deployment is about 7.5 minutes. Chaff deployment as an aid to radar tracking occurs 3.0 seconds after ejection handles are actuated if the chaff control lever is in the ON position. The crew module repositions to its horizontal landing attitude and the emergency UHF antenna erects 7.0 seconds after recovery parachute deployment. A mild explosive report will be heard and a sudden raising of the nose of the crew module will occur upon repositioning. Landing impacts are absorbed by the impact attenuation bag, which is fully inflated 7.25 seconds after recovery parachute deployment. Canopy hatches may be opened during descent, but prior to landing hatches should be closed in case of overturning. If restraint harnesses are loosened, the crew member should assume ejection posture and tighten the harness prior to landing. Oxygen masks should be worn until the module has been vented of possible toxic gases. Crew members may expect moderate landing impact decelerations for the nominal weight crew module. It is recommended that the severance & flotation handle be pulled immediately after ground or water impact; this will expose the chute release handle that should be pulled to avoid being dragged or overturned in high winds.

WARNING

- Do not actuate the severance and flotation handle prior to impact. To do so will result in severe post landing gyrations, if on land or rupture the bags if on water.
- Do not actuate the severance and flotation handle when personnel are within 100 feet of the crew module because of explosive severance of metal covers.

EJECTION WITH SUSPECTED PITOT-STATIC SYSTEM FAILURE.

Ejection with a failure in the pitot static system will affect crew module performance in a number of ways.

1. If pitot-static system speed inputs are erroneous and are less than 300 KIAS the crew module "q"-actuated selector will program the ejection in the low speed mode regardless of aircraft speed. In this event at actual aircraft speeds greater than 300 knots, high spinal loading, serious structural damage and recovery parachute failure may occur. At actual speeds below 300 knots, ejection will not be affected.
2. If pitot-static system speed inputs are erroneous and are greater than 300 KIAS the crew module "q"-actuated selector will program the ejection in the high speed mode regardless of aircraft speed. In this event at actual aircraft speeds less than 300 knots, an additional 0.6 seconds will be added to recovery sequence times and the rocket motor will fire in the high speed mode resulting in degraded recovery height performance. At actual speeds above 300 knots, ejection will not be affected.

DITCHING ESCAPE SEQUENCE.

If the aircraft is ditched, crew module severance and flotation bag deployment may be initiated manually by pulling the severance and flotation handle. When the handle is pulled, the following sequence of events occurs: An initiator is fired to (1) fire the flexible linear shaped charges to separate the crew module from the aircraft, (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.

WARNING

Pulling the severance and flotation handle and the auxiliary flotation handle does not disable the rocket motor; it will still fire if either ejection handle is pulled. To preclude inadvertent firing of the rocket motor during the ditching sequence, both ejection handle safety pins must be installed.

Seat Fore and Aft Adjustment Lever.

The seat fore and aft adjustment lever (1, figure 1-48), 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 5 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.

Vertical adjustment of each seat is provided by a switch located on each left and right sidewall (2, figure 1-18 and 3, figure 1-42) 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. The headrest does not move with vertical seat adjustment.

Headrest Adjustment Lever.

A headrest adjustment lever (6, figure 1-48), 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.

The inertia reel control handle (7, figure 1-48), located on the left side of each seat headrest, is provided to lock or unlock the inertia reel.

Ejection Handles.

Two ejection handles (21, figure 1-47), 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 ½ inch will fire the initiator to start the ejection sequence.

Note

The ejection handle safety pins provided must be installed with the T-handles inboard, as shown on figure 1-16, to preclude interference of the pins with seat adjustment.

Recovery Parachute Deploy Handle.

The ring-shaped recovery parachute deploy handle (24, figure 1-47), 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 deploy the recovery parachute.

Recovery Parachute Release Handle.

The T-shaped recovery parachute release handle (27, figure 1-47), located on the canopy center beam assembly, is provided to release the recovery parachute from the crew module after landing. Pressing a release button on either side of the handle and 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.

WARNING

Do not pull the handle prior to landing as the parachute will separate from the crew module allowing the crew module to free fall.

Auxiliary Flotation Handle.

The T-shaped auxiliary flotation handle (25, figure 1-47), 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.

The severance and flotation handle (26, figure 1-47), located on the canopy center beam assembly, is provided for escape in the event the aircraft is ditched. Pressing a release button and pulling the handle out

will fire the flexible linear shaped charges and guillotines, separating the crew module from the aircraft, and will inflate the aft flotation bags and the self righting bags. Pulling the handle will also activate the emergency oxygen system. The rocket motor will not ignite in this sequence, however, it is not disabled and will still fire if either ejection handle is pulled.

Bilge/Flotation Bag Inflation Pump.

The bilge/flotation bag inflation pump (figure 1-49), provides simultaneous inflation of the flotation bags and pumping of water overboard. The pump is operated by fore and aft motion of the control stick. After landing, the bilge pump lock 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. Movement of the stick will then operate the pump.

WARNING

On aircraft modified by T.O. 1F-111-550, do not engage the bilge pump until after ejection. To do so will prevent the cabin pressure regulator from relieving cabin pressure and the canopy hatches may be blown off when they are opened during normal operation.

Chaff Dispenser Control Lever.

The chaff dispenser control lever (14, figure 1-47), located on the aft bulkhead, is used to arm or disarm the crew module chaff dispenser. The lever is labeled CHAFF and has two positions marked ON and OFF. Placing the lever to the ON position opens a mechanical interrupt to allow explosive train propagation to the chaff dispenser release mechanism. When the crew module is ejected, the explosive train releases the chaff dispenser and the slip stream dispenses the chaff. Placing the lever to the OFF position closes the mechanical interrupt, thereby disarming the dispenser.

Note

The lever should be ON over friendly territory and placed to OFF as directed by tactical requirements.

On aircraft modified by T.O. 1F-111-613, when the lever is in the ON position the radio beacon set will be automatically activated as the crew module ejects.

Bilge/Flotation Bag Inflation Pump

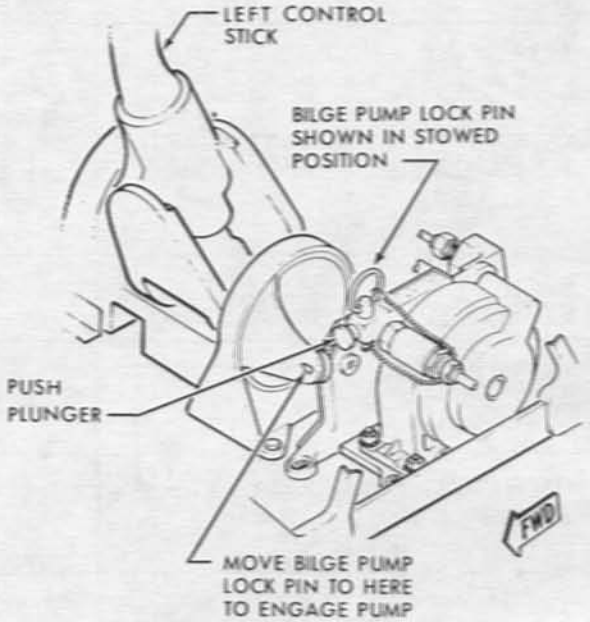


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Figure 1-49.

COMMUNICATION EQUIPMENT.

For a listing and function of communications equipment see figure 1-50.

UHF COMMAND RADIO (AN/ARC-109).

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 unit, a control panel, an antenna selector, blade type upper and lower antenna, and a loop ADF antenna. See figure 1-69 for antenna locations. There are 3500 channels available in 50 kilohertz increments in the frequency range from 225.00 to 399.95 megahertz. The receiver-transmitter unit and guard receiver are located in the right forward equipment bay. The receiver section of the receiver-transmitter unit provides ADF bearing signals to the number two pointer of the bearing distance heading indicator and audio to the interphone when the ADF function is selected. Refer to Section IV for ADF operating procedures. The guard receiver monitors the guard frequency of 243.0 megahertz when guard function is selected. The control panel allows selection of 20 preset channels and manual selection

Communications and Avionics Equipment

Type	*Designation*	*Function*	*Crew Station*	*Range*
UHF RADIO	AN/ARC-109	Air-to-air and air-to-ground voice communication	Both	Line-of-Sight
UHF ADF	AN-ARA-50	Provides bearing information to selected UHF transmissions	Both	Line-of-Sight
HF RADIO	AN/ARC-123	Air-to-air and air-to-ground long range voice communications	Right	5000 miles
RADIO BEACON SET	AN/URT-27 or –33	Provides a tone signal for rescue aircraft to home on	Right or Both	Line-of-Sight
INTERPHONE	AN/AIC-25	Interphone between crew members and monitoring of all communications facilities	Both	
IDENTIFICATION RADAR (IFF-SIF)	AN/APX-64	Provides coded IFF replies to an interrogating ground radar station	Both	Line-of-Sight
TACAN	AN/ARN-52	Provides bearing and distance information to TACAN stations	Both	Line-of-Sight up to 300 NM
ILS	AN/ARN-58	Provides visual indications for ILS approaches	Right	Localizer 45 NM Glide Slope 25 NM
RADAR ALTIMETER	AN/APN-67	Provides precise altitude measurements from 0 to 5,000 feet	Left	0-5000 feet
TERRAIN FOLLOWING RADAR	AN/APQ-110	Provides all weather, low altitude terrain following, obstacle avoidance and blind letdown capability	Left	Line-of-Sight up to 15 miles
BOMBING-NAVIGATION SYSTEM	AN/AJQ-20	Provides integrated bombing and navigation capabilities in conjunction with other systems in the aircraft	Right	
ATTACK RADAR	AN/APQ-113	All weather navigataion, fix-taking, bombing, and air-to-air attack	Right	Line-of-Sight up to 160 miles
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	Left	Line-of-Sight

Figure 1-50.

of any frequency in the frequency range of the radio. The upper and lower antenna complement 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.

Note

When operating the UHF in the automatic antenna selection mode, the antenna selector has a transmission memory circuit which automatically connects the transmitter to the antenna last used for transmission. If the channel or frequency is changed to another station or the aircraft position has changed relative to the station, this may be the wrong antenna for the next transmission and difficulty will be encountered in gaining contact. Should this occur, manually select the upper or lower antenna and repeat the transmission to gain contact.

The antennas also serve the TACAN. 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 main dc bus.

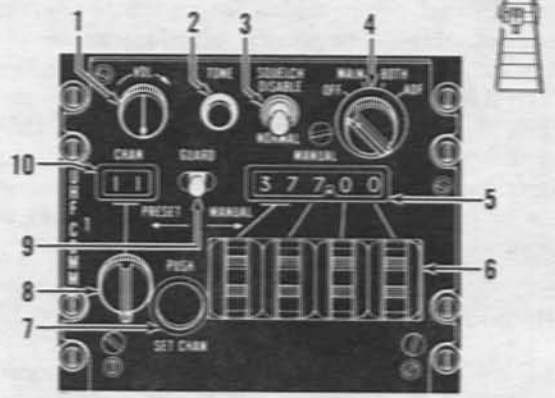
UHF Radio Function Selector Knob.

The UHF radio function selector knob (4, figure 1-51), located on the UHF radio control panel, has four positions marked OFF, MAIN, BOTH, and ADF. Rotating the knob to the MAIN position activates the receiver-transmitter unit for normal transmission and reception on the channel selected; the guard receiver is inoperative. Rotating the knob to the BOTH position also activates the receiver-transmitter 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 bearing distance heading indicator and interphone respectively. Audio range is reduced considerably when operating on the ADF antenna and it may be necessary to return the knob to MAIN or BOTH for better reception. 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 antenna is disabled until the microphone switch is released.

Note

The ADF position of the switch is inoperative when operating on emergency electrical power.

UHF Radio Control Panel



1. Volume Control Knob.
2. Tone Button.
3. Squelch Switch.
4. Function Selector Knob.
5. Frequency Indicator Window.
★ 6. Manual Frequency Selector Knobs (4).
7. Channel Set Pushbutton.
8. Preset Channel Selector Knob.
9. Mode Selector Switch.
10. Preset Channel Indicator Window.

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Figure 1-51.

UHF Radio Mode Selector Switch.

The three position UHF radio mode selector switch (9, figure 1-51), located on the UHF radio control panel, permits selection of the desired operating mode. The switch is marked PRESET, GUARD and MANUAL. The PRESET position is used when selecting one of the 20 preset frequencies. The MANUAL position is used when utilizing frequencies that are selected by the manual frequency selector knobs. The GUARD position tunes the main receiver-transmitter unit to the guard frequency of 243.0 megahertz.

UHF Radio Preset Channel Selector Knob.

The preset channel selector knob (8, figure 1-51), 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. The number of the channel selected is displayed in a window above the knob. Frequencies for each channel are written on a channel frequency log located under the glare shield on the right side. Frequencies of the preset channels can be changed during flight.

UHF Radio Manual Frequency Selector Knobs.

Four thumb actuated UHF radio manual frequency selector knobs (6, figure 1-51), located on the UHF radio control panel, are provided for manually selecting frequencies. Manual frequency selection can be made in steps of 50 kilohertz from 225 through 399.95 megahertz. The first selector knob will select the first two digits of the desired frequency. The second, third and fourth knobs select the third, fourth and fifth frequency digits respectively. 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 (1, figure 1-51), located on the UHF radio control panel, increases the volume of the receiver when turned clockwise and decreases it when turned counterclockwise.

UHF Squelch Switch.

The squelch switch (3, figure 1-51), located on the UHF radio control panel, is a two-position switch marked DISABLE and NORMAL. 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 DISABLE turns off the squelch. Placing the switch to NORMAL turns the squelch on.

UHF Tone Button.

The tone button (2, figure 1-51) is located on the UHF radio control panel. With the UHF radio in operation, depressing the button will interrupt reception and transmit a continuous wave (CW) 1000 hertz tone signal on the selected frequency.

UHF Channel Set Pushbutton.

The channel set pushbutton (7, figure 1-51), located on the UHF radio control panel, is used to set or change preset channel frequencies. The button is only effective when the mode selector switch is in the PRESET position. With the mode selector switch in the PRESET position and with the preset channel selector knob set to the desired channel, depressing the button will set the frequency selected in the manual frequency window into the desired channel. The button is recessed in a guard to prevent inadvertent actuation.

Transmitter Selector Knobs.

Two transmitter selector knobs (3, figure 1-53), labeled 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 pivot type microphone switch, marked TRANS and INPH with an unmarked off position, is located on each right throttle (5, figure 1-4). The top position actuates the TRANS position for radio transmission, and the bottom position actuates the INPH for interphone operation. The INPH position is in parallel with the INT monitor knob on the interphone panel. Each actuated position is spring loaded to the off position.

UHF Radio Antenna Selector Switch.

The three position UHF radio antenna selector switch (1, figure 1-58), 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 1-51), located on the UHF radio control panel, indicates the manual frequency selected for transmission or receiving. The window has five digits, which are set by frequency selector knobs below the window.

HF RADIO (AN/ARC-123).

Aircraft 29, 32, 34 ▸ and those modified by T.O. 1F-111-863, are equipped with an HF radio. 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: SSB, single side band; AME, amplitude modulation equivalent, and FSK, frequency shift keying. The FSK position is inoperative at this time. There are 280,000 channels available in 100 Hz increments in the frequency range of 2,000 through 29,999.9 kilohertz. Components of the system include a receiver-transmitter unit, amplifier power supply, control panel, antenna, coupler, antenna coupler control, and remote capacitor. The receiver-transmitter 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 remote capacitor is located at the forward tip of the dorsal fin. Refer to figure 1-69 for antenna location. The antenna is impedance matched to the receiver-transmitter. The system incorporates self test features for maintenance trouble-shooting. The control panel is located at the navigator's station on the right console. Once the

system is placed in operation either crew member can use the equipment. Refer to Section IV for HF radio operation. The radio operates on 115 volt ac power from the right main ac bus.

HF Radio Mode Selector Knob.

The HF radio mode selector knob (7, figure 1-52), located on the HF radio control panel has four positions marked OFF, SSB, AME and FSK. In the OFF position the system is deenergized. Rotating the knob to SSB, provides single side band capability of operation. The AME position provides the capability of amplitude modulation equivalent capability of operation. The FSK position is inoperative at this time.

HF Radio Frequency Selector Knobs.

Six HF radio frequency selector knobs (1, figure 1-52), located on the HF radio control panel, provide a means of setting desired frequencies. Each knob has an indicator line drawn to the window it controls on the frequency indicator. To prevent selectin of frequencies below 20 megahertz, the 10 megahertz and 1 megahertz knobs are interlocked, and in order to select a 1 or 0 on the 1 megahertz frequency indicator, a 1 or 2 must be present on the 10 megahertz frequency indicator.

HF Radio Frequency Indicator Window.

The HF radio frequency indicator window (3, figure 1-52), located on the HF radio control panel, has six digits indicating the frequency selected for transmission or receiving. Each window has an indicator line drawn to its corresponding frequency selector knob.

HF RF Gain Control Knob.

The radio frequency gain control knob (6, figure 1-52), located on the HF radio control panel is labeled RF GAIN. The knob is used to adjust the radio frequency gain of the receiver as desired during normal operation, to provide better signal to noise ratio.

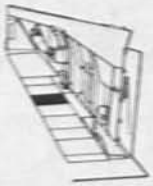
Note

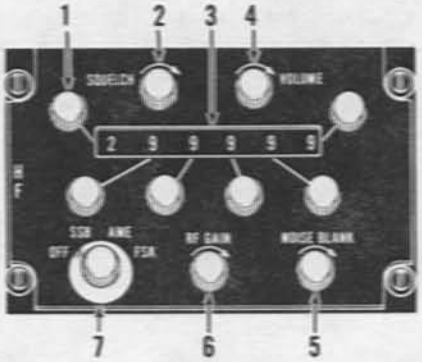
The RF GAIN control knob should not be used for adjusting the audio volume.

HF Noise Blanking Control Knob.

The noise blanking control knob (5, figure 1-52), located on the HF radio control panel is labeled NOISE BLANK. The knob is used to reduce the effects of impulse noise, either natural or man-made, including ECM.

HF Radio Control Panel





1. Frequency Selector Knobs (6).
2. Squelch Control Knob.
3. Frequency Indicator Window.
4. Volume Control Knob.
5. Noise Blanking Control Knob.
6. RF Gain Control Knob
7. Mode Selector Knob.

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Figure 1-52.

Note

Adjustment of the noise blanking control knob is not critical. If high impulse noise is encountered, rotation of the knob will blank the noise without affecting voice signals.

HF Volume Control Knob.

The volume control knob (4, figure 1-52), located on the HF radio control panel, is labeled VOLUME. When the knob is rotated clockwise the audio output is increased.

Note

- For most applications the volume control knob should be set to obtain approximately the same audio level as the UHF to balance the audio out of the interphone. After this balance has been obtained, any additional audio volume corrections should be made with the interphone volume controls.
- Care should be exercised in the use of this control as audio can be disabled if this control is set too far counterclockwise.

HF Squelch Control Knob.

The squelch control knob (2, figure 1-52), located on the HF radio control panel, is labeled SQUELCH and is used to adjust the threshold of squelch operation in the SSB or AME mode. Adjustment of the squelch must be done after the other controls have been set. First place the squelch control fully clockwise. Then with no signal on the channel, turn the squelch control counterclockwise slowly until the noise is subdued. Rotate the control knob slowly clockwise and find the place where the set breaks in and out of squelch. This is the proper squelch adjustment for the prevailing noise conditions. If the noise conditions change, the squelch must be readjusted. If desired signals are found to break in and out of squelch, adjust to a higher clockwise position or operate without squelch.

Note

Care must be exercised in the use of this control as the receiver audio can be disabled if the squelch is set too far counterclockwise.

RADIO BEACON SET.

A radio beacon set, located on the right console, is provided for use as a survival radio to aid in crew rescue after ejection. The radio operates on a self-contained battery. The set is connected to a crew module mounted antenna which is automatically erected when the crew module ejects. An on-off switch on the face of the radio is provided to arm the set. A safety plug located adjacent to the on-off switch must be removed to place the set in operation. When the switch is positioned to ON and the safety plug removed, the radio will transmit an intermittent modulated tone signal for the rescue aircraft to home on. The radio can also be removed from the console and used as a portable rescue aid. A telescoping antenna, stowed in the radio, can be extended when the radio is used as a portable. On aircraft modified by T.O. 1F-111-613 the radio is located behind the left seat. When the on-off switch is placed to ON and the chaff control lever is ON, the radio will be automatically actuated whenever ejection occurs.

INTERPHONE (AN/AIC-25).

The interphone provides the following functions: Communications between crew members and between crew members and ground crew; monitoring and volume control UHF radio, HF radio, TACAN, ILS, RHAW and missile tone reception, and hot mic and call capability. Two identical interphone control panels (figure 1-54) located on the left and right consoles are provided for each crew member. Interphone stations for ground crew operation are located in the nose wheel well, main landing gear well and ground power receptacle. The interphone operates on 28 volt dc power from the essential dc bus. Power is applied to the interphone whenever power is on the aircraft.

Monitor Knobs.

Eight push-pull monitor knobs (1, figure 1-53), 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
TACAN	— TACAN Identification
RHAWS	— Radar Homing and Warning System
MISSILE	— Missile Tones
HOT MIC LISTEN	— Hot Mic Transmissions

Other signals fed to the interphone panel are the landing gear warning signal and, on aircraft modified by T.O. 1F-111-891, the stall warning signal. 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.

Interphone Control Panel

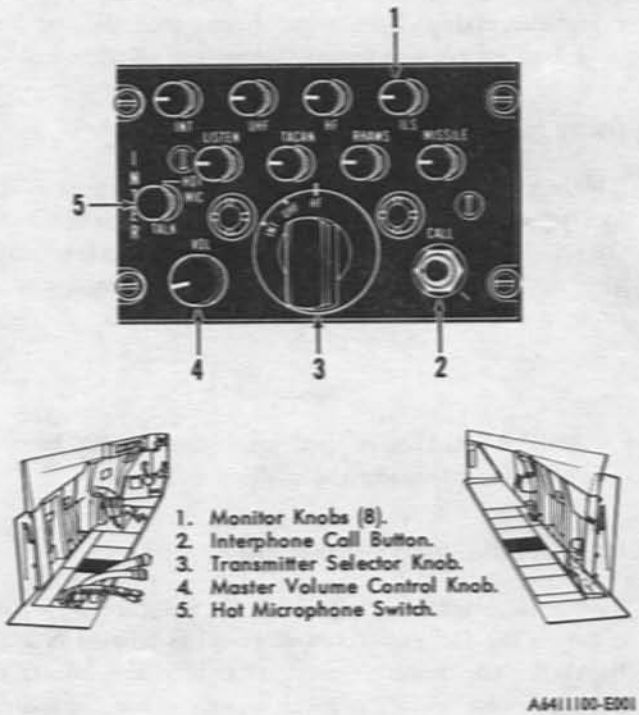


Figure 1-53.

Master Volume Control Knob.

A master volume control knob (4, figure 1-53), 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 Talk Switch.

A push-pull (HOT MIC) hot microphone talk switch (5, figure 1-53), located on each interphone control panel, provides a continually operating microphone when it is pulled. When this switch is pulled, the crew member can transmit without using the microphone switch, however, the crew member at each station must have his hot mic listen monitor knob pulled to receive the transmission.

Note

The hot mic talk signal is subject to cross talk from all head set signals being monitored. Also, high background noise in the cockpit will interfere with reception when using hot mic. To eliminate this problem, the microphone switch at each crew station should be used instead of hot mic talk.

Interphone Call Button.

The interphone call button (2, figure 1-53), located on the interphone control panel, permits either crew member to call the other crew member or the ground crew. Depressing either call button boosts the interphone volume of the other stations and reduces the operator's side tone level, allowing the call signal to override the other station's reception. The call signal will override the reception at the other station regardless of the position of the communications monitor knob or transmitter selector knobs at either station.

Transmitter Selector Knobs.

A three position transmitter selector knob (3, figure 1-53), located on each interphone control panel, is provided to select either UHF or HF radio. 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 to the TRANS position. In addition, the UHF or HF position will allow continuous monitoring of the respective receiver (UHF or HF) regardless of the position of the communications monitor knobs. Regardless of the position of the transmitter selector switch, the interphone may be used by moving the microphone switch on the throttle to the INPH position. The INT position of the transmitter selector switch has no operational function.

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. Aircraft modified by T.O. 1F-111-677 have an additional exterior interphone station installed in the ground power receptacle.

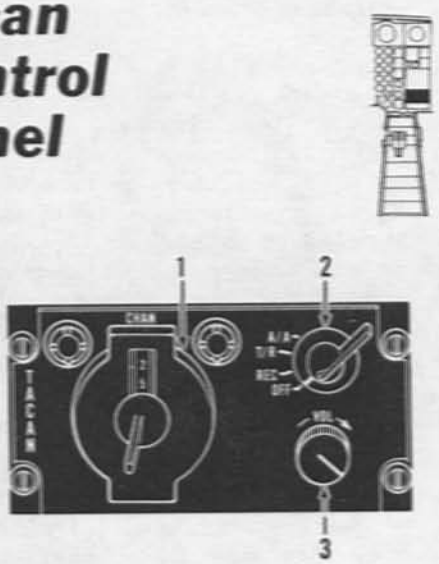
TACTICAL AIR NAVIGATION SYSTEM (AN/ARN-52).

The tactical air navigation system (TACAN) enables the aircraft 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 1-69), 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. When TACAN is selected on the instrument system coupler all heading information is supplied by the AFRS. For all other instrument system coupler selections (except AIR/AIR), TACAN bearing information is still provided by auxiliary flight reference system but all other integrated flight instrument system heading data is supplied by the bomb nav system. The system operates on 28 volt dc from the main dc bus and 115 volt ac from the left main ac bus. The TACAN control panel is located on the right main instrument panel. Refer to Section IV for TACAN operating procedures.

TACAN FUNCTION SELECTOR KNOB.

The function selector knob (2, figure 1-54), 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

Tacan Control Panel



1. Channel Selector.
2. Function Selector Knob.
3. Volume Control Knob.

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Figure 1-54.

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 to and receive 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). This was designed to provide distance information for up to five aircraft homing to a single aircraft.

TACAN CHANNEL SELECTOR.

The channel selector (1, figure 1-54), 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 digit(s) of the desired channel and the inner control to select the last digit.

TACAN VOLUME CONTROL KNOB.

A volume control knob (3, figure 1-54), 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 1-58), 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.

INSTRUMENT LANDING SYSTEM.

The instrument landing system (ILS) provides the capability of making instrument 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 megahertz 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 aircraft. The ILS operates on 28 volt dc power from the 28 volt dc main bus. Refer to "Instrument Flight Procedures," Section VII for instrument landing system operating procedures. Refer to figure 1-34 for ILS indications in the various knob positions.

ILS FREQUENCY SELECTOR KNOB.

The frequency selector knob (2, figure 1-55), located on the ILS control panel, allows individual selection of

20 ILS channels ranging in localizer frequencies from 108.1 to 111.9 megahertz in 0.2 megahertz 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 megahertz. The frequency of each channel selected is displayed in a digital window to the left of the knob.

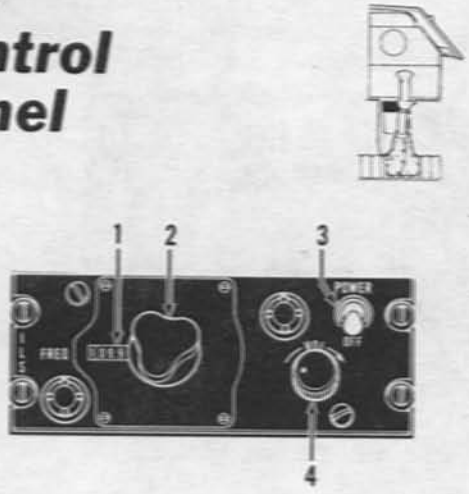
ILS POWER SWITCH.

The power switch (3, figure 1-55), 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 1-55), located on the ILS control panel, adjusts the volume of the localizer station identification signal. Clockwise rotation increases volume.

ILS Control Panel



1. Frequency Window.
2. Frequency Selector Knob.
3. Power Switch.
4. Volume Control Knob.

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Figure 1-55.

MARKER BEACON LAMP.

The marker beacon lamp (18, figure 1-5), located on the left main instrument panel, provides a visual station signal when the aircraft is over a marker beacon facility. When lighted the words MARKER BEACON are displayed in green.

Note

The marker beacon lamp may blink during HF radio transmission on some frequencies due to electromagnetic interference. This is not a malfunction of either the marker beacon or the HF radio and should be no cause for alarm.

IFF SYSTEM (AN/APX-64). (AIMS)

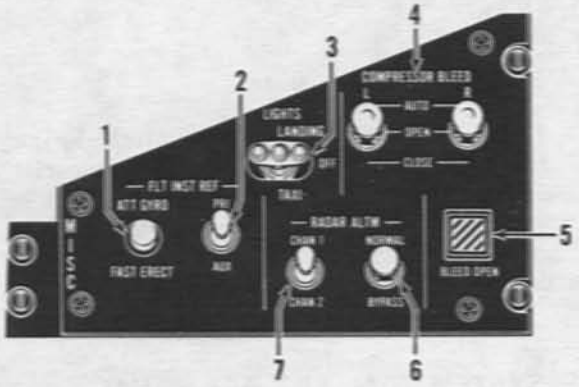
Note

AIMS includes the features of and is derived from:

- Air traffic control radar beacon system (ATCRBS)
- IFF (SIF)
- MK 12 IFF
- System

The air-to-ground IFF system provides for Mark X IFF with selective identification feature (SIF), automatic altitude reporting, and Mark XII (mode 4) encrypted IFF. Operation is possible in any one of five 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, Mode 3/A—Traffic Identity, Mode 4—Encrypted Identity and Mode C—Altitude Interrogation. The equipment consists of an IFF control panel, a transmitter-receiver, a mode 4 computer, an antenna lobing switch, and two radiator-type antennas. 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 1-64 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 aircraft 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. Mode 2 and 4 code settings are set into the

Miscellaneous Switch Panel



1. AFRS Gyro Fast Erect Button.
2. Flight Instrument Reference Select Switch.
3. Landing and Taxi Light Switch.
4. Compressor Bleed Valve Control Switches.
5. Compressor Bleed Valve Position Indicator.
6. Radar Altimeter Bypass Switch.
7. Radar Altimeter Channel Selector Switch.

A0000000-E002

Figure 1-56.

receiver-transmitter on the ground and thus are fixed for any one flight. Mode 1 and 3/A codes are set up at the control panel. All modes can be turned on or off at the control panel. Replies to modes 1, 2, 3/A, 4 and C interrogations, as well as to I/P and emergency replies, are shown on the ground station equipment. The system will go to code 7700 in mode 3/A when operating in the emergency mode. 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 aircraft conditions. The mode C provides altitude information from the CADC to the ground in 100 foot increments. An optional low sensitivity setting provision restricts sensitivity so that replies are made only to local interrogations. Refer to Section IV for IFF operating procedures. Electrical power is supplied to the IFF system from the 115 volt ac essential bus and the 28 volt dc essential bus.

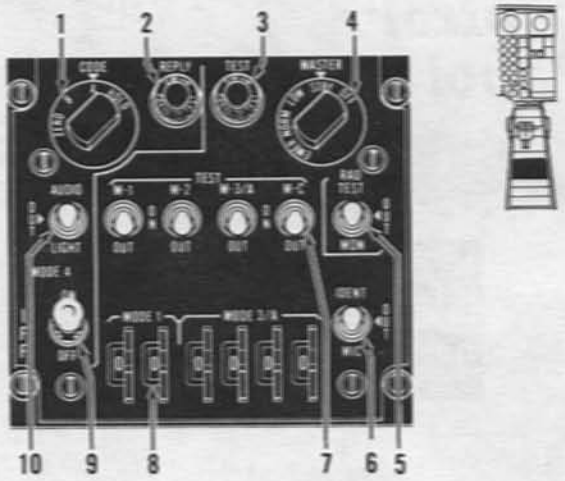
IFF MASTER CONTROL KNOB.

The five-position IFF master control knob (4, figure 1-57) is located on the IFF control panel. The knob positions are marked OFF, STBY, LOW, NORM and EMER. The OFF position removes power from the set and also zeroizes mode 4 code settings. When positioned to STBY, 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 reply occurs. Transmitted power from the IFF system is the same for both the LOW and NORM positions. The knob must be pulled outward to position it to EMER. When the knob is positioned to EMER, an emergency-indicating pulse group is transmitted each time a mode 1, 2, or 3/A interrogation is recognized.

IDENTIFICATION-OF-POSITION SWITCH.

The identification-of-position (I/P) switch (6, figure 1-57), located on the IFF control panel, is used to control transmission of I/P pulse groups. The switch has three positions marked MIC, OUT, and IDENT. When the switch is momentarily held in the spring-loaded IDENT position, the I/P timer is energized for 15-30 seconds. If a mode 1, 2, or 3/A interrogation is recognized within this 15-30 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

IFF Control Panel



1. Mode 4 Code Control Knob.
2. Reply Lamp.
3. Test Lamp.
4. Master Control Knob.
5. Rad Test/Monitor Switch.
6. Identification-of-Position Switch.
7. Mode Select/Test Switches (4).
★ 8. Code Selector Wheels(6).
9. Mode 4 Control Switch.
10. Mode 4 Monitor Control Switch.

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Figure 1-57.

in reply to a mode 1 or 3/A interrogation as long as a microphone switch is held to the TRANS position and for 15-30 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.

IFF ANTENNA SELECTOR SWITCH.

The two-position antenna selector switch (3, figure 1-58), located on the antenna select panel, is marked AUTO and LOWER. When the switch is placed to AUTO, the antenna lobing switch rapidly cycles contact of the receiver-transmitter between the upper and lower antenna to provide thorough antenna pattern coverage. When the antenna selector switch is placed to LOWER only, the lower antenna will be used to receive and reply to interrogation signals.

MODE SELECT/TEST SWITCHES.

Four mode select/test switches, (7, figure 1-57), located on the IFF control panel, are marked TEST, ON and OUT. The switches are labeled M-1, M-2, M-3/A and M-C from left to right to correspond to mode 1, mode 2, mode 3/A and mode C. The OUT position for each switch disables the transmitter-receiver for the mode selected. The ON position for each switch enables the transmitter-receiver to reply to interrogations for the mode selected. If more than one switch is placed to ON the transmitter-receiver will reply to interrogations for all modes selected. The switches are spring-loaded to the ON position from the TEST position. The TEST positions are inoperative at this time.

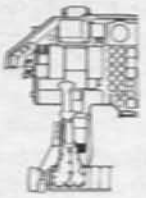
RAD TEST/MONITOR SWITCH.

The three position rad test/monitor switch (5, figure 1-57), located on the IFF control panel, is used for control of the radiation—test and monitor provisions. The switch has three positions marked RAD TEST, OUT and MON. The switch is spring-loaded from the RAD TEST to the OUT position. The MON position and in-flight test capability is inoperative at this time. When the switch is placed to OUT, the radiation test and monitor circuits are inoperative. The RAD TEST position is used for preflight check of the equipment.

CODE SELECTOR WHEELS.

Two sets of thumb actuated code selector wheels (8, figure 1-57), located on the IFF control panel, are provided to set mode 1 and mode 3/A codes. The set of wheels labeled mode 1, consists of two wheels which allow selection of 32 different codes. The set of wheels labeled mode 3/A consists of four wheels which provide the capability of setting 4096 codes. Code digits on each wheel are read in windows recessed in the face of the panel.

Antenna Select Panel



1. UHF Radio Antenna Selector Switch.
2. TACAN Antenna Selector Switch.
3. IFF Antenna Selector Switch.

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Figure 1-58.

MODE 4 CONTROL SWITCH.

The mode 4 control switch (9, figure 1-57), located on the IFF control panel, has two positions marked ON and OUT. Mode 4 operation is enabled by placing the switch to ON. Placing the switch to OUT disables mode 4 operation. The switch toggle must be pulled out in order to move the switch between the ON and OUT positions.

MODE 4 CODE CONTROL KNOB.

The mode 4 code control knob (1, figure 1-57), located on the IFF control panel, has four positions marked ZERO, A, B, and HOLD. The knob must be pulled out before it can be moved to the ZERO posi-

tion, and is spring-loaded from HOLD to the A position. Positions A and B select the preset code for the present and succeeding code periods, respectively. Placing the knob to ZERO will zeroize both code settings if the master control knob (4, figure 1-57) is in any position except OFF. Both codes will be automatically zeroized when the IFF is turned off after landing. However, both code settings can be retained by momentarily holding the knob in the spring-loaded HOLD position prior to turning the IFF off. The HOLD function is operative only when the landing gear handle is in the DN position, and requires that system power remain on for at least 15 seconds after HOLD is selected to allow mechanical latching of the code settings.

MODE 4 MONITOR CONTROL SWITCH.

The mode 4 monitor control switch (10, figure 1-57), located on the IFF control panel, has three positions marked AUDIO, OUT, and LIGHT. In the AUDIO position, monitoring of mode 4 interrogations and replies is provided by an audio tone on the interphone and by illumination of the reply lamp on the IFF control panel. The audio tone is controlled by the UHF-2 mixer switch on each communications panel. Placing the switch to LIGHT switches out the audio tone and provides monitoring only by the reply lamp. In the OUT position, both the audio tone and the reply lamp are inoperative.

REPLY LAMP.

The reply lamp (2, figure 1-57), located on the IFF control panel, lights to indicate mode 4 replies. This lamp is operative only when the mode 4 monitor switch is in AUDIO or LIGHT.

IFF CAUTION LAMP.

The IFF caution lamp is located on the main caution lamp panel (figure 1-37). The lamp will light whenever an inoperative mode 4 capability is detected, provided the mode 4 computer is installed in the aircraft and the master control knob is not in the OFF position. Specific discrepancies monitored by the IFF caution lamp are:

- Mode 4 codes zeroized
- Failure of the system to reply to proper interrogation
- Automatic self-test function of the mode 4 computer reveals a faulty computer

The letters IFF are visible on the lamp when lighted.

TEST LAMP.

The test lamp (3, figure 1-57), located on the IFF control panel, is used for functional testing and trouble shooting of the IFF equipment.

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

The bombing-navigation (bomb nav) system is a self-contained dead reckoning analogue inertial system. The system consists of a stabilized platform (SP), a navigation computer (NC), a remotely located flux valve, a remotely located bomb nav distance time indicator (BNDTI), and an automatic ballistics computer unit (BCU). 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.
- An inertial true heading plus handset magnetic variation to the pilot's flight instruments.
- Automatic steering signals to the autopilot, 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 offset radar sighting.
- Position correction via radar fix-taking.
- Determination of the coordinates of unknown radar locations detected by the homing and warning radar.
- Pushbutton over fly 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.
- Glide path or dive angle deviation steering signals for use in making airborne instrument landing approaches or dive bomb runs.
- A backup capability in case of stabilized platform failure.
- Automatic computation of ballistic parameters for selected weapons and prevailing attack attitude, airspeed and altitude for accurate weapon delivery under tactical conditions.
- Simple self-test features to isolate system troubles to the stabilized platform, navigation computer or ballistics computer while the system is still installed in the aircraft.

STABILIZED PLATFORM.

The stabilized 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, vertical acceleration, 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 the following methods:

- Gyrocompass alignment (the normal method).
- Alignment to magnetic variation.
- 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 methods of alignment are rapid alignment modes and may be used when the correct aircraft true heading has been stored in a navigation computer (NC) from a previous operation or when the correct magnetic variation is known. These modes bypass the relatively slow gyrocompass process to determine true north, and brings the SP to a ready condition in less than ¼ the time required for gyrocompass alignment (refer to "Principles of Operation," this section). 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. An accelerometer mounted in the vertical plane will sense only vertical acceleration. The vertical accelerometer output is supplied to the ballistics computer for use in continuously updating ballistics parameters. The north and east accelerometer 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 computed aircraft position information. Critical signal loops within the SP are constantly monitored by a go-no-go circuit which, for most of the likely failures, turns the SP off, lights the platform error indicator lamp on the bomb nav control panel, and results in automatic transfer to the AFRS.

CAUTION

The go-no-go circuit is designed to light the platform error lamp in event of a platform malfunction; however, since some failures of the SP and related circuits are not automatically detected, other cockpit indications should be monitored throughout flight to ascertain proper SP operation. Refer to "Stabilized Platform Malfunction Analysis," this section.

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, 28 volt dc mode control signals from the NC, and synchro analogue data from the NC corresponding to aircraft latitude.

NAVIGATION COMPUTER.

The navigation 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 stabilized 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 last computed wind 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.

BALLISTICS COMPUTER.

A ballistics computer unit (BCU), located in the right equipment bay, contains the electronics and electromechanical computing components to automatically compute weapon trail and time of fall. LCOS pipper azimuth and elevation positioning signals are also furnished to continuously sight on weapon impact point for the existing conditions of airspeed, altitude, attitude, wind and ground-speed. Ballistics computations are accomplished only when the nav computer mode selector knob is in VISUAL CCIP or AUTO BOMB. The computer utilizes pre-set weapon loading data in conjunction with inputs of indicated airspeed, true airspeed and pressure altitude rate from the CADC; true heading, vertical acceleration, east groundspeed, pitch and roll from the SP; radar range from the attack

radar; burst lead or height data from the burst control panel at the pilot's station, groundspeed, altitude and drift data from the NC; and time delay data from the dual bombing timer. The preset weapon loading data for each weapon station is programmed into the computer by hand-set controls located on the ballistics selection panel on the front of the computer. Computer operation is controlled by the bomb-nav mode selector knob. With the knob in the OFF or HEAT positions power to the computer is off. Computer power is on in all other positions.

CONTROLS AND INDICATORS.

Bomb Nav Mode Selector Knob.

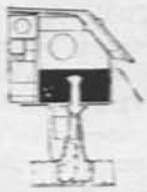
The eleven position bomb nav mode selector knob (13, figure 1-59), located on the bomb nav control panel is labeled MODE SEL. The knob controls warmup, power turn on, and system operating modes. The HEAT and ALIGN positions set the system up for operation. The GREAT CIRCLE through RANGE BOMB/MAN BAL positions provide primary navigation. The auxiliary position is labeled AUX NAV CHECK and allows checks of the computer in the aux nav mode of operation. The knob must be rotated counterclockwise to turn the system off. This knob has detents at all positions and requires a pullout to rotate from ALIGN to any normal navigation position and from the normal navigation range to the bomb modes. The knob markings and functions are as follows:

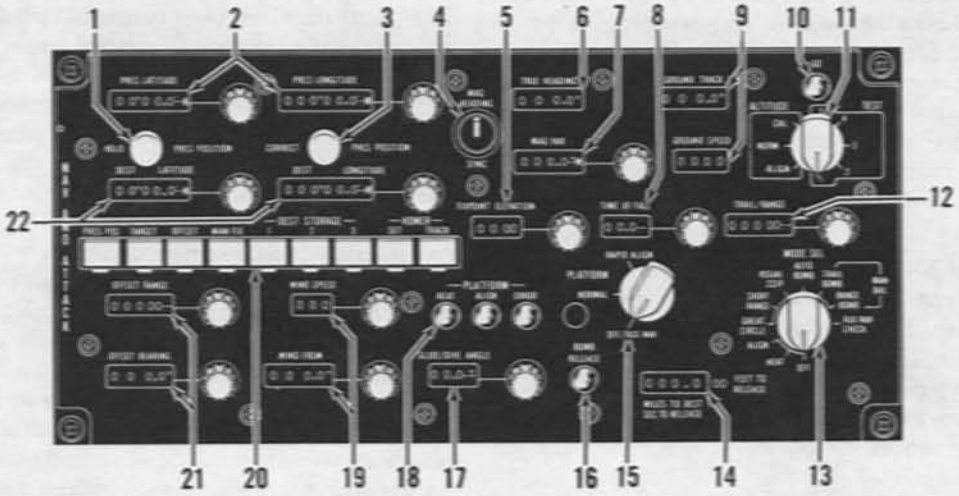
1. OFF—All power off.
2. HEAT—All power off except for inertial platform heater power—provided the platform alignment control knob is in any position except PLATFORM OFF.
3. ALIGN—All power on and stabilized platform sequenced through alignment cycle—provided the platform alignment control knob is in any position except OFF/AUX NAV. Navigation computer is operative.
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 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 operative.
6. VISUAL CCIP (Continuously computed impact point)—This mode activates the BCU, which then supplies continuously updated pipper azimuth and elevation positioning signals to the LCOS for use in the DIV BOMB and LEV BOMB modes of LCOS operation to effect a visual CCIP attack. The azimuth and elevation signals are continuously updated for the selected weapon and prevailing airspeed, altitude, attitude, vertical velocity, groundspeed, wind and drift conditions so as to continuously sight the LCOS pipper at the point where the weapon will impact when released. The NC operates in a non-attack mode equivalent to SHORT RANGE, with no provisions for automatic bomb release, therefore, weapon release must be made with weapon release button on the control stick.
7. AUTO BOMB—The BCU is activated, which then supplies the LCOS with the same signals supplied in VISUAL CCIP. The NC is also supplied with continuously updated trail and time-of-fall signals to effect a trail bomb type of attack. The NC uses the updated signals instead of the data set into the time of fall and trail/range counters; operations are otherwise identical to trail bomb.
8. TRAIL BOMB/MAN BAL—The NC is utilized in the short range configuration to develop the bombing equation and the radar sighting computer is 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 and is compensated for wind existing at release altitude.
9. RANGE BOMB/MAN BAL—The NC operates in the basic short range mode, continuously displaying the range to the release point but does not compensate for wind existing at release altitude. The bomb release lamp lights when the preselected release range is reached. The radar sighting computer is operative.
10. AUX NAV CHECK—Permits operation in the aux nav mode while the SP is on and places the NC in a configuration of short range/aux nav mode. This switch position is provided for check purposes only. Operation in the aux nav mode is possible only when the SP turns itself off automatically or is turned off manually by placing the platform alignment control knob to OFF/AUX NAV.

Bomb Nav Fix Mode Selector Buttons.

Nine bomb nav fix mode selector pushbuttons (20, figure 1-59) 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

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. Windspeed and Wind From Counters.
20. Bomb Nav Fix Mode Selector Buttons
21. Offset Range and Offset Bearing Counters (2).
22. Destination Position Counters.

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Figure 1-59.

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. If all buttons are out (no mode engaged), the system automatically reverts to TARGET. 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. The pushbutton is not effective if the bomb-nav mode selector knob is in the AUTO BOMB, TRAIL BOMB or RANGE BOMB positions.
2. TARGET—Selects position set in the destination position counters as the radar sighting point and the navigation (or bombing) destination.
3. OFFSET—Selects position established by offset range and bearing settings relative to the destination position as the radar sighting point, while maintaining the destination position as the navigation (or bombing) destination.
4. MAN FIX—Applies energizing voltages to the present position hold and present position correction buttons. Removes attack radar cursors from display, de-activates the target bearing, slant range, course angle, and distance to destination servos, and enables control of present position counters by their respective control knob.
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 deactivates 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 latitude or longitude position counters will not slew more than 18 degrees from the destination latitude or longitude indicated at the time a fix mode dest storage selector button is depressed. The counter(s) will drive to an erroneous position should this limitation not be observed during operation. However, even though the limitations are exceeded the stored information will not be lost, unless changed by handsetting, and will drive the counters to the correct stored position when the 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 horizontal triangulation to improve accuracy of fixing the interrogating radar coordinates. The cursors on the RHAW and attack radar scopes 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 1-59), 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 up-dating 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, engaging 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 mode should be deactivated (button disengaged) by depressing any other fix mode selector button.

Present Position Hold Button.

The present position hold button (1, figure 1-59), 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 a 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 1-59), located on the bomb nav control panel, is a three-position rotary knob marked OFF/AUX NAV, NORMAL and RAPID ALIGN. The knob must be pulled out before it can be rotated from any position. The knob controls the alignment modes of the stabilized platform when the bomb nav mode selector knob is positioned to ALIGN. In the OFF/AUX NAV position, the stabilized platform is completely de-energized. In the NORMAL position, the platform may be aligned two ways, (1) gyrocompass or (2) alignment to preset magnetic variation. The gyrocompass alignment is the normal alignment, and is much more precise. In the gyrocompass alignment, the stabilized platform goes through two modes of azimuth alignment: (1) Alignment to true north as defined by the bomb nav system flux valve magnetic heading plus handset magnetic variation, and (2) Refinement of alignment or gyrocompassing to true north as precisely defined by the direction where the "East" gyroscope does not sense any earth rotation. Alignment to preset magnetic variation may be used when a fast alignment is desired. The operator manually places the bomb nav mode selector knob to a navigate mode after the stabilized platform has positioned to computed true heading and before the gyrocompass phase of alignment begins. The use of rapid align depends on whether or not gyrocompass heading information had previously been stored in the system and the aircraft not moved. In this mode, the platform will align to the stored heading and will be ready for operation in less than ¼ the time required for gyrocompass alignment. (Refer to "Principles of Operation," this section.)

Altitude/Test Selector Knob.

The altitude/test selector knob (11, figure 1-59), located on the bomb nav control panel, is labeled ALTITUDE TEST. The knob has five positions for computer self-testing marked 1, 2, 3, 4 and 5; and three positions marked CAL, NORM and ALIGN. The knob must be pulled out to go into or out of the test sector. The CAL position provides automatic calibration of pressure altitude by the radar altimeter if an altitude good signal is present from the radar altimeter, or for semi-automatic calibration when above terrain of known elevation with the attack radar. The ALIGN position provides for setting the pressure altitude correction term to zero. The NORM position is the normal NC operating mode. Test 1, 2 and 3 positions are for system operational ground checkout; test positions 4 and 5 are for BCU ground checkout.

Note

Do not set the switch to any of the five test positions in flight. To do so would furnish erroneous heading information to the flight instruments.

Platform Indicator Lamps.

Three platform indicator lamps (18, figure 1-59), 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 stabilized platform (SP). When the system is activated, the lamp will light until the gyroscope temperature reaches the required operating level (160°F). When the lamp is lighted, the SP will not enter the gyrocompass cycle. 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, signaling 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 prior to taxiing.

Note

If aircraft moves prior to placing the mode selector knob to a nav mode, platform alignment will be degraded in proportion to aircraft movement, thereby rendering the stable platform unreliable for use as an attitude heading reference.

With the platform alignment control knob in RAPID ALIGN, the align indicator lamp will remain out until alignment is complete at which time the lamp will begin flashing. The error indicator lamp provides a monitor of platform reliability when the bomb nav mode selector knob is in ALIGN or any of the normal navigation positions. The lamp will light when the SP is off, unless the platform alignment control knob is in the OFF/AUX NAV position. The lamp will light, should the platform turn itself off, when the bomb nav mode selector knob is in any of the normal navigation modes.

Primary Attitude/Heading Caution Lamp.

The primary attitude/heading caution lamp, located on the main caution lamp panel, (figure 1-37), will light when either attitude or heading information from the stabilized platform is interrupted or becomes unreliable due to a platform malfunction. The lamp will light: (1) Any time the bomb nav system is in an align mode. (2) Any time the bomb nav system ceases to supply an attitude ready signal to the flight director system. (3) Any time the flight instrument reference select switch is in the AUX position and, (4) by placing the bomb nav mode selector knob to AUX NAV CHECK. When the lamp lights the letters PRI ATT/HDG will be visible.

Go Lamp.

The green GO lamp (10, figure 1-59), located on the bomb nav control panel, is labeled GO and provides for monitoring the navigation computer self-test circuits. The lamp will light when the altitude/test selector knob is placed in TEST positions 1, 2 or 3, provided the proper settings have been set on the panel, and the bomb nav system is operational. When the altitude/test selector knob is in the ALIGN position the lamp will light when pressure altitude correction term is zero. The lamp will light when the altitude/test selector knob is in the CAL position provided the automatic altitude calibration by the radar altimeter is completed, and the bomb nav system is operational.

Present Position Counters.

Two present position counters (2, figure 1-59), 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 aircraft position coordinates. The counters are continuously and automatically up-dated by inputs of true north and east velocity components from the stabilized 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, handset or last computed wind data, and auxiliary flight reference system heading plus hand-set magnetic variation. Control knobs are provided for electrically slewing the counters to set in initial position or to insert corrections. The control knobs are activated only when the fix-mode selector man fix button is depressed to prevent accidental slewing. The speed at which the counter is driven is proportional to the degree the control knob is turned. Under certain conditions the counters may also be automatically driven with the attack radar tracking handle.

Fixpoint Elevation Counter.

The fixpoint elevation counter (5, figure 1-59), located on the bomb nav control panel is labeled FIXPOINT ELEVATION. An adjacent control knob (not labeled) provides control of the counter, independent of the altitude/test selector knob. The counter indicates the fixpoint elevation data programmed into the computer for computation of radar sighting values for slant range and depression angle. In operational usage, when the fixmode selector target, homer set or homer track buttons are depressed, the counter should be set to the

elevation of the position indicated by the destination position counters, and when the fixmode selector OFFSET button is depressed, the counter should be set to the elevation of the offset aimpoint. The counter need not be preset prior to selection of the altitude/test selector knob ALIGN position, as altitude alignment is automatic, independent of the counter setting, and the counter may be set to the elevation of the calibration terrain either before or after selection of the altitude/test selector knob CAL position. Normally, the counter can be set to any reading from 0000 to 9990.

Destination Position Counters.

The latitude and longitude coordinates of aircraft destination, radar fixpoints or targets must be set into the destination latitude and longitude counter in order to use the course computation, radar cursor laying, or automatic bombing functions. Two destination position counters (22, figure 1-59), located on the bomb nav control panel, are labeled DEST LATITUDE and DEST LONGITUDE. The counters, which display destination position in 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 for electrically slewing the counters to set in destination position. The speed at which the counter is driven is proportional to the degree the control knob is rotated. The last drum of the destination counters is numbered in tenths of minutes and has intermediate markings at 0.05 minute intervals. These counters can be set to one-half of the half tenths divisions or 0.025 minutes. For accurate radar fixtaking or bombing of targets by techniques other than direct radar sighting, these counters should be set to the nearest 0.025 minutes.

True Heading Counter.

The true heading counter (6, figure 1-59), located on the bomb nav control panel, is labeled TRUE HEADING. Except in self-test, the counter continuously displays the computer aircraft heading relative to true north as derived from (1) inputs of true heading from the stabilized 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 1-59), 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 may be adjusted until the magnetic heading synchronization indicator indicates a null. At this time, the counter will indicate the actual 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 navigation computer true heading.

Magnetic Heading Synchronization Indicator.

The magnetic heading synchronization indicator (4, figure 1-59), 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 the bomb nav system flux valve input. During the normal navigation modes, the indicator may be maintained at null by periodic manual correction of handset magnetic variation to correct computed magnetic heading to agree with flux valve data. The indicator is unusable in the auxiliary navigation modes. In the platform gyro-compass mode, the indicator provides an indication of the stability and accuracy of platform azimuth alignment. The indicator monitors the vertical accelerometer output during the level phase of platform alignment and a null indicates this output is good.

Groundtrack and Groundspeed Counters.

Note

Zero velocity may be indicated on the GROUNDSPEED counter by any reading from 9950 up to 9999, or 0000 to 0002. A groundspeed reading in the range of 9950 to 9999 may also be due to mechanical slippage between the mechanical data shaft and the counter; this possibility may be investigated by placing the altitude/test selector knob to TEST 1 or 2 and checking that the counter drives to 0800 (±0003), or by observing the counter for correct indications during taxi. If the counter is checked in TEST 1 or 2, the altitude/test selector knob must be repositioned out of the TEST sector prior to taxi, as these test positions inhibit NC and SP automatic latitude tracking, introducing SP alignment and navigation errors proportional to position changes.

The groundtrack and groundspeed counters (9, figure 1-59), located on the bomb nav control panel, are labeled GROUNDTRACK and GROUNDSPEED. Ex-

cept during self-test 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 AFRS handset magnetic variation, and either last computed or handset wind information during all auxiliary navigation modes.

Wind Speed and Wind From Counters.

The wind speed and wind from counters (19, figure 1-59), 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. Wind speed limits are from 0 to 250 knots and wind direction is 0 to 360 degrees.

Glide/Dive Angle Counter.

The glide/dive angle counter (17, figure 1-59), located on the bomb nav control panel, is labeled GLIDE/DIVE ANGLE. The counter displays the preselected glide, dive or loft angle set into the computer. The computer compares selected glide, dive or loft angle with pitch angle and generates an error signal for vertical steering in AILA, dive and loft bombing modes. The resultant steering information is displayed on the attitude director indicator and lead computing optical sight. The glide/dive angle counter limits are from 00.0 to 99.9 degrees.

Offset Range and Offset Bearing Counters.

The offset range and offset bearing counters (21, figure 1-59), 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.

Destination Distance/Time Counter.

The destination distance/time counter (14, figure 1-59) 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 VISUAL CCIP, SHORT RANGE or GREAT CIRCLE, MILES TO DEST will light and display continuous computations of the distance in nautical miles from present position to any destination set into the destination counters (provided the destination is within 200 to 4000 nautical miles of the present position for GREAT CIRCLE or 0-200 nautical miles for SHORT RANGE). When the mode selector knob is placed to RANGE BOMB, the words, FEET TO REL and two zeroes (00) will light, and display continuous computation of the range to the bomb release point in feet from zero to 400,000 feet. When the mode selector knob is placed to TRAIL or AUTO BOMB, SEC TO REL will light, and display continuous computation of the seconds remaining before bomb release ranging from zero to 400 seconds.

Trail/Range Counter.

The trail/range counter (12, figure 1-59), 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, short range, visual CCIP or auto bomb modes. Trail range setting limits are from 0 to 99,990 feet.

Time-of-Fall Counter.

The time-of-fall counter (8, figure 1-59), 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 manual ballistics 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 manual ballistics mode. Time of fall setting limits are from 0 to 99.9 seconds.

Burst Altitude Counter.

The burst altitude counter (1, figure 1-60), located on the burst control panel, is labeled BURST ALT. A control knob is provided to set in a weapons burst altitude of 0 to 9990 feet above target. When the target elevation switch is positioned to SAME AS AIMPOINT, the counter is set to the burst altitude above target. When the target elevation switch is positioned to ABOVE AIMPOINT or BELOW AIMPOINT, the counter is set to the differential altitude between the bombing target and the radar aimpoint.

Train Lead Counter.

The train lead counter (3, figure 1-60), located on the burst control panel, is labeled TRAIN LEAD. A control knob is provided to set in weapons train lead from 0 to 9990 milliseconds.

Bomb Release Lamps.

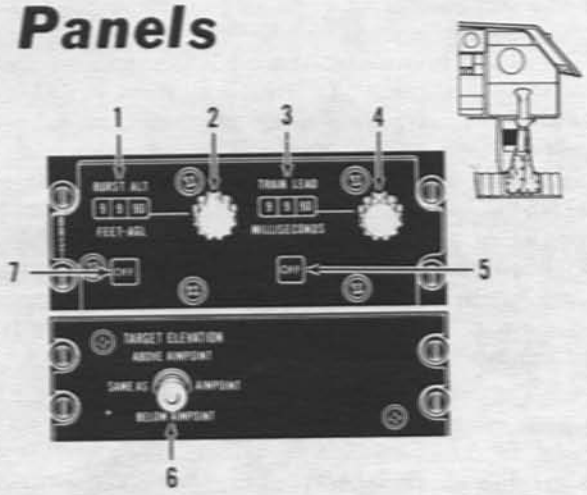
The green bomb release lamp (16, figure 1-59), located on the bomb nav control panel is labeled BOMB RELEASE. The lamp provides monitoring for the automatic release signal that is computed and provided by the navigation computer to other equipment for weapons release. The lamp will light when the computer is in the auto bomb, trail or range bomb configuration and the seconds or feet to release has driven to zero. An additional green bomb release lamp is located on the left main instrument panel (18, figure 1-5) and when lighted will display the words BOMB RELEASE. This lamp will light to indicate a release signal is present for the following conditions:

- When man delivery mode is selected, the master arm and release switch is ON and the control stick weapon release button is depressed.
- When timer delivery mode is selected, the master arm and release switch is ON, the dual bombing counters have driven to zero and the control stick weapon release button is depressed.
- When angle delivery mode is selected, the master arm and release switch is ON, the control stick weapon release button is depressed and the aircraft pulls up to the angle preset into the bomb nav system.
- When nav delivery mode is selected, the master arm and release switch is ON, a control stick weapon release button is depressed and the bomb nav system is transmitting a weapon release signal (NC bomb release lamp on).

WARNING

Do not operate any of the bombing system controls when a release signal is present as indicated by the bomb release lamp. Operation of the station select indicator/pushbuttons, monitor and release knob, delivery mode indicator/pushbuttons, or release option indicators/pushbuttons during presence of a release signal, may result in inadvertent store or bomb rack release.

Burst Control and Target Elevation Panels



1. Burst Altitude Counter.
2. Burst Altitude Selector Knob.
3. Train Lead Counter.
4. Train Lead Selector Knob.
5. Train Lead Indicator.
6. Target Elevation Switch.
7. Burst Altitude Indicator.

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Figure 1-60.

Target Elevation Switch.

The target elevation switch (6, figure 1-60), located on the target elevation panel is labeled TARGET ELEVATION. The switch has three positions marked ABOVE AIMPOINT, SAME AS AIMPOINT and BELOW AIMPOINT and is lever-locked in all positions. The switch permits the ballistics computer unit to be programmed to compute weapon ballistics for a target that is at an elevation either above or below the elevation of the radar aimpoint.

Note

When the target elevation switch is in the SAME AS AIMPOINT position, the ballistics computer uses the same altitude above target as is used for radar sighting, and the computer recognizes the burst altitude setting only when an air burst weapon has been selected.

Radar Bomb Scoring (RBS) Tone Switch.

The RBS tone switch (12, figure 1-4), located on the center throttle panel is provided to control a 1020 cps tone transmitted by the UHF radio for scoring simulated bomb runs. The switch is a lever lock type

with two positions marked RBS TONE (in the ON position) and OFF. The switch is spring loaded to the OFF position and is locked in the OFF position. The switch toggle must be pulled out to move the switch from OFF to ON. Placing the switch to the ON position turns on the tone and energizes a switch solenoid to hold the switch in the ON position. Regardless of the delivery mode either weapon release button must be held depressed to allow the release signal to break the circuit to the switch solenoid to turn off the tone. The tone can be manually turned off by placing the switch to the OFF position.

Bomb Nav Distance-Time Indicator.

The bomb nav distance-time indicator (BNDTI) (27, figure 1-5), located on the left main instrument panel, is a remote indicating type instrument. The indicator displays digital time or 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 navigation 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 operational computing 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 or visual CCIP, the tape will display short range/miles and the digital display will indicate distance in nautical miles to destination. When operating in trail bomb or auto 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.

Train Lead Indicator.

The train lead indicator (5, figure 1-60) is located on the burst control panel. The indicator is an on/off flip flop with the ON display being visible whenever a train mode has been selected on the armament select panel. With an ON indication present, train lead information will be directed to the BCU whenever the bomb nav mode selector knob is in the AUTO BOMB or VISUAL CCIP positions.

Burst Altitude Indicator.

The burst altitude indicator (7, figure 1-60), located on the burst control panel is an on/off flip flop indicator. An ON indication, indicates the burst altitude information is being used by the BCU when the bomb nav mode selector knob is in AUTO BOMB or VISUAL CCIP positions.

PRINCIPLES OF OPERATION.

The bombing-navigation (bomb nav) system generally utilizes electro-mechanical analogue computing circuits to provide continuous solutions to navigation and bombing problems, as represented by operator inserted destination/target position data and derived or sensed data for aircraft position, attitude and velocity. (Refer to figure 1-61.) The stabilized platform SP is a precise inertial reference for aircraft attitude, heading and velocity. The navigation computer (NC) accomplishes all of the computing tasks and provides all the system operating controls except those on the burst control and target elevation panels. The ballistic computer unit (BCU) provides weapons ballistic data. The bomb nav flux valve is the system's primary magnetic heading reference. The BNDTI provides the left crew member with a display of distance, time and mode information from the NC. A more detailed discussion of each unit and its interface with other units and systems is given below. For bombing navigation system operating procedures, refer to Section IV.

Stabilized Platform. (SP)

Components. The stabilized platform contains a reference platform on which two precision gyroscopes and three precision accelerometers are mounted. One accelerometer is mounted on the platform to serve as a vertical acceleration reference for the BCU. Each gyro is constructed so it will sense rotation of its case about either of two sensitive axes, and to provide an electrical signal output for each axis in proportion to the rotational rate about that axis. The gyros are mounted on the platform such that one sensitive axis of each coincides with one sensitive axis of the other, with the remaining sensitive axis of each at right angles to the remaining sensitive axis of the other, thus constituting an orthogonal three axis reference system. The accelerometers are constructed with only one sensitive axis each, and are mounted on the platform so that their sensitive axes are at right angles to each other and parallel with the right angle axes established by the gyros. The third accelerometer is mounted on the platform so that its sensitive axis is parallel to the coincident gyro axis. With the gyros and accelerometers mounted as described, they are capable of very sensitive detection of any angular rotation or inertial acceleration about any of the platform's three reference axes. The platform is supported within the SP by a gimbal system which isolates the platform from angular rotations of the SP case, so that in operation the SP case may be rotated through any angle without affecting the orientation of the platform with respect to inertial space. The gimbal axes are mechanized with synchro devices to measure and trans-

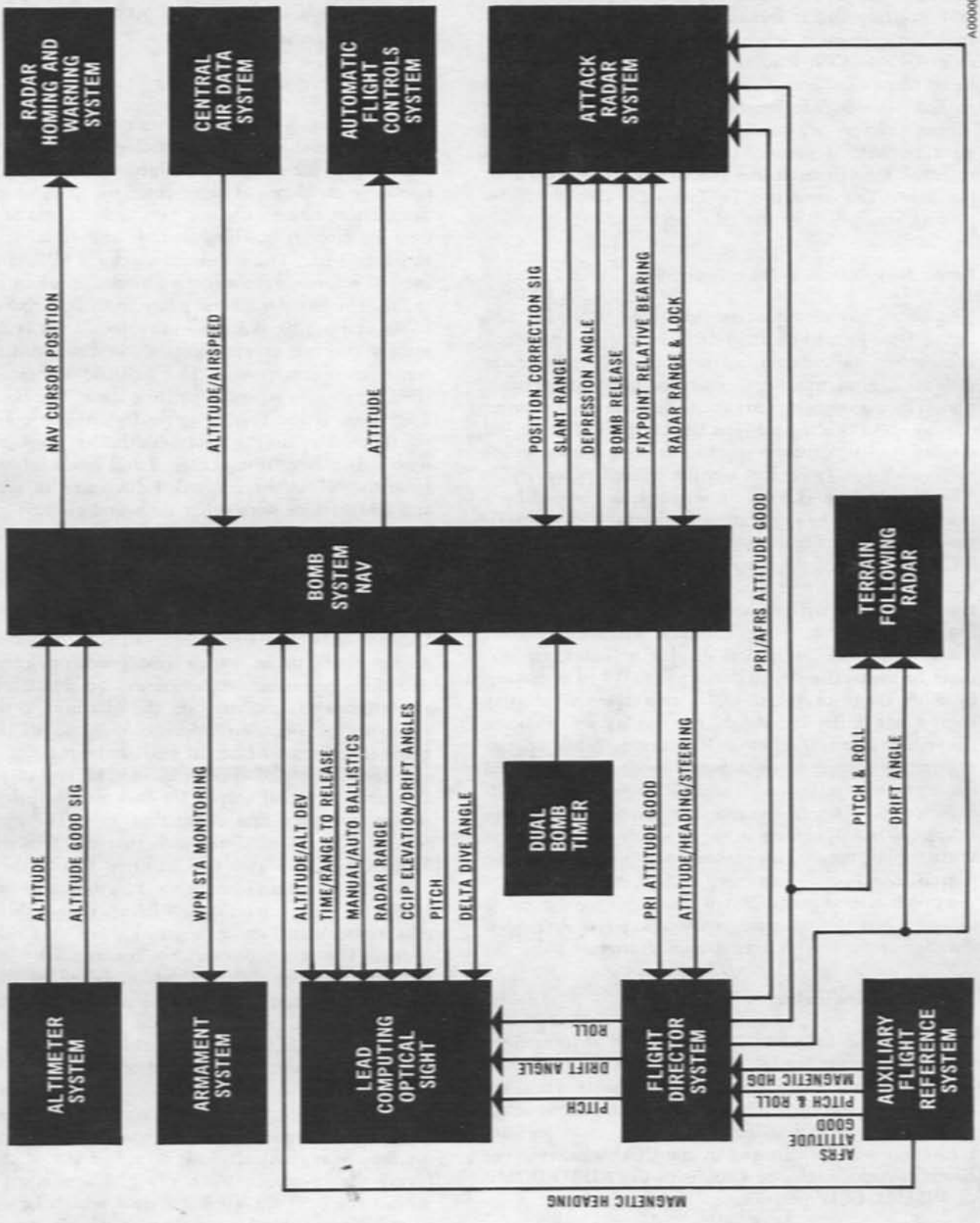
Bomb Nav - Subsystem Tie-Ins
RADAR HOMING AND WARNING SYSTEM
CENTRAL AIR DATA SYSTEM
AUTOMATIC FLIGHT CONTROLS SYSTEM
ATTACK RADAR SYSTEM
NAV CURSOR POSITION
ALTITUDE/AIRSPEED
ATTITUDE
POSITION CORRECTION SIG
SLANT RANGE
DEPRESSION ANGLE
BOMB RELEASE
FIXPOINT RELATIVE BEARING
RADAR RANGE & LOCK
BOMB SYSTEM NAV
PRI/AFRS ATTITUDE GOOD
TERRAIN FOLLOWING RADAR
ALTIMETER SYSTEM
ALTITUDE
ALTITUDE GOOD SIG
ARMAMENT SYSTEM
WPN STA MONITORING
LEAD COMPUTING OPTICAL SIGHT
ALTITUDE/ALT DEV
TIME/RANGE TO RELEASE
MANUAL/AUTO BALLISTICS
RADAR RANGE
CCIP ELEVATION/DRIFT ANGLES
PITCH
DELTA DIVE ANGLE
DUAL BOMB TIMER
PRI ATTITUDE GOOD
ATTITUDE/HEADING/STEERING
PITCH & ROLL
DRIFT ANGLE
ROLL
DRIFT ANGLE
PITCH
FLIGHT DIRECTOR SYSTEM
MAGNETIC HDG
PITCH & ROLL
AFRS ATTITUDE GOOD
AUXILIARY FLIGHT REFERENCE SYSTEM
MAGNETIC HEADING
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Figure 1-61.

mit the angular displacement of each gimbal with respect to its supporting gimbal, and each gimbal is driven by a motor which in operate modes is controlled by signals derived from computed latitude data from the NC, pressure altitude rate data from the CADC, and aircraft rotation and acceleration data from the gyros and accelerometers on the reference platform. The latitude data is used to derive signals with which to precess the gyros to compensate for the earth's rotation, the pressure altitude rate data is used to develop signals to compensate the accelerometer outputs for coriolis type accelerations associated with altitude changes, the accelerometer signals are used to develop gyro precession signals to compensate for aircraft translational motion relative to earth, and the compensated gyro outputs drive the motors.

Power Requirements. The SP contains its own power supply and the necessary electronics to control the reference platform orientation, and to process the information derived therefrom for transmission to using systems. Three phase ac power is made available to the SP from both the left and right main ac buses. The SP power supply, however, contains a circuit that continuously monitors the quality of the power supplied from the left bus. When left bus power is within specifications, this circuit selects the left bus as the power source, but when left bus power is not within specifications, this circuit switches to the right bus supply. Since the NC supplies the SP with updated latitude data and the NC requires left bus power, the SP is mechanized to cease to track the updated latitude data when left bus power is not being used, and SP accuracy will therefore be degraded in proportion to latitude changes made under these conditions. This circuit will automatically turn the SP off in the event that power from both buses is outside specification requirements, and to require that the control switches be recycled through the turn-on sequence in order to reactivate the SP if it has been automatically turned off while in an align or operate mode.

Temperature Controls. The SP is cooled by forced air from the air conditioning system. To prevent possible damage from over-heating, the cooling air should be on at all times during SP operation except for initial heating cycle. When the SP is on and cooling air is supplied, the power supply and electronics are cooled continuously, but the air to the reference platform is controlled by a self-contained temperature control circuit. This circuit constantly monitors temperatures within the reference platform and controls the application of cooling air and the operation of self-contained heating circuits as required to achieve and maintain the temperature levels required for accurate operation.

Error Detection Circuits. The SP also contains a self-test circuit that operates continuously when the unit is on. This circuit monitors the gyro and accelerometer outputs and the precision frequency gyro spin rate control voltage. When the signals at any of the monitored points exceed design levels, this circuit will automatically turn the SP off. This circuit is capable only of detection of major failures of the circuits most critical to operation of the reference platform, and therefore cannot be relied upon to detect all SP malfunctions. Since accuracy of SP performance is essential to safe accomplishment of some missions, operators should accustom themselves to continuous evaluation of SP performance by observation of those system characteristics that reflect SP outputs.

Heat Cycle. Operation of the SP is controlled by the platform alignment control knob and bomb nav mode selector knob on the NC, and by self-contained automatic circuits. The previously noted temperature control circuit is activated when the bomb nav mode selector knob is in any position other than OFF, and the platform alignment control knob is in any position other than OFF/AUX NAV, and the automatic control circuit has not turned the SP off. When the bomb nav mode selector knob is placed to the HEAT position, or any position clockwise from HEAT, with the platform alignment control knob in NORMAL or RAPID ALIGN, a signal is routed through the bomb nav mode selector knob to the SP to activate the temperature control circuits. The voltage supplied by the HEAT position also sets SP internal control logic to enable activation of the SP power supply by a signal supplied separately by the NC when the bomb nav mode selector knob switch is rotated to any position clockwise from HEAT. Therefore, either on initial alignment, or to reinitiate alignment, after some system failure, it is necessary to position the bomb nav mode selector knob to HEAT, at least momentarily, to activate the SP, before going to ALIGN. This step is not necessary if bomb nav system operation has been normal. To set the SP in rapid align mode after landing refer to Section II. On receiving this signal, the temperature control circuits will commence to heat the platform at a rate of temperature increase of approximately 35 degrees F per minute, until operating temperature is reached. During this time, the SP supplies a signal to the NC to light the platform heat lamp. After operating temperature is reached, the temperature control circuits will cycle the heaters and cooling air on and off as required to maintain the temperature, until the SP is turned off.

Align Mode Mechanization. When the bomb nav mode selector knob is in ALIGN, and the platform alignment control is in NORMAL or RAPID ALIGN, and the automatic control circuit has not turned the SP off, the reference platform control circuits are switched into a configuration to automatically effect self-alignment. When the bomb nav mode selector knob is rotated to ALIGN, a signal is routed to the SP to activate the SP power supply. If the HEAT position signal has been transmitted as previously described, the SP power supply will be activated and the

SP will start an automatic self-alignment sequence. During the alignment cycle, the SP supplies signals to the NC to indicate status of alignment, and for use in NC logic control circuits. A signal is supplied to indicate that the SP is on. When the NC receives this SP on signal, a relay is activated to substitute this SP on signal for the aircraft 28 vdc as the SP control voltage source, and other relays are set so that loss of this SP on signal will light the platform error lamp. The SP on signal also enables the development of a separate signal for the transmission to the flight directory system to indicate that the primary attitude heading reference system is ready for use. If the reference platform has not completed warm-up, 28 vdc will continue to be supplied to illuminate the platform heat lamp. For the first 30 seconds of alignment, a "coarse align" signal will be supplied, and after coarse align and until the next phase of alignment is complete, another "levelling mode" signal will be supplied. Each of these signals is used in the NC to inhibit transmission of the primary attitude heading reference ready signal, and to set the NC true heading circuits to a configuration to compute true heading from flux valve and magnetic variation data. One SP circuit controls a ground return circuit for the NC platform align lamp. This circuit holds the ground return open until alignment has reached the gyrocompass phase, or until alignment is complete. If a gyrocompass alignment is being accomplished, this circuit will close the ground return path, causing the align lamp to light, when the gyrocompass phase of alignment begins, and will cycle the ground return path open and closed, causing the align lamp to flash on and off, when alignment has reached specification accuracy requirements. If a rapid alignment is being performed, with the platform alignment control knob in RAPID ALIGN, the ground return path will be held open until alignment is complete to specification accuracy requirements, at which time it will be cycled open and closed to flash the align lamp. The SP supplies another variable level dc signal that is indicative of the degree of stability and accuracy of azimuth alignment. When the coarse align and levelling phases of alignment are complete, with the platform alignment control knob in NORMAL and the bomb nav mode selector knob switch in ALIGN, the NC switches this signal to drive the NC mag heading sync meter to provide the operator with a qualitative indication of status of alignment. At this time, the NC also switches its true heading circuits to position to the heading indicated by the synchro data being received from the SP. Also, the SP supplies ac and dc signals proportional to the velocities indicated by the integral of the accelerometer outputs. In the align mode, the NC disregards the dc signals, but uses the ac signals to compute and display the groundtrack and groundspeed indicated. As platform alignment progresses, the NC groundspeed counter will drive to indicate zero velocity and the groundtrack counter will drift randomly, indicating that groundtrack is indeterminate. When the bomb nav mode selector knob is rotated to an operate mode after coarse alignment is complete, and the platform alignment control knob is in NORMAL or RAPID ALIGN, and the automatic control circuit has not turned the SP off, the reference platform control circuits are switched into a configuration to maintain the orientation established in the alignment mode. When alignment has been properly completed, one set of gyro and accelerometer axes will be aligned with true north in the local horizontal plane, another set will be aligned with true east in the local horizontal plane, and the coincident set of gyro and accelerometer axes will be aligned with the local plumb-bob vertical.

Gyrocompassing Principles. In alignment modes, alignment to the local plumb-bob vertical is automatically effected by utilizing the outputs of the north and east accelerometers to drive the gimbal system motors until these accelerometers sense no component of acceleration due to the earth's gravitational field. Due to accelerometer sensitivity, this condition can only occur when the accelerometer axes are at right angles to the gravitational field. Alignment to true north is effected by either of two methods, as elected by the operator. In one method, the output of one axis of one of the gyros is utilized to drive the azimuth gimbal motor until this gyro senses no rotation about this axis. Due to the earth's rotation and gyro sensitivity, this condition can only occur when the reference platform north axis is very precisely aligned with true north. This method is called gyrocompassing, and is relatively time consuming although it provides the most accurate heading and velocity. The SP will automatically sequence through the gyrocompassing method when the bomb nav mode selector knob is set to ALIGN, with the platform alignment control switch in NORMAL.

Rapid Align and Alignment to Pre-set Magnetic Variation. In these methods, the platform is driven in azimuth until its true heading output agrees with the true heading data in the NC. These methods are much faster than the gyrocompassing method, but should be used only when fast alignment is required. The SP will automatically sequence through the rapid align method when the bomb nav mode selector knob is placed to ALIGN, with the platform alignment control knob in RAPID ALIGN. The alignment to pre-set magnetic variation is effected by manual operator termination of the automatic gyrocompass alignment sequence, after the alignment mode has progressed sufficiently to achieve an accurate alignment and before the automatic sequencing switches azimuth alignment control to the north-seeking gyro. In order to use this mode effectively, the magnetic variation counter must first be set to a reading known to yield a correct true heading result when combined with the magnetic heading flux-valve data under the local magnetic environment.

Coarse Align. The first phase of all of the alignment methods is called coarse alignment and always occurs for the first 30 seconds after the bomb nav mode select knob is placed to ALIGN, provided that the platform alignment control knob is in NORMAL or RAPID ALIGN and the bomb nav mode selector knob has been cycled through the HEAT position (not necessary if setting SP to rapid align mode after landing if bomb nav system has been normal). The purpose of this phase is to quickly orient the reference platform close to the ultimate alignment orientation and thereby reduce total alignment time. In this phase, the output of the pitch and roll synchros are utilized to drive the gimbal motors until zero pitch and roll conditions are achieved, and the NC true heading data is used to drive the platform in azimuth until its true heading output agrees with the NC true heading data. It should be noted that this phase will continue even if the bomb nav mode selector knob is advanced to an operate mode before the 30 second time period has expired. It is possible, either on the ground or while airborne, to bring the SP to a rough alignment in 30 seconds without going through a complete alignment cycle; however, because the alignment is inaccurate to the extent that the aircraft is not level, and because the resultant output velocities and other signals such as drift angle will be greatly in error, therefore use of this feature should be limited to emergency conditions where an attitude reference is required and the AFRS cannot be used.

Leveling. When the 30 second time period has expired, with the bomb nav mode selector knob in ALIGN, the SP will automatically switch into the next phase, called levelling. In this phase, control of vertical alignment is switched to the accelerometers, while azimuth alignment continues to be controlled by NC true heading. The acccuracy of alignment attained in the levelling mode is thus dependent on the accuracy of the true heading data supplied by the NC. When the platform alignment control knob is in RAPID ALIGN and the bomb nav mode selector knob is not in an operate mode, the NC true heading data is locked. Thus, by placing the platform alignment control knob to RAPID ALIGN while the bomb nav mode selector knob is in ALIGN, the operator may then set the bomb nav mode selector knob to OFF and thereby store the existing NC heading data for subsequent use in the rapid align mode. If a gyrocompass alignment has been accomplished immediately preceding this heading storage procedure, the stored heading will be very accurate, and the SP will therefore achieve a very accurate realignment in the rapid align mode, provided realignment is accomplished with the aircraft on exactly the same heading that existed when the heading data was stored. If the operator has selected the rapid align mode, the SP will signal the operator when leveling has achieved the required accuracy, by causing the ALIGN lamp to flash on and off, and the levelling mode will continue until the bomb nav mode selector knob is rotated to an operate mode.

Gyrocompass Align and Pre-set Magnetic Variation. If the operator has not selected the rapid align mode, the SP will automatically switch control of azimuth alignment to the north-seeking gyro when levelling is completed, as determined by the level detection circuits, and will signal the operator by causing the ALIGN lamp to light steadily. This phase is called the gyrocompass phase and is timed to continue for a minimum of 5 minutes. After the 5 minutes have lapsed and the self-contained level detection circuits sense that required accuracy has been attained, the SP will signal the operator by causing the align lamp to flash on and off. At this point, alignment accuracy is sufficient to assure specification performance; however, the SP will continue the gyrocompass process to an increasingly accurate degree until the bomb nav mode selector knob is placed to an operate mode. The operator should therefore leave the SP in the alignment mode as long as available time permits. When the NC platform alignment control knob is in NORMAL, and the bomb nav mode selector knob is in ALIGN, and the levelling phase of alignment has not been completed, the NC true heading data is computed from the summation of the magnetic heading flux valve input, plus the setting of the NC magnetic variation data. If the local magnetic variation (including deviations due to the local magnetic environment) is accurately known, the NC true heading data can be set accurately by inserting this value into the magnetic variation counter prior to, or early in the alignment cycle, and a fast, accurate alignment can be accomplished in the alignment to preset magnetic variation mode. The procedural principle of this mode is to manually set true heading to an accurate value and then to start a gyrocompass alignment sequence to allow the coarse align and levelling phases to reach an accurate alignment, and then to set the bomb nav mode selector knob to an operate mode before the timed gyrocompass phase begins.

Alignment Times. Due to internal timing, it is theoretically possible that the gyrocompass phase could begin within 44 seconds after the align mode is initiated, but due to the probable conditions of aircraft, attitude, ambient temperature and magnetic heading accuracy, the gyrocompass phase will most likely be automatically inhibited by the level detection circuits for a minimum of 110 seconds, so that, for the best probability of an accurate alignment, the bomb nav mode selector knob should be set to an operate mode 110 seconds after alignment is initiated. The time required to complete alignment is a function of magnetic heading accuracy (or stored heading accuracy), aircraft attitude, ambient temperature, latitude, and tolerances of internal timing circuits; if magnetic heading (or stored heading) is precise, aircraft is level, ambient temperature is warm, alignment latitude is 45 degrees, and all internal timing is minimum, a rapid alignment may be accomplished in 50 seconds or a gyrocompass alignment may be completed in approximately 5¾ minutes, while if magnetic heading is 2 degrees in error

(or stored heading is not correct), aircraft is 10 degrees off level, temperature is cold, alignment latitude is greater than 45 degrees and all internal timing is maximum, a rapid alignment may require 4 minutes and a gyrocompass alignment may require 16 minutes, and proportionately more time may be required under worse conditions. If this switching is not accomplished before gyrocompassing begins, slight heading transients due to automatic mode switching will degrade the accuracy of alignment.

Operate Modes and Outputs. When alignment is completed, the SP is oriented to serve as a very accurate, inertial attitude, heading and velocity data source. As long as the bomb nav mode selector knob is in ALIGN, however, any movement of the aircraft will be interpreted by the reference platform control circuits to be due to earth rotation, and the platform will be positioned in error to offset the movement. Thus, the SP is not ready for use until the bomb nav mode selector knob is placed to an operate mode, switching the platform control circuits to the configuration required to maintain alignment orientation. The flight director system is therefore mechanized to inhibit use of SP data by other systems, and to light the primary attitude/heading lamp on the master caution lamp panel, until a primary attitude heading ready signal is received from the NC. The NC logic control circuits are mechanized to supply this signal only when the SP is on, coarse align has been completed, and the bomb nav mode selector knob is in an operate position not designated as aux nav. The SP reference platform gimbal assembly contains one set of very accurate pitch and roll synchros exclusively for interface with the flight control systems; the input terminals of these synchros are connected directly to an excitation voltage supplied by the flight control system, and the output terminals are connected exclusively to the flight control system input circuits. The platform gimbal assembly also contains another set of very accurate pitch and roll synchros that serve three functions, (1) they are used internally as the verticality reference for the coarse alignment phase of alignment, (2) they are used internally to drive repeating servos which contain less accurate synchros that serve to supply pitch and roll data externally to the flight director system for distribution to the attack radar antenna pitch control circuit, the LCOS pitch deviation indicator circuit, the ADI pitch and roll circuits, the LCOS pitch and roll indicator circuits, and the flight director computer roll circuits when a primary attitude heading ready signal is present from the NC, (3) they are supplied externally directly to the flight director system for distribution to the TFR pitch channel and the attack radar roll plate control circuits when a primary attitude heading ready signal is present from the NC. The platform gimbal assembly also contains a very accurate azimuth synchro that is connected exclusively to the NC. This signal is connected to the BCU, where it is routed to the NC or replaced by a test signal in self-test modes. The platform gimbal assembly also contains a very accurate azimuth synchro that is connected exclusively to the NC. In the coarse align and levelling phases of alignment, the NC uses this data to develop a signal to return to the SP to drive the platform in azimuth until platform azimuth agrees with NC computed true heading. In all other modes, except those designated aux nav or test, when the SP on signal is present, the NC true heading computer utilizes the platform azimuth synchro signal as the primary true heading. The SP contains circuits which integrate the acceleration signals supplied by the north and east accelerometers and drive the output contacts of self-contained potentiometers to positions corresponding to inertial velocity north and east. In each channel, one potentiometer is excited by a precision dc voltage from the NC, and another is excited by a precision ac voltage from the NC. The output contacts of these potentiometers are connected exclusively to the NC, where the dc signals are used as inputs to the NC integrating circuits that continuously update the present position data, and the ac signals are used to compute groundtrack and groundspeed and, in combination with airspeed data supplied by the central air data computer, to compute wind speed and wind from. One set of synchros in the present latitude computer in the NC is connected exclusively to the SP latitude repeater circuit. If left main ac bus power is within specifications, and the NC altitude/test selector knob is not in a test position, the SP latitude repeater repeats the NC latitude data. As noted previously, the SP latitude circuit contains computing elements to develop the gyro precession and acceleration correction signals that are functions of latitude.

Navigation Computer. (NC)

Components. The navigation computer contains a power supply, electronics, electro-mechanical servomechanism, indicators, and most of the system manual operating controls.

Cooling System. The NC is cooled by forced air from the aircraft air conditioning system. To avoid equipment damage, cooling air should be applied at all times that the NC is on.

Power Requirements. Three phase ac power is supplied from the left main ac bus and aircraft 28 vdc is supplied from the main dc bus. The ac power is utilized to drive the NC power supply, and the dc power is used to provide system turn-on control. When the NC power supply is activated, it generates all of the dc and ac power otherwise required for NC operation. The power supply is activated when the bomb nav mode selector switch is in ALIGN, or any position above ALIGN, and NC computations are continuous as long as the power supply is activated.

Present Position/Destination Counters. NC computations are keyed primarily to the present position and destination position data programmed into the computer as indicated by the reading of the present position and destination position counters. Actual programmed data for each channel will be within 0.025 minute of the counter reading. Adjacent control knobs allow the operator to manually set these counters to any desired reading. The control knobs are spring loaded to a detented position at which they have no effect on computer operation. Rotation of the control knobs away from the detented position will cause the counters to slew in a direction and rate proportional to direction and magnitude of knob rotation. The knobs are mechanically limited to a maximum angular displacement of approximately 60 degrees in either direction. At this maximum displacement angle, the counters will slew at a rate of approximately 10 degrees of data a minute. Slewing is accomplished by mechanically coupling the mechanical data shaft to a slew motor when the control knob for the data shaft is operated. One motor is used for either present position latitude or destination latitude counter and another is used for either present position longitude or destination longitude counter. Each motor can be driven by only one control at a time, with the present position control having precedence over the destination position control, so that both latitude counters or both longitude counters cannot be set simultaneously. The present position data shafts can be driven by their control knobs only when the fix mode man fix selector button is depressed. The motors are powered by ac voltage from the NC power supply, so that the counters can be set only when the bomb nav mode selector knob is in ALIGN or any position above ALIGN.

Great Circle/Short Range Computations. The latitude and longitude channels contain electronic and synchro devices for computing the various functions involved in the trigonometric solution for ground range and course from the present position data point to the destination position data point. As controlled by the bomb nav mode selector knob, these devices are connected in one configuration to supply spherical trigonometric functions, or in another configuration to supply plane trigonometric functions, including a correction term for longitude convergence. When the bomb nav mode selector knob is in GREAT CIRCLE, spherical trigonometric functions are provided; in all other positions of the bomb nav mode selector knob, except OFF and HEAT, plane trigonometric functions are provided. Spherical trigonometric functions are not limited in range, but due to resolution inaccuracies associated with their mechanization to cover long ranges, these functions become inaccurate at the shorter ranges of less than 200 nautical miles, so that the great circle modes should not be used for navigation legs of less than 200 nautical miles. However, the plane trigonometric functions are limited to approximately a 200 nautical mile range, so that the great circle modes must be used for all navigation legs in excess of 200 nautical miles. The spherical trigonometric functions yield range and course solutions for a great circle navigation leg, while the plane trigonometric functions yield range and course solutions for a flat earth navigation leg. The trigonometric functions generated within the latitude and longitude channels are continuously supplied in parallel to two course and range computers, referred to as the course angle and fixpoint bearing modules.

Ballistics Computer Unit.

The automatic ballistics computer located in the right equipment bay, contains its own power supply and sufficient electro-mechanical computing elements to automatically compute weapon trail and time-of-fall ballistic parameters, when supplied with signals representative of existing ballistic conditions. This computer receives three-phase ac power from the left main bus and is cooled by forced air from the air conditioning system. The computer receives inputs, as follows:

- True airspeed from the CADC
- Rate of change of pressure altitude from the CADC
- Vertical acceleration from the SP
- True heading from the SP
- Inertial velocity east from the SP
- Pitch compensated airspeed from the SP
- Pitch and roll from the flight director system
- Range lock signal from the attack radar
- A time delay signal from the dual bomb timer
- Groundspeed from the NC
- Velocity east from the NC
- Train lead and burst altitude from the burst control panel
- Drift angle from the flight director system
- Primary attitude heading ready signal from the NC
- CCIP and auto bomb signals from the NC
- Weapon station monitor signals from the armament system
- Dive mode signal from the LCOS.

The computer provides outputs as follows:

- Weapon trail to the NC
- Groundspeed times time-of-fall to the NC
- Indicated airspeed reference to the CADC
- True airspeed to the SP and NC separately
- Roll and pitch compensated drift angle to the LCOS
- Roll and pitch compensated depression angle to the LCOS
- Range rate to the LCOS
- Groundspeed to the dual bomb timer.

The BCU power supply is activated when the bomb nav mode selector knob is in any position other than OFF or HEAT, but power is not applied to the computing servo control motors except when a CCIP or auto bomb signal is received from the NC, CCIP or AUTO BOMB should be selected a minimum of 15 seconds prior to release. During preflight, weapon settings must be set into the ballistics computer unit in the right forward electronic equipment bay to pre-program the computer for the type weapon loaded at each station. Station selection signals supplied from the armament select panel provide remote control for selection of the proper preprogrammed data for each station. In the CCIP and auto bomb modes, the computer utilizes the above noted inputs to compute the proper trail and groundspeed times time-of fall parameters for the selected weapon under the existing ballistic conditions. These signals are supplied to the NC for use in computing the weapon release point in the auto bomb mode. In addition, the computer computes a roll/pitch compensated drift angle and a roll/pitch compensated depression angle to drive the LCOS pipper positioning circuits so as to continuously position the pipper on the weapon impact point. Thus these modes may be used in LCOS attack modes to indicate when a weapon release will effect impact on target, by maneuvering the aircraft to position the pipper on target.

Stabilized Platform Malfunction Analysis.

Most of the likely stabilized platform malfunctions will be automatically detected and the platform error lamp will light; however, it is recommended that observation for the following conditions be made throughout the flight to insure against possible undetected malfunctions. The presence of any of the following conditions may indicate SP malfunction.

1. Abnormal disagreement between primary and standby attitude/heading indicators.
2. Sudden rolling or pitching of the aircraft when on autopilot.
3. Unusually large or unexpected drift indications (difference between HSI indicated course and heading/indicated ground track and heading).
4. Aircraft flying either low or high during terrain following, or flying low with 3 "g" fly-ups.
5. Abnormally high rates of attack radar ground velocity or ground auto mode cursor drift.
6. Unequal attack radar side-to-side video uniformity.

Presence of the following conditions may also indicate possible SP malfunction, however, due to meteorological effects which might cause these same conditions, confirmation should be made by checking for presence also of one of the above conditions.

7. Unusual or unexpected change in windspeed and direction.
8. Unusually large or unexpected difference between groundspeed and true airspeed.
9. Excessive buildup of position error.

Note

With one or more of the above indications; disengage/do not engage autopilot, pull-up to safe altitude/do not conduct terrain following operation, do not attempt instrument approaches using the primary ref system, until malfunction is cleared.

Refer to "Caution Lamp Analysis," Section III, for primary attitude/hdg reference failure.

ARMAMENT SYSTEM.

The armament capability of the aircraft includes the delivery of conventional and nuclear weapons in various configurations and air-to-ground and air-to-air gunnery. An M61A1 gun is installed in the right hand weapons bay. Bombing and launching equipment (pylons and stores release system) weapons bay doors, the weapons themselves, and the gun are considered as the armament system. For detailed description and operation of the armament system refer to the applicable delivery manual, T.O. 1F-111E-25-1 for nuclear delivery and T.O. 1F-111E-34-1-1 for conventional delivery.

BOMBING AND LAUNCHING EQUIPMENT.

Bombing and launching equipment consists of the various bomb racks, stationary and pivoting wing pylons, and the release systems. Stores can be 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 stores carriage in various configurations. 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.

Master Arm and Release Switch.

The master arm and release switch (4, figure 1-14), located on the auxiliary gage panel, is labeled MASTER ARM & RELEASE. The switch has two positions marked ON and OFF. The switch must be placed on the ON position before weapon release can be obtained. The OFF position prevents weapon release.

IR Missile Indicator/Pushbutton.

The IR (infrared radiation) missile indicator/pushbutton (1, figure 1-62), located on the weapons control panel, is labeled IR MISSILE. If IR missiles are loaded on the aircraft the letters WPN will be displayed on the pushbutton. When the pushbutton is depressed the letters SEL will appear and each weapon station indicator/pushbutton will display the letters WPN if an IR missile is loaded at that station.

Air/Air IR Missile Switch.

The air/air IR missile switch (13, figure 1-5), located on the left main instrument panel, is marked A/A IR MSL and OFF. When the switch is placed to A/A IR MSL the IR missile can be launched, with the weapon release button. Placing the switch to OFF will prevent launching IR missiles. A red guard must be raised to gain access to the switch.

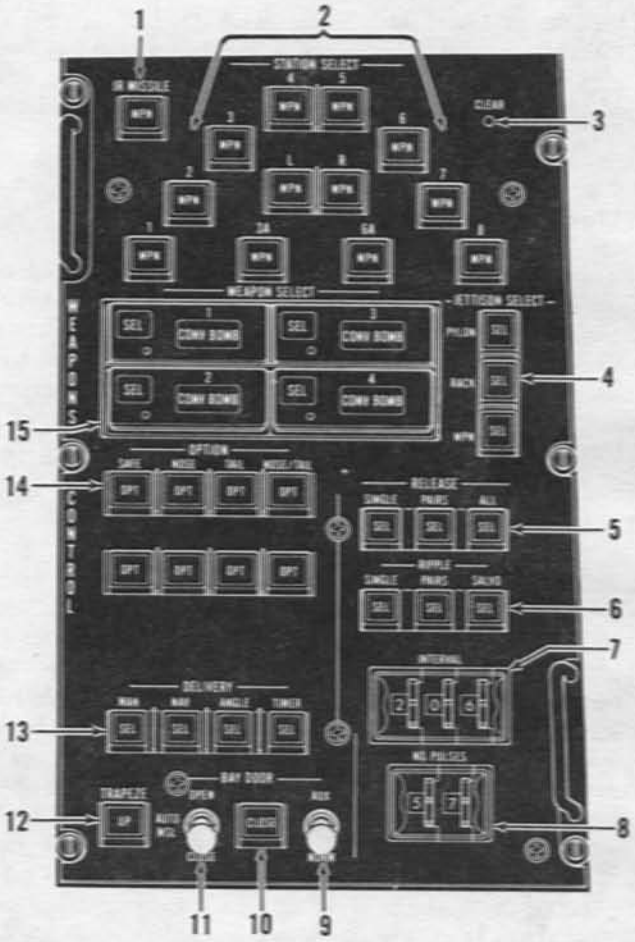
Weapons Station Indicator/Pushbuttons.

Twelve weapons station indicator/pushbuttons (2, figure 1-62), located on the weapons control panel and labeled STATION SELECT, are provided for selection of weapons stations on the aircraft. The pylon stations are numbered 1 thru 8 and correspond to their respective pushbuttons. Pushbuttons 3A and 6A are for outboard shoulder loaded AIM-9B launchers on pylons 3 and 6. Pushbuttons L and R are for the right and left weapons bay stations. Each indicator/pushbutton will display the letters SEL, WPN or blank depending on the condition of selection. When an indicator/pushbutton is enabled by a weapon select cassette/indicator, either WPN or SEL will appear on the pushbuttons. A SEL display indicates that the station has been selected for release and has been accepted by the CPU. A WPN display indicates that the weapon station has been enabled, a weapon is present at that station, and that station has not been selected for release. Depressing the indicator/pushbutton will change the display from WPN to SEL for release. When the indicator/pushbutton is blank the station has not been selected by the weapons select cassette/indicators or a weapon is not present at that station.

Weapon Select Cassette/Indicators.

Four weapon select cassette/indicators (15, figure 1-62), located on the weapons control panel, are labeled WEAPON SELECT. The cassette/indicators provide identification and control of all weapons loaded on the aircraft, and are marked 1, 2, 3 and 4. Cassette no. 1 can control weapon stations 1, 2, 3, 4, 5, 6, 7, and 8, no. 2 can control stations 3, 4, L, R, 5 and 6, no. 3 can control stations 3, 4, 5 and 6, and no. 4 can control stations 1, 2, L, R, 7 and 8. Each cassette contains a weapon type identification window and a code generat-

Weapons Control Panel



1. IR Missile Indicator/pushbutton.
2. Weapons Station Indicator/pushbuttons (12).
3. Weapons Memory Clear Switch.
4. Jettison Select Indicator/pushbuttons (3).
5. Weapon Release Option Indicator/pushbuttons (3).
6. Weapon Ripple Option Indirator/Pushbuttons (3).
7. Intervalometer Set Wheels.
8. Number of Pulse Set Wheels.
9. Weapons Bay Door Auxiliary Control Switch.
10. Weapons Bay Door Position Indicator.
11. Weapons Bay Door Control Switch.
12. Trapeze Position Indicator.
13. Delivery Mode Indicator/pushbuttons (4).
14. Bomb Arming Option Indicator/pushbuttons (4).
15. Weapons Select Cassette/Indicators (4).

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Figure 1-62.

ing tape with a capacity for 21 different weapons. The cassette program tapes have provisions for 14 weapon programs at this time. The ground crew will set the weapon code tape when weapons are loaded on the aircraft. The indicator on the cassette/indicator will display the letters SEL, READY or blank, depending on the selected mode. To select a weapon for program, depress and hold the applicable weapon cassette. At the same time depress the applicable weapons station indicator/pushbuttons for the weapon to be released. The letters SEL will appear on the cassette and the weapon stations. The letters READY will appear on the cassette when necessary logic for release has been met and the master arm and release switch has been placed to the ON position. When the cassette/indicator is blank that weapon has not been selected.

Weapon Status Indicator.

The weapon status indicator (3, figure 1-14), located on the auxiliary gage panel, is labeled WEAPON STATUS. The indicator will display the letters READY when necessary weapon logic for weapon release has been set on the weapons control panel and the master arm and release switch has been placed to the ON position.

Jettison Select Indicator/Pushbuttons.

Three jettison select indicator/pushbuttons (4, figure 1-62), located on the weapons control panel, are labeled JETTISON SELECT and are provided for selective jettison of fixed pylons, bomb racks, and weapons. The pushbuttons are marked PYLON, RACK and WPN, which display SEL when depressed or will be blank when not selected. When the pylon pushbutton is depressed to SEL, all weapons station indicator/pushbuttons will display WPN if a store is present. To jettison the fixed pylons, stations 1, 2, 7 and 8 must be depressed to SEL, the master arm and release switch must be on, and the weapon release button depressed. The MAU-12 C/A racks will fire on all other selected stations in this condition.

CAUTION

Pylon jettison can be accomplished with the flaps extended, however, flap damage will probably occur.

The rack pushbutton operates the same as the pylon except only the MAU-12 C/A bomb rack will fire and jettison all loaded BRU/TER racks, tanks and weapons. The weapon pushbutton works the same as the rack pushbutton except only the weapons will jettison from the BRU/TER, or MAU-12 C/A bomb racks.

Delivery Mode Indicator/Pushbuttons.

The four delivery mode indicator/pushbuttons (13, figure 1-62), located on the weapons control panel, are labeled DELIVERY. The pushbuttons provide a means of selecting the source of signal for weapon release. Each pushbutton will display the letters SEL if selected or will go blank if another mode is selected. The pushbuttons are marked and function as follows:

Note

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

MAN — provides manual release using the weapon release buttons mounted on either control stick grip.

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

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.

Weapon Release Option Indicator/Pushbuttons.

The three weapon release option indicator/pushbuttons (5, figure 1-62), located on the weapons control panel, are labeled RELEASE. The pushbuttons provide a means of selecting one or more weapons to be released at the same time. Each pushbutton will display the letters SEL if selected or will go blank if another release option is selected. The pushbuttons are marked and function as follows:

- SINGLE—A weapon will be released each time the weapon release button is depressed. Weapon release will follow an outboard to inboard station sequence. When corresponding left and right stations are selected, the first station selected will release first and then alternate until both stations are empty. Also if corresponding stations are selected all weapons must be dropped prior to system stepping inboard. Weapon release will alternate between symmetrical stations. If unsymmetrical stations are selected (1 and 5) the most outboard station weapons must be dropped prior to system stepping inboard. Whenever the difference between the total number of selected left and right fixed pylons (1 and 2 or 7 and 8) exceeds one, release cannot be accomplished.

- PAIRS—Same as singles except releases occur from each station of a symmetrical pair simultaneously. Both stations of a symmetrical pair must be selected for a release to occur.
- ALL—Same as pairs except releases will occur from all stations symmetrically.

Bomb Arming Option Indicator/Pushbuttons.

Four bomb arming option indicator/pushbuttons (14, figure 1-62), located on the weapons control panel, are labeled OPTION. The pushbuttons operate in conjunction with the weapons select indicator/pushbuttons to provide 28 volt dc power from the main dc bus to arm or safe the fuzing systems of conventional bombs. The pushbuttons will display SEL when selected for an arming option or will go blank if no conventional bombs are selected. The pushbuttons are marked and function as follows:

- SAFE—Will allow both nose and tail arming wires to pull out of the arming solenoids and stay in the fuzes as the bombs are released, thereby rendering the weapon safe.
- NOSE—Provides electrical power to retain only the nose fuze arming wire, thereby arming the nose fuze at bomb release.
- TAIL—Provides electrical power to retain only the tail fuze arming wire, thereby arming the tail fuze at bomb release.
- NOSE/TAIL—Provides electrical power to retain both nose and tail fuzes arming wires, thereby arming both at bomb release.

Weapon Ripple Option Indicator/Pushbuttons.

The three weapon ripple option indicator/pushbuttons (6, figure 1-62), located on the weapons control panel, are labeled RIPPLE. The pushbuttons are marked SINGLE, PAIRS and SALVO and when selected will display the letters SEL for that mode. The selection of a ripple release sequence will enable the intervalometer to generate a given number of release pulses at a given interval for selected release.

Intervalometer Set Wheels.

Three intervalometer set wheels (7, figure 1-62), located on the weapons control panel, are labeled INTERVAL. A digital counter located by each set wheel provides a readout for setting the desired weapon release time interval. The three set wheels have a range of 10 to 999 milliseconds in 1 millisecond intervals. Due to the rack characteristics the minimum release interval is 50 milliseconds. The intervalometer is activated any time one of the weapon ripple option indicator/pushbuttons is depressed to SEL.

Number of Pulse Set Wheels.

The number of pulse set wheels (8, figure 1-62), located on the weapons control panel, are labeled NO. PULSES. A digital counter is located adjacent to each set wheel to provide readout for selected settings. The set wheels work in conjunction with the intervalometer, to generate a desired number of time interval signals. The set wheels have a range of 0 to 90 in 1 increment steps.

Weapons Memory Clear Switch.

The weapons memory clear switch (3, figure 1-62), located on the weapons control panel, is labeled CLEAR. The switch is used to clear the memory logic when each weapons select indicator/pushbutton is depressed to SEL. When the switch is depressed, the weapons select cassette/indicators and associated weapons station indicator/pushbuttons will go blank. The switch is depressed by inserting a sharp object into the hole and must be pushed down for each activation.

Nuclear Consent Switch.

The nuclear consent switch (5, figure 1-14), located on the auxiliary gage panel 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 selected. Placing the switch to REL ONLY position enables unlocking of the bomb racks at all stations selected. With the switch in the OFF position all power is removed from the bomb rack inflight safety lock and nuclear weapon arming circuits. A red guard must be raised to gain access to the switch. When nuclear weapons are carried the guard, which is a slotted flat plate, is safetied and sealed down to hold the nuclear consent switch in the OFF position. The switch position is to be monitored visually whenever nuclear weapons are carried on the aircraft.

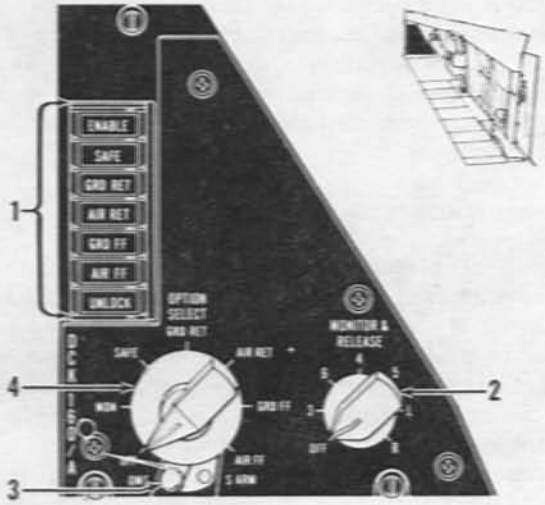
Nuclear Weapons Monitor and Release Knob.

The nuclear weapons monitor and release knob (2, figure 1-63), located on the nuclear weapons control panel has seven positions marked OFF, 3, 6, 4, 5, L & R. Placing the knob to OFF opens the monitoring and release circuits to all stations. Position 3, 6, 4, 5, L & R 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 (4, figure 1-63), located on the nuclear weapons control panel, is labeled OPTION SELECT. The knob has seven positions marked OFF, MON (monitor), SAFE, GRD RET (ground re-

Nuclear Weapons Control Panel



1. Nuclear Weapons Monitor Lamps (7).
2. Nuclear Weapons Monitor and Release Knob.
3. Nuclear Weapons Arm Knob Lock Lever.
4. Nuclear Weapons Arm Knob.

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Figure 1-63.

tard), AIR RET, GRD FF (ground freefall), and AIR FF. When the knob is in the OFF position, 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 and station select buttons 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 (3, figure 1-63), 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 1-63), 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 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 Caution Lamp.

The nuclear caution lamp, located on the main caution lamp panel (figure 1-37), will light when one or more of the following conditions exist:

- A nuclear weapon rack is unlocked when not commanded.
- Loss of the monitoring circuit continuity to any nuclear weapon being monitored.
- The bomb arming device remains in an intermediate (not armed, not safe) condition.
- If the weapon is armed when not commanded.

When lighted the word NUCLEAR will appear in the face of the lamp.

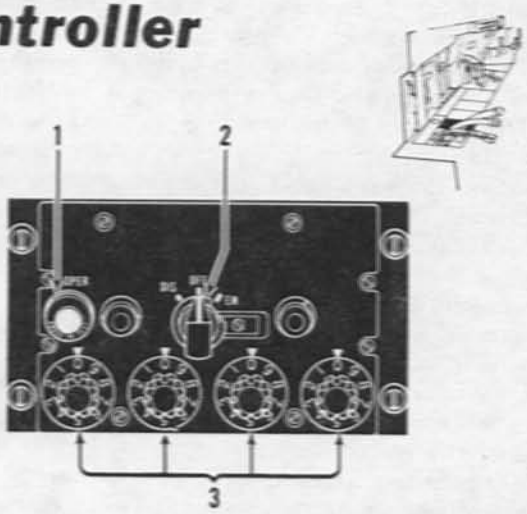
Permissive Action Link (PAL) Controller.

Aircraft modified by T.O. 1F-111E-506, are equipped with the permissive action link (PAL) controller, figure 1-64), located on the left console. The PAL controller allows enabling of nuclear bombs so that they can be prearmed. For detailed description and operation of the PAL controller refer to T.O. 1F-111E-25-1.

Weapon Release Buttons.

Two weapon release buttons (1, figure 1-24), 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 aircraft.

Permissive Action Link (PAL) Controller



1. Operate Lamp
2. Enable Control Knob
3. Code Select Knobs (4)

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Figure 1-64.

External Stores Jettison Button.

The external stores jettison button (4, figure 1-5), located on the left main instrument panel, is a flush mounted pushbutton labeled EXT STORES JETTISON. Depressing the button, with power to the dc essential bus, will not jettison the pivot pylons. Depressing and holding the button will fire the MAU-12C/A racks after a 0.25 second safing delay. If the flaps are fully retracted, the fixed pylons will be jettisoned 0.50 second after MAU-12C/A firing. Nuclear weapons will be jettisoned only if the MAU-12C/A is unlocked. The button will not jettison the AIM-9B missile.

Pylons.

The aircraft can be equipped with eight detachable pylons mounted along the lower surface of the wing. The pylons are designed to accommodate the MAU-12C/A bomb rack or 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. After the weapons carried on these pylons positions are expended, the fixed pylons can be jettisoned 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 54 degrees with certain weapons loaded on the inboard pivoting pylons to prevent damage to the fuselage. Pylon weapon station indicator/pushbuttons, located on the weapons control panel, are provided to select individual pylons stations for release or launch. Each button is numbered corresponding to the station it controls.

Stores Release System.

The various stores carried on the aircraft are released by electrical signals generated by the bomb nav system, lead computing optical sight, dual bombing timer or by manually depressing the weapons release button. To accomplish a release, except jettison, the applicable weapon cassette/indicator and weapons station indicator/pushbuttons must be depressed to SEL, the master arm and release switch must be on, the applicable weapons select cassette/indicators and weapon status indicator read READY and either weapon release button on the control stick depressed. Refer to "Bombing-Navigation System" this section, for the type of release that can be made for the various stores carried.

Dual Bombing Timer. The dual bombing timer (figure 1-65), 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 counters marked PULL UP and RELEASE. Each may be set in increments of 0.1 of a second by means of knobs on each side of the timer. The pull up counter must be set at a minimum of 0.2 seconds to assure lighting of the pull up lamp. The release counter must be set at a minimum of 0.2 seconds to assure bomb release. The timer is used with the delivery mode timer indicator/pushbutton when employing loft or straight flyover bombing delivery tactics. For a loft bombing delivery 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 counters and the weapon is released at the expiration of these two times. For a fly over release the precomputed time from IP to target may be divided and set in both counters. When only a release signal is required the predetermined time to release may be divided into two parts and set into the bombing timer counters in any manner as long as the release counter setting is greater than 0.2 seconds. If the time to release is equal or less than 30 seconds, all of this time should be set into the release counter with the pull up counter set to zero. The total of the two must equal the time

Dual Bombing Timer

1. Pull Up Counter.
2. Release Counter.
3. Release Counter Set Knob.
4. Pull Up Counter Set Knob.

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Figure 1-65.

from IP to weapon release. When making a bomb run, either weapon release button 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 he pull up counter a lamp 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 a manual constant 4 "g" pull up using the airspeed mach indicator accelerometer for "g" reference must be initiated. The weapon will be released at the expiration of release time and a green release lamp will momentarily light displaying the words BOMB RELEASE. If a straight fly over laydown delivery is being accomplished, continue to hold altitude, heading, and airspeed through the expiration of release time to obtain weapon release. BCU inputs to the LCOS are frozen when the weapons release button is depressed in TIMER delivery mode. The timer receives 28 volt dc power from the 28 volt dc essential bus through the master power switch on the armament select panel.

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 2½ seconds. The alternate system takes approximately 30 seconds to open or close the doors. A ground safety switch and lockpin are provided to assure safe ground crew operations. The weapon bay doors are controlled by a weapons bay door switch. The right weapons bay door is replaced by the weapon bay gun module when it is installed. Refer to Section IV for normal and alternate operating procedures for the weapons bay door(s).

CAUTION

To prevent damage to the door actuator motor during alternate operation, there must be a 10-second interval between opening and closing the door(s) and no more than 3 complete door opening and closing cycles within a 15 minute period.

Weapons Bay Door Control Switch.

The weapons bay door control switch (11, figure 1-62), located on the weapons control panel, is a three position switch marked OPEN, AUTO MSL and CLOSE. When the switch is placed to OPEN the doors will open. Placing the switch to CLOSE will close the doors. The AUTO MSL position is inoperative.

Weapons Bay Door Auxiliary Control Switch.

The weapons bay door auxiliary control switch (9, figure 1-62), located on the weapons control panel, has two positions marked AUX and NORM. When the switch is placed to the NORM position the weapons bay doors operate on hydraulic pressure from the utility hydraulic system. Placing the switch to AUX provides electrical power to operate the weapons bay doors.

Weapons Bay Door Position Indicator.

The weapons bay door position indicator (10, figure 1-62), located on the weapons control panel, is a flip-flop type indicator. When the weapons bay doors are closed, the letters CLOSE will be displayed. When the

doors are open the letters OPEN will be displayed. A crosshatch pattern indicates the doors are opening or closing.

Trapeze Position Indicator.

The trapeze position indicator (12, figure 1-62), located on the weapons control panel, is inoperative since a trapeze is not installed.

Weapons Bay Door Ground Safety Switch and Lockpin.

A ground safety switch and a ground safety lockpin are located behind an access door just aft of the left weapon bay door. See figure 1-16. The switch is labeled WEAPON BAY DOOR and has two positions marked SAFE and NORMAL. When the switch is in the SAFE position, electrical and hydraulic bay door operation is disabled. Placing the switch in the NORMAL position enables normal weapon bay door operation. The ground safety lockpin mechanically prevents rotation of the drive shaft thereby inhibiting operation of the weapon bay doors. The access door covering these safety devices is constructed so that it cannot be closed and latched with either the ground safety lockpin installed or with the ground safety switch positioned to SAFE.

WEAPONS.

Refer to Section V for a list of authorized munitions and delivery limitations.

GUNNERY EQUIPMENT.

The aircraft is equipped to carry a weapons bay gun in the right side of the weapons bay.

Weapons Bay Gun.

The weapons bay gun provides both air-to-air and air-to-ground gunnery capability. The gun 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 millimeter 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 safety switch on the main landing gear prevents firing the gun on the ground. In flight, the gun will not fire unless the weapons bay area is vented. On aircraft 68 ▸ and those modified by T.O. 1F-111-645, the gun vent ports will open automatically each time the gun is fired. The gun is fired by 28 volt dc power from the essential dc power panel. A rounds counter provides an indication of remaining ammunition.

Gun/Camera Control Switch.

The gun/camera control switch (12, figure 1-5), located on the left main instrument panel, is marked GUN/CAMERA, OFF and CAMERA ONLY. With the switch in the OFF position the gun cannot be fired. Placing the switch to GUN/CAMERA enables firing the gun. The CAMERA ONLY position is inoperative.

Rounds Counter.

The rounds counter (2, figure 1-14), located on the auxiliary gage 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.

Gun Triggers.

Two gun triggers (5, figure 1-24), one located on each control stick grip, are provided to fire the guns. Depressing either trigger will fire the weapon bay gun depending on the position of the gun/camera control switch.

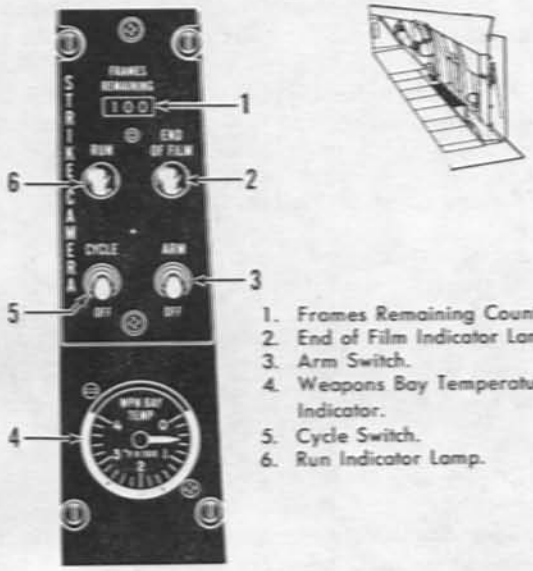
Gun Hot Caution Lamp.

The amber gun hot caution lamp (11, figure 1-5), located on left main instrument panel is used to monitor the ambient temperature of the gun. The lamp will come on at approximately 240 degrees F. The gun should not be fired when the lamp is lighted.

Weapons Bay Temperature Indicator.

The weapons bay temperature indicator (4, figure 1-66), located on the strike camera control panel, provides an indication of the ambient temperature in the weapons bay. The indicator has three colored bands to provide indications from 0 to 400 degrees F. The green band indicates from 0 to 160 degrees, amber band from 160 to 260 degrees, and red band from 260 to 400 degrees.

Strike Camera Control Panel



1. Frames Remaining Counter.
2. End of Film Indicator Lamp.
3. Arm Switch.
4. Weapons Bay Temperature Indicator.
5. Cycle Switch.
6. Run Indicator Lamp.

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Figure 1-66.

STRIKE CAMERA (KB-18A).

The aircraft is equipped with a KB-18A strike camera to provide low level daylight photographic reconnaissance. The camera is mounted in the nose of the aircraft in the forward electronics bay. It consists of the KB-18A camera, the film magazine and a control box. Panoramic photo coverage of an area 180 degrees along and 40 degrees across the flight path is provided. This is accomplished by rotating a double dove prism in front of the lens on an axis perpendicular in line with the direction of flight and simultaneously advancing the film across a narrow aperture at the focal point of the camera. The film advance is synchronized with the prism rotation so that the image is transferred to the film as the prism scans fore to aft. The format size of the resulting photographs is 2.25 by 9.4 inches. Camera speeds of 1, 2, or 4 frames per second are provided. Camera overrun time of 0 to 20 seconds (in 2 second increments) and 32 seconds are available and exposure index for different film speeds (40, 64, 80 and 200) can be selected for the type of film being used. These data must be set into the system by ground maintenance personnel. On aircraft modified by T.O. 1F-111-856, the camera has been removed.

CONTROLS AND INDICATORS.

Arm Switch.

The arm switch (3, figure 1-66), located on the strike camera control panel, is marked ARM and OFF. When the switch is placed to ARM, power is applied to the camera arming circuit for the camera to operate. The OFF position removes power from the camera arming circuit. The camera will operate automatically when a weapon release signal is received from the armament system. Operation will continue for the preset overrun time.

Cycle Switch.

The cycle switch (5, figure 1-66), located on the strike camera control panel is marked CYCLE and OFF. Placing the switch to CYCLE provides a means of manually operating the camera.

Run Indicator Lamp.

The run indicator lamp (6, figure 1-66), located on the strike camera control panel is labeled RUN. The lamp will come on any time the camera is operating.

End of Film Indicator Lamp.

The end of film indicator lamp (2, figure 1-66), located on the strike camera control panel is labeled END OF FILM. The lamp will come on any time the film is broken or the camera is out of film.

Frames Remaining Counter.

The frames remaining counter (1, figure 1-66), located on the strike camera control panel is labeled FRAME REMAINING. The counter displays the number of frames remaining in the camera to be exposed. A maximum of 300 frames is available when the camera is fully loaded.

ATTACK RADAR (AN/APQ-113).

The attack radar provides all weather navigation, air-to-ground and air-to-air attack capability. Basic components of the radar set consist of an antenna assembly, an antenna roll unit, and an antenna control, all located within the nose radome; a modulator-receiver-transmitter (MRT) and an electrical synchronizer, located in the left forward electronics equipment bay; and a radar scope panel, a radar control panel, and a tracking control handle, all located at the right crew member's station. The antenna automatically scans in azimuth either ±45 degrees about the longitudinal axis of the aircraft (or ±10 degrees about a movable azimuth cursor) and is automatically stabilized in pitch and roll by signals from the bomb nav system. The antenna may

be positioned in elevation (tilt) within ± 30 degrees of the horizontal by a depression angle signal provided by the bomb nav system or by manual operation of a tilt control. For location of the antenna, see figure 1-69. The roll unit is attached to the most forward bulkhead of the aircraft and serves primarily as a mounting and roll-stabilized platform for the antenna assembly and for two antenna-receiver units of the terrain following radar (TFR). The antenna control provides for the proper positioning of the antenna and the roll platform. It senses and compares antenna and roll platform position with the bomb nav system command and pilot command inputs for movement about the four gimbals; azimuth, pitch, tilt, and roll. If a difference exists between sensed position and input positioning commands, the antenna control provides drive power to correct the applicable position. The MRT modulator provides high voltage pulses for generating high power RF energy from the magnetron, which energy is then transmitted in pulses at a random or set frequency between 16.0 and 16.4 gigahertz. The returned echo signal is received, amplified, and applied as video to the radar scope for display. The picture seen is a plan position indicator (PPI) display with the origin or aircraft position offset one radius to the bottom of the radar scope during operation in two ground modes (ground manual and ground automatic) and in the air mode, or offset up to six radii in the other mode (ground velocity). During random frequency operation (AFC-1), the magnetron output frequency sweeps through the frequency band which provides a measure of immunity to many types of jamming and improves the stability of returns on the display. The electrical synchronizer provides system timing for the attack radar, generates range marks for the radar scope display, provides automatic angle tracking of air targets and supplies range and range rate information to the lead computing optical sight (LCOS) for gun and missile firing control, generates precision range and azimuth cursors, supplies the receiver with signals for automatic gain control, and monitors radar operation for in-flight malfunction detection and isolation of the malfunction during self test operation. The radar scope panel provides range data, a radar scope display, and tuning controls for the radar scope. Power, mode, and function controls on the control panel cause corresponding signals to be sent to the units of the attack radar. The tracking control handle is used to control antenna tilt in air mode of operation, change antenna scanned sector from ±45 degrees to ±10 degrees, provide rapid slewing of the range cursor, and position the azimuth and range cursors for fix-taking, bombing, and target tracking. Self test features incorporated into the radar are used for preflight, inflight, and maintenance malfunction analysis and trouble shooting. The attack radar operates on 115 volts ac power from the left main ac bus and 28 volts dc power from the main dc bus. The system has three ground modes of operation (ground manual, ground automatic, and ground velocity) and one air mode (air). Tie-ins between the attack radar and other aircraft systems are shown in figure 1-74.

ATTACK RADAR SCOPE RECORDING CAMERA.

A recording camera located in the attack radar scope panel is provided to take exposures of the radar scope display. 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. The camera is activated by depressing the manual photo button on the radar scope panel, or by a weapon release signal from the bomb nav system, and will operate for one scan starting at the first antenna turn around after the operate signal is received from either source. On aircraft modified by T.O. 1F-111E-507, automatic radar scope photography capability is provided. A scope photo control panel (figure 1-68) located on the aft end of the center console contains an intervalometer and an associated power switch marked POWER and OFF. The intervalometer knob provides a time setting from 1 to 24 seconds. When automatic radar scope photography is desired, the knob is set to the time interval desired and the power switch is placed to POWER. This will generate a timed pulse to the scope camera to take a picture at the interval setting. A photo malfunction lamp on the attack radar scope panel will light while an exposure is made. The film magazine provides a maximum of 1300 frames of 35-millimeter film and a readout window on the magazine shows the percentage of film remaining. Simultaneous film exposure of a 24 hour clock, data slate and 12 code lamps is made with each scope exposure to identify each frame of the film as shown in figure 1-67. The clock shows time of exposure and the data slate may be filled in to show operator's name, date, aircraft number, target etc. The camera film magazine is mounted through the front of the attack radar scope panel by positioning the end slot of the knurled drive shaft on the rear of the magazine horizontally when the magazine is in the normal mounting position, then inserting the magazine in its respective hole and pushing in until contact is made with the internal electrical connections. The film magazine removal handle is then rotated clockwise approximately 90 degrees and folded down to engage its latching mechanism. To remove the magazine depress the pushbutton to unlatch the magazine removal handle. Rotate the handle approximately 90 degrees counterclockwise and pull out.

ATTACK RADAR AIR MODE.

The air mode is used to detect and track airborne targets. During the search and detection phase, the radar forms a pencil beam antenna pattern and the antenna is programmed for a box scan of 2 degrees in

Photo Data Recording

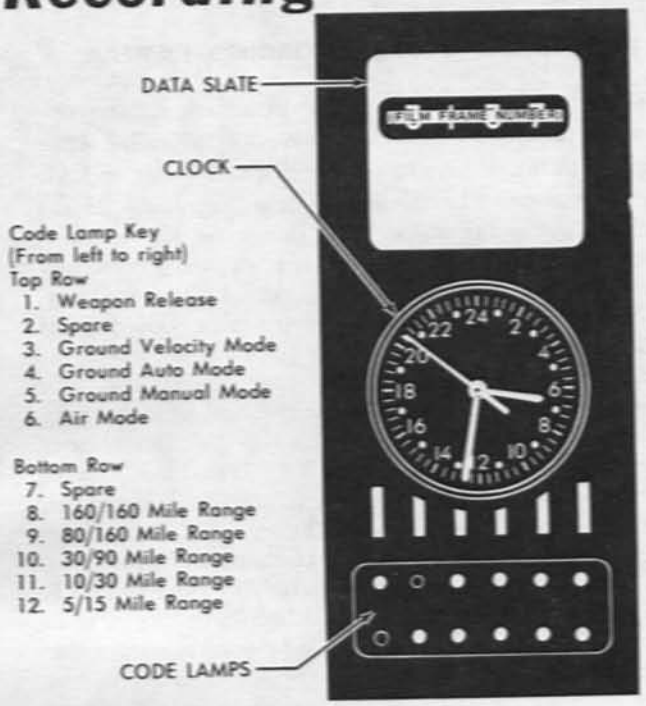


Figure 1-67.

elevation and 90 degrees in azimuth. The pointing elevation of the antenna, to a maximum of ±30 degrees with reference to the horizon under normal stabilization, is controlled by fore and aft movement of the tracking control handle with the enable switch depressed. Selection of a target to be tracked is accomplished by using the tracking handle to place the azimuth cursor within 10 degrees of the desired target and the range cursor to a lesser range than the range to the target. Acquisition is initiated by placing the sector switch on the tracking handle to the forward position, which reduces the area scanned in azmuth to 20 degrees centered about the azimuth cursor position. The range cursor begins to step out in range at each antenna azimuth turnaround and the range gate samples the area scanned for the chosen target. Once the target falls within the range gate, the range cursor will cease to step in range and will disappear from the scope presentation. A portion of the azimuth cursor will be blanked at the range of the target and the cursor will be automatically repositioned to be in azimuth alignment with the target. After the target is acquired, the lock indicator lamp will light within four seconds and tracking of the target is automatically performed in range, azimuth, and elevation. The operation of the box scan during the tracking operation is such that it keeps the elevation of the antenna pointed in a direction to keep the target in the vertical center of the azimuth scan. This is indicated by the lighting of both elevation tracking lamps (arrows) located on the bezel of the radar scope. Information supplied to the LCOS during the tracking operation consists of a signal indicating that a target is being tracked, the range to the target, and the rate of change of range to the target. The range to the target is also displayed on the range readout on the radar scope panel. Target lock will be broken if any of the following occurs: (1) wide antenna scan is selected, (2) the range cursor is moved off the target, (3) a mode other than air mode is selected, (4) or loss of detection occurs during a four seconds interval. If lock is broken because of loss of detection or because the range cursor position is changed, the range cursor will begin to step out in range in search of a target. In the air mode, the IF gain is a fixed value.

ATTACK RADAR GROUND MODES.

Three ground modes of operation are provided primarily to enable target identification for radar navigation, radar fixtaking, and fixed angle or automatic synchronous bombing. The use of these ground modes,

Scope Camera Control Panel

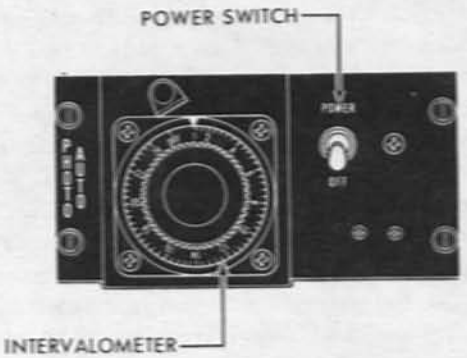
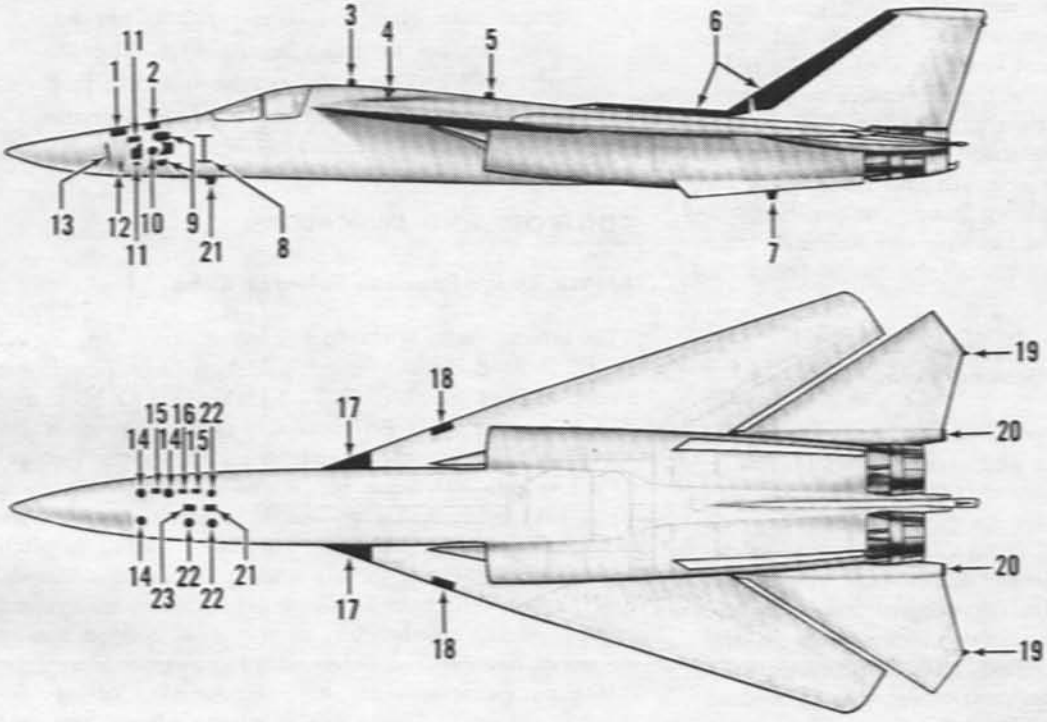


Figure 1-68.

Antenna Locations



1. Glide Slope.
2. ADF.
3. IFF (Upper) and UHF Data Link.
4. Radio Beacon Set (AN/URT-27).
5. UHF #1 and TACAN Upper.
6. HF.
7. IFF Lower.
8. Localizer.
9. Low and Medium Frequency Radar Homing (4).
10. Forward Radar Warning.
11. High Frequency Radar Homing (4).
12. TFR (2).
13. Attack Radar.
14. AN/ALQ-94 (3).
15. Radar Altimeter.
16. ALR-41 (1).
17. AN/ALQ-94 (12).
18. ALR-41 (4).
19. Aft Radar Warning (2).
20. AN/ALQ-94 (6).
21. UHF #2 and TACAN Lower.
22. AN/ALQ-94(3).
23. Marker Beacon.

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Figure 1-69.

in conjunction with other avionics systems, also enables the radar to be employed for airborne instrument low approach (AILA) landings, altitude calibrations, and air-to-ground ranging in which the attack radar computes range and range rate, for use by the LCOS, to targets detected by the TFR. The ground modes are ground manual (GND MAN), ground automatic (GND AUTO), and ground velocity (GND VEL). In each of the three ground modes of operation, a pie-shaped radar map of the terrain ahead of the aircraft is displayed on the radar scope with a maximum range of from 5 to 160 nautical miles selectable in five steps by the crew member. The antenna forms a fan beam vertical pattern which is optimized to cover a range of from 8 to 80 nautical miles at an altitude of 40,000 feet. The depression angle (tilt) of the antenna is adjusted either manually or automatically, depending upon the ground mode selected, to cover the range selected with the optimized antenna pattern. The fan beam scans a 90-degree azimuth sector ahead of the aircraft centered on the aircraft heading. The radar scope presentation is normally stabilized with drift angle signals supplied from the bomb nav system, so that the direction of the aircraft ground track is displayed vertically from the sweep origin (aircraft position).

Note

During ground operation, with the attack radar operating in either ground manual with the antenna stabilized (cage lamp out) or the ground auto or ground velocity mode, the sweep vertex on the radar scope may swing to the left or right or both. With the mode selector switch in ground manual and the antenna caged (cage lamp lighted), the sweep vertex shall be stabilized. In the ground auto or ground velocity mode of operation, if the sweep vertex is unstable, it shall stabilize when the bomb-nav system supplies a drift angle (difference between the true heading and ground track counters on the navigation computer) to the attack radar. To keep the sweep in view on the radar scope, the drift angle should be limited to an angle of between ± 45 degrees. To connect the drift signal from the bomb nav system to the attack radar system, while the aircraft is on the ground, it is necessary to depress the instrument test switch. This connection is accomplished automatically, through the landing gear ground safety switches, when aircraft weight is off the gear.

Ground Manual Mode.

The ground manual mode is a backup mode used principally when failure occurs in the other two ground modes which renders them unuseable. In ground manual the antenna depression angle is controlled manually by use of a tilt control knob located on the radar scope panel. This permits the crew member to position the fan beam to scan terrain closer in or farther out from the aircraft for optimization of the display. The crew member may obtain azimuth and range data for a target by using the tracking handle to position the cursors over the target and reading the azimuth from the bezel of the radar scope and the range from the readout on the panel.

Ground Automatic and Ground Velocity Modes.

The ground auto and ground velocity modes are used primarily during fixtaking and synchronous bombing. The depression angle of the antenna is adjusted automatically and the crosshairs are positioned by signals from the bomb nav system. Angular corrections to the depression angle may be made for display refinement by manually adjusting the tilt control knob on the radar scope panel. The navigation computer in the bomb nav system is supplied with information by movement of the tracking control handle to update destination or present position latitude and longitude (the destination/present position selector switch on the control panel determines which latitude and longitude are updated). This in turn updates the target or nav fixpoint range bearing. The attack radar operates the same in either ground auto or ground velocity mode except that in the latter mode the target or fixpoint and the crosshairs are automatically maintained at the center of the radar scope presentation if crosshair synchronization is correct. The display is ground velocity stabilized so that the radar map seen on the display is stationary. The display is also magnified in the ground velocity mode by as much as three (3) times since the operator may select any of the following miles diameter/range settings with the range selector: 5/15, 10/30, 30/90, 80/160, or 160/160. In defining this terminology, the 5/15 setting will be used as an example. The maximum offset of the origin or aircraft position from the target (or fixpoint) and crosshairs is 15 nautical miles and the useful display diameter represents 5 nautical miles centered about the crosshairs. Movement of the tracking handle during operation in the ground auto mode will drive the destination or present position counters in the bomb nav system for repositioning the azimuth and range cursors. Movement of the tracking handle during operation in the ground velocity mode will drive the destination or present position counters in the bomb nav system to move the display in azimuth and range, with the azimuth and range cursors remaining fixed in the center of the display.

CAUTION

Do not leave the radar in GND VEL and narrow scan after the turn point/target has passed aft of the radar scan limits. The antenna will oscillate against the limiting stop and may result in reduced life of the antenna azimuth drive.

CONTROLS AND INDICATORS.

Attack Radar Function Selector Knob.

The attack radar function selector knob (3, figure 1-70), 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 1-70), located on the attack radar control panel, has four positions marked GND MAN (ground manual), GND AUTO (ground automatic), GND VEL (ground velocity), and AIR. In the GND MAN position the range and azimuth cursors are positioned with the tracking control handle and antenna tilt is positioned with the antenna tilt control knob independently of the navigation computer. In the GND AUTO position the cursors and antenna tilt are automatically positioned by signals from the bomb nav system. The tracking control handle is used to correct the bomb nav system present position and destination counters and the tilt control knob is used to refine the scope display. 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 remains in the center of the display. In the AIR position the antenna is programmed for a box scan which can be raised or lowered in elevation with the tracking control handle during search, detection, and acquisition of an air target. Once a target has been located on the radar, automatic tracking in range, azimuth, and elevation can be acquired, i.e., lock-on.

Note

When rotating the attack radar mode selector knob allow 1 second for each mode. Rapid cycling of the mode selector knob through two or mode modes may produce conditions requiring a master reset action. Necessity for reset may be indicated by an abnormal radar display such as distorted display, cursor hang-up, or erratic range readout. Master reset is an internally generated signal that places the attack radar computer circuitry in a known state of valid operation and is initiated by placing the mode selector knob briefly in the GND AUTO or GND VEL position.

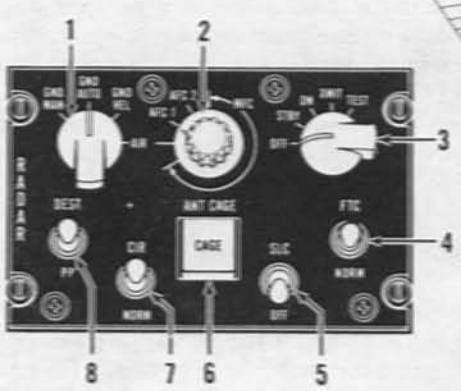
Attack Radar Frequency Control Knob.

The frequency control knob (2, figure 1-70), 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 tune control knob, located on the radar scope panel. The MFC position of the knob is variable over a range between the 12 and 9 o'clock positions. In this position the transmitter operates in a mid-band fixed frequency and the receiver is manually tunable by adjusting the knob over the MFC range. Due to system design two positions are available in the MFC range, both of which provide a radar display. Coarse tuning the receiver to either position may be assisted by observing the system malfunction lamp. This lamp will light when the knob is turned to MFC, and go out when the knob is in either of the two possible positions. Fine tuning the receiver is then carried out by observing the display.

Attack Radar Fast Time Constant Switch.

The fast time constant switch (4, figure 1-70), 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. This function is particularly important in breaking out a specific target situated in a large industrial complex. Returns from targets very close together tend to overlap on the scope making positive identification of the desired target difficult. Placing the switch in the FTC position provides leading edge discrimination of all returns which highlights the leading edge of targets, blanks

Attack Radar Control Panel



1. Mode Selector Knob.
2. Frequency Control Knob.
3. Function Selector Knob.
4. Fast Time Constant Switch.
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 1-70.

out the trailing edge, and provides a much clearer assessment of the relative position or pattern of the complex. The FTC position of the switch is also used to minimize the effects of jamming in any mode of operation. In the NORM position anti-jamming capabilities are inoperative.

Note

FTC should be used in the air mode of operation when obvious jamming signals are present on the attack radar display or to reduce the effect of ground clutter during target detection.

Attack Radar Side Lobe Cancellation Switch.

The side lobe cancellation (SLC) switch (5, figure 1-70), 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.

Antenna Polarization Switch.

The antenna polarization switch (7, figure 1-70), 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 1-70), located on the attack radar control panel, is a two position switch marked DEST (destination) and PP (present position). The switch position determines which bomb nav system counters are updated in GND AUTO and GND VEL modes.

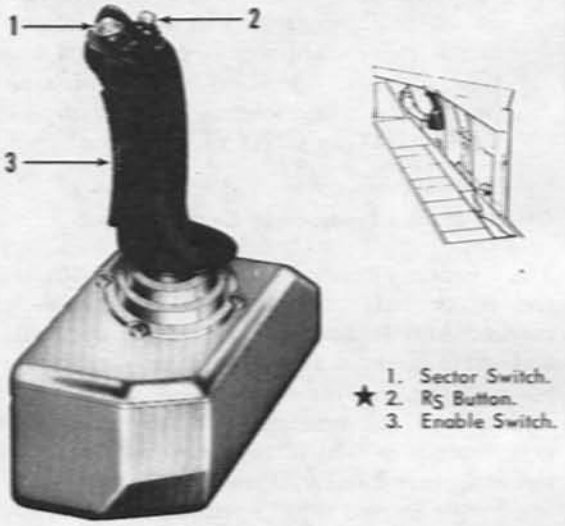
Attack Radar Tracking Control Handle.

The tracking control handle (figure 1-71) is mounted on a pivot pedestal on the right side of the right crew 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 after depressing the pivot pin. A blade type enable switch (3, figure 1-71), recessed in the front of handle must be depressed and held to activate the handle. When operating in the air mode 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 (R_s) Button.

The range search button (2, figure 1-71), located on the right top of the tracking control handle, is used in the air mode of operation only. The button has a red cap, is labeled Rs, and must be depressed to be effective. With the sector switch (1, figure 1-71), in the aft position (wide antenna scan), depressing the range search button permits the range cursor to be slewed rapidly to any desired position on the sweep to a maximum range of 124 nautical miles, by moving the tracking handle fore or aft. When the button is released, the range cursor will remain stationary after slewing. With the sector switch in the forward position (narrow scan), depressing the range search button overrides range lock (if established) and permits the range

Attack Radar Tracking Control Handle



1. Sector Switch.
★ 2. R_S Button.
3. Enable Switch.

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Figure 1-71.

cursor to be slewed rapidly to any desired position on the sweep to a maximum range of 30 nautical miles, by moving the tracking handle fore and aft. When the button is released, range seaching will resume from the point at which the range cursor was positioned by slewing.

Attack Radar Sector Switch.

The sector switch (1, figure 1-71), located on the left top of the tracking control handle, is labeled SECTOR and is a two position, thumb actuated, toggle 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 air mode, the sector switch must be placed in the forward position to cause the range tracking circuitry to commence tracking a target.

Attack Radar Manual Photo Button.

The manual photo button (7, figure 1-72), 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.

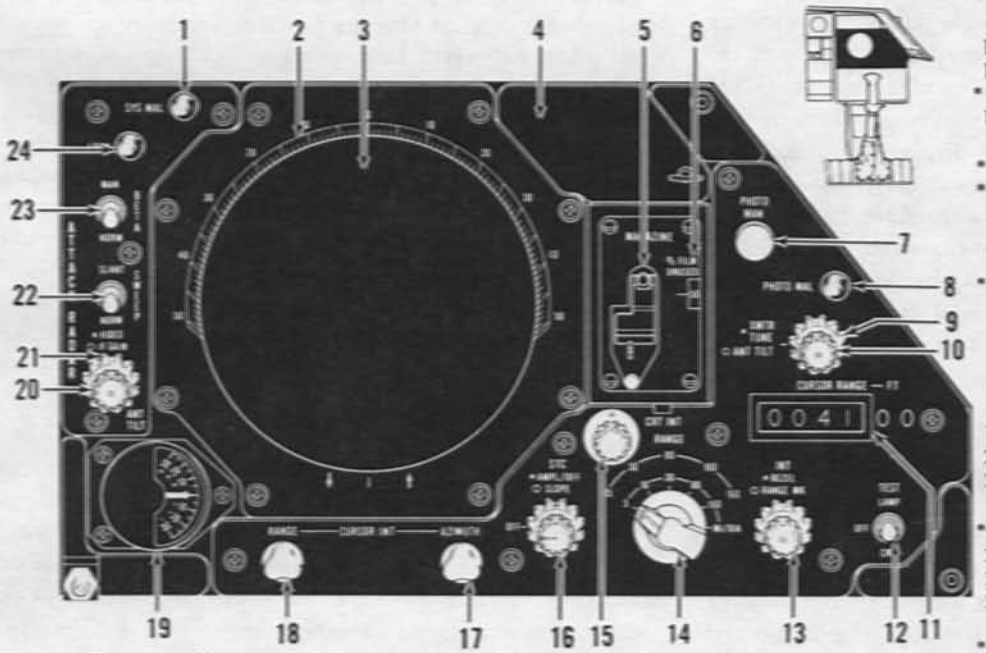
Intermediate Frequency Gain Control Knob.

The intermediate frequency gain control knob (21, figure 1-72), located on the radar scope panel, is labeled IF GAIN and permits adjustment of receiver gain when operating in the ground modes. This control determines maximum useable sensitivity of the receiver circuits and functions primarily as a brightness control. It can best be set without the transmitter firing (function selector in the ON position) but in order to set IF gain the video gain control must be advanced to a position to allow noise video to be painted on the scope even though the optimum setting of the video gain control must be made with the function selector knob in XMIT. With the video control in about the mid-range position, advance the IF gain control clockwise until the presentation shows a snowy noise appearance. The control should now be turned counterclockwise until the noise just disappears. If the control is set too low, weak returns cannot be distinguished on the scope because of the lowered sensitivity of the receiver. The intermediate frequency gain control knob should be normally near the top of its range but may have to be decreased as a point target is approached in order to reduce background clutter.

Antenna Tilt Control Knob.

The antenna tilt control knob (9, figure 1-72), 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 when the radar is operating in GND MAN mode, or operating in GND AUTO or GND VEL mode with the beta switch in the MAN position. The detent indicates zero tilt correction when the radar is operating in GND AUTO or GND VEL mode with the beta switch in the NORM position. Rotating the knob fully counterclockwise tilts the antenna up to +30 degrees and rotating the knob fully clockwise tilts the antenna down to −30 degrees. However, with the terrain following radar (TFR)

Attack Radar Scope Panel (Typical)



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. Antenna Tilt Control Knob.
10. Transmitter Tuning Control Knob.
* (Intermediate Frequency Gain Control Knob.)
11. Cursor Range Counter.
12. Test Switch.
* (Sensitivity Time Control Knobs.)
13. Bezel/Range Marks Intensity Control Knobs.
* (Video Adjustment Knob.)
* (Transmitter Tuning Control Knob.)
14. Range Selector Knob.
15. Scope Intensity Control Knob.
16. Sensitivity Time Control Knobs.
* (Bezel/Range Marks Intensity Control Knobs.)
17. Azimuth Cursor Intensity Control Knob.
18. Range Cursor Intensity Control Knob.
19. Antenna Tilt Indicator.
20. Video Adjustment Knob.
21. Intermediate Frequency Gain Control Knob.
* (Test switch.)
22. Sweep Control Switch.
23. Beta Switch.
24. Range Lock Indicator Lamp.

*Note
Control location on aircraft prior to modification by T.O. 1F-111-780.

A7321500-E001B

Figure 1-72.

operating the antenna can only be physically adjusted to —10 degrees (pitch plus tilt) to prevent interference with the TFR. Antenna position is indicated on the antenna tilt indicator, 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. The antenna sighting angle has a significant effect on the adjustment position of the other radar controls, therefore, the sighting angle (tilt) should be adjusted prior to optimizing the intermediate frequency gain, video and sensitivity time control knobs. The antenna pattern in the ground modes is optimized to provide even ground paint from 8 to 80 miles range with the aircraft at 40,000 feet of altitude and the toe of the antenna beam at 80 miles range. At 40,000 feet of altitude the antenna tilt should be approximately —5 degrees. At altitudes less than 5000 feet the sighting angle should be adjusted with the tilt control for approximately 0 degrees. One method of adjusting the sighting angle is to turn the range selector knob to 80, place the beta switch to MAN, and adjust the tilt control knob until video returns are available at 80 miles on the radar scope display.

Antenna Tilt Indicator.

The antenna tilt indicator (19, figure 1-72), located on the attack radar scope panel, provides an indication of the commanded sighting angle (tilt) of the antenna beam from horizontal on a single-pointer type meter graduated in 5-degree increments from zero to ± 30 degrees. In the GND MAN mode, and in GND AUTO and GND VEL with the beta switch in the MAN position, the indicator displays the sighting angle of the antenna as commanded by the antenna tilt control knob. In ground automatic and ground velocity modes, with the beta switch in NORM, the sighting angle displayed is that commanded by the bomb nav system as modified by any correction from the antenna tilt control knob. In air mode the indicated tilt angle is that commanded by the tracking handle tilt control as modified by the automatic elevation tracking function and the superimposed box scan. Both the pitch and tilt antenna axes are singly capable of operating to the limits of ±30 degrees. However, when operated simultaneously, the total antenna travel is limited to a minimum of +30 degrees and —50 degrees with respect to the aircraft longitudinal axis. The lower limit of the combined displacement is further restricted to —10 degrees with respect to the aircraft longitudinal axis when the TFR on signal is received by the radar set. If the combined displacement of pitch and tilt exceeds the values given above, the tilt gimbal is commanded to back off. The tilt indicator will continue to indicate sighting angle of the antenna beam from horizontal, but tilt may have been modified by the pitch plus tilt limit circuitry to the extent that tilt may have little relation to that commanded by the operator. The system malfunction lamp will light to indicate that the pitch and tilt electrical limits have been exceeded. Certain GO-NO-GO confidence tests can be performed using the antenna tilt indicator. Transmitter and receiver performance can be checked if a malfunction occurs, or during flight for extended periods over water, or during continuous operation in the air mode. When the function selector knob is placed in the TEST position, six receiver crystal currents, three magnetron currents, and AFC operation can be verified using the tilt meter as a readout device, with the individual parameters selected by use of the range selector knob and the SLC switch. Figure 1-73 shows the command sequence required to display the desired parameter on the tilt indicator and the desired readings to indicate a GO condition.

Attack Radar Scope Intensity Control Knob.

The scope intensity (CRT INT) control knob (15, figure 1-72), located on the radar scope panel, provides an adjustment of scope baseline intensity from zero to full brightness. To set the scope intensity properly, operate the radar with the function selector in the ON position. With the intermediate frequency gain and video controls fully ccw advance the knob clockwise (increasing brightness) to a point where the sweep is on the ragged edge between being visible and invisible. Since this control sets the baseline intensity on which all other video will be painted on the scope, it is somewhat critical in adjustment for best target presentation. If set too high, where the sweep is brightly visible as it follows the antenna scan, the scope phosphor will be excited to the point of complete masking or overriding of weak targets. If set too low, weak targets will not have the power to excite the phosphor to the point of visibility. The control setting can normally be left at the optimum position but may need refining after switching range scale.

Bezel/Range Marks Intensity Control Knobs.

Two coaxial knobs (13, figure 1-72), 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. The range marks represent slant range (straight line) distance from the radar antenna and are present on all range scales in ground manual and air modes but only on the three longest range scales in ground auto and ground velocity. The range mark intensity should normally be adjusted at a slightly different level from the range cursor intensity to prevent confusion.

MRT Parameter Checks

RANGE SELECTOR KNOB	*SLC SWITCH*	*PARAMETER*	*ANTENNA TILT INDICATOR READING*
5	OFF	AFC No. 1 crystal current	+ 10 to + 25 or — 10 to — 25
10	OFF	Main No. 1 crystal current	+ 10 to + 30 or — 10 to — 30
30	OFF	Mag current (narrow pulse width)	— 10 to — 25
80	OFF	Mag current (medium pulse width)	— 10 to — 25
160	OFF	Mag current (wide pulse width)	— 10 to — 25
5	SLC	*AFC No. 2 crystal current	+ 10 to + 25 or — 10 to — 25
10	SLC	*Main No. 2 crystal current	+ 10 to + 30 or — 10 to — 30
30	SLC	Omni No. 1 crystal current	+ 10 to + 30 or — 10 to — 30
80	SLC	*Omni No. 2 crystal current	+ 10 to + 30 or — 10 to — 30
160	SLC	Present volts error of AFC	Between ± 25

Note:

1. Mode selector knob must be in GND MAN position.
2. Function selector knob must be in TEST position.
3. *No. 2 crystal currents are in opposite polarity from respective No. 1 crystal currents

Figure 1-73.

Range and Azimuth Cursor Intensity Control Knobs.

Two cursor intensity control knobs (17, 18, figure 1-72), 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. Adjustment of the range and azimuth cursor intensity to the lowest useable value will allow more precise placement of the cursors over the target. Placement of the range cursor requires that the edge of the cursor nearest the vertex be placed precisely on the desired impact point. This is normally the edge of the selected target return nearest the vertex. Too much brightness of the range cursor will result in a bombing error with the bomb generally hitting beyond the target. The azimuth cursor should split the selected return and if too bright, the cursor will totally obscure small targets.

Attack Radar Test Switch.

The test switch (12, figure 1-72), 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. In the LAMP position, the switch can be used to test the following lamps: SYS MAL, LOCK, PHOTO MAL, and the up and down elevation tracking lamps (arrows). The momentary CKT position is used for energizing the system malfunction test circuit, and places the circuit in self test. The system malfunction lamp will light continuously when the test switch is held in the CKT position.

Attack Radar Sweep Control Switch.

The sweep control switch (22, figure 1-72), 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 a map-like presentation in the NORM position and a linear presentation in the SLANT position. The switch is inoperative in the air mode.

Attack Radar Beta Switch.

The beta switch (23, figure 1-72), 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 positioned by signals from the bomb nav system and the antenna tilt control knob can be used to refine this position. 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.

Sensitivity Time Control Knobs.

Two coaxial rotary sensitivity time control (STC) knobs (16, figure 1-72), 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. Slope control is at the minimum with the knob fully clockwise. The STC slope function is inopeartive in the AIR mode. With STC on, an altitude compensation signal is provided to maintain even ground paint as long as the toe of the antenna beam remains at 80 nautical miles. With the toe of the beam at shorter distances, the slope and amplitude controls must be adjusted for compensation. The amplitude control sets the receiver sensitivity level at zero range and the slope control adjusts the time required after zero range for the receiver to regain full sensitivity. This time would be less if the toe of the beam was on a point 30 miles ahead of the airplane than for a condition with the toe 80 miles ahead. As an example of practical use of the STC function, assume a low level bomb run on a target just inside the 30-mile range. Ground returns at short range would be very bright without STC on. The prime target would grow brighter as it were approached also. By turning STC on, adjusting amplitude to cut down the intensity of the near targets, and adjusting slope to give even paint throughout the 30-mile range, the bomb run can be completed without requiring a reduction in video gain or IF gain. This setting of STC will normally hold throughout a low level bomb run because antenna sighting angle will not change appreciably as range decreases. On medium to high altitude bomb runs the sighting angle changes appreciably as range decreases, so it is advantageous to place the beta switch in MAN and leave the antenna tilt angle fixed at the value where STC was adjusted. The STC function is fixed in air mode but must be turned on with the outer STC knob labeled AMPL/OFF.

Video Adjustment Knob.

The video adjustment control knob (20, figure 1-72), located on the attack radar scope panel, provides a means of adjusting the video signal. The knob is labeled VIDEO and will increase the amplitude of the video signal supplied to the attack radar scope when it is turned clockwise. The video control determines the brightness of target returns as opposed to the CRT intensity control setting the overall baseline brightness of the scope. To set the video control properly advance the function knob to XMIT and adjust antenna tilt to see the most returns on the scope. Adjust the video control until the target returns are sharp and bright against the picture background and give an overall optimum contrast to the picture. Video gain may need to be decreased slightly to prevent blooming of the target on a bomb run as the range decreases and the return grows stronger.

Transmitter Tuning Control Knob.

The transmitter tuning control knob (10, figure 1-72), located on the attack radar scope panel allows continuous tuning of the transmitter over its entire frequency range. The knob labeled XMTR TUNE may be used when the frequency control knob is in the AFC 2 position.

Attack Radar Range Selector Knob.

The range selector knob (14, figure 1-72), 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 is displaying a fractional part of a range. For example, when the switch is set to the 5/15 mile position, the scope is displaying 5 miles of the 15 mile range.

Attack Radar Scope.

The radar scope (3, figure 1-72) 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 except in ground velocity mode. The scope is 7 inches in diameter. The sector displayed is a 90 degree area ahead of the aircraft when in wide scan and a 20 degree area centered on the azimuth cursor when in narrow scan. An azimuth bezel around the top of the scope is graduated in one degree increments

Attack Radar - Subsystem Tie-Ins

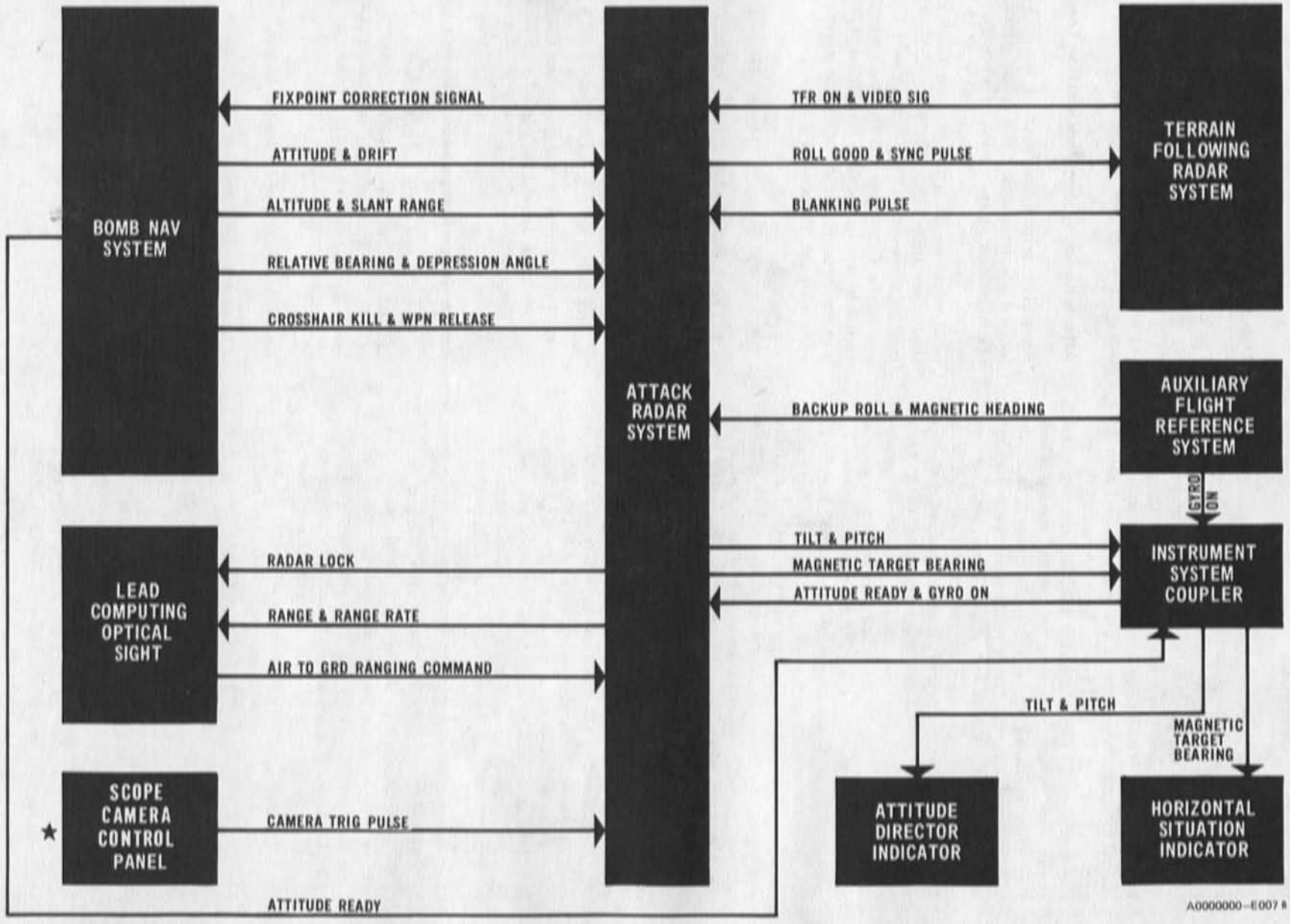


Figure 1-74.

with each 10 degrees marked to show azimuth displacement up to 50 degrees either side of the aircraft heading or ground track. When operating in the air mode or when the antenna is caged in ground manual mode, zero degrees on the scale represents aircraft heading. In any of the ground modes the scan is displaced in azimuth to compensate for drift, and zero degrees represents ground track. Two arrows at the bottom of the bezel indicate target position relative to antenna scan, when operating in the air mode. When the arrow pointing up is lighted, the target is in the upper scan. When the arrow pointing down is lighted, the target is in the lower 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 by the tracking control handle in ground manual and air and by the bomb nav system in ground auto and ground velocity. 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 or ground velocity modes when in 5 and 10 or 15/5 and 30/10 range scales, respectively. 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. During the air to ground ranging, the attack radar provides the TFR system with its computer function to compute the slant range to a ground point for use in the LCOS. For the attack radar computer to accurately compute the air to ground range, it is necessary that the attack radar computer be synchronized with the TFR transmitter. The TFR, therefore, supplies the attack radar computer with a pulse indicating the fixing time of the TFR transmitter and a sample of the video to which the range is desired. The pulse from the TFR, indicating fixing time, will determine when the sweep on the attack radar begins. However, the video display on the attack radar scope is provided by the attack radar receiver. Since the video and the sweep on the attack radar scope are not synchronized, a range error exists for any arbitrary returns displayed making the video presentation unsuitable for data extraction. The range readout on the attack radar will indicate the slant range to the video pulse supplied by the TFR. The tracking handle for the attack radar is rendered inoperative and the plan position indicator presentation on the TFR is blanked during air to ground ranging.

Attack Radar Cursor Range Counter.

The cursor range counter (11, figure 1-72), located on the attack radar scope panel indicates slant range in all modes of operation. The counter has four electronic digital readout windows capable of indicating distances up to 799,900 feet in increments of 100 feet.

Attack Radar System Malfunction Lamp.

An amber system malfunction lamp (1, figure 1-72), located on the attack radar scope panel, provides the operator with an indication of a possible failure in the system. The lamp is labeled SYS MAL and will not light when the function selector knob is in any position other than ON, XMIT or TEST (TEST only if in AFRS operation). The lamp will light indicating any of the following: (1) function selector knob inadvertently left in the TEST position; (2) failure in the antenna system, or input to the antenna system; (3) failure in the transmitter system.

Note

- The attack radar system malfunction lamp will light any time the combined antenna pitch plus tilt angle exceeds the vertical limits. Vertical limits with TFR on are −10 to +30 degrees. It will be normal for the attack radar system malfunction lamp to light intermittently during TFR operation.
- If the system is not usable the function selector knob should be placed to STBY to stow the antenna.

Attack Radar Range Lock Indicator Lamp.

A green range lock indicator lamp (24, figure 1-72), located on the attack radar scope panel is labeled LOCK. The lamp will light when a target is acquired while operating in the air mode, or during air-to-ground ranging operation.

Photo Malfunction Indicator Lamp.

The amber photo malfunction indicator lamp (8, figure 1-72), located on the attack radar scope panel, is labeled PHOTO MAL, and 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.

Note

In case of film stoppage, the malfunction can usually be cleared by removing the film magazine and manually turning the drive shaft two or three turns.

Tracking Handle Operation

Attack Radar Control Setting	TRACKING HANDLE MOVEMENT	
	Left — Right	*Fore — Aft*
GND MAN	Slews azimuth cursor left and right	Slews range cursor out and in (range)
GND AUTO-PP	Provides correction voltage to aircraft navigational computer unit for updating navigational fix bearing Azimuth cursor controlled by aircraft navigational computer unit	Provides correction voltage to aircraft navigational computer unit for updating navigational fix range Range cursor controlled by aircraft navigational computer unit
GND AUTO-DEST	Provides correction voltage to aircraft navigational computer unit for updating target spot bearing (azimuth cursor) Azimuth cursor controlled by aircraft navigational computer unit	Provides correction voltage to aircraft navigational computer unit for updating target spot range (range cursor) Range cursor controlled by aircraft navigational computer unit
GND VEL-PP	Provides correction voltage to aircraft navigational computer unit for updating navigational fix bearing (azimuth cursor) Slews display left or right (in azimuth) Fixpoint and cursors at center of radar scope Display ground stabilized Sweep origin offset to aircraft position (off radar scope)	Provides correction voltage to aircraft navigational system for updating navigational fix range (range cursor) Slews display out or in (range) Fixpoint and cursors at center of radar scope Display ground stabilized Sweep origin offset to aircraft position (off radar scope)
GND VEL-DEST	Provides correction voltage to aircraft navigational computer unit for updating target spot bearing (azimuth cursor) Slews display left or right (in azimuth) Target and cursors at center of radar scope Display ground stabilized Sweep origin offset to aircraft position (off scope)	Provides correction voltage to aircraft navigational computer unit for undating target spot range (range cursor) Slews display out or in (in range) Target and cursors at center of radar scope Display ground stabilized Sweep origin offset to aircraft position (off scope)
AIR	Slews azimuth cursor left and right	Controls antenna tilt. Controls range cursor if R_a button is depressed

Figure 1-75.

Unused Film Indicator.

The unused film indicator (6, figure 1-72), located on the attack radar scope panel, is a digital readout indicator that displays the percent of film remaining in the magazine. When the indicator reads 100 percent a maximum of 1300 frames of film is present.

Antenna Cage Pushbutton Indicator Lamp.

The antenna cage pushbutton indicator lamp (6, figure 1-70), located on the attack radar control panel, is labeled ANT CAGE and when the indicator lamp in the pushbutton is lighted, the word CAGE will be displayed. The pushbutton provides a means of caging the antenna and the lamp provides an indication that the antenna is caged, either due to manually pushing the button or due to failure of the automatic pitch and roll stabilization circuitry. Normally the antenna is stabilized automatically in pitch and roll by a primary signal from the bomb nav system. In the event of absence of the primary signal, due to a failure in the bomb nav system, the antenna will be caged automatically in pitch (indicator lamp will not light) and stabilized automatically in roll by an auxiliary signal from the AFRS. The antenna will be caged automatically in pitch and roll and the indicator lamp will light when both the primary signal from the bomb nav system and the auxiliary signal from the AFRS are not present or, if both signals are present and the flight instrument reference select switch is positioned to AUX. The antenna will continue to sector in azimuth and tilt can be adjusted. The antenna can be caged at any time in pitch and roll by depressing the pushbutton. If pitch or roll stabilization signals are present, depressing the button again after the antenna has been manually caged will uncage the antenna and the lamp will go out.

Note

When the antenna is caged in roll the terrain following radar will generate a fly-up signal.

ELECTRONIC COUNTER COUNTERMEASURES (ECCM) FEATURES.

The attack radar design incorporates electronic counter-countermeasures (ECCM) features, some of which are effective during certain conditions of normal operation and others of which may be activated when an active jamming environment is encountered. The ECCM features which may be activated by the operator are as follows:

- Transmitter frequency—The attack radar is normally operated in the AFC 1 mode. In this mode, the transmitter operates in a frequency agility mode in which the transmitter output frequency is swept through the frequency band with random reversals. The changing frequency and rapid scanning rate not only provide immunity to many types of jamming environment but also improve stability and legibility of the radar scope display.
- Side lobe cancellation (SLC)—SLC is not used in ground modes unless the radar encounters a heavy jamming environment in which jamming signals or spurious inputs are received through the antenna side lobes. In such an environment, SLC is activated by a switch on the radar set control in order to reduce the interference. SLC may also be used to reduce ground clutter when the radar is operating in the air mode.
- Receiver response characteristics—Fast time constant (FTC) may be introduced into the receiver by selection of the FTC switch on the radar set control. This is a two-position switch normally placed in the NORM position. When the receiver is saturated by a jamming environment consisting of continuous wave or modulated continuous wave signals, the switch may be positioned to FTC for improved radar scope display.
- Polarization—CIR—NORM—When this switch is in the NORM position, the radar operates with horizontal polarization in the ground modes and with vertical polarization in the air mode. To improve operation in rain, this switch may be set to the circular (CIR) position, which changes operation to circular polarization. Circular polarization reduces the clutter return from rain and minimizes the fogging effect on the radar scope display.

ALTERNATE OPERATION.

The attack radar is designed for fail-safe operation in certain modes. A failure in noncommon control circuits or functional circuits does not affect operation in the fail-safe mode. In the event the antenna pedestal fails to stabilize, the attack radar should be operated with the antenna caged. In the cage mode, roll and pitch are driven to and maintained in alignment with the aircraft coordinate system. When operating in ground manual mode with the antenna caged, the radar scope displays aircraft heading at zero degrees azimuth and the range sweep becomes slant range instead of the normal ground range (ground automatic/velocity modes are not affected in this way). In the event of failure in the ground auto or ground velocity modes, the attack radar should be operated in ground manual mode. In ground manual mode, the attack radar operates independently of the inertial bombing-navigation system and the radar scope display is read directly by the operator. If the transmitter frequency agility should fail in AFC 1, the attack radar may be operated in AFC 2. In AFC 2 mode, the transmitter may be manually tuned anywhere within the frequency band. Receiver tuning is performed automatically, as in AFC 1 mode. If the receiver automatic frequency control fails in both the AFC 1 and AFC 2 modes, the

attack radar should be operated in manual frequency control position (MFC). In MFC, the transmitter operates at a midband fixed frequency, while the receiver is tuned manually.

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 radar altimeter system will provide fly-up signals upon reaching the preset altitude during ILS or AILA. The system is composed of two receiver-transmitter units; two antennas, one for transmitting and one for receiving; a distribution box; a radar altimeter indicator and the necessary controls. The receiver-transmitter units are located in the forward electronic equipment bay. When the system is placed in operation, one receiver-transmitter 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 receiver-transmitter unit must be manually selected. The receiver-transmitter unit in operation is connected to the antennas and its outputs are distributed to other aircraft systems by circuits in the distribution box. A pressure operated switch in each receiver-transmitter unit will place the operating unit to standby when above approximately 38,000 feet pressure altitude. The radar altimeter should maintain lock to bank angles of 45° and pitch angles of ±20°. 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 1-69 for antenna location.

WARNING

High frequency radar waves can penetrate snow, sand, dry lake beds, etc. When operating in these areas, the radar altimeter may indicate a greater terrain clearance than actually exists.

RADAR ALTIMETER CHANNEL SELECTOR SWITCH.

The radar altimeter channel selector switch (7, figure 1-56), located on the miscellaneous switch panel, is labeled RADAR ALTM and has two positions marked CHAN 1 and CHAN 2. Placing the switch in either position will allow the receiver-transmitter unit in the respective channel to transmit and receive.

RADAR ALTIMETER BYPASS SWITCH.

The radar altimeter bypass switch (6, figure 1-56), located on the miscellaneous switch panel, 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 INDICATOR.

The radar altimeter indicator (23, 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, when the altitude of the airplane is over 5000 feet above the terrain, if the pitch or roll limits of the system are exceeded or if the system malfunctions.

WARNING

If power is lost on the system, the OFF warning flag will appear on the dial and the pointer will remain at the last indication.

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 95 ±12 feet if the receiver-transmitter unit is operating properly.

CAUTION

The radar altimeter must be turned off after each flight to prevent damage to the receiver-transmitter unit, should power be applied without cooling air.

The self-test feature may be used at any time and at any altitude below approximately 38,000 ±5000 feet.

RADAR ALTITUDE LOW WARNING LAMP.

The radar altitude low warning lamp (25, figure 1-5), located on the left main instrument panel, will light when the absolute altitude of the aircraft is at or below the minimum altitude set into the radar altimeter. When lighted the letters RADAR ALT LOW are displayed on the face of the lamp in red.

TERRAIN FOLLOWING RADAR (AN/APQ-110).

The terrain following radar (TFR) provides low altitude terrain following, terrain 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, central air data computer and flight control system. Refer to figure 1-79. For TFR operating procedures, refer to Section IV. 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 aircraft to be flown manually or automatically at a preselected terrain clearance. Climb and dive signals generated in the manual and automatic mode can be coupled into the attitude director indicator (ADI) and lead computing optical sight (LCOS). In the manual mode, the set clearance can be maintained by flying pitch steering commands on the ADI or LCOS. In the automatic mode, the climb and dive signals are coupled into the pitch channel of the flight control system. Refer to figure 1-78. The TF mode can also be used to make blind let-downs to a preselected terrain clearance in either the manual or automatic mode. Should the aircraft descend to below 68 percent of the selected terrain clearance altitude setting, or if a TF malfunction is detected, the TFR will indicate a fly-up command on the ADI/LCOS and the flight control system will initiate a fly-up maneuver if the aircraft is being flown manually or is in auto TF.

Note

The initial aircraft response to the fly-up command may be as much as +3.8 absolute "g"s.

SITUATION MODE (SIT).

This mode of operation is used in conjunction with the TF mode for obstacle avoidance. In this mode, the antenna is pitch and drift stabilized such that the antenna scans in the horizontal plane thirty degrees on each side of the aircraft ground track. Returns from terrain that extend above the altitude of the aircraft are displayed on the scope in a one-radius offset plan position indication with range selections of 5, 10, and 15 miles with fixed cursors provided for range reference at 1, 2, and 5 miles respectively. The information displayed on the scope allows the pilot to choose a flight path which avoids major obstacles.

GROUND MAP (GM) MODE.

The antenna scan and stabilization, and type of scope display in this mode are the same as in the SIT mode. The targets displayed in this mode are those which are being illuminated by the complete pencil beam from the antenna. The antenna tilt can be adjusted to optimize the target display by rotating the antenna tilt knob on the TFR control panel. The range of tilt control is from 0 degree to 15 degrees below the horizontal reference.

AIR-TO-GROUND RANGING MODE.

This mode is used in conjunction with the attack radar to provide range-to-target data to the LCOS when an air-to-ground weapon delivery mode is selected on the LCOS. The mode enabling command signal is routed within the TFR to whichever channel is in the SIT or GM mode.

TFR CONTROLS AND INDICATORS.

TFR Channel Mode Selector Knobs.

Two five-position rotary channel mode selector knobs (2, figure 1-76), 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. 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 after a momentary fly-up occurs. If the R channel is in control and a failure occurs (even a momentary failure) the L channel will take over and the R channel will go to standby. Another failure will switch the operation back to the R channel. Any subsequent fail-

ures will not cause automatic switchover until the L channel is recycled to STBY position and back to TF. In other words it will cycle R-L-R and stay in R. If the L channel starts out in control a fail will cause a switchover to the R and stop there until the L channel is recycled.

CAUTION

If both channels are not placed in STBY or above when operating the TFR or attack radar, the crystal diodes in the receiver of the channel in OFF may be damaged.

Terrain Clearance Knob.

The terrain clearance knob (3, figure 1-76), located on the TFR control panel, has six positions marked 200, 300, 400, 500, 750 and 1000. Rotating the knob clockwise increases the altitude clearance setting corresponding to the position selected and vice versa.

Note

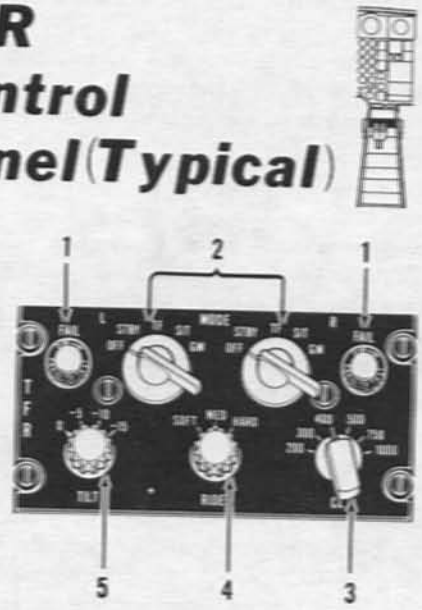
When flying at one clearance setting and the knob is positioned to a higher setting, a TF failure and fly-up may be generated until the aircraft is maneuvered outside the 68 percent radar altimeter fly-up range.

The TFR should maintain flight within the following tolerances. If these tolerances are not maintained the equipment should be written up after the flight. If an undershoot occurs to the extent that a 68 percent fly-up occurs, change to the other channel. If the fly-up again occurs select the next higher setting. Do not operate at the lower setting where the 68 percent fly-up occurred.

Selected Clearance	Terrain Clearance Min.	Max.
200	170	300
300	260	425
400	350	550
500	440	650
750	675	950
1000	900	1200

During automatic terrain following flight over level to rolling terrain or other water with HARD ride selected, the radar altimeter should indicate the altitude above the terrain corresponding to each clearance knob setting within these tolerances. Terrain clearances when cresting peaks will usually be slightly less than the stabilized, level terrain clearances. The above tolerances are not directly applicable to terrain following flight over rugged terrain; however, terrain clearances when cresting peaks should not consistently be below these tolerances. Clearances may be slightly higher in SOFT or MED relative to the HARD ride clearances.

TFR Control Panel (Typical)



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.

A7341400-E001A

Figure 1-76.

Ride Control Knob.

The ride control knob (4, figure 1-76), located on the TFR control panel is a three position rotary knob marked SOFT, MED, and HARD. The negative commanded g's will be limited to zero for HARD, 0.5 for MED, and 0.75 for SOFT. The fail-safe fly-up signal is not affected by the position of this switch. Progression of the ride control from HARD to SOFT will compute an earlier anticipatory command upon approach to an obstacle.

Antenna Tilt Control Knob.

The antenna tilt control knob (5, figure 1-76), 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. The tilt control is inoperative when the channel mode selector knob is in the STBY, TF, or SIT positions.

Range Selector Knob.

The range selector knob (5, figure 1-77), 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 Tuning Control Knobs.

Four radar scope tuning control knobs (4, figure 1-77), 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. To obtain the proper video threshold on the TFR scope adjust the contrast control in the E mode until a thin vertical line along the right side of the scan becomes discernible and then adjust the video control for optimum target display. With this procedure, the video paint on the E scope is at approximately the same threshold level as that required by the system for proper commands from the forward-looking radar, and, when switching between any of the positions on the range selector knob (5, 10, 15, E), the scope display should not bloom or fade excessively.

Autopilot Release Lever.

When flying auto or manual TF the autopilot release lever (6, figure 1-24), located on the control stick grip, may be used for overriding fly-up maneuvers induced by loss of the data good signal from the TFR. When the autopilot release lever is depressed, a pseudo data good signal is sent to the flight control system to interrupt the fly-up. The TF failure warning lamp will not go out when the autopilot release lever is depressed. The fail caution lamp on the TFR control panel will remain lighted as long as the fail condition is present in the TFR. For description of autopilot related functions of the lever refer to "Autopilot" this section.

Auto Terrain Following Switch.

The auto terrain following (auto TF) switch (3, figure 1-27), located on the autopilot/damper panel, is a two position lever lock switch marked AUTO TF and OFF. The switch is locked in the OFF position and must be pulled out to move from OFF to AUTO TF. When the switch is in the OFF position and either TFR channel mode selector knob is in the TF position, the aircraft must be flown manually using the pitch steering commands on the ADI and LCOS to hold the terrain clearance selected on the TFR terrain clearance knob. With the switch in the OFF position the reference not engaged lamp will remain on. When the switch is placed to the AUTO TF position and either TFR channel mode selector knob is in the TF position signals from the TFR will control the pitch damper and series trim to automatically fly the aircraft on the terrain clearance setting selected by the terrain clearance knob. The reference not engaged lamp will go out when the auto TF mode is engaged.

Note

When auto TF is selected at least one TFR channel must be in the TF mode to prevent abnormal series trim operation. The fly-up off caution lamp and the reference not engaged lamp will be lighted for this configuration.

When the AUTO TF position is selected, and the autopilot release lever is not held, parallel trim will center and the control stick may move. If auto TF mode is selected, the pitch trim function of the stick trim button will be inoperative.

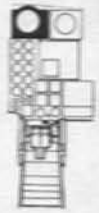
Instrument System Coupler Pitch Steering Mode Switch.

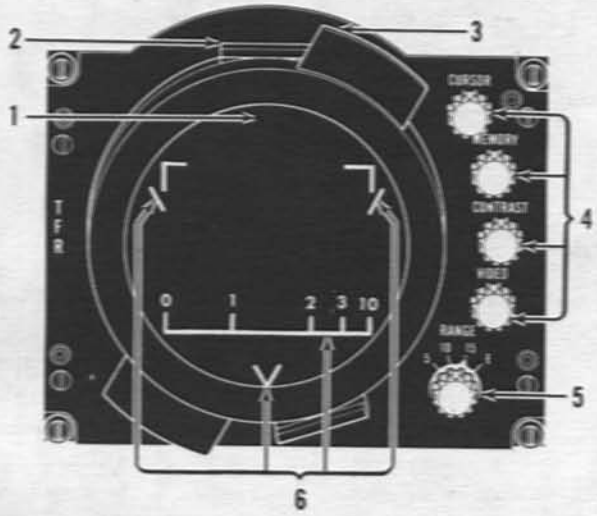
The instrument system coupler pitch steering mode switch, located on the instrument system coupler control panel (14, figure 1-5), has three positions marked ALT REF (altitude reference), OFF and TF (terrain following). Placing the switch to the TF position couples the manual commands from the TFR to the pitch steering bars on the LCOS and ADI. One or both TF channels must be in the TF mode before the switch will hold in the TF position. The switch will not hold in the TF position if the ISC mode selector knob is in the ILS, AILA or AIR/AIR positions. It is recommended that the switch be placed in the TF position when performing auto TF in order that the aircraft response to the automatic commands can be compared to the manual command displays. The switch will not hold in the ALT REF position if either TFR channel is in the TF mode and is checking safe. For a description of other functions of the switch refer to Instruments this section.

Instrument System Coupler Mode Selector Knob.

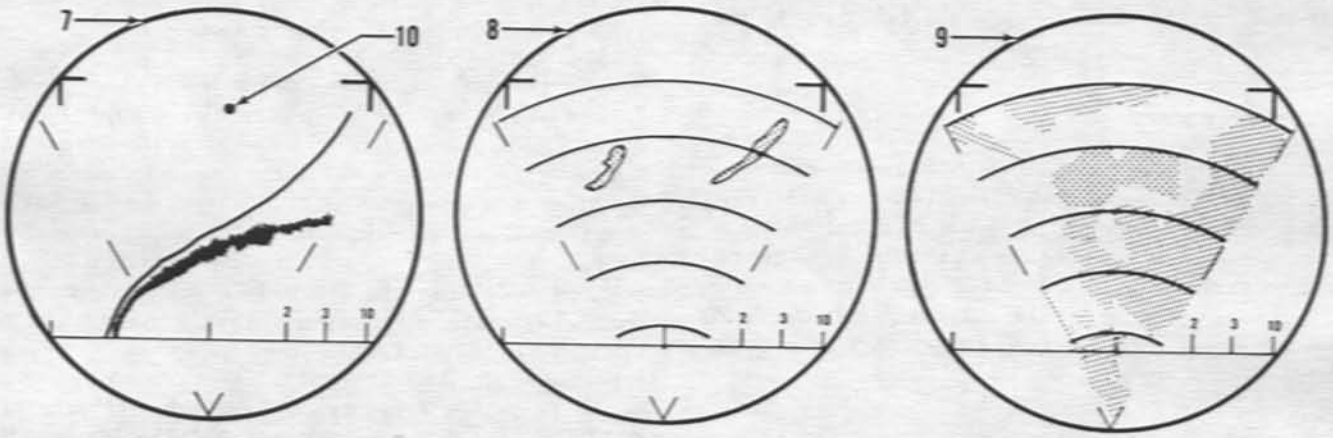
The instrument system coupler mode selector knob, located on the instrument system coupler control panel (14, figure 1-5), has nine marked positions. TFR opera-

TFR Scope Panel and Presentations (Typical)





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.
7. TF Mode Terrain Following E Display.
8. Sit Mode Lateral Terrain Search PPI Display.
9. GM Mode Ground Mapping PPI Display.
10. Test Pulse.



Test Patterns

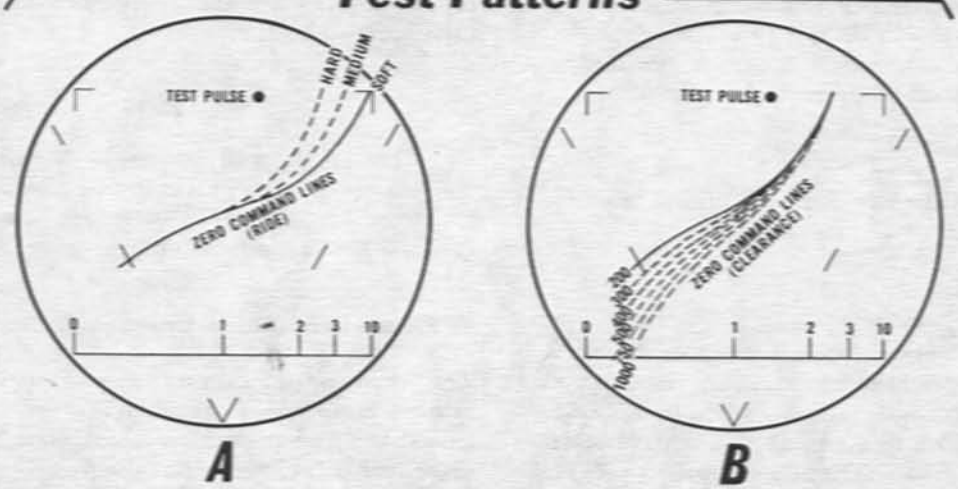


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Figure 1-77.

tion is compatible with all positions of the knob except ILS, AILA and AIR/AIR. With the knob in any of these three positions the ISC pitch steering mode switch will not hold in the TF position. The knob should be placed in the NAV position for TFR operation. With the knobs in the CRS SEL NAV or NAV positions a pullup signal from the armament system will cause the ISC pitch steering mode switch to unlatch and return to the OFF position. For a description of the other functions of the knob refer to Instruments this section.

Flight Instrument Reference Select Switch.

The flight instrument reference select switch (2, figure 1-56), located on the miscellaneous switch panel, has two positions marked PRI and AUX. Normal TFR operations should be performed with this switch in the PRI position. Selecting the AUX position while the bomb nav system is operating normally, will cause the attack radar antenna to cage and the TFR to fail. Should the bomb nav stabilized platform system fail, or be turned off, the switch over to the AFRS will be automatic whether the switch is in the PRI or AUX position. For complete description of the functions of this switch refer to auxiliary flight reference system this section.

Radar Altimeter.

During terrain following operation it is recommended that the radar altimeter index pointer be set to 80 percent of the clearance to be flown in order that the radar altitude low warning lamp can serve as a caution lamp to alert the aircrew to more closely monitor the TFR system operation and, if descent is continued, to expect a fly-up at 68 percent of the selected clearance plane.

Note

Performing a self test of the radar altimeter while flying TFR will cause a fly-up, since the false altitude the altimeter locks on during self test is less than 68 percent of any selected clearance.

Refer to "Radar Altimeter System" this section for complete description of the radar altimeter.

Radar Scope.

The radar scope (1, figure 1-77), 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. A self-test pulse (10, figure 1-77) is located at approximately 1.5 miles range at the top of the E scope presentation when the system is in TF mode of operation. Absence of this pulse, in TF mode, indicates improper system operation. 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.

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, TFR ground mapping or situation scope presentations will not display a sweep. If one channel is in TF mode, the E scope presentation will be normal.

TFR Channel Failure Caution Lamps.

Two amber channel failure caution lamps (1, figure 1-76), 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 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 knob in TF, SIT, or GM position, indicates: (1) a malfunction in the TFR channel, (2) loss of one or more of the three data good signals from other subsystems (CADC, radar altimeter, roll pedestal), or (3) in TF mode only, the aircraft has descended below 68 percent of selected clearance. A press-to-test feature allows each lamp to be checked.

Note

The lamps will momentarily blink when changing the TFR terrain clearance knob from low to higher clearance settings. This is a normal indication.

TF Fly-up Off Caution Lamp.

The TF fly-up off caution lamp, located on the main caution lamp panel (figure 1-37), provides an indication that fly-up protection is not available. The fly-up maneuver is inhibited until the fly-up circuit is armed. The fly-up off lamp will be lighted during TF mode selection until the fly-up circuit is armed. Arming of

Auto TF Flight Control Schematic

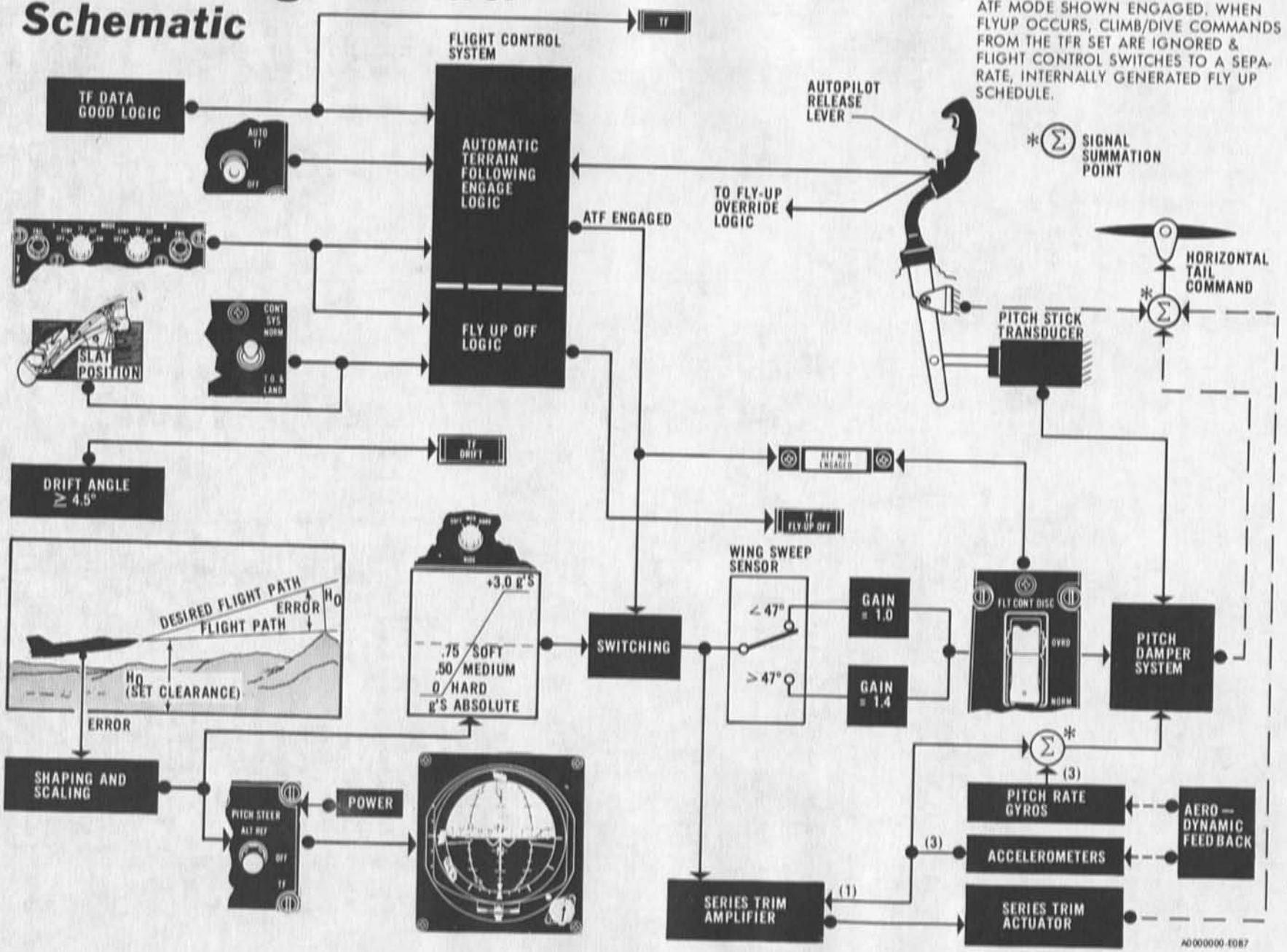


Figure 1-78.

the fly-up circuit will result in the TF fly-up off lamp going out. The fly-up circuit can be armed by either of two ways: (a) the TF set must initially check safe (TF fail warning lamp goes out) or (b) the autopilot release lever is momentarily depressed. After the fly-up circuit is armed, a subsequent TF fail signal (TF fail warning lamp comes on) will result in a fly-up maneuver. The flyup circuitry is interlocked with the flight control system switch and slat extension mechanism to prevent inadvertent fly-ups during take-off and landing operations. When airborne and with the L or R TFR channel mode selector knob in the TF position, the lamp will light when the flight control system switch is (1) in the T.O. & LAND position, or (2) in the NORM position and the slats are extended. Certain power failures within the TFR system will also cause the lamp to light. The lamp will also light if the auto TF switch is placed in the AUTO TF position and neither TFR channel is in a TF mode; however, this is an abnormal switching configuration and should be avoided. Terrain following flight should not be attempted when the lamp is lighted.

Reference Not Engaged Caution Lamp.

The reference not engaged caution panel is located on the left warning and caution panel (figure 1-37). The following conditions must be satisfied to put the reference not engaged caution lamp out during TFR operation.

1. Either TFR channel mode selector knob in the TF position,
2. The auto TF Switch in the AUTO TF position, and
3. "G" commands controlling the aircraft with TFR checking safe.

WARNING

While operating in auto TF the lamp can come on due to conditions not associated with TFR operation. For example, when operating in auto TF with constant track or heading navigate autopilot engaged, rolling into a turn will disengage the autopilot mode and cause the lamp to light. This is normal, however with the lamp on, a subsequent malfunction in the auto TF mode which could also light the lamp will go undetected. For this reason, any time the lamp lights during TF operation immediate action must be taken to put the lamp out or terminate auto TF.

For detailed description of the lamp and its functions with other systems refer to "Autopilot System," this section.

TFR Failure Warning Lamp.

A TFR failure warning lamp (18, figure 1-5), located on the left main instrument panel, provides a more apparent indication of TFR channel malfunctions. When lighted, the lamp will display the legend "TF." 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.

TF Drift Caution Lamp.

The TF drift caution lamp located on the main caution lamp panel (figure 1-37), provides an indication that the TFR antenna in the channel operating in the TF mode is displaced from aircraft centerline by 4.5 ($\pm$0.5) degrees or greater. The letters TF DRIFT are visible on the lamp when it is lighted. Terrain following flight must not be attempted when this lamp is lighted, until the complete drift angle accuracy check has been accomplished and verified in accordance with "TFR Inflight Checks," Section IV (see "TF Drift" under "TFR Operational Considerations," this section).

TFR OPERATIONAL CONSIDERATIONS.

Terrain Following Mission Planning.

WARNING

The following considerations should be taken into account during TFR preplanning and flight operation.

The aircraft is equipped with a terrain following radar (TFR) system that when properly used will give the aircraft a contour following flight path which will afford the maximum in surprise and terrain concealment. However, to gain the most benefit from the TFR, careful pre-flight planning must be accomplished to assure safe low level operation. TF mission planning data is provided in Appendix I. The following steps should be used as a guide in preparing for a mission involving TFR operation.

1. Determine aircraft configuration and required radius of action.
2. Select the initial recommended maximum angle-of-attack and corresponding airspeed from the TF planning charts in Appendix I. Maximum angle-of-attack is provided for the purpose of defining the minimum airspeed at which a TF failure may be

Terrain Following Radar - Subsystem Tie-Ins

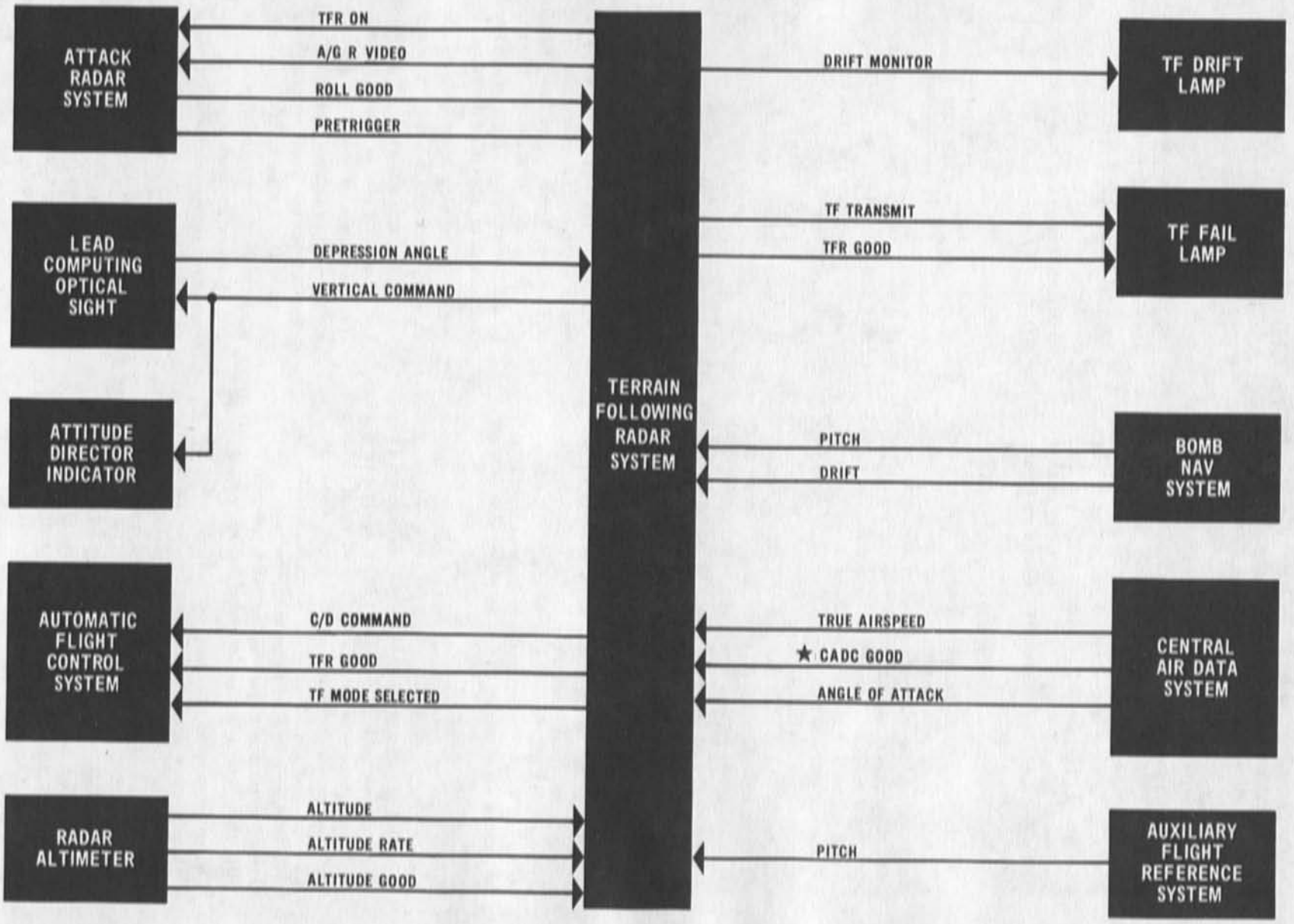


Figure 1-79.

experienced without exceeding the angle-of-attack limits presented in Section V. With this or a higher airspeed, calculate the cruise performance for the required radius of action to assure an adequate fuel reserve.

3. Next, carefully select the route for TFR operation. When possible, select a course parallel to ridge lines and along valleys in an attempt to keep terrain features between your aircraft and detection devices or ground to air weapons as long as practical. The route should be planned to avoid sand dunes and man made obstacles such as TV or radio towers, high power lines between valleys, smoke stacks and water towers, since tall, slender objects, sand dunes, or deep snow may not be detected by the TFR.

4. After the route has been tentatively selected to take advantage of terrain features for surprise and concealment, a close study should be made to determine the highest terrain feature above the stabilized flight altitude. The "delta" altitude is important in preflight planning for three reasons: (1) It will determine whether you will need military or afterburner power to go over the obstacle. (2) It will enable you to plan the type of pull up maneuver if the TFR should fail approaching the obstacle. (3) It might require a new route selection if the afterburners must be used to clear the obstacle and cruise performance or night visual detection are critical to mission accomplishment. The TF data presented in Appendix I will enable you to determine the delta altitude that can be cleared.

5. With the actual route established and outlined on a map, a minimum enroute altitude should be determined and indicated along the route. This minimum enroute altitude should provide at least 1000 feet above all obstacles five miles either side of the route.

6. Route turning points need special consideration during TFR operation because of the limitations of the TFR system during turns. While flying TF mode, do not exceed 10 degrees bank angle unless terrain clearance can be assured. Therefore, if heading changes are in excess of 10 degrees, the route must be carefully checked for terrain clearance if over 10 degree banked turns are to be used. An alternate method is to pull up to MEA to make the turn, then return to the desired clearance setting.

7. The recommended operating configuration for the TFR is to use one channel in the TF mode and the other channel in the SIT mode for maximum utilization of the terrain following and terrain avoidance capabilities of the system. The TFR mode of operation should be determined primarily as follows: (1) For day VFR conditions, one channel in TF and one channel in SIT with frequent monitoring of the E-scope. (2) For night and IFR conditions, one channel in TF and one channel in SIT with the E-scope primarily monitored.

Note

In IFR conditions, the E-scope should be monitored for weather returns so as to anticipate a flyup command as a result of weather.

8. The terrain clearance setting will normally be determined by the following considerations:

   (a) VFR or night and IFR conditions.
   (b) Terrain profile and differential altitude.
   (c) Height of trees or man-made obstacles.
   (d) Mission requirements, i.e. training or combat.

CAUTION

If at any time an undershoot of the set clearance occurs under night or IFR conditions, immediately go to the next highest clearance setting for operation and assure that the undershoot is within limits. Do not operate at the lower clearance where the undershoot occurred.

9. The desired TFR ride quality, i.e., soft, medium, and hard, determines how closely the flight path follows the desired clearance over the contour of the terrain. Medium ride is recommended for most TFR flight conditions especially during turbulence, night or IFR operation. Hard ride can be safely used; however, pushovers of zero to negative 0.5 "g" can be expected. These pushovers can be disconcerting and loose articles and dirt in the cockpit can cause serious distraction.

10. Preflight planning must include consideration of enroute weather. If heavy rain or thunderstorm activity is forecast, auto TF may not be usable. Use of attack radar to vector around weather cells is recommended. The E scope should be monitored for video returns from weather during TFR flight. The back scatter from moderate to heavy precipitation will often be visible on the E scope. If the operator cannot determine where the terrain ends and the precipitation begins on the E scope, the automatic signal detection circuitry will also be incapable of discrimination and a climb command will result. As video returns from rain approach the zero command line from the right side of the E scope, a climb command can be expected. When a climb command occurs, the pilot should not allow the auto TF to exceed the angle-of-attack limits or exceed a 20 degree climb attitude and should add power to maintain airspeed. If either of these angle limits are encountered, depress the autopilot

release lever, maintain the climb, and level off at a safe altitude (not less than MEA). When the video from weather disappears from the E scope and normal ground return is present, TFR operation can be resumed.

WARNING

- Certain weather conditions can cause blanking (no video presentation) of the E scope and/or attack radar with no associated fly-up. This blanking eliminates radar returns from terrain and precipitation. The TFR system interprets the blanking as the ground return from a flat, low radar energy return surface such as a body of water and reverts to radar altimeter override mode. In this mode, the TFR will not provide safe flight over other than known level terrain.
- If E scope blanking due to weather conditions is observed or suspected, an immediate climb must be initiated to MEA.

11. The automatic channel switchover feature (both channel mode selector knobs in TF) should not be used at clearances below 500 feet. Should a failure occur at the lower clearances in the channel operating in the TF mode, a clearance of 1000 feet should be flown after manually switching to TF in the opposite channel until proper performance is confirmed. It is also recommended that the index pointer in the radar altimeter be set to 80 percent of the clearance to be flown. If the low altitude warning lamp lights, closely monitor the TFR system and if descent continues, expect a fly-up at 68 percent of selected plane clearance. If a fly-up does not occur at 68 percent immediately take control of the aircraft and perform the 68 percent confidence check at a safe clearance on both TFR and radar altimeter channels before resuming TF flight. When flying auto or manual TF the radar altimeter indicator must be monitored for an altitude error, especially while flying over terrain having low radar return. The altitude information from the radar altimeter is used to generate the TF climb/dive command when forward video is lost. If there is an error on the radar altimeter indicator, e.g., indicating approximately 3000 feet while flying level at less than 1000 feet, and forward video is lost the climb/dive command will be a dive and the 68 percent fly-up will not occur because the altitude being in error indicates a condition of being above selected set clearance.

WARNING

Terrain following flight should not be attempted when there is an altitude error indicated on the radar altimeter indicator.

12. When changing selected clearances on the TFR to a higher setting, the clearance control knob should be progressively rotated through each intermediate setting and the aircraft allowed to stabilize at each intermediate clearance until the desired clearance is reached. Failure to select one clearance setting at a time will result in a climb angle which may cause the radar altimeter to lose range track. In this event the aircraft will not level off at the desired clearance setting and the fly-up must be overridden by manual control.
13. Do not fly in TF mode on AFRS reference, except under day VFR conditions.
14. When operating in auto TF mode, manually overriding auto TF commands will cause the series pitch trim actuator to drive to an improper position. This will result in degraded "g" response until the series pitch trim has had time to regain its correct position. When manually maneuvering the aircraft, the autopilot release lever should be depressed and held until the stick is returned to neutral to ensure that the series pitch trim does not drive to an improper position.
15. Pitch trim inputs should not be made when the auto TF switch is in the AUTO TF position, even when the autopilot release lever is held depressed.
16. Over calm water or flat terrain such as dry lake beds, dry wheat fields, or smooth sand, or snow covered areas, there will be little energy returned back to the radar. When forward video is lost while flying over smooth water or certain terrrain, terrain following will be commanded by inputs from the radar altimeter. The radar altimeter looks only below the aircraft 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.
17. When flying over smooth sloping areas of low reflectivity such that radar altimeter commands are controlling the aircraft, the aircraft will fly at an offset from the selected clearance proportional to the magnitude of the slope. If the flight path is up an extended slope, the aircraft will fly below set clearance by 87 feet/degrees of slope and the 68 percent fly-up threshold may be reached. When flying down an extended slope, the aircraft will fly at an offset above the set clearance.

Blind Let-Down.

When initiating a blind let-down in the TF mode from above 5000 feet absolute, the radar altimeter will not be locked and will drop out its data good signal to the TFR. To prevent this signal from generating a fly-up, the altimeter by-pass switch must be positioned to BY-PASS to give the TFR a pseudo altitude and data good signal. Upon passing through 5000 feet the altimeter should lock and the by-pass switch will drop out and normal altitude and data good signals will be supplied to the TFR. If the terrain is mountainous, the clearance may suddenly increase beyond 5000 feet and a fail signal will result. Should this occur immediately reposition the by-pass switch to BY-PASS and resume the let-down. If the let-down is being made to 1000 feet set clearance, the dive angle will be limited to 10 degrees until the radar altimeter locks. At this time the dive angle may increase to 12 degrees. At other set clearances the dive limits are 12 degrees above and below 5000. It is recommended that an initial set clearance of 1000 feet be used for all blind letdowns. When a letdown is made from high altitude in the vicinity of mountainous or rugged terrain the following is recommended to minimize the annoying radar altimeter break-lock when passing through the lock-on altitude and also to reduce the rate of descent at the set clearance level-off. Manual descent to MEA is recommended with the autopilot release lever depressed, while the aircrew monitors the pitch steering bar and E scope display. After the aircraft is established at MEA and the attack radar and E scope reveal no abrupt ridges along the aircraft flight path, an auto TF descent should then be initiated. Aircrews should be aware that while flying at MEA, the 5000 feet limit of the radar altimeter may be exceeded resulting in a TF fail-safe fly-up. After level-off at 1000 feet the desired operating clearance can be selected.

WARNING

- When a blind let-down is made, the radar altimeter must be monitored to assure that it locks on after passing through 5000 feet absolute. If the let-down is made over water or areas of low radar energy return, the radar altimeter is the only source of a signal to compute a level-off at the set clearance.
- Let-downs started below 5000 feet absolute, do not verify proper radar altimeter operation.

Low Altitude Fly-Up Recovery.

The fly-up command generated by descending below 68 percent of the set clearance will be terminated when the aircraft climbs up to the 68 percent point. If the condition that caused the aircraft to initially descend below 68 percent is still present and the TFR is allowed to control the aircraft, a pushover can result which will dive the aircraft back through the 68 percent point at an angle from which a fly-up command cannot recover.

Overriding Fly-Up Maneuvers.

Should a TF fail occur and a fail-safe fly-up maneuver be initiated, the maneuver should be overridden by depressing the autopilot release lever on the control stick as the aircraft attains a 20-degree pitch angle or limit angle-of-attack (whichever is less). Maintain this angle-of-attack or attitude until sufficient altitude has been gained to clear all known obstacles in the immediate area. The fail-safe fly-up maneuver will require approximately 2 seconds to attain the limit pitch attitude or limit angle-of-attack. Before releasing the autopilot release lever, the pilot should check that the TF failure warning lamp and the channel failure caution lamp for the channel in the TF mode are not lighted. If the lamps are still on, releasing the autopilot release lever will allow the fly-up maneuver to be resumed. Should the fly-up maneuver not be terminated by depressing the autopilot release lever, the TFR should immediately be switched out of the TF mode, the auto TF switch turned to OFF, and the fly-up overridden by stick force if necessary. The pitch trim function of the stick trim button is disabled and the control stick is centered during TFR flyup maneuvers.

Aircraft Pitch Attitude.

If the pitch attitude of the aircraft exceeds 20 degrees of pitch during a fly-up command, the radar altimeter may break lock. If this occurs, the fly-up must be manually overridden until the radar altimeter can regain track and allow the TFR to check safe.

Turning During TF Flight.

The TFR antennas in the TF mode are scanning in a vertical plane along the ground track. When turns are made while under TF control, the aircraft is turning into terrain which the antenna is not scanning. This could result in not obtaining climb commands in sufficient time to clear an obstacle. Turns exceeding 10 degrees of bank angle should be accomplished only under VFR conditions where terrain into which the turn is made can be observed or at minimum enroute altitude. If in the process of making heading correc-

tions during auto TF, stick force is applied in pitch without depressing the autopilot release lever, subsequent aircraft response to auto TF commands will be degraded up to 5 seconds after the stick control forces are released.

Pitch Trim During Auto TF.

Auxiliary pitch trim inputs must be zeroed for proper auto TF operation. If nose up trim is used, it will cause the series trim actuator to drive nose down until the effect is neutralized. This will reduce the amount of remaining down elevator available and when a subsequent down elevator command is called for there may be insufficient elevator available to keep the aircraft on the set clearance.

WARNING

Trim the aircraft prior to engaging auto TF and do not use pitch trim while operating in the auto TF mode. To do so will result in degraded auto TF performance such as ballooning on the back side of hills.

The pitch trim function of the stick trim button is inoperative when auto TF is engaged.

Stick Pitch Inputs During Auto TF.

If in the process of making small heading corrections during auto TF, some stick force is applied in pitch, the aircraft response to auto TF commands may be degraded up to 5 seconds after the stick control forces are released. During TF fail safe fly-ups, control of the aircraft should be achieved by depressing the autopilot release lever and using stick force until a decision is reached regarding continued TF operation. Angle of attack should not be allowed to exceed operating limits included in Section V. Maintain safe terrain clearance during this period.

Use of E Scope.

The E-scope presentation is intended for an advisory display and is not a primary command display. The pilot can fly manual TF by keeping the video below the command line but due to the lack of proper feedback to the display it should not be used as the primary manual display command below 500 feet set clearance; however, the E-scope display should be monitored at all clearances to determine if forward video returns are being received.

CAUTION

- When the video on the TFR scope becomes weak or barely visible, the commands will be generated from the radar altimeter. Under these conditions the pilot must assure that the surface is water or other low reflectivity smooth surface by visual means or by comparison with the other TFR channel.
- If the E-scope presentation is unusable for any reason, terrain following operation during night or IFR conditions should be terminated.

SIT Mode Display.

The SIT mode of the TFR provides a scope display of targets that are level with, or higher than the altitude of the aircraft. Since this mode does not provide a margin of vertical clearance, the scope display should only be used as a reference for avoiding obstacles and not for overflying obstacles.

TF Drift.

The TF drift caution lamp provides an indication that the TFR antenna in the channel operating in the TF mode is displaced from aircraft centerline by 4.5 ($\pm$0.5) degrees or greater. This antenna displacement can be caused by a malfunction in the TFR, an erroneous drift signal from the bomb nav system, or actual aircraft drift of 4.5 ($\pm$0.5) degrees or greater, e.g., with a true airspeed of 450 knots, 35 knots of direct crosswind will give 4.5 degrees of drift. Terrain following flight must not be attempted when this lamp is on steady, until the complete drift angle accuracy check has been accomplished and verified in accordance with "TFR Inflight Checks," Section IV. The lamp may come on momentarily during turning maneuvers. This is a normal condition. If the lamp is on steady prior to or during a blind letdown, the aircraft must not descend below the minimum enroute altitude until the complete drift angle accuracy check has been accomplished. If this lamp comes on during a blind letdown after the aircraft is below the minimum enroute altitude or after terrain following flight has been initiated, the aircraft must be flown to the minimum enroute altitude as soon as possible. After stabilization at the minimum enroute altitude, the drift angle accuracy check must be performed prior to resumption of terrain following flight. The condition of this lamp, either on or off, does not exempt the aircrew from performing the drift angle accuracy check prior to terrain following flight. Drift accuracy should be monitored at frequent intervals during TF operation.

Primary Attitude Reference.

The TFR is dependent upon the bomb nav system for primary aircraft attitude and drift reference signals. Since all sources of possible error are not continuously tested, the flight crew should monitor all available cockpit indications (such as cross-checking primary vs. auxiliary attitude indicators, large differences between heading and ground track, unrealistic winds, radar scope stabilization etc.) to detect any malfunctions that may not be detected by the bomb nav system. If an abnormal condition is observed, TF flight should be terminated.

Auxiliary Attitude Reference.

Terrain following flight should be limited to day VFR conditions when the AFRS is furnishing attitude signals to the aircraft subsystems in order that the pilot can maintain a visual reference to the ground since the primary attitude reference will not be available for a comparative check.

Auxiliary Navigation Modes.

Terrain following flight in the auxiliary navigation modes should be limited to day VFR conditions since drift errors may develop if accurate wind data is not maintained in the navigation computer. During TF flight in the auxiliary navigation modes, the flight crew should monitor all available cockpit indications to insure that excessive drift errors do not build up.

Fuel Low Caution Lamp Indications.

During operation over mountain terrain with hard ride selected, the negative "g's" encountered during pushover maneuvers may cause the fuel low caution lamp to light.

WARNING

Prolonged pushovers could result in engine flameout due to fuel starvation. Selecting medium or soft ride will reduce the possibility of this occurrence.

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

The LCOS is integrated with other systems in the aircraft to, (1) deliver bombs, missiles and gun fire and (2) provide homing, navigation and landing information. The system consists of the optical display sight and control panel, located at the left crew station, a lead and launch computing amplifier and a lead computing gyro package, both located in the forward electronics bay. The optical display sight provides indications in the form of two presentations: an aiming reticle, lighted in red, and a set of command steering bars, lighted in green. Refer to figure 1-80 for aiming reticle and command steering bar presentations. The aiming reticle consists of a 2-milliradian center pipper, a 30-milliradian diameter circle, roll reference tabs, a 50-milliradian diameter circle, analog bar reference tabs and range scale, an analog bar presentation, and two deviation indicators. All elements of the aiming reticle are fixed with respect to one another as the aiming reticle display moves about on the sight combining glass. The reticle will move about on the sight in relation to the handset depression angle in elevation, and the drift angle correction in azimuth. The analog bar represents range and appears as a bar of light on the lower half of the 50-milliradian circle. The 50-milliradian circle has fixed indices located at the 3, 4, 5, 6, 7, 8 & 9 o'clock positions, each index denoting a 1000 foot difference in range. The four indices to the left and right of the 6 o'clock index represent increments of 250 feet of range. The three movable indices of the reticle display are the roll tabs. The roll tabs provide an indication of the roll attitude of the aircraft. Roll tab reference in-

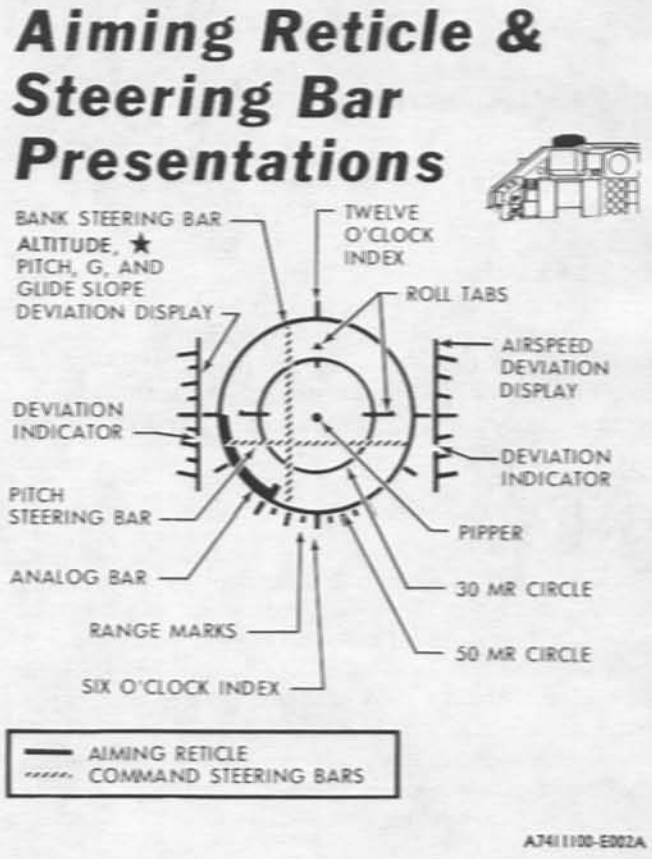


Figure 1-80.

dices are located at 9, 12, and 3 o'clock positions of the 30-milliradian circle. Deviation indicators to the left and right of the reticle rings provide indications of aircraft performance deviations from preset conditions. The left deviation indicator provides pitch, "g," altitude, or glide-slope "fly-to" indications, depending on the selected mode of system operation. The right deviation indicator provides indications of TAS deviations from a preset value; low indications show TAS below preset value and vice versa. Each graduation mark represents 10 knots, and a full scale deviation is 30 knots. The command steering bars consist of a pitch (horizontal) steering bar and a bank (vertical) steering bar superimposed over the aiming reticle. The center pipper of the aiming reticle is the zero reference for the steering bars and when the bars are centered over the pipper a part of the bars is blanked out to provide a 6-milliradian window to aid in keeping the pipper on target. When using the attack radar in the AIR/AIR mode in conjunction with the LCOS the steering bars may be used to provide steering commands to aid in target location. The steering bars function independently of the LCOS mode select knob and provide duplicate indications of the ADI steering bars. Large right azimuth deviations without a corresponding right roll maneuver may cause the bank command bar to drive out of view to the right. Large down pitch steering deviations without a down pitch rate may cause the pitch command bar to drive out of view at the bottom of the sight reticle. Mechanical stops prevent the bars from driving out of view to the left or to the top of the sight reticle. The LCOS functions in nine modes of operation. The system utilizes 28 volt dc power from the main dc bus and 115 volt, three phase, 400 hertz, ac power from the left main ac bus.

Note

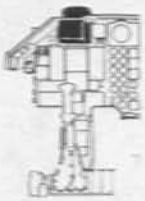
The LCOS reticle will blink when the presentation reaches a horizontal sighting limit in all modes of operation.

CONTROL AND INDICATORS.

LCOS Mode Select Knob.

The LCOS mode select knob (8, figure 1-81) has 10 positions marked OFF, COMM (command), GAR-8, GUN AA (gun air-to-air), GUN AG (gun air-to-ground), RKT AG (rockets air-to-ground), DIV BOMB (dive bomb), LOF BOMB (loft bomb), LEV BOMB

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 Lever.
7. Command Bar Brightness Knob.
8. Mode Select Knob.
9. Aiming Reticle Brightness Knob.
10. Test Switch.
11. ★ Range/Altitude Set Knob.
12. Preset Range Indicator.

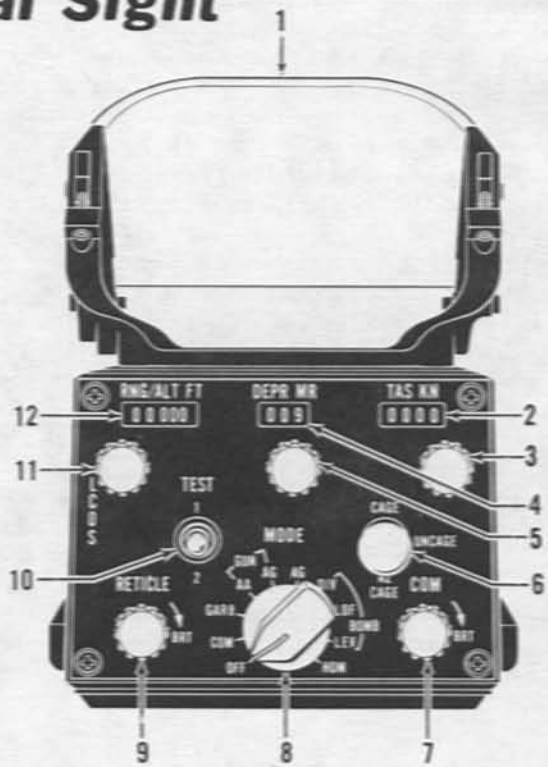


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Figure 1-81.

(level bomb), and HOM (homing). In the OFF position all functions of the LCOS are inoperative. Refer to figure 1-82 for LCOS indications in the various knob positions. Each knob position provides the following indications and functions:

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.

- The COM (command) position is used in conjunction with the instrument system coupler in the ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS and MAN HDG, to provide duplicate indications of those presented on the ADI during operation in these ISC modes. For detailed explanation of LCOS command steering bar indications refer to "Instruments" this section.
- The GAR-8 position is used in conjunction with delivery of GAR-8 (AIM-9B) missiles. When the knob is placed to this position the aiming reticle is caged, and the pipper is positioned approximately 52 MR down in elevation and zero MR in azimuth. The analog bar represents the launch envelope of the missile. When the analog bar is between six o'clock and three o'clock the missile is out of effective launch range, as this area represents 20,000 feet in range. The 6 o'clock position represents the maximum launch range, and the 9 o'clock position represents the minimum launch/breakaway range. The attack radar in the AIR/AIR mode is used in conjunction with the LCOS to provide radar ranging. When the attack radar locks on a target, the analog bar will slew from 9 o'clock toward the 3 o'clock position. If radar lock-on is lost at any time in the attack, the analog bar will drive to the stowed (out-of-sight) position. If target is not in visual contact, the instrument system coupler in the AIR/AIR mode will provide steering bar commands to maneuver the aircraft until visual contact is established. After visual contact has been established, the aircraft is maneuvered to keep the pipper superimposed on the target. The missile may be launched any time the analog bar is between 6 o'clock and 9 o'clock, and the missile is locked on the target. The missile is locked on the target when a tone is heard on the interphone.

Note

If the aircraft is pulling a "g" value in excess of missile turning capabilities, a "g" limit flag will blank out the six o'clock area of the reticle and the missile should not be launched.

- The GUN A/A (guns air-to-air) position is used for air-to-air gunnery attack. When the knob is positioned to GUN A/A position, the preset range indicator is manually set to the maximum or desired firing range of the guns. The 3 o'clock position of the analog bar represents manually set range plus 3000 feet. The 6 o'clock position represents manually set range. The 9 o'clock position represents manually set range minus 3000 feet. If the fixpoint elevation counter is set above the aircraft altitude, the LCOS reticle will stow to the full up position. The reticle is positioned by the lead computing gyro, and the pipper is positioned to the computed lead angle. The left and right deviation indicators will be zeroed. The attack radar in the AIR/AIR mode is used in conjunction with the LCOS to provide radar ranging. When the attack radar locks on a target the analog bar will slew from the stowed (9 o'clock) position toward the 3 o'clock position. If the target is not in visual contact the instrument system coupler in the AIR/AIR mode will provide steering bar commands to maneuver the aircraft until visual contact is established. With visual contact established, the target is tracked by maneuvering the aircraft to keep the pipper superimposed on the target. As range is decreased the analog bar will recede toward the 9 o'clock position. If the manual range selected represents maximum gun firing range, the target will be at maximum firing range when the analog bar is at the 6 o'clock position, and the guns may be fired.

Note

- The pipper should be held on the target for approximately 2 seconds after initial acquisition to allow the system to stabilize.
- If radar lock-on is lost, the analog bar will slew to the stowed (out of sight) position and manually set range is automatically used for lead angle computations.

- The GUN A/G (guns air-to-ground) position is used for attacks on ground targets. The attack conditions of dive angle, true airspeed, and firing range are preselected, and from these conditions the aiming reticle depression angle (lead angle) is determined. The values of depression angle, range, and true airspeed are manually set into the LCOS. Dive angle minus angle of attack is manually set into the glide/dive angle counter on the bomb-nav control panel. The TFR in conjunction with the attack radar, generates the radar range input to the LCOS. The aircraft is maneuvered into the dive attack at a predetermined range from the target. During the dive, the aircraft is maneuvered to zero the left and right deviation indicators and to position the aiming reticle pipper on the target. When the attack radar system locks on, the analog bar slews from the 9 o'clock position toward

LCOS Mode Select Knob Positions Versus Indications ★

	AIMING RETICLE				ANALOG BAR			RIGHT DEVIATION	LEFT DEVIATION	RETICLE BLINK	COMMAND BARS
	Azimuth		Elevation								
	Manual	Auto Ball	Manual	Auto Ball	Manual	Visual CCIP	Manual/ Auto Bomb	Airspeed	Altitude, Pitch, G, Glide Slope	Azimuth Limit	
OFF	—	—	—	—	—	—	—	—	—	—	Pitch and bank steering bars present with instrument system coupler mode selector knob in ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS, MAN HDG and AIR/AIR position.
COM (Command)	Cage	Cage	Man Dep.	Man Dep.	Stowed	Stowed	Time/ Range-To-Release	True Airspeed	ILS, AILA, Glide Slope* Pitch Climb Deviation	Signal blinks if reticle in azimuth limit	
GAR-8 (AIM-9 Missile)	Cage	Cage	Cage	Cage	Range Envelope of GAR-8 Missile			Inoperative	Inoperative		
GUN AA (Gun Air/Air)	Positioned to Computed Lead Angle				Deviation From Preset Maximum Gun Range			Inoperative	Inoperative		
GUN AG (Gun Air/Ground)	Cage	Cage	Man Dep.	Man Dep.	Deviation From Preset Range			True Airspeed	Pitch Dive Deviation		
RKT AG (Rocket Air/Ground)	Cage	Cage	Man Dep.	Man Dep.	Deviation From Preset Range				Pitch Dive Deviation		
Div Bomb (Dive Bombing)	Drift Angle	CCIP	Man Dep.	CCIP	Deviation From Preset Range	Stowed	Deviation From Preset Range		Pitch Dive Deviation		
Lof Bomb (Loft Bombing)	Drift Angle	CCIP	Man Dep.	Man Dep.	Deviation from Preset Pitch Angle				4 G Command		
Lev Bomb (Level Bombing)	Drift Angle	CCIP	Man Dep. Pitch Stab.	CCIP	Stowed	Stowed	Time/ Range-To-Release		Altitude Deviation		*With ISC Mode Selector Knob in ILS or AILA
HOM (Homing)	Radar Homing				Stowed	Stowed	Stowed		Pitch Deviation		

Figure 1-82.

the 3 o'clock position and displays range deviation from the manually set range. The aircraft is out of optimum firing range when the analog bar is between 3 and 6 o'clock. When the analog bar recedes to 6 o'clock, the aircraft is at firing range.

Note

If radar lock-on is lost, the analog bar will slew to the stowed (out of sight) position.

- The RKT A/G (rockets air-to-ground) position is used when firing rockets air to ground. Operation in this position is the same as operation in guns air-to-ground.
- The DIV BOMB (dive bomb) position is used to deliver bombs on a target in diving attacks. The attack conditions of dive angle, airspeed, and altitude are preselected, and from these conditions the aiming reticle depression angle is determined. Values of release range, airspeed, and depression angle are set on the LCOS. Dive angle minus angle-of-attack is set in the bomb nav glide/dive angle counter on the bomb nav control. The armament select panel is set for the type of release desired. The TFR in conjunction with the attack radar will generate a radar range input to the LCOS. The aircraft is maneuvered into a dive at a predetermined range and altitude from the target. During the dive, the aircraft is maneuvered to zero the left and right deviation indicators and to position the aiming reticle pipper on the target at weapon release. The aiming reticle is positioned in azimuth by drift angle correction. When radar lock-on occurs, the analog bar will slew from the 9 o'clock position toward the 3 o'clock position and display radar range deviation from manually set release range. As the aircraft closes on the target, the analog bar will recede toward the 9 o'clock position. When the analog bar reaches the 6 o'clock position the weapon is manually released.
- The LOF BOMB (loft bomb) position is used for bombing targets using the loft bombing maneuver. For loft bombing the LCOS generates a release signal at the proper point in the loft bomb maneuver for weapon release. Either the dual bombing timer or bomb-nav system is selected as the source for pullup signal. The delivery mode selector knob is positioned to ANGLE, and the release angle is set into the bomb nav system glide/dive angle counter. At the IP the weapon release button is depressed to start the timer and held throughout the loft maneuver to effect a release. The aircraft is flown at a predetermined altitude and airspeed to the pullup point. When the pullup signal is generated, the PULL UP lamp on the left main instrument panel lights and the aircraft commander initiates a 4 "g" pullup maneuver. At pullup, the signal input to the pitch steering bar becomes "g" deviation from 4 "g's". To achieve the desired 4 "g" pullup, the aircraft is placed in a climb to center the pitch steering bar and the left deviation indicator. The pitch steering bar represents an error of plus and minus 3 "g's" at full displacement, while the left deviation indicator represents a 1 "g" error at full displacement. Therefore, the left deviation indicator should be used to hold the 4 "g" pullup after it has been achieved using the pitch steering bar. As the release angle is approached, the range analog bar will recede toward the 9 o'clock position. When the bar reaches the 6 o'clock position (aircraft pitch angle equals the release angle set into the bomb-nav system), the LCOS will generate a release signal, the RELEASE lamp on the left main instrument panel will light and the weapon will be released.
- The LEV BOMB (level bomb) position is used for visually delivering bombs on a target from level flight. During level bomb operation in AUTO BOMB or visual CCIP the aiming reticle and pipper are positioned in elevation and azimuth to the continuously computed impact point. For manual operation the delivery mode selector knob is positioned to MAN. In the level bomb mode, the LCOS provides drift angle correction to position the reticle in azimuth. The attack conditions of airspeed and altitude are selected and from these conditions the depression angle is determined. The airspeed, altitude (AGL), and depression angle are manually set in the LCOS. The aircraft is flown so that the target moves down the aiming reticle from the 12 o'clock position toward the pipper. When the pipper and target coincide the weapon can be released by depressing the weapon release button on the control stick grip.

Note

During level bombing when the bomb nav altitude/test selector knob is in CAL, the left deviation indicator should not be used for altitude reference. For left deviation altitude to be accurate, fixpoint elevation setting and altitude calibration must be correct.

- The HOM (homing) position is used to aid in visual detection of ground radar targets. The radar homing and warning system will generate signals representative of the angular position of the ground radar. These signals are used in the LCOS to position the reticle over the ground radar within the field of view of the LCOS sight. The crew member will visually search for the ground radar in the vicinity of the reticle pipper. When visual contact is made, the target can then be attacked using the desired procedures.

Note

The LCOS reticle will blink when the RHAW is not tracking the target.

LCOS Range/Altitude Set Knob.

The range/altitude set knob (11, figure 1-81), located on the optical sight control panel, is used to set in the six o'clock range index (air-air mode) or the left deviation indicator (level bombing mode). A numerical preset range indicator (12, figure 1-81), located directly above the knob, will display, in feet, the selected setting.

LCOS Reticle Depression Set Knob.

The reticle depression set knob (5, figure 1-81), 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 1-81), 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 1-81), located on the optical sight and control panel, is used to set in a desired true airspeed. A preset true airspeed indicator (2, figure 1-81), 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 1-81), 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 aircraft 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 Lever.

The aiming reticle cage lever (6, figure 1-81), 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 (1, figure 1-4), located under the forward contour of the left crew member'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 1-81), 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.

Command Bar Brightness Knob.

The command bar brightness knob (7, figure 1-81), 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.

PENETRATION AIDS.

Three penetration aids control panels are shown in figure 1-83. For all further information pertaining to penetration aids, refer to classified supplement, T.O. 1F-111E-1-2.

MISCELLANEOUS EQUIPMENT.

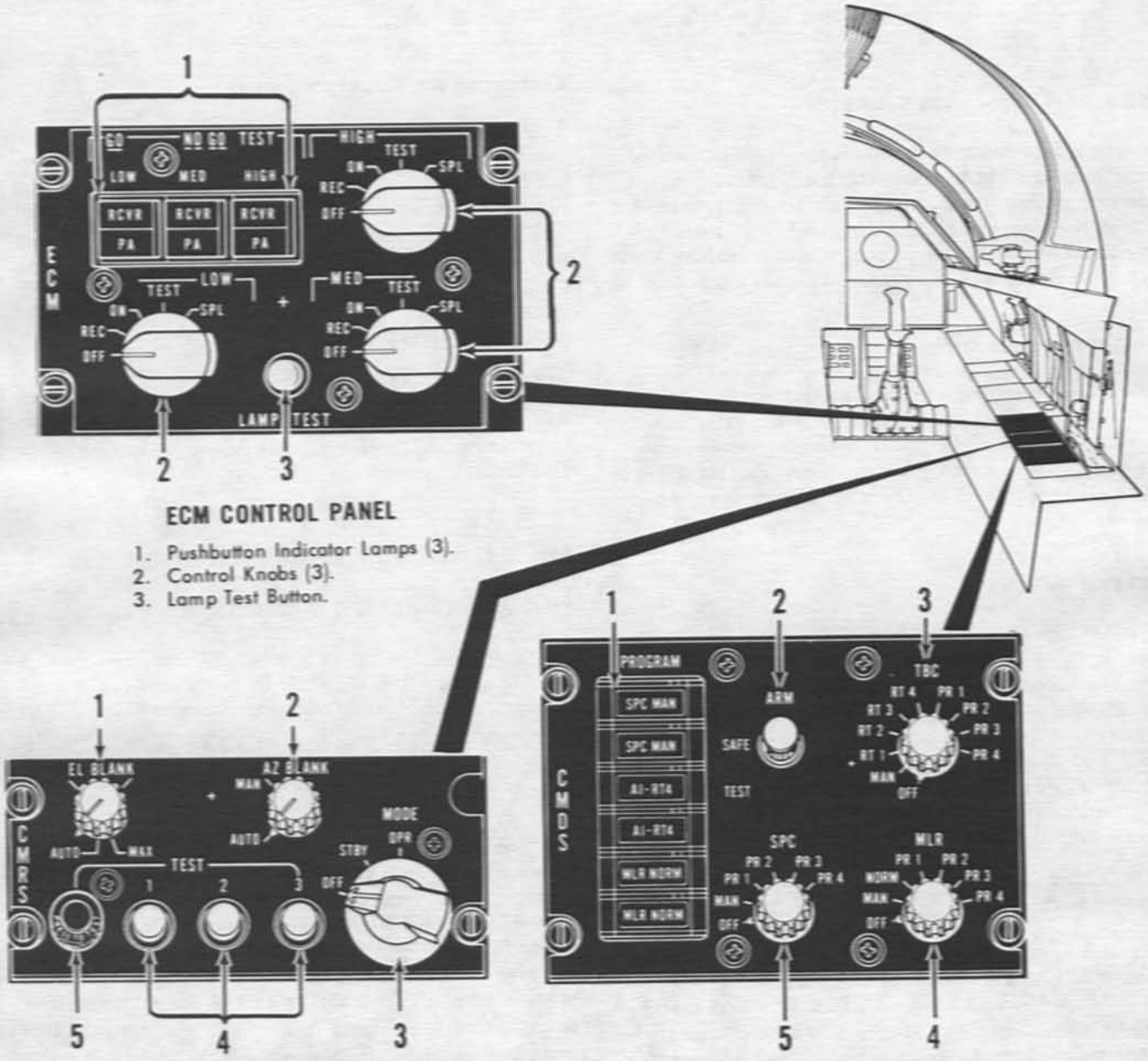
THERMAL RADIATION PROTECTION.

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.

The side curtains (15, figure 1-2) are mounted along the upper edge of each canopy hatch on either side of the center canopy beam. When stowed the curtains are folded as an accordion in the shape of a fan with the hinge forward. As each 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.

Penetration Aids Control Panels (Typical)



ECM CONTROL PANEL

1. Pushbutton Indicator Lamps (3).
2. Control Knobs (3).
3. Lamp Test Button.

INFRARED RECEIVER (IRRS) CONTROL PANEL

1. Elevation Blanking Control Knob.
2. Azimuth Blanking Control Knob.
3. Function Selector Knob.
4. Test Buttons (3).
5. Ready/Test Indicator Lamp.

COUNTERMEASURES DISPENSER (CMDS) CONTROL PANEL

1. Dispenser Program Module Indicator Lamps (6).
2. CMDS Arming Switch.
3. Track Break Chaff Mode Selector Knob.
4. Missile Launch Response Selector Knob.
5. Special Purpose Chaff Mode Selector Knob.

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Figure 1-83.

Forward Panel.

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 panels are 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 panels will fall forward on the glare shield. The slide catches on each side 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 panels.

CREW ENTRANCE STEPS.

Crew entrance steps located externally on each side of the cockpit are provided for use with a ground support equipment crew entrance ladder. The steps are manually extended or stowed from the ground.

ANTI-G SUIT.

Each anti-G suit is connected to the aircraft pressure source by an anti-G suit hose (5, figure 1-45), located on each crew station oxygen-suit control panel. Pressure for the anti-G suit is supplied from the engine compressor section. A test button (2, figure 1-45) marked ANTI-G PUSH-TO-TEST, 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 (13, 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 1-18 and 4, figure 1-42). 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. 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.

CHART STOWAGE.

Two chart stowage compartments are located on each side of the lighting control panel (45 and 50, figure 1-2). 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.

EJECTION SYSTEM SAFETY PIN STOWAGE.

A stowage compartment (10, figure 1-18), located at the aft end of the left sidewall, is provided for stowing the ejection system safety pins.

SPARE LAMP AND FUSE HOLDER STOWAGE.

A stowage compartment (5, figure 1-42), located at the aft end of the right sidewall, is provided for stowing spare light bulbs and fuses.

CHECKLIST STOWAGE.

A space for stowing the checklist is provided on the left sidewall (1, figure 1-18). A nylon strap retains the checklist in place.

FOOD STOWAGE COMPARTMENT.

A food stowage compartment (52, figure 1-2) is provided for the crew on the left side of the aft bulkhead. The door of the compartment is held closed by a spring-loaded latch.

RELIEF CONTAINER STOWAGE.

Relief containers for each crew member are located in small compartments (53, figure 1-2) 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.

A hood stowage compartment (41, figure 1-2), located on the right side of the aft bulkhead, is provided to store the attack radar scope hood.

Servicing Diagram

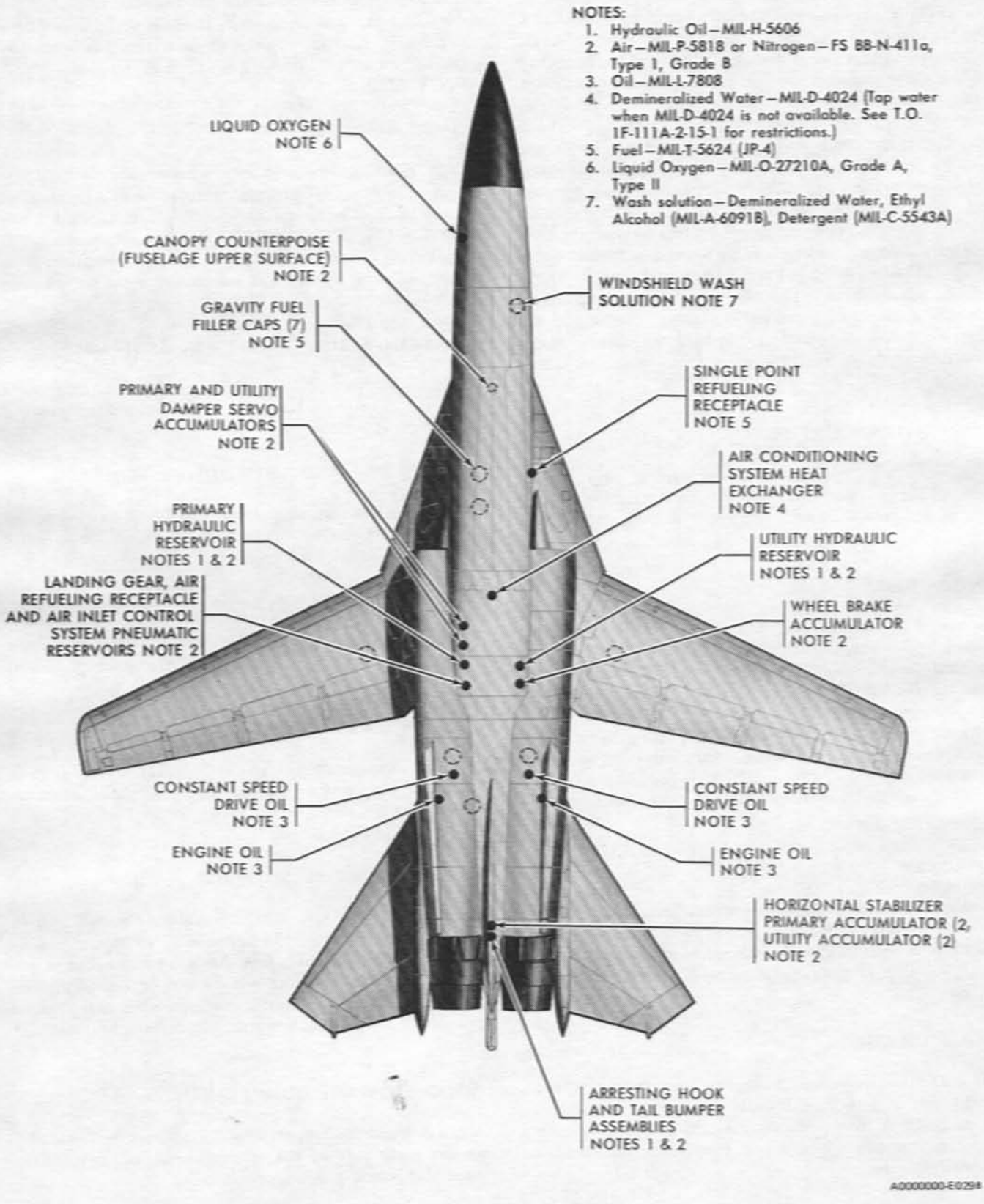

Figure 1-84.

CORRECTION CARD HOLDERS.

Card holders are provided under the glare shield for EPR setting, compass correction and UHF frequency channels. Each card holder is attached by spring tensioned hinges riveted to the glare shield. The card holders are pulled out into position for reading purposes and spring back against the lower side of glare shield when released.

CHART HOLDER.

A chart holder is provided to clearly display approach charts where they can be easily followed during instrument letdowns. The holder is a rectangular transparent pane the size of an approach 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 the holder. The holder is stowed in a receptacle (50, figure 1-2), located in the aft console, when not in use.

LIQUID CONTAINERS.

Two insulated liquid containers (43, figure 1-2) provide the crew with hot or cold liquids during flight. The containers are stowed in recessed receptacles in the aft bulkhead, outboard of each head rest. A spring-loaded latch on the front of each receptacle holds the respective containers 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.

This is the last page of Section I.

SECTION II

NORMAL PROCEDURES

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Note

- It is the responsibility of the aircraft commander to assure that all checklist procedures are complied with.
- For all weapon related checks, refer to the appropriate Weapons Manual.

PREPARATION FOR FLIGHT.

FLIGHT RESTRICTIONS.

Refer to Section V for the operating limitations imposed on the aircraft.

FLIGHT PLANNING.

Refer to the Performance Appendix to determine takeoff, cruise control, fuel planning and management, and landing data necessary to complete the mission.

TAKEOFF AND LANDING DATA CARDS.

Refer to the Performance Appendix for information necessary to complete the Takeoff and Landing Data Card in the Flight Crew Checklist, T.O. 1F-111E-1CL-1. If the takeoff distance exceeds one-half the available runway length, the acceleration check portion of the takeoff and landing data card shall be completed.

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 aircraft and crew module loading information.

WARNING

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. In the event that combined crew weight, including personal equipment, exceeds 430 pounds, or the weight differential between the two occupants exceeds 65 pounds, low altitude safe escape will be compromised and landing impact acceleration will increase. To assure stability of the crew 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. Flight Crew Checklist T.O. 1F-111E-1CL-1 is issued as a separate document.

PREFLIGHT CHECK.

BEFORE EXTERIOR INSPECTION.

Check Form 781 for aircraft status and release. Check aircraft and crew module weight and balance chart C for correct limits.

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 postflight; therefore, duplicate inspections and operational checks of systems have been eliminated except for those needed in the interest of flight safety: 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 and all protective covers removed. Check that gravity fuel filler caps are flush. Check gear pins removed, and uplock assembly for proper positioning (as indicated by the red tip of the hook assembly being aft of the uplock roller guide). Check angle-of-attack and side slip probes for cleanliness and freedom of movement. Check the fuel/dump vent valve for security and freedom of movement.

Note

- For normal operations, the speed brake/main landing gear door ground lock should be left installed until one engine has been motored or started. This will prevent sagging of the door after the ground lock is removed.
- The stores station inspection and bomb preflight will be accomplished concurrently with the aircraft exterior inspection.
- The landing gear emergency system may be actuated by contact with the lever actuator located in the main wheel well on the left side. Exercise care when inspecting this area.

BEFORE ENTERING COCKPIT. (AC)

1. Ejection handle safety pins (2)—Installed.
2. Canopy center beam safety pins (3)—Installed.
3. Bilge pump lockpin—Stowed.
4. Quick rescue kit—Stowed. (If applicable)
5. Emergency oxygen bottle pressure — Check, 1400-2500 psi.
6. All circuit breakers—In.
7. Ground check panel—Check and close door.
   - Computer power switches (3)—On.
   - Central air data computer power switch—POWER.
   - Ground ignition cutoff switch—NORM.
   - Gyros power switch—GYROS.
   - Mach trim test switch—NORM.
   - Fire detect switches—NORM.
8. Utility light—Stowed as desired.
9. Radio beacon set—ON or as applicable.

BEFORE ENTERING COCKPIT. (WSO)

1. Ejection handle safety pins (2)—Installed.
2. Canopy center beam safety pins (3)—Installed.
3. Survival equipment compartment door—Closed and sealed.
4. Crew module chaff dispenser control lever—As required.

   The lever should be ON over friendly territory and OFF as directed by tactical requirements.
5. Publications—Check.

INTERIOR INSPECTION.

Power Off. (AC)

1. Harness and leads—Attach and check.
   - Insure that the yoke of the restraint harness is adjusted firmly against the neck with head against headrest and sitting erect to allow full reel-in, in the event of subsequent ejection.
   - Restraint harness and inertia reel—Checked. Check the condition of the restraint harness. Check operation of the inertia reel in the locked and unlocked position.
   - Oxygen regulator—Inserted in harness receptacle.

CAUTION

Valve port screens are easily damaged by improper/careless handling or placing fingers on screens.

   - Anti-g suit hose—Connect and check. (As applicable)
2. Pressure suit ventilation knob—Full CCW. (As applicable)
3. Oxygen system and personal equipment—Checked.
   - Oxygen mask and communications cord—Connected.

- Oxygen control lever—OFF then ON.
  Check the control lever is OFF and after several breaths note that breathing becomes more difficult due to the restriction of the anti-suffocation valve. Also observe that the anti-suffocation valve unseats with each inhalation. Place the oxygen control lever to ON.
- Oxygen control knob—Checked, then NORM.
- Oxygen control knob—NORM.
  Inhale and check movement of the diluter valve through the screen in the side of the regulator.
- Oxygen control knob—100 percent.
  Inhale and check that the diluter valve does not move.
- Oxygen control knob—EMER.
  Check that a positive pressure is felt in the mask and check that the diluter valve does not move.

4. Interphone panel—Set.
5. Auxiliary pitch trim switch—STICK.
6. Flap/slat switch—NORM.
7. Auto TF switch—OFF.
8. Rudder authority switch—AUTO.
9. Flight control disconnect switch—NORM. (Cover down)
10. Throttles—OFF.
11. Speed brake switch—IN.
12. Ground roll spoiler switch—OFF.
13. Anti-skid switch—OFF.
14. Flight instrument reference select switch—PRI.
15. Landing/taxi lights switch—OFF.
16. Compressor bleed valve control switches—AUTO.
17. Flares/chaff dispense switch—OFF.
18. Nuclear consent switch—Cover down and safetied.
19. Master arm and release switch—OFF.
20. Landing gear handle—DN.
21. Auxiliary brake handle—Pull. (If previous landing not within two hours)
22. Utility hydraulic system isolation switch—NORM.
23. Control system switch—NORM.
24. Arresting hook handle—In.
25. ECM pod control switch—STBY. (If installed)
26. Air/air IR missile switch—OFF, guard down.
27. Gun selector switch—OFF.
28. Lead computing optical sight control panel—Set.
    - Mode selector knob—COM.
    - Caging lever—CAGE.
29. Radar altimeter control knob—OFF.
30. Instrument system coupler mode selector knob—NAV.
31. Dual bombing timer—Set to zero.
32. Auxiliary flight reference system compass mode selector knob—Slaved and set. (Present latitude)
33. Windshield wash/rain removal selector switch—OFF.
34. Pitot/probe heater switch—OFF/SEC.
35. Engine/inlet anti-icing switch—AUTO.
36. Antenna select panel (UHF, TACAN, and IFF)—AUTO.
37. Landing gear emergency release handle—In.
38. Fuel panel—Set.
    a. Fuel dump switch—OFF.
    b. Air refueling switch—CLOSE.
    c. Fuel tank pressurization selector switch—AUTO.
    d. Fuel transfer knob—OFF.
    e. Engine feed selector knob—OFF.
39. TFR channel mode selector knobs (2)—OFF.
40. Spike control switches (2)—NORM.
41. Ground start switch—OFF.
42. Electrical control panel—Set.
    a. Generator switches (2)—RUN.
    b. Battery switch—OFF.
    c. External power switch—OFF.
    d. Emergency generator switch—AUTO.
    e. Emergency generator indicator/cutoff push-button—In. (Safetied)
43. Air conditioning control panel—Set.
    - Temperature control knob—As desired.
    - Air source selector knob—BOTH.
    - Mode selector switch—AUTO.
    - Pressurization selector switch—NORMAL.
    - Air flow selector switch—NORMAL. (If installed)
    - Exchange exit air control switch—NORMAL. (If installed)
44. AC—Ready for electrical power.

Power On. (AC)

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. If the engines are to be started using battery power the following "Power On" checks must be delayed until at least one generator is on the line.
2. External power—Connected. (If applicable) (GO)

3. 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.

Note

- If external power is not obtained when ON is selected, the external power source should be replaced or a battery start made. OVRD should not be selected unless all electronic equipment is off.
- The FWD EQUIP HOT lamp will light in 120 seconds after power on if cooling is not available.

4. Caution lamps—Check.
   a. Following lamps will be lighted:
      PRI ATT/HDG
      PITCH, ROLL, YAW DAMPER
      ANTI-SKID
      α/β PROBE HEAT
      L & R PRI HYD
      L & R UTIL HYD
      L & R ENG OVERSPEED
   b. Following lamps may be lighted:
      CANOPY
      PITCH, ROLL, YAW CHANNEL (must reset)
      AUX ATT (Until initial erection)
      L & R FUEL PRESS
      If the caution lamps in a. are not lighted or lamps other than in b. are lighted, a malfunction is indicated and should be checked prior to starting engines. If the pitch, roll or yaw channel caution lamps are lighted, depress the damper reset button. If lamps remain lighted a malfunction is indicated.
5. Lighting control panel—Check. (If required)
   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 with GO.
6. Malfunction and indicator lamps and stall warning system—Check. (GO)
   - Pitot/probe heater switch—OFF/SEC.

WARNING

If pitot/probe heater switch has been in the HEAT position, residual heat in the probe may be sufficient to cause injury to ground personnel.

- Alpha probe slots—Full up. (Lowest angle-of-attack value)
- Malfunction and indicator lamps test button—Depress and check all malfunction and indicator lamps light, check for intermittent (landing gear) audible warning tone through headset.
- With malfunction and indicator lamps test button depressed.
- Warning horn silence button operation.
- Stall warning system—Check. (After T.O. 1F-111-891)
- Alpha probe slots—Full down. (Highest angle-of-attack value)
- With malfunction and indicator lamps test button depressed, check stall warning lamp flashing, steady audible warning tone through headset, and rudder pedals shaker activated.

Note

When the lamps test button is depressed, the rudder may deflect due to AYC input and the yaw channel caution lamp may light. This is normal.

- Malfunction and indicator lamps test button—Release.

Note

When the lamps test button is released, the yaw channel caution lamp may remain lighted, in which case, reset to put lamp out.

7. Seat and headrest—Adjusted.
8. Anti-skid switch—ON then OFF.
   Check anti-skid caution lamp operation.
9. Flap/slat handle—Corresponds with surface position.
10. Wing sweep handle—Corresponds with wing position.
11. Oxygen quantity—Check.
    Check that oxygen quantity is adequate for mission. Depress oxygen quantity button: Oxygen quantity indicator should decrease to zero. Note that the oxygen quantity caution lamp lights when indication is approximately 2 liters or below. Release the test button and note that the caution lamp goes out at 2 liters and that the quantity indication returns to original value.
12. External air conditioning unit—Check. (If applicable)

Check that air conditioning is connected and functioning properly to provide equipment cooling.

13. Fire detect circuit—Checked.
    Hold the agent discharge/fire detect test switch to FIRE DETECT TEST and check that the wheel well hot caution lamp, the fuselage fire warning lamp, and both engine fire warning lamps are lighted. Release the switch.
14. Landing gear position indicator lamps—Checked.
15. AFRS synchronization indicator—Nulled.
16. Oil quantity indicators—Check, 12 to 16 quarts.
    Check that indicators show 12 to 16 quarts, depress the oil quantity indicator test button, and check that indicators show decrease to 5 quarts on the left indicator and 5.7 quarts on the right indicator. Check that the oil low caution lamp lights. Release test button and check that indicators return to original readings and that the oil low caution lamp goes out.
17. Engine feed selector knob—FWD, then AFT.
    Check that the appropriate fuel pump low pressure indicator lamps light and go out and that the fuel pressure caution lamps go out (if on).
18. Fuel quantity indicators—Check.
    If forward or aft tank pointers or totalizer fail to test or all tank quantities do not add up to the total fuel indication (±1000 pounds), a malfunction is indicated.

Abbreviated Check:

- Fuel quantity indicator test button—Depress and hold until the fwd and aft tank pointers are out of indicator distribution limits to the extent that the fwd pointer will require more than 15 seconds to return to distribution limits.

If for any reason aircraft commander desires to perform the complete fuel quantity gage test, the following checks may be performed.

Complete Check:

- Fuel quantity indicator test button—Depress and hold for the following indications:
- Forward and aft tanks—2000 (±400) pounds.
- Select tank—2000 (±100) pounds.
- Total fuel—2000 (±1250) pounds.

  The external tanks positions need not be checked unless fuel is loaded at any of these positions. If the fuel gage select switch is positioned to an external tank position and no tank is installed at that station, the selected fuel quantity gage should drive to below zero, against the mechanical stop.

Continue the abbreviated or complete check as follows:

- Check that forward and aft tank fuel quantity indicator pointers, totalizer, and select tank pointer move smoothly.

WARNING

If either forward or aft tank fuel quantity indicator pointers indicate a malfunction, do not fly the aircraft.

- Fuel distribution caution lamp—Lighted after 12 seconds.
- Fuel quantity indicator test button—Release.
- Fuel distribution caution lamp — Remains lighted for 10 to 15 seconds, then goes out.

19. Engine feed selector knob—AUTO.
    Select AUTO when the forward tank pointer is approximately 2000 pounds outside the bar index of the fuselage fuel quantity indicator.

Note

If fuel tank expansion space has been reduced due to fuel overfill or thermal expansion, some fuel venting may occur while the fuselage fuel quantity indicators are returning from the test indications if the engine feed selector knob is positioned to AUTO too soon. Fuel venting must cease prior to takeoff.

- Fuel distribution caution lamp—Lighted until distribution is within limits.

Note

If a malfunction is indicated in the fuel distribution system, position the engine feed selector knob to OFF to preclude possible fuel venting.

- Appropriate fuel pump low pressure indicator lamps—Light and go out.
- All indicators—Return to original indications.

20. Fuel transfer knob—AUTO, or as applicable.
    Fuel pump low pressure indicator lamps 7 thru 12 should light and go out unless the tank is empty and then the lamp should remain lighted.
21. Report ready for engine start. (Verified by GO)

Power Off. (WSO)

1. Harness and leads—Attach and check.
   - Insure that the yoke of the restraint harness is adjusted firmly against the neck with head against headrest and sitting erect; to allow full reel-in, in the event of subsequent ejection.
   - Restraint harness and inertia reel—Checked. Check the condition of the restraint harness. Check operation of the inertia reel in the locked and unlocked position.
   - Oxygen regulator—Inserted in harness receptacle.

CAUTION

Valve port screens are easily damaged by improper/careless handling or placing fingers on screens.

   - Anti-g suit hose—Connect and check. (As applicable)
2. Oxygen system and personal equipment — Checked.
   - Oxygen mask and communication cord—Connected.
   - Oxygen control lever—OFF then ON. Check the control lever is OFF and after several breaths note that breathing becomes more difficult due to the restriction of the anti-suffocation valve. Also observe that the anti-suffocation valve unseats with each inhalation. Place the oxygen control lever to ON.
   - Oxygen control knob—Checked then NORM.
   - Oxygen control knob—NORM. Inhale and check movement of diluter valve through the screen in the side of the regulator.
   - Oxygen control knob—100 percent. Inhale and check that the diluter valve does not move.
   - Oxygen control knob—EMER. Check that a positive pressure is felt in the mask and check that the diluter valve does not move.
3. Pressure suit ventilation knob—Full CCW. (As applicable)
4. Electronic countermeasures destruct control panel—Lockout pin installed. (If applicable)
5. Countermeasures dispense system arm switch—SAFE.
   - Mode selector knobs (3)—OFF.
6. Infrared receiver system control knob—OFF.
7. Electronic countermeasures pod operate knobs (2)—OFF. (If installed)
8. Interphone panel—Set.
9. HF control switch—OFF.
10. Strike camera switches (2)—OFF.
11. Attack radar function selector knob—OFF.
12. Electronic countermeasures (ALQ) control knob —OFF. (If installed)
13. Weapons control panel—Set:
    a. Weapons bay auxiliary control switch—NORM.
    b. Weapon bay door control switch—Checked. Check that the weapon bay door control switch position agrees with the position of the weapon bay doors.

WARNING

If the position of the weapon bay doors disagrees with the position of the switch, the doors will actuate (open/close) when power is applied to the aircraft.

14. Nuclear control panel knobs (2)—OFF.
15. Instrument landing system power switch—OFF.
16. Burst altitude/train lead counter—As required.
17. Target elevation selector switch—As required.
18. Bomb nav mode selector knob—OFF.
19. Bomb nav magnetic variation counter—Set, to best known local value.
20. Attack radar indicator recorder panel—Set.
    - Switches—OFF/NORM.
    - Antenna tilt control knob—Detent.
    - Sensitivity time control knob—OFF.
    - All other control knobs—Fully CCW.
21. RHAW threat panel—Both knobs CCW.
22. RHAW scope panel—Set.
    - Memory control knob—CCW.
    - Sensitivity knob—CW.
    - Reticle/scope intensity knob—CCW.
    - Gate select knob—N.
23. UHF radio—OFF.
24. TACAN—OFF.
25. IFF master control knob—OFF.
26. Scope camera power switch—OFF.
27. WSO—Report ready for electrical power.

Power On. (WSO)

1. UHF radio—On and set.
2. Bomb nav present position—Set. (As required)
   - Platform alignment knob—OFF.
   - Bomb nav mode selector knob—ALIGN.
   - Man fix mode selector button—Depressed.
   - Present position latitude and longitude—Set.
3. Bomb nav mode selector knob—HEAT.

Note

When ambient temperature exceeds 110 degrees F, do not leave bomb/nav mode selector knob in HEAT for more than 5 minutes without external air conditioning.

4. Platform alignment knob—NORMAL.
5. Platform heat indicator lamp—Lighted.
6. Altitude/test selector knob—NORM.
7. Attack radar system camera magazine—Check. Fill in the identification with the desired information and check the clock for proper operation.
8. Report ready for engine start.

BEFORE STARTING ENGINES.

Refer to figure 2-1, "Danger Areas," for the extent of engine intake and exhaust hazard areas, and the engine turbine and starter turbine planes of rotation.

1. Ground crew report—Ready for engine start. Fire guard posted, engine and run area clear, chocks in place, nacelle vent and fire access doors checked for hinge integrity and freedom of movement.

DEFINITIONS.

Hot Start—TIT indicates engine ignition but exceeds the limit specified in Section V. If at any time during start the TIT increases at an abnormally rapid rate or approaches within 50 degrees C of the limit and is still climbing, a hot start can be anticipated.

False or Hung Start—TIT indicates engine ignition but rpm will not increase to IDLE within 2 minutes.

Failure to Start—TIT does not indicate ignition within 20 seconds after throttle advance. RPM will stabilize at the maximum for starter output.

Cartridge Start Misfire—Cartridge fails to ignite as indicated by lack of smoke at the starter exhaust port. There will be no engine rpm indication.

Cartridge Start Hangfire—Cartridge ignites as indicated by smoke at the starter exhaust port, however there will be little or no rpm indication.

If any of the above conditions occur return the throttle to OFF and investigate. The engine should be inspected for residual fuel before a second start is attempted. If no fuel is visible a second start may be attempted. The engine should be motored until TIT is approximately 100 degrees C before advancing the throttle to minimize the possibility of a hot start. If visible fuel or vapors are found the engine must be cleared using the pneumatic starter as follows:

ENGINE CLEARING.

- Engine ground start switch—PNEU.
- Affected engine throttle—Lift.
  Lift the throttle of the affected engine out of the OFF detent to motor the engine. This may be done any time rpm is below 20 percent.

CAUTION

To avoid a possible hot start do not advance the throttle.

- Affected engine throttle—Release.
  Release the throttle to OFF prior to the time limit specified for starter operation in Section V.

CAUTION

To prevent possible engine damage due to overtemperature, do not attempt to restart the engine until TIT is below 100 degrees C.

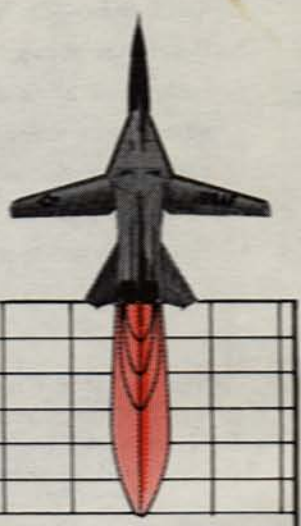
STARTING ENGINES. (AC)

Engine starts can be accomplished by using air pressure from a ground source or by a pyrotechnic cartridge. Only the left engine has cartridge starting capability. Either engine may be started by the use of external air when supplied by an adequate source; however, when using the MA-1A starter cart, left engine starting capability is marginal. For normal flight operations it is recommended that the right engine be started first with external air due to the higher starter torque available. With either engine operating, the remaining engine may then be started by pneumatic crossbleed. Electrical power required for engine starting may be supplied either by the aircraft battery or by an external source.

Danger Areas (Typical)

ENGINE: TF30-P-3
DATA BASIS: ESTIMATED
DATE: 1 APRIL 1971

DISTANCE FROM JET NOZZLE (FEET)	MAX AB THRUST TEMPERATURE (FAHRENHEIT)	MAX AB THRUST VELOCITY (MPH)	MILITARY THRUST TEMPERATURE (FAHRENHEIT)	MILITARY THRUST VELOCITY (MPH)
20	2000°	1750	500°	1100
40			300°	455
60	600°	410	200°	210
80				
100	400°	200	150°	105
120				
140	300°	125	125°	70
160			110°	50
180				
200	200°	55		
JET WAKE RADIUS (FEET)	30 15 0 15 30			



DISTANCE FROM JET NOZZLE (FEET)	IDLE THRUST TEMPERATURE (FAHRENHEIT)	IDLE THRUST VELOCITY (MPH)
30	150°	70
40		
50	125°	40
60		
70		
80		
JET WAKE RADIUS (FEET)	30 15 0 15 30	

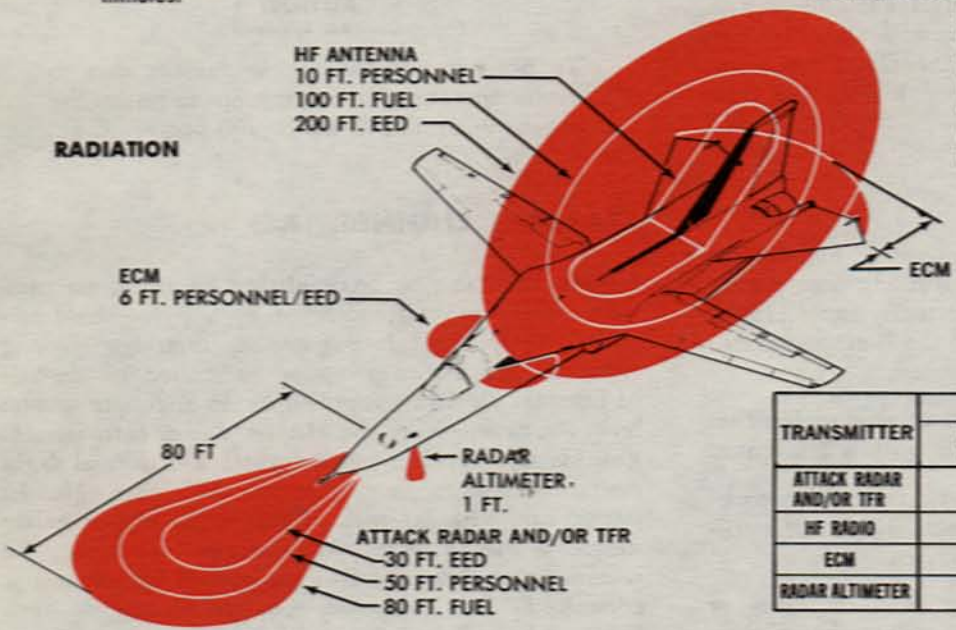
ROTATING PLANES OF ENGINE TURBINES

TIRE AVOIDANCE

If landings are made which for some reason require maximum braking to stop the aircraft, avoid tire area for 1 hour and 15 minutes after aircraft has stopped. If necessary, approach from the front or rear only. If thermal release plugs have blown allowing tires to deflate, danger of explosive failure is minimal; however, danger of fire exists for at least one hour.

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.



TRANSMITTER	SAFE DISTANCE FROM ANTENNAS		
	PERSONNEL HEALTH HAZARD	ELECTRO EXPLOSIVE DEVICES (EED)	FUEL
ATTACK RADAR AND/OR TFR	50 FT.	30 FT.	80 FT.
HF RADIO	10 FT.	200 FT.	100 FT.
ECM	6 FT.	6 FT.	---
RADAR ALTIMETER	1 FT.	---	---

A0000000-E064C

Figure 2-1.

WARNING

- Do not attempt a pneumatic start or fly the aircraft with an unfired cartridge in the breech. Abnormal cartridge conditions of an explosive nature could be generated due to the combination of vibration and high temperatures that can exist in the engine nacelle.
- 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.

CAUTION

- If engine has had insufficient time to cool from a previous operation, do not attempt a restart until TIT is below 100 degrees C. Motoring of the engine will reduce the temperature.
- If either hydraulic hot caution lamp lights during ground operation, failure of hydraulic ground cooling is indicated and the engines should be shut down as soon as possible.

1. Engine ground start switch—PNEU or CARTRIDGE. (As applicable)
2. Applicable engine throttle—Lift to start position.
   a. On cartridge start advance the throttle to IDLE immediately.

WARNING

In the event of aborted start during a cartridge start due to misfire, hangfire, or slow burning cartridge, the breech will not be opened until a time period of 5 minutes has elapsed after attempted start and no smoke can be observed emitting from the starter exhaust.

Note

If battery power only is used during start, only TIT indicators and tachometers are operating until one engine driven generator is supplying power to the ac buses.

   b. Oil pressure—Checked.

Note

- Oil pressure should be indicated within 10 seconds after first indication of rpm.
- During second engine start, check that the engine ground start switch moves to OFF prior to reaching 50 percent engine rpm. Cooling air will not be available if the switch is in any position other than OFF.

3. Engine throttle—IDLE.
   On a pneumatic start advance the throttle to IDLE after the engine rpm reaches 17 percent.

Note

Turbine inlet temperature rise should occur within 20 seconds after advancing throttle.

4. Nacelle vent ejector system—Check.
   During engine start the ground crew will observe the nacelle vent and fire access doors to assure that during engine start and stabilization at idle, that a positive movement in the open direction occurs indicating that the nacelle vent ejector system is functioning. If this door movement is not observed, it is indicative of a broken nacelle vent system duct or a faulty nacelle vent system pressure regulating and shut off valve.
5. Engine instruments—Check.
   a. Fuel flow—1100 pph maximum.
   b. Turbine inlet temperature indicator—710 degrees C maximum.
   c. Idle rpm—57 to 69 percent.
   d. Hydraulic pressure indicators — 2950-3250 psi, caution lamps out.
   e. Idle oil pressure—30 to 50 psi.
   f. Nozzle position—Open.
6. Engine overspeed caution lamp—Out.
7. Generator switch — START (pause), then release to RUN; check caution lamp out.

Note

If the generator caution lamp remains lighted, place the generator switch to OFF-RESET, hold to START (pause) then release to RUN.

8. Power flow indicator—TIE or NORM. (As applicable)
   The power flow indicator will read NORM if a ground power unit is plugged in or TIE if battery was used or if the left engine was started first.

9. Speed brake ground lock—Removed. (GO)
10. External power switch—OFF.
11. External power and air—Disconnected, if crossbleed is used. (GO)
12. Power on checks—Complete. (If battery start accomplished)
13. Air refueling receptacle—Check. (If required) (GO)
14. Remaining engine—Start. (Repeat steps 1 thru 8)

CAUTION

If left engine is started with external air at high ambient temperatures, starting TIT must be closely monitored for possible overtemperature.

Note

If crossbleed is being used for starting the second engine, obtain ground clearance and then advance the throttle to 80-85 percent on the operating engine, depending on ambient conditions, until second engine reaches 50 percent or until pneumatic ground start switch cuts off, then retard throttle to IDLE.

15. Engine ground start switch—OFF.
16. Power flow indicator—NORM.
17. Emergency generator switch—TEST, ON, then AUTO.

    Place the emergency generator switch to TEST. The emergency generator indicator lamp will light within one second, indicating that the emergency generator is operating within limits. The power flow indicator should display a crosshatch. Check operation of T/R units by noting that the angle-of-attack indexers and the LCOS reticle lamps are lighted. Place the emergency generator switch to ON, check power flow indicator displays NORM. Place the emergency generator switch to AUTO. Check that indicator lamp goes out and that the power flow indicator displays NORM.

AFTER ENGINE START. (AC)

1. Radar altimeter—Set 80 feet.
2. L & R TFR channels—STBY.
3. Wing sweep handle—Set for takeoff.
4. Wing sweep handle lockout controls—ON. (If applicable)

   If fixed stores or multiple weapon racks are being carried, place the respective lockout control to ON.
5. Ground crew check flight controls—Clear. (GO)
6. Flight control and damper system—Check. (GO optional)

Note

During the following checks, the required flight control surface positions will be verified by the control surface position indicator or the ground observer.

a. Slats—Extended.
b. Takeoff trim—Set.
c. Damper switches (3)—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.

d. Flight controls—Checked.

   - Move the control stick aft, then left wing down, right wing down: check for freedom of movement and verify that the control surfaces and surface position indicators correspond with control stick movement. Check that pitch and roll channel caution lamps do not light.
   - Move the control stick full forward, then rapidly full left through the detent to the forward left corner and hold firmly for one second. Verify that the right horizontal stabilizer indicates 12 to 18 degrees down while the stick is held in this extreme position.
   - Move the control stick rapidly full right through the detent to the forward right corner, firmly holding forward pressure. Verify that the left horizontal stabilizer indicates 12 to 18 degrees down while the stick is firmly held for one second in this extreme position, then release.
   - Rudder pedals—Check for more than 25 degrees of rudder in each direction.

e. Damper switches (3)—DAMPER.
f. Damper reset button — Momentarily depressed. (If necessary)

   Check that the pitch, roll, and yaw damper caution lamps go out.

g. Trim—Checked. (Optional)

   Move auxiliary pitch trim switch to OFF, actuate stick trim button to NOSE DOWN

and NOSE UP and check for no movement of stabilizers. Move auxiliary pitch trim switch to NOSE DN, then NOSE UP; check control surfaces travel in response to switch positions. Move auxiliary pitch trim switch to STICK and check trim button NOSE DOWN, NOSE UP, RWD, LWD, and rudder trim left and right; check control surfaces give proper response to trim inputs. Leave control surfaces out of center for subsequent check of takeoff trim system.

7. Flaps/slats—Retracted.

Note

When the control system switch is in NORM and the slats are retracted, a small oscillation may occur in the horizontal stabilizers which will be transmitted through the airframe. This condition is normal and will disappear when the slats are extended.

8. Series trim—Check.
   - Takeoff trim—Set.
   - Trim nose up for one second.
   - Wait for the horizontal stabilizers to stop driving at more than 8 degrees trailing edge up before completing the next step.
9. Auto TF switch—AUTO TF.
   The control stick shall drive slightly forward, the TF fly up off caution lamp shall light and the reference not engaged lamp shall light. These checks are valid whether TF is operational or not.

CAUTION

Do not initiate the next step unless both stabilizers indicate more than 8 degrees trailing edge up. If necessary, place the auto TF switch to OFF and repeat "Series Trim" checks.

10. Surface motion test—Complete. (GO optional)
    - Stability augmentation test switch—SURFACE MOTION, and hold until next step is completed.
    - Flight control master test button—Depress and hold for the following checks:
      - Rudder moves to right, then to the left.
      - Left horizontal stabilizer drives to near zero degrees.
      - Right horizontal stabilizer drives to approximately 10 degrees down.
      - Control system caution lamps do not light.
    - Flight control master test button—Release.
      - Rudder returns to neutral.
      - Both horizontal stabilizers may drift together in pitch.
11. Surface motion and light test—Complete. (GO optional)
    - Stability augmentation test switch—SURFACE MOTION & LIGHTS and hold until next step is completed.

CAUTION

Do not initiate the next step unless the horizontal stabilizers are more than 8 degrees trailing edge up. If necessary, place the auto TF switch to OFF and repeat "Series Trim" checks.

    - Flight control master test button—Depress and hold for the following checks:
      - Rudder initially drives right, then returns to neutral.
      - Left horizontal stabilizer drives to near zero degrees.
      - Right horizontal stabilizer drives to approximately 10 degrees down.
      - Pitch, roll, and yaw damper, channel, and pitch and roll gain changer caution lamps light (8).
    - Master test button—Release.
      - Rudder initially drives left then returns to neutral.
      - Both horizontal stabilizers may drift together in pitch.

Note

If all the lamps do not light, cycle the control system switch to T.O. & LAND and return to NORM, then repeat the "Surface Motion and Light Test" checks. If all lamps still do not light, a malfunction is indicated and correction will be required before flight.

12. Flap/slat handle—Set for takeoff.
13. Auto TF switch—OFF.
14. Damper reset button—Depress momentarily.
15. All caution lamps—Out.
16. Spoiler monitor test—Checked. (GO optional)
    - Flight control master test button—Depress and hold.

- Spoiler test switch—OUTBD and hold until:
  - Outboard spoilers momentarily extend, then retract.
  - Spoiler caution lamp lights.
- Spoiler reset button—Depress.
  Check spoiler lamp out.
- Spoiler test switch—INBD and hold until:
  - Inboard spoilers momentarily extend, then retract.
  - Spoiler caution lamp lights.
- Flight control master test button—Release.
- Spoiler reset button—Depress.
  Check spoiler caution lamp out.

17. Ground roll spoilers/throttles—Check. (GO optional)
    - Ground roll spoiler switch—BRAKE.
      Check all spoilers extend.
    - Left throttle—Advance slightly, then IDLE.
      Check all spoilers retract, then extend.
    - Right throttle—Advance slightly, then IDLE.
      Check all spoilers retract, then extend.
    - Ground roll spoiler switch—OFF.
      Check all spoilers retract.
18. Autopilot—Checked. (Optional)
19. Radar altimeter—Checked.
    Depress and hold radar altimeter control knob; check for an indication of 95 (±12) feet and radar altitude low warning lamp goes out. Select other channel and repeat test.
20. TFR operational check:

WARNING

Do not transmit with the TFR if personnel or equipment are within the dangerous radar emission area. See Figure 2-1.

Note

- If time prohibits pilot accomplishing this check on the ground, both crew members must accomplish inflight prior to TF operation.
- This check must be accomplished on the ground or above low altitude radar altimeter range (5000 feet absolute) to obtain proper light indications.
- When switching channels, or changing clearance plane settings, a momentary TF fail and fly-up maneuver may occur. The autopilot release lever can be held depressed to prevent the fly-up maneuver from occurring.

a. Antenna cage pushbutton indicator lamp—Out.

b. TF, SIT, and GM mode check—Complete.

Note

If, on the ground, the TF warning lamps stay lighted, check angle-of-attack indicator. If the reading is not in the range of plus 2 to plus 6 degrees, moving probe into this range will put the lamps out.

(1) TFR channel mode selector knobs—L TF, R SIT.
  (a) Channel fail caution lamp—Lighted.
      The channel fail caution lamp of the channel in TF should be ON, and the lamp of the channel in SIT should be OFF.
  (b) Reference not engaged caution lamp —Lighted.
  (c) TF fly-up off caution lamp—Lighted.
  (d) TF fail warning lamp—Lighted.

(2) Instrument system coupler pitch steering mode switch—TF.

(3) Radar altimeter bypass switch — BYPASS.
    If check is performed on the ground the switch must be held in the bypass position.
  (a) Check TFR channel fail caution lamps—Out.
  (b) Check TF fail warning lamp—Out.
  (c) TF fly-up off caution lamp—Out.
  (d) Reference not engaged caution lamp —Lighted.

(4) Radar altimeter bypass switch—Release to NORMAL. (Ground check only)
    Any time the aircraft is below 5000 feet absolute with radar altimeter operating, this switch will automatically release to normal.

(5) E scope—Checked.
    Adjust the contrast control until a thin vertical line along the right side of the E scan is discernible. Adjust the memory control knob so the sweep is repainted just prior to the fade point. Set the video knob to midpoint (adjust for optimum target display when at low altitude).

(6) Self-test pulse—Checked.
    Check for the presence of a test pulse.

(7) Zero command line—Check.

(a) Ride control knob—Checked.
Rotate thru each position. Check the zero command line position for proper movement and a smooth curve for the three ride settings.

(b) Terrain clearance knob—Checked.
Rotate thru each position. Check the zero command line position for proper movement and a smooth curve for all clearance settings.

(8) SIT and GM check—Checked.
Rotate the range selector knob from E to 5, checking for following indications: In 15 mile position, scope should show 15 mile range with three cursors evenly spaced. Check 10 and 5 for proper range and 5 evenly spaced range cursors. Switch to GM and check for antenna tilt in 5 NM range. Return range selector knob to E and check range and cursors in GM 5, 10, and 15 as above.

(9) TFR channel mode selector knobs — STBY.

c. Radar altimeter channel selector switch — Opposite channel.

d. Repeat TF, SIT, and GM mode check with TFR channels reversed.

21. Takeoff trim—Set. (GO optional)
After the takeoff trim lamp lights, check the control surface position indicator to insure that the horizontal stabilizer and rudder are within tolerance for takeoff and/or have the ground observer verify.

22. Pitot/heat, angle-of-attack, and engine inlet anti-ice probes—Check. (If applicable)

AFTER ENGINE START. (WSO)

Note

If cooling air is available, this checklist portion may be performed prior to engine start. Loss of cooling air during normal engine start is not sufficient to damage equipment.

1. Bomb nav mode selector knob—ALIGN.
2. Attack radar control panel—Set.
   - Function selector knob—STBY.
   - Antenna tilt indicator—Check +30.
   - Antenna cage pushbutton indicator lamp—Check.
   - Function selector knob—ON.
   - Mode selector knob—GND AUTO or GND VEL.
   - Antenna tilt indicator—Check (±2) degrees.
3. RHAW power audio knob—CW, out of detent.
   The RHAW system requires approximately 5 minutes for warm up.
4. TACAN—T/R and set.
5. IFF—STBY and set.
6. ILS switch—POWER.
7. Scope camera power switch—Set.
8. Bomb nav destination storage—Set.
   - Fix mode destination storage 1 button—Depress.
     When counters stop driving enter destination storage coordinates into the destination position counters. Repeat for destination storage 2 and 3.
9. Fix mode target selector button—Depressed and set.
   Depress the target button and set destination counters to coordinates to destination or first steering point.
10. Ballistics and offset data—Set.
11. Attack radar—Tune.

WARNING

Do not place attack radar to XMIT in parking area unless the area ahead of the aircraft has been checked and cleared of all personnel.

   - Cathode ray tube intensity knob—CW until sweep is just visible on scope.
   - Video control knob—Advance to mid-range.
   - IF gain knob—CW to obtain snow then decrease until snow just disappears.
   - Bezel/range marks—Check and set.
   - Range/azimuth cursors—Checked and set.
12. Attack radar camera—Expose a minimum of 8 clearing frames.
13. RHAW system—Check. (If required)
14. Countermeasures dispense system—Check. (If required)
15. Infrared receiver system—Check. (If required)
16. Electronic countermeasures system (ALQ) control knobs—STBY. (If installed)
17. Electronic countermeasures pod control knobs—STBY. (If installed)
18. HF radio—Set. (If required)

WARNING

HF system electromagnetic radiation may trigger unshielded ejector cartridges, 20mm ammo, rockets or other explosive devices within 200 feet of the aircraft. In addition no personnel should be within 4 feet of the aircraft during ground transmissions.

19. Weapon bay door(s)—As required.
    To close weapon bay door(s):
    a. Weapon bay area—Clear.
    b. Weapon bay door switch—CLOSE.
20. Attack radar system antenna cage lamp—Out.
21. Altitude calibration—Complete.

BEFORE TAXIING. (BOTH)

1. Altimeters—Set.

WARNING

Do not push in on standby altimeter set knob when setting barometric pressure as disengagement of the gear train between the indicating pointers and the barometric scale may occur, resulting in erroneous altimeter readings. Observe that the pointers and barometric scale move simultaneously with set knob rotation.

2. EPR's—Set.
3. TACAN—Checked.
4. Bomb nav mode selector knob—GREAT CIRCLE or above. (When alignment complete)
5. Heading crosscheck—Complete.
   Primary and auxiliary headings should agree within 3 degrees.
6. Ejection handle and center beam safety pins—Remove, display to ground crew, and stow.
7. Chocks removed—Ready for taxi.
   One engine should be advanced to the point where the FWD EQUIP HOT caution lamp goes out. All non-essential forced air cooled avionics equipment (including ECM) should be turned off. If it is necessary to exceed 78 percent rpm, a system discrepancy is indicated.

TAXIING. (COMMAND RESPONSE)

Note

For "Turning Radius" during taxi operations, see figure 2-2.

1. Auxiliary brake handle—In.
2. 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.

Note

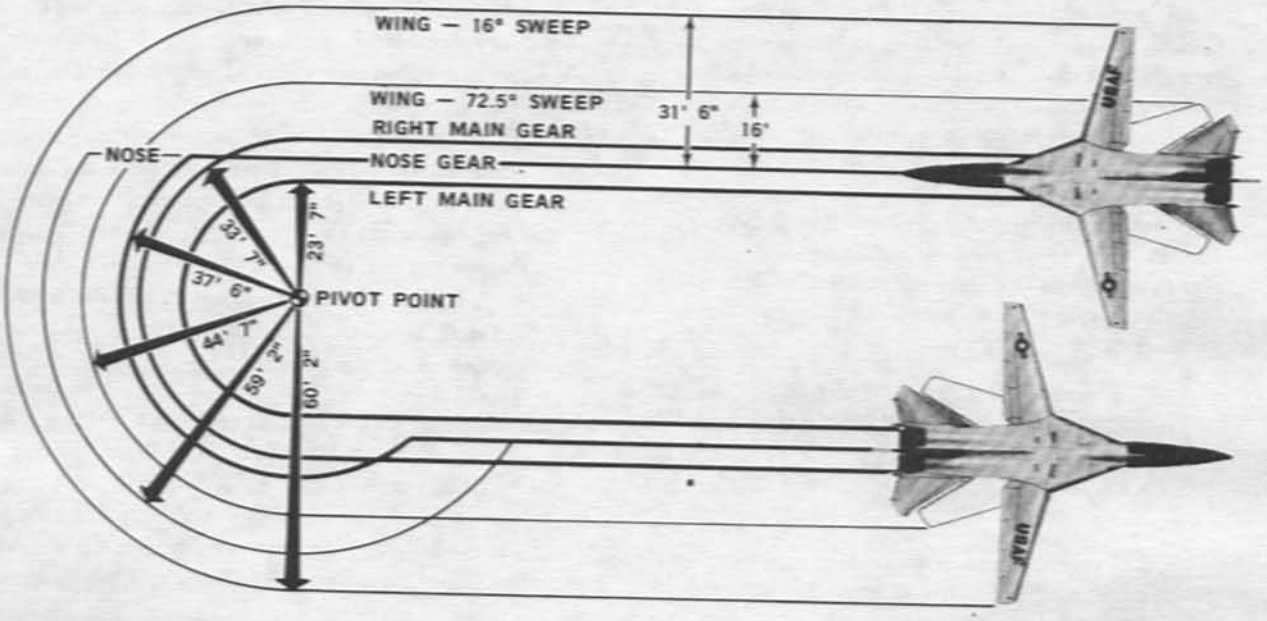
- Full nose wheel steering will not be available when slats are retracted and the flight control switch is in NORM. If full rudder authority is desired, place the rudder authority switch to FULL. Use of T.O. & LAND position is not recommended.
- If turn radius is exceeded, range switch will automatically disengage controlled steering from rudder pedals and NWS/AR lamp will go out.

3. Hydraulic pressure—Checked.
   Check for 2950 to 3250 psi indication.
4. Brakes—Checked.
   Depress brake pedals and check for proper braking.
5. Flight instruments—Checked.
   Check the flight instruments for proper operation during taxi.
6. Weapons bay door—Check closed.

BEFORE TAKEOFF. (COMMAND RESPONSE)

1. Wing sweep, flaps, and slats—Set for takeoff.
   Check the surface position indicator for selected wing, flap, and slat settings.
2. Ground roll spoiler switch—BRAKE.
3. Speed brake switch—IN.
4. Control system switch—NORM.
5. Pitot/probe heater switch—OFF/SEC.
6. Takeoff trim button—Check.
   Depress and hold until takeoff trim lamp lights.

Turning Radius



A0000000-E066A

Figure 2-2.

WARNING

A possible malfunction is indicated if trim has drifted during taxi out. Once trim has been set, with slats extended as indicated by the T.O. trim lamp lighting, and slats have not been retracted the lamp should light immediately when the T.O. trim button is depressed.

7. Fuel quantity and distribution—Checked.
8. Fuel panel—Checked.
9. Anti-collision light — On, position light STEADY.
10. Canopies—Closed and latched, unlock warning lamp out and the lock tab flush.
11. Warning and caution lamps—Checked.
    Check that all warning lamps are out and that caution lamps are compatible with mission. The α/β probe heat and anti-skid caution lamps will be lighted if the switches are left in the OFF position. The α/β probe heat caution lamp will go out when the aircraft weight is off the gear.
12. Helmet visors—Lowered. (As practicable)
13. Takeoff brief—Completed.

Note

If required remove electronic countermeasures destruct lockout pin prior to flight.

TAKEOFF.

RUNWAY CHECK. (BOTH)

1. Anti-skid switch—ON, caution lamp out.
2. IFF—As required.
3. Attack radar video—Check.
4. Flight instruments—Check.

WARNING

Do not take off if the airspeed mach indicator reads greater than mach 0.42. An erroneous CADC output can result in improper mach trim functions of the engine fuel control unit, causing a significant reduction in engine thrust (as much as 40 percent) on both engines when the landing gear handle is placed to UP after takeoff. In the event of a sudden thrust reduction when the landing gear handle is placed to UP, with an accompanying abnormal mach indication, recover normal thrust by returning the landing gear handle to DN and land as soon as practicable.

5. Throttles—MIL. (Check bleed valve indicator indicates NONE)

Note

If additional engine instrument checks are desired, MAX AB power may be selected prior to brake release.

6. Brakes—Release.
7. Throttles — MAX AB. (Check engine instruments)

NORMAL TAKEOFF.

Normal takeoffs will be accomplished with wing sweep positioned at 16 degrees and 25 degrees flaps. The recommended flap setting provides an optimum trade-off between single engine rate of climb at takeoff speed and ground roll. After lining up on the runway, hold brakes and complete necessary pre-takeoff checks. With brakes held, engines may be operated up to maximum afterburner at gross weights in excess of 60,000 pounds with no skidding of the tires. Below 60,000 pounds skidding may be encountered and reduced afterburner thrust should be used. To begin takeoff roll, smoothly release brake pedals. It is recommended that maximum afterburner thrust be used for all normal takeoffs. Asymmetric afterburner operation presents no directional control problem and can easily be controlled with nose wheel steering or rudder as required. Differential wheel braking will extend takeoff roll. Nose wheel steering should be disengaged as the rudder becomes effective (50 to 70 knots). At 15 knots below takeoff speed initiate back stick pressure to achieve a rotation rate that will result in a takeoff attitude at the recommended takeoff speed. Adequate longitudinal control may be available to lift the nose wheel from the runway at lower speeds, but it is recommended that this not be done since it will lengthen the takeoff distance slightly due to increased drag.

Note

- Rotational characteristics of the aircraft will vary with gross weight, center-of-gravity position and external stores loading. Certain combinations (light gross weight and/or aft center-of-gravity location) will result in a fairly rapid rotation when aft stick force is applied. With a heavy aircraft and/or a forward center-of-gravity location immediate rotation may not occur with aft stick movement and a much slower rate of rotation may be experienced. In some cases, takeoff attitude may not be achieved until takeoff speed is reached. Therefore, takeoff should not be aborted due to failure to rotate until takeoff speed is attained.

- If obstacle clearance is required, aircraft pitch attitude should be increased after takeoff to 15 degrees (not to exceed 13 degrees angle-of-attack). Do not retract flaps or slats until the obstacle has been cleared, pitch attitude reduced, and angle-of-attack is within recommended flap retraction limits.

Immediately after nose wheel lift-off, a forward stick motion may be required to arrest the rotation of the aircraft, and the stick should be adjusted to maintain 10 degrees of pitch attitude for aircraft lift-off. Landing gear retraction should be initiated when safely airborne. After lift-off, maintain this attitude constant and, as the aircraft accelerates, retract the flaps/slats incrementally at a rate which will result in an angle-of-attack not to exceed 10 degrees. During heavy gross weight takeoff conditions (above 90,000 pounds), it will be necessary to maintain an angle-of-attack between 8 and 10 degrees to avoid exceeding the flap limit speed.

WARNING

- Excessive angles-of-attack may result from retracting the flaps too rapidly in this flight regime.
- Maneuvering flight at angles-of-attack greater than 10 degrees should be avoided.

For typical takeoff, see figure 2-3. Refer to the Performance Appendix for takeoff data.

Takeoff (Typical)

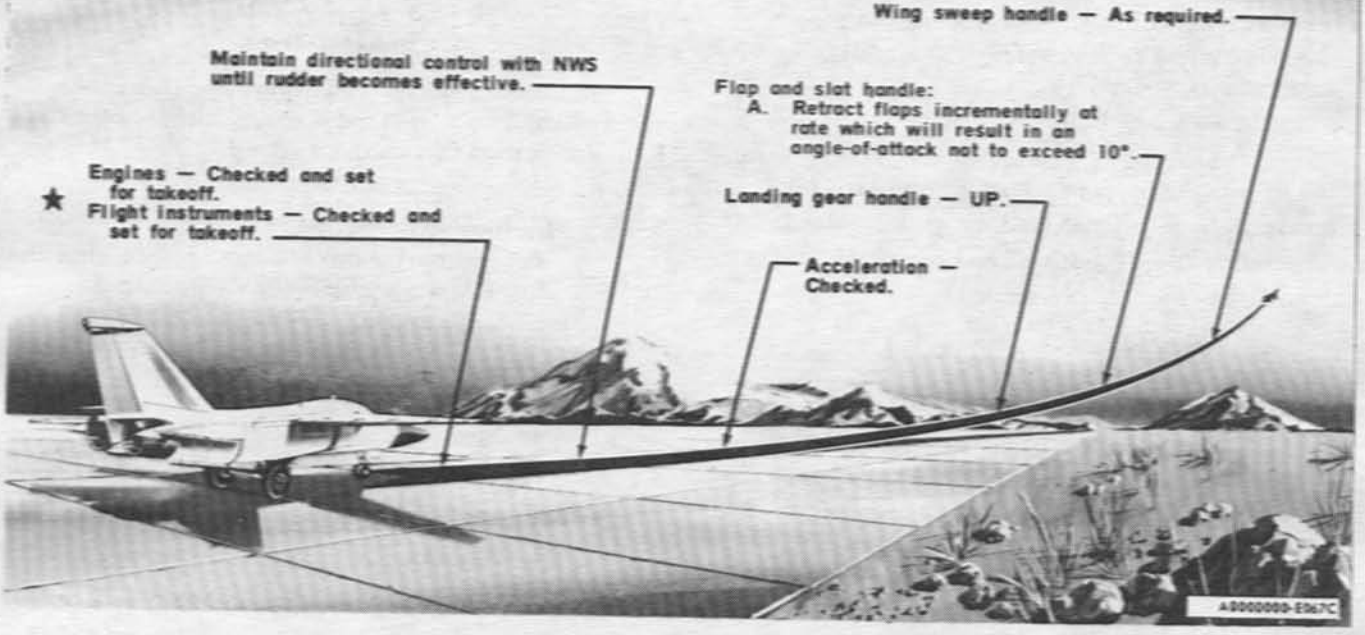


Figure 2-3.

CAUTION

Failure to arrest rapid rotation rates generated at nose wheel lift off can result in aircraft tail bumper and/or engine nozzle leaves contacting the runway.

CROSSWIND TAKEOFF.

Under crosswind conditions, the aircraft tends to weather-vane into the wind. The weather-vaning tendency can be easily controlled with nose wheel steering until the rudder becomes effective. As forward speed increases, the weather-vaning tendency decreases. At speeds above approximately 50 knots rudder effectiveness will normally be sufficient to maintain directional control. Use of roll control will aid directional control and keep the wings level. Care should be exercised, however, to prevent inducing an excessive wing-low attitude at lift-off.

Note

Application of roll control may delay rotation due to a slight reduction in available pitch control.

After the aircraft leaves the ground, it should be crabbed into the wind, wings level, to maintain runway alignment. Refer to "Crosswind Takeoff and Landing Limits," Section V.

AFTER TAKEOFF. (BOTH)

1. Landing gear handle—UP.
   When the aircraft is definitely airborne, retract the landing gear. Check that the landing gear position indicator lamps and the warning lamp in the landing gear handle go out. The landing gear and landing gear doors should be up and locked before reaching 295 KIAS.

WARNING

If it is necessary to depress the landing gear handle lock release button to move the handle to the UP position, the crew member should suspect a malfunction of the landing gear ground safety switch. Refer to Section III.

Note

The fuel tank pressurization caution lamp may light when the landing gear handle is moved to the UP position and remain lighted until the tanks are pressurized.

2. Flap/slat handle:
   a. Flaps—Retract flaps incrementally at a rate which will result in an angle-of-attack not to exceed 10 degrees.

WARNING

- Excessive angle-of-attack may result from retracting flaps too rapidly.
- If aircraft starts to roll off while retracting the flaps immediately return the flap/slat handle to original position and make no further attempts to operate the flaps. Sufficient lateral control may not be available to counter an asymmetrical flight condition. Refer to the appropriate procedure under "Landing With Flap And Slat Malfunctions," Section III.
- If any malfunction is indicated or suspected during flap retraction it is recommended, flight conditions permitting, that the flaps not be further actuated until over an approved drop or unpopulated area. A landing utilizing normal landing procedures can be accomplished if the slats/flaps return to original position. If they do not, refer to appropriate procedure under "Landing With Flap And Slat Malfunctions," Section III.

   b. Slats—UP, and verified.
      Retract slats after verifying flaps are full up. Check that slat/aux flap indicator displays UP.

Note

Maintain 1 "g" until slats/flaps are fully retracted.

3. Wing sweep handle—As required.

CAUTION

If the slat/aux flap indicator displays crosshatch, do not sweep the wings without other verification that the flaps are up.

4. Throttles—As required.
   For military power climb, reduce throttles to MIL when climb speed is attained.
5. Engine instruments—Checked.
6. Fuel quantity indicators—Checked.
   Check the fuel quantity indicators for normal fuel usage.
7. Oxygen and cabin altitude—Checked.
8. Altimeters—As required.

CLIMB.

The recommended climb speed, as shown in the Performance Appendix should be followed.

CRUISE.

After transfer of all external, weapon bay, and wing tank fuel, check fuselage fuel quantity indicators for normal distribution and usage. Forward and aft together should equal totalizer (±1000) pounds.

WARNING

Failure of either fwd or aft indicator pointers will cause improper fwd and aft tank fuel distribution if engine feed is in AUTO. Do not remain in AUTO. Fuel distribution must be controlled manually to maintain cg within safe limits. On aircraft prior to T.O. 1F-111-673, the fuel distribution caution lamp is inoperative in manual modes. Refer to "Abnormal Fuel Distribution," Section III.

Refer to the Performance Appendix for cruise operating data, and to Section I for complete fuel system operation.

DESCENT FOR OTHER THAN LANDINGS. (COMMAND RESPONSE)

1. Wing sweep handle—As required.
   Check wing position indicator to assure wings moved to position selected.
2. Radar altimeter—Set minimums.
3. Magnetic variation—Set.
4. Altimeters—Set.

DESCENT/BEFORE LANDING. (COMMAND RESPONSE)

Note

The use of slats during the descent/before landing phase is optional.

1. Wing sweep handle and lockout control—Set 26° or forward, ON.
   Check wing position indicator to assure wings moved to position selected.
2. Radar altimeter—Set minimums.
3. Altimeters—Set.
4. Ground roll spoiler switch—BRAKE.
5. Anti-skid switch—ON.
6. Landing and taxi lights switch—LANDING.
7. Fuel quantity and fuel distribution—Check.
   Check fuselage fuel indicator totals against totalizer reading (±1000) pounds. If engine feed is in AUTO, verify normal distribution. If aft tank is empty (pump lamps lighted) switch to FWD.

WARNING

If any fuselage fuel quantity gage abnormality is noted, do not remain in AUTO. Place the engine feed selector knob to AFT and refer to "Landing With Abnormal Fuel Distribution," Section III.

8. Elevator check—Complete (if an abnormal cg is indicated or suspected).
   The elevator check may be performed at mach 0.70 or below and below 20,000 feet MSL in level flight with 26 degrees wing sweep, slats up, and speed brake retracted. If the elevator trailing edge deflection is between 2 degrees trailing edge up and 1 degree trailing edge down, 26 degrees sweep may be used for landing. If the elevator trailing edge deflection is greater than 2 degrees trailing edge up, the wing should be swept forward until an elevator position of 2 degrees trailing edge up or 16 degrees is reached. This wing sweep may be used for landing. For trailing edge deflections greater than 1 degree trailing edge down, refer to "Landing With Abnormal Fuel Distribution," Section III.
9. CMDS—Checked, SAFE and OFF.
10. Magnetic variation—Set.
11. Landing brief—Completed.
12. Autopilot/damper switches—DAMPER.

LANDING PATTERN. (BOTH)

1. Wing sweep handle—Set for landing.
   Check wing position indicator to assure wings move to position selected.

Note

- Wing sweep selected for landing will depend on gross weight and runway conditions as long as elevator position limits in the landing configuration are maintained.
- The wings must be at 26 degrees or less to allow flap/slat extension.

2. Landing gear—DN and check.
   Extend the landing gear after airspeed is below 295 KIAS. Check that warning light in landing gear handle is out and landing gear position indicator lights are lighted, and hydraulic pressure is normal.

WARNING

- After placing the landing gear handle to on, selection of slats/flaps during decelerating flight should not be delayed and extension of slats should be accomplished while landing gear is in the extend cycle. The command augmentation feature masks stall warning characteristics and rapid drag rise as airspeed decreases without flaps and slats extended. This may result in a rapid increase in angle-of-attack which the pilot may not be able to arrest before critical angle-of-attack limits are exceeded.
- Under landing conditions wherein airspeed may be above the gear warning horn setting 160 (±12) KIAS exercise caution to insure the landing gear is down and locked.

Note

The pitch and roll gain changer caution lamps will light when the gear is extended and will remain lighted until the slats are extended to approximately 70 percent.

3. Slats—Extend. (240 KIAS minimum)
   Extend slats while gear is in the extend cycle by positioning the flap/slat handle to the slat gate and make positive verification of slat position using the wing sweep flap/slat position indicator, visual check of slats and/or observation of the gain changer caution lamps. Since the gain changer caution lamps will remain lighted until the slats have extended to aproximately 70 percent, this will provide an indication of slat position. When the gain changer caution lamps go out, extend the flaps. If the gain changer lamps remain lighted and 70 degree slat extension cannot be verified by other means, do not extend flaps. (Refer to Section III)

WARNING

- For normal operation, slats should be extended by a minimum airspeed of 240 KIAS. Do not roll or execute abrupt maneuvers with slats only extended.
- Do not extend flaps by normal or emergency method until approximately 70 percent slat extension has been verified. To do so could result in the flaps being locked at approximately 15 degrees with zero (or partial) slat extension. Flight in this configuration could result in stall or uncontrolled roll off. If the system locks, refer to "No (Or Partial) Slats and Partial Flaps Landing," Section III.

Note

- Airloads may prevent full slat extension of airspeed approaching the slat limit speed; however as airspeed is reduced resultant lowering of airloads will allow full slat extension.
- Maintain 1 "g" wings level until slats/flaps are extended to the desired position.

4. Flaps—Down and verified.
   - Flaps—Down to 15 degrees.
   - Flaps—Full down.

WARNING

In the event of an asymmetrical flap condition with flaps extended beyond the 15 degree position, lateral control available to effect recovery may be marginal. (Refer to "Asymmetric Flap In Flight," Section III)

- Center-of-gravity—Check elevator position.
  At 10 degrees angle-of-attack check elevator position. If the elevator position is between 10 degrees trailing edge up (forward limit) and 1.5 degrees trailing edge up (aft limit) at 26 degrees wing sweep or between 10 degrees trailing edge up and 4 degrees trailing edge up at 16 degrees wing sweep, the aircraft is within the center-of-gravity limits. For wing sweeps between 26 and 16 use linear interpolation in determining the elevator position for the aft limit. If the elevator position is not in the above envelope, sweep the wing until it is. As the wing is swept forward from 26 degrees, the elevator required to trim will move in the down direction.

Note

The cg elevator position range will provide safe operation for all landing wing sweeps and store loadings. For the aft limit for landing with a specific configuration refer to Section V.

LANDING.

See figure 2-4 for a typical landing pattern. Brakes should be used as required compatible with runway available. Engage nose wheel steering as required for landing roll. For Landing Data, refer to Performance Appendix.

NORMAL LANDING.

Normal landing should be accomplished with the wings at 16 to 26 degrees sweep, full flaps, and the pattern flown as illustrated in figure 2-4. Enter the pattern as local policies dictate. On the downwind leg, reduce thrust to obtain 250 KIAS or below. Extend the landing gear and slats. Do not decelerate below 240 KIAS prior to full extension of slats. Flaps should be extended by a two step procedure, first, extend flaps to 15 degrees. Continue to slow aircraft to approxi-

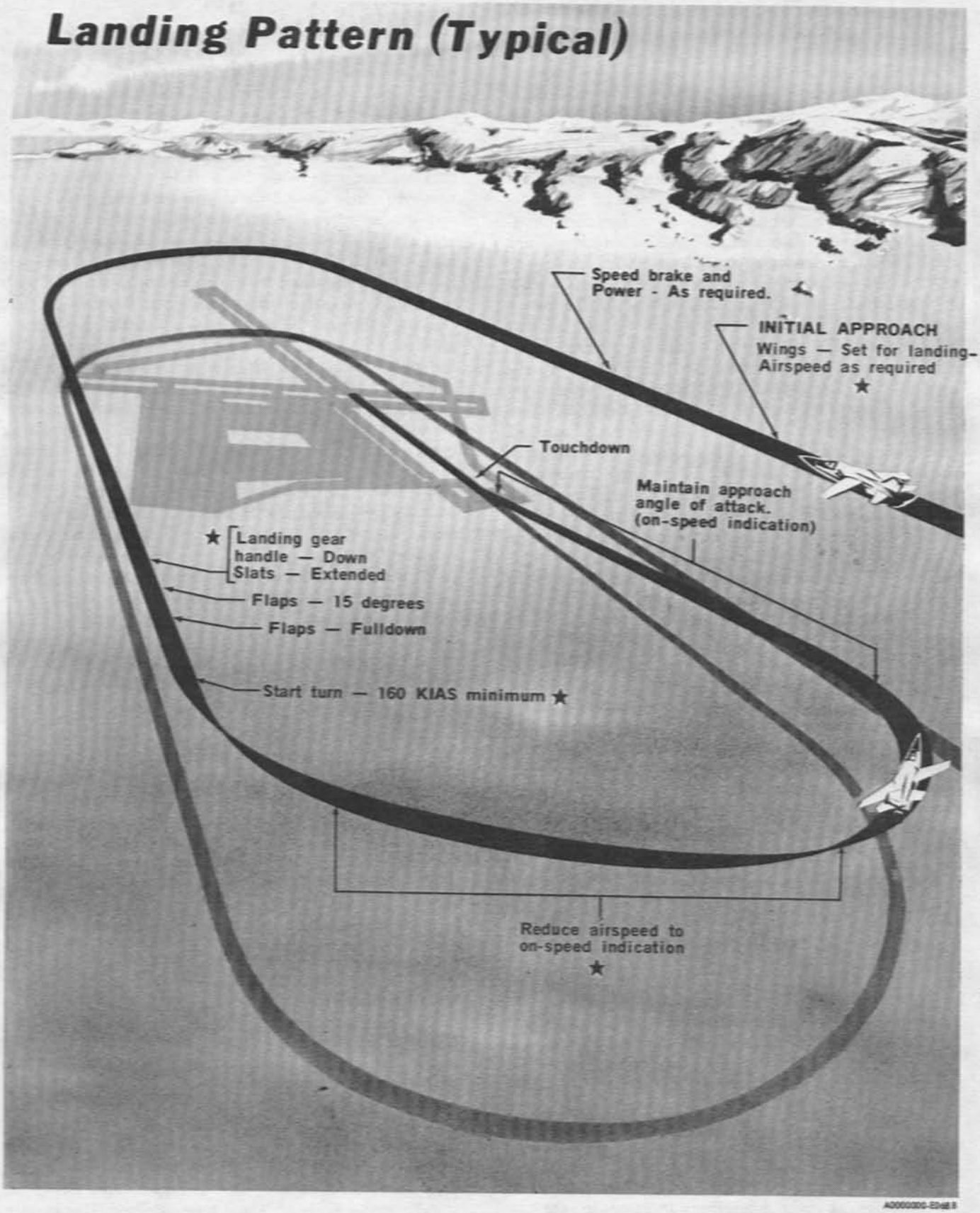

Landing Pattern (Typical)
Speed brake and
Power - As required.
INITIAL APPROACH
Wings — Set for landing-
Airspeed as required
★
Touchdown
Maintain approach
angle of attack.
(on-speed indication)
★ Landing gear
handle — Down
Slats — Extended
Flaps — 15 degrees
Flaps — Fulldown
Start turn — 160 KIAS minimum ★
Reduce airspeed to
on-speed indication
★

Figure 2-4.

mately 220 KIAS and lower flaps to full down. Trim changes associated with gear and flap extension are small, but a noticeable decrease in angle-of-attack and pitch attitude will be evidenced as slats and flaps are extended. Start turn on to base leg at a minimum of 160 KIAS. On base leg, airspeed should be reduced to obtain an "on-speed" indication upon rolling out on final. On final, establish and maintain "on-speed" indication by adjusting power (to maintain glide slope/rate of descent) and pitch (to maintain angle-of-attack). A normal glide slope will produce a rate of descent of approximately 600-700 feet per minute. The "on-speed" indication represents optimum angle-of-attack and airspeed for all pattern work including maneuvering, and will automatically adjust airspeed for the gross weight of the aircraft. In addition, the "on-speed" indication provides sufficient airspeed cushion for gusts that may be encountered in the pattern. After the aircraft enters ground effect (approximately 30 to 50 feet above the ground), the aircraft will tend to rotate in the nose-down direction. Aircraft pitch attitude should be maintained to touchdown. Power should be reduced to IDLE immediately upon touchdown and nose wheel lowered to runway as soon as possible. Upon touchdown, landing roll brakes will extend, and the aircraft nose will tend to fall through due to the center of rotation being shifted from the aircraft center-of-gravity to the main gear. Directional control can be maintained with rudder and differential braking down to the lower speed regions. Differential braking and nose wheel steering, if required, may be used in the lower speed regions (rudder loses effectiveness below approximately 50 knots). Nose wheel steering should be engaged at minimum speeds, when necessary for directional control, with the rudder pedals at or near neutral. Brakes can be used as required throughout the landing roll. Ease the control stick aft when brakes are applied to utilize aerodynamic braking effects of the horizontal tail. The stick can be held full aft without unsticking the nose wheel at speeds below approximately 90 KIAS.

WARNING

Under no circumstances, during the landing phase, should the 14 degree angle-of-attack limit or stall warning activation be exceeded. Possible inadvertent stall and post-stall gyrations can result from exceeding these limits.

CAUTION

Rapid or abrupt lateral or longitudinal stick motions can cause momentary increases in rate of sink and therefore should be avoided.

SHORT FIELD LANDING.

A short field landing is accomplished in the same manner as a normal landing except that particular attention must be given to precise airspeed, angle of attack, and glide slope control. Touchdown should be as close to the end of the runway as possible with no landing flare. Observe sink rate limits. Refer to Section V. Reduce the power to IDLE at touchdown if this has not previously been done and allow the airplane to settle on the main gear and the ground roll spoilers to extend. After the spoilers have extended and the nose wheel is firmly on the runway, apply maximum antiskid braking. Maximum braking performance is obtained in the three-point attitude with maximum weight on the main gear. Because of this, the stick should be eased aft when the brakes are applied, but caution should be exercised to insure that the nose wheel does not rise from the runway. The stick can be brought to the full aft position without unsticking the nose wheel at speeds below approximately 90 KIAS. Be prepared to lower the arresting hook and engage the runway barrier if the airplane cannot be stopped prior to reaching the end of the runway. Maximum braking should be released, if practical, at approximately 25 knots to prevent the brakes from fusing and immobilizing the airplane on the runway. At light gross weights the antiskid system cycling will be quite extreme and continue throughout the ground roll until just before the airplane is stopped. At heavier gross weights, little antiskid cycling will be noted. If safety or operational considerations dictate that the ground roll must be the absolute minimum possible, touchdown can be made with full antiskid braking applied.

HEAVY GROSS WEIGHT LANDING.

A heavy gross weight landing will be accomplished with a 16 degree wing sweep (if cg permits) in the same manner as a normal landing except that, maintaining an on-speed indication will result in higher approach and touchdown speeds. These higher speeds, due to heavier weights, result in increased braking requirements and stopping distances. Refer to Appendix I for landing data.

HYDROPLANING.

Dynamic hydroplaning is a condition where the tires of the airplane are separated from the runway surface by a fluid. Under conditions of total dynamic hydroplaning, the hydrodynamic pressures between the tires and runway lift the tires off the runway to the extent that wheel rotation slows or actually stops. The major factors in determining when an airplane will hydroplane are groundspeed, tire pressure, and depth of water on the surface. To a lesser degree, the surface texture, type of tire, and tire tread depth influence the

total hydroplaning speed. Total dynamic hydroplaning in this airplane with recommended tire pressure and .1 inch or more of water or slush on the runway can be expected at approximately 115 knots groundspeed (main landing gear) and 150 knots groundspeed (nose wheel) considering a takeoff gross weight of 86,000 to 90,000 pounds. These speeds will change as tire pressure is varied for takeoff gross weight. Partial dynamic hydroplaning occurs to varying degrees below these speeds. When an airplane is subjected to hydroplaning to any degree, directional control becomes difficult. Under total dynamic hydroplaning conditions, nose wheel steering is ineffective and wheel braking is nonexistent. In addition to dynamic, two other types of hydroplaning can occur. Viscous hydroplaning can occur on a damp runway and at speeds less than those associated with dynamic hydroplaning, and is caused by a thin film of water mixed with rubber deposits and/or dust. Reverted rubber hydroplaning is caused by skid which boils the water on the runway, causing the rubber to revert to its natural latex state and seals the tire grooves, delaying water dispersal. Reverted rubber hydroplaning can occur at very low airspeeds. When possible hydroplaning conditions exist, pilots should be aware of the following:

1. Smooth tires tend to hydroplane with as little as .08 inch of water. New tires tend to release hydro-dynamic pressures and will require in excess of .2 inches of water depth to hydroplane.
2. Takeoffs with crosswinds on water covered runways should be made with caution. An aborted takeoff on a wet runway initiated at or near hydroplaning speed will require considerably more runway than a dry runway abort and directional control of the airplane will be critical until the speed has decreased below hydroplaning velocity.
3. In the absence of accurately measured runway water depths, pilots may use the following information to determine the possibility of hydroplaning when landing must be accomplished on a wet runway that does not have a porous surface or is not grooved:
   a. Rain reported as LIGHT—Dynamic hydroplaning unlikely, viscous and reverted rubber hydroplaning are possible.
   b. Rain reported as MODERATE—All types of hydroplaning are possible. Smooth tires will likely hydroplane; however, new tires are less likely to hydroplane.
   c. Rain reported as HEAVY—Hydroplaning will occur.

LANDING ON SLIPPERY RUNWAYS.

WARNING

If hydroplaning conditions exist the landing roll will be increased an indeterminate amount, therefore, be prepared for a departure end barrier engagement.

The technique for a slippery runway landing is essentially the same as that for a short field landing. During the high speed portion of the landing roll, particularly under wet or icy conditions, little braking capability will be available. This is because of the low coefficient of friction available due to hydroplaning or a very low RCR. Maximum aerodynamic braking should be used throughout the landing roll to aid in decelerating the aircraft. To avoid inhibiting wheel spin-up, and to improve wet runway wheel cornering capability, insure that the aircraft is firmly on the runway and positively under control prior to applying brakes. On wet runways during the high speed portion of the roll, little deceleration will be felt due to rapid anti-skid cycling. As speed decreases, braking potential on a wet runway will increase and brakes should be applied as required to stop the aircraft. On an icy runway, the coefficient of friction will remain fairly constant throughout the landing roll and brakes should be applied as required. Aerodynamic control, differential braking and nose wheel steering may be used to maintain directional control. Nose wheel steering should not be required until aerodynamic control becomes ineffective. If planned stopping distance indicates that a stop on the runway is doubtful, divert or make either an approach end or departure end barrier engagement, depending on the severity of the situation. Refer to Appendix I for ground roll distance for various runway conditions.

CROSSWIND LANDING (DRY RUNWAY).

When crosswind conditions are encountered, a crab technique on final approach should be used to compensate for drift. Under visual conditions a wing-low drift correction technique may be used, however, airspeed and glidepath control becomes more difficult. Additionally, when the aircraft sideslips to the right, airflow to the angle of attack sensor begins to be blanked by the aircraft nose at a sideslip angle of approximately 10 degrees. As the sideslip angle is increased beyond this point, the angle-of-attack sensor indicates increasingly lower values of angle-of-attack. Therefore, it is recommended that steady-state rudder inputs be kept below seven degrees as inputs of a larger magnitude may result in erroneous angle-of-attack indications.

Sideslip to the left will not affect the angle-of-attack sensor; therefore, the aircraft may sideslip to the left to the limits presented in Section V. During the transition to touchdown (approximately 75 feet above the ground), the drift correction technique should shift gradually from a crab to a wing low crabbed correction at touchdown. The pilot should attempt to touch down with no drift and the longitudinal axis of the aircraft aligned with the runway, which will minimize sideloads on the landing gear. However, if the crosswind components is excessive, it will be necessary to land in a combination wing-low crabbed attitude, not to exceed 10 degrees yaw or crab angle at touchdown.

CAUTION

External tanks at stations 2 or 7 will contact the ground at a bank angle of 15 degrees.

During touchdown from a wing-low crabbed approach, the pilot may experience the sensation of bouncing from gear to gear which may be aggravated by use of roll control in attempting to keep the wings level. The probability of this occurring will be reduced if a firm touchdown at the recommended angle of attack is accomplished. If this condition is encountered, minimize use of roll control until the aircraft has settled through the struts and is firmly on the ground. After touchdown, the pilot should use rudder, roll control and differential braking as required to maintain directional control. Roll control effectiveness may be increased significantly by "cracking" a throttle, thereby retracting the spoiler brakes and allowing the spoilers to function as an aid to roll control. When the desired directional control change is achieved, return the throttle to idle to extend the spoiler brakes. If nosewheel steering is required, it should be initiated with the rudder pedals at or near neutral, since the nosewheel will rapidly assume a position relative to the rudder pedal position at engagement. Unless required for directional control, nosewheel steering should not be engaged until the aircraft has slowed to taxi speed and just prior to turning off the runway. When landing with slats/flaps up, refer to "Crosswind Takeoff And Landing Limits," Section V, for recommended touchdown technique and limits. When landing with augmentation off, refer to "Dampers Off Landing," Section III.

CROSSWIND LANDING (SLIPPERY RUNWAY).

The problem of maintaining directional control on a slippery runway becomes more difficult as the effective crosswind is increased. Consequently, aircraft flight path alignment with the runway must be established during the approach to prevent drift at touchdown. Restricted visibility, poor ground references, and crab angle will further complicate the task of establishing alignment during the approach. Pilots should be aware that excessive maneuvering during the final phase of the approach may induce misalignment and/or drift and may make it impossible for the pilot to determine actual aircraft track.

WARNING

Proper runway alignment for approaches and landings under low RCR conditions is extremely critical. Avoid excessive maneuvering on final approach under these conditions. Aircraft drift or flight path misalignment at touchdown increases susceptibility to skidding or hydroplaning, which may cause loss of directional control during landing roll. If aircraft drift is not corrected prior to touchdown, execute a missed approach.

Plan the landing pattern to be established on final approach using a crab technique to correct for drift. This will insure that the aircraft is tracking straight down the center line of the runway. Establish a normal rate of descent and plan to touch-down approximately 500 feet down the runway or at the glide slope/runway interception point (if applicable). Make a firm touchdown with no flare (observe sink rate limitations, Section V) while maintaining the drift correction. Touching down in a crab will help insure that the runway center line track is maintained. Due to visibility restrictions that may occur with a crabbed approach, a combination crabbed/wing-low technique may be necessary during the transition to touchdown. Immediately after touchdown, retard throttles to idle and lower the nose to the runway. Aerodynamic (rudder and roll) control, differential braking, and nose-wheel steering may be used to maintain directional control; however, nose-wheel steering should not be required until aerodynamic control becomes ineffective. Roll control effectiveness will be increased significantly by "cracking" a throttle, thereby retracting the spoiler brakes and allowing the spoilers to function as an aid to roll control. When the desired directional control change is achieved, return the throttle to idle to extend the spoiler brakes. If nose-wheel steering is engaged, inputs should be kept small as steering effectiveness diminishes rapidly with nose-wheel deflections of more than 10°.

Note

- If directional control cannot be established or maintained, immediately advance power as required to accomplish a go-around.
- After directional control is well established, use the technique described under "Landing on Slippery Runways," this section, to stop the airplane.

LANDING WITH PARTIAL FLAPS.

A partial flap landing is accomplished in the same manner as a normal landing except that, maintaining an "on-speed" indication will result in higher approach and touchdown speeds (approximately 1.7 knots increase in airspeed for each degree of flap less than full flaps with 16 to 26 degree wing sweep). Due to the higher approach and touchdown speeds, braking requirements as well as stopping distances will be significantly increased.

LANDING WITH SLATS EXTENDED AND FLAPS RETRACTED OR WITH SLATS AND FLAPS RETRACTED.

Approaches with wings and flaps in other than normal landing configuration will necessitate a long shallow, straight-in approach. If it is necessary to land the aircraft in this configuration, refer to "No Flap Landing," Section III.

TOUCH AND GO LANDING.

Touch and go landings should be accomplished using the same technique as presented in the "Normal Landing and Takeoff" procedures this section. After touchdown power should be reduced to IDLE to allow the aircraft to decelerate and the nose wheel lowered to the runway. Directional control should be maintained with the rudder pedals. After the nose wheel has been lowered to the runway, smoothly advance the throttles to MIL or AB power as required. Check engine instruments for normal indications and caution lamps for malfunction warning. Lift nose wheel off runway 10 knots below previous approach speed.

SIMULATED SINGLE ENGINE LANDING.

Simulated single engine landing should be flown with one engine at idle rpm, following the "Single Engine Go-Around," procedure, Section III.

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 and continue the approach because a touchdown may be necessary. As the aircraft accelerates, rotate the nose to a climbing attitude and when the altimeter and vertical velocity show a definite rate-of-climb proceed with the normal after takeoff checklist. Fly clear of the runway as soon as practicable. (See figure 2-5.) In the accomplishment of a go-around from the approach condition at light gross weight, application of MAX AB on both engines will result in a significant nose-up pitching moment. The forward stick movement to counter the induced nose-up moment, plus the normal forward stick required to maintain level flight as the aircraft accelerates, results in a large forward stick deflection. Forward stick trim authority may not be sufficient to correct this nose-up tendency, and forward control stick application may be required. However, adequate longitudinal control is available to maintain level flight.

BEFORE CLEARING RUNWAY. (COMMAND RESPONSE)

1. Attack radar function selector knob—ON.

WARNING

The attack radar must not be left in the XMIT position after landing due to the possible danger to personnel in front of the aircraft.

2. TFR channels—STBY.
3. Nose wheel steering—As required.
4. Anti-skid switch—OFF.
   Place anti-skid switch to OFF when aircraft speed is reduced to approximately 20 knots or below.

AFTER LANDING. (COMMAND RESPONSE)

CAUTION

To prevent damage to the canopy, do not open canopy with cabin pressurized. Prior to opening canopy, check cabin pressure altimeter agrees with field elevation. If cabin is pressurized, place the pressurization selector switch to DUMP prior to opening canopy. (Equipment cooling is not affected with this switch in DUMP.)

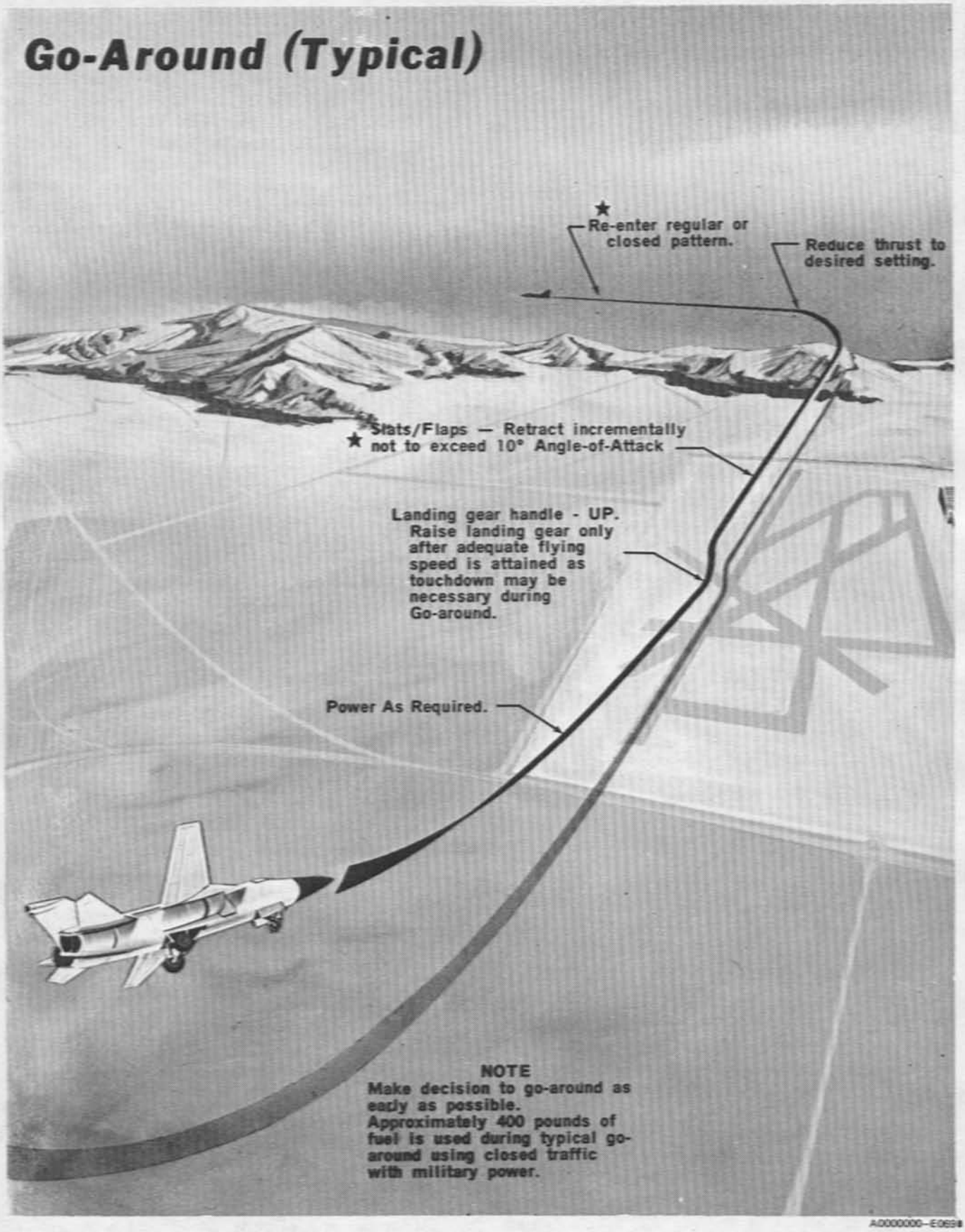

Go-Around (Typical)
★ Re-enter regular or closed pattern.
Reduce thrust to desired setting.
★ Slats/Flaps — Retract incrementally not to exceed 10° Angle-of-Attack
Landing gear handle - UP. Raise landing gear only after adequate flying speed is attained as touchdown may be necessary during Go-around.
Power As Required.
NOTE
Make decision to go-around as early as possible.
Approximately 400 pounds of fuel is used during typical go-around using closed traffic with military power.
A0000000-E0691

Figure 2-5.

1. Nose wheel steering—Engaged.

Note

Nose wheel steering should be engaged prior to turning off runway.

2. Anti-skid switch — OFF. (Approximately 20 KIAS)
3. Radar function knob—ON.

WARNING

The attack radar must not be left in the XMIT position after landing due to the possible danger to personnel in front of the aircraft.

4. Fuselage and landing lights—As required.
   - Landing lights—OFF or TAXI. (If necessary)
   - Anti-collision light—OFF.
   - Fuselage lights—FLASH.
5. Radar altimeter—OFF.
6. Ground roll spoiler switch—OFF.
7. Flap/slat handle—As required. (Normally extended)

   If slats are retracted, place rudder authority switch to FULL to insure full nose wheel steering.
8. Wing sweep—Set. (Normally 16 degrees)
9. Pitot/probe heater switch—OFF/SEC.
10. IFF mode 4 control knob—HOLD or ZERO.
11. IFF master control knob—OFF.
12. Ejection handle and center beam safety pins—Inserted.

Note

- The ejection handle safety pins provided must be inserted from center console outboard to preclude interference of the pins with seat adjustment.
- Destruct panel lockout pin installed if previously removed.

ENGINE SHUTDOWN. (BOTH)

CAUTION

To prevent damage to the brakes from overheating, do not pull the auxiliary brake handle.

1. Wheels—Chocked. (GO)
2. Wing sweep handle—As required.
3. Flap/slat handle—As required.
4. Weapon bay door(s)—As required. (GO)
   - Ground crew check weapon bay doors—Checked, clear.
   - Weapon bay door(s) switch—OPEN. (As required)
   - Ground crew report weapon bay door(s)—OPEN.
5. Bomb nav computer—Set. (If required)
   - Present position latitude and longitude—Corrected.
   - Rapid alignment. (If applicable)
6. Bomb nav mode selector knob—OFF.
7. Pitch, roll and yaw damper switches—OFF.
8. Computer power switches(3)—OFF.
9. Applicable throttle—OFF.

   Place the throttle of first engine started to OFF.

Note

If shutdown is after operation above 80 percent, reduce power to IDLE for approximately five minutes.

10. Nacelle vent ejector system—Check.

    Prior to engine shut down the ground crew will observe that the nacelle vent and fire access doors are in an open position indicating that the nacelle vent system is functioning. A positive movement of the doors from open to close should occur during engine shut down. If the doors are closed prior to shut down, it is indicative of a broken nacelle vent system duct or a faulty nacelle vent system pressure regulating and shut off valve.
11. Hydraulic pressure—Check.

    Check for 2950 to 3250 psi indication.
12. Remaining throttle—OFF.

CAUTION

To prevent possible engine damage due to overtemperature, do not attempt to restart the engine until TIT is below 100 degrees C.

Note

Do not move the control stick after shutting down the last engine. To do so will invalidate the following horizontal tail droop check.

13. Horizontal tail droop check—Completed.
    Do not move the control stick. A minimum trailing edge down position of 12 inches must occur within 60 seconds of zero hydraulic pressure. Zero hydraulic pressure is considered to be the point at which the audible whine associated with emergency generator operation begins to decay.
14. Emergency generator—Checked.
    - Emergency generator OPR light—Lighted, within one second.
    - Emergency generator lamp—Out.
      Light remains on until utility hydraulic pressure drops. The emergency generator indicator lamp will light as the last engine driven generator disconnects from the ac buses. The lamp will go out when hydraulic pressure driving the emergency generator is depleted.
15. Battery switch—OFF.
16. All switches and controls—OFF, NORMAL or SAFE.

WARNING

If oxygen lever is left on and regulator is set to EMER, liquid oxygen may flow through the regulator creating a potentially hazardous situation.

17. Camera magazine—Removed (if required).

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 aircraft 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 aircraft electrical systems. The following check list supplements the normal operating procedures and includes items that would normally be accomplished by the ground crew.

AFTER SHUTDOWN.

1. Ground safety pins—Installed.
2. 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.

3. Hydraulic reservoirs—Checked.
   Check the utility and primary hydraulic reservoirs for specified accumulator preload and fluid level in accordance with placard.
4. Refueling—Accomplish. (As required)
   If high ambient temperatures (above 100 degrees) are anticipated, a full fuel load should not be taken on. Excessive fuel venting will occur which may create a hazardous condition.
   a. Single point refueling:

Note

- External electrical power may be connected during refueling if desired for monitoring instruments; otherwise, external power is not necessary.
- The lower engine access doors should be closed prior to ground refueling. The added fuel weight may preclude closing the doors after refueling.

(1) Aircraft and refueling equipment—Grounded.
    Insure that the aircraft and all refueling equipment are statically grounded.
(2) Nose gear chocks and work stands—Removed.
    Remove nose gear chock and 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) Fueling hose ground cable—Connected.
    Connect the grounding cable from the fueling hose to the aircraft.
(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 SEC. (As applicable)

Within a few seconds after fuel flow is indicated, position all precheck selector valves to PRI or SEC as applicable. The fuel flow should drop to less than 20 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 aircraft to tip up.

Note

If fuel flow drops to 20 gpm or less, proceed to step 9. If fuel flow does not drop, determine which refuel valve has malfunctioned by reducing flow from servicing unit to a minimum and operate valves individually 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.

(9) 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.

(10) Precheck selector valves—REFUEL.

Continue refueling operation.

(11) Position lights/stores refuel battery power switch—STORES REFUEL. (If external tanks are installed)

(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) Aircraft and refueling equipment—Grounded.

Insure that the aircraft 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 selector switch —AUTO.

If tanks are pressurized, place the tank pressurization selector 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 engine feed selector knobs to the FWD position and allow fuel pumps to operate for approximately 2 minutes to assure a full reservoir tank.

Note

If forward tank initially had less than 4000 pounds, perform step 5 after the forward tank has been filled above 4000 pounds, and then continue filling.

(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.

(7) Gravity refuel the remaining tanks in the following order:
   (a) Bay A-1
   (b) Bay A-2
   (c) Wing tanks (and external tanks if required)

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.
- There are no provisions for gravity refueling the weapon bay tanks.

(8) To refuel external tanks, fill internal wing tanks to approximately half full and proceed as follows:
   (a) External electrical power—On.
   (b) Fuel transfer knob—WING.
   (c) Position lights/stores refuel battery power switch—STORES REFUEL.
   Continue refueling the wings to maintain internal wing tanks approximately half full.
   (d) Fuel transfer knob—OFF.
   (e) Position lights/stores refuel battery power switch—NORM.
   Continue refueling internal wing tanks.

(9) Fuel filler caps—Secure.

POSTFLIGHT.

1. Exterior inspection—Complete:

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.
- Tires for condition.
- Reservoirs and accumulators for proper servicing.
- Ground area around aircraft for cleanliness.

Aircraft 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. The following systems should be checked and serviced as required. Upon completion, follow normal procedures this section. Complete required Form 781 entries prior to takeoff.

1. Preflight and Form 781—Completed.
2. Liquid oxygen—Checked.
3. Pneumatic pressure—Checked.
   The accumulators or reservoir pneumatic pressures should be checked for required pressure range specified for the ambient temperature.
4. 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.

a. Refill very slowly until oil level reaches the bottom of the green band. Shut off oil supply to avoid overfilling, and allow oil to equalize. As much as 5 minutes may be required.
b. Repeat preceding step until oil level is stabilized in the green band.

5. 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.
   a. Check the hydraulic reservoir pneumatic pressurization system for proper service.
   b. Using the aircraft hydraulic hand pump, 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 hydraulic fluid bleed valve (lower aft end of reservoir) sufficiently to bleed trapped air from the reservoir chamber.
   e. Check the reservoir quantity indicator for proper fluid level.
   f. Repeat step d and e until the reservoir is fully serviced and free of air.
6. Horizontal stabilizer accumulator—Check.
   Check the utility and primary hydraulic accumulators preload in accordance with instruction placard.
7. Landing gear shock struts—Checked.
   Check nose landing gear shock strut and main landing gear shock struts inflated in accordance with strut instruction placard.
8. Tires—Checked. (See figure 2-6)
9. Angle-of-attack and side slip probes—Checked for slot cleanliness and freedom of movement.

Tire Inflation Pressures

DATE: 24 APRIL 1970

Aircraft Gross Weight	*Inflation Pressure MLG Tire Size 47 x 18-18*	*Inflation Pressure NLG Tire Size 22 x 6.6-10*
60,000 & below	115-125 psi	175-185 psi
60,000-66,000	125-135 psi	195-205 psi
66,000-72,000	135-145 psi	215-225 psi
72,000-78,000	145-155 psi	235-245 psi
78,000-84,000	155-165 psi	255-265 psi
84,000-91,500	165-175 psi	275-285 psi
Either nitrogen or air may be used.		

Figure 2-6.

AIRCRAFT EMERGENCY MOVEMENT CHECKLIST.

CAUTION

Pilot will ascertain from ground crew or Form 781 that the aircraft status will allow ground movement.

STARTING ENGINE.

1. Connect external power and air or install cartridge—Accomplished.
2. Ground ignition cutoff switch—NORM.
3. Battery and/or external power switches — ON. (As applicable)
   Engine start may be accomplished using either battery or external power, or both.
4. UHF radios—ON and set.
5. Engine feed selector knob—AUTO.
6. Position lights—BRT and FLASH.
7. Auxiliary brake handle—Pulled.
8. 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.
9. Throttle—START, then IDLE:
   a. On a cartridge start, advance the applicable throttle to IDLE immediately.
   b. On a pneumatic start, advance the applicable throttle to IDLE after engine rpm reaches 17 percent.
10. Engine instruments, caution lamps, and hydraulics—Checked.
11. Engine ground start switch—OFF.
12. Generator switch—START (pause), release to RUN.

Note

If the generator caution lamp remains lighted, place the generator switch to OFF/RESET, hold to START (pause), then release to RUN.

13. Air refueling switch—CLOSE.
14. External power switch—OFF, power unit removed.
15. Rudder authority switch—FULL.

TAXIING.

1. Auxiliary brake handle—In.
2. Nose wheel steering—Engaged.

   Check that the nose wheel steering indicator lamp is lighted. Check engagement of nose wheel steering by slight movement of rudder pedals.
3. Brakes—Checked.

   Depress brake pedals and check for proper braking.
4. Hydraulic pressure—Checked. (2950 to 3250 psi)

ENGINE SHUTDOWN.

1. Wheels—Chocked.
2. Generators—OFF.
3. Throttle—OFF.

BEFORE LEAVING AIRCRAFT.

1. All switches and controls—OFF, normal or safe.

This is the last page of Section II.

SECTION III

EMERGENCY PROCEDURES

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This section contains procedures to be followed in the event of 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. Although the procedures contained herein are considered the best available, the pilot must exercise sound judgment when confronted with an emergency. 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 pilot (aircraft commander) should notify the other crew member 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 the situation dictates.

Note

It is impossible to establish a pre-determined set of instructions that would provide a ready-made decision applicable to all situations. The emergency conditions combined with the pilot's analysis of the conditions of the aircraft, type of emergency and his proficiency are of prime importance in determining the urgency to land. The following provides general guidance.

Land As Soon As Possible:

An emergency will be declared. A landing should be accomplished at the nearest suitable airfield considering the severity of the emergency, weather conditions, field facilities, aircraft gross weight and command guidance.

Land As Soon As Practicable:

The mission should be terminated. Landing will be accomplished at a suitable alternate or the planned destination at the discretion of the pilot.

Note

- During any inflight emergency when structural damage is known or suspected that may adversely affect aircraft handling characteristics, a controllability check should be performed at a safe altitude. This check should be performed in the anticipated landing configuration. If adequate control response is available, and if it is practicable, maintain this configuration for landing.
- All odors not identifiable by the flight crew shall be considered toxic. Immediately go on 100% oxygen. Properly vent the aircraft and land as soon as practicable. Do not take off when unidentified odors are detected.
- 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.

1. **THROTTLE(S)—OFF.**
2. Motor engine.
   Affected engine throttle—Lift for 20 seconds. (If air is available)

If Fire Persists:

3. Fire pushbutton for affected engine—Depress.
4. Agent discharge—Actuate.

WARNING

The fire extinguishing agent is available for one actuation only. Depressing the engine fire pushbutton the second time will reopen the fuel shutoff valve and disarm the fire extinguisher agent discharge valve.

5. Abandon the aircraft.

EMERGENCY AIRCRAFT BRAKING.

In the event normal pedal braking is not available, several methods may be used to obtain braking and/or control aircraft speed to prevent damage to the aircraft and other equipment. The following are suggested solutions. However, the sequence will be dictated by the situation.

1. Anti-skid—OFF.
2. Thrust reduced.
3. Barrier engagement.
4. Auxiliary brake handle—Pull.

ABANDONING THE AIRCRAFT ON THE GROUND.

In an emergency requiring ground abandonment, the primary concern should be to leave the immediate area of the aircraft as soon as possible. Salvaging emergency and survival equipment should not be considered. To abandon the aircraft, disconnect personal leads and harness, open canopy hatches and exit over the side of the cockpit.

1. Throttles—OFF.
2. Fire pushbuttons—Depress.
3. Battery switch—OFF.

EMERGENCY ENTRANCE.

Emergency entrances are shown in figure 3-5.

TAKEOFF EMERGENCIES

ABORT/BARRIER ENGAGEMENT.

1. **THROTTLES—IDLE. (OFF FOR FIRE)**
2. **EXTERNAL STORES—JETTISON.** (If required)

Note

Nuclear stores will not be jettisoned.

3. **HOOK—EXTEND.** (If required)
4. Shoulder harness—Locked.

ABORTED TAKEOFF (WET OR DRY RUNWAY).

Full pedal deflection anti-skid braking with control stick full aft and centered will give the most effective deceleration for both dry and wet runways at normal take-off gross weights. Nose wheel steering may be used throughout the roll, except during barrier engagement. The chances for a successful barrier arrestment are greatly reduced by tire failure (blowout). The rim of the affected wheel normally snags or damages the cable, causing a missed engagement or cable failure. When a barrier engagement is anticipated, brake application of a severity great enough to cause tire blowouts should be avoided. Wet runway aborts are essentially the same as dry runway aborts with a noticeable exception: nose wheel steering and differential braking may be necessary to maintain directional control.

WARNING

- Hot brakes will usually occur during any maximum braking abort, wet or dry. Refer to Section V, "Brake Energy Limits." Do not set parking brakes after a maximum braking abort.
- If excessive braking is used at high speeds, the wheel blowout plugs may relieve tire pressure within 15 minutes after stop. Provisions should be made to cope with wheel fires which may start shortly after the blowout plugs relieve.
- Call the fire department after any emergency landing which results in hot wheels or brakes or use of the tail hook. Do not shut down the engines until after the fire trucks arrive. Fuel venting from the engines after shutdown may be ignited by the affected hot part.

Note

It is recommended that the heels be located below the pedals prior to brake application for wet runway aborts to prevent the foot from sliding up on the pedal during large differential rudder deflection.

BARRIER ENGAGEMENT.

On center engagements of the BAK-9, BAK-12, extended runout BAK-12, dual BAK-12, and BAK-13 barrier present no special problem to the aircrew. Tests show that with off center engagement the aircraft will be pulled to the "off center" side of the runway during the run out. Barrier contact should be made with nosewheel steering disengaged. No attempt to correct yaw or roll tendencies during the arrestment should be made until the aircraft is slowed sufficiently to insure aircraft control. Due to inherent stretch characteristics of the nylon tape used on the BAK-12, extended runout BAK-12, and BAK-13 barriers, a roll back occurs at the end of the tape run out. The aircraft will be rolled backwards from 10 to 200 feet, depending on the energy absorbed during the engagement. When roll back occurs after an engagement the aircraft will roll back parallel to the center line of the runway for either "on center" or "off center" engagements. Roll back may be shortened by the use of even braking; however, difficulties may be experienced in maintaining aircraft alignment with braking while it rolls back. The following recommended steps will aid in successful barrier engagements:

WARNING

Use of the MA-1 and MA-1A barrier with this aircraft has not been tested, therefore, results of their engagement cannot be accurately predicted.

- Disengage nosewheel steering prior to barrier contact.
- Do not attempt to correct yaw or roll tendencies during the arrestment until the aircraft is slowed sufficiently to insure aircraft control.
- Apply light braking at the end of the arrestment when possible to minimize roll back without causing the aircraft to pitch up.

ENGINE FAILURE DURING TAKEOFF.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort/Barrier Engagement" procedures, this section.

If Takeoff Is Continued:

1. **THROTTLES—MAX.**
2. **EXTERNAL STORES—JETTISON.** (If required)

Note

Nuclear stores will not be jettisoned.

3. Maintain takeoff speed until obstacles are cleared.
4. Landing gear handle—UP, when safely airborne.
5. Air source selector knob—OFF. (If required)

Note

- Significant thrust is gained with the air source selector knob OFF.
- With air source OFF, no servo air will be available for throttle boost or fuel tank pressurization. Lack of tank pressurization will degrade fuel dump rate.

6. Fuel dump—As required.
7. Flap/slat handle—As required.
   a. Flap/slat retraction—Maintain established pitch attitude and retract flaps/slats at a rate to maintain 8.5 degrees angle-of-attack.

WARNING

Excessive angle-of-attack may result from retracting flaps too rapidly.

Note

A normal reduction of rudder authority will occur as slats are retracted. This may be felt as a kick-back on the rudder if more than 7.5 degrees rudder deflection is being held.

8. Air source selector knob—As required.
9. Throttle of failed engine—OFF.
10. Attempt airstart if failure was nonmechanical and engine appears normal.
11. Land as soon as practicable.

SINGLE ENGINE RATE OF CLIMB.

Due to temperature, pressure altitude, gross weight, pilot technique, etc., the time and distance required to accelerate to best single engine climb speed is widely variable. The altitude attainable at a specific close-in obstacle is unpredictable unless takeoff speed is maintained until such obstacles are cleared. After close-in obstacles are cleared, maintain an attitude which will clear terrain and accelerate to best single engine climb speed. This speed will equate to 8.5 degrees angle-of-attack. Maintain established pitch attitude and retract the flaps/slats at a rate to maintain 8.5 degrees angle-of-attack.

ENGINE FIRE DURING TAKEOFF.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort/Barrier Engagement" procedures, this section.

If Fire Persists:

2. Fire pushbutton—Depress.
3. Agent discharge—Actuate.
4. Abandon the aircraft.

If Takeoff Is Continued:

1. **THROTTLE GOOD ENGINE—MAX.**
2. **THROTTLE BAD ENGINE—OFF.**
3. **EXTERNAL STORES—JETTISON.** (If required)

Note

Nuclear stores will not be jettisoned.

4. **FIRE PUSHBUTTON—DEPRESS.**

WARNING

- Use caution to prevent inadvertently depressing the wrong pushbutton and shutting down the good engine. Even though the button may be depressed again to open the fuel shutoff valve and allow restarting the engine, the hydraulic shutoff valves cannot be reopened to provide hydraulic power for flight control system operation.
- Depressing engine fire pushbutton the second time will reopen the fuel shutoff valve and disarm the fire extinguisher agent discharge valve.

5. **AGENT DISCHARGE—ACTUATE.**
6. Landing gear handle—UP, when safely airborne.
7. Maintain takeoff speed until obstacles are cleared.
8. If fire is confirmed and continues—Eject.

If Fire Is Extinguished:

9. Air source selector knob—OFF. (If required)

Note

- Significant thrust is gained with the air source selector knob OFF.
- With air source off, no servo air will be available for throttle boost or fuel tank pressurization. Lack of tank pressurization will degrade fuel dump rate.

10. Fuel dump—As required.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the aircraft. This should cause no concern however, since the fire will remain behind the aircraft. Observers may interpret this as an aircraft fire.

11. Flap/slat handle—As required.
    a. Flaps/slats retraction — Maintain established pitch attitude and retract flaps/slats at a rate to maintain 8.5 degrees angle-of-attack.

WARNING

Excessive angle-of-attack may result from retracting flaps too rapidly.

Note

A normal reduction of rudder authority will occur as slats are retracted. This may be felt as a kick-back on the rudder if more than 7.5 degrees rudder deflection is being held.

12. Air source selector knob—As required.
13. Fire detect circuit—Check.
14. Land as soon as possible.

AFTERBURNER FAILURE DURING TAKEOFF.

Full afterburner thrust will be required for normal takeoff. If afterburner fails during takeoff the thrust loss is significant. Abort the takeoff if failure occurs prior to being committed to takeoff. If failure occurs after takeoff is committed, attempt to regain AB by recycling the throttle to MIL and back to AB.

TIRE FAILURE DURING TAKEOFF.

Directional control is not difficult with a blown tire if nose wheel steering and differential braking are used properly. The aircraft will lean significantly to the side of the blown tire. The brake on the good tire should be used normally. Do not lock the brake on the wheel with the blown tire.

If Decision Is Made To Stop:

1. **ABORT.**
   Refer to "Abort/Barrier Engagement" procedures, this section.
2. Nose wheel steering—Engaged.

If Takeoff Is Continued:

1. External stores—Jettison. (If required)

Note

Nuclear stores will not be jettisoned.

2. Do not retract landing gear.

CAUTION

If gear is retracted with a blown tire, possible damage to the wheel well area may occur.

3. Instruments—Check.
   Check hydraulic, fuel and engine instruments to determine possible damage resulting from the disintegrated tire.
4. Fuel dump—As required.
5. Land as soon as practicable.

LANDING GEAR RETRACTION MALFUNCTION.

LANDING GEAR UP AND LOCKED INDICATION NOT OBTAINED.

If after 15 seconds following landing gear up selection the landing gear warning lamp remains lighted:

1. Landing gear control circuit breakers—Check.
2. Speed brake switch (left crew station)—In.
   Recycle speed brake to correct possible out-of-sequence condition.
3. Speed brake hydraulic valve circuit breaker—Pulled, then reset.
4. Obtain visual confirmation of landing gear position if possible. Check for possible malsequence of nose gear and uplock malsequence.
5. Do not recycle landing gear handle or place the utility system isolation switch to PRESSURIZE.

WARNING

Recycling the gear or placing the isolate switch to PRESSURE could result in damage to the gear, nose wheel steering mechanism or the aircraft, which may jeopardize safety for subsequent landing.

If The Landing Gear Warning Lamp Is Still Lighted:

6. Landing gear handle—DN.
   Obtain visual confirmation of landing gear position, if possible.
   a. If normal indications are not present, refer to "Landing Gear Malfunctions," this section.
   b. If normal indications are present, check landing gear doors, struts, steering linkages, and tires for proper extension, alignment, and security.
7. If any gear or steering abnormality exists or is suspected, an approach end barrier engagement is recommended. Refer to "Approach End Barrier Engagement," this section.

INFLIGHT EMERGENCIES

CAUTION LAMP ANALYSIS.

See figure 3-4 for analysis and suggested corrective action to be taken whenever a caution lamp is lighted.

AIR CONDITIONING SYSTEM MALFUNCTIONS.

CABIN OVERHEAT.

If Uncontrolled Cabin Overheat Occurs, Try To Close The Cabin Warm Air Valves As Follows:

1. Mode selector switch—OFF.
2. Air source selector knob—OFF.
   Immediately descend and decelerate to the "Ram or Emer Mode Flight Limits", Section V.
3. Air source selector knob—EMER. (Use RAM if EMER not installed)

Note

- For supersonic flight under conditions of high total temperature readings, placing the air source selector knob to EMER or RAM will result in excessive cabin temperature.
- With air source in EMER or OFF, servo air will not be available for throttle boost, fuel tank, or cabin seal pressurization.

If Cabin Temperature Is Still Uncontrollable:

4. Air source selector knob—OFF.

If Cabin Overheat Continues:

5. Left engine—Reduce to IDLE power.

If Cabin Overheat Persists:

6. Left engine—Power as required.
7. Right engine—Reduce to IDLE power.

The Following May Be Used As A Last Resort:

If hot airflow is not reduced to a bearable level, reduce airspeed. (10 degrees angle-of-attack with full flaps)

8. Canopy—Open one or both. (Be prepared for wind blast)
9. Land as soon as practicable.

CABIN TOO COLD.

Excessive Cooling: (High Air Flow)

If the cabin temperature becomes uncontrollably cold due to excessive cooling airflow in the MAN or AUTO modes, cabin cooling airflow can be reduced as follows:

1. Air source selector knob—L ENG.
2. Left engine—Reduce to IDLE power.
3. Right engine—Power as required.
4. ECM selector knobs—ON. (All 3 bands)

Loss of Heating: (Low Air Flow)

If the cabin temperature becomes uncontrollably cold in the MAN or AUTO modes due to a loss of heating airflow:

1. Set power on one engine (left or right) to MIL for 20 seconds, then return as required.

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 uncontrollable conditions, eject at least 15,000 feet above the terrain whenever possible. If the aircraft is controllable, attempt to decelerate as much as practical prior to ejection by zooming the aircraft, thus trading airspeed for altitude. If the aircraft 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, 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. As with all aircraft ejection systems, safe ejection is enhanced by establishing the best conditions possible prior to ejection. For "Minimum Terrain Clearance For Ejection" and "Crew Module Ejection Sequence" see figures 3-1 and 3-2.

WARNING

- Under certain conditions of crew module weight and/or tail wind, zero altitude and zero airspeed ejection capability may be prejudiced. Because of the variables involved, ejection is not recommended at zero altitude with less than 50 KIAS.
- When ejection is necessary with a known or suspected pitot-static failure, every attempt should be made to slow the aircraft to below an estimated 300 knots. At aircraft gross weights up to 100,000 pounds and with the wings forward of 50 degrees, a 10 degree or greater angle-of-attack in one "g" stabilized flight will ensure that the airspeed is below 300 knots. At altitudes above 20,000 feet MSL, slowing the aircraft before ejection is less critical but is still recommended. In all cases, ejection should be initiated before stall/departure is reached.
- Ejection performance is valid only when crew module gross weight and center-of-gravity is within limits specified in T.O. 1-1B-40 and reflected in Form 781.

BEFORE EJECTION. (IF TIME PERMITS).

1. Reduce airspeed. (As practicable)
2. Advise crew member of situation.
3. Transmit MAYDAY. (Give position)
4. Emergency radio beacon plastic plug pulled out. (If applicable)
5. IFF master control knob—EMERG.

Crew Module Ejection Sequence
(Typical Low Speed Ejection)

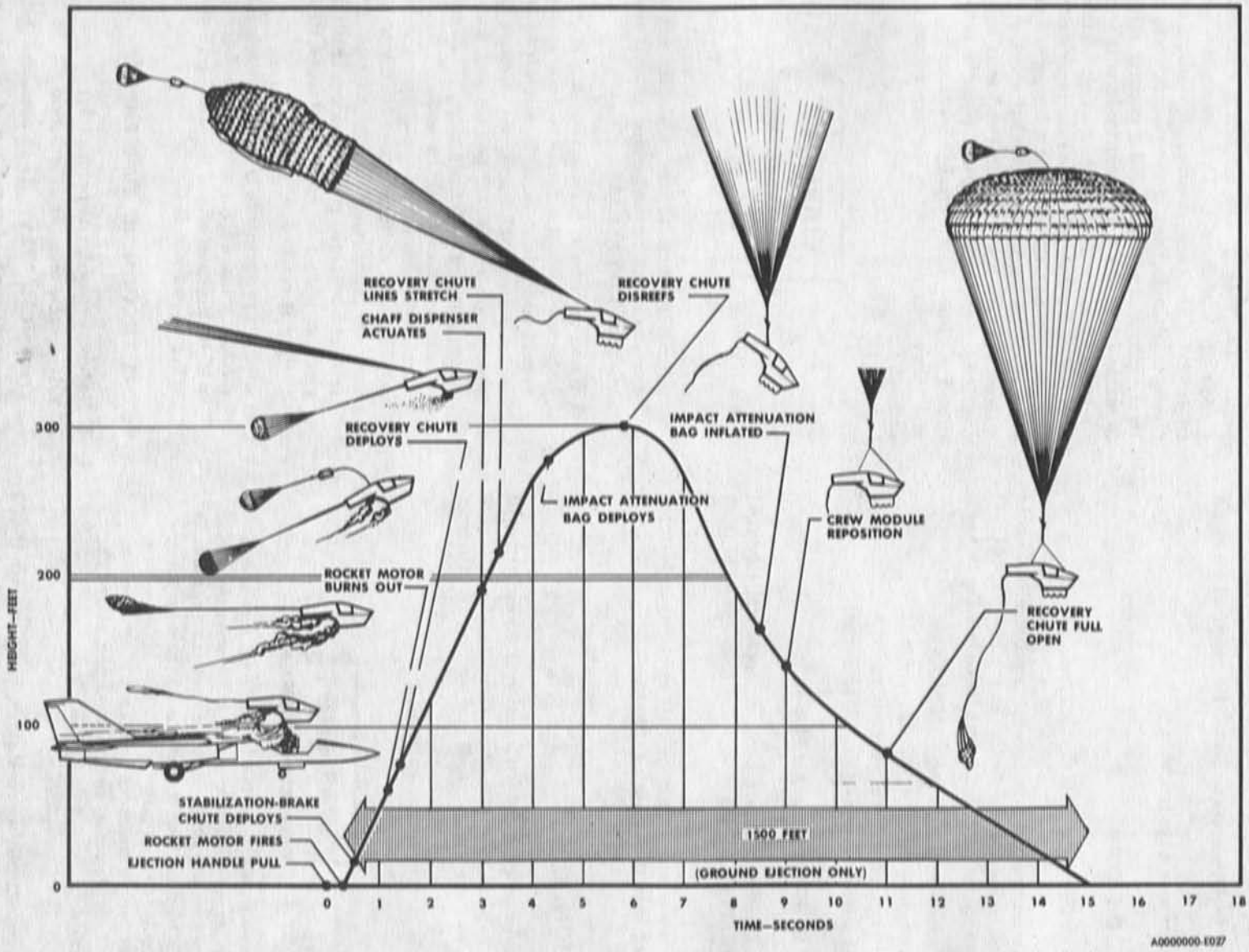


Figure 3-1.

Minimum Terrain Clearance For Ejection

DATA BASIS:
1. CREW MODULE GROSS WEIGHT AND CG WITHIN LIMITS SPECIFIED IN T.O. 1-1B-40.
2. INITIAL YAW ANGLE LESS THAN 5 DEGREES FOR SPEED BELOW 500 KNOTS AND LESS THAN 1 DEGREE FOR SPEEDS ABOVE 500 KNOTS.
3. SEA LEVEL ALTITUDE
4. FOR DIVING TURNS THE CLEARANCE REQUIREMENTS ARE ADDITIVE PLUS 300 FEET.
5. NO ALLOWANCE FOR CREW REACTION TIME INCORPORATED. ★

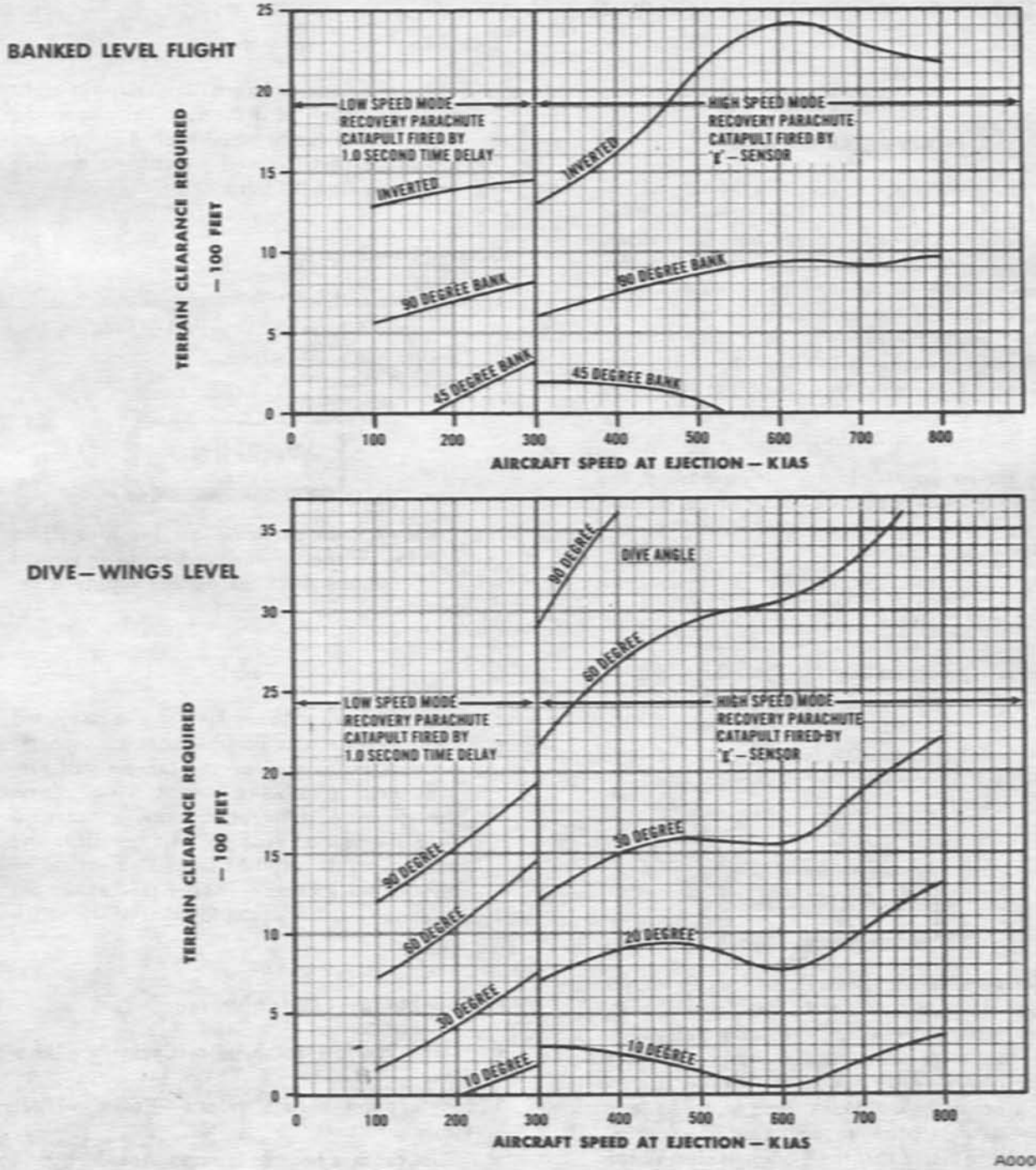


Figure 3-2.

6. Chaff dispenser control lever—ON. (OFF for tactical considerations only)
7. Oxygen—100 percent.
8. Inertia reel control handle—LOCKED.
9. Oxygen mask and fittings—On and checked. Keep mask on during ejection and descent.
10. Adjust seat full down and aft.

EJECTION.

1. EJECTION HANDLE—SQUEEZE AND PULL.

WARNING

- A three tenths of a second delay should be expected between the time the ejection handle is pulled and the rocket motor fires. This delay allows the enertial reels to retract and lock the crew member in the upright position. Injury could occur if the crew member is not in a firm upright position with head against headrest when ejection is initiated. Ejecting with the seat high enough to cause the shoulders or back to contact the headrest will increase probability of spinal injury.
- To prevent possible injury during landing, restraint harness should be cinched tight during descent and remain tight until module comes to rest after impact.

DURING DESCENT.

1. Emergency oxygen handle—Pull. (If required)
2. Oxygen control knob—EMER or 100%. (If required)
3. Parachute deploy handle (when below 15,000 feet as indicated by the standby altimeter)—Pull. (If required)

WARNING

If ejection occurs above 15,000 feet, do not actuate the parachute deploy handle until passing through that altitude, if terrain elevation permits, to prevent loss of recovery parachute and to assure sufficient oxygen supply. If ejection occurs below 15,000 feet, allow at least five seconds prior to actuating the deploy handle to assure that the module is clear of the aircraft and to allow deceleration of the module. Do not attempt to beat the system.

Note

- Oxygen mask should be on prior to impact.
- If smoke or fumes are present in the cockpit, they should be vented by opening canopy hatch(es) as soon as the main parachute has deployed. The hatch(es) must be closed and locked prior to impact.

AFTER LANDING.

Note

The right flotation bag will overlap the right hatch and must be deflated to fully open the hatch. Egress can be accomplished by opening the hatch partially and puncturing the bag or exiting thru the left hatch.

Ground Landing.

1. Severance and flotation handle—Pull.
2. Parachute release handle—Pull. (If required)
3. Canopy hatch(es)—Open.

WARNING

Crew exposure to toxic gas resulting from ejection in the unventilated cabin is limited to 15 minutes after the oxygen supply is exhausted.

Note

After descending below 8000 feet in an ejected crew module it is possible that atmospheric pressure differential on the canopy will prevent them from being opened. To eliminate this pressure differential, remove the caps from the snorkel vent ports (if installed) and push in on the exposed tube. This should be done on the ground only after the module has come to rest, due to the location of the ventilation ports.

If Crew Module Lands Inverted:

4. Inertia reel control handle—Cycle to relieve tension.
5. Single point harness release—Rotate 90 degrees, either direction.

   Use caution because lap and shoulder straps will release simultaneously, requiring crew member to support himself.

6. Open canopy hatch as far as possible and lock in place by moving the latch handle to lock detent (midpoint) position.

Water Landing.

1. Severance and flotation handle—Pull.
2. Parachute release handle—Pull.
3. Canopy hatch(es)—Open.

WARNING

Crew exposure to toxic gas is limited to 15 minutes after oxygen depletes. In the event of ejection over water, if it is necessary for crew members to remain in the module for an extended period of time awaiting recovery both canopies should be opened approximately every three hours for ventilation.

4. Bilge pump—Engage and operate. (If required)
5. Auxiliary flotation handle—Pull. (If required)

Note

Survival two way radio antenna must be projected out of the module in order to function properly.

ALTERNATE ESCAPE PROCEDURE.

WARNING

The following procedures are presented as a last resort action for the crew to depart the aircraft and no in-depth studies have been made to confirm whether or not they will be successful. Crew safety may be jeopardized using these procedures; however, this may be preferable to remaining with the aircraft.

Each ejection handle has an explosive crossover to ensure SMDC line initiation and operation. If one handle fails to initiate ejection, the other handle should be pulled. If neither handle works, the crew module may still be severed from the aircraft using the severance and flotation handle. (Emergency oxygen provision will precede FLSC severance.) This action will not retract the harness inertia reels, the racket motor will fire, and the stabilization brake chute will not deploy. To ensure separation, a negative "g" maneuver should be performed, circumstances permitting, before pulling the severance and flotation handle. Once clear of the aircraft the parachute deploy handle can then be operated. If the module does not clear the aircraft after the severance and flotation handle has been pulled, the parachute deploy handle may also assist in module separation. It should be remembered, however, that the capsule will free fall without brake chute stabilization. For this reason it is highly desirable that module separation be accomplished within the recommended parachute deployment envelope (maximum altitude 20,000 feet preferably below 15,000 feet, and below 300 KIAS).

1. Ejection handle—Squeeze and pull.
2. Second ejection handle—Squeeze and pull.
3. Slow aircraft to below 300 KIAS and descend to below 20,000 feet if possible.
4. Inertia reel control handle—LOCKED.
5. Perform negative "g" maneuver (preferably inverted). (If possible)
6. Adopt ejection posture.
7. Severance and flotation handle—Pull.
8. Passing 15,000 feet—Pull parachute deploy handle.

WARNING

Operation of the severance and flotation handle will unguard the parachute release handle. Pulling this handle will explosively release the recovery parachute and the module will free fall.

9. Brace for landing.

Note

A rough landing with severe post landing gyrations will probably occur due to the inflated flotation bags activated by the severance and flotation handle.

ENGINE EMERGENCIES INFLIGHT.

SINGLE ENGINE FAILURE.

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 using "Single Engine Landing" procedures, this section.

SINGLE ENGINE FLIGHT CHARACTERISTICS.

Single-engine flight characteristics are essentially the same as the normal flight characteristics due to the proximity of the thrust lines to the center of the aircraft. With one engine inoperative, slight rudder deflection is required to prevent yaw toward the failed engine. The aircraft design is such that no one system is entirely dependent upon a specific engine, thus loss of one engine will not result in the loss of a complete system. Aircraft service ceiling and/or range for single-engine operation (military or afterburner thrust) is a function of aircraft configuration and gross weight. For best range, set military power on the good engine (observing TIT limits) and maintain approximately 285 KIAS, 26 degree wing sweep and allow aircraft to descend to but not below best single-engine cruise altitude as shown in Appendix I. As a rule of thumb, when descent is made using this power setting and airspeed, the best single-engine cruise altitude is that altitude where rate-of-descent stops. If a climb is required, best range can be obtained by anticipating altitude requirements in advance so as to allow a gradual climb with minimum change to power setting (285 KIAS). If time does not permit a gradual climb, use afterburner thrust on the good engine (trading range for altitude). During single-engine operation with various landing gear and wing flap configurations, care must be exercised to avoid rapid airspeed bleed-off and/or excessive sink rates. Limited thrust available makes airspeed response to power much slower than normal two-engine operation.

DOUBLE ENGINE FAILURE.

1. Fuel panel—Checked.
2. Throttles—As required.
3. Airstart button(s)—Depress.
4. Eject. (If airstart cannot be accomplished)

AIRSTART.

Satisfactory airstarts may be accomplished throughout the flight envelope. Approximately 300 KIAS may be required for ram airstarts. Below these airspeeds, engine crossbleed may be required to restart an engine.

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.
- If the throttle has been retarded below IDLE position, the airstart ignition button must be depressed to provide 55 seconds of ignition to the engine to be restarted.

1. Fuel panel—Checked.
   Check fuel feed selection and fuel quantities to assure that fuel is available to the engine.
2. Throttle of affected engine—OFF.
3. Throttle of affected engine—IDLE.
4. Airstart ignition button—Depress momentarily. (Check for relight within 20 seconds)
5. Generator switch of affected engine—OFF-RESET, START (pause), then release to RUN.
   If the generator caution lamp remains lighted, after engine start, place the generator switch to OFF-RESET, then to START (pause), then release to RUN and check that the lamp goes out and the power flow indicator reads NORM.

Note

The generator switches are lever-locked in OFF-RESET, spring-loaded from START to RUN, and the generator caution lamp will go out when the switch is held in START.

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 position.
9. Throttle of affected engine (at 17 percent)—IDLE. (Check for relight within 20 seconds.)
10. Generator switch of affected engine—OFF-RESET, START (pause), then release to RUN.
    If the generator caution lamp remains lighted place the generator switch to OFF-RESET, START (pause), then release to RUN and check that the lamp goes out and the power flow indicator indicates NORM.
11. If start is not obtained:
    a. Emergency generator switch—As required.
       The emergency generator switch should be left in AUTO to prevent a drain on the hydraulic system when flight is being conducted under visual flight conditions. Emergency electrical power will be available within 1 second if the other generator fails. Position switch to ON during instrument flight and prior to descent for night landings.
    b. Engine ground start switch—OFF.

ENGINE STALL.

In the event of engine stall on one or both engines, proceed as follows:

1. Throttle of affected engine(s):
   a. If in AB—Retard to MIL.
   b. If at MIL or below do not change power setting. When engine recovers, select desired power.

Note

In the event that a compressor stall and/or afterburner blowout occurs in afterburner operation, but fully stalled engine condition does not follow, an afterburner relight from MIL power may be initiated at any flight condition.

If Stall Does Not Clear Within 10 Seconds:

2. Shut down stalled engine (one only). Note that rpm is decreasing.
3. Perform airstart and set power as desired.

If Stall Is Not Cleared By Above Procedure:

4. Shut down stalled engine a second time. Note that rpm is decreasing.
5. Decelerate to below mach 0.90 or 415 KIAS. (Whichever is less)
6. Perform airstart and set power as desired.

ENGINE FIRE DURING FLIGHT.

1. **THROTTLE BAD ENGINE—OFF.**
2. **FIRE PUSHBUTTON—DEPRESS.**

WARNING

- Use caution to prevent inadvertently depressing the wrong pushbutton and shutting down the good engine. Even though the button may be depressed again to open the fuel shutoff valve and allow restarting the engine, the hydraulic shutoff valves cannot be reopened to provide hydraulic power for flight control system operation.
- Depressing the engine fire pushbutton the second time will reopen the fuel shutoff valve and disarm the fire extinguisher agent discharge valve.

3. **AGENT DISCHARGE—ACTUATE.**

Note

Do not attempt restart.

4. If fire is confirmed and continues—Eject.

Note

Trailing smoke as viewed by another aircraft or ground observer may be used as an indication of fire. Engine smoke should not be confused with an engine or aircraft fire.

5. If fire ceases, land as soon as possible using "Single Engine Landing" procedures, this section.
6. Fire detect circuit—Check.

FUSELAGE FIRE DURING FLIGHT.

1. **FUSELAGE FIRE PUSHBUTTON—DEPRESS.**
2. **AGENT DISCHARGE—ACTUATE.**
3. Air source selector knob—EMER. (Use OFF if EMER not installed)

Note

- The fuselage fire warning lamp may not go out immediately after discharging the fire extinguishing agent. Agent effectiveness may be verified by observing the weapons bay temperature indicator, additional caution lamp indications, or other symptoms of degradation.
- An overheat condition may also cause the fuselage fire warning lamp to light. If time permits, consider opening the weapons bay door(s) in an attempt to cool the area or blow out the fire. This may be done at the discretion of the aircraft commander.
- In the event nuclear weapons are aboard the aircraft, pull out the nuclear master and nuclear unlock circuit breakers.
- A rain removal duct failure could cause the fuselage fire warning lamp to light and remain lighted until the hot air source is shut off.

4. If fire is confirmed and continues—Eject.
5. If fire ceases—Land as soon as practicable.

SAFE JETTISON PROCEDURES.

In the event an aircraft emergency precludes landing with bombs aboard or Command Tactical Doctrine procedures dictate a safe release of nonexpended bombs, proceed as follows:

WARNING

External fuel tanks must have been jettisoned prior to jettisoning of external bombs.

NUCLEAR AND CONVENTIONAL WEAPONS.

For safe jettison of nuclear and conventional bombs, refer to Section V and the appropriate weapons delivery manual.

EXTERNAL TANK/PYLON JETTISON PROCEDURES.

If jettison of external tanks/pylons is required, proceed as follows after selecting a safe jettison area (if possible). Empty tanks or tanks with 1800 pounds or more fuel remaining may be jettisoned.

1. Wing sweep—16 to 26 degrees.
2. Altitude—10,000 feet or less.
3. Airspeed—300 KIAS or less.
4. Angle-of-attack—10 degrees or less.
5. Flaps/slats—Extended or retracted.
6. External stores jettison button—Depress.
7. Tanks may be jettisoned simultaneously.
8. Fixed pylons will jettison 0.50 second after tanks if flaps are retracted.

FLAP/SLAT MALFUNCTIONS.

If the flaps stop at an intermediate position during retraction or extension, a likely cause is a dislodged or broken flap vane. Further flap actuation could result in extensive flap damage or loss of the malfunctioning flap vane. It is recommended that further flap operation not be attempted, and a landing made with the existing flap setting, provided landing conditions are acceptable (RCR, ceiling, etc.). Placing the flap/slat switch to EMER will relieve hydraulic pressure to the flap motor and could prevent further damage. If marginal flap-up landing conditions exist, flap extension may be attempted. If practical this should be accomplished over a designated drop area or unpopulated area. If landing is made with existing flap position refer to "Landing With Partial Flaps," Section II.

ASYMMETRIC SLAT.

If it is determined that an asymmetric slat extended condition exists, proceed as follows:

1. Flap/slat handle—Position to obtain symmetric slat extension.
   Extend or retract slats, monitoring operation, in an attempt to obtain equal slat deflection on each side, and check for a reduction in degree of rolloff.

CAUTION

Do not place the flap/slat handle beyond the SLAT DOWN position. If asymmetrical slat extended condition exists, and the flap and slat handle is in FLAP DOWN or the flap and slat switch is in EMER, upon initial travel of the flaps the asymmetry device will cause the torque shaft brakes to engage preventing any further flap or slat movement.

If Rolloff Is Reduced To Zero:

2. Slat deflection—Checked.
   Check for equal slat deflection on each side. If slats are extended, observe the slats extended airspeed and limit maneuver load factors limitations. With slats extended or retracted, refer to "No Flap Landing," this section.

If Rolloff Is Still Present:

3. Observe the slats extended airspeed and limit maneuver load factors limitations and refer to "Landing with Asymmetric Slats," this section.

ASYMMETRIC FLAP.

An asymmetric flap condition probably exists if during takeoff or landing when flaps/slats are retracted or extended, the aircraft starts to roll off. Flight controls should be applied as required to maintain wings level and the flaps handle should be returned toward its previous position (25 degrees for takeoff or UP for landing) until rolloff stops. This should place the flaps back in a symmetrical condition. For asymmetric flap condition proceed as follows:

1. Flap/slat handle—Return towards previous position until rolloff stops.
2. Fuel dump—As required.
3. Land as soon as practicable. Refer to "No Flap Landing," this section.

FLIGHT CONTROL SYSTEM MALFUNCTIONS.

Note

The flight control system emergency procedures presented in this section are the immediate steps required to correct critical abnormal flight characteristics. The sequence of these steps is determined by the degree of emergency. It may not be necessary to accomplish the complete procedure to correct a particular malfunction.

Various flight control system malfunctions are indicated by the lighting of an associated caution lamp. Refer to figure 3-4 for analysis of all caution lamps.

LOW FREQUENCY OSCILLATIONS IN PITCH OR ROLL AXIS.

Although the pitch and roll gain changers are redundant, certain malfunctions may occur which cause the pitch or roll adaptive gain to become high enough to cause the pitch or roll damper servo to drive the horizontal tails in a limit cycle oscillation at a frequency between 1.7 and 3 cps. Under certain conditions, this oscillation may also appear in the control stick. A decrease in airspeed or increase in altitude will alleviate the oscillation. The pitch or roll gain can be reset to its minimum value by cycling the appropriate damper switch to OFF and back to DAMPER. If oscillation occurs proceed as follows:

1. Decrease airspeed or increase altitude, and/or cycle the appropriate damper switch to OFF and return to DAMPER.

Note

Momentarily cycling the damper switch to OFF does not necessitate going to the "Damper Off" operating region.

UNSCHEDULED PITCH PARALLEL TRIM.

A malfunction in the pitch parallel trim system will cause the trim to drive nose up or down and will be evidenced by the stick driving at a normal system rate. If condition occurs proceed as follows:

1. Auxiliary pitch trim switch—OFF, then trim nose up or down as required.
2. Leave auxiliary pitch trim switch out of the stick position for remainder of flight.
3. Terminate pitch autopilot, manual and auto TF operation.
4. Land as soon as practicable.

UNSCHEDULED SERIES TRIM.

Malfunctions within the series trim circuits can cause the actuator to stop, or drive nose up or down. The maximum rate of drive will be approximately 1.4 degrees per second. Series trim driving will not cause the control stick to be driven. For unscheduled driving of the series trim proceed as follows:

1. Control stick trim button—Trim parallel trim to counter the maneuver.
2. Terminate pitch autopilot, manual and auto TF operation.
3. Land as soon as practicable.

UNSCHEDULED ROLL TRIM.

Should a malfunction occur in the roll trim circuit, it can be overcome by initially applying stick force. Placing the roll damper OFF will remove the roll trim input. If unscheduled roll trim is encountered proceed as follows:

1. Roll damper switch—OFF.
2. Flight control disconnect switch—OVRD. (If desired)

Note

If desired, roll trim inputs may be removed by placing the flight control disconnect switch to the OVRD position. Placing the disconnect switch to OVRD position removes the following inputs from the system: pitch and roll autopilot commands, pitch damper trim inputs, TFR climb/dive commands, adverse yaw compensation commands, and pedal shaker inputs.

3. Roll damper switch—DAMPER. (If desired)
   If roll damping is desired after placing flight control disconnect switch to OVRD, return roll damper switch to DAMPER.

UNSCHEDULED YAW TRIM.

Should a malfunction occur during flight in the yaw trim circuit which causes the trim unit to drive hard over, pedal force must be used to oppose the ensuing side slip. The required forces can be reduced by placing the rudder authority switch to FULL position. Should an unscheduled yaw trim condition occur, proceed as follows:

1. Rudder trim—Retrim. (If possible)
2. Rudder authority switch—FULL.
3. Land as soon as practicable.

AUTOPILOT DISCONNECT PROCEDURE.

If a malfunction should occur while on autopilot, the autopilot should be disengaged through use of the autopilot release lever. However, under certain malfunctions, this procedure may not fully suffice. The autopilot switches can be placed to the "DAMPER" position if disengagement has not occurred, then position the flight control disconnect switch to the OVRD position. These actions will terminate autopilot use. If a malfunction occurs, proceed as follows:

1. Autopilot release lever—Depress.
2. Autopilot damper switches—DAMPER.

If Disengagement Has Not Occurred:

3. Flight control disconnect switch—OVRD.

Note

Placing the flight control disconnect switch to OVRD position removes the following inputs from the system: pitch and roll autopilot commands, roll trim commands, pitch damper auxiliary trim inputs, TFR climb/dive commands, adverse yaw compensation commands, and pedal shaker inputs.

UNSCHEDULED PITCH MANEUVER.

The following procedure applies in event of an unscheduled pitch maneuver that cannot be attributed to a TFR fly-up. This type of maneuver may or may not be abrupt.

1. Autopilot release lever—Depress and hold.
2. Pitch damper switch—OFF.
   With pitch damper off, observe damper off limits.

If Control Of The Aircraft Cannot Be Achieved:

3. Eject.

UNSCHEDULED ROLL/YAW MANEUVER.

The following procedure applies in the event of an unscheduled roll/yaw maneuver. This type of maneuver may or may not be abrupt. The maneuver will be characterized by the inability to hold constant heading, a build-up of lateral acceleration, and the requirement for increasing lateral control to maintain wings level flight. For unscheduled roll/yaw maneuvers proceed as follows:

WARNING

- Engine stall may occur on one or both engines due to side slip.
- If unscheduled roll/yaw maneuver is due to hardover rudder which cannot be compensated for by use of rudder command, the aircraft cannot be landed and speed should not be allowed to go below mach 0.60 as adequate aircraft control will not be available.

1. Autopilot release lever—Depress and hold.
2. Rudder authority switch—FULL.
3. Roll or yaw damper switch—OFF. (As required)
   Cycle the appropriate damper switch to OFF to check if this eliminates the unscheduled maneuver. If it does, leave the damper OFF; if not, return the switch to DAMPER.

If Control Is Available:

4. Wing sweep—Sweep wings forward to obtain spoilers. (Within airspeed limits)
   Sweep the wings forward of 45 degrees to provide spoilers to assist in controlling the roll or yaw maneuver. The wings should be swept slowly, when between 50 and 45 degrees in anticipation of a possible abrupt lateral transient when the spoilers are activated, especially if a large lateral control stick displacement is being held to counteract the maneuver.

If Control Cannot Be Achieved:

5. Eject.

FUEL SYSTEM MALFUNCTIONS.

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 reservoir tank or number 5 in the aft tank) or the two inboard wing transfer pumps and number 12 transfer pump in the weapons bay tank. The transfer pumps cannot be operated while one of the fuselage booster pumps is operating. When the engine feed selector switch is in FWD, only the number 4 pump in the reservoir tank will be operating and will supply fuel to both engines. When the engine feed selector switch 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 switch 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 until the differential reduces to less than 8250 pounds. Number 5 pump will then supply fuel to the engines until the differential again increases to 8500 pounds. The above operation will repeat with the number 4 and 5 pumps alternately supplying 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 or weapons bay tanks, the engine feed selector switch must be turned OFF and the fuel transfer switch placed to WING, BAY, or AUTO. 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 transfer. If distribution gets out of tolerance, it can be corrected by positioning the engine feed selector switch to AUTO. During suction feed, the fuel manifold low pressure caution lamps will light. Refer to "Gravity Feed," this section.

Engine Feed.

1. Engine feed selector knob—AUTO.
   Closely monitor fuel quantity in the fuselage tanks to maintain 8200 (±400) pounds fuel differential.
2. Fuel transfer knob—OFF.
3. Fuel tank pressurization selector switch—PRESSURIZE.

Wing Or Weapons Bay Tank Fuel Transfer.

1. Fuel transfer knob—As required.

WARNING

When aft tank boost pumps are not operating, the fuel in the aft tank cannot be transferred. Refer to "Landing with Aft Abnormal Fuel Distribution," this section.

Note

When the wings are swept aft during fuel transfer on emergency generator power, 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.

2. Engine feed selector knob—OFF.
   Monitor fuel quantities in the fuselage tanks to maintain a satisfactory fuel differential. Refer to "Gravity Feed," this section.

External Tank Transfer.

External tank fuel can be transferred while on AUTO engine feed.

1. Engine feed selector knob—AUTO.
2. Fuel transfer knob—AUTO.
   Particular tank sets may also be selected by positioning transfer knob to OUTBD, CENTER or INBD.
   Monitor fuel quantity in the fuselage tanks. To maintain 8200 (±400) pounds fuel differential, it may be necessary to periodically position transfer knob to OFF.

Gravity Feed.

During gravity feed, sufficient fuel pressure is available to allow operation within the following ranges of conditions:

1. Military power—Zero to 30,000 feet altitude, up to maximum airspeed with or without fuel tank pressurization.
2. Max AB—Zero to 30,000 feet altitude.
   a. Zero to 300 KIAS—Without fuel tank pressurization.
   b. Zero to 1.30 mach—With fuel tank pressurization.

WARNING

During gravity feed the engines are fed from the forward tank only. Refer to "Abnormal Fuel Distribution," this section.

Fuel Dump.

1. External fuel tank—Jettison. (If required)

Note

External tank fuel can be transferred while on AUTO engine feed. Refer to "External Tank Transfer," this section.

2. Establish an airspeed of 0.7 mach or less and maintain 1 "g" flight.
3. Position wings slowly to 26 degrees.
4. Engine feed selector knob—AUTO.

5. Airspeed—Established, no greater than 350 KIAS or mach 0.75, whichever is less.
6. Fuel dump switch—DUMP.
7. Fuel distribution—Monitor.
   Fuel will be dumped from the forward tank faster than emergency power can transfer fuel from the aft tank. When the differential fuel distribution between forward and aft tank is approximately 2,000 pounds, stop dumping until the differential approaches 8,000 pounds again.
8. Fuel dump switch—OFF.
9. Engine feed selector knob—OFF.
10. Fuel transfer knob — AUTO. (Repeat steps 5 through 8 if required)
11. Fuel transfer knob—OFF.
12. Fuel dump switch—OFF.
13. Engine feed selector knob—As required.
14. Land as soon as practicable.

FUEL PRESSURE CAUTION LAMP LIGHTED.

L Fuel Press and/or R Fuel Press Caution Lamp(s) Lighted:

1. Throttles—Set minimum power practical.
2. Engine feed selector knob—Checked.
   Check engine fuel feed selection to insure that fuel is available to the engine(s). Check fuel pump low pressure indicator lamps for evidence of boost pump failure(s) or an empty tank.
3. Fuel tank pressurization selector switch—PRESSURIZE.

If Either/Both Fuel Press Caution Lamp(s) Remain Lighted:

4. Refer immediately to "Excessive Fuel Depletion Procedure," this section.

If The Fuel Press Caution Lamp(s) Do Not Remain Lighted:

5. Check for a possible loss of fuel by comparing:
   a. Planned fuel on board versus actual.
   b. Flowmeters with each other (and against normal flowrate for flight condition).
   c. Totalizer fuel drop versus fuel flowmeters.
      If no discrepancy is noted, continue mission. If any portion of this check reveals a loss of fuel, consult "Excessive Fuel Depletion Procedure," this section.

EXCESSIVE FUEL DEPLETION PROCEDURE.

Some fuel system failures can result in fuel depletion rates that are capable of exhausting the entire aircraft fuel supply in minutes. It is highly recommended that the following steps be accomplished without delay while enroute to the nearest suitable airfield. If an excessive fuel depletion rate is known or suspected proceed as follows:

WARNING

Due to the fire hazard from fuel impinging on the fuselage, afterburner thrust will not be used during or following an excessive fuel depletion, or sooner than one minute after completing air refueling operations, unless the additional thrust is necessary to sustain flight.

1. Throttles—Set minimum power practical.
2. Fuel transfer knob—OFF.
   Terminate all transfer from weapon bay, external and wing tanks. If this stops the excessive fuel depletion, the leak/failure was in a transfer line. Normal fuel procedures may be used, however fuel should be transferred from weapon bay, external and wing tanks only if necessary to reach a suitable airfield.

Note

To avoid unnecessary loss of fuel do not allow transfer system to operate with fuselage tanks full. If transfer is necessary, allow total fuselage fuel to deplete to approximately a 20,000 pound maximum and select the desired tank(s) manually. (Do not use AUTO transfer.)

3. Fuel flowmeters—Checked.
   Determine if either fuel flow is excessive by comparing:
   a. Flowmeters with each other and against normal flow rate for flight condition.
   b. Engine instruments versus throttle position.
   c. Engine response to throttle movement.
   If neither fuel flow is considered excessive, proceed to step 4. If either fuel flow is excessive, proceed as follows:
   d. Throttle of affected engine—OFF.
   e. Fire pushbutton—Depress.
   f. Speed brake—Extend momentarily.
      Extend speed brake to ventilate the main wheel area.
   g. Weapons bay doors—Open, close.
   h. Land as soon as possible.

If An Excessive Fuel Depletion Rate Exists:

4. Altitude—Checked.
   Gravity feed with fuel tanks unpressurized may be accomplished at any altitude below 30,000 feet provided the throttles are set at MIL power or below.
5. Engine feed selector knob—OFF.
6. Tank pressurization switch—OFF.
7. Fuel tank depletion rate—Checked.
   If the excessive depletion rate is reduced or stopped, maintain gravity feed except for periodic use of AUTO engine feed to establish proper fuel differential between the forward and aft tanks. If the use of AUTO fails to obtain proper fuel distribution, refer to "Landing With Abnormal Fuel Distribution," this section.

ABNORMAL FUEL DISTRIBUTION/INDICATION.

Suspected Fuel Quantity Indicator(s) Malfunction.

Continued operation with a fuselage fuel quantity indicator malfunction and with the engine feed selector knob in the AUTO position may result in a fuel imbalance and a shift in center-of-gravity. Manual fuel management is necessary to keep the desired 8200 lb. fuel differential between the forward and aft fuselage tank.

1. Engine feed selector knob—AFT.
2. Fuel quantity indicators—Test.
   Check forward, aft, and total fuel quantity indications:
   a. If an indicator fails to test, it should be considered inoperative. Monitor the other two indications to determine the fuel distribution and operate fuel system manually to maintain at least 8200 pound differential.

Note

If the total/select fuel quantity indicator is considered inoperative, but both the forward and aft tank indicator pointers operate normally, auto feed may be continued.

   b. If more than one fuel quantity indicator fails to test, remain on aft tank feed and burn the aft tank empty. Select forward tank feed when the aft tank pump lamps light and reduce forward tank fuel quantity to below 8000 pounds prior to landing.

If Aft Tank Is Not Feeding:

3. Do not dump fuel.

If Elevator Position Is More Than 1 Degree Down:

4. Wing sweep—Aft until 1 degree or less is obtained.

WARNING

- If the aft tank is not feeding, an aft center-of-gravity problem may result from continued flight. Land as soon as possible. Do not dump fuel. Jettison external stores if necessary to reduce gross weight.
- If the elevator position is more than 1 degree down position wings aft until one degree or less is obtained.

Note

For aircraft with inoperative surface position indicators, a wing sweep of 40 degrees will provide adequate safety margin for the most adverse fuel distribution that can be encountered with no external stores.

5. External stores—Jettison. (If required)
6. Land as soon as possible. Refer to "Landing With Aft Abnormal Fuel Distribution," this section.

Forward Abnormal Distribution.

Crosscheck elevator position and fuel quantity indicators. After confirming an abnormal forward distribution, select FWD engine feed until desired distribution is obtained. Dump can be utilized if required. Refer to "Landing With Forward Abnormal Fuel Distribution," this section.

GENERATOR FAILURES.

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 generator caution lamp light proceed as follows:

Note

The flight control system computers operate on 115 volt ac power from the essential ac bus. The essential ac bus is in-turn normally fed by the left generator. An interruption of power to the essential ac bus, such as loss or shutting down of the left generator or switching from left generator to external power will cause a mild shifting of the flight controls. This may also be accompanied by stick movement. Usually this will be felt as a mild air frame disturbance and should not be cause for concern.

1. Electrical control panel—Check.
   Check electrical control panel for proper position of switches and that the power flow indicator reads TIE.

If Power Flow Indicator Reads TIE Or The Generator Caution Lamp Is Lighted:

2. Emergency generator switch—As required.
   The emergency generator switch should be left in AUTO to prevent a drain on the hydraulic system when flight is being conducted under visual flight conditions. Emergency electrical power will be available within one second if the other primary generator fails. Position the switch ON during instrument flight and prior to descent for night landings.

WARNING

Failure of the ac sensing relay will connect the emergency generator to the essential bus when ON is selected. This failure will be indicated if the flow indicator reads EMER when emergency generator switch is placed to ON, and one engine driven generator is still operative.

Note

If the emergency generator lamp does not light, when the emergency generator switch is placed to ON, depress the emergency generator indicator/cutoff button. If the lamp still does not light, and the bulb is good, the emergency generator is inoperative and there is no back-up for the operative engine driven generator.

Before proceeding to step 3, check the power flow indicator. If it reads TIE, proceed to step 3. If it reads EMER, accomplish the following steps prior to proceeding to step 3.

   a. Emergency generator indicator/cutoff pushbutton—Pulled.
   b. Battery switch—OFF. (Leave OFF for remainder of flight)
   c. Emergency generator switch—AUTO.
   d. Emergency generator indicator/cutoff pushbutton—In.

3. Pitch damper switch—OFF. (Within dampers off region)
   Adjust flight envelope, as necessary, to dampers off region and place pitch damper switch OFF until the malfunctioning generator can be restored to normal operation or until all attempts to reset the generator are completed. Turning the pitch damper OFF will prevent possible flight control transient commands, through the pitch damper, resulting from electrical power surges, during attempts to reset generators.
4. Applicable generator switch—OFF-RESET, START (pause), then RUN.
   Place the generator switch to OFF-RESET then hold to START, pause approximately one second and release the switch to RUN. (Attempts to reset generator may be repeated if desired)

If Power Flow Indicator Reads NORM And The Generator Caution Lamp Is Out:

5. Emergency generator switch—AUTO.
6. Pitch damper switch—DAMPER.

If Power Flow Indicator Reads TIE:

7. Generator switch—OFF-RESET.
8. Decouple pushbutton—Depress.
9. Pitch damper switch—DAMPER.

DOUBLE GENERATOR FAILURE WITH BOTH ENGINES OPERATING.

Double generator failure will not result in a total loss of electrical power for more than the maximum of one second required for the emergency generator to provide power for the essential ac and dc buses. During operation on emergency generator power the airspeed mach indicator, the altitude vertical velocity indicator the angle-of-attack tape (and indexers prior to T.O 1F-111-891) will be inoperative. Yaw and roll trim will be inoperative. Refer to Section I for list of equipment that is powered by the essential buses.

WARNING

- If the pitch damper has been turned OFF prior to loss of power, the switch will return to DAMPER prior to the emergency generator coming on the line.
- Power interruption will cause the auxiliary flight reference system (AFRS) gyros to revert to automatic fast erection. This will be indicated by the auxiliary attitude (AUX ATT) caution lamp lighting, the OFF flag on the standby attitude indicator, and the ADI.
- The angle-of-attack indicator will be inoperative when operating on emergency generator power even though angle-of-attack indications appear normal. The angle-of-attack indexers, however, will be operative. (After T.O. 1F-111-891).

1. Emergency generator switch—ON.
2. Electrical control panel—Check.
   Check electrical control panel for proper position of switches and that the power flow indicator reads EMER.
3. Maintain 1 "g" flight.

WARNING

To assure adequate hydraulic pressure for emergency generator operation, do not open speed brake. Maintain a minimum of 90 percent rpm on both engines while closing speed brake, if it is open, and for wing sweep and landing gear operation.

4. Pitch damper switch—OFF. (Within dampers off region)
   Adjust flight envelope as necessary, to damper off region, and place pitch damper switch OFF until the malfunctioning generator can be restored to normal operation or until all attempts to reset the generator are completed. Turning the pitch damper OFF will prevent possible flight control transient commands through the pitch damper resulting from electrical power surges.
5. Generator switches (individually)—OFF-RESET, START (pause), then RUN.
   Individually place the generator switches to OFF-RESET then to START, pause approximately one second and release switches to RUN.
6. Pitch damper switch—DAMPER.
   For continued operation with one or both engine driven generators or the emergency generator supplying power, place the pitch damper switch to DAMPER.

If Power Flow Indicator Reads EMER:

7. Fuel panel—Check.
   With only the emergency generator providing electrical power, only fuel boost pumps 4 or 5 or transfer pumps 7 and 8 or 12 will be operable. Refer to "Fuel System Operation On Emergency Electrical Power," this section.
8. Land as soon as practicable. Refer to "Fuel System Operation On Emergency Electrical Power," this section.

EMERGENCY GENERATOR OPERATION WITH ONE ENGINE SHUT DOWN.

In the event of a generator failure on the operating engine with the other engine shut down the following should be accomplished.

WARNING

The angle-of-attack indicator will be inoperative when operating on emergency generator power even though angle-of-attack indications appear normal. The angle-of-attack indexers, however, will be operative (aircraft after T.O. 1F-111-891).

1. Emergency generator switch—ON.
2. Establish and maintain nominal 1 "g" flight and an airspeed of 350 KIAS or less. Then, maintain a minimum of 90 percent rpm on the operating engine.
3. Do not open or close speed brake.
4. Sweep wings forward to 26 degrees by moving the wing sweep handle at a smooth rate not to exceed 1 degree of sweep per second.

WARNING

Flight control damper transients may be experienced if hydraulic demands cause an interruption of the emergency generator power.

5. Land as soon as possible using "Single Engine Emergency Generator Landing" procedures, this section.

COMPLETE ELECTRICAL FAILURE.

In the event of complete loss of electrical power the aircraft will be flyable, but should be landed as soon as possible. The following are considerations to be applied as necessary. (1) Airspeed should be maintained within "damper off operating limits" as stability augmentation is not available. (2) Special attention should be given to setting wing sweep for landing to compensate for a possible aft cg condition as fuel is available from the forward tank (suction feed) only and the cg will shift aft as fuel is consumed. (3) If the wing sweep handle is moved to the 16 degree detent, the handle will lock at that position and the wing cannot be moved aft. The wing sweep position indicator will be inoperative. (4) Slats and flaps can be extended using normal extension procedures. The flap/slat position indicator is inoperative. (5) The landing gear must be extended using emergency extension procedures. No gear down indication will be available. (6) Utility lights only will be operational if battery power is available. (7) Tachometers and the standby altimeter and airspeed indicator will be operational; all other flight instruments are inoperative. (8) Estimate fuel consumption as closely as possible to aid in setting wing sweep for landing. (9) Radio communication may be attempted with the emergency radio contained in the quick rescue kit.

GLASS PANEL FAILURE OR UNLOCKED CANOPY INDICATION.

Loss of a glass panel, windshield, and/or canopy will not, of itself, cause the aircraft to become uncontrollable or unstable. However, such a failure may result in conditions whereby one or both crew members may be incapacitated and aircraft control degraded. Conditions that can be expected to occur instantaneously include severe wind blast, unbearable noise levels, loss of intercom and radio communications, limited visibility and possible personal injury, and/or aircraft damage due to flying debris. Should windshield/glass panel failure occur, the first aircrew reaction should be directed at maintaining control of the aircraft. Either crew member that is capable of maintaining control should take control of the aircraft, level the wings, reduce airspeed as required, and climb/descend to a safe flight level. If aircraft control cannot be achieved or maintained, initiate ejection procedures. If time and conditions permit, the following actions should be taken.

1. Visors—Down.
2. Oxygen mask and fitting—On, 100 percent.
3. Canopy latch handle—Check locked.
4. Obtain a safe altitude and airspeed.
5. Pressurization selector switch—COMBAT.
6. Land as soon as practicable.

HYDRAULIC SYSTEM FAILURE.

PRIMARY OR UTILITY HYDRAULIC SYSTEM FAILURE.

Failure of either hydraulic system will cause the pitch, roll, and yaw damper caution lamps and the hydraulic low pressure caution lamps to light. If the failure occurs with the flight control system in the cruise mode, rudder authority will revert to full, causing the rudder authority caution lamp to remain lighted until the slats are extended or the control system switch is placed to T.O. & LAND. The damper servo-actuators will operate as non-redundant servos. As the hydraulic pressure drops and the damper caution lamps light, forces may be felt in the control stick. Loss of either hydraulic system will result in the loss of automatic control and normal operation of all hydraulically operated components except flight controls and wing sweep. The emergency generator, nosewheel steering, speed brake and weapons bay gun will be completely inoperative. Back-up systems are provided to operate the spikes, landing gear extension, flaps and slats, wheel brakes, air refueling system and weapons bay doors.

1. Wing sweep handle—26 degrees.
   Maintain wing sweep position compatible with airspeed and sweep wings to 26 degrees when at appropriate airspeed. Minimize flight control movement during wing sweep.

CAUTION

Maintain nominal 1 "g" flight while changing wing position. Change wing sweep position by moving the wing sweep handle at a smooth rate not to exceed 1 degree of sweep per second to avoid depleting hydraulic pressure.

Note

If supersonic and wings are aft of 50 degrees, retard throttles and sweep wings forward slowly to 50 degrees to enhance deceleration. When reaching subsonic speeds sweep wings forward slowly to 26 degrees.

2. Maintain airspeed within the damper off operating limits.
3. Depress the damper reset button only if the affected system pressure returns to normal.
4. Land as soon as possible using "Primary Or Utility Hydraulic Failure Landing," this section.

COMPLETE HYDRAULIC SYSTEM FAILURE.

1. Eject.

WARNING

If both hydraulic systems fail during flight, the flight control system will be inoperative and flight cannot be continued.

LANDING GEAR GROUND SAFETY (SQUAT) SWITCH FAILURE.

If these switches fail in the closed position, the first indication to the crew will normally be that the landing gear handle will not move out of the DOWN position without first depressing the landing gear handle lock release button. If these switches fail in the closed position, these conditions will exist:

- The landing gear handle lock will hold the handle DOWN (depressing button will allow gear retraction).
- Ground roll spoilers remain armed even with gear retracted.
- AYC will be inoperative.
- Weapons firing will be inoperative.
- Cowl anti-icing will be inoperative.
- Artificial stall warning will be inoperative.
- The secondary alpha/beta probe heater will be inoperative.
- The engine nozzles will open if the respective throttle is within 3 degrees of IDLE.
- Anti-skid touchdown control will be inoperative.
- Vortex destroyers will continue to operate during flight.
- Nose wheel steering will remain operational during flight.
- Ejector air hydraulic and engine oil cooling will continue to operate during flight.
- Flight control ground test panel will remain armed.

If A Malfunction Of The Landing Gear Ground Safety (Squat) Switches Is Suspected, Proceed As Follows:

1. Landing gear safety release circuit breaker—Pull.
   This should return all squat switch functions to normal operation, except that AYC channel 2 is disabled (clean configuration) while channels 1 and 3 are operational; the yaw channel lamp may light; and stall warning above 17 degrees is disabled.

As Additional Precautions, Accomplish The Following:

2. Ground roll spoiler switch—OFF.
3. Pitot/probe heater switch—HEAT. (If required)
4. Do not retard either throttle within 3 degrees of IDLE until landing is accomplished.
5. Do not actuate flight control ground test switches.
6. Monitor flight conditions carefully as artificial stall warning may be inoperative.

If Mission Requirements Dictate, The Aircraft May Be Flown In This Configuration And The Mission Completed. For Landing, Continue As Follows:

7. Assume that AYC is inoperative. Fly a straight-in approach avoiding abrupt control inputs; do not exceed 60 degree bank.
8. Do not apply brakes until firmly on the runway.

After Touchdown:

9. Throttles—IDLE.
10. Landing gear safety release circuit breaker—In.

Note

Nose wheel steering will be inoperative until this circuit breaker is pushed in.

11. Ground roll spoiler switch—BRAKE.
12. Normal "After Landing" checklists—Complete.

OIL SYSTEM MALFUNCTIONS.

An oil system malfunction on either engine is recognized by a change in oil pressure, a complete loss of oil pressure, or excessive oil temperature. In general, it is advisable to shut down the engine 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 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 BETWEEN 30 AND 40 PSI. (EXCEPT AT IDLE)

1. Throttle of affected engine—IDLE.
2. Monitor oil pressure.

OIL PRESSURE ABOVE 50 PSI.

1. Throttle of affected engine—Retard.
   Reduce thrust on affected engine. If oil pressure can be maintained below 50 psi continue to operate engine at the reduced power setting. If oil pressure cannot be reduced below 50 psi, shut down the engine.

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 will most probably be evidenced by inlet buzz and/or compressor stall.

1. Airspeed—Reduce until inlet buzz or compressor stall disappears.

SMOKE AND FUME ELIMINATION.

1. **OXYGEN—100 PERCENT.**
2. Air source selector knob—L. ENG, R. ENG.
   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 to next step:
3. Air source selector knob—BOTH.

Attempt To Isolate Source Of Smoke Or Fumes As Follows:

4. Non-essential electrical equipment—OFF.

Note

If condition still exists, do not accomplish step 5, proceed to step 6.

5. Electrical equipment—ON. (As required)
   Turn on electrical and lighting equipment one system at a time, and check for smoke until source is determined.
6. If smoke persists, position the air source selector knob to OFF.
7. If smoke or fumes persist, position the air source selector knob to RAM/EMER.

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.

SPEED BRAKE MALFUNCTIONS.

Speed Brake Fails To Retract:

1. Speed brake hydraulic valve circuit breaker—Pull out.
2. Before extending gear—Push circuit breaker in.

OUT-OF-CONTROL RECOVERY PROCEDURES.

Detailed out-of-control characteristics are described in Section VI under "Stall/Loss of Control Characteristics." In general, stalls, post stall gyrations and spins are the result of exceeding the angle-of-attack limits in Section V and recovery is achieved by reducing angle-of-attack to within limits.

WARNING

Near the angle-of-attack limits, induced drag increases rapidly and may cause the total drag to exceed the total thrust available. This results in rapid airspeed decrease and/or increase in sink rate. The resulting increase in angle-of-attack may be sufficient to cause overshoot of the angle-of-attack limits and subsequent departure from controlled flight.

Conventional aerodynamic stall warning such as a sudden "g" break, stick force changes or other pronounced cues are not available to warn the pilot of impending departure from controlled flight. The departure will occur as a smooth but uncommanded yawing and rolling motion. Unless the pilot monitors angle of attack and observes artificial stall warning, aircraft control may be lost.

WARNING

The command augmentation feature of the flight control system will attempt to maintain the stick commanded level of pitch rate, "g" force, and roll rate independent of airspeed variations. For instance, during flight conditions where airspeed is decreasing, the horizontal stabilizer will be commanded to increase angle-of-attack without additional pilot input. Unless the pilot is monitoring angle-of-attack increase, airspeed decrease, and/or artificial stall warning indications, the flight control system will allow the aircraft to achieve an out-of-control condition.

1. **STICK—FULL FORWARD AND CENTERED.**
2. **RUDDER—NEUTRAL.**
3. **ROLL DAMPER SWITCH—OFF.**

WARNING

Action of the roll damper after departure can delay recovery of aircraft control.

4. Throttles—As required.
   If in afterburner, reduce power to MIL. If below afterburner range, do not reposition throttles. To do so may result in engine stalls.

WARNING

- Hold recovery controls until all significant angular motions have damped, the nose is well below the horizon, airspeed is above 200 KIAS and increasing, and angle-of-attack is below 15 degrees and decreasing. Care must be taken to assure that airspeed is increasing and angle-of-attack is decreasing. Erroneous values may be occasionally presented on the angle of attack indicator when the aircraft angle-of-attack is above the indicator limit (25 degrees).
- Engine stall and resulting loss of hydraulic power may occur in an out-of-control condition. To conserve hydraulic power, do not change aircraft configuration (flaps, wing sweep, etc.). If engine rpm drops below 35 percent on both engines, hydraulic power may be insufficient for recovery.
- During recovery from out-of-control conditions, it may be necessary to obtain an air speed of as much as 300 KIAS to recover a stalled engine.
- High descent rates (up to 20,000 feet per minute) may exist during out-of-control maneuvers.
- If aircraft control has not been recovered at an altitude of 15,000 feet above the terrain —Eject.

If Recovery Is Effected:

5. Roll damper switch—DAMPER.

SPIN RECOVERY PROCEDURE.

If the out-of-control recovery procedure has not produced a recovery, the aircraft is probably spinning. A spin is indicated by an angle of attack above stall angle of attack (indicator generally at 25 or minus 2 to 3 degrees), low airspeed (140 KIAS or lower), and turn needle fully displaced in the direction of spin rotation. If these conditions are confirmed, perform the following recovery steps:

1. Stick—Forward and full with turn needle. (Pitch and roll control centered, if inverted)

WARNING

- Both full lateral control and forward stick are required to effect spin recovery. In order to obtain full lateral control deflection, less than full forward stick will have to be held.
- Jettisoning of external stores can result in store/aircraft collision.

2. Rudder authority switch—FULL.
3. Rudder—Full oposite turn needle.

When Aircraft Rotation Stops:

4. Stick—Forward and centered. (Pitch and roll control centered, if inverted).
5. Rudder—Neutral.
6. Roll damper switch—DAMPER.

WARNING

- Immediately after rotation stops, the aircraft will unload and negative "g's" may be encountered. This unloading is to be expected during recovery and can be moderated by reducing forward stick deflection. Longitudinal oscillations may continue for several cycles after the aircraft unloads and no attempt should be made to counter these oscillations. During the recovery process, the aircraft will be in a near vertical attitude and external visual cues may be confusing. Continual monitoring of airspeed, angle-of-attack, and altitude is mandatory. Hold recovery controls until the angle-of-attack remains below 15 degrees and airspeed continues to increase. During this period, residual pitch oscillations and a slow roll may exist even though the aircraft has recovered and is flyable. The aircraft pitch response will track with the control stick when the aircraft is positively under control. If positive control has not been attained by 15,000 feet above the ground, eject.

- Do not exceed 15 degrees angle-of-attack during recovery.

- During recovery from either a poststall gyration or spin, the aircraft may occasionally encounter a series of uncommanded, rapid rolls near or below the angle of attack limit due to inertia coupling. These rolls may demonstrate roll rates as high as 180 degrees per second, even though roll control is centered. The key to recognizing this situation is building airspeed above 200 KIAS and an angle-of-attack below stall. If maintaining full forward stick does not produce a rapid reduction of this high roll rate condition, neutralize all controls, and roll rate should begin to decrease immediately. Although uncommanded rolling will continue for 1 to 2 turns, recovery should be complete in 5 to 10 seconds. If uncommanded roll rate has not subsided within 5 to 10 seconds, rudder should be applied opposite the roll direction. As the roll rate slows near zero, ease the stick forward to further reduce angle of attack and neutralize rudder.

- If lateral control with the spin direction is maintained even though the yaw rotation has stopped and angle-of-attack has reduced to a value below stall, the aircraft will begin to roll because of the lateral command and building airspeed. The pilot may mistake this rolling motion as a continuation of the spin and incorrectly hold in full aileron. The key to recognizing when to neutralize these controls is a building airspeed above 200 KIAS and an angle of attack well below stall. Failure to neutralize roll and yaw controls will result in an excessive altitude loss after the initial out-of-control condition has been corrected.

7. Throttles—As required.
8. Air start button—Depress.
9. After controlled flight and normal engine operation are restored, the wings, if aft of 45 degrees, should be swept forward to minimize altitude loss and excessive speed buildup.
10. If still out of control by 15,000 feet above ground level—Eject.

THROTTLE MALFUNCTIONS.

EXCESSIVE THROTTLE FRICTION.

When high throttle friction or unsteady (jerky) throttle movement is encountered, an impending throttle binding problem may be indicated. If this condition occurs, move the affected throttle to maintain approximately 80 percent rpm and use the other throttle to control airspeed.

FROZEN THROTTLE IN AFTERBURNER RANGE.

1. Relax force on throttle for 20-30 seconds.
2. Attempt slight throttle advance; then apply force to retard throttle to 80 percent rpm.

ENGINE SHUTDOWN WITH A FROZEN THROTTLE.

1. Engine fire pushbutton—Depress.

Note

- Engine may be shut down from any power setting.
- If engine is shut down in flight with the fire pushbutton, refer to "Single Engine Landing" and "Single Engine Go-Around," this section.
- If throttle is frozen above the IDLE position, ground roll spoilers will not be available on landing roll.

WHEEL WELL OVERHEAT DURING FLIGHT.

The most probable cause of a wheel well overheat condition would be a ruptured engine bleed air duct. The detection system will indicate a hot condition and light the wheel well hot caution lamp. A fire condition may exist, and as it progresses will probably be verified by loss or degradation of the hydraulic and electrical systems and/or a smoke trail. Note that the corrective crew action includes shutting off the engine bleed air source; therefore, equipment cooling and cabin pressure will not be available. Airspeed should be reduced to achieve favorable conditions for emergency ram air cooling and no cabin pressure. After the hot light goes out, a visual inspection should be made of the wheel well and surrounding area (by chase aircraft or tower fly-by) and the aircraft should be landed as soon as practicable. If the wheel well hot caution lamp lights, proceed as follows:

1. Air source selector knob—OFF or EMER. (As applicable)
   For supersonic flight with high total temperature indications, place the selector knob to OFF and decelerate and descend to decrease total temperature indication and establish an altitude where cabin pressure is not required. If total temperature and/or altitude is not a consideration, place the knob to EMER and descend as outlined above. (Refer to "Ram or Emer Mode Flight Limits," Section V.)

Note

Placing the air source selector knob to OFF shuts off engine bleed air but does not dump cabin pressure. The EMER position of the knob shuts off bleed air, opens the ram air scoop and dumps cabin pressure. With high total temperature indications, opening the ram air scoop will result in excessive cabin temperature.

2. If lamp does not go out 10 seconds after placing the air source selector knob to OFF or EMER, open the speed brake door and when airspeed permits, extend the landing gear.
3. Land as soon as practicable.

LANDING EMERGENCIES

APPROACH END BARRIER ENGAGEMENT.

Approach end arrestments are considered practicable and should be attempted when directional control and stopping distance are questionable and when a malfunction presents a threat to directional control and there is sufficient runway in front of the barrier on which to land and lower the nose wheel prior to barrier contact. Consideration should also be given to the engaging speed limits to prevent structural failure of the arresting barrier or the aircraft.

Note

Fly a straight-in approach when possible to insure an accurate touchdown point on the runway. Considerations should also be given to actions taken if engagement is missed; i.e., go-around and barrier engagement on the other end of the runway.

1. Reduce gross weight.
   Time permitting, dump and/or burn fuel to reduce gross weight as low as practicable.
2. Normal Procedures Checklists—Complete.

3. Throttle friction—Reasonably tight.
   Deceleration forces could cause throttles to be thrown forward if not tight.
4. Arresting hook—Extend, check hook lamp lighted.
   If time permits, extend hook where cover may be recovered and will not cause injury to persons on the ground.
5. Shoulder harness—Locked.
6. Touch down in center of runway at least 400 feet short of cable without landing flare.
7. Lower nose immediately.
   Do not make an attempt to steer aircraft to center of barrier. Off center engagement may cause the aircraft to veer off course to the off-center side of the runway. Barrier contact should be made with nosewheel steering disengaged. No attempt to correct yaw or roll tendencies during the arrestment should be made until the aircraft is slowed sufficiently to insure aircraft control.
8. Throttles—IDLE.
   Reduce power to idle at touchdown to insure spoiler brake operation.
9. Engage barrier with brakes off.
   Roll back may occur depending on type of barrier used. Roll back will be parallel to runway center line for either on center or off center engagements. Light braking should be applied at the end of the arrestment when possible to minimize rollback without causing the aircraft to pitch up.
10. Keep engines running until crash crews arrive and signal for engine shutdown.
11. If emergency evacuation is required, pull the auxiliary brake handle, shut down engines and abandon the aircraft.

WARNING

During emergency engine shutdown for evacuation some fuel will be dumped overboard in the proximity of the main wheel area and could cause a fire hazard.

BLOWN TIRE LANDING.

If barrier engagement is to be made, refer to "Approach End Barrier Engagement," this section.

MAIN GEAR TIRE.

1. Normal Procedures Checklists—Complete.
2. Fly a straight-in approach.
3. Anti-skid switch—ON.
4. Touch down on side of the runway opposite the blown tire.
5. Lower nose and use nose wheel steering and differential braking as required to keep the aircraft on the runway. The brake on the good tire should be used normally. Do not lock the brake on the wheel with blown tire.

NOSE GEAR TIRE.

1. Normal Procedures Checklists—Complete.
2. Ground roll spoiler switch—OFF.
3. Fly a straight-in approach.
4. Delay lowering nose to runway until just prior to losing flight control effectiveness. Then, apply aft stick for aerodynamic braking effect but keep nose wheel on runway.
5. Use differential braking as required for directional control.

DAMPERS OFF LANDING.

If a landing is to be made with any or all of the dampers OFF proceed as follows:

1. Land using a straight-in approach.
   With any single damper failure, fly a long, straight-in approach at 10 degrees angle-of-attack. Aircraft handling qualities associated with multiple damper failure may be improved by flying a higher than normal approach speed corresponding to 8.5 degrees angle-of-attack.
2. Avoid large or abrupt control inputs.
   Crosswind landings with the pitch or roll dampers inoperative require no special considerations or techniques other than observing those limitations specified under "Flight With Dampers Off," Section V. However, a crosswind landing with the yaw damper inoperative, especially under gusty wind conditions, requires special techniques and considerations. It is recommended that the pilot establish a crabbed drift correction on the final approach. Do not attempt to assume a wing-low drift correction during the transition and touchdown phase. Instead, maintain the required crab drift correction through touchdown, not to exceed 10 degrees yaw or crab angle. In addition, minimize yaw inputs or corrections on final approach, especially during the transition phase just prior to touchdown. Because the air craft has low directional damping in this configuration, rudder inputs to correct for yaw variations resulting from gusts or lateral control inputs should be kept small to avoid yaw overshoot in the opposite direction.

DITCHING.

It is recommended that ejection be accomplished rather than ditching. If ditching is unavoidable, proceed as follows:

1. Fuel dump—As required.
2. Oxygen—100 percent.
3. Flaps/slats—Extended.
4. Landing gear—Retracted.
5. Approach at an angle-of-attack of 12 degrees as indicated on the angle-of-attack indicator.
6. Adjust power to maintain angle-of-attack of 12 degrees with minimum sink rate. (Not to exceed 200 feet per minute)
7. Hold constant angle-of-attack and do not flare the aircraft before touchdown.

Upon Water Contact:

8. Stick—Neutral.
9. Throttles—OFF.
10. Pull severance and flotation handle.

Should crew module structure rupture or canopy transparency break during the course of ditching and severance, cabin flooding beyond the capability of the bilge pump may result. Proceed as follows:

11. Continue to wear oxygen mask.

Note

Emergency oxygen will be automatically supplied when SEVERANCE & FLOTATION handle is actuated (manual actuation is also possible by means of EMERGENCY OXYGEN handle).

12. Auxiliary flotation handle—Pull.
13. Insert safety pins into ejection handles.

WARNING

Pulling the severance and flotation handle and the auxiliary flotation handle will sever the module from the aircraft bypassing the rocket motor. The rocket will fire if the ejection handle is pulled or accidentally activated.

Note

Pulling the auxiliary flotation handle will cause even a flooded crew module to float with sufficient freeboard to open canopy hatches.

LANDING WITH ABNORMAL FUEL DISTRIBUTION.

The proper center-of-gravity for landing depends upon correct fuel distribution. The following procedures are used for landing with conditions of abnormal fuel distribution.

LANDING WITH AFT ABNORMAL FUEL DISTRIBUTION.

1. Between 250 and 300 KIAS in steady flight, sweep wings forward until 26 degrees wing sweep or until the stabilizer indication is 1 degree down elevator, whichever is reached first. Do not confuse with 16/26 degree wing sweep procedure under "Normal Landing", Section II.

If 26 Degrees Wing Sweep Is Reached And 1 Degree Down Not Exceeded:

2. Flaps/slats—Extend.
3. Make a normal landing using a straight-in approach.

WARNING

Do not sweep wings forward of 24 degrees in attempting to extend flaps. Do not make a landing approach with wings forward of 26 degrees and/or average elevator positions more than 1 degree down, as sufficient aircraft nose down elevator authority may not be available to maintain control of the aircraft. Increased attention to airspeed and angle-of-attack control will be required since the aircraft is statically unstable at elevator deflections greater than 1 degree trailing edge down.

If 1 Degree Down Is Reached Prior To 26 Degrees Wing Sweep:

4. Maintain wing sweep obtained at 1 degree down elevator.
5. Perform landing in accordance with "No Flap Landing" procedures, this section.

LANDING WITH FORWARD ABNORMAL FUEL DISTRIBUTION.

If 16 Degree Wing Sweep Is Reached, And Elevator Position Is Greater Than 2 Degrees Up:

1. Establish landing configuration.
2. Slow the aircraft until 10.5 degrees trailing edge up or until final approach angle-of-attack is reached.
3. Do not allow airspeed to decrease below this value until after touchdown.

LANDING WITH FLAP AND SLAT MALFUNCTIONS.

If conditions require landing with flap or slat malfunction, factors such as gross weight, approach speed, ground roll distance and runway condition must be considered. Diversion to a suitable alternate or approach and barrier engagement may be necessary. If flap position can be confirmed to be full up, landing should be made utilizing "No Flap Landing" procedures provided slats are symmetrical. If flaps can be confirmed to be full up but slats are asymmetrical, use "Landing With Asymmetric Slats," this section. If slats cannot be verified by gain changer lamps or visually to be approximately 70 percent down and flap position is confirmed or suspected to be other than full up, land using "No (Or Partial) Slats With Partial Flaps Landing," this section.

WARNING

- If flaps are confirmed or suspected to be other than full up and the slats cannot be verified by gain changer lamps or visually to be approximately 70 percent down, landing should be made utilizing "No (Or Partial) Slats With Partial Flaps Landing", this section.
- Flap position indicator is unreliable if malfunction is due to a slat/flap sequencing mechanism failure.

CAUTION

Placing the emergency flap/slat switch to the emergency position, relieves the hydraulic pressure to the flap motor and isolates the function of the flap handle and may prevent further damage.

FLAPS AND SLATS EMERGENCY EXTENSION.

1. Reduce airspeed to 180 KIAS or 10 degrees angle-of-attack, whichever is higher.
2. Flap/slat switch—EMER.
3. Emergency flap/slat switch—EXTEND and hold. (Emergency extension requires up to 60 seconds or more)

WARNING

- Prior to T.O. 1F-111-824, emergency flap extension with M-117 weapons installed on pylons 3 or 6 or fuel tanks installed on pylons 4 or 5 can result in interference if 34 degree extension is exceeded.
- Make a positive check that all slats are extended by visual observation of the slat position indicator and the slats themselves, prior to proceeding with flap extension. Flap extension without prior slat extension or with asymmetric slat extension can result in a mild to uncontrollable pitchup, stall, and rolloff depending on the magnitude of flap and slat deflections.

4. Flap/slat handle—DOWN.

LANDING WITH ASYMMETRIC SLAT.

1. Do not extend flaps.
2. Fuel dump—As required.
3. External stores—Jettison. (If required)
4. Normal Procedures Checklists—Complete.
5. Control system switch—T.O. & LAND. (Below 300 KIAS or mach 0.45, whichever is less)

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system switch in T.O. & LAND can result in loss of control of the aircraft.

6. Fly long, shallow straight-in approach at 11 degrees angle-of-attack. (On final approach maintain ground track with rudder trim and lateral control)

WARNING

- Desired rate of descent should be established at beginning of approach and abrupt maneuvers, large throttle motions or flight in excess of 1 "g" should be avoided.
- Maintain an airspeed compatible with aircraft configuration and gross weight to insure that 10 degrees angle of attack is not exceeded during maneuvering flight prior to final approach phase.

Note

Maintain constant ground track by use of rudder trim and lateral control. Rudder trim up to full authority should be used to reduce lateral control requirements.

7. Landing.

Full pedal deflection anti-skid braking with control stick full aft and centered will give the most effective deceleration for both dry and wet runways. If ground roll distance is a consideration, utilize the "Short Field Landing" procedure, Section II. Refer to figure 3-3 for approach speeds and landing roll distance. If stopping distance is not critical, utilize normal landing procedures, Section II.

WARNING

Aircraft will tend to veer in the direction of the extended slat upon touchdown if the lateral control is centered or the spoiler brakes extended. Lateral control, augmented with rudder as necessary, should be initiated upon touchdown to maintain desired ground track.

8. Hook—As required.

Note

If a strong crosswind exists or directional control is difficult on the landing roll, turn ground roll spoiler switch OFF.

WARNING

- If excessive braking is used at high speeds, the wheel blowout plugs may relieve tire pressure within 15 minutes after stop. Provisions should be made to cope with wheel fires which may start shortly after the blowout plugs relieve.
- Call the fire department after any emergency landing which results in hot wheels or brakes or use of the tail hook. Do not shut down the engines until after the fire trucks arrive. Fuel venting from the engines after shutdown may be ignited by the affected hot part.

NO FLAP LANDING.

Landing with no flaps or with slats only should be made from a long, shallow, straight-in approach. Excessive shallow approaches make judgment of the touchdown point more difficult and may result in a soft landing which will not close the squat switches for immediate spoiler extension. Excessively steep approaches require a low power setting which increases engine acceleration time. Refer to figure 3-3 for approach speed and angle-of-attack. (Approach speeds obtained from chart must be adjusted if carrying stores: Add 1 KIAS for each 1000 pounds in weapons bay; add 0.5 KIAS for each pylon station carrying weapons or tanks.) Approach angles-of-attack should be established by use of the angle-of-attack indicator on the AMI. Do not use the angle-of-attack indexer. For wing sweeps between 16 and 45 degrees, approach at 11 degrees angle-of-attack. For wing sweeps aft of 45 degrees, approach at 12 degrees angle-of-attack. Under these approach conditions, care should be exercised to avoid tail strikes at touchdown. Landings with wing sweeps greater than 45 degrees can be made with up to 20 knots of crosswind but roll response will be reduced to less than half due to spoiler lock-out. The stability augmentation system must be operating for crosswind landings with wing sweep greater than 45 degrees.

WARNING

- Desired rate of descent should be established at beginning of approach and abrupt maneuvers, large throttle motions or flight in excess of 1 "g" should be avoided. Any of the above can result in excessive sink rate build-up which may be difficult to arrest at approach altitudes.
- Maintain an airspeed compatible with aircraft configuration and gross weight to insure that 10 degrees angle-of-attack is not exceeded during maneuvering flight prior to final approach phase.

1. Fuel dump—As required.

Because of the high approach and touchdown airspeeds required during landing with wings at 26 degrees or greater and no flaps, burn or dump as much fuel as practicable prior to entering traffic pattern.

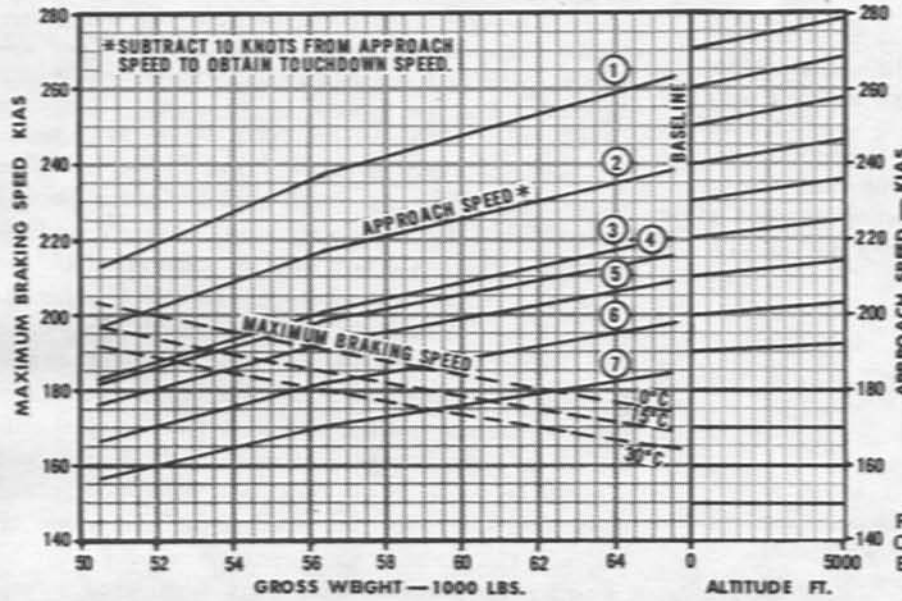
Emergency Landing Airspeeds and Ground Roll Distances

CONDITIONS:
- C.G. = AUTO ENGINE FEED
- MAXIMUM BACK STICK DURING LANDING ROLL
- NO GROUND ROLL SPOILERS

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONFIGURATION:
SLATS RETRACTED
FLAPS RETRACTED

DATA BASIS: ESTIMATED
DATE: 19 MAY 1972

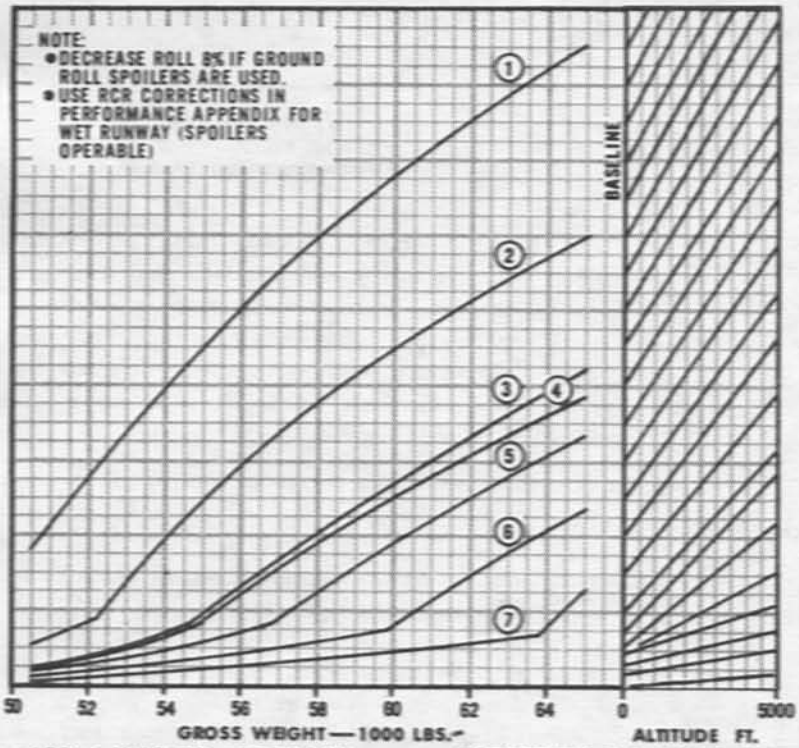


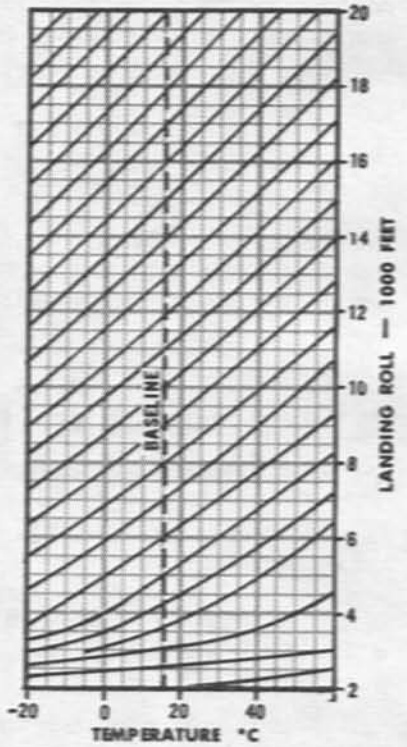
WING SWEEP		ANGLE OF ATTACK
1	72.5°	12°
2	60°	12°
3	50°	12°
4	40°	11°
5	35°	11°
6	26°	11°
7	16°	11°

NOTE:
DECREASE MAXIMUM BRAKING SPEED 4 KNOTS/1000 FEET ALTITUDE.

WARNING

REFER TO SECTION V FOR CENTER OF GRAVITY LIMITS AND BRAKE ENERGY LIMITS.





NOTE:
- ADD 1 KIAS TO APPROACH SPEED FOR EACH 1000 POUNDS IN BAY.
- ADD 0.5 KIAS FOR EACH PYLON STATION WITH WEAPONS OR TANKS PRESENT.
- STOPPING DISTANCE GIVEN CONSIDERS AERODYNAMIC DECELERATION TO MAX. BRAKE APPLICATION SPEED AND THEN APPLICATION OF MAX WHEEL BRAKES

A0000000-E081C

Figure 3-3.

WARNING

If landing is to be accomplished due to abnormal fuel distribution and fuel feed from aft tank cannot be confirmed, do not dump fuel.

2. External stores—Jettison. (If required)
3. Normal Procedures Checklists—Complete.
   a. Compute emergency landing airspeed from figure 3-3.
4. Flight control disconnect switch—OVRD. (When aft of 26 degrees wing sweep)

   Placing the flight control disconnect switch to OVRD deactivates adverse yaw compensation, auxiliary pitch trim, roll trim, TFR climb/dive commands, autopilot and pedal shaker.

Note

Once the flight control disconnect switch is placed to OVRD the pitch and roll gain changer lamps will remain on even though the control system switch is subsequently placed to T.O. & LAND, because AYC is not activated.

5. Control system switch—T.O. & LAND. (Below 300 KIAS or mach 0.45, whichever is less)

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system switch in T.O. & LAND, can result in loss of control of the aircraft.

6. Landing.

   Full pedal deflection anti-skid braking with control stick full aft and centered will give the most effective deceleration for both dry and wet runways. If ground roll distance is a consideration, utilize the "Short Field Landing" procedure, Section II. Refer to figure 3-3 for approach speeds and landing roll distance. If stopping distance is not critical, utilize "Normal Landing" procedures, Section II.

Note

- Ground roll spoilers will not be available at wing sweep angles of 45 degrees or greater.
- If possible, sweep wings forward to obtain ground roll spoiler operation.

7. Hook—As required.

WARNING

If excessive braking is used, the wheel blowout plugs may blow out, relieving tire pressure within 15 minutes after stopping. Do not shut down the engines until the fire trucks arrive as fuel venting from the engines after shutdown may result in a fire.

NO (OR PARTIAL) SLATS WITH PARTIAL FLAPS LANDING.

Landing with no (or partial) slats with partial flaps will require a long, shallow, and straight-in approach. Approach angle-of-attack should be 7 degrees and should be established by use of the angle-of-attack indicator on the airspeed mach indicator. Do not use the angle-of-attack indexer. Landing can be accomplished at 16 to 26 degrees wing sweep depending on center-of-gravity considerations. Approach speed will be greater than normal full flap landing speeds by approximately 20 percent at 16 degrees and 30 percent at 26 degrees, based on no slat extension and 15 degrees flaps extension. Speeds will vary with actual configuration, however, angle-of-attack is the primary speed reference for approach and landing. Figure 3-3 may be used to approximate ground roll distance.

WARNING

- Desired rate of descent should be established at the beginning of the approach and abrupt maneuvers, large throttle motions or flight in excess of 1 "g" should be avoided. Any of the above can result in excessive angle-of-attack build up (above 12 degrees) which could result in stall or uncontrolled roll off.
- Maintain an airspeed compatible with aircraft configuration and gross weight to insure that 7 degrees angle of attack is not exceeded during maneuvering flight prior to final approach.

1. Wing sweep—Set as required.
   At 7 degrees angle-of-attack in stabilized flight, check the elevator position. If the elevator position is between 7 degrees up and 0 degrees, maintain the existing wing sweep. If the elevator position is greater than 7 degrees up, the wings should be swept forward or fuel dumped until the elevator position is between 7 degrees up and 0 degrees. If the elevator position is below 0 degrees, sweep wings aft (limit 26 degrees) until elevator position is between 0 and 7 degrees up. If elevator position is below 0 degrees with wing sweep at 26 degrees, refer to "Abnormal Fuel Distribution" this section.
2. Fuel dump—As required.
   Fuel may be dumped as required to reduce landing speeds.
   Fuel should be dumped at 7 degrees angle-of-attack in stabilized flight and the elevator position monitored to assure that the elevator position is within the limits of step 1.
3. External stores jettison button—Depress. (If required)

Note

Wing sweep may require readjustment after fuel is dumped or external stores jettisoned.

4. Normal Procedures Checklists—Complete.
   Wing sweep for landing is to be determined in accordance with step 1 of this procedure.
5. Control system switch—T.O. & LAND. (Below 300 KIAS or mach 0.45, whichever is less)

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system switch in T.O. & LAND, can result in loss of control of the aircraft.

6. Approach angle-of-attack—7 degrees.
7. Land as soon as praticable.

LANDING WITH ASYMMETRIC STORES.

If a large or heavy store asymmetry exists, landing can be accomplished using a 10 degree angle-of-attack approach. Landing configurations and approach speed should be established with sufficient altitude remaining to determine specific flying qualities prior to the final approach. A straight-in approach is recommended making full use of roll and rudder trim to establish an acceptable balance of lateral control force and angle of sideslip. As speed is decreased, the lateral trim required may exceed roll damper authority of ±6 degrees. Lateral control forces may be reduced through the use of rudder trim and/or by increasing final approach speed to obtain 8.5 degrees angle-of-attack. If a significant cross wind exists, land with the heavy wing up-wind if conditions permit. Using rudder to align the aircraft with the runway centerline may require full lateral control to hold the wings level. Do not exceed 360 feet per minute sink rate at touchdown. Normal braking technique may be used during the landing roll.

WARNING

As speed is decreased or load factors increased, the asymmetric effects become more pronounced.

HYDRAULIC SYSTEM FAILURE LANDING.

PRIMARY OR UTILITY HYDRAULIC FAILURE LANDING.

An approach end barrier engagement should be considered. Fly an extended downwind leg sufficiently long to provide time for lowering the landing gear and flaps by the emergency method. After touchdown, braking will be available until the brake accumulator pressure has been reduced to approximately 1100 psi (After approximately 10-14 full brake applications). Brake pedals will move to the fully deflected position as the accumulators deplete. To minimize consumption of brake accumulator hydraulic fluid, braking should be accomplished with as few brake applications as possible. A single moderate and steady brake application should be applied at the lowest speed practical to stop on the available runway. If the accumulator pressure has been reduced to less than 1100 psi, normal braking will not be available. If the barrier cannot be engaged, it will be necessary to pull the auxiliary brake handle to stop the aircraft. Only one set of spoilers will be available. If the runway is wet or icy, application of brakes at high speeds would result in anti-skid cycling, and depletion of the hydraulic accumulators, thereby losing the use of normal brakes for directional control and stopping. Dampers will not reset unless pressure returns to normal.

1. Anti-skid switch—OFF.
2. Spike control switches—OVERRIDE. (Below 0.9 mach)
3. Wing sweep—Adjust for landing.
4. Emergency extension of flaps/slats.

5. Emergency extension of landing gear.
6. Utilize aerodynamic braking and maintain directional control with the rudder as long as effective.
7. Hook—Extend. (If required)
8. Auxiliary brake handle—Pull. (If required)

CAUTION

With the auxiliary brake handle pulled, the brakes will lock under some conditions, such as a wet or icy runway. On a dry runway the brakes should not lock.

9. Stop straight ahead on runway.

LANDING GEAR MALFUNCTIONS.

LANDING AFTER NOSE LANDING GEAR RETRACTION FAILURE.

Failure of the nose landing gear to achieve a proper up and locked indication after landing gear UP selection may be caused by malsequence between the nose gear and uplock mechanism. Such malsequence can result in damage to the nose wheel steering linkage. To avoid directional control difficulty during the landing rollout after nose landing gear retraction failure has been confirmed:

1. Landing gear handle—DN.

Note

Do not recycle gear handle.

2. Landing gear emergency release landing—Pull.
   This will remove all hydraulic pressure to the nose steering system and will allow the nose gear to align with the runway.
3. Consider barrier engagement. Refer to "Approach End Barrier Engagement," this section.

UNSAFE GEAR INDICATION.

Landing gear unsafe (not down and locked) is indicated by either or both green landing gear position indicating lamps not being lighted after gear down selection. However, failure of one or both landing gear position indicator lamps to light, together with failure of the landing gear handle warning lamp to light after the gear has been lowered, indicates a probable malfunction of the gear position indicator lamp system. If the landing gear handle warning lamp remains lighted with nose and main gear down and locked, refer to "Landing Gear Handle Warning Lamp Lighted," this section.

Note

Each of the following steps will overcome a particular malfunction and should be followed in sequence.

1. Malfunction and indicator lamps—Checked.
2. Circuit breakers in—Checked.
   a. Landing gear control
   b. Landing gear warning
   c. Speed brake hydraulic valve
3. Utility hydraulic isolation switch—PRESSURIZE.
4. Obtain visual gear check; if main and nose gear appear to be properly extended, refer to "Landing With Unsafe Gear Indication," this section.

If Either Nose Or Main Gear Is Not Extended, Or Visual Check Was Not Possible:

5. Landing gear handle—Recycle.
   a. Alternately impose a negative 1.5 "g" and a positive 2.0 "g" load on the aircraft and check for gear down indication.

CAUTION

Do not exceed 2.0 positive "g" during gear extension attempts.

6. If unable to obtain safe gear indication, refer to "Landing Gear Emergency Extension," this section.

LANDING GEAR HANDLE WARNING LAMP LIGHTED.

If the landing gear handle warning lamp is lighted, with nose and main gear down and locked, the speed brake may be mis-positioned.

1. Speed brake hydraulic valve circuit breaker—Pull.
   If proper speed brake position cannot be verified visually:
2. Landing gear emergency release handle—Pull.
   If proper speed brake position/indication is not achieved:
3. Landing gear emergency release handle—In.

Note

After the landing gear emergency release handle is pulled, nose wheel steering will be inoperative and the nose wheels will be cocked approximately 40 degrees to the right. During landing roll the nose wheels will align and present no directional control problems. If the landing gear emergency release handle has been pushed in, reduced airloads during the landing roll may allow the speed brake to extend and drag the runway.

LANDING GEAR EMERGENCY EXTENSION.

1. Reduce speed to 160 KIAS or normal approach speed, whichever is higher. (Using flaps and slats as required)
2. Landing gear handle—UP. (DN if main gear is down and locked and speed brake is in trail)
3. Landing gear emergency release handle—Pull.

WARNING

If the landing gear door/main landing gear partially extends and stops before full extension, do not push the landing gear emergency release handle back in. To do so will deplete the pneumatic pressure and reduce the possibility of completing gear extension. Leave the handle pulled out and check for positive down and locked indication. Time required to obtain this indication may exceed 10 minutes.

4. Landing gear handle—DN.
5. Landing gear position indicator lamps—Check.

If The Landing Gear Handle Warning Lamp Remains Lighted With Gear Down And Locked:

6. Landing gear emergency release handle—In.

CAUTION

If the landing gear emergency release handle is pushed in, the weight of the speed brake door and the reduced airloads during the landing roll will allow the door to extend and drag the runway.

Note

After the landing gear emergency release handle is pulled, nose wheel steering will be inoperative and the nose wheels will be cocked approximately 40 degrees to the right. During landing roll the nose wheels will align and present no directional control problem.

7. If landing gear is still unsafe, refer to "Landing With Unsafe Gear Indication," this section.

LANDING WITH UNSAFE GEAR INDICATION.

1. Perform landing gear emergency extension procedure. (If required)
2. Consider barrier engagement.
   Refer to "Approach End Barrier Engagement," this section.

If Approach End Barrier Engagement Is Not Used:

3. Fuel dump—As required.
4. External load—Jettison. (If required)
5. Normal Procedures Checklists—Complete.
6. Battery switch—OFF.
7. Shoulder harness—Locked.

If Nose Gear Is Unsafe:

8. Ground roll spoiler switch—OFF.
9. Fly normal pattern and landing.
   Stop aircraft on the runway and insert landing gear ground safety pins.

Note

Touch down at normal landing attitude; do not try to hold the aircraft off the runway. If spoilers are turned off, aerodynamic braking may be obtained by holding the nose off the runway. Light braking may be used in conjunction with aerodynamic braking. Lower the nose gently to the runway while sufficient longitudinal control is still available.

If Nose Or Main Gear Collapses:

10. Throttles—OFF. (After nose is on the runway)
11. Fire pushbuttons—Depress.
12. Abandon the aircraft as soon as possible.

LANDING WITH NOSE/MAIN GEAR RETRACTED.

Approach end barrier engagement with nose, main or both gear retracted is recommended but must take into consideration the barrier availability and type, runway, weather, and aircraft conditions. For all gear retracted landings the ground roll spoiler brake switch should be off. Consideration must be given to missed barrier procedures based on the nature of the gear problem. After the aircraft has engaged the barrier for an arrested landing or after speed has been reduced so that aerodynamic control is not effective, the engines should be shut off by use of fire pushbuttons as this will shut off hydraulic and fuel lines and lessen chance for fire from fuel drainage or hydraulic fluid leakage. Place the throttle to OFF after the fire pushbuttons are depressed to keep the engines from running on residual fuel downstream of the shutoff valve. If time is available, foam the runway 3000 to 4000 feet starting at the barrier to reduce fire hazard.

For Nose, Main Or Both Gear Retracted:

Fly a straight-in normal landing pattern with final approach at minimum sink rate.

1. Fuel dump—As required.
2. Normal Procedure Checklists—Complete.
3. Ground roll spoiler switch—OFF.
4. Battery switch—OFF.
5. All nonessential equipment—OFF.
   This should include engine feed selector, fuel tank pressurization and fuel transfer switches.
6. Hook—Down. (If barrier engagement planned)
7. Shoulder harness—Lock.
8. Throttle friction—Reasonably tight.

Nose Gear Retracted:

For barrier engagement, touch down in center of runway, 400 to 600 ft. short of the cable. Prior to engagement, lower nose to a level attitude. Do not apply brakes. Immediately after cable engagement lower nose smoothly to runway. If a barrier engagement procedure is not utilized, lower the nose to the runway while control effectiveness still exists, and apply maximum braking.

Main/Both Gear Retracted:

For barrier engagement touch down in center of runway so that hook makes contact just short of barrier cable. If a barrier engagement procedure is not utilized, maintain directional control with rudder.

9. Fire pushbutton—Depress.
10. Throttles—OFF.
11. Abandon aircraft.

SINGLE ENGINE LANDING.

During single engine operation, utility and primary hydraulic system flow is reduced by almost 50 percent. Aircraft response to normal control inputs will not be adversely affected unless other hydraulic demands such as landing gear speed brake or wing sweep, etc. are being simultaneously utilized. Since the flight controls use both utility and primary hydraulic pressure the wings should be swept only in 1 "g" flight, and at reduced rate of 1 degree per second. Fuel should be dumped down to a minimum to reduce approach speed and gross weight. A long, moderately shallow straight-in approach should be flown with flaps set at 25 degrees. This is the optimum flap setting in case of a go-around. Maintain 8.5 degrees angle-of-attack until landing is assured. When landing is assured, increase angle-of-attack slowly to "on-speed." Operate engine as high as practical until touchdown. Throughout the approach maintain engine rpm above 85 percent, below this power setting sufficient hydraulic pressure may not be available.

1. Fuel dump—As required.
2. Hydraulic pressure—Checked.
3. Operating engine—Maintain 85 percent rpm minimum.
4. Wing sweep—Adjust slowly for landing. (1 degree per second)
5. Normal Procedures Checklists—Complete.
6. Emergency generator—As required.
7. Landing gear handle—DN.
   Allow gear to fully extend before initiating slat extension.
8. Flap/slat handle—25 degrees.
   If runway length is critical, full flaps may be used.
9. Final approach:
   a. Angle-of-attack—8.5 degrees.

Note

Approach speed will be approximately 20 knots above computed full flap approach speed.

   b. Glide slope—Normal. (Approximately 600 fpm)
   c. Angle-of-attack—On-speed when landing is assured.

SINGLE ENGINE EMERGENCY GENERATOR LANDING.

During operation on emergency generator power the airspeed mach indicator, the altitude vertical velocity indicator and the angle-of-attack tape (and indexers prior to T.O. 1F-111-891) will be inoperative. Yaw and roll trim will be inoperative. The aux pitch trim switch must be used to trim the aircraft longitudinally. Hydraulic system pressures should be monitored closely throughout the approach and landing. To reduce demand on the hydraulic system, do not open or close the speed brake. Refer to "Fuel System Operation On Emergency Electrical Power," "Single Engine Landing" and "Single Engine Go-Around" procedures, this section.

1. Fuel dump—As required. Refer to "Fuel System Operation on Emergency Electrical Power" procedures, this section.
2. Hydraulic pressure—Checked.
3. Operating engine—Maintain 90 percent rpm minimum.

4. Wing sweep—Adjust slowly for landing. (1 degree per second)
5. Normal Procedures Checklists—Complete. (As applicable)
6. Flap/slat handle—25 degrees. (Normal system)
   If runway length is critical, full flaps may be used.
7. Emergency extension of landing gear. Refer to "Landing Gear Emergency Extension," this section.
8. Final approach:
   a. Fly approximately 20 knots above computed full flap approach speed to obtain an 8.5 degree angle-of-attack.
   b. Glide slope—Normal. (Approximately 600 fpm)
   c. Angle-of-attack indexers—On-speed when landing is assured. (After T.O. 1F-111-891)

SINGLE ENGINE GO-AROUND.

Note

Engine acceleration time is severely affected by the amount of compressor discharge air being bled from the engine and by outside temperature. In flight this effect is minimized but during final approach for landing, engine acceleration may require as much as 10-15 seconds to increase thrust from IDLE to MIL with full bleed from the accelerating engine.

1. Throttle—Maximum. (Operating engine)
2. Air source selector knob — EMER. (Use OFF if EMER not installed) (If required)

Note

With air source in OFF or EMER, no servo air will be available for throttle boost or fuel tank pressurization. Lack of tank pressurization will degrade fuel dump rate.

3. Climb.
   a. Maintain approach airspeed until gear is retracted and all obstacles are cleared.
   b. Flaps/slats retraction—Maintain established pitch attitude and retract flaps/slats at a rate to maintain 8.5 degrees angle-of-attack.

WARNING

Excessive angle-of-attack may result from retracting flaps too rapidly.

PITOT PROBE ICING.

In the event airspeed and mach indications return to minimum values during icing conditions, the angle-of-attack indication will be correct. If the airspeed and mach indications should remain fixed during icing conditions the angle-of-attack indicator may be used for landing approach. With the mach indicator fixed at the following values, fly the angle-of-attack indicator as shown in order to maintain 10 degrees angle-of-attack.

Mach Indicator	Angle-of-Attack Indicator
0.45 thru 1.25	12 degrees
1.25 thru 1.40	11 degrees

ANGLE-OF-ATTACK PROBE ICING.

WARNING

If angle-of-attack probe icing occurs (as indicated by no change on the angle-of-attack tape), do not use the TFR. On approach, engage flight control disconnect switch prior to positioning control system to T.O. & LAND or lowering slats. Do not use angle-of-attack indexers.

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
α/β PROBE HEAT (This lamp is disabled above mach 1.10.)	On the Ground:	
	1. Pitot/probe heater switch OFF/SEC.	1. Momentarily place the heater switch to HEAT to verify that heaters are functioning as indicated by the lamp going out.
	2. Primary heater in angle-of-sideslip or angle-of-attack probe overheated.	2. Place switch to OFF/SEC and allow probe to cool. Lamp will remain lighted due to being in OFF/SEC.
	In Flight:	
	1. Primary heater in angle-of-attack or angle-of-sideslip probe not functioning with probe heat sw in HEAT.	1. Place pitot/probe heater switch to OFF/SEC.
	2. Secondary heater in angle-of-attack or angle-of-sideslip probe not functioning with probe heat switch in OFF/SEC.	2. Cycle switch back to HEAT. In icing conditions, if lamp remains lighted, consider angle-of-attack indicator, TFR, and AYC inoperative. Place flight control disconnect switch to OVRD prior to extending slats or placing flight control switch to T.O. & LAND for landing.
ANTI-SKID	Indicates gear down with switch off or anti-skid inoperative.	Check switch on. Recycle to OFF then on. If lamp remains on, place switch to OFF and avoid hard braking during landing roll.
AUX ATT	1. AFRS attitude info unreliable.	1. Verify flight instrument reference select switch is in PRI. The standby attitude indicator will be unreliable. Verify that the AFRS circuit breaker is set whenever the auxiliary attitude caution lamp goes out after being lighted. Failure of the B/N system with AFRS circuit breakers out, results in inaccurate signals to the ADI.
	2. Elec power interruption causing AFRS gyros to fast erect (off flag in view) or an intentional fast erect of AFRS using the fast erect button. ★	2. Maintain unaccelerated straight and level flight during the AFRS fast erection period (normally 2 minutes) to prevent erection to a false vertical. ★
CABIN PRESS	Cabin altitude above 10,000 feet.	Check oxygen equipment. Assure oxygen is on. Check that pressurization selector switch is in NORM. Don oxygen mask and descend to 25,000 feet-or below before continuing flight.

Figure 3-4. (Sheet 1)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
CADS	One of CADC monitors indicates malfunction. Also indicates loss of power to MSMA.	Cross check flight instruments to determine if any are inoperative. Use standby instruments in lieu of malfunctioning primary instruments. Use mach or altitude hold modes with caution. Also suspect loss of power to MSMA and observe structural limit speeds.
ROLL CHANNEL AND/OR PITCH CHANNEL AND/OR YAW CHANNEL	1. One of the triple redundant channels is in error. 2. Pitch, roll or yaw computer power supply failure. ★	1. Depress the damper reset button momentarily. If lamp resets, continue normal operation. For a pitch channel lamp that will reset, verify that the lamp does not come on during an intentionally induced fly-up maneuver at MEA or above before continuing TF operation. If lamp does not reset, change speed to a stability augmentation off region, turn the affected damper OFF and land as soon as practicable. 2. For a yaw channel lamp that does not reset, do not fly auto or manual TF since aircraft response to climb/dive signals, loss of TF fly-up capability, loss of ref not engaged and fly-up off caution lamps may have occurred. In addition, the pitch and roll autopilot operation may be affected. Therefore, autopilot performance should be closely monitored if engaged. 3. If any of the 3 channel lamps remain lighted, change speed to a stability augmentation off region, turn the affected damper OFF and land as soon as practicable. ★
ALL 3 LIGHTED	Loss of one ac pwr source and loss of redundancy.	Decelerate to less than 320 KIAS and land as soon as practicable.
YAW CHANNEL (With slats extended)	One of the redundant AYC signals has a single failure.	Depress the damper reset button; if lamp resets continue normal operation. If lamp does not reset place the flight control disconnect switch to OVRD which terminates AYC. Reset the lamp. If lamp resets, continue operation. If slats are subsequently retracted, place the flight control disconnect switch to NORM and continue. If the lamp does not reset, turn the affected damper OFF and land as soon as practicable.

Figure 3-4. (Sheet 2)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
ROLL DAMPER OR PITCH DAMPER OR YAW DAMPER	One of the triple redundant commands to a damper servo is in error.	Depress damper reset button momentarily, if lamp resets, continue normal operation. For a pitch damper lamp that will reset, verify that the lamp does not come on during an intentionally induced fly-up maneuver at MEA or above before continuing TF operation. If lamp does not reset, reduce speed to the applicable stability augmentation off limits, turn affected damper off and land as soon as practicable. If pitch damper lamp will not reset, do not fly manual or auto TF. ★
ROLL, PITCH & YAW DAMPER (With both PRI or both UTIL HYD sys caution lamps)	One hydraulic system pressure is low.	Reduce speed to damper off operating region. Monitor hydraulic pressure. Depress damper reset button only if affected system pressure returns to normal. Damper operation will not be affected. Follow normal operating procedures. Sweep wings forward at reduced rate to prevent hydraulic pressure depletion. Refer to "Hydraulic System Failure," this section.
PITCH GAIN CHANGER AND ROLL GAIN CHANGER	1. Gear handle DN but flight control system not in takeoff and land configuration. 2. Slats retracted and control system still in takeoff and land configuration.	1. Extend slats. If lamps stay on place control system switch to T.O. & LAND to override the automatic switching. If lamps still remain lighted, place rudder authority sw to FULL to insure full nose wheel steering. If the flt control sw is in OVRD, the lamps will remain lighted. 2. Check that control system switch is in NORM, and gear handle is up. If lamps remain lighted, the flight control system is locked in T.O. & LAND. Place the flight control disconnect switch to OVRD if the wing sweep is aft of 26 degrees. Do not exceed 300 KIAS and land as soon as practicable. Do not fly TFR.
PITCH OR ROLL GAIN CHANGER	One of the redundant roll or pitch gain changers is in error.	Depress damper reset button momentarily. If lamp resets, continue normal operation. If lamp does not reset, decrease speed to less than 425 KIAS/mach 0.80, whichever is less. If subsequent 2 cps oscillation occurs, decelerate.
RUDDER AUTHORITY	Rudder authority differs from that programmed when the control system switch is in the T.O. & LAND position or differs from that called for by slat position when control system switch is in NORM.	Check rudder authority switch 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 lighted, rudder and nose wheel steering authority may be limited.

Figure 3-4. (Sheet 3)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
FUEL DISTRIB	Fuel distribution out of limits. Fuel distribution control system failure. Alternate fuel distribution monitoring system failure.	Select AFT feed. If lamp goes out, indication is from automatic fuel distribution control system. If lamp remains on, indication is from alternate fuel distribution monitoring system. Refer to "Abnormal Fuel Distribution" this section.
FUEL LOW	Usable fuel in fuselage reservoir tank is 2300 (±235) 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 lights. Evaluate the condition and take necessary action.
L FUEL PRESS R FUEL PRESS	Affected fuel manifold pressure is less than 15.5 psia. Improper engine feed selector. Boost pump malfunction.	Check eng feed selector knob, fuel tank pressurization switches and fuel pump pressure lamp. If the fuel pump pressure lamp is lighted, reduce throttle and recheck the L fuel press and R fuel press caution lamps for indication. If lamps remain on, refer to "Fuel Pressure Caution Lamp Lighted," this section.
FUEL 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.
FWD EQUIP HOT	On the Ground: 1. Low airflow at low engine power settings. 2. Icing of water separator during prolonged idle operation with high humidity condition. In Flight: 1. Low airflow at low engine power settings. 2. Reduced cooling air flow due to air flow sel sw being in NORMAL with normal pwr settings. (Most likely during summertime low-level operation). Lighting of lamp will be preceded by a gradual reduction of airflow in the cockpit.	 1. Increase eng pwr slowly (80% rpm on both engines is sufficient) until lamp goes out. 2. Direct GO to depress self-test button on the −65° hot air cont valve located on the aft bulkhead of the wpns bay, or place engine/inlet anti-icing sw to MAN. Lamp should go out within 2 minutes, then reposition anti-icing sw as required. 1. Increase eng pwr slowly and maintain until lamp goes out. 2. Place air flow sel sw to MAX. (If flt conditions permit, reduce eng rpm 80-85% prior to repositioning sw, then adjust pwr as required.) Reduce cooling demand by turning off ALQ-94 (if non-essential). After lamp goes out, reposition air flow & ALQ-94 switches as desired. Pwr reduction not necessary when repositioning air flow sel sw from MAX to NORMAL.

Figure 3-4. (Sheet 4)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
FWD EQUIP HOT (Cont)	3. Low air flow due to icing of water separator with normal pwr settings. (Most likely after setting pwr following prolonged low or idle pwr descent and when operating in or near cloud bases at low level). Lighting of lamp will be preceded by a fairly rapid reduction of airflow to the cockpit.	3. Place engine/inlet anti-icing sw to MAN to thaw water separator. When lamp goes out, return to AUTO or OFF, as desired.
	4. Low air flow due to failed cabin temp control.	4. Select MANUAL temp cont & depress cabin temp cont knob against full COOL stop. Holding kb depressed against stop for 45 seconds will reposition system valves from full warm to full cool. Allow up to 90 seconds with the lamp lighted; then turn off non-essential equipment. The following equipment is listed in order of heat generation: ECM, Attack Radar, TFR, HF Radio (transmit), IRRS, RHAW, UHF, Radio (transmit), Bomb Nav, TACAN, Radar Altimeter, HF Radio (receive), IFF, UHF Radio (receive). ★
AFT EQUIP HOT		Not operable.
LOW EQUIP PRESS	Pressure to forward equipment bay pressurized components is less than 12.5 (±0.5) psi.	Turn TFR to standby if above 15,000 feet altitude to prevent equipment damage. ★
L GEN R GEN	Indicated generator has malfunctioned and has disconnected from its ac bus.	Check power flow indicator indicates TIE. The operating generator will automatically connect to the inoperative bus. If light remains on, refer to "Generator Failures," this section.
HOOK DOWN	Hook is not up and locked.	Land past the approach end barrier. Hook cannot be retracted in flight.
PRI HOT UTIL HOT	Indicated hydraulic system fluid temperature is above 230°F (110°C).	Reduce demand on the hydraulic system. Reduce speed and land as soon as practicable.
L PRI HYD R PRI HYD	Pressure output of the indicated primary hydraulic pump is below 400 to 600 psi.	Monitor hydraulic pressure. If it is normal, land as soon as practicable. If abnormal pressure, refer to "Hydraulic System Failure," this section. Damper oper will not be affected.

Figure 3-4. (Sheet 5)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
L UTIL HYD R UTIL HYD	Pressure output of the indicated utility hydraulic pump is below 400 to 600 psi.	Monitor hydraulic pressure. If it is normal, land as soon as practicable. If abnormal pressure, refer to "Hydraulic System Failure," this section. Damper oper will not be affected.
ICING	1. Icing condition sensed by ice detector.	1. Check that engine inlet anti-icing system is operational by placing engine/inlet anti-icing switch to OFF then to AUTO. If system is operational, above 8000 feet there will be a 300 to 500 foot fluctuation in cabin pressure when cycling the switch. There will also be a noticeable decrease in EPR when system is turned OFF and back to AUTO. If not, go to MAN. Lamp will remain lighted until 60 seconds after icing condition ceases.
	2. Malfunction of ice detection system.	2. If icing conditions are not present, turn anti-icing system off.
IFF	Mode 4 inoperative or improperly comparing code.	1. Check that master control knob is in NORM, Mode 4 control switch is in ON, and proper A or B code is selected. 2. Take action to obtain IFF identification on other modes. ★
INLET HOT	Anti-icing air temperature excessive.	Shut off engine inlet anti-icing. Lamp should go out. If not, slow aircraft to reduce total temperature.
NUCLEAR	Refer to T.O. 1F-111E-25-2.	
L ENG OIL HOT R ENG OIL HOT	1. Oil temperature of affected engine exceeds 250°F (121°C).	1. If oil pressure drops below 30 psi, or lamp persists for more than 10 seconds after retarding to IDLE, shut down the engine and land as soon as practicable. With normal oil pressure following a thrust reduction, advance throttle to a higher setting, if possible. If lamp persists for two minutes, retard to IDLE and monitor oil pressure. If lamp persists for more than 10 seconds after retarding to IDLE, shut down the engine and land as soon as practicable. During ground operation, advance throttle to a higher setting. If lamp persists for two minutes, retard to IDLE and monitor oil pressure. If lamp persists for more than 10 seconds after retard to IDLE, shut down the engine.

Figure 3-4. (Sheet 6)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
L ENG OIL HOT R ENG OIL HOT (Cont)	2. Under some conditions this could also be caused by a broken hot air bleed line.	2. If under steady state conditions, the lamp lights, consideration should be given to shutting down the engine.
OIL LOW	Oil level in either engine down to 4 quarts.	Check oil quantity indicators. Shut down affected engine if not needed. If engine needed, shut down when oil pressure starts to drop.
L ENG OVERSPEED R ENG OVERSPEED	Excessive low press comp rpm. (As a self test feature, lamp is lighted when eng is below idle rpm.)	Retard throttle of affected engine. Lamp should go out at reduced power. If lamp remains lighted, operate engine at reduced power.
OXY	Total liquid oxygen remaining is two liters or less or pressure is 42 psi or less.	Descend to a safe altitude and monitor oxygen supply.
PRI ATT/HDG	The lamp indicates: (1) Flight instrument reference selector switch in AUX. (2) B/N mode selector knob in OFF, HEAT, ALIGN or an AUX NAV CHECK mode. (3) Certain failures of the SP or B/N computer.	Check switch positions. If light persists, place flight instrument reference selector switch to AUX. Cross check attitude indicators and discontinue autopilot operation.
L ENG SPIKE R ENG SPIKE	Mach 0.35 or below, and the affected spike has not contracted or is not full forward.	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.
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.
SERIES TRIM		Not operable.
TF FLY-UP OFF	TF fly-up is not available due to one of the following conditions: (1) Control system switch in T.O. & LAND. (2) Slats are extended. (3) Auto TF switch is in AUTO TF but TFR set is not in TF. (4) The fly-up circuit not armed.	Check switch positions. If light persists, do not fly manual or auto TF.

Figure 3-4. (Sheet 7)

Caution Lamp Analysis

Indicator	*Cause*	*Corrective Action*
TF DRIFT	TFR antenna displaced in azimuth more than 4.5 (±0.5) degrees.	Fly the aircraft to the MEA (if above, do not descend below the MEA). Perform and verify the drift angle accuracy check.
TOTAL TEMP	Total temp above 153°C.	Monitor total temperature indicator for "seconds to go" (five minutes allowable). Reduce speed after five minutes or when the REDUCE SPEED warning lamp lights.
L TIT HOT R TIT HOT		Not operable.
WHEEL WELL HOT	Wheel well area overheat condition. (Possible rupture of engine bleed air duct).	Position air source selector to EMER and decelerate to subsonic. If lamp persists for more than 10 seconds extend speed brakes, lower gear. Land as soon as practicable.
WINDSHIELD HOT	Rain removal air exceeds 450°F.	Place rain removal switch to OFF and reduce pwr below 80%. If after 15 seconds the caution lamp is still lighted, place the air source selector to EMER (RAM if EMER not installed) and observe "Ram or Emer Mode Flight Limits," Section V.

Figure 3-4. (Sheet 8)

Emergency Entrance

1. Push plunger to unlock internal handle.
2. Push in on external handle to extend.
3. Grasp handle to raise hatch.

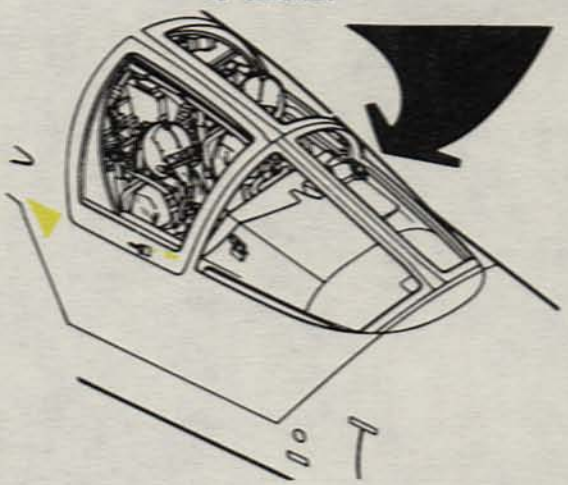


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Figure 3-5.

This is the last page of Section III.

SECTION IV

CREW DUTIES

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The purpose of this section is to provide a compact collection of material wherein each crew member can readily determine his duties in relation to the accomplishment of the over-all mission. Instructions relating to crew duties do not include information which is already covered in other sections.

CREW COORDINATION.

Coordination of actions within a crew is of prime importance to insure the optimum degree of mission success and safety during all phases of operation. This coordination is not necessarily limited to actions alone. Complete familiarity with one's crew position, the responsibilities thereof and a working knowledge of the other crew member's duties will contribute immeasurably toward crew coordination. Each crew member must be constantly on the alert and should notify the responsible crew member of any deviation or discrepancy which will affect successful accomplishment of the mission. Liaison between individuals concerned must be established prior to initiating any action or procedure which will alter aircraft configuration or require correlation of activities between crew members. Prior to flight both crew members must be thoroughly familiar with all aspects of the assigned mission as pertains to their crew specialty to include:

1. Applicable instructions in the flight information publications.
2. Route of flight.
3. Navigation.
4. Air refueling information.
5. Bombing.
6. ECM activities.
7. Normal and emergency communications procedures.
8. Penetration, approach, mixed approach, landing patterns, altitudes, and obstructions at both destination and alternate airfields.

Prior to accomplishment of any of the following, coordination between crew members will be required when:

1. Changing fuel control settings.
2. A function or mode is selected that could affect aircraft control or command steering displays.
3. Changing TFR controls.

AIRCRAFT COMMANDER'S DUTIES.

The AC is responsible for the aircraft and for successful accomplishment of the mission as prescribed by appropriate command directives. In no instance will the safety of the aircraft or crew be compromised.

WEAPONS SYSTEM OFFICER'S DUTIES.

The WSO will insure that sufficient mission preparation is accomplished, in accordance with command directives, to successfully accomplish the briefed mission. During all phases of flight, the WSO will insure that all required equipment is operating correctly for the task being performed and will monitor the aircraft position. During operations with malfunctioning equipment, or in emergency situations, the WSO must respond with the required corrective actions and checklists. During critical phases of flight, the WSO will monitor aircraft configuration, flight and engine instruments, and terrain clearance to insure immediate recognition of a dangerous condition. The AC will be advised immediately of any impending situation or condition which may compromise the safety of aircraft or crew.

TFR INFLIGHT CHECKS.

PRE-TFR ENGAGEMENT.

The following check will be accomplished prior to commencing TF operation. These checks should be initiated over level terrain if possible, and at a minimum of 1000 feet AGL. If the aircraft is above 5000 feet AGL, the radar altimeter bypass switch must be placed to BYPASS.

1. Wing sweep—As required.
2. Radar altimeter index pointer—Set 800 feet.
3. Altimeters—Set.
4. TFR operational check—Complete. (If not previously accomplished)
5. AMI and AVVI command markers—Set.
   Set desired mach, KIAS, and minimum enroute altitude.
6. Radar altimeter bypass switch—BYPASS (if above 5000 ft. AGL).
7. TFR control panel—Set.
   a. Best TFR channel—TF.
   b. Other TFR channel—SIT.
   c. Ride control knob—As desired.
   d. Terrain clearance knob—Set 1000 ft.
8. ISC pitch steering switch—TF.
   a. ADI and LCOS pitch steering bars—Indicate dive.
9. Range selector knob—E position.
10. E-scope—Checked.
    a. Adjust tuning controls for best presentation.
11. Self-test pulse—Checked.

WARNING

If the self-test pulse is absent, do not fly TF under night or IFR conditions.

12. Fly-up check—Complete.
    a. Autopilot release lever—Hold depressed.
    b. Radar altimeter control knob—Depress and hold temporarily.
       (1) Bypass switch—Returns to NORMAL.
       (2) Off warning flag—Out of view.
       (3) TF failure warning and channel fail caution lamps—Lighted.
       (4) ADI and LCOS pitch steering bars—Full fly-up command.
    c. Autopilot release lever—Release.
       (1) Fly-up maneuver—Initiated.
    d. Autopilot release lever—Depressed.
    e. Radar altimeter bypass switch—BYPASS.
    f. Autopilot release lever—Release.
13. Drift accuracy—Check.
    • TF drift caution lamp—Out.

WARNING

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

    • NCU drift—Check.

14. Airspeed/angle-of-attack—Within limits for TFR operation. (Refer to Section V.)
    • Angle-of-attack—Check.
      Prior to operating in TF mode, attain sufficient airspeed by adjusting power with the aircraft in 1 "g" level flight to maintain an angle-of-attack equal to or less than that computed from TF planning data in Appendix 1. Maintaining an angle-of-attack in excess of that shown in Appendix I may result in exceeding the limits in Section V.

15. Descent data—Confirmed.
    - Computed altitude (MSL) for radar altimeter lock-on.
    - Computed level-off altitude (MSL).
16. Helmet visors—Lowered.

TFR OPERATION.

1. Roll autopilot—Engaged.
2. Auto TF switch—AUTO TF and check:
   a. Aircraft response.

      Check that the aircraft response corresponds to the climb and dive commands present. If making a blind letdown, the pushover response should correspond to the ride selected and the aircraft should obtain a dive of 10 (±1.0) degrees.

Note

If a fly-up is commanded during the letdown due to rain, the pilot should manually continue the letdown, while depressing the autopilot release lever, with a 10 degree dive angle until 1000 feet above the minimum enroute altitude. At this time, he should decrease the dive angle and level off at the desired minimum enroute altitude. The letdown to 1000 feet, and subsequent lower settings can be resumed as the rain return disappears from the E-scope presentation.

   b. Reference not engaged caution lamp—Out.
   c. ADI and LCOS pitch steering bars—Centered.
   d. E-scope—Check ground returns.

WARNING

- If the E-scope presentation is unusable for any reason, terrain following operations during IFR conditions should be terminated.
- Both crew members must monitor aircraft terrain clearance during TF flight. During periods when forward video is lost (due to water or low surface reflectivity) aircraft altitude will be monitored by using outside visual reference (if possible), AVVI interpretation, and close observation of TF system operation. The crew should be especially watchful for unexplained descents while over water or level terrain.

Note

When flying at either 1000 or 750 feet set clearance and the terrain clearance is very near the selected value, the terrain video painted on the scope may penetrate the zero command line slightly for approximately the first ¾ mile. At the lower set clearance (200, 300 and 400) a separation of approximately ¼ inch between the zero command line and the terrain video can be expected. These conditions are normal and will not affect the vertical climb/dive signal to the ADI, LCOS or to the flight control system.

3. 5000 foot check:
   a. Radar altimeter—Lock-on.
   b. Radar altimeter bypass switch — Returns to NORMAL.
   c. Dive angle—Increases to 12 degrees.
4. Level-off check:
   a. Aircraft begins rotation toward level flight at approximately 2000 feet AGL.
   b. Airspeed/angle-of-attack—Within limits for TF operation. (See Section V.)
   c. Clearance—900 to 1200 feet AGL.

Note

When switching between channels, or when selecting lower clearances, momentary TF fail and fly-up maneuver may occur. The autopilot release lever can be held depressed while changing channels to prevent the fly-up maneuver from occurring.

LOW ALTITUDE TFR (68%) FLY-UP CHECK.

This check should be accomplished in visual weather conditions and over level terrain/water if possible. During night operations it will only be flown in the auto TF mode.

1. Auto TF switch—AUTO.
2. Radar altimeter index pointer—Set 680 feet.
3. Clearance plane—Set 750 feet momentarily, then 500 feet and check:
   a. Radar altitude low warning lamp — Lighted when passing through 680 feet.
   b. Aircraft levels within limits of the 500 foot setting.
4. Clearance plane—Set 750 feet momentarily, then 1000 feet and check:
   a. Fly-up initiated.
   b. TF failure warning lamp—Lighted until aircraft passes through 680 feet AGL.

c. Pitch steering bars—Indicate fly-up.
d. Radar altitude low warning lamp—Out at 680 feet.

5. Autopilot release lever—Depress and hold while leveling aircraft at 1000 feet, then release.
6. Repeat above steps with TFR and LARA channels reversed.

WARNING

If the 68 percent fly-up capability is not operational in a TF or LARA channel, do not use that channel for TF operation.

7. Low altitude radar altimeter—Reset to 80 percent of desired clearance plane.
8. Roll autopilot—Engaged.

TF FLY-UP CAUSES.

The TF fail fly-up, with the exception of the 68 percent set clearance fail fly-up, is caused by either a detected malfunction within the TFR system or a loss of a data good input signal to the TFR from another system. The 68 percent set clearance fail fly-up occurs any time the radar altimeter altitude is less than 68 percent of the selected set clearance. The failures that will cause a TF fail fly-up are as follows:

1. A detected malfunction internal to the TFR system.
2. Loss of the roll data good signal from the ARS.
3. Loss of the CADS data good signal from the CADS.
4. Loss of the altimeter data good signal from the radar altimeter.
5. Altitude (AGL) is less than 68 percent of selected set clearance.

MALFUNCTION ANALYSIS.

Malfunctions which are most frequently experienced during TFR operation can be classed in two categories: (1) lighting of a fail lamp and (2) improper performance with no fail indications. These two categories can be divided as to malfunctions that occur in one channel only, or both channels. Since the TFR is a dual channel system, and each channel is independent of the other, if a malfunction occurs in one channel, the other channel should be checked under the same conditions to see if the malfunction is present. If the malfunction is present in both channels, the most probable cause is from a signal that is common to both channels. Refer to figure 4-1 for malfunction analysis of TFR system malfunctions. The fail conditions noted in figure 4-1 are assumed to be steady failures; however, these failures may also appear as intermittent, or momentary failures. Short duration fail indications may be normal under certain conditions such as exceeding 45 degrees roll angles, selecting a higher clearance when at the lower clearance setting, performing a self-test of the radar altimeter while in the TF mode, caging the attack radar antenna, etc. A momentary fail will occur immediately after either channel begins operating in the TF mode. Generally, a failure induced by the aircraft descending below 68 percent of selected clearance will also be a momentary failure since the fly-up maneuver resulting from the failure will cause the aircraft to fly up higher than the 68 percent fail threshold. In the event of an intermittent fail condition, the pilot should observe the "E" display for evidence of the test video pulse fading out and discontinuities in the zero command line, and try to determine if the failures can be correlated to the top of the "E" scan. If the intermittent condition is present in both channels in the TF mode, the radar altimeter should be observed for erratic operation. During flight in moderate to heavy turbulence, rapid oscillations of the angle-of-attack probe may also cause intermittent failures in the TFR. In this event, the TFR failures should be coincident with blinking of the CADS caution lamp. The following malfunctions are exhibited as poor performance of the TFR, but do not necessarily cause a fail lamp to light. As stated previously, if a malfunction is suspected in one channel of the TFR, the other channel should be checked under the same conditions to see if the same malfunction is present in that channel.

1. Offset errors above or below the selected clearance. The allowable tolerances for terrain following flight at each selected clearance are listed under terrain clearance knob this section.

   Probable Cause: Alignment errors in the antenna assembly or computer, or errors in pitch or angle of attack inputs to the TFR.

   Check: Compare the offset errors between both channels of the TFR over level terrain and over water. If the offset error is approximately the same in both channels of the TFR, the cause is most likely an error in one of the input signals to the TFR. A significant difference in offset error over land and over water indicates the probability of an alignment error in the antenna assembly or computer.

   Note

   If an error in the angle-of-attack or pitch inputs to the TFR is suspected of causing an offset error, the following flight vector check can be performed to possibly confirm the presence of an error.

TFR Malfunction Analysis

Indication	*Probable Cause*	*Corrective Action*
L and R channel failure caution lamps lighted with mode selector knobs in TF, SIT or GM.	Loss of one of the data good signals from radar altimeter, CADC or attack radar roll pedestal.	Check radar altimeter off flag in view. If in view and altitude is below 5000 feet absolute, select other altimeter channel. Check CADS lamp. If lighted terminate TFR operating. Check attack radar malfunction lamp and/or antenna cage pushbutton indicator lamp lighted. If roll pedestal is not stabilized scope displays for attack radar and TFR in SIT or GM will be blank on one side of scan. If antenna cannot be uncaged check primary altitude/heading and auxiliary attitude caution lamps lighted. Terminate TFR operation for any of these conditions.
L and R channel failure caution lamps lighted with mode selector knob in TF mode only.	Probable cause — aircraft below 68 percent terrain clearance setting or loss of altitude signal from radar altimeter.	Compare radar altimeter indication with terrain clearance setting. If the radar altitude appears correct in relation to the terrain clearance setting, the altitude signal from the radar altimeter to the TFR is in error or has been lost. Select the other radar altimeter channel. If this does not correct the problem, terminate TF operation.
Either channel failure caution lamp lights with mode selector knob in TF, SIT or GM.	Malfunction in antenna receiver, synchronizer transmitter or power supply.	Check scope for test pulse. Switch to STBY and back to an operating mode. If this does not correct malfunction use other channel.
Either channel failure caution lamp lights with mode selector knob in SIT or GM.	Malfunction in antenna receiver, synchronizer transmitter or power supply.	Check scope in 5, 10 and 15 mile ranges. Use opposite channel.
Either channel failure caution lamp lights with mode selector knob in TF only.	Malfunction in antenna receiver or computer. Malfunction in sychronizer transmitter. Malfunction in amplifier power supply or antenna assembly. Aircraft below 68 percent terrain clearance setting.	Check E-scope display for test pulse at top end of scan. Check E-scope display for discontinuities in the zero command line, or complete lack of display. Check E-scope display for proper scan. Compare radar altimeter indication with terrain clearance setting. Use opposite TFR channel for all of above conditions.

Figure 4-1. (Sheet 1)

TFR Malfunction Analysis (Cont'd)

Indication	*Probable Cause*	*Corrective Action*
Either channel failure caution lamp lighted with mode selector knob in STBY, TF, SIT or GM. In TF mode TF failure warning lamp will also light and the G/S warning flag on the ADI will appear.	Blown fuse in amplifier power supply in that channel.	Use other channel.
TF fail when hard ride selected, set clearance 200 or 300 feet and airspeed zero.	Unrealistic condition fed into computer since airspeed is near zero. (This is normal operation and will occur during ground checks only.)	Fail may be cleared by selecting a higher set clearance, selecting soft or medium ride, or by turning the angle-of-attack probe in a counter clockwise direction until fail lamp goes out.
TF failures are induced (or will not clear) during climb maneuvers. ★	"g" loads on the TFR antennas cause false faliure indications. ★	Override the fly-up maneuver, as described under "Overriding Fly-up Maneuvers," this section. ★

Figure 4-1. (Sheet 2)

a. Establish stabilized 1 "g" flight in smooth air. It is important that airspeed be held constant and vertical velocity be held as close to zero as possible.

b. LCOS—Select LOF bomb mode.

c. Adjust the glide/dive angle counter while the aircraft commander observes the analog bar on the LCOS reticle. Set the glide/dive angle counter to position the analog bar to the 6 o'clock position. The glide/dive angle counters should now indicate the aircraft pitch attitude.

d. Compare the glide/dive angle counter reading. If indicated angle-of-attack is more than one degree greater than aircraft pitch attitude, the result may be a high offset error in both TFR channels. The flight crew should be aware that this check, although it is a reasonably accurate check of the aircraft flight vector, should not be considered conclusive since there is the possibility of the existence of errors that are not made apparent with this check. Further, the TFR requires a greater degree of accuracy than can be determined from this check.

2. Deviation from, or porpoising about, the clearance plane.

a. Probable Cause: Present in one channel only—may be caused by alignment errors in the receiver or computer, or from weak video returns due to low reflectivity of the terrain.

Check: E-scope for adequate video returns.

Corrective Action: Use opposite channel unless weak video is due to type of terrain.

b. Probable Cause: Present in both channels—may be caused by adaptive gains in flight control system being too high. This condition usually will correct itself after a short period as the gains drive down to the proper value. This condition is most often encountered upon initial engagement of the auto TF mode after extended flight through very smooth air. This condition may also be caused by weak video returns from terrain of low reflectivity.

Check: E-scope video for adequate returns.

Corrective Action: If problem persists, terminate auto TF flight.

3. Slow response to TFR commands.

Probable Cause: This condition may be caused by improper alignment of the TFR receiver or computer, or certain malfunctions in the TFR computer or flight control system.

Check: Observe manual command displays on the ADI and LCOS to insure that auto TF commands to the aircraft correlate with command bars.

Corrective Action: If condition is present on only one TFR channel, use opposite channel. If condition is present on both channels, terminate auto TF.

4. Intermittent or erratic climb commands

Probable Cause:

a. Side lobe bleed through due to improper alignment of the TFR receiver.

Check: E-scope display for spurious targets or video spikes on top of the normal video.

Corrective Action: Use opposite TFR channel.

b. Certain malfunctions in the TFR computer may allow cross-talk from the attack radar transmit pulse to trigger the command circuits.

Check: Turn the attack radar out of the XMIT mode briefly to see if the erratic commands are corrected.

Corrective Action: Use opposite TFR channel.

c. Certain malfunctions in the radar altimeter may cause a brief failure in the TFR. Frequently these failures are of such a short duration that lighting of the fail lamps may not be apparent, but the ADI and LCOS command bars will deflect upward, and a slight pulse may be felt in the airframe as the fly-up maneuver is momentarily initiated.

Check: These malfunctions are very difficult to detect. The only visual indication will be large, rapid excursions of the pointer in the radar altitude indicator; therefore, it may be difficult to determine if an excursion is due to a malfunction, or variations in the terrain below the aircraft.

Corrective Action: Use opposite radar altimeter channel.

d. Two or more aircraft flying in company may experience erratic commands due to cross-talk between the TFR systems on the aircraft.

Check: E-scope display for evidence of radar interference.

Corrective Action: If this condition occurs, the aircraft should coordinate the selection of channels and modes to minimize the interference.

e. The backscatter from clouds may cause erratic commands.

Check: The E-scope display for the presence of video from the clouds.

Corrective Action: Selecting the opposite channel may correct this problem under marginal conditions; however, if the clouds are sufficiently dense, both channels may command similarly from them. Refer to "Operational Considerations," Section I.

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, VISUAL CCIP, AUTO BOMB, or manual ballistics TRAIL BOMB or RANGE BOMB positions as appropriate to the phase of the mission. The heading, groundtrack, and groundspeed counters are controlled by outputs from the stabilized 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 manual fix modes. Range and course to target or destination are continuously computed and displayed. 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 lamp will light if the SP fails, at which time the NC will automatically switch into auxiliary navigation.

DESTINATION SET PROCEDURE.

Note

- If destination is to be set prior to system operation or preflight alignment, place the platform alignment control knob to OFF/AUX NAV before moving the bomb nav mode selector knob from OFF.
- Do not move the bomb nav mode selector knob from OFF on a system that has been preset for stored heading rapid alignment until ready for alignment.

1. Bomb nav mode selector knob—ALIGN position or above.
2. Fix mode TARGET selector button—Depress.
3. Destination counter—Set.
   Set the destination counters to the desired coordinates.

DESTINATION STORAGE.

Note

If the platform alignment control knob is in RAPID ALIGN do not set storages until bomb nav mode selector knob is in ALIGN.

1. Bomb nav mode selector knob—ALIGN position or above.
2. Fix mode DEST STORAGE 1, 2, or 3 selector button—Depress.

Note

Computed course and miles to destination will remain at the computed values existing when the button is depressed. The attack radar cursors will be absent in the ground velocity and ground auto modes.

3. Destination counters—Set.
   Set the destination counters to the coordinates desired for storage.

4. Repeat steps 2 and 3 for each destination storage desired.
5. Fix mode TARGET selector button—Depress.

STORED DESTINATION RECALL PROCEDURE.

Note

Do not depress the destination storage button unless desired stored destination is within 18 degrees latitude and longitude of indicated destination.

1. Fix mode DEST STORAGE 1, 2, or 3 selector button—Depress.
   Computed course and miles to destination will remain at the computed values existing when the button is depressed. The attack radar cursors will be absent in the ground velocity and ground auto modes.
2. Destination counters—Stop driving.
3. Fix mode TARGET selector button—Depress.
   Computed course and miles to destination will resume to new destination. The attack radar ground velocity and ground auto mode cursors will appear on the new destination if it is in range and the bomb nav mode selector knob is in any position other than GREAT CIRCLE, HEAT or OFF.
4. Destination counters—Check and refine as desired.

ALTITUDE CALIBRATION.

Altitude calibration is necessary prior to position updating, radar bombing and AILA letdown. It is recommended that, circumstances permitting, calibration be made at pressure altitude, and speed at which the up-dating and/or bombing will be performed. For an AILA, calibrate at the altitude and speed at which glide slope interception is anticipated. Due to system design, increased accuracy will result if the calibration is within 43 miles of the destination counters.

Low Altitude Calibration (Below 5000 Feet).

1. Radar altimeter—On, and supplying a good signal.
2. Fixpoint elevation counter—Set.
   Set the fixpoint elevation counter to the known elevation of the terrain where the calibration is to be accomplished.
3. Altitude/test selector knob—CAL.
4. GO lamp—Lighted.
   If the calibration attempt results in a blinking GO lamp, the calibration is good but accuracy is slightly degraded. If the lamp does not light, large errors exist.
5. Altitude/test selector knob—NORM.

High Altitude Calibration (Using Attack Radar).

Altitude calibration should be accomplished in accordance with this procedure when flying over level terrain of known elevation at altitudes normally above 5000 feet or below 5000 feet if a radar altimeter good signal is not present.

1. Radar altimeter—Off.
2. Fixpoint elevation counter—Set.
   Set the fixpoint elevation counter to the known elevation of the terrain where the calibration is to be accomplished.
3. Bomb nav mode selector knob—SHORT RANGE.
4. Fix mode TARGET selector button—Depress.
5. Attack radar function selector knob—XMIT.
6. Attack radar mode selector knob—GND AUTO or GND VEL.
7. Attack radar sector switch—Wide scan.
8. Range select knob—Set minimum range compatible with altitude.
9. Sensitivity time control knob—OFF.
10. Attack radar beta switch—NORM.
11. Altitude/test selector knob—CAL.
12. Antenna tilt—Minus 30°. (—10° TFR on)
13. Using the attack radar tracking handle place the radar range cursor coincident with first ground return and note reading in cursor range counter.
14. Altitude/test selector knob—NORM.
    Calibration is complete.
15. Sensitivity time control knob—Set.

MAGNETIC VARIATION UPDATING.

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

1. Magnetic variation counter—Set to null the magnetic synchronization indicator.

Note

- For accurate magnetic variation updating, the aircraft must be straight and level.
- When in auxiliary navigation modes, magnetic variation control knob adjustments update true heading and do not affect the magnetic heading synchronization indicator.
- When in the align mode, the magnetic heading synchronization indicator indicates align status and cannot be used for magnetic variation updating.

2. AUX NAV operation—Set magnetic variation of the aircraft present position.

AUXILIARY MODE OPERATION.

Note

Selection of AUX NAV CHECK 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 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 last computed or hand set winds are substituted for the stabilized platform outputs. Navigational computer true heading is derived from the auxiliary flight reference system (AFRS) and hand set magnetic variations. Magnetic heading for the horizontal situation indicator (HSI) and the attitude director indicator (ADI) is supplied directly from the AFRS. The magnetic heading synchronization indicator is not operative. The platform error indicator lamp will be out at all times when the platform alignment control knob is in OFF/AUX NAV.

AIRCRAFT POSITION UPDATING.

The need for aircraft position updating is indicated primarily by a position error observed on the attack radar scope. Attack radar updating of present position should be done only after the operator has assessed that the apparent position error is due primarily to the performance of the bomb nav system. There are other reasons why the attack radar cursors may not coincide with the selected radar return identified as the destination coordinates. The accuracy of the coordinates and fixpoint elevation should also be considered. If the attack radar system is not operating, manual position updating should be accomplished periodically. Aircraft position updating will be required more often when operating in an auxiliary navigation mode since nav system accuracy will be degraded. This requirement may be reduced by updating the handset wind with winds found by other navigational means.

Radar Fix.

The following radar fix is applicable only if the attack radar is operating.

1. Altitude calibration—Completed.
2. Fix mode TARGET selector button—Depress.
3. Destination position counters—Set.
4. Fixpoint elevation counter—Set to fixpoint elevation.
5. Bomb nav mode selector knob—SHORT RANGE, or above.
6. Attack radar mode selector knob—GND AUTO or GND VEL.
7. Destination/present position selector switch—PP.
8. Attack radar range selector knob—Use lowest range setting possible.
9. Positively identify target.
10. Attack radar display—Tune for best presentation.
11. Radar cursors—Synchronize on target.

Radar Target Position Determination Procedure.

1. Altitude calibration—Complete.
2. Fixpoint elevation counter—Set to best known elevation.
3. Fix mode PRES POS selector button—Depress.
4. Attack radar mode selector knob—GND VEL or GND AUTO.
5. Destination/present position selector switch—DEST.
6. Bomb nav mode selector knob—SHORT RANGE.
7. Fix mode TARGET selector button—Depress, after destination position counters have stopped slewing.
8. Attack radar range selector knob—Use lowest range setting possible.
9. Radar cursors—Synchronize on target.
10. Destination counters—Record values, then set as desired.

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 cursors will disappear from the scope.
3. Present position correction button—Depress at fixpoint overfly.

   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.

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.

STABLE PLATFORM ALIGNMENT.

Normal turn on, gyrocompassing platform alignment procedures are contained within this section as well as within the appropriate portions of Section II. Rapid alignment to stored gyro compass heading and alignment to stored magnetic variation are covered for special conditions.

Gyrocompass Alignment Procedure.

1. Magnetic variation counter—Check and set to local variation.
2. Platform alignment control knob—OFF.
3. Bomb nav mode selector knob—ALIGN.
4. MAN FIX pushbutton—Depress.
5. Present position latitude and longitude counters—Check and set.
6. Bomb nav mode selector knob—HEAT.
7. Platform alignment control knob—NORMAL.
8. Platform heat indicator lamp—ON.

Note

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

9. Altitude/test selector knob—NORM.
10. Bomb nav mode selector knob—ALIGN.
11. Platform align indicator lamp—On steady within one minute after heat lamp goes out and flashing within an additional five minutes minimum.
    If the aircraft 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 aligned.

Note

If the magnetic heading synchronization 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 aircraft is not to be moved immediately, the mode selector knob may be left in the ALIGN position until just before aircraft movement. This will prevent any system error buildup during the waiting period.

12. Bomb nav mode selector—GREAT CIRCLE or above.

Rapid Alignment to Stored Gyrocompass Heading.

Normally used as a quick reaction procedure.

Pre-Setting Procedure.

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

Note

Once pre-setting is complete the aircraft must not be moved.

Alignment Procedure.

1. Altitude/test selector knob—NORM.

Note

The platform alignment control knob must be in RAPID ALIGN position; if not the system should be considered as not properly preset and alternate align mode should be used.

2. Platform alignment control knob—Check, RAPID ALIGN.
3. Bomb nav mode selector knob—ALIGN.
4. Present position counters—Checked.
5. Platform align indicator lamp—Flashing, within approximately 104 seconds.
   The time required is a function of ambient temperature, local latitude, aircraft attitude and component tolerances. Under optimum conditions the time may be as low as 50 seconds and under worst conditions the time can be as long as 4 minutes.
6. 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 aircraft.
7. Rapid alignment control knob—NORMAL.

Alignment to Magnetic Variation.

Use this procedure as a last resort, when conditions will not permit gyrocompass alignment or rapid alignment to stored gyrocompass heading, since it will result in a low accuracy alignment.

Local Magnetic Variation Determination.

Position the aircraft 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—Recorded.

Note

If local variation is not accurately known, set best available magnetic variation.

Alignment Procedure.

1. Altitude/test selector knob—NORM.
2. Platform alignment control knob—NORMAL.
3. Magnetic variation counter—Check and set prerecorded 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 with MAN FIX pushbutton depressed.

At 100 seconds.

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

Note

Any movement of the aircraft such as that caused by operation of the flight controls should be avoided during the last 10 seconds of alignment, as this may induce a heading error in the system. If the ALIGN lamp comes on before the 100 seconds time has elapsed, it is recommended that the bomb nav mode selector knob be left in ALIGN, and a normal gyrocompass alignment accomplished, or rotate the knob to HEAT, then to ALIGN and move the knob to an operate mode after 60 seconds in ALIGN.

Gyrocompass Alignment Procedure with Flux Valve Inoperative or Malfunctioning.

1. Platform alignment control knob—OFF/AUX NAV.
2. Bomb nav mode selector knob—GREAT CIRCLE.
3. Altitude/test selector knob—NORM.
4. Magnetic variation counter control knob—Check and set.
5. True heading counter—Check that counter drives to approximate value of true heading. If not, adjust magnetic variation until true heading counter indicates correct true heading.
6. Platform alignment control knob—RAPID ALIGN.
7. Present position latitude—Set.
8. Bomb nav mode selector knob—HEAT.
9. Bomb nav mode selector knob—ALIGN.

CAUTION

To prevent equipment damage allow 5 minutes for gyro spin down prior to returning the knob to ALIGN.

10. Platform align lamp—Flashing.
11. Platform alignment control knob—NORMAL.
12. Platform align lamp—On steady, then flashing after gyrocompassing.
13. Magnetic heading synchronizer indicator—Nulled and steady (if time permits).
14. Reset magnetic variation to correct value.

Backup Gyrocompass Alignment Procedure.

WARNING

The following ground and airborne procedures are for alternate use only, when time or other considerations prohibit any other alignment. Attitude information may deteriorate rapidly.

Stabilized Platform Coarse Alignment. (Ground)

1. Bomb nav magnetic variation counter—Set to local value.
2. Bomb nav platform alignment control knob—NORMAL or RAPID ALIGN.
3. Bomb nav mode selector knob—GREAT CIRCLE or SHORT RANGE.
4. Primary attitude/heading caution lamp—Out.
   If flight instrument reference switch is in PRI, the lamp should go out in 36 seconds maximum after step 3, and SP will provide pitch and roll data based on existing aircraft level attitude instead of local plumb bob vertical, the pitch and roll data should be used with caution as a visual attitude reference only, and should not be used for any terrain following or autopilot mode. If the lamp fails to extinguish within 36 seconds, the SP cannot be used and all attitude information will be supplied by the AFRS.
5. Proceed with caution.

Stabilized Platform Coarse Alignment. (Airborne)

1. Flight instrument reference switch—PRI (primary attitude/heading caution lamp will be on if SP is off).
2. Bomb nav magnetic variation counter—Set to local value.
3. Bomb nav platform alignment control knob—NORMAL.
4. Aircraft attitude—Establish zero pitch and roll, using visual horizon reference if possible; hold straight and level flight.
5. Bomb nav mode selector knob—OFF or HEAT momentarily, then GREAT CIRCLE or SHORT RANGE.
6. Primary attitude/heading caution lamp—Out.
   If SP successfully completes coarse alignment, the lamp will go out within 36 seconds after step 5, and SP will supply pitch and roll data based on aircraft level attitude instead of the local plumb-bob vertical; the pitch and roll data should be used with caution as a visual attitude reference only, and should not be used for any autopilot mode. If the lamp fails to go out, the SP cannot be used and all attitude information will be supplied by the AFRS.
7. Proceed with caution.

MALFUNCTION ANALYSIS. (AIRBORNE)

Malfunction of the SP may be indicated in any of the following ways:

1. A critical failure of the stabilized platform control circuits may trigger the SP no-go circuit causing the SP to turn itself off and inhibiting the primary attitude heading ready signal to the flight director system. In this event, the primary/attiude heading caution lamp and the platform error lamp will light. The NC will automatically switch to aux nav operation and the flight director computer will automatically select the AFRS for pitch and roll reference for all using systems. The operator may turn off the platform error lamp and initiate computer auxiliary navigation operation by selecting OFF/AUX NAV on the platform alignment control knob. The operator should immediately handset wind speed to the best known value, or to zero if winds are unknown. Navigation accuracy will be degraded, requiring more frequent position fixes, and introducing this inaccuracy into course angle, fix point bearing, slant range and weapon release computations not based on direct radar sighting.
2. The SP may degrade or fail to an unacceptable condition without triggering its no-go circuit. This type of malfunction can be detected by observation only, so that the operator should continuously evaluate system performance against the following criteria:
   a. Navigation accuracy should be good, such that the present latitude and present longitude counters continuously track aircraft position, and the bomb nav generated cursors closely track selected targets on the attack radar display in GND AUTO and GND VEL modes. Note that radar cursor tracking drift can also be introduced by altitude errors, so that altitude data sources should also be checked when excessive cursor drift is observed. If cursor drift rate exceeds approximately 10 feet/second while observing the same radar target, the operator should be alerted for possibility of more serious SP malfunctions.

b. Computed drift angle should be accurate, as indicated by the difference between the true heading and groundtrack counters on the NC, and as displayed by the difference between magnetic heading and groundtrack on the HSI in Nav, Man Crs, Course Select Nav and AILA flight director modes, and as also reflected on the attack radar display in GND AUTO and GND VEL modes by the targets tracking from top to bottom on the scope during straight and level flight.

c. Computed wind data should be accurate, as indicated by the wind speed, wind from, groundspeed and groundtrack counters, in comparison, with true airspeed data.

d. Flight director pitch and roll data as indicated on the ADI should be accurate when selected to the primary reference. Also, good roll information is required for roll stabilization of the attack radar and TFR, so that when roll data is inaccurate, the radar display will reflect a washout effect as targets are painted from one side of the scope to the other. A quick cross-check with AFRS pitch and roll data may also be made by comparing the ADI and HSI displays, and/or on the ADI alone, by switching the flight director between primary and aux reference. If this cross-check indicates differences between SP and AFRS data, the operator should check all other criteria to determine if the difference is due to SP or AFRS performance, or output signal channels.

As erroneous indications may be due to causes other than SP, all criteria should be taken into account when evaluating SP performance. Malfunctions of the NC can be detected only through observation of errors in computed data displays, or failure to perform per design. BNDTI malfunctions will be reflected by failure to indicate the proper bomb-nav mode or to display the NC range/time data. Ballistic computer malfunctions will appear as erroneous SEC TO RELEASE data in the auto bomb mode, or erroneous LCOS pipper placement in the visual CCIP or auto bomb modes.

PENETRATION AIDS EQUIPMENT CHECKS/OPERATION.

OPERATION OF THE RADAR HOMING AND WARNING SYSTEM.

Turn-On Procedure.

1. CMDS arming switch—SAFE.
2. Power/audio control knob—CW out of OFF detent.

   Allow approximately 5 minutes warmup.
3. Gate select knob—N.
4. BRT knob—Full CW.
5. RTL knob—Full CW.
6. Sensitivity knob—Full CW.
7. Memory control knob—Full CCW.
8. View control knob—Full CCW.
9. RHAW scope filters—As desired.
10. ALR-41 S/L SCAN pushbutton—SCAN.

WARNING

Refer to classified supplement T.O. 1F-111E-1-2.

Confidence Check Procedure.

1. RHAW test knob—LAMP.
   a. All threat lamps, cryo fail lamp, and remote threat lamps—Lighted.
2. RHAW test knob—DISPLAY.
3. RHAW mode selector knob—IRT.
   a. Target—Centered.
4. RHAW mode selector knob—OMO.
   a. Scope display—Checked.
5. RHAW mode selector knob—H3.
   a. Scope display—Checked.
6. RHAW mode selector knob—H2.
   a. Scope display—Checked.
7. RHAW mode selector knob—H1.
   a. Scope display—Checked.
8. RHAW test knob—SYSTEM.
   a. Scope display and threat lamps—Checked, —3 degrees.

      Center the target on the azimuth cursor and adjust to —3 degrees elevation. Monitor the threat display panel for the appropriate warning lamps.
9. Audio—Adjust.

   With the power/audio control knob adjusted to minimum (full ccw), adjust the interphone RHAWS monitor knob to provide an MA warning tone at the desired level. Adjust the power/audio control knob to provide the desired level for threat signals.
10. RHAW mode selector knob—H2.
    a. Scope display and threat lamps—Checked, —4 degrees.

       Center the target on the azimuth cursor and adjust to —4 degrees elevation. Monitor the threat display panel for the appropriate warning lamps.

11. RHAW mode selector knob—H3.
    a. Scope display and threat lamps—Checked, −5 degrees.
       Center the target on the azimuth cursor and adjust to −5 degrees elevation. Monitor the threat display panel for the appropriate warning lamps.
12. RHAW mode selector knob—OMT-F.
    a. Scope display and threat lamps—Checked.
       Check system for proper multiplex displays.
13. Gate select knob—A.
14. View control knob—Positioned at band 1 target. (Azimuth only)
    a. Applicable forward threat lamps—Lighted.
    b. Aft threat lamps—All lighted.
15. View control knob—Positioned at band 2 target. (Azimuth only)
    a. Applicable forward threat lamps—Lighted.
    b. Aft threat lamps—All lighted.
16. View control knob—Positioned at band 3 target.
    a. Applicable forward threat lamps—Lighted.
    b. Aft threat lamps—All lighted.
17. Gate select knob—N.
18. RHAW mode selector knob—OMT-A.
    a. Scope display and threat lamps—Checked.
19. Gate select knob—A.
20. View control knob—Positioned at band 3 target.
    a. Forward threat lamps—All lighted.
    b. Applicable aft threat lamps—Lighted.
21. View control knob—Positioned at band 2 target.
    a. Forward threat lamps—All lighted.
    b. Applicable aft threat lamps—Lighted.
22. View control knob—Positioned at band 1 target.
    a. Forward threat lamps—All lighted.
    b. Applicable aft threat lamps—Lighted.
23. Gate select knob—N.
24. RHAW mode selector knob—OMO.
    a. Scope display and threat lamps—Checked.
25. RHAW test knobs—OFF.

Note

The test knob must be placed to OFF to permit normal RHAWS operation.

Threat Analysis Procedure.

1. RHAW mode selector knob—OMO.
2. Select a radar return for analysis.
3. Turn aircraft toward selected return. (If desired)
4. RHAW mode selector knob—OMT-F or OMT-A. (As applicable)
   Non-threat signals will disappear from the scope.
5. Gate select knob—A. (If desired)
6. View control knob—Frame target. (If desired)
7. Gate select knob—N.

Note

The gate select knob must be in the N position to provide complete warning capability.

Passive Ranging Procedure.

1. Threat analysis procedure—Completed.
2. Radar target—Zero azimuth on RHAW scope. (By turning aircraft)
3. RHAW mode selector knob—As desired.
   Place mode selector knob to H1, H2, or H3 as applicable.
4. Using depression angle, compute range to target as follows:

$$\frac{\text{Altitude above emitter in thousands of feet}}{\text{Depression angle in degrees}} \times 10 = \text{Range in NM}$$

Homer Set Procedure.

1. Threat analysis procedure—Completed.
2. Bomb nav mode selector knob—SHORT RANGE.
3. Attack radar mode selector knob—GND AUTO or GND VEL.
4. Fixpoint elevation counters—Set. (If desired)
   Fixpoint elevation is not critical if homer set is to be followed by a homer track. Fixpoint elevation will affect only the range accuracy and will not affect the bearing accuracy of homer set.
5. Radar target—Zero azimuth on RHAW scope. (By turning aircraft)
6. RHAW mode selector knob—As desired.
   Place mode selector knob to H1, H2, or H3 as applicable.
7. Present position fix mode selector button—Depressed. (If required)
   The destination coordinates must be placed within 20° of the nose of the aircraft. This can be accomplished by one of two methods:
   - By depressing PRESS POS pushbutton then HOMER SET pushbutton, and by using the tracking handle to drive the destination coordinates forward of the aircraft.
   - By going direct to HOMER SET and observing the heading marker (captains bars) on the HSI and the Bomb Nav Distance Time Indicator range, then using the tracking handle to drive the coordinates to the desired position.

8. Homer set fix mode selector button—Depress.
9. Tracking handle—Center the circular cursor (donut) over the target. (If desired)
   If the homer set is to be followed by a homer track, aligning the cursor in elevation is not critical. Align the cursor in azimuth first, and if time and circumstances permit, align the cursor in elevation.

Homer Track Procedure.

1. Homer set procedure—Completed.
2. Homer track fix mode selector button—Depressed.

Note

The homer track fix mode selector button must be depressed before the aircraft is turned from the homer set heading if the homer track solution is to be valid.

3. Inbound track—15° bearing change. (Minimum)
   The greater the bearing change (up to 90°) the more accurate the homer track solution will be.
4. Radar target—Zero azimuth on RHAW scope. (By turning aircraft)
5. Tracking handle—Align the cursor (vertical bar) with the target azimuth.

RHAW/LCOS Homing Procedure.

1. Threat analysis procedure—Completed.
2. Target fix mode selector button—Depress.
3. Radar target—Centered on RHAW scope. (By turning aircraft)
4. RHAW mode selector knob—As required.
5. LCOS mode selector knob—HOM.
6. RHAW gate select knob—N. (If desired)
7. RHAW view control knob—Frame radar return. (If desired)
8. RHAW gate select knob—T.

OPERATION OF THE INFRARED RECEIVER SET.

Turn-on Procedure.

1. Function selector knob—STBY.
2. RHAW mode selector knob—IRT. (If desired)
3. Blanking control knobs—AUTO.
4. Ready/test indicator lamp—Lighted.
   If the ready/test indicator lamp does not light within 8 minutes, the system should be checked for proper servicing.
5. Function selector knob—OPR.
6. Cryo fail lamp—OFF.

Confidence Check Procedure.

1. Turn-on procedure—Completed.
2. CMDS arming switch—TEST.
3. Test button 1—Depress and hold until ready/test lamp lights (approximately 5 seconds), then release.
4. Test button 2—Depress and hold until ready/test lamp lights (approximately 5 seconds), then release.
5. CMDS MLR mode selector knob—NORM.
6. Test button 3—Depress and hold.
   a. IR TGT lamp—Lighted.
   b. IR MLD lamp—Lighted for approximately 5 seconds.
   c. FLARE and TBC dispense indicator lamps blink once while MLD lamp is lighted.
   d. MLD warning tone—Activated.
   e. RHAW scope—Check.
      Target should be approximately half way between center and right edge of scope.
   f. RHAW mode selector knob—IRS.
      Hold until target moves from 0° to +30°.
   g. RHAW mode selector knob—As desired.
7. Test button 3—Released.
8. CMDS arming switch—SAFE.

OPERATION OF THE COUNTERMEASURES DISPENSER.

Operation of the CMDS is automatic once it has been turned on and set to the desired rate or program.

Turn-On Procedure.

1. Arming switch—ARM.
2. TBC mode selector knob—Desired mode.
3. SPC mode selector knob—Desired mode.
4. MLR mode selector knob—Desired mode.
5. Dispenser program module indicator lamps—Monitor.
6. Chaff/flares remaining counters—Monitor.

Confidence Check Procedure.

1. CMDS arming switch—TEST.
2. TBC mode selector knob—MAN.
3. TBC dispense pushbutton—Depress.
   a. TBC dispense indicator lamp—Blinks once.
4. TBC mode selector knob—RT 4.
   a. TBC dispense indicator lamp—Blinks 10 times per second while depressed.
5. TBC mode selector knob—OFF.
6. MLR mode selector knob—MAN.
7. Flare dispense pushbutton—Depress.
   a. Flare and TBC dispense indicator lamps—Blink once each.

8. MLR mode selector knob—OFF.
9. SPC mode selector knob—MAN.
10. SPC dispense pushbutton—Depress.
    a. SPC dispense indicator lamp—Blinks once.
11. SPC mode selector knob—OFF.
12. CMDS arming switch—SAFE.

OPERATION OF THE ELECTRONIC COUNTERMEASURES SET (AN/ALQ-94).

WARNING

Prior to turning on equipment advise all ground crew personnel to move away from the aircraft, minimum distance 6 feet.

Note

The ALQ-94 ECM system, operating in ON or TEST (prior to initiation of self test), may create interference with other radars in the local area. This interference will be indicated by lighting of the XMIT threat indicator lamp(s) for the appropriate band. Self test should be initiated immediately after placing the ECM control knob to the TEST position.

1. ECM control knobs (3)—REC.
    a. RCVR/PA indicator lamps (3) light and remain lighted until system warmup time expires. (Approximately 3 minutes)
2. ECM control knobs (3)—TEST.
3. RCVR/PA pushbutton indicator lamps (3)—Depress and hold for 30 seconds; then release.
    a. RCVR/PA indicator lamps will light and remain lighted within one minute. (Blinking light indicates malfunction)
4. ECM control knobs (3)—REC or ON. (As desired)

Note

In the event of a malfunction indication in REC or ON modes, the system can be reset by placing the control knob to OFF and returning the knob immediately (less than ⅛ second) to REC or ON. This will reset the system without necessitating a warmup delay. If the malfunction has cleared, the system will operate normally. If this procedure does not correct the malfunction, turn the system off for at least 3 minutes, then repeat turn-on procedure.

DUAL BOMBING TIMER CHECK.

1. Upper left weapons select cassette/indicator—Depress.
2. Weapons station indicator/pushbutton—Depress station 1.

   The cassette must be held depressed when selecting the weapon station. Check for SEL on the cassette and station indicator.
3. Delivery mode indicator/pushbutton—Timer selected.
4. Master arm and release switch—ON.
5. Pull-up counter—Set 5.0 seconds.
6. Release counter—Set 5.0 seconds.
7. Weapon release button—Depress and hold.
8. Stopwatch—Check pull-up and release time.
9. Pull-up counter—Set zero.
10. Release counter—Set 30.0 seconds.
11. Weapon release button—Depress and hold.
12. Stopwatch—Check release time.
13. Upper left weapons select cassette/indicator—Deselected.
14. Master arm and release switch—OFF.
15. Dual bombing timer counters—Set to zero.

HF RADIO OPERATION.

WARNING

- Ensure that no personnel or equipment remains in the vicinity of the vertical fin or dorsal antenna sections while the HF radio is transmitting. Be sure that no fuel, oil, or oxygen carts are connected to the aircraft while operating the HF radio. Refer to Section II for danger areas.
- Electromagnetic interference from HF radio transmission, on some frequencies, may cause a fly-up maneuver when operating the TFR in the TF mode. This interference may also cause degradation of the TFR scope displays. If HF radio use is essential and interference is noted when operating in the TF mode, the terrain should be cleared visually or, if this is not possible, the aircraft climbed to the minimum enroute altitude.

Note

If ground operation of the HF system is required, electromagnetic radiation may produce excessive harmonic distortion in the external power monitor, resulting in the power monitor rejecting ground power. Should this occur the external power switch should be selected to OFF and then OVRD.

1. Transmitter selector knob—HF.
2. HF monitor knob—On.
3. Mode selector knob—Desired mode.
4. Adjust volume control to obtain audio balance between HF and UHF radios.
5. RF gain control knob—Maximum clockwise.
6. Squelch control knob—Maximum clockwise.
7. Desired frequency—Set.
8. Microphone switch—TRANS.

   After a frequency change, a 1 kilohertz tone will be heard when the microphone switch is first placed to TRANS. This indicates that the amplifier power supply unit and antenna coupler group are tuning. When the tone ceases, the tuning cycle is complete and a sidetone will be heard when transmitting. Lack of sidetone indicates coupler mistune or an incorrect adjustment of the volume control knob.
9. RF gain control knob—Adjusted.

   Establish contact and then adjust RF gain control knob to obtain optimum signal to noise ratio.

Note

If receiver operation is unsatisfactory, rotate the volume control, RF gain control, and squelch control knobs to the maximum clockwise position.

10. Squelch control knob—Adjust.

AUTOPILOT CHECK.

1. Pitch and roll autopilot/damper switches—AUTOPILOT.

   Control stick motion may occur.
2. Altitude hold and constant track switches—Engaged.

   Reference not engaged caution lamp lights.
3. Reference engage button—Depressed.

   Reference not engaged caution lamp goes out.
4. Move stick forward or aft and release.

   Reference not engaged caution lamp lights and remains lighted.
5. Reference engage button—Depressed.

   Reference not engaged caution lamp goes out.
6. Move stick left or right.

   Reference not engaged caution lamp lights and remains lighted.
7. Reference engage button—Depressed.

   Reference not engaged caution lamp goes out.

FUEL TANK JETTISON PROCEDURE.

Selective jettison of fuel tanks must be accomplished in straight and level flight, with gear and flaps up, at an angle-of-attack less than 10 degrees. Jettison tanks outboard to inboard. Tanks must be empty, or have more than 1,800 pounds of fuel remaining. Refer to "Store Limitations," Section V, for tank release limits.

1. Rack jettison select indicator/pushbutton—Depress.

   The jettison select indicator will read SEL. Station indicators will read WPN, if loaded.
2. Station indicator/pushbutton — Selected. (As applicable)

   Select station with the tank to be released. When the station to be released has been selected SEL will appear in the station indicator.
3. Master arm and release switch—ON.
4. Left or right weapon release button—Momentarily depress.
5. Repeat steps 2 and 4 as necessary.

WEAPONS BAY GUN OPERATION.

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

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

WEAPONS BAY DOOR(S) OPERATION.

NORMAL OPERATION OF WEAPONS BAY DOOR(S).

Electrical power must be on and utility hydraulic system pressure available to operate the weapons bay doors with the normal system.

1. Weapons bay auxiliary control switch—NORM.
2. Weapons bay door control switch—OPEN or CLOSE. (As desired)
3. Weapons bay door position indicator—OPEN or CLOSE (as applicable), after 2½ seconds.

ALTERNATE OPERATION OF WEAPONS BAY DOOR(S).

To operate the weapons bay doors with the auxiliary system electrical power only must be available.

1. Weapons bay auxiliary control switch—AUX.
2. Weapons bay door control switch — OPEN or CLOSE. (As desired)
3. Weapon bay door position indicator—OPEN or CLOSE (as applicable), after 30 seconds.

AIR REFUELING PROCEDURES.

Refer to T.O. 1-1C-1 for general air refueling procedures, and to T.O. 1-1C-1-18 for specific air refueling procedures for this aircraft.

TACAN OPERATION.

1. Function selector knob—As required (REC, T/R, or A/A).
2. Antenna selector switch—AUTO.
3. Channel selector—As required.
4. Volume control knob—Adjust for desired volume level.
5. ISC mode select knob—TACAN.
6. Desired HSI course—Set.
7. Check that the HSI and BDHI bearing pointers lock onto the ground station and that the pointer bearings agree within 3 degrees.
8. Set the HSI course deviation indicator to zero. Check that the course readout and the bearing pointers agree within 3 degrees.
9. Check that the TO/FROM indicator indicates the bearing selected in relation to the ground station.
10. Check the audio signal for readability.
11. If the TACAN signal becomes unusable, check that the OFF flags appear on the course deviation indicator and on the range indicator.
12. The range indicator shall have a distance accuracy of ±0.1 mile plus 0.2 percent of the distance from the station.
13. The air-to-air mode of the TACAN system shall be checked against another aircraft with compatible equipment. The range indicator readings between aircraft shall agree within 1.5 nautical miles.
14. Monitor attitude director indicator, lead computing optical sight and horizontal situation indicator for proper indications.

Note

It is possible that improperly adjusted or malfunctioning ground or airborne TACAN equipment may lock on to a false bearing. This error will probably be plus or minus 40 degrees or multiples of 40 degrees. This is an inherent error in the TACAN system; consequently, bearing information should be cross-checked against other navigation aids whenever possible. When false lock on occurs, it is possible to correct the malfunction by switching to another channel and back to the desired channel or turning the set off and back on again. This deficiency does not affect the range display.

AUTOMATIC DIRECTION FINDER OPERATION.

1. UHF function selector knob—ADF.
2. UHF mode selector knob—PRESET or MAN. Select the desired frequency with the preset channel selector knob or with the manual frequency selector knobs.
3. When transmissions are received on the selected frequency bearing information will be displayed by the number 2 pointer on the bearing distance heading indicator.

Note

During ADF operation audio level will be significantly reduced and considerable background noise will be evident.

IFF CHECK.

1. IFF antenna selector switch—AUTO.
2. Master control knob—NORM.
3. Make contact with the nearest ground control intercept site and request IFF checks. Comply with instructions from the ground control intercept site.
4. Check modes 1, 2, 3A, C and 4 on NORM and LOW power.
5. Check I/P and MIC.
6. Check emergency IFF.
7. At the end of IFF check, leave controls as required for remainder of flight.

This is the last page of Section IV.

SECTION V

OPERATING LIMITATIONS

Note

The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and therefore is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual since it is read directly from the instrument.

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INTRODUCTION.

This section includes limitations that must be observed for safe and efficient operation of the engines and the aircraft. 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.

Note

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 percent of aircraft structural design limits.

MINIMUM CREW REQUIREMENTS.

The minimum crew for normal flight is two.

ENGINE LIMITATIONS.

GROUND OPERATION.

Engine idle speed:

TF30-P-3 — 57 to 69 percent.

Maximum IDLE time is unlimited.

Maximum time at MIL power—45 minutes.

Afterburner operating time limits:

CAUTION

AFTERBURNER	*ONE ENGINE See Note 1*	*BOTH ENGINES See Note 2*
Zone 1, 2, 3	6 Minutes	6 Minutes
Zone 4 & 5 (MAX)	90 Seconds	30 Seconds

NOTES:

1. Rudder must be deflected at least 11 degrees away from the operating afterburner when operating one engine in AB power.
2. Rudder must be centered when operating both engines in AB power.
3. Upon reaching any of the above limits, retard throttle(s) to MIL or below for 6 minutes before further AB operation. AB operation which includes a combination of any of the above conditions shall be limited to a total time of 6 minutes.

Instrument Markings

DATE: 24 APRIL 1970

TF30-P3 ENGINES
BASED ON JP-4 FUEL

TACHOMETER

Idle RPM — percent	57 to 69
Normal operating range — percent	57 to 105
Maximum operating speed — percent	105

TURBINE INLET TEMPERATURE

	TEMPERATURE	TIME
Normal operating range	300 TO 1130°C	NOT APPLICABLE
Starting	710°C	MOMENTARY
Maximum Military operation	1130°C	45 MINUTES
Maximum and Partial Afterburner Limit (No Reset)	1130°C	45 MINUTES
Maximum and Partial Afterburner Limit (In Reset)	1190°C	45 MINUTES
(Unmarked) During an engine acceleration or within 2 minutes following a throttle advance	1160°C	2 MINUTES
(Unmarked) Maximum Continuous	1000°C	UNLIMITED

OIL PRESSURE

- 40 to 50 psi — Normal range.*
- 30 psi — Minimum during idle.
- 50 psi — Maximum.

HYDRAULIC PRESSURE

- 2950 to 3250 psi — Normal range.
- 3250 psi — Maximum.

★

A0000000-E035-C

Figure 5-1.

INFLIGHT OPERATION.

Engine operation should be conducted within the military rating and maximum rating time limits whenever practicable. However, if the mission or flight conditions require operation in excess of these time limits, thrust should not be reduced for only a short interval and then advanced to the high thrust level. Operation at the high thrust level should be continued until conditions permit a reduction in thrust. Overtime operation can be sustained without immediate adverse results, but the total operating life of the engine will be shortened. Operating continuously for one slightly longer period instead of using two or more shorter periods will avoid an additional heat cycling of the engine, which is detrimental to engine life. The engine may be operated continuously, with no time limitation, as long as the turbine inlet temperature limit for continuous operation is not exceeded.

ENGINE TIT TRANSIENT LIMITS.

Refer to figure 5-1.

ENGINE OVERSPEED LIMIT.

Refer to figure 5-1.

ZERO "G" AND NEGATIVE "G" TIME LIMIT.

Engine Fuel Supply.

To prevent possible flameout of both engines, do not exceed 10 seconds under zero "g" or negative "g" flight condition.

WARNING

Do not initiate a zero or negative "g" maneuver when the fuel low caution lamp is lighted. To do so could result in a flameout of both engines.

Note

The fuel low caution lamp may light during a negative "g" maneuver.

Oil Pressure.

The TF30-P-3 engine can be operated with zero indicated oil pressure for 60 seconds under zero "g" or negative "g" flight condition.

ALTERNATE FUEL.

Refer to figure 5-2 for specific limitations for alternate and emergency fuels.

OIL TEMPERATURE LIMITATIONS.

Maximum temperature is 120 degrees C (248 degrees F). Engine oil hot caution lamp will light at 121 degrees C (250 degrees F).

Note

Engine oil overheat may occur during supersonic operation with one engine at MIL or below, since the cooling action of the AB fuel oil cooler is not available.

STARTER LIMITATIONS.

The left starter is limited to 2 cartridge starts in a 15 minute period. Both starters are limited to 5 consecutive pneumatic starts after which a 1 hour cooling period must be observed. Starters are limited to the following periods of continuous operation after which a 15 minute cooling period must be observed:

Left Starter	10 minutes
Right Starter	2 minutes

AIRSPEED LIMITATIONS.

AIRSPEED AND ALTITUDE OPERATIONAL LIMIT ENVELOPES.

The airspeed restrictions for the aircraft with flaps retracted and gear up are presented in figure 5-3. With wings swept between 16 and 49 degrees the airspeed limits shown in figure 5-3 coincide with the limits programmed into the maximum safe mach assembly (MSMA). With the wings swept between 50 and 72.5 degrees, the maximum airspeeds presented are permitted. The maximum sustained speed is coincident with a total temperature of 153 C (308 F) degrees. The maximum dash speed is coincident with a total temperature of 214 C (418 F) degrees or mach 2.50, whichever is less. Flight at speeds which result in total temperatures greater than 153 C (308 F) degrees is limited to 5 minutes per flight.

FUEL DUMP LIMIT SPEED.

Do not dump fuel at airspeeds above 350 KIAS or mach 0.75, whichever is less. To do so may cause dumped fuel to reenter the fuselage, resulting in a fire hazard.

AIR REFUELING RECEPTACLE LIMIT SPEED.

Do not exceed 400 KIAS or mach 1.0 whichever is less, with the air refueling receptacle in any position other than fully closed.

Approved Fuels

	APPROVED/ALTERNATE FUELS					EMERGENCY FUEL
Fuel Specification	MIL-T-5624 Grade JP-4	1. MIL-T-5624 Grade JP-5 2. NATO F-44	1. ASTM D1655 Type A 2. NATO F-30	1. ASTM D1655 Type A-1 2. NATO F-34 or 35	1. ASTM D1655 Type B 2. NATO F-40	MIL-G-5572 Grade 115/145 Gasoline blended with 3 percent MIL-L-6082 Grade 1100 Petroleum Oil
Limitations	None	See Note A	See Note A for item 1	See Note A	See Note A	See Note B

Note A:

Since this fuel does not contain an anti-icing additive and the engines are not equipped with fuel heaters, an anti-icing additive must be blended with the fuel if extensive operation is to be performed where fuel temperatures may reach 0 degrees C or less. The additive will prevent ice from accumulating in the fuel controls and strainers.

Note B:

1. This fuel is approved for a one flight emergency situation only. An alternate fuel should be used if available.
2. Fuel tank pressurization selector switch must be selected to PRESSURIZE prior to takeoff. The fuel tank pressurization caution lamp will be lighted when the landing gear is down or the refuel receptacle is extended.
3. Throttle movements should be as slow as practical.
4. Altitude should remain as low as practical and must not exceed 35,000 feet.
5. Engine thrust available may be reduced approximately 10 percent.
6. The aircraft should be filled with fuel at a temperature of less than 100 degrees F and maintained as cool as possible thereafter. Supersonic flight should be avoided.
7. It is permissible to mix this fuel with a preferred or alternate fuel in the aircraft. However the above restrictions are still applicable.

Figure 5-2.

SLATS/FLAPS LIMIT SPEEDS.

The slats and flaps in the extended position can structurally withstand higher airspeeds than they can be driven against.

1. Flap limits are as follows:

During Extension

• Flaps—0 to 25 degrees	250 KIAS or 0.62 mach, whichever is less
• Flaps—26 degrees to full down	220 KIAS or 0.48 mach, whichever is less

Static Extended Condition or During Retraction

• Flaps—0 to 25 degrees	270 KIAS or 0.62 mach, whichever is less
• Flaps—26 degrees to full down	245 KIAS or 0.48 mach, whichever is less

2. Slat limit speed is 295 KIAS or 0.62 mach, whichever is less.

WEAPONS BAY DOOR(S).

1. Do not open weapons bay door(s) at airspeed in excess of mach 2.0 or exceed mach 2.0 with the doors open.
2. If weapon bay gun ammunition is installed, refer to "Stores Limitations" this section, for door open time limitations.
3. Do not open the weapons bay doors with external stores installed on the pivot pylons above mach 0.90.
4. Due to buffeting, do not open the weapons bay doors with the speed brake extended.
5. Observe 0 to 4 "g" limit during operation of the weapons bay doors.

LANDING GEAR OPERATION LIMIT.

Do not exceed 1.20 "g" during landing gear extension. The maximum speed for landing gear extension, flight with the landing gear extended or for retraction is 295 KIAS.

Airspeed Limitations

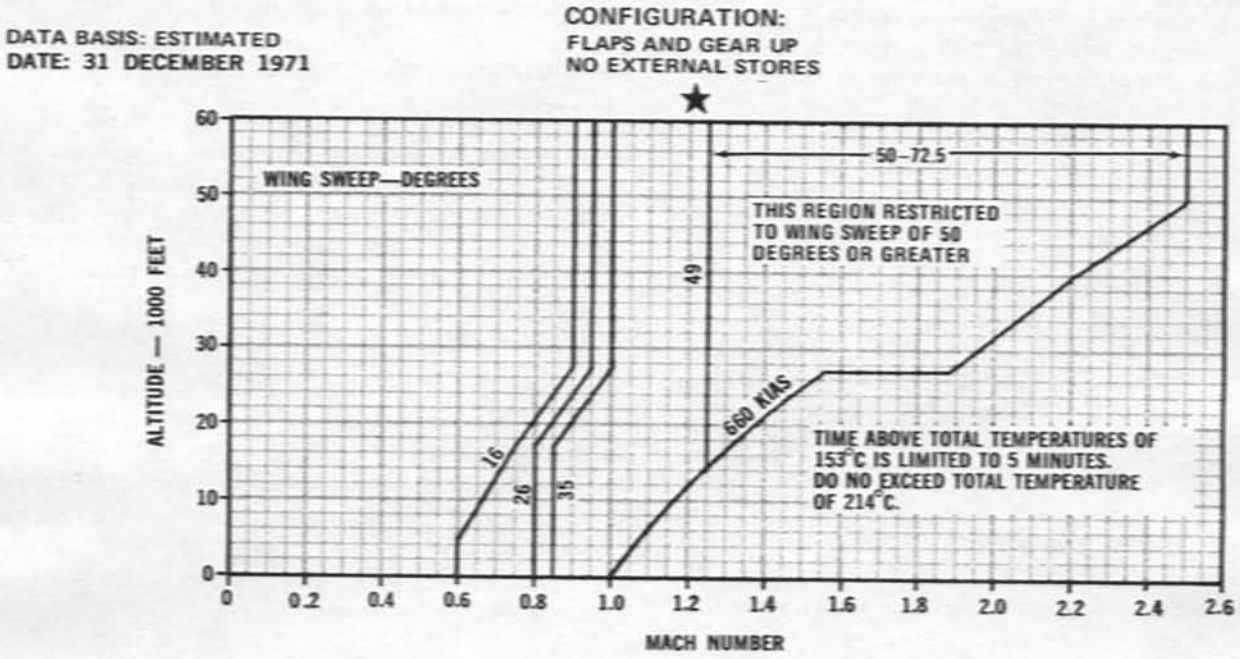

Figure 5-3.

RAM OR EMER MODE FLIGHT LIMITS.

Structurally RAM or EMER mode can be selected anywhere in the flight envelope, however to insure equipment cooling and crews comfort when operating in a ram air mode (RAM or EMER), do not exceed an altitude of 25,000 feet. Airspeed should not be above 460 KIAS or below 260 KIAS.

CAUTION

During ram air operation, the IRRS, RHAWS and ECM equipment must be turned off immediately. Other non-essential electronic equipment should be turned off and the forward equipment hot caution lamp monitored. Refer to "Caution Lamp Analysis," Section III.

TIRE LIMIT SPEED.

The following maximum tire speeds are applicable to all gross weights.

Note

The following tire limit speeds are based on ground speeds with no less than 25 degree flap extension. To obtain KIAS, the speeds must be corrected for temperature, altitude and wind.

1. Main landing gear tire—196 knots ground speed.
2. Nose landing gear tire—196 knots ground speed.
3. Emergency landing maximum tire speed—217 knots ground speed.

Note

Some tires have lower limits. These tires are so marked.

FLIGHT CONTROL SYSTEM LIMITS.

Do not exceed 300 KIAS or mach 0.45, whichever is less, with the control system switch in T.O. & LAND. With the flaps retracted, and wings aft of 26 degrees, do not place the control system switch to T.O. & LAND without first placing the flight control disconnect switch to OVRD.

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system switch in T.O. & LAND, can result in loss of control of the aircraft.

TAXI SPEED.

Maximum taxi speeds:

1. 25 knots straight away
2. 10 knots turning

These limits are based on the possibility of overheating the tires during prolonged straight away taxiing and preventing excessive side loads on the landing gear when turning.

ARRESTING HOOK ENGAGING SPEED.

For maximum arresting hook engaging speed, refer to figure 5-4.

CAUTION

The maximum barrier cable the arresting hook will accept is 1⅜ inches in diameter.

MINIMUM FLYING SPEEDS.

The minimum flying speeds are defined by the maximum angle-of-attack limits presented in figure 5-6. For a discussion of minimum flying speeds, refer to "Minimum Recommended Flying Speeds," Section VI.

SINGLE GENERATOR OPERATION—MINIMUM FLYING SPEEDS FOR CSD OIL COOLING.

Minimum airspeeds/altitudes for constant speed drive oil cooling for continuous single generator operation are as follows:

1. 250 KIAS—20,000 feet and above.
2. 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.

LIMIT MANEUVER LOAD FACTORS.

Limit maneuver load factors as determined from structural considerations are presented in figure 5-5.

ROLL LIMITATIONS.

The following roll limitations are based on 80 percent limit strength values. It should be noted that full normal lateral stick deflection is indicated by a force detent. These limits apply with or without external stores. For other more restrictive limits with external stores refer to "Stores Limitations," this section.

Roll Limitations at All Wing Sweep Angles.

1. Do not exceed the force detent at any mach number at any altitude except under emergency conditions requiring more than normal lateral control.
2. Do not exceed roll angles greater than 360 degrees.
3. Do not perform rolling maneuvers at load factors less than 1 g.
4. Do not rapidly reduce load factors while performing a rolling maneuver.

Roll Limitations at Wing Sweep Angles Where Spoilers Are Operational. (Sweep Angles of 45 Degrees or Less).

1. On aircraft 1 ↓ 19 until modified by T.O. 1F-111-617, do not exceed ½ normal lateral stick deflection at airspeeds between 375 and 580 KIAS. Spoiler commands at airspeeds in excess of 580 KIAS are prohibited.
2. On airplanes 20 ↓ and those modified by T.O. 1F-111-617, do not exceed ½ normal lateral stick deflection at speeds greater than 450 KIAS at any altitude. With fuel in wings, do not exceed ½ normal lateral stick deflection at any mach number at any altitude.

Roll Limitations at Wing Sweep Angles Where Spoilers Are Not Operational. (Sweep Angles Greater Than 45 Degrees).

1. At altitudes less than 25,000 feet, do not exceed ½ normal lateral stick deflection at speeds greater than 525 KIAS.

ANGLE-OF-ATTACK AND RUDDER DEFLECTION (SIDESLIP) LIMITATIONS.

The angle-of-attack and rudder deflection limitations presented in figure 5-6 must be observed. When in longitudinal maneuvering flight, large nose-up pitch rates can be developed if excessively large and/or abrupt aft stick movements are made. Under such conditions, it could be possible to overshoot the allowable

Maximum Arresting Hook Engaging Speed

DATA BASIS: ESTIMATED ABOVE 90,000 POUNDS
DATE: 29 OCTOBER 1971

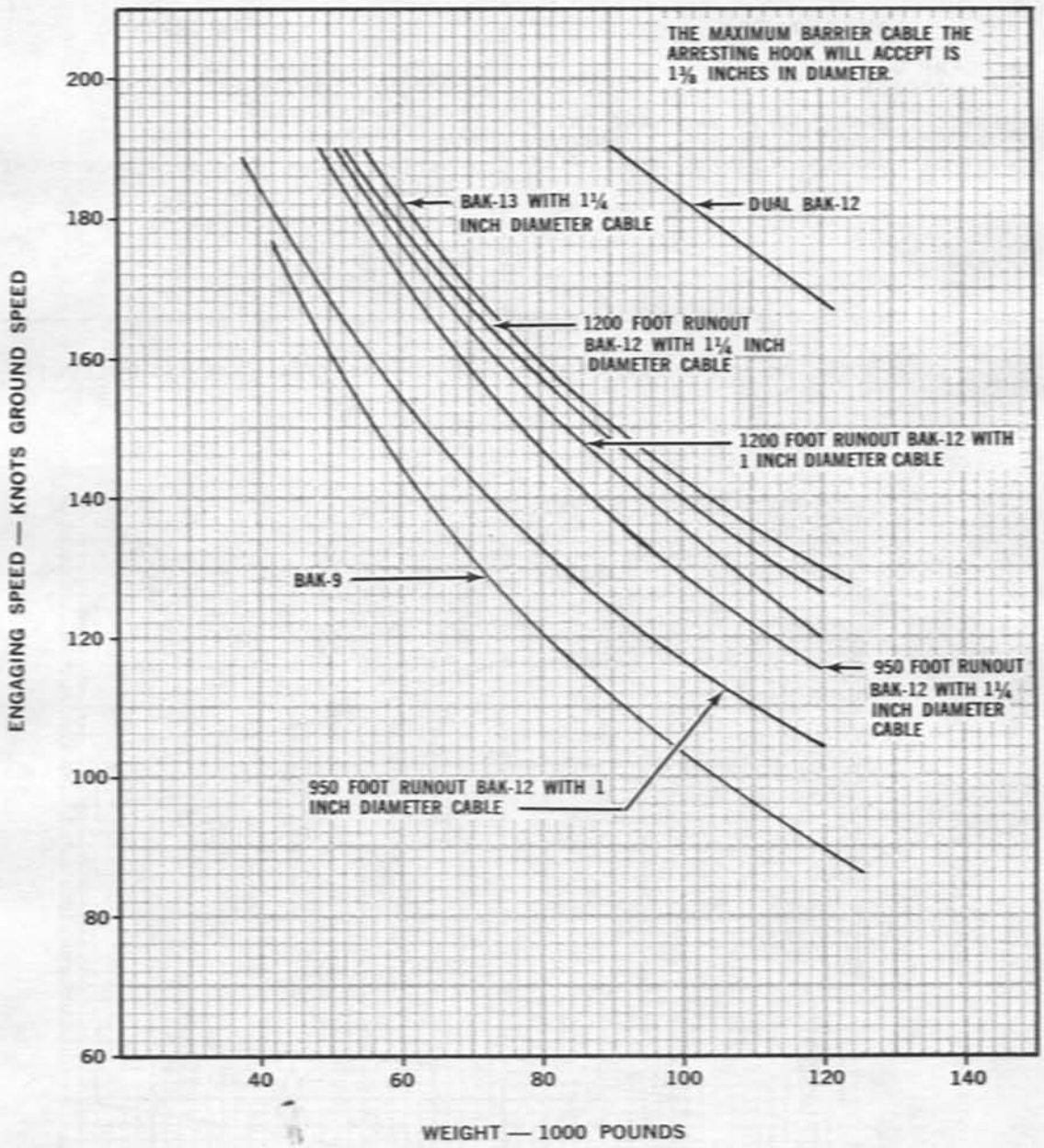

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Figure 5-4.

Limit Maneuver Load Factors

DATA BASIS: FLIGHT TEST
DATE: 30 MARCH 1973

CONFIGURATION:

FLAPS AND GEAR UP
NO EXTERNAL STORES

(A) SYMMETRICAL MANEUVER AT ANY WING SWEEP
(B) ASYMMETRICAL (ROLLING PULLOUT) MANEUVER
(C) SYMMETRICAL MANEUVER DURING WING SWEEP

CONFIGURATION:

GEAR UP OR DOWN
SLATS ONLY EXTENDED
OR FLAPS EXTENDED
NO EXTERNAL STORES

(D) SYMMETRICAL MANEUVER 16-26 DEGREES WING SWEEP

NOTE: DO NOT SWEEP WINGS DURING ASYMMETRICAL (ROLLING PULLOUT) MANEUVER.

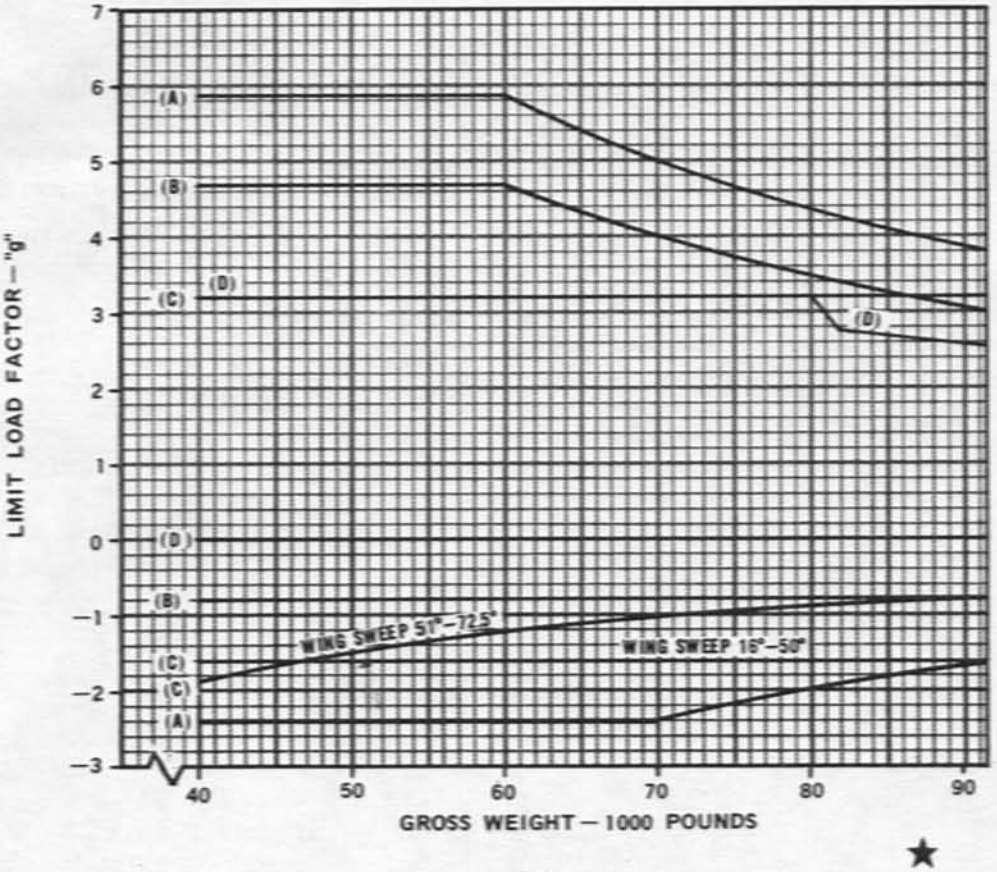


Figure 5-5.

Angle-of-Attack and Rudder Deflection (Sideslip) Limitations ✶

WING SWEEP (Degrees)	CONFIGURATION (With or without weapons)	ANGLE-OF-ATTACK	RUDDER DEFLECTION LIMIT Applicable for all angles-of-attack
16—26	Gear and slats/flaps down.	Below 0.40 mach: 18 degrees or stall warning system activation, whichever occurs first. Above 0.40 mach: 1. 10 degrees for flaps greater than 15 degrees. 2. 12 degrees for flaps at 15 degrees or less. 3. Up to 15 degrees with slats only.	Yaw Damper On. 15 degrees. Yaw Damper Off. 12 degrees, do not make abrupt rudder inputs.
16—45	Gear and slats/flaps up.	15 degrees or stall warning whichever occurs first.	Yaw Damper On. 6 degrees below mach 0.80. 3 degrees above mach 0.80. Yaw Damper Off. No intentional sideslip.
46—72	Gear and slats/flaps up.	18 degrees or stall warning whichever occurs first.	Yaw Damper On. 6 degrees below mach 0.80. 3 degrees above mach 0.80. Yaw Damper Off. No intentional sideslip.

Figure 5-6.

angle-of-attack. As the angle-of-attack increases, the pitch rate of the aircraft should be moderated by forward stick movement to avoid exceeding the angle-of-attack limit. On aircraft prior to T.O. 1F-111-891 the rudder pedal shaker will activate when a combination of the values of pitch rate in degrees per second and wing angle-of-attack in degrees total 18 (±1). On aircraft modified by T.O. 1F-111-891, stall warning will activate at 14 degrees wing angle-of-attack for wing sweeps less than 50 degrees. For wing sweeps greater than 50 degrees, stall warning will be activated only above 14 degrees wing angle-of-attack when the combination of pitch rate in degrees per second and wing angle-of-attack in degrees total 18 (±1). Sideslip limitations are given in terms of rudder surface deflection limits since no direct method exists to determine sideslip angles. Sideslip limits are set to assure proper engine operation and should not be intentionally exceeded.

WARNING

- The rudder required to maintain coordinated flight increases as angle-of-attack increases. Attention should be given to coordinating rudder and lateral control when maneuvering at angles-of-attack above 10 degrees.
- Exceeding the rudder deflection limits to perform sideslips or rolling maneuvers can result in loss of control of the aircraft due to the roll and yaw characteristics of the aircraft and subsequent rapid build-up of angle-of-attack. When full rudder authority is available, care should be taken to assure that the rudder deflection limits are not exceeded.

FLIGHT WITH DAMPERS OFF.

Figure 5-7 presents the damper off operating limits. For a complete discussion, refer to "Flight With Dampers Off," Section VI. In the event of a flight control system malfunction necessitating turning the pitch, yaw, or roll damper off in flight, the aircraft speed should be reduced to that commensurate with figure 5-7 and the affected damper turned off. Continuing flight should be accomplished with a wing sweep of 26 degrees observing the airspeed limitations for this sweep presented in figure 5-7, and landing should be accomplished as soon as practical. In the event of damper failure with the gear down, flaps and slats extended, land as soon as practical. If retraction of flaps and slats is necessary, observe the limits in figure 5-7.

Dampers Off Operating Limits

DO NOT EXCEED THE FOLLOWING AIRSPEEDS/ALTITUDES:		
PITCH DAMPER OFF		
Wing Sweep	*Altitude*	*Airspeed*
16 Degrees	No Restriction	Mach 0.40
26—45 Degrees	No Restriction	400 KIAS or mach 0.75 ★ whichever is less
46—72.5 Degrees	Sea Level to 20,000 feet	Mach 0.70
	Above 20,000 feet	Wing sweep airspeed limits
YAW AND/OR ROLL DAMPER OFF ★		
Wing Sweep	*Altitude*	*Airspeed*
16—45 Degrees	No Restriction	Wing sweep airspeed limits
46—72.5 Degrees	No Restriction	Mach 1.50

Figure 5-7.

Crosswind Takeoff and Landing Limits (All Wing Sweeps) ★

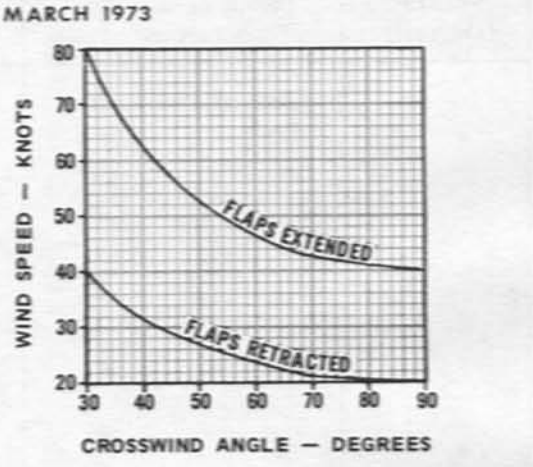


Figure 5-8.

WARNING

- During flight with pitch, yaw, or roll damper off, large and/or abrupt stick and/or rudder inputs should be avoided. Rolling maneuvers should be limited to 60 degree bank angle.
- For landing with pitch and/or yaw damper off, approach at 10 degree angle-of-attack (approach indexer setting). Approaches with the pitch damper off will require increased pilot attention to airspeed and angle-of-attack control.

CROSSWIND LANDINGS.

The maximum allowable crosswinds during landings with flaps extended and/or retracted are presented in figure 5-8. A crabbed or wing-low-crabbed approach is recommended. Crosswind landing with the flaps retracted will require a crabbed attitude at touchdown. Crosswind landings with flaps extended at wind speeds above approximately 75 percent of the maximum specified on the figure will also require a crabbed attitude at touchdown. Do not exceed a crab or yaw angle of 10 degrees at touchdown.

WARNING

With the flaps retracted do not attempt to align the aircraft with the runway prior to touchdown as sufficient roll control may not be available to hold the wings level.

After touchdown align the aircraft with the runway prior to lowering the nose gear to the ground. Refer to Section II, Crosswind Landings, for a discussion of the proper operating procedures.

PROHIBITED MANEUVERS.

The following maneuvers are prohibited:

1. Spins
2. Stalls
3. Flight into heavy buffet.

CENTER-OF-GRAVITY LIMITATIONS.

AFT CENTER-OF-GRAVITY.

For a detailed discussion of aft center-of-gravity, refer to "Determination of the Aft Allowable Center-of-Gravity Position," Section VI. A simplified method for determining aft center-of-gravity limits, is presented in figure 5-9, as a function of maximum allowable down horizontal stabilizer position and minimum fuel remaining.

CAUTION

Loadings with stores on fixed pylons or fixed and pivoting pylons may result in takeoff conditions where the center-of-gravity is behind the aft limit at 16 degrees wing sweep. If fixed pylons are to be used, caution should be exercised in determining the center-of-gravity/aft limit relationship for takeoff. Refer to Determination of Aft Allowable Center-of-Gravity Positions, Section VI.

For wing sweeps greater than 35 degrees at airspeeds below mach 2.0, there is no aft center-of-gravity limit within the normal fuel and/or store loading capability of the aircraft. Maximum allowable down horizontal stabilizer position and/or minimum fuel remaining values are not provided for wing sweeps aft of 26 degrees with stores loaded on the fixed pylons only, or the fixed and pivoting pylons because store carriage of these configurations is not permitted at wing sweeps aft of 26 degrees. For aircraft with empty pylons or pylons and empty racks attached, use the maximum allowable down horizontal stabilizer position and/or minimum fuel remaining values for the basic aircraft. Maximum allowable down horizontal stabilizer deflection is not used as a limit at airspeeds above mach 2.0 because the limitation in this airspeed range is based on directional stability rather than longitudinal stability. Figure 5-9 is used to determine if the aircraft center-of-gravity is within the aft limit by use of the maximum allowable down horizontal stabilizer position for airspeeds less than mach 2.0, and the minimum fuel remaining for airspeeds of mach 2.0 and above. The maximum allowable down horizontal stabilizer positions and minimum fuel values presented are for stabilized flight, with speed brake retracted. The maximum allowable down horizontal stabilizer positions are "do not exceed" values. Speed brake extension causes a nose up pitching moment which will require a down horizontal stabilizer correction to arrest; therefore, for flight with the speed brake extended, the maximum allowable down horizontal stabilizer position shown in figure 5-9 is not applicable and may be temporarily exceeded. It is recommended that the speed brake be periodically retracted and the maximum allowable down horizontal stabilizer position limit checked if prolonged flight is conducted with the speed brake extended. The maximum allowable down horizontal stabilizer position is a value which is dependent on wing sweep and pylon loading only and can be used to determine if the aircraft is within the aft center-of-gravity limit for a specific pylon loading regardless of aircraft ballast, fuel distribution, weapon bay payload, airspeed or store weight. For example, with stores loaded on the pivoting pylons only and 26 degree wing sweep, the maximum allowable down horizontal stabilizer position presented in figure 5-9 (1.0 degree down) does not change as the total store weight is reduced or airspeed is changed. The maximum allowable down horizontal stabilizer position is independent of airspeed for a given wing sweep due to the fact that as the airspeed changes the center-of-gravity limit also changes, thereby allowing a single value of horizontal stabilizer to define the aft limit throughout the entire speed range of that wing sweep. The minimum fuel values presented for flight at airspeeds below mach 2.0 are primarily for mission planning purposes and may be attained before or after reaching the specified maximum allowable down horizontal stabilizer position limit provided the surface position indicator is operating properly. If the minimum fuel value is reached prior to attaining the maximum allowable down horizontal stabilizer position, it is permissible to continue flight until reaching the maximum allowable down horizontal stabilizer position limit even though the actual fuel remaining will be less than the minimum value presented provided the horizontal stabilizer position indicator is working properly. (If in doubt,

Aft Center-Of-Gravity Limits (Based on Stabilizer Position/ Minimum Fuel)

DATA BASIS: ESTIMATED
DATE: 30 MARCH 1973

FUEL GRADE: JP-4
ENGINES: TF30-P-1/P-3

THE MAXIMUM STABILIZER POSITION ALLOWABLE IS THE MAXIMUM CLOCKWISE AVERAGE POSITION OF THE POINTERS ON THE CONTROL SURFACE POSITION INDICATOR. IF THE AVERAGE ELEVATOR EXCEEDS THIS VALUE IN A CLOCKWISE DIRECTION WHILE IN 1.0 "g" FLIGHT, THE AIRCRAFT CENTER OF GRAVITY HAS EXCEEDED THE LIMIT FOR THAT COMBINATION OF WING SWEEP AND PYLON LOADING. THE HORIZONTAL STABILIZER POSITION IS A FUNCTION OF WING SWEEP AND PYLON LOADING ONLY.

CONSIDERATIONS:

Minimum fuel remaining is based on:

a. An 8200 pound fuel differential has been maintained, and/or all the remaining fuel is in the forward tank.

b. The weapon bay gun (with no ammunition) or equivalent ballast is the only weapon bay loading.

c. The aircraft is ballasted at 47.5 percent MAC.

d. The total store weight does not include pylons and racks.

		GEAR AND FLAPS DOWN		GEAR AND FLAPS UP							
	WING SWEEP	16 DEGREES	26 DEGREES	16 DEGREES		26 DEGREES		35 DEGREES	AFT WING SWEEP		
	AIRSPEED	Flaps — Full	Flaps — 1° to Full	Mach 0.60 or Less	Mach Greater Than 0.60	Mach 0.70 or Less	Mach Greater Than 0.70	To Limit Airspeed	MACH 2.0	MACH 2.2	MACH 2.5
CONFIGURATION	TOTAL STORE WEIGHT	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Maximum Stabilizer / Minimum Fuel-Pounds	Minimum Fuel-Pounds	Minimum Fuel-Pounds	Minimum Fuel-Pounds
BASIC AIRCRAFT	0	3.0° Up / 6,500	1.0° Up / 0	1.0° Down / 7,500	1.0° Down / (1)	1.5° Down / 1,500	1.5° Down / 3,500	1.0° Down / 0 (4)	0 (2)	4,000 (2)	8,000 (3)
STORES ON PIVOTING PYLONS ONLY	5,000	3.5° Up / 7,000	1.5° Up / 1,500	0.5° Down / 8,000	0.5° Down / (1)	1.0° Down / 3,000	1.0° Down / 5,000	0.5° Down / 0 (4)	0 (2)	8,000 (3)	(1)
	10,000	3.5° Up / 7,000	1.5° Up / 0	0.5° Down / 8,000	0.5° Down / (1)	1.0° Down / 2,000	1.0° Down / 4,000	0.5° Down / 0 (4)	0 (2)	8,000 (3)	(1)
	20,000	3.5° Up / 4,000	1.5° Up / 0	0.5° Down / 6,500	0.5° Down / (1)	1.0° Down / 0	1.0° Down / 3,000	0.5° Down / 0 (4)	3,000 (2)	19,000 (3)	(1)
STORES ON FIXED PYLONS ONLY	5,000	3.5° Up / 28,000	1.5° Up / 5,000	0.5° Down / (1)	0.5° Down / (1)	1.0° Down / 5,000	1.0° Down / 7,000				
	10,000	3.5° Up / (1)	1.5° Up / 6,000	0.5° Down / (1)	0.5° Down / (1)	1.0° Down / 6,500	1.0° Down / 13,000				
	20,000	3.5° Up / (1)	1.5° Up / 8,000	0.5° Down / (1)	0.5° Down / (1)	1.0° Down / 16,000	1.0° Down / (1)				
STORES ON PIVOTING AND FIXED PYLONS	10,000	4.0° Up / (1)	2.0° Up / 9,000	0 / (1)	0 / (1)	0.5° Down / 8,000	0.5° Down / 20,000				
	20,000	4.0° Up / (1)	2.0° Up / 7,000	0 / (1)	0 / (1)	0.5° Down / 8,000	0.5° Down / 20,000				
	40,000	4.0° Up / 29,000	2.0° Up / 7,000	0 / (1)	0 / (1)	0.5° Down / 20,000	0.5° Down / (1)				

Notes

(1) Using the minimum fuel remaining considerations above, there is no fuel loading at which the center of gravity does not exceed the aft limit.

(2) For aircraft with the forward or forward and aft QRC pods on the fuselage, add 3,000 pounds of fuel. No change is required with only the aft QRC pod on the fuselage.

(3) For aircraft with the forward or forward and aft QRC pods on the fuselage, there is no fuel loading at which the center of gravity does not exceed the aft limit. With only the aft QRC pod installed on the fuselage, observe the minimum fuel remaining values presented.

(4) For wing sweeps greater than 35 degrees at airspeeds below mach 2.0, there is no aft center of gravity limit within the normal fuel and/or store loading capability of the aircraft.

Figure 5-9. ★

push takeoff trim button and check for 3.8 degrees trailing edge up.) For example, this condition may occur when stores or weapon bay gun ammunition is loaded in the weapon bay. The minimum fuel values are based on the following considerations:

a. An 8200 pound fuel differential (automatic fuel feed) has been maintained, and/or all the remaining fuel is in the forward tank.
b. The weapon bay gun (with no ammunition) or equivalent ballast is the only weapon bay loading.
c. The aircraft is ballasted to 47.5 percent MAC.
d. The total store weight does not include pylons and racks.

WARNING

- When the maximum allowable down horizontal stabilizer position is reached, the aircraft is at the aft center-of-gravity limit for that specific wing sweep/airspeed/pylon loading even though the minimum fuel values may have been exceeded. The aft center-of-gravity limit will be exceeded if flight is continued, and the wing sweep, airspeed, and/or pylon loading is not changed to maintain the center-of-gravity within the limits specified in figure 5-9.
- The minimum fuel values presented for flight at airspeeds below mach 2.0 are primarily for mission planning purposes and may be attained before or after reaching the specified maximum allowable down horizontal stabilizer position limit.

It should be noted from figure 5-9 that as wing sweep is reduced from 35 to 16 degrees an increased total fuel loading will be required to maintain the center-of-gravity within the aft limit. Also, when stores are carried on the pivoting pylons only and store weight is decreased from 20,000 to 5,000 pounds, an increased total fuel loading will be required to maintain the center-of-gravity within the aft limit. When the total store weight carried is between the values presented, interpolation is required to determine minimum fuel remaining values. For example, gear and flaps up, 16 degrees wing sweep, mach 0.60 or less, and a 12,000 pound total store weight carried on the pivoting pylons only. By interpolation, the minimum fuel remaining to determine the aft center-of-gravity limit would be 7400 pounds with no change in the maximum allowable horizontal stabilizer limit. For total store weights of 5,000 pounds and less (loading on the pivoting pylons only or on the fixed pylons only), or 10,000 pounds and less (loading on the fixed and pivoting pylons) the minimum fuel value and/or maximum allowable horizontal stabilizer position presented remains constant, and interpolation cannot be used. This is due to the fact that for store weights less than the above values there is no appreciable change in the center-of-gravity position.

FORWARD CENTER-OF-GRAVITY.

Current loadings of the aircraft do not result in center-of-gravity positions forward of 23 percent MAC. If future loadings result in center-of-gravity positions forward of 23 percent MAC, increase rotation speed one knot for each one percent forward of 23 percent. The forward center-of-gravity limits for landing with full flaps as a function of wing sweep, in terms of maximum allowable trailing edge up horizontal stabilizer position, are as follows:

1. 16 degree wing sweep 10.5 trailing edge up
2. 26 degree wing sweep 9.5 trailing edge up

The above limit is applicable only at 10 degrees angle-of-attack. Once the landing configuration and approach attitude (10 degrees angle-of-attack) have been established, monitor the control surface position indicator to determine if the aircraft is within the forward center-of-gravity limit. For certain combinations of fuel remaining and bay and/or external loadings, it may be necessary to land with the wing sweep forward of 26 degrees in order to attain a center-of-gravity within the forward center-of-gravity limits. If it is necessary to sweep the wings for landing, monitor elevator position to assure the aft center-of-gravity limits in this section are not exceeded. The above limits are based on maintaining sufficient longitudinal control to achieve at least 18 degrees angle-of-attack with flaps and slats extended and full back stick.

Note

- The maximum allowable horizontal stabilizer position specified above is the maximum average trailing edge up position of the pointers (mid position between the pointers) on the control surface position indicator.
- In view of the small differences in the horizontal stabilizer limits shown and the fact that the control surface position indicator cannot be easily read any closer than one degree, a value of 10 degrees trailing edge up may be used as a rapid reference for any sweep between 16 and 26 degrees. Using this reference value will maintain an adequate safety margin under any condition.

CREW MODULE CENTER-OF-GRAVITY LIMIT.

WARNING

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. In the event that combined crew weight, including personal equipment, exceeds 430 pounds, or the weight differential between the two occupants exceeds 65 pounds, low altitude safe escape will be compromised and landing impact acceleration will increase. To assure stability of the crew module in event of ejection, it must be loaded in accordance with T.O. 1-1B-40.

GROSS WEIGHT–CENTER-OF-GRAVITY LIMITATIONS FOR TAXI AND GROUND OPERATION.

Loadings which result in an aft center-of-gravity in excess of 60 percent MAC can cause the aircraft to tip back when brakes are released with AB power. At light gross weights, forward wing sweep angles will minimize nose wheel steering difficulties.

GROSS WEIGHT LIMITATIONS.

MAXIMUM GROSS WEIGHT.

Maximum gross weight limits are as follows:

1. Taxi and takeoff—91,500 pounds.
2. Inflight—100,000 pounds.
3. Landing—80,000 pounds.

AIRCRAFT SINK RATE AT TOUCHDOWN.

The allowable sink rate at touchdown shall not exceed 600 feet per minute at landing gross weights up to the maximum allowable with any authorized weapon and/or stores loading, except if any usable fuel remains in the external tanks, the allowable sink rate shall not exceed 360 feet per minute.

BRAKE LIMITATIONS.

BRAKE APPLICATION SPEED LIMIT.

Brake energy limits with slats, flaps and spoilers extended are presented in figure 5-10. The example lines explain how to determine the amount of energy absorbed by the brakes during a stop.

Note

Subtract 50 percent of the headwind component, measured by the tower, from the indicated airspeed. A tailwind component must be added to the indicated airspeed.

Example: Full Stop Landing.

Given: Gross weight = 60,000 pounds.
Airspeed when brakes applied = 95 knots indicated airspeed.
Tower reported wind velocity = 5 knot tailwind.
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 14.0 million foot-pounds per brake.

CAUTION

If maximum braking capacity is utilized (danger zone), wheel blowout plugs will relieve tire pressure within 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 aircraft is brought to a complete stop with a single continuous application of the brakes.

MISCELLANEOUS OPERATIONAL LIMITATIONS.

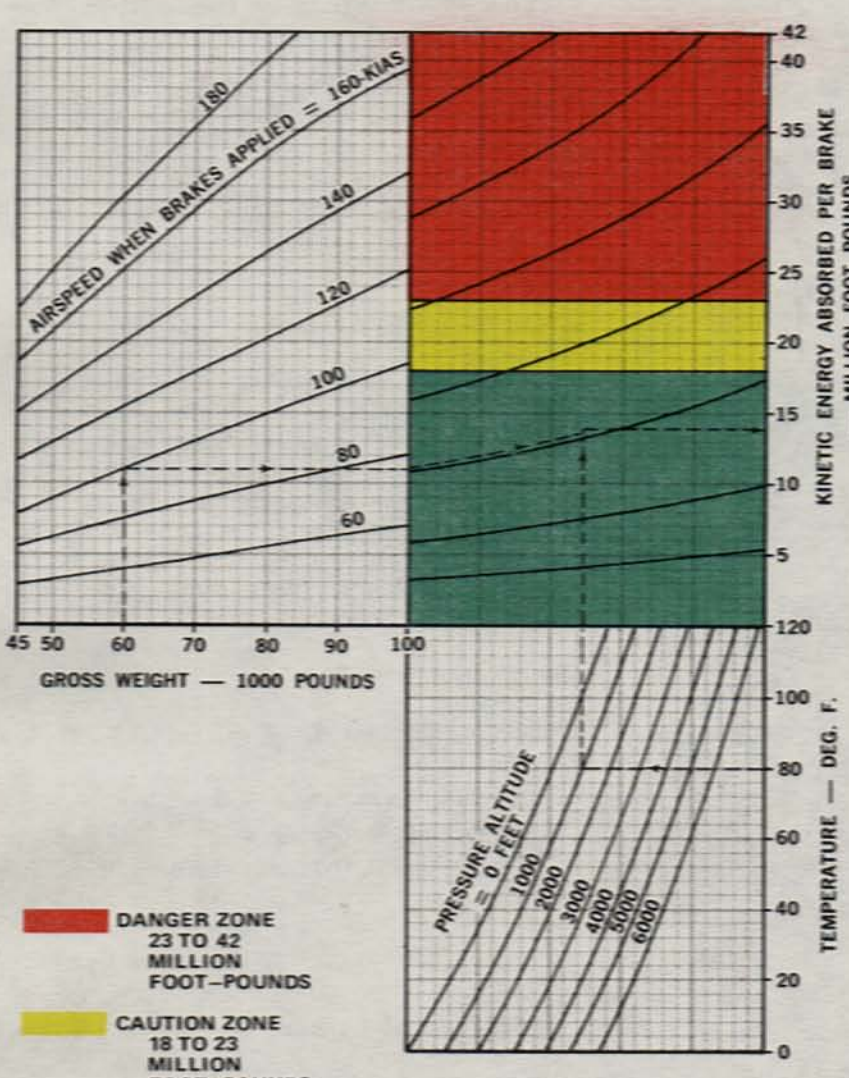
SPEED BRAKE LIMITS.

1. Speed brake operation is limited to 600 KIAS or mach 2.0, whichever is less.
2. With ECM pods installed on weapon bay doors, do not extend the speed brake at speeds in excess of 500 KIAS or mach 1.20, whichever is less.
3. No evaluation has been made with speed brakes and weapons bay doors open; therefore, do not attempt simultaneous operation.
4. For speed brake limits with external stores installed refer to "Stores Limitations" this section.

Brake Energy Limits

DATA BASIS: FLIGHT TEST
DATE: 30 MARCH 1973

CONDITIONS:
FLAPS/SLATS/SPOILERS – EXTENDED
WING SWEEP – 26 DEGREES OR LESS
ONE CONTINUOUS BRAKE APPLICATION



DANGER ZONE
23 TO 42 MILLION FOOT–POUNDS

CAUTION ZONE
18 TO 23 MILLION FOOT–POUNDS

NORMAL ZONE
0 TO 18 MILLION FOOT–POUNDS

★

NOTE:
- FLIGHT CREWS WILL MAKE NECESSARY ENTRIES IN FORM 781 IF BRAKE LIMITS ARE EXCEEDED.

THE FOLLOWING INFORMATION EXPLAINS ACTION TO BE TAKEN WHEN A STOP IN THE DANGER, CAUTION, OR NORMAL ZONE IS PERFORMED.

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.
5. ★ If possible, delay engine shutdown until arrival of fire-fighting equipment. If immediate evacuation is required, shut down engines as required. In either case, after engine shutdown, evacuate aircraft immediately. Leave immediate vicinity, keeping forward of the aircraft.

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.

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Figure 5-10.

CANOPY HATCH OPERATING SPEED.

Do not open canopy hatch or taxi with the canopy hatch open when the relative wind is in excess of 60 knots.

TERRAIN FOLLOWING RADAR OPERATION.

Terrain following radar operation is limited to the following:

1. Strict observance of minimum TF operation airspeed (Appendix I).
2. Wing sweep angles of 26 to 72 degrees.
3. Blind letdown to an initial clearance of 1000 feet (After level-off at 1,000 feet, the desired clearance can be selected.)
4. A minimum altitude of 500 feet when in manual mode using the E scope only.
5. Bank angles of 10 degrees or less in either manual or auto TF whenever terrain clearance cannot be assured.
6. Set clearances for each route segment, during night or IFR conditions, that will be at least 200 feet higher than any obstacle in the flight path that may not provide a reliable radar return.

Note

The 200 feet clearance should not normally be used at night or in IFR conditions over mountainous terrain.

7. Operation with external stores as stated under "Stores Limitations," this section.

In addition to the above limitations, do not attempt or continue auto or manual TF operation if any of the following flight control system malfunctions exist:

8. Any known pitch trim malfunction or any pitch axis caution lamp that will not reset.

Note

For a pitch channel or pitch damper caution lamp that will reset, verify that the lamp does not come on during an intentionally induced fly-up maneuver before continuing TF operation.

9. The yaw channel caution lamp will not reset.
10. The TF fly-up off caution lamp on.
11. The reference not engaged caution lamp will not reset.

CAUTION

These limits are peculiar to TF operation. Other emergency procedures regarding caution lamps must be followed at all times.

SLIP ON FERRY WING TIPS.

On aircraft with ferry wing tips installed, observe the following limitations:

1. Do not exceed 26 degrees wing sweep.
2. Do not exceed 330 KIAS or 0.95 mach, whichever is less.
3. TFR operations are prohibited.
4. Refer to figure 5-11 for limit maneuver load factors with ferry tips installed.

STORES LIMITATIONS.

The authorized store loadings are identified in figure 5-12 except for SUU-21/A. The table refers to figures 5-13 through 5-22, as appropriate, for maximum values of airspeeds, load factors and roll rates allowed for carriage of stores in addition to release and jettison limits and specific release data. Other more restrictive limits which apply to the aircraft must be observed and the miscellaneous stores limits must be used in conjunction with figure 5-12.

SUU-21/A LIMITS.

1. Loading configurations for the SUU-21/A are as follows:
   a. Pylons on stations 3, 4, 5 and 6.
      (1) Four SUU-21/A's
      (2) Two SUU-21/A's—Stations 3 and 4;
          Stations 4 and 5;
          Stations 3 and 5;
          Stations 4 and 6.
      (3) One SUU-21/A—Station 3, 4, 5, or 6.
   b. Pylons on Stations 3 and 6 with Stations 4 and 5 clean.
      (1) Two SUU-21/A's—Stations 3 and 6.
      (2) One SUU-21/A—Station 3 or 6.
2. Carriage Limits.
   a. Airspeed.
      (1) 630 KIAS to 19,000 feet.
      (2) 1.3 mach above 19,000 feet.
   b. Load Factor.
      (1) Plus 3.83 to minus 1.0 "g" symmetric.
      (2) Plus 3.0 to minus 0.8 "g" unsymmetric.

Limit Maneuver Factors - Slip-On Ferry Tips Installed

DATA BASIS: ESTIMATED
DATE: 6 JUNE 1969

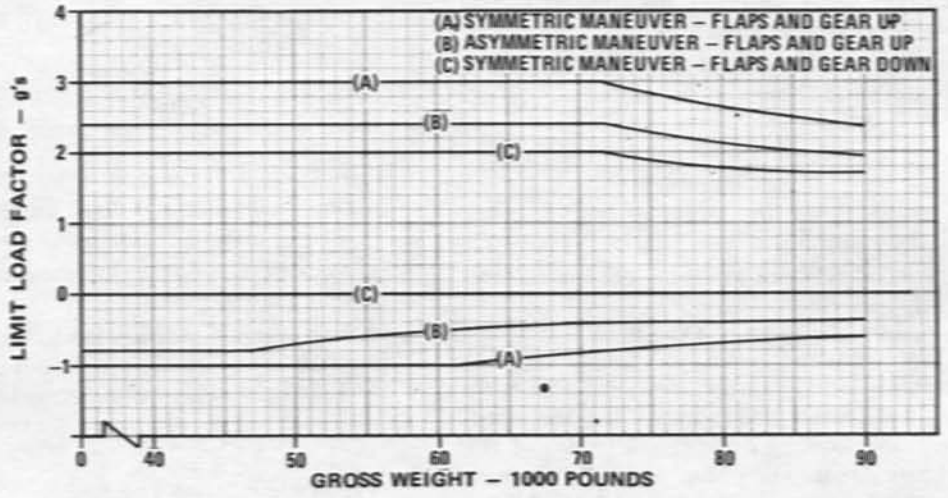

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Figure 5-11.

c. Wing Sweep.
   (1) 16 to 68 degrees.

3. Release Limits.
   a. BDU-33B/B.
      (1) Airspeed: .95 mach to 15,000 feet.
      (2) Wing Sweep: 26 to 60 degrees.
      (3) Pitch Angle: Plus 45 to minus 45 degrees.
      (4) Bank Angle: 5 degrees.
      (5) Roll Rate: Zero.
      (6) Load Factor: Plus 4 to plus 0.707 "g."
   b. MK 106 MOD 1.
      (1) Airspeed: 560 KIAS to 7,000 feet; .95 mach between 7,000 and 15,000 feet.
      (2) Wing Sweep: 26 to 60 degrees.
      (3) Pitch Angle: Plus 45 to minus 20 degrees.
      (4) Bank Angle: 5 degrees.
      (5) Roll Rate: Zero.
      (6) Load Factor: Plus 4 to minus 0.9 "g."
4. Emergency Jettison.
   a. 250 KIAS below 10,000 feet.
   b. Flaps and slats up or down.
   c. Wing Sweep: 16 to 26 degrees.
   d. Speed brake retracted.

Note

- Maximum wing sweep for 45 degree climb or 45 degree climb plus 4 "g" pull up is 50 degrees.
- Angle-of-attack must be less than 10 degrees for release of munitions.
- SUU-21/A doors must be closed above .95 mach.

MISCELLANEOUS STORES LIMITS.

Weapons.

1. MK-82 Snakeyes (HD) with MK-15 MOD, —1 and —2 fins are restricted from use.
2. Only MK-82 Snakeyes (HD) with MOD 3, 3A, and 4 fins will be used.
3. MK-82 Snakeyes (HD) with MOD 3, 3A, and 4 fins will be used in the high drag mode only to a maximum airspeed of 500 KIAS.

Pylons.

1. On aircraft 6, 10 and 13, flight with outboard pivot pylons is prohibited until T.O. 1F-111-876 has been complied with.

2. Fixed pylons must not be jettisoned, except in case of emergency. Refer to Section III.
3. When only pivoting pylons are attached, the clean aircraft airspeed and load factor limits apply.

Racks.

1. MAU-12C/A, BRU-3A/A and AERO-3B are the only authorized racks.
2. Whenever BRU-3A/A racks are installed, pivoting pylons must have damper installed.
3. All weapon bay installations will use MAU-12C/A racks with bomb lug chocks installed.

Tanks.

1. Outboard tanks must be released before using fuel from inboard tanks.
2. All tanks must be released prior to weapon release.

Internal and External Combined Loading.

1. Nuclear weapon bay loadings shown in the table may be combined with any external nuclear weapon or tank pylon loading designated, provided the most restrictive limits are observed. The weapons or tanks on the pylons must be dropped before the weapon bay loadings are dropped. In any event, do not open the weapon bay doors above mach 0.90 with external stores installed.
2. Any external load shown in the table may be combined with the weapon bay gun as long as the most restrictive limits are observed.

Carriage.

WARNING

Failure to observe the maximum allowable roll rates during rolling maneuvers may result in loss of the stores. With heavy store loading, excess roll rate may cause structural damage to or failure of the weapons racks, pylons or wing-pylon attachment.

Release.

WARNING

Normal acceleration of the aircraft at weapon release must be held constant or increased for two seconds after weapons have been released to prevent weapons from striking the aircraft. Roll maneuvers within one second after last store is separated, in a particular drop, are prohibited.

CAUTION

The use of release modes other than those authorized on the "Release and Jettison Limits" charts (especially for weapon cluster or heavy single store loadings) could result in extreme lateral loading asymmetry or hung store configurations. Damage to, or failure of racks, pylons or wing pylon attachments may occur.

Bomb clusters should be released in the authorized modes in the following manner:

1. Select stations in symmetric pairs.
2. Set intervalometer and number-of-pulses to completely unload the selected stations.
3. Depress weapon release button until all selected stores are released.
4. Step singles release mode is authorized for four (4) weapon clusters if step released with no more than one store asymmetry between opposite stations.

Asymmetric Store Load.

Sufficient roll control is available to permit flight and landing with up to and including full wing asymmetry, if necessary, for all authorized store loadings. Most single store asymmetric loads (exceptions noted below) and any fuel load asymmetry between stations 4 and 5 or fuel load asymmetry of 2000 pounds or less on other stations, require no limits in addition to the normal carriage limits.

Asymmetric Store Load — Flight Limitations. The following configurations require limits more restrictive than the normal carriage limits.

1. For any asymmetry of a loading that includes a six store cluster load: or dual station asymmetry of any four store cluster load: or a fuel load asymmetry greater than 2000 pounds between stations 2 and 7 or 3 and 6; observe:
   a. Airspeed—Sweep wings to 26 degrees and observe the airspeed carriage limit for that takeoff store loading configuration.
   b. Load Factors:
      (1) Symmetric—Zero to +2 "g".
      (2) Asymmetric—Nominally one "g".
      (3) Limit bank angle to less than 30 degrees. (Avoid abrupt control inputs.)
   c. If moderate to heavy turbulence is encountered, reduce speed to 300 KIAS or 0.60 mach, whichever is less.
   d. If weapons are loaded on stations inboard of the asymmetric fuel load, and are to be dropped, jettison the tanks using the "Fuel Tank Jettison Procedure", Section I.

2. For a single station asymmetry of any four store cluster load, observe, in addition to normal carriage limits:
   a. Symmetric—0 to 4 "g" (3 "g" above 70,000 pounds gross weight)
   b. Asymmetric—0 to 3 "g" (2 "g" above 70,000 pounds gross weight)
   c. Limit roll rate to 70 degrees/second, avoiding abrupt control inputs.
3. With BDU-8/B, BDU-18/B, MK-84, or M-118 in a takeoff load of one each on stations 3, 4, 5, and 6:
   a. A single station asymmetry imposes no limit in addition to the normal carriage limits.
   b. A dual station asymmetry imposes the limits of the preceding item 1.

Asymmetric Store Load – Selective Jettison. If selective jettison is desired, to relieve an asymmetric condition, observe:

1. Wing sweep—26 degrees.
2. Attitude—Straight and level.
3. Angle-of-attack—Not to exceed 10 degrees.
4. Flaps—Up.
5. Release—(For release, refer to the following items a, b, c, or d, as appropriate).
   a. Fuel tanks, empty or with more than 1800 pounds of fuel—Release at 300 KIAS or less, below 10,000 feet.

WARNING

Do not jettison tank(s) with 50-1800 pounds of fuel remaining. To do so may result in tank(s) colliding with the aircraft.

   b. Single weapon loads on pylon (Jettison by firing MAU-12C/A rack) release at 0.80 mach or less.
   c. All takeoff loadings (Flaps up or down) or empty BRU-3A/A rack—Release at 250 KIAS or less below 10,000 feet.
   d. Cluster load on BRU-3A/A (Full rack of six weapons)—Release at 0.60 mach or less below 25,000 feet.—No jettison approved for partially, in flight, down loaded rack.

Hung Store Jettison.

Non-nuclear loadings other than takeoff loadings, i.e., hung bombs or partial down loads, should be retained if at all possible. If emergency jettison is deemed necessary, refer to the associated "Release and Jettison Limits" chart, this section.

WARNING

- Emergency jettison at wing sweep angles other than 26° may result in damage to the aircraft.
- Emergency jettison of hung bombs, partial down loadings, or any loadings other than the authorized takeoff loadings may result in damage to the aircraft.

Weapon Bay Gun.

1. Refer to figure 5-23 for weapon bay gun carriage and firing envelopes.
2. Do not fire the weapon bay gun until after accomplishment of T.O. 1F-111-798.

TFR Operation With External Stores.

TFR operation with stores is permitted as noted for specific stores loadings on figure 5-12.

Authorized Stores Loading Table

Date: 30 March 1973

WARNING

Weapons configurations not specifically stated in this table are unauthorized. Refer to "Miscellaneous Stores Limits," this section.

Stores	STATIONS										Notes			Fig. No.		Sht. No.	
	1	2	3	4	L	R	5	6	7	8	SpBrake	TFR	ECMPod	Car.	Rel.	Car.	Rel.
			2	1			1	2			3, 5	1	2	5-13	5-13	1	2
			1	1			1	2			3, 5	1	2	5-13	5-13	1	2
			1	1			1	1			3, 5	1	2	5-13	5-13	1	2
			2	OP			OP	2			3, 5	1	2	5-13	5-13	1	2
AIM-9B			1	OP			OP	2			3, 5	1	2	5-13	5-13	1	2
			OP	1			1	2			3, 5	1	2	5-13	5-13	1	2
			1	1			1	OP			3, 5	1	2	5-13	5-13	1	2
			OP	1			1	OP			3, 5	1	2	5-13	5-13	1	2
			OP	OP			1	OP			3, 5	1	2	5-13	5-13	1	2
AIM-9B			M	M			A	A			3, 5	1	2	5-14	5-14	1	2
mix			OP	M			A	OP			3, 5	1	2	5-14	5-14	1	2
TDU-11/B			M	OP			OP	A			3, 5	1	2	5-14	5-14	1	2
					1						3, 4	1	2	5-15	5-15	1	2
					1	1					3, 4	2	2	5-15	5-15	3	4
						1					3, 4	1	2	5-15	5-15	1	2
BDU-8/B			1					1			2, 5	1	1	5-15	5-15	5	6
or			OP					1			2, 5	1	1	5-15	5-15	5	6
BDU-18/B			1	1			1	1			2, 5	1	1	5-15	5-15	5	6
			OP	1			1	1			2, 5	1	1	5-15	5-15	5	6
			OP	1			1	OP			2, 5	1	1	5-15	5-15	5	6
			OP	OP			1	OP			2, 5	1	1	5-15	5-15	5	6
					1						3, 4	1	2	5-16	5-16	1	2
					1	1					3, 4	2	2	5-16	5-16	1	2
BDU-12/B						1					3, 4	1	2	5-16	5-16	1	2
or			1					1			2, 5	1	1	5-16	5-16	1	3
BDU-19/B			OP					1			2, 5	1	1	5-16	5-16	1	3
or			1	1			1	1			2, 5	1	1	5-16	5-16	1	3
BDU-38/B			OP	1			1	1			2, 5	1	1	5-16	5-16	1	3
			OP	1			1	OP			2, 5	1	1	5-16	5-16	1	3
			OP	OP			1	OP			2, 5	1	1	5-16	5-16	1	3
CBU-24H/B or			4*	4*			4*	4*			2, 5	2	1	5-19	†	1	†
CBU-29H/B or			P	4*			4*	P			2, 5	2	1	5-19	†	1	†
CBU-49H/B or CBU-52H/B or CBU-58/B or CBU-71/B			6					6			2,5	2	1	5-19	5-19	1	3

★

Figure 5-12. (Sheet 1)

Authorized Stores Loading Table (Cont'd)

Stores	STATIONS										Notes			Fig. No.		Sht. No.	
	1	2	3	4	L	R	5	6	7	8	SpBrake	TFR	ECMPod	Car.	Rel.	Car.	Rel.
MK-82 or MK-82 Snakeye (LD)			6	6			6	6			2, 5	2	1	5-17	5-17	6	5
			OP	6			6	OP			2, 5	2	1	5-17		6	5
			4*	4*			4*	4*			2, 5	2	1	5-17	†	3	†
			6	4			4	6			2, 5	2	1	5-17	5-17	3	5
			P	4*			4*	P			2, 5	2	1	5-17	5-17	3	5
			6					6			2, 5	2	1	5-17	5-17	1	2
MK-82 Snakeye (HD) or MK-36			4*	4*			4*	4*			2, 5	2	1	5-17	†	3	†
			6	4			4	6			2, 5	2	1	5-17	5-17	3	4
			P	4*			4*	P			2, 5	2	1	5-17	5-17	3	4
			6					6			2, 5	2	1	5-17	5-17	1	2
MK-84			1					1			2, 5	2	1	5-18	5-18	1	2
			OP					1			2, 5	2	1	5-18	5-18	1	2
			1	1			1	1			2, 5	2	1	5-18	5-18	1	2
			OP	1			1	1			2, 5	2	1	5-18	5-18	1	2
			OP	1			1	OP			2, 5	2	1	5-18	5-18	1	2
			OP	OP			1	OP			2, 5	2	1	5-18	5-18	1	2
M-117 with MAU-103A/B fins			4*	4*			4*	4*			2, 5	2	1	5-19	†	4	†
			P	4*			4*	P			2, 5	2	1	5-19	†	4	†
			6					6			2, 5	2	1	5-19	5-19	1	2
			6	6			6	6			2, 5	2	1	5-19	†	5	†
			P	6			6	P			2, 5	2	1	5-19	†	5	†
M-117R or M-117D			4*					4*			2, 5	2	1	5-19	5-19	6	7
			4*	4*			4*	4*			2, 5	2	1	5-19	†	4	†
			P	4*			4*	P			2, 5	2	1	5-19	†	4	†
M-118			1					1			2, 5	2	1	5-20	5-20	2	3
			P					1			2, 5	2	1	5-20	5-20	2	3
			1	1			1	1			2, 5	2	1	5-20	5-20	1	3
			P	1			1	1			2, 5	2	1	5-20	5-20	1	3
			P	1			1	P			2, 5	2	1	5-20	5-20	1	3
			P	P			1	P			2, 5	2	1	5-20	5-20	1	3
SUU-20A/A or SUU-20A/M or SUU-20B/A			OP	1			1	OP			3	1	2	5-21	5-21	1	2
			OP	OP			1	OP			3	1	2	5-21	5-21	1	2
			1	OP			OP	1			3	1	2	5-21	5-21	1	2
			OP	OP			OP	1			3	1	2	5-21	5-21	1	2

★

Figure 5-12. (Sheet 2)

Authorized Stores Loading Table (Cont'd)

Stores	STATIONS										Notes			Fig. No.		Sht. No.	
	1	2	3	4	L	R	5	6	7	8	SpBrake	TFR	ECMPod	Car.	Rel.	Car.	Rel.
Tanks or Tanks with Nuclear Weapons (BDU's)			T					T			1	2	2	5-22	5-22	2	4
			T			N		T			1	2	2	5-22	5-22	2	4
			T	N				T			1	2	2	5-22	5-22	2	4
			T	N			N	T			1	2	2	5-22	5-22	3	4
			T	T			T	T			1	2	2	5-22	5-22	3	4
			P	T			T	P			1	2	2	5-22	5-22	3	4
			T	T	N		T	T			1	2	2	5-22	5-22	3	4
			P	T	N		T	P			1	2	2	5-22	5-22	3	4
		T							T		1	2	2	5-22	5-22	1	4
TDU-11/B				1			1				3, 5	1	2	5-14	5-14	1	2
				OP			1				3, 5	1	2	5-14	5-14	1	2

Symbols

* — Downloading of slant-4 loadings to two stores remaining on BRU-3A/A rack outboard shoulder stations is authorized as a takeoff loading provided that the aircraft lateral symmetry is maintained.

† — No release envelope available.

P — Unloaded pylon required on that station.

OP — Pylon optional on that station.

4 — Mounted on BRU-3A/A rack (slant) outboard shoulder and bottom positions only.

6 — Fully loaded BRU-3A/A rack.

T — 600 gallon fuel tank.

N — Nuclear weapon or equivalent dummy (BDU).

A — AIM-9B

M — TDU-11/B

Blank — No pylon permitted on that station.

Notes

Speed Brake Usage.

1. Not allowed for this loading.
2. Authorized for nominal 1 "g" flight conditions up to 400 KIAS/0.90 mach or store carriage limitations (whichever is less).
3. Authorized within the clean aircraft speed brake limits.
4. Not allowed with weapon bay doors open.
5. Not allowed during separation of stores.

Figure 5-12. (Sheet 3)

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TFR Operations.

1. Authorized to clean aircraft TFR limits.
2. Authorized only within 10 degrees bank. No TFR above 91,500 pounds allowed.

ECM Pods.

1. Either or both allowed. Carriage limited to mach 1.30 and separation to mach 0.90 when forward pod installed.
2. Aft only allowed.

★

Figure 5-12. (Sheet 4)

Carriage Limits - AIM - 9B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 4 AND 5 WITH 1 OR 2 EACH ON 3 AND/OR 6, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6, OR;
1 OR 2 EACH ON 3 AND 6, W-W/O PYLONS ON 4 AND 5.

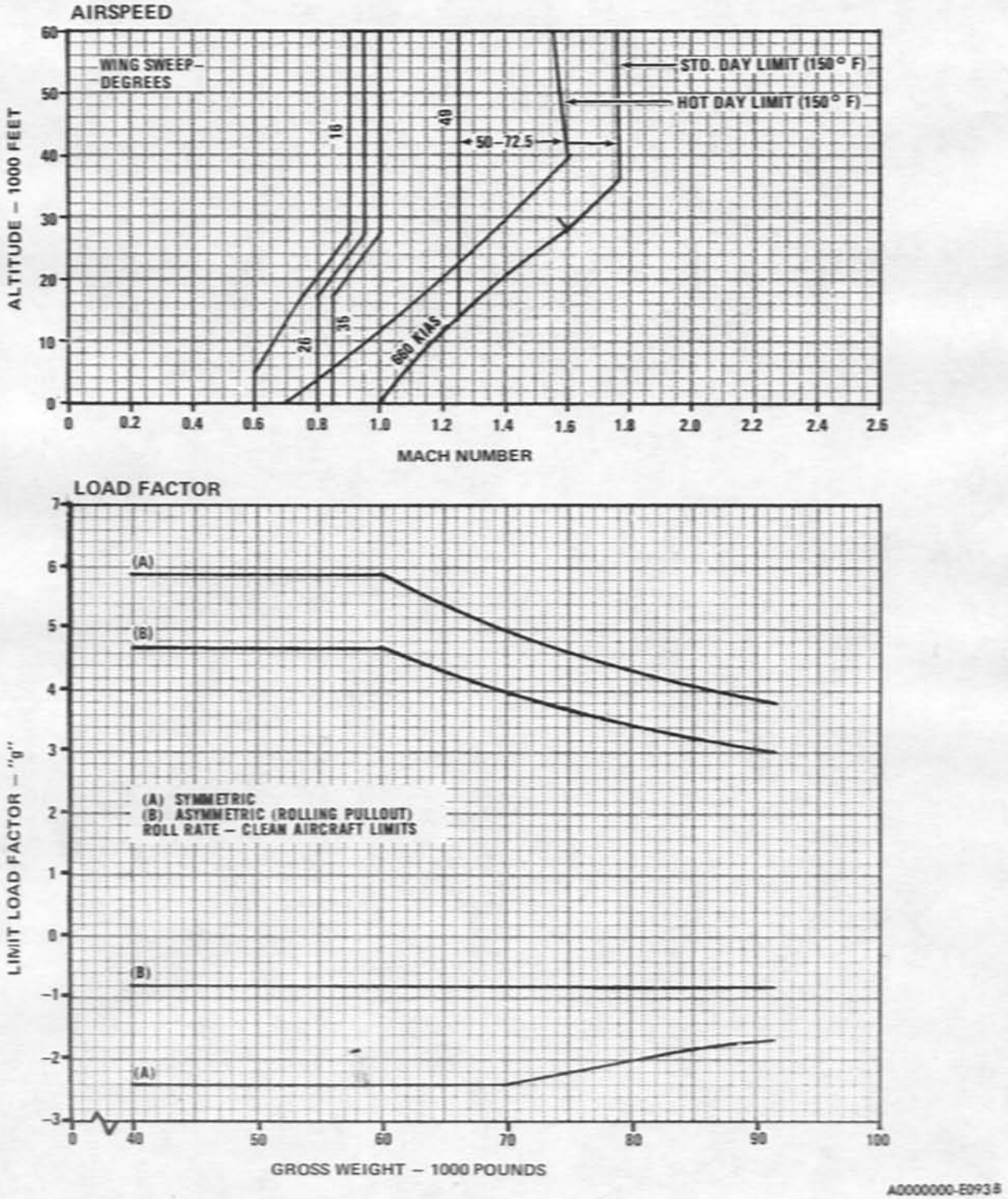


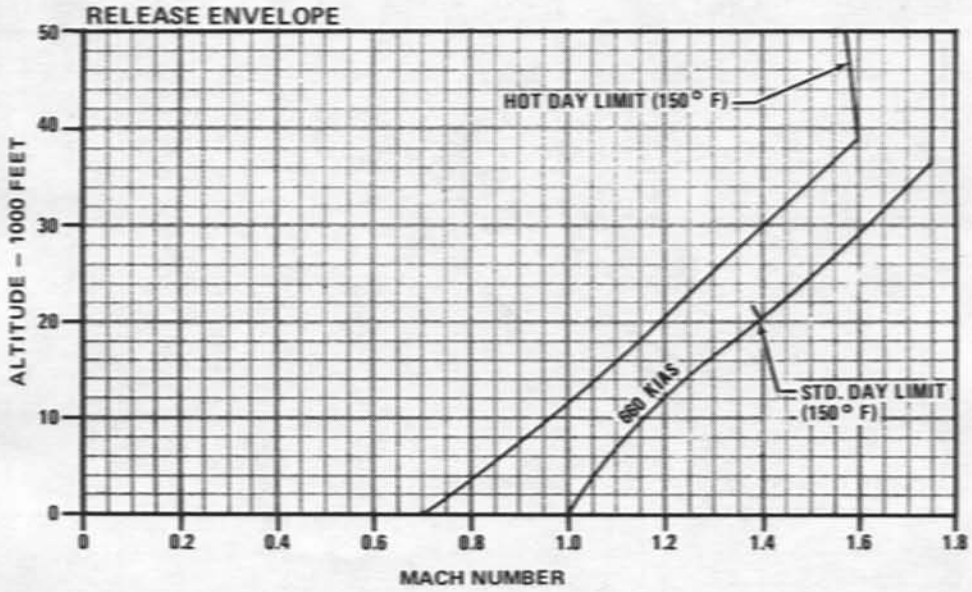
Figure 5-13. (Sheet 1)

Release and Jettison Limits - AIM - 9B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

1 EACH ON 4 AND 5 WITH 1 OR 2 EACH ON 3 AND/OR 6, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6, OR;
1 OR 2 EACH ON 3 AND 6, W-W/O PYLONS ON 4 AND 5.



FIRING LIMITATIONS

PARAMETERS	AIM-9B
WING SWEEP	26 DEGREES TO 60 DEGREES
DIVE ANGLE	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	UNLIMITED
ROLL RATE	70 DEGREES PER SECOND OR LESS
NORMAL "G"	+0.7 TO +4.0

FIRING SEQUENCE IS OUTBOARD PYLONS TO INBOARD PYLONS ONLY.

EMERGENCY JETTISON LIMITS –SAME AS NORMAL FIRING

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL

A0000000—E0948

Figure 5-13. (Sheet 2)

Carriage Limits - TDU - 11/B W-W/O AIM-9B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH TDU-11/B ON 4 AND 5, OR;
1 EACH TDU-11/B ON 4 OR 5; W-W/O PYLON ON THE OTHER STATION
1 EACH TDU-11/B ON 3 AND 4 WITH 1 EACH AIM-9B ON 5 AND 6, OR;
1 EACH TDU-11/B ON 3 OR 4 WITH 1 EACH AIM-9B ON 5 OR 6;
W-W/O PYLONS AT OTHER PIVOT STATIONS

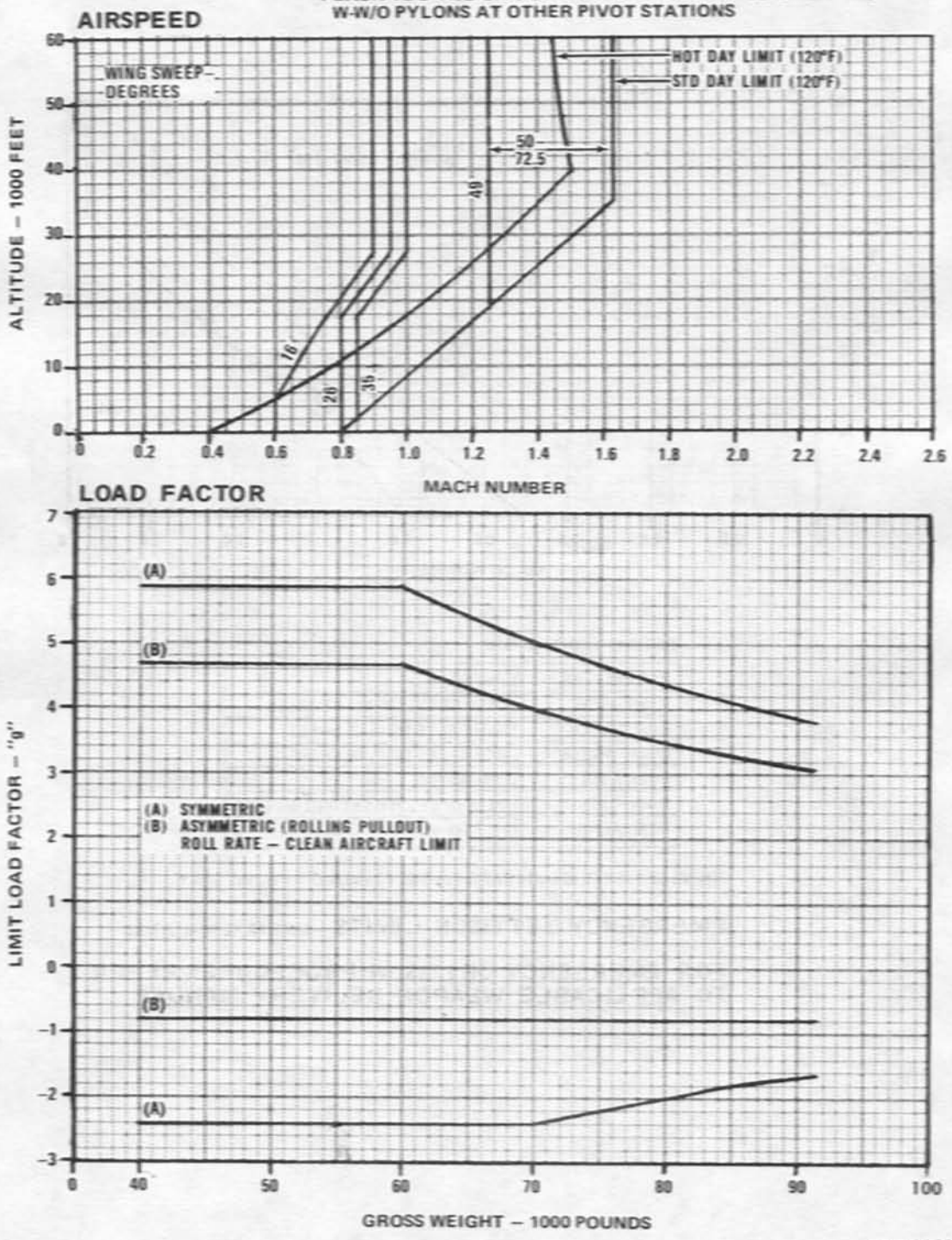


Figure 5-14. (Sheet 1)

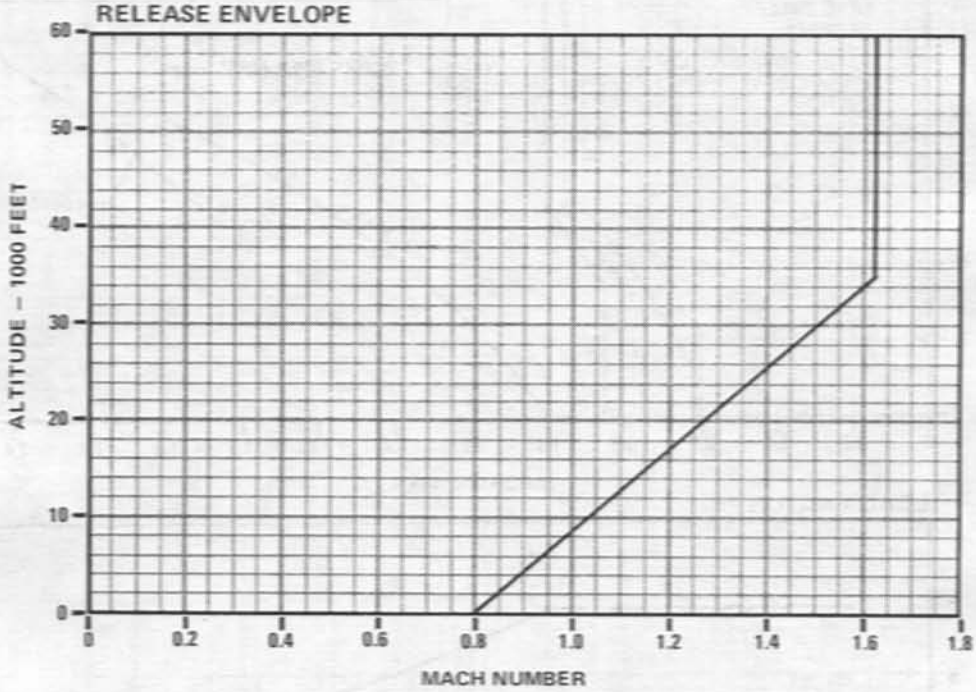
Release and Jettison Limits - TDU - 11/B W-W/O AIM-9B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

1 EACH TDU-11/B ON 4 AND 5, OR;
1 EACH TDU-11/B ON 4 OR 5, W-W/O PYLON AT THE OTHER STATION, OR;
1 EACH TDU-11/B ON 3 AND 4, WITH 1 EACH AIM-9B ON 5 AND 6, OR;
1 EACH TDU-11/B ON 3 OR 4, WITH 1 EACH AIM-9B ON 5 OR 6,
W-W/O PYLONS AT OTHER PIVOT STATIONS

RELEASE ENVELOPE



FIRING LIMITATIONS

PARAMETERS	TDU-11/B W-W/O AIM-9B
WING SWEEP	26 DEGREES TO 60 DEGREES
DIVE ANGLE	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	-60 DEGREES TO +60 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.7 TO +1.7
SPEEDBRAKE	RETRACTED

FIRING SEQUENCE IS OUTBOARD PYLONS TO INBOARD PYLONS ONLY.
DO NOT FIRE MISSILE WITH FORWARD ECM POD INSTALLED.

EMERGENCY JETTISON LIMITS –SAME AS NORMAL FIRING

A0000000—E1108

Figure 5-14. (Sheet 2)

Carriage Limits - BDU - 8/B or BDU - 18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 IN LEFT WEAPON BAY, OR; 1 IN RIGHT WEAPON BAY, OR;
1 IN LEFT WEAPON BAY PLUS GUN IN RIGHT WEAPON BAY

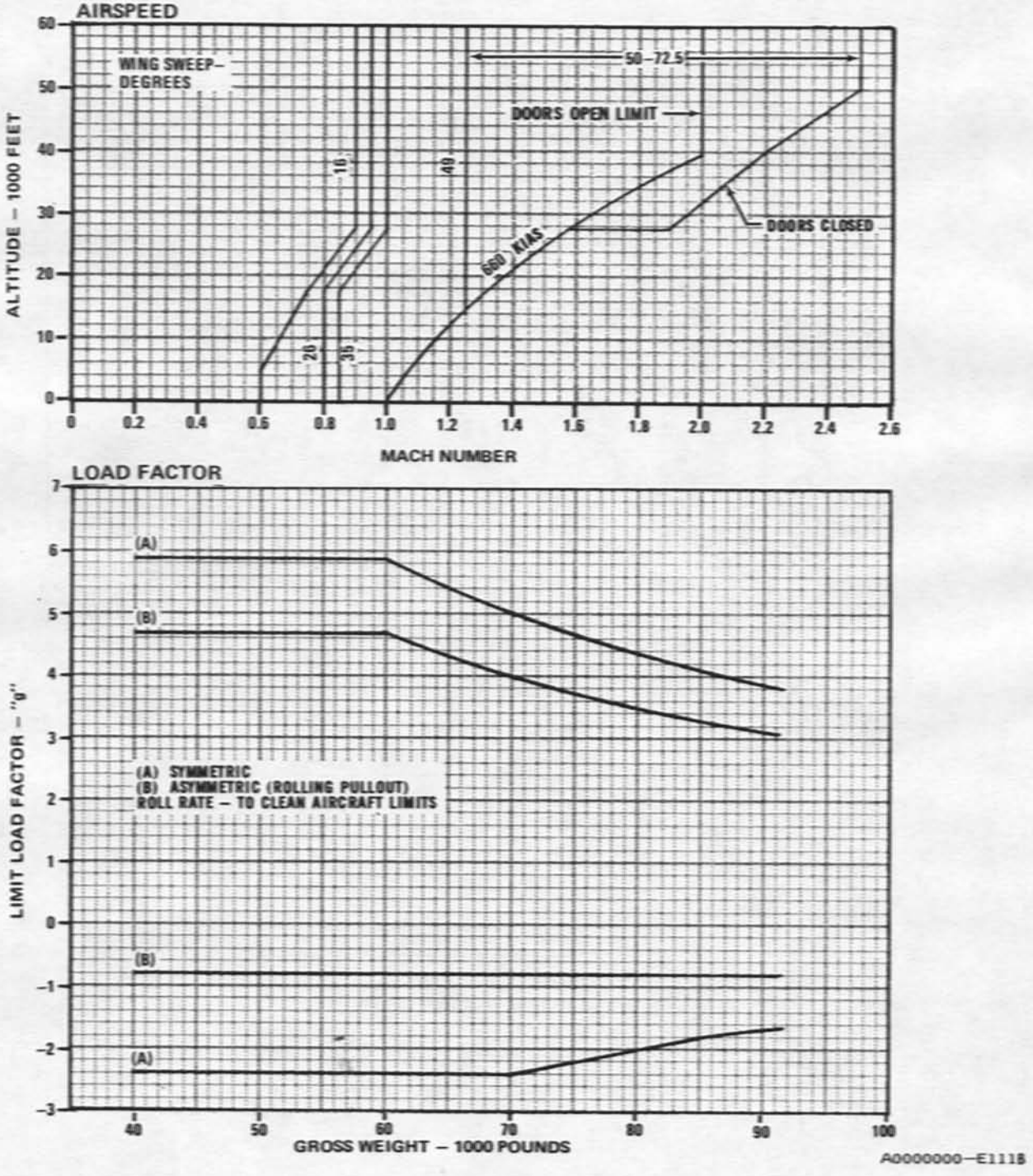
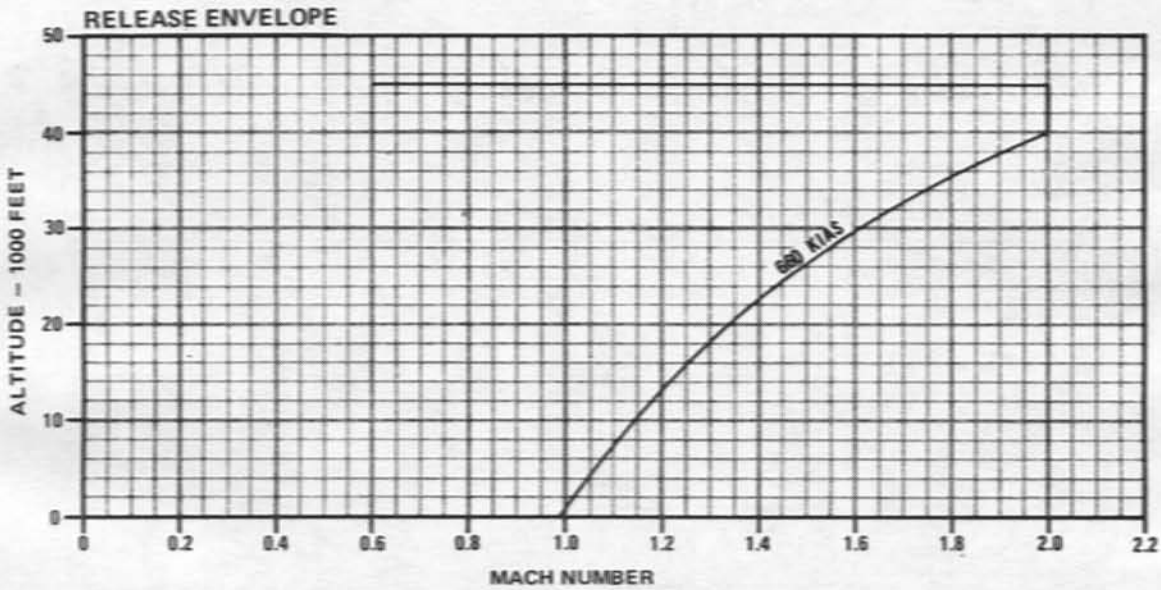


Figure 5-15. (Sheet 1)

Release and Jettison Limits - BDU - 8/B or BDU - 18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 IN LEFT WEAPON BAY, OR; 1 IN RIGHT WEAPON BAY, OR;
1 IN LEFT WEAPON BAY PLUS GUN IN RIGHT WEAPON BAY



RELEASE LIMITATIONS

PARAMETER	BDU-8/B OR BDU-18/B
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 72 1/2 DEGREES
DIVE ANGLE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +4.0

AUTHORIZED RELEASE MODE IS RELEASE SINGLE ONLY.

EMERGENCY JETTISON LIMITS –SAME AS NORMAL RELEASE

FOR EMERGENCY JETTISON PROCEDURES
REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000-E112C

Figure 5-15. (Sheet 2)

Carriage Limits - BDU-8/B or BDU-18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

1 EACH IN LEFT AND RIGHT WEAPON BAY

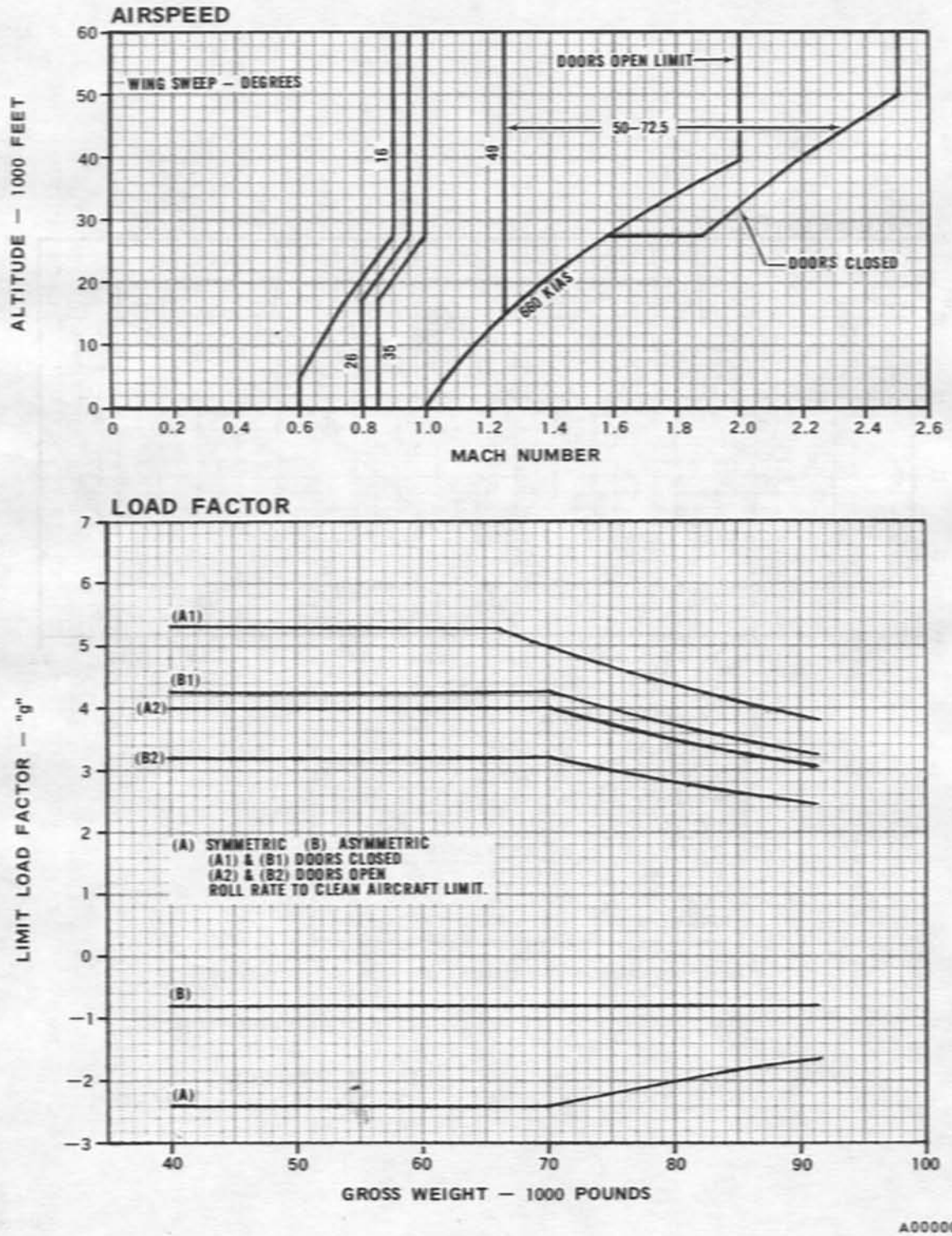


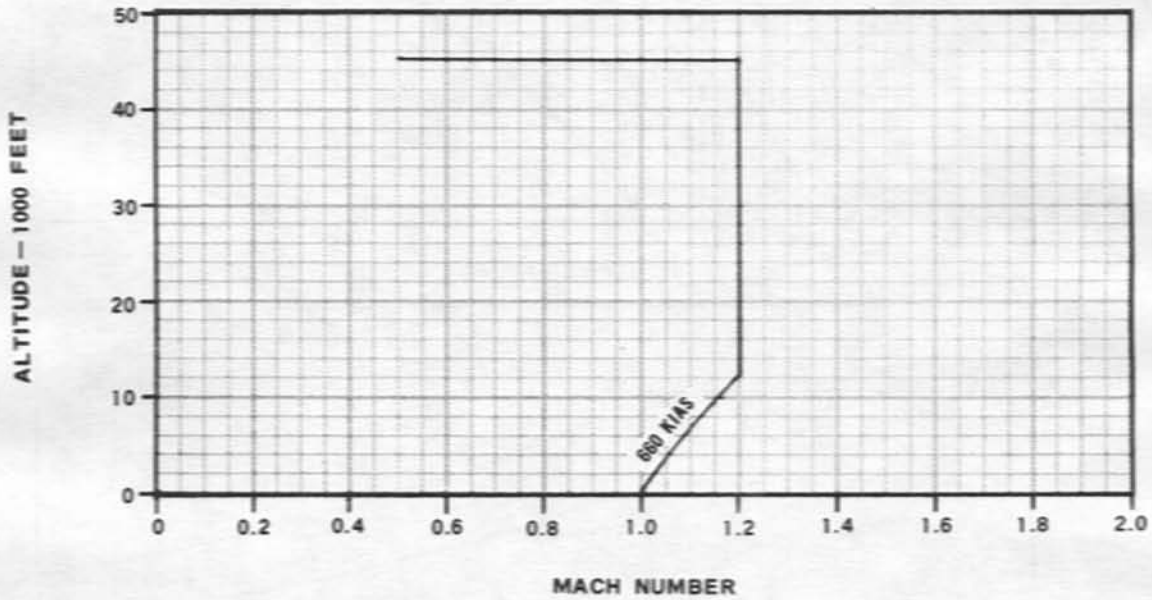
Figure 5-15. (Sheet 3)

Release and Jettison Limits - BDU-8/B or BDU-18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH IN LEFT AND RIGHT WEAPON BAY

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETER	BDU-8/B OR BDU-18/B
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 72-1/2 DEGREES
DIVE ANGLE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +4.0

AUTHORIZED RELEASE MODE
IS RELEASE SINGLE ONLY

FOR EMERGENCY JETTISON PROCEDURES, REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

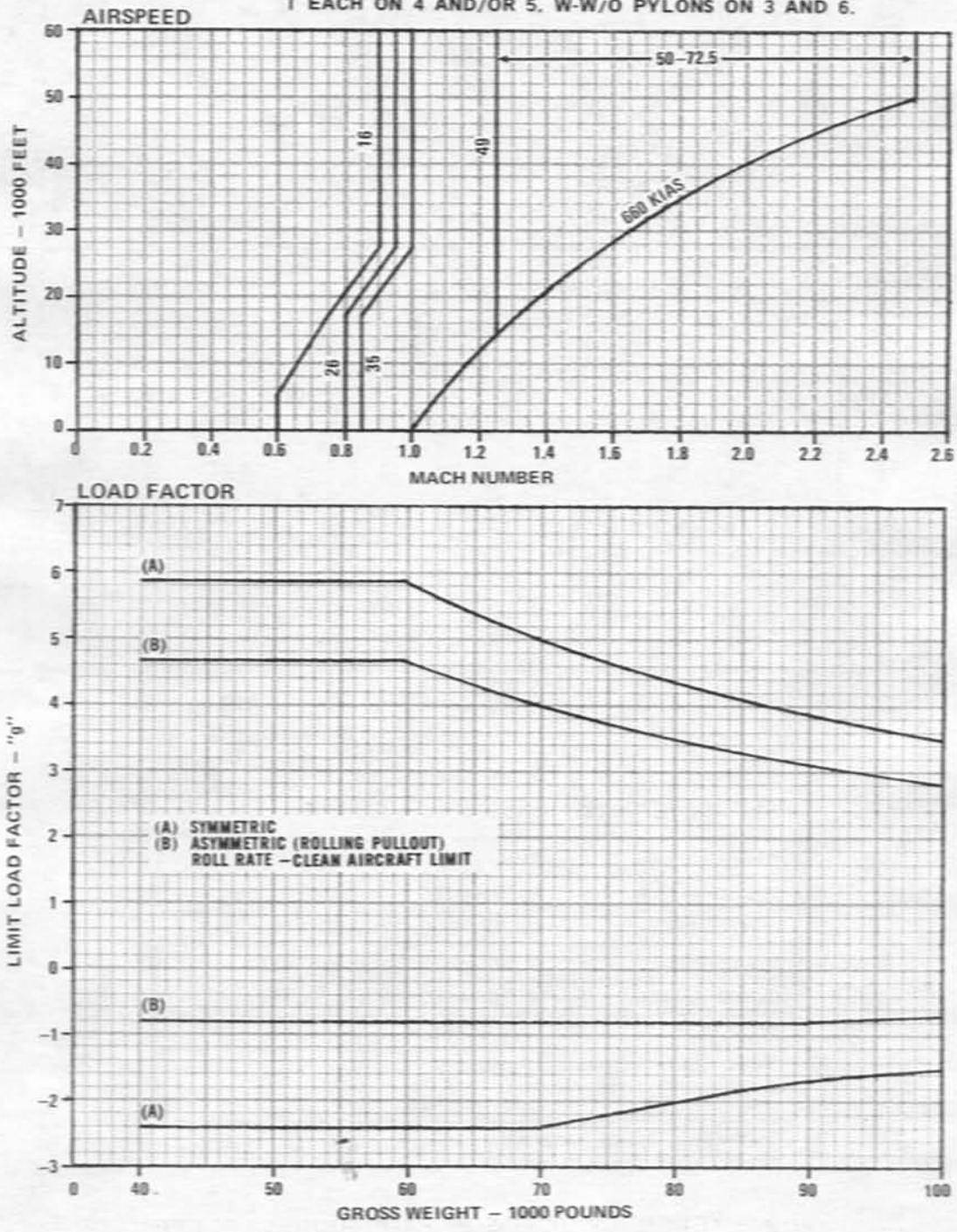
A0000000-E133

Figure 5-15. (Sheet 4)

Carriage Limits - BDU - 8/B or BDU-18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5, OR;
1 EACH ON 3, 4, 5 AND 6, OR;
1 EACH ON 4 AND 5 AND 3 OR 6, W-W/O PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6.



A0000000-E1138

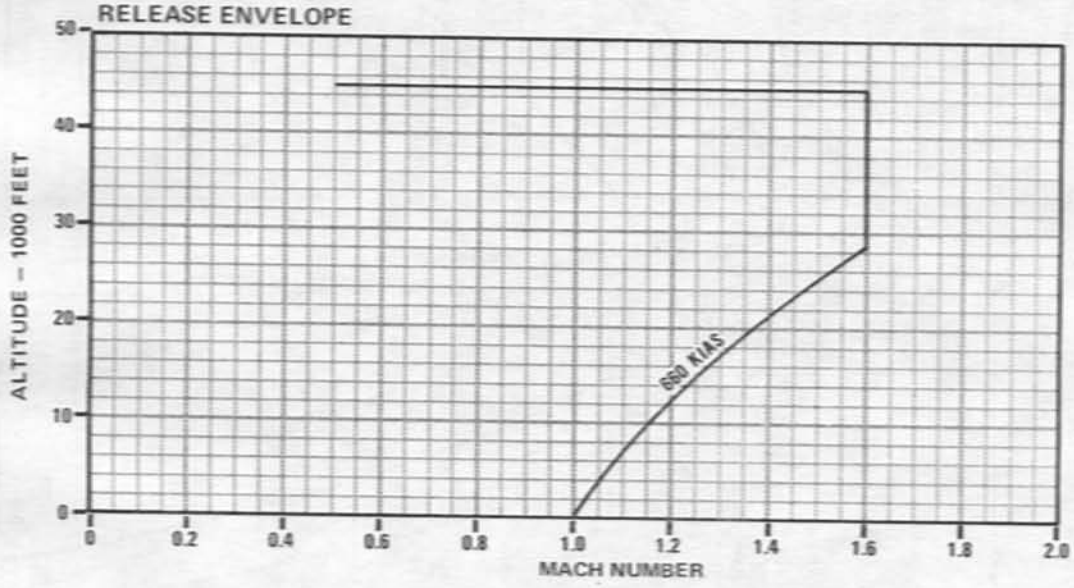
Figure 5-15. (Sheet 5)

Release and Jettison Limits - BDU - 8/B or BDU - 18/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5 OR;
1 EACH ON 3, 4, 5 AND 6, OR;
1 EACH ON 4 AND 5 AND 3 OR 6
W-W/O PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6,



RELEASE LIMITATIONS

PARAMETER	BDU-8/B OR BDU-18/B
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 60 DEGREES
SPEED OR MACH	AS SHOWN ABOVE
DIVE ANGLE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +4.0

- AUTHORIZED RELEASE MODE IS RELEASE SINGLE ONLY.
- RELEASE OF WEAPON IN THE PRESENCE OF ECM POD LIMITED TO MACH 1.30.

EMERGENCY JETTISON LIMITS –SAME AS NORMAL RELEASE

FOR EMERGENCY JETTISON PROCEDURES
REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000–E1148

Figure 5-15. (Sheet 6)

Carriage Limits - BDU - 12/B or BDU - 19/B or BDU - 38/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH IN LEFT AND/OR RIGHT WEAPON BAY, OR;
1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5, OR
1 EACH ON 3, 4, 5 AND 6, OR;
1 EACH ON 4 AND 5 AND 3 OR 6
W-W/O PYLON ON OTHER PIVOT STATION , OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6.

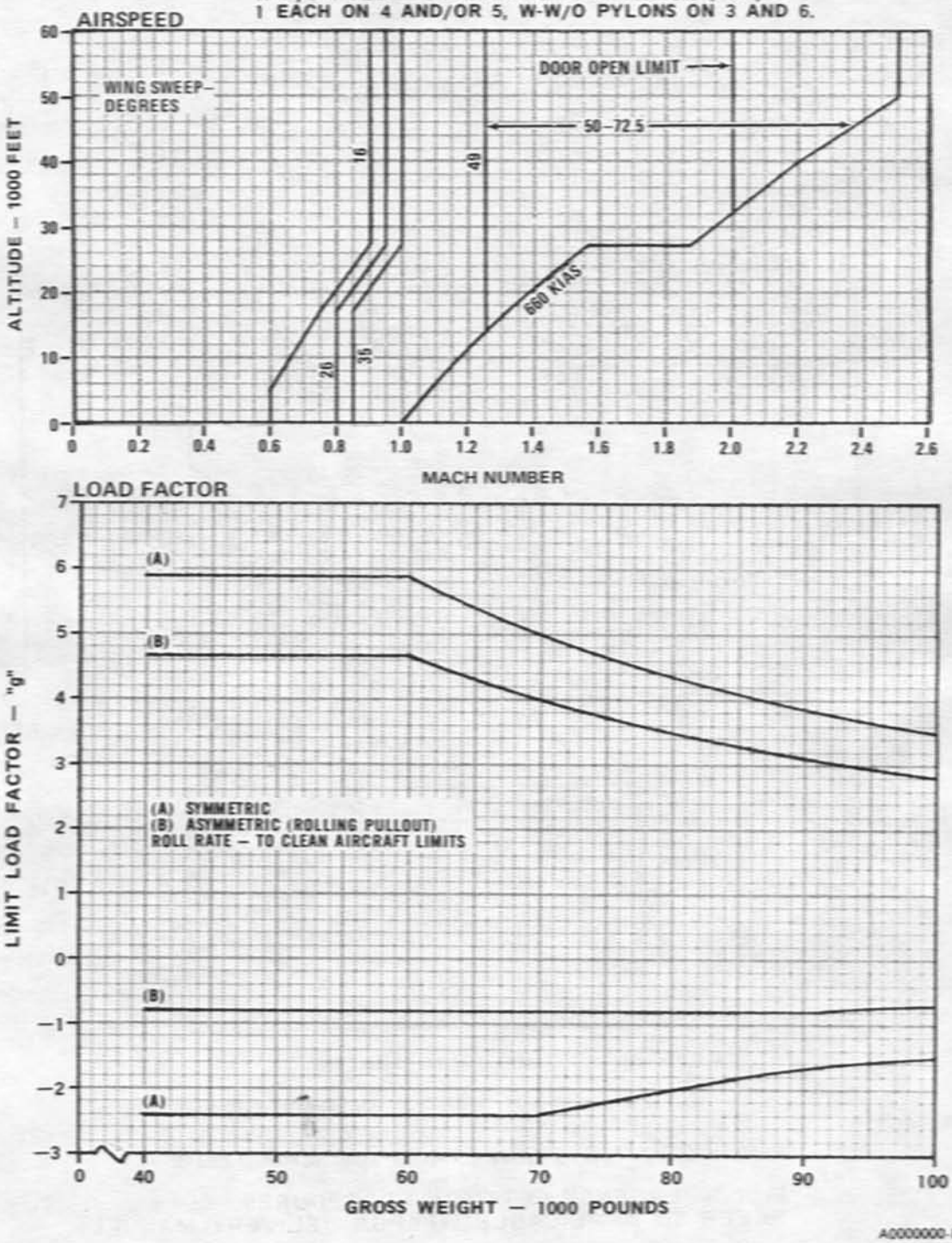
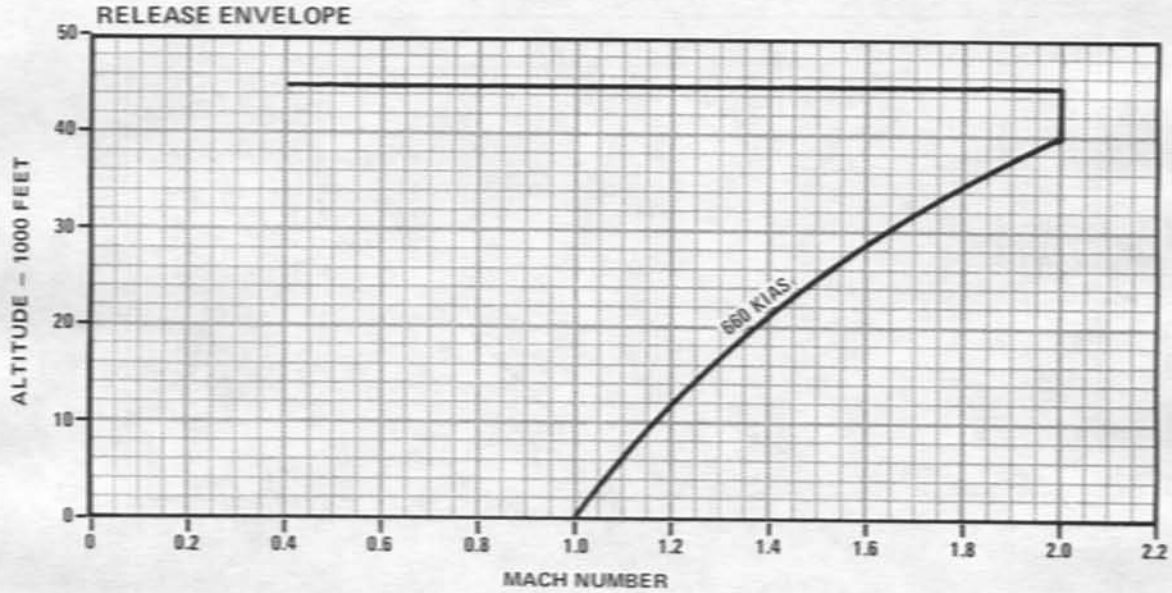


Figure 5-16. (Sheet 1)

Release and Jettison Limits - BDU - 12/B or BDU-19/B or BDU-38/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH IN L AND/OR R WPN BAY



RELEASE LIMITATIONS

PARAMETER	BDU-12/B OR BDU-19/B OR BDU-38/B
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 72 1/2 DEGREES
DIVE ANGLE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +4.0

- AUTHORIZED RELEASE MODE IS RELEASE SINGLE ONLY.
- NOT CLEARED FOR RELEASE WITH WEAPON BAY GUN INSTALLED

EMERGENCY JETTISON LIMITS –SAME AS NORMAL RELEASE

FOR EMERGENCY JETTISON PROCEDURES
REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

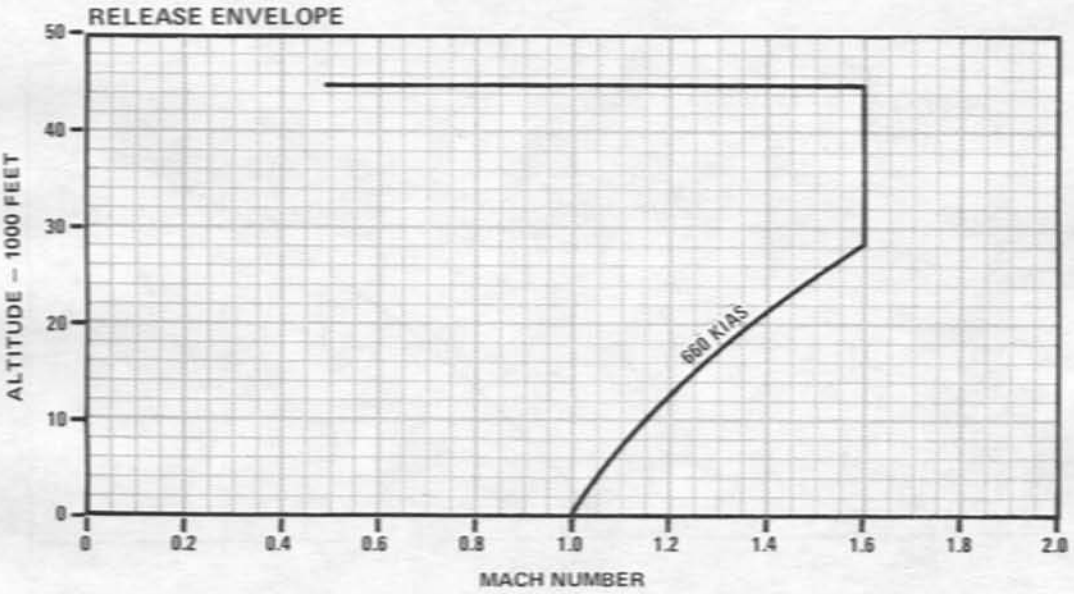
A0000000—E1178

Figure 5-16. (Sheet 2)

Release and Jettison Limits - BDU - 12/B or BDU - 19/B or BDU - 38/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 3, 4, 5 AND 6, OR;
1 EACH ON 4 AND 5 AND 3 OR 6
W-W/O PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6, OR;
1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5



RELEASE LIMITATIONS

PARAMETER	BDU-12/B OR BDU-19/B OR BDU-38/B
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 60 DEGREES
SPEED OR MACH	AS SHOWN ABOVE
DIVE ANGLE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +4.0

- AUTHORIZED RELEASE MODES IS RELEASE SINGLES ONLY.
- RELEASE OF WEAPON IN THE PRESENCE OF ECM PODS LIMITED TO MACH 1.30

EMERGENCY JETTISON LIMITS –SAME AS NORMAL RELEASE

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

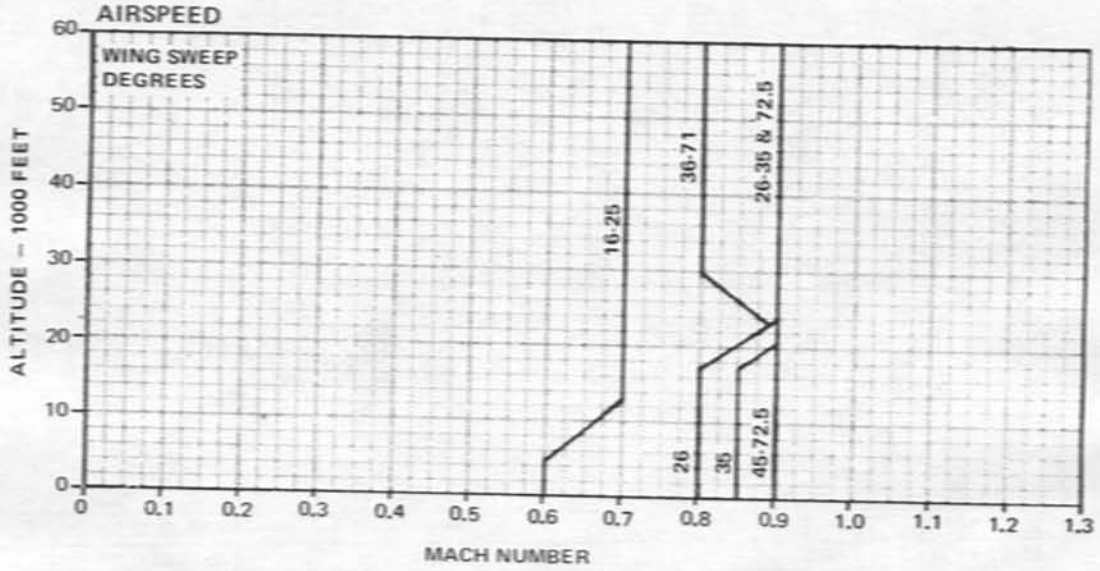
A0000000–E1168

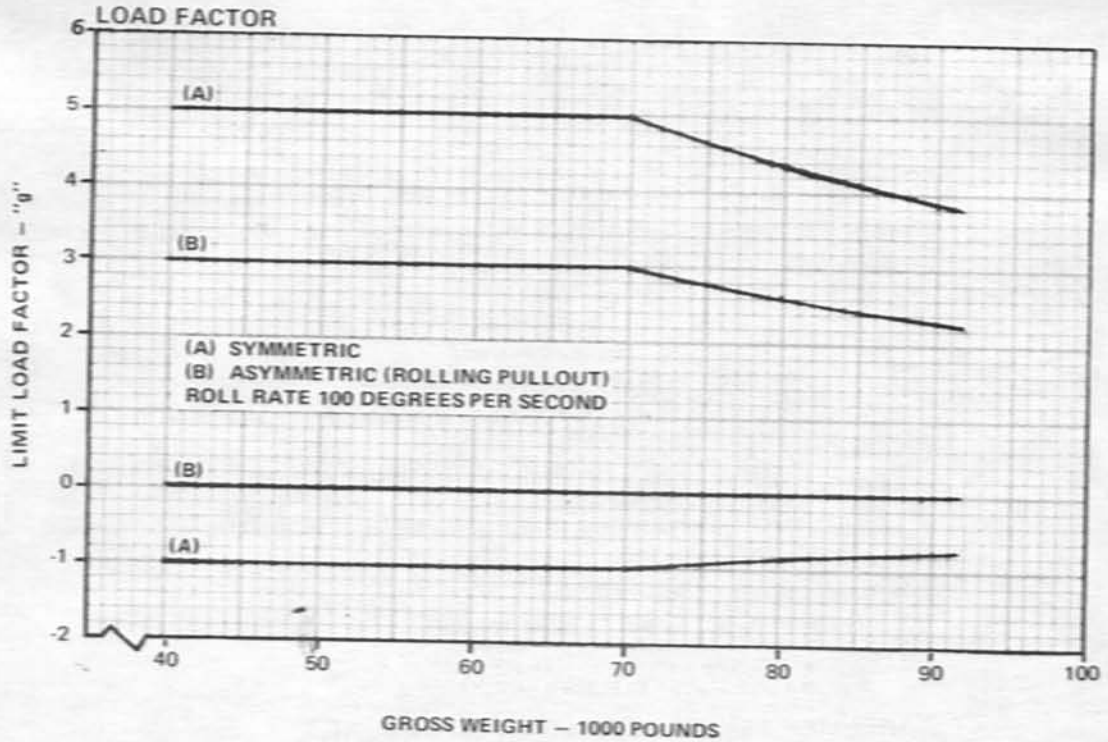
Figure 5-16. (Sheet 3)

Carriage Limits - MK - 82 or MK - 82 Snakeye (LD or HD) or MK-36

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3 AND 6
W/O PYLONS ON 4 AND 5





A0000000-E1018

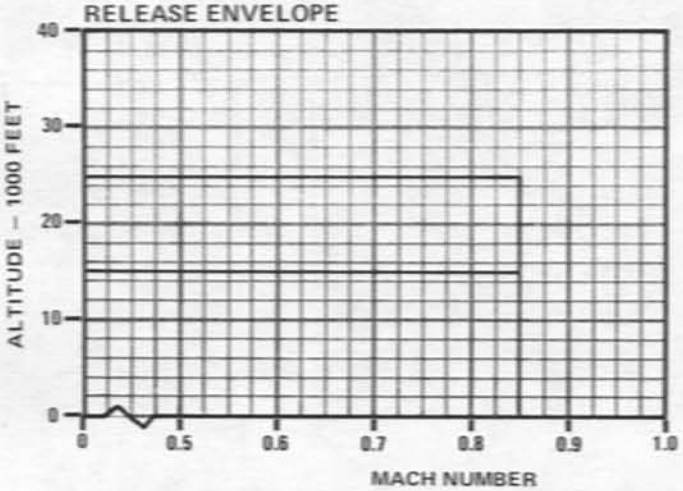
Figure 5-17. (Sheet 1)

Release and Jettison Limits - MK - 82 or MK-82 Snakeye (LD only)

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3 AND 6, W/O PYLONS ON 4 AND 5

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	LEVEL	AT ALTITUDE	DIVE OR CLIMB
SPEED BRAKE	RETRACTED	RETRACTED	RETRACTED
WING SWEEP	26 DEGREES TO 54 DEGREES	26 DEGREES TO 54 DEGREES	26 DEGREES TO 45 DEGREES
ALTITUDE–FEET	0 TO 15,000	15,000 TO 25,000	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 10 DEGREES	NONE	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 15 DEGREES	0 DEGREES TO 15 DEGREES	0 DEGREES TO 30 DEGREES
ROLL ANGLE	±5 DEGREES	±5 DEGREES	±5 DEGREES
ROLL RATE	ZERO	ZERO	ZERO
NORMAL "G"	+0.8 TO +2.0	+0.9 TO +1.1	+0.8 TO +4.0

AUTHORIZED RELEASE MODES:
- RIPPLE SINGLES – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE PAIRS – RELEASE PULSES MUST BE SET TO COMPLETELY UNLOAD THE SELECTED STATIONS.
- RIPPLE SALVO – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY AND RELEASE PULSES MUST BE SET TO COMPLETELY UNLOAD SELECTED STATIONS.
- MINIMUM INTERVALOMETER SETTING IS 50 MILLISECONDS.

WARNING

FOR RIPPLE PAIRS AND/OR SELECT SALVO RELEASE THE TIME INTERVAL BETWEEN WEAPON RELEASES SHOULD BE GREATER THAN THE MINIMUM SPECIFIED TO REDUCE THE POSSIBILITY OF INTERBOMB COLLISION AFTER WEAPON SEPARATION FROM THE AIRCRAFT.

NOTE

FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATIONS.

EMERGENCY JETTISON LIMITS:
- WING SWEEP – NOT TO EXCEED 26 DEGREES
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000-E103B

Figure 5-17. (Sheet 2)

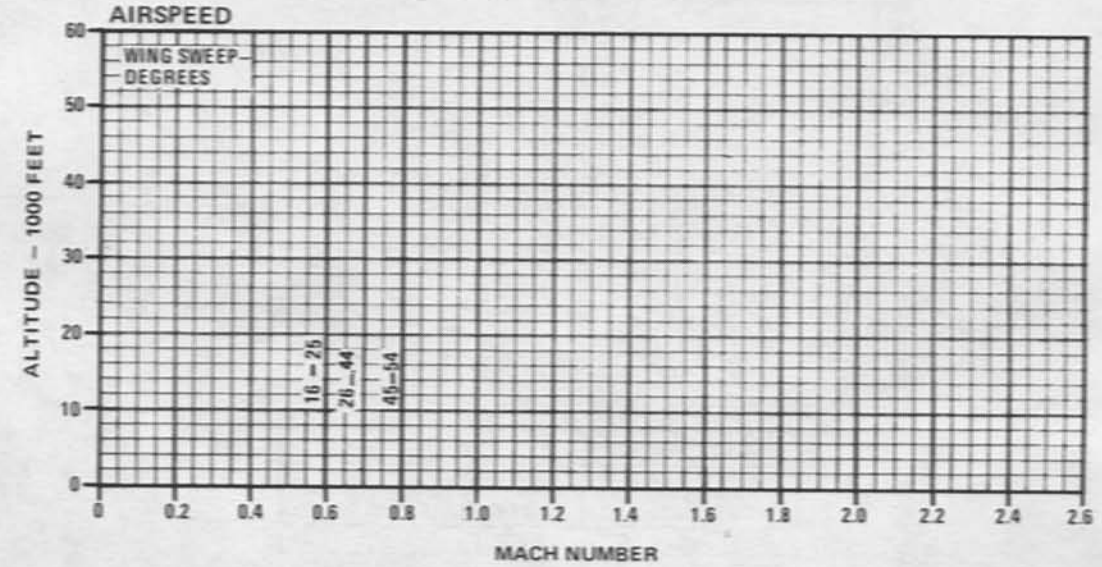
Carriage Limits - MK - 82 or MK - 82 Snakeye (LD or HD) or MK-36

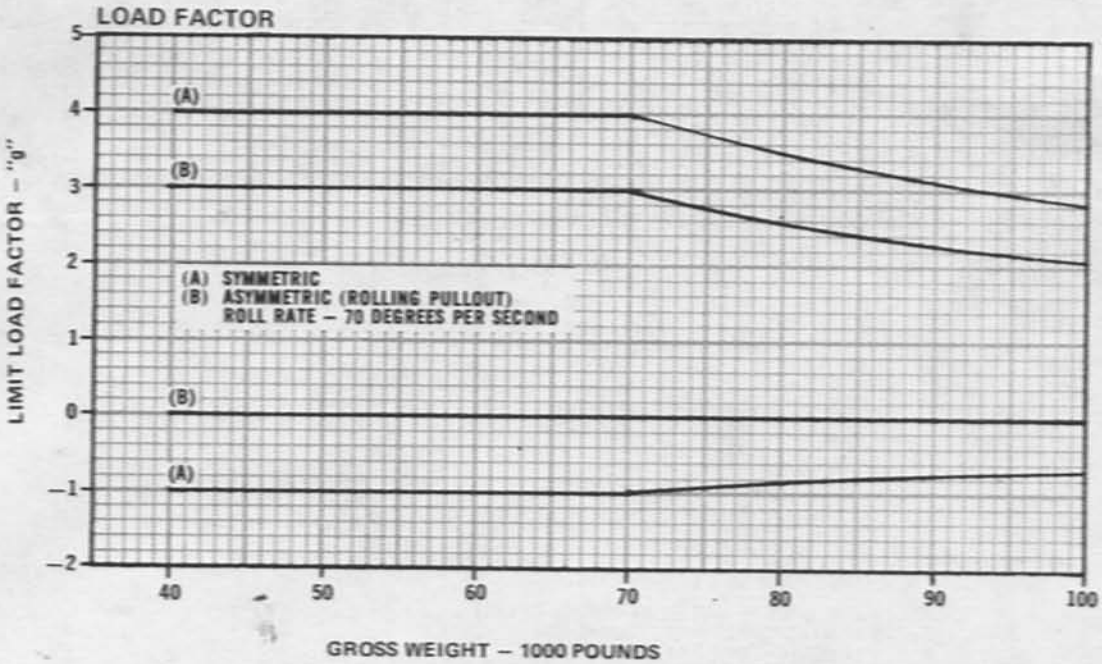
DATE: 30 MARCH 1973

CONFIGURATION:

4 EACH ON 3, 4, 5, AND 6 (SLANT) OR;
6 EACH ON 3 AND 6 AND 4 EACH ON 4 AND 5 (SLANT) OR;
4 EACH ON 4 AND 5 (SLANT), W PYLONS ON 3 AND 6.

★ NOTE: FOR THE 4 WEAPONS PER BRU LOADING THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND OUTBOARD SHOULDER POSITIONS.





A0000000-E105D

Figure 5-17. (Sheet 3)

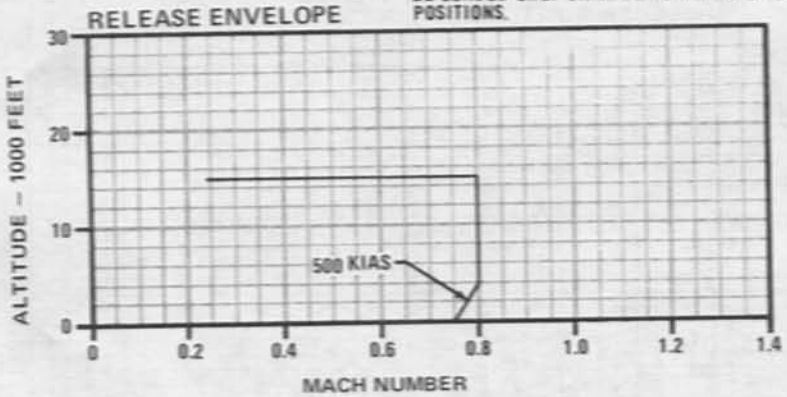
Release and Jettison Limits - MK - 82 Snakeye (HD) or MK-36

DATE: 8 SEPTEMBER 1972

CONFIGURATION
6 EACH ON 3 AND 6, W/O PYLONS ON 4 AND 5 OR;
6 EACH ON 3 AND 6 AND 4 EACH ON 4 AND 5 (SLANT), OR;
4 EACH ON 4 AND 5 (SLANT), W PYLONS ON 3 AND 6

NOTE: FOR THE 4 WEAPONS PER BRU LOADING THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND OUTBOARD SHOULDER POSITIONS.

RELEASE ENVELOPE



RELEASE LIMITATIONS:

PARAMETERS	LEVEL
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 45 DEGREES
ALTITUDE – FEET	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 15 DEGREES
CLIMB ANGLE	0 DEGREES TO 15 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +2.0

AUTHORIZED RELEASE MODES.
- RIPPLE SINGLES OR PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RELEASE OUTBOARD TO INBOARD ONLY.
- MINIMUM INTERVALOMETER SETTING IS 250 MILLISECONDS.

WARNING

6 WEAPONS ON STATIONS 3 AND 6, INTERBOMB COLLISION MAY BE A PROBLEM AFTER SEPARATION.

NOTE

- FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATIONS.
- FOR RIPPLE PAIRS RELEASE, THE TIME INTERVAL BETWEEN WEAPON RELEASES SHOULD BE GREATER THAN THE MINIMUM SPECIFIED TO REDUCE THE POSSIBILITY OF INTERBOMB COLLISION AFTER SEPARATION FROM THE AIRCRAFT.

EMERGENCY JETTISON LIMITS:
- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

AC000006-E106C

Figure 5-17. (Sheet 4)

Release and Jettison Limits - MK - 82 or MK-82 Snakeye (LD only)

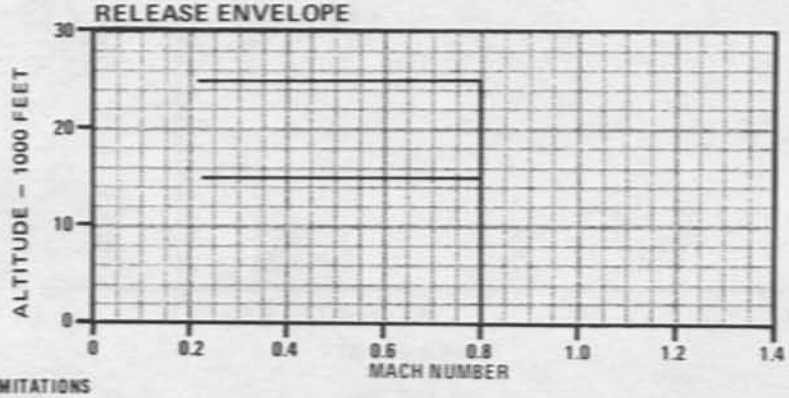
DATE: 30 MARCH 1973

CONFIGURATION:

★ 6 EACH ON 3, 4, 5 AND 6, OR;
6 EACH ON 4 AND 5, W-W/O PYLONS ON 3 AND 6, OR;
4 EACH ON 3, 4, 5 AND 6 (SLANT), OR;
6 EACH ON 3 AND 6, AND 4 EACH ON 4 AND 5 (SLANT), OR;
4 EACH ON 4 AND 5 (SLANT), W PYLONS ON 3 AND 6

NOTE: FOR THE 4 WEAPONS PER BRU LOADING, THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND OUTBOARD SHOULDER POSITIONS.

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	LEVEL	AT ALTITUDE	DIVE OR CLIMB
SPEED BRAKE	RETRACTED	RETRACTED	RETRACTED
WING SWEEP	26° TO 45°	26° TO 45°	26° TO 45°
ALTITUDE–FEET	0 TO 15,000	15,000 TO 25,000	0 TO 15,000
DIVE ANGLE	0 to 15°	NONE	0 TO 45°
CLIMB ANGLE	0 TO 15°	0 TO 15°	0° TO 30°
ROLL ANGLE	±5°	±5°	±5°
ROLL RATE	ZERO	ZERO	ZERO
NORMAL "G"	+0.8 TO +2.0	+0.9 TO +1.1	+0.8 TO + 4.0

AUTHORIZED RELEASE MODES.
- **RIPPLE SALVO** – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- **RIPPLE PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.**
- RELEASE OUTBOARD TO INBOARD ONLY.
- MINIMUM INTERVALOMETER SETTING IS 50 MILLISECONDS.

NOTE

FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATION.

- **RELEASE SINGLES OR PAIRS** ALSO AUTHORIZED WITH FOUR STORE CLUSTERS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.

WARNING

FOR RIPPLE PAIRS AND/OR RIPPLE SALVO RELEASE THE TIME INTERVAL BETWEEN WEAPON RELEASES SHOULD BE GREATER THAN THE MINIMUM SPECIFIED TO REDUCE THE POSSIBILITY OF INTERBOMB COLLISION AFTER WEAPON SEPARATION FROM THE AIRCRAFT.

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

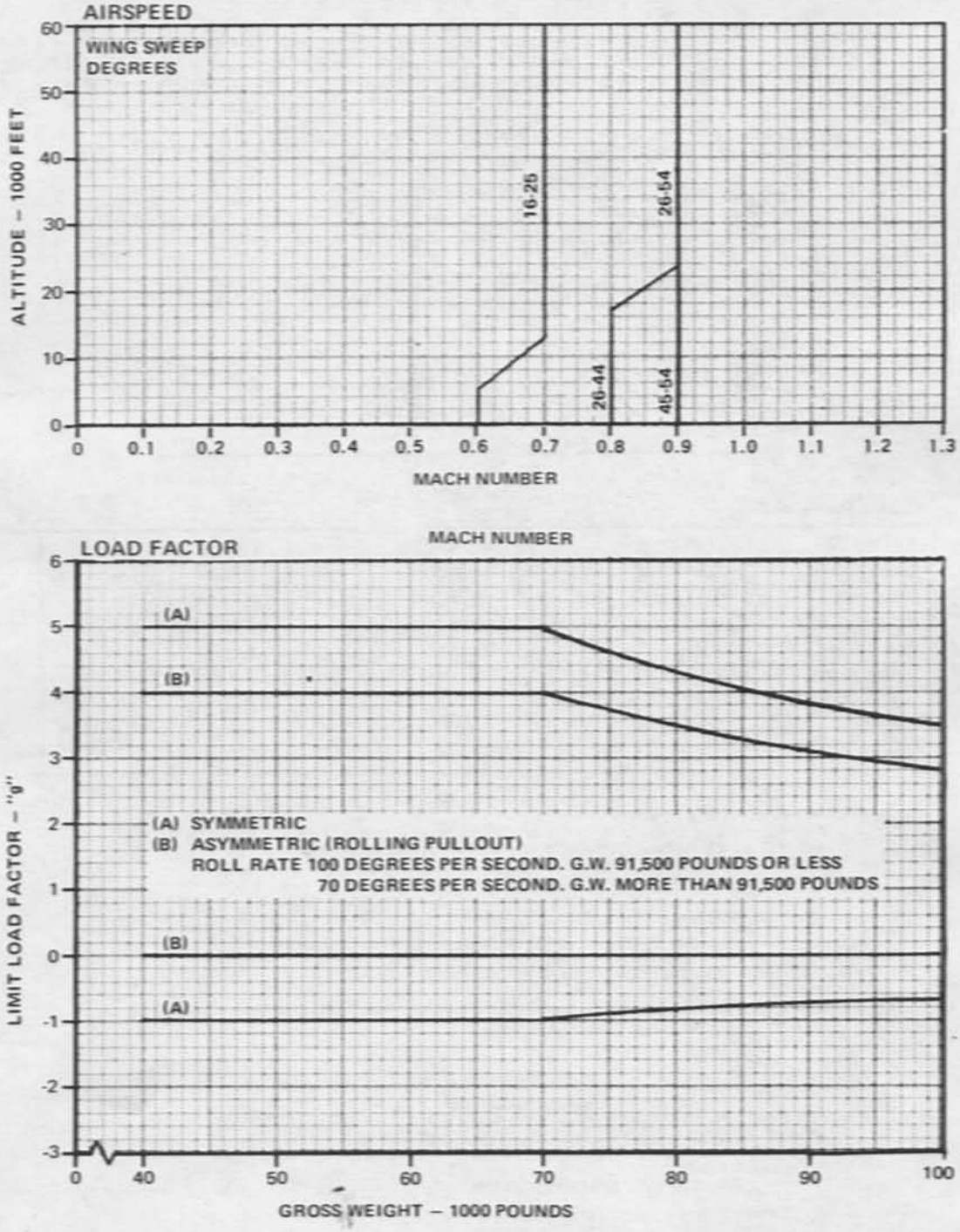
A0000000-E135A

Figure 5-17. (Sheet 5)

Carriage Limits - MK - 82 or MK-82 Snakeye (LD only)

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3, 4, 5 AND 6, OR;
6 EACH ON 4 AND 5, W-W/O PYLONS ON 3 AND 6



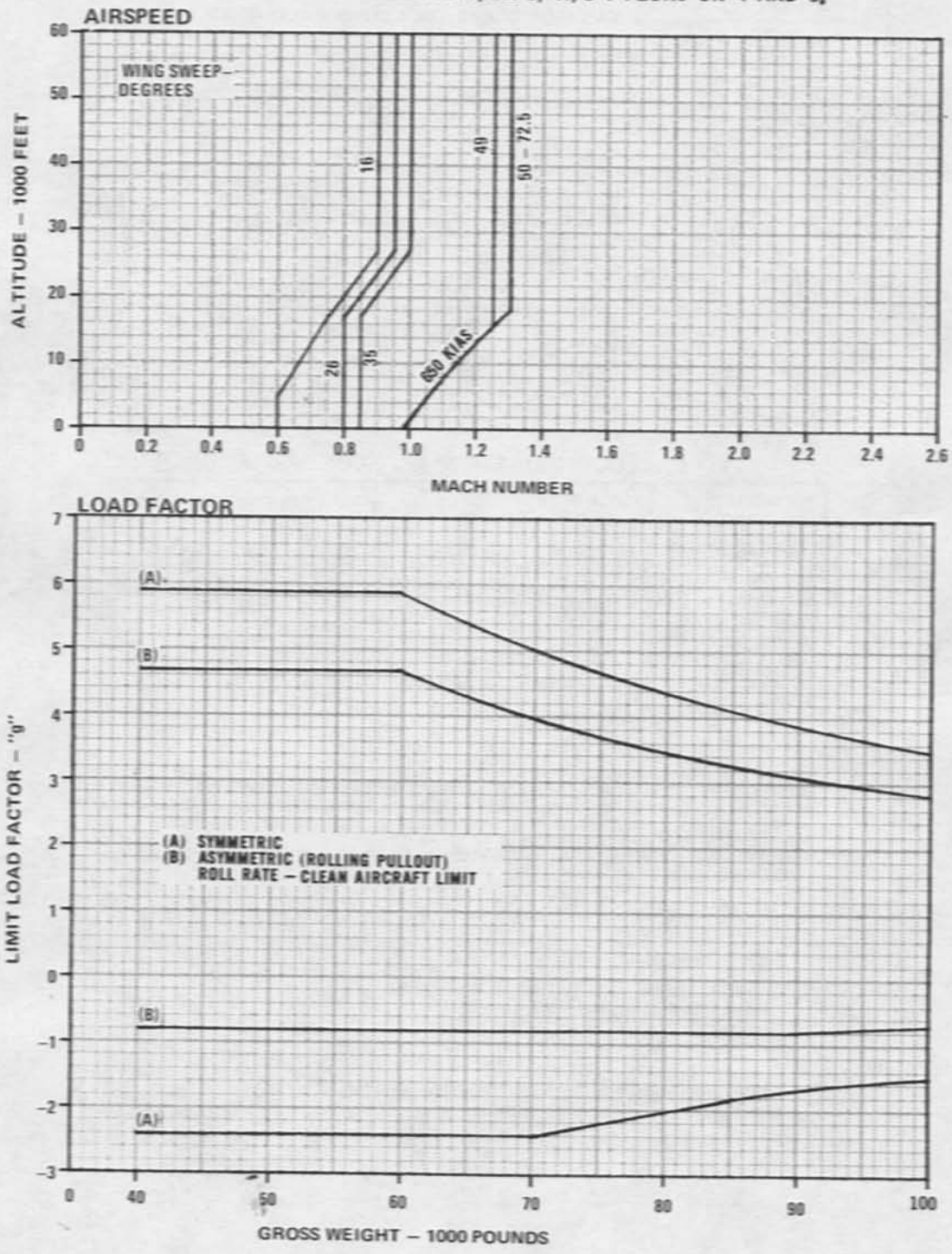
A0000000-E102C

Figure 5-17. (Sheet 6)

Carriage Limits - MK - 84

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 3, 4, 5 AND 6, OR;
1 EACH ON 4 AND 5 AND 3 OR 6, W-W/O PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6, OR;
1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5,



A0000000–E1078

Figure 5-18. (Sheet 1)

Release and Jettison Limits - MK - 84

DATE: 8 SEPTEMBER 1972

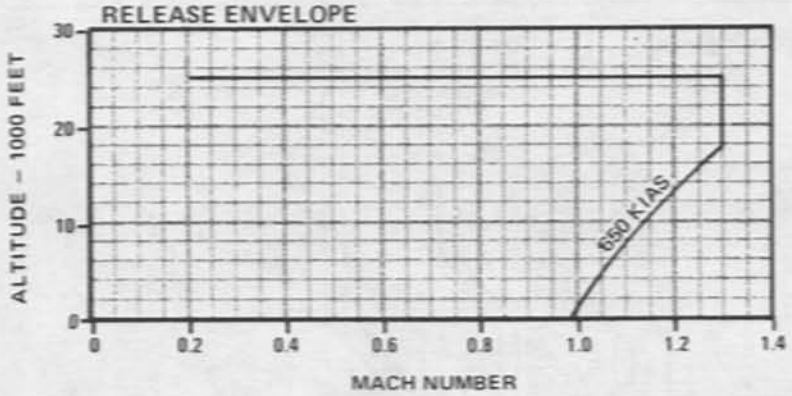
CONFIGURATION I

1 EACH ON 4 AND 5 AND 3 OR 6, W-W/O PYLON ON OTHER PIVOT STATION, OR:
1 EACH ON 4 AND/OR 5, W-W/O PYLONS ON 3 AND 6.

CONFIGURATION II

1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5.

RELEASE ENVELOPE



PARAMETERS	CONFIGURATION I LEVEL AND DIVE	CONFIGURATION II LEVEL AND DIVE
SPEED BRAKE	RETRACTED	RETRACTED
WING SWEEP	26 DEGREES TO 60 DEGREES	26 DEGREES TO 72.5 DEGREES
ALTITUDE-FEET	0 TO 25000	0 TO 25000
DIVE ANGLE	0 DEGREES TO 45 DEGREES	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 45 DEGREES	0 DEGREES TO 45 DEGREES
ROLL ANGLE	±5 DEGREES	±5 DEGREES
ROLL RATE	ZERO	ZERO
NORMAL "G"	+0.8 TO 4.0	+0.8 TO 4.0

AUTHORIZED RELEASE MODES.
- RELEASE SINGLES OR PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE SINGLES OR PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE SALVO – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RELEASE OUTBOARD TO INBOARD ONLY.
- PYLONS 4 AND 5 MUST BE RELEASED SIMULTANEOUSLY AT AIRSPEEDS ABOVE 1.0 MACH.
- RELEASE OF WEAPON IN THE PRESENCE OF ECM PODS LIMITED TO MACH 1.30

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES
REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

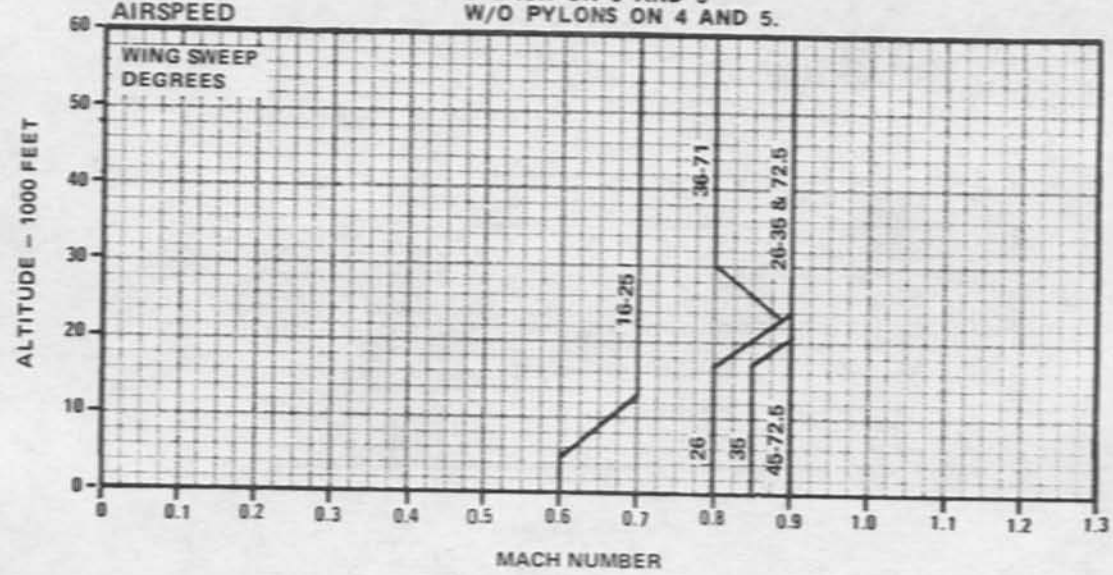
A0000000-E108B

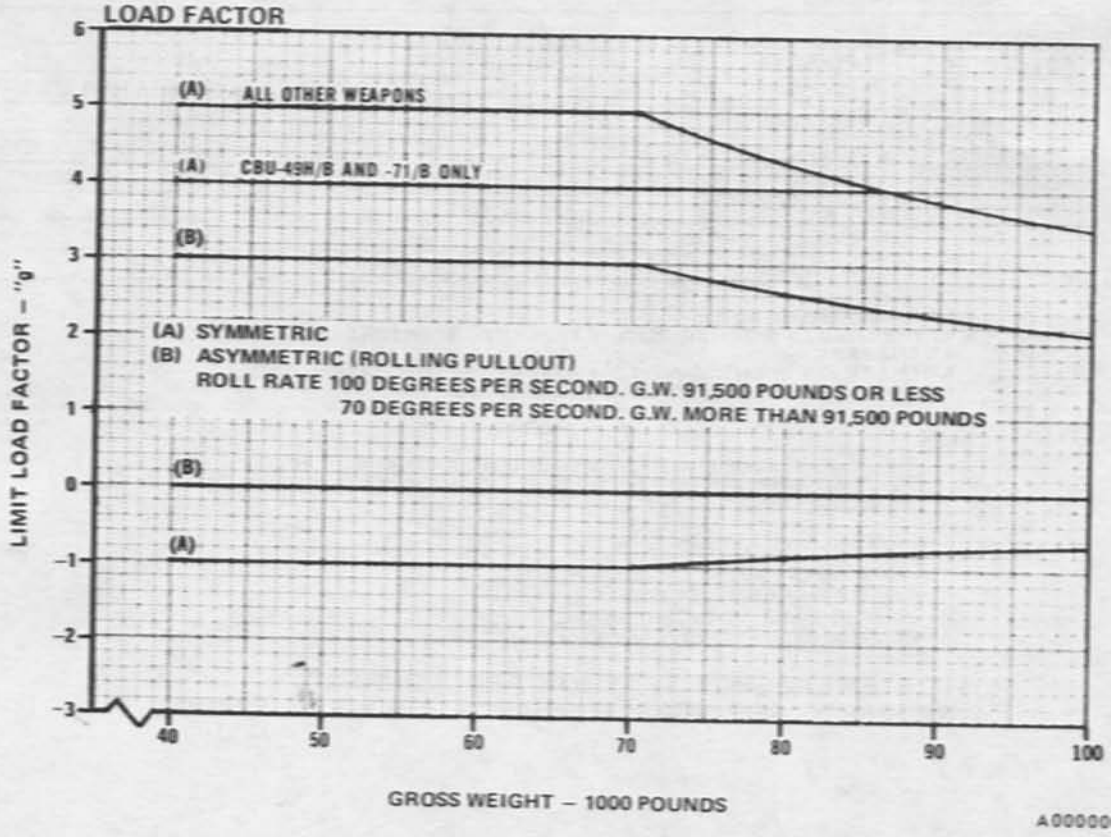
Figure 5-18. (Sheet 2)

Carriage Limits - M-117 w MAU-103A/B or CBU-24H/B,-29H/B, -49H/B, -52B/B, -58/B, -71/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3 AND 6
W/O PYLONS ON 4 AND 5.





A0000000-E0968

Figure 5-19. (Sheet 1)

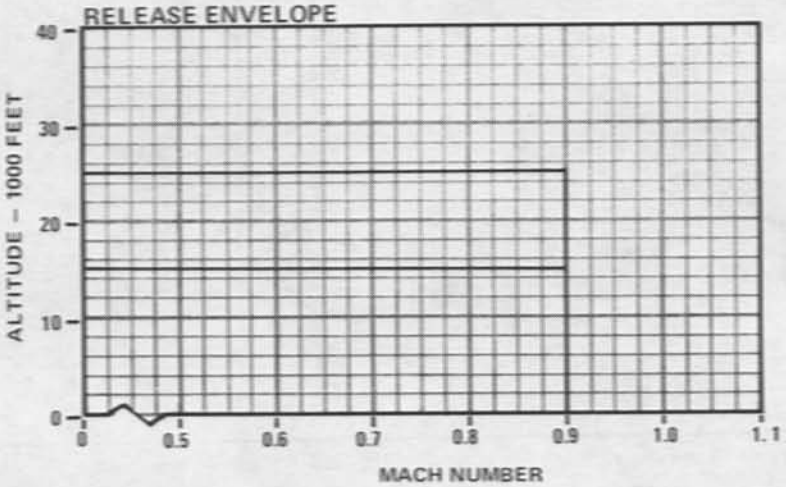
Release and Jettison Limits - M - 117 w MAU-103A/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

6I EACH ON 3 AND 6, W/O PYLONS ON 4 AND 5

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	LEVEL	AT ALTITUDE	DIVE OR CLIMB
SPEED BRAKE	RETRACTED	RETRACTED	RETRACTED
WING SWEEP	26 DEGREES TO 65 DEGREES	26 DEGREES TO 65 DEGREES	26 DEGREES TO 45 DEGREES
ALTITUDE–FEET	0 TO 15,000	15,000 TO 25,000	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 15 DEGREES	NONE	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 15 DEGREES	0 DEGREES 15 DEGREES	0 DEGREES TO 30 DEGREES
ROLL ANGLE	±5 DEGREES	±5 DEGREES	±5 DEGREES
ROLL RATE	ZERO	ZERO	ZERO
NORMAL "G"	+0.8 TO +2.0	+0.9 TO +1.1	+0.8 TO +4.0

AUTHORIZED RELEASE MODES:
- RIPPLE SINGLES – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE PAIRS
- RIPPLE SALVO – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- MINIMUM INTERVALOMETER SETTING IS 80 MILLISECONDS.

NOTE

FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATIONS.

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000–E0978

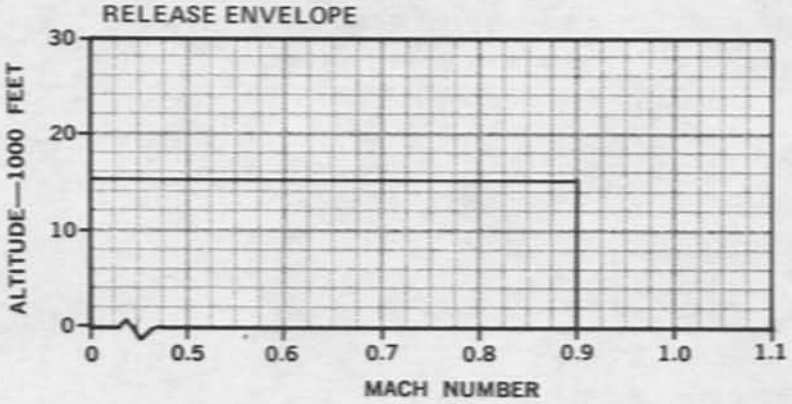
Figure 5-19. (Sheet 2)

Release and Jettison Limits - CBU-24H/B, -29H/B, -49H/B, -52B/B, -58/B, -71/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3 AND 6
W/O PYLONS ON 4 AND 5

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	LEVEL	DIVE OR CLIMB
SPEED BRAKE	RETRACTED	RETRACTED
WING SEEEP	26 DEGREES TO 54 DEGREES	26 DEGREES TO 45 DEGREES
ALTITUDE-FEET	0 TO 15,000	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 15 DEGREES	0 DEGREES TO 45 DEGREES
CLIMB ANGLE	0 DEGREES TO 15 DEGREES	0 DEGREES TO 30 DEGREES
ROLL ANGLE	±5 DEGREES	±5 DEGREES
ROLL RATE	ZERO	ZERO
NORMAL "G"	+0.8 TO +2.0	+0.8 TO +4.0

AUTHORIZED RELEASE MODES:
- RIPPLE PAIRS
- MINIMUM INTERVALOMETER SETTING IS 50 MILLISECONDS.

NOTE

- FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATIONS.
- FOR RIPPLE PAIRS RELEASE THE TIME INTERVAL BETWEEN WEAPON RELEASES SHOULD BE GREATER THAN THE MINIMUM SPECIFIED TO REDUCE THE POSSIBILITY OF INTERBOMB COLLISION AFTER SEPARATION FROM THE AIRCRAFT.

- RELEASE AUTHORIZED WITH FORWARD ECM POD

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES
- ALTITUDE – 10,000 FEET OR BELOW
- AIRSPEED – NOT TO EXCEED 250 KIAS
- FLAPS/SLATS – EXTENDED OR RETRACTED

FOR EMERGENCY JETTISON PROCEDURES, REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000-E136

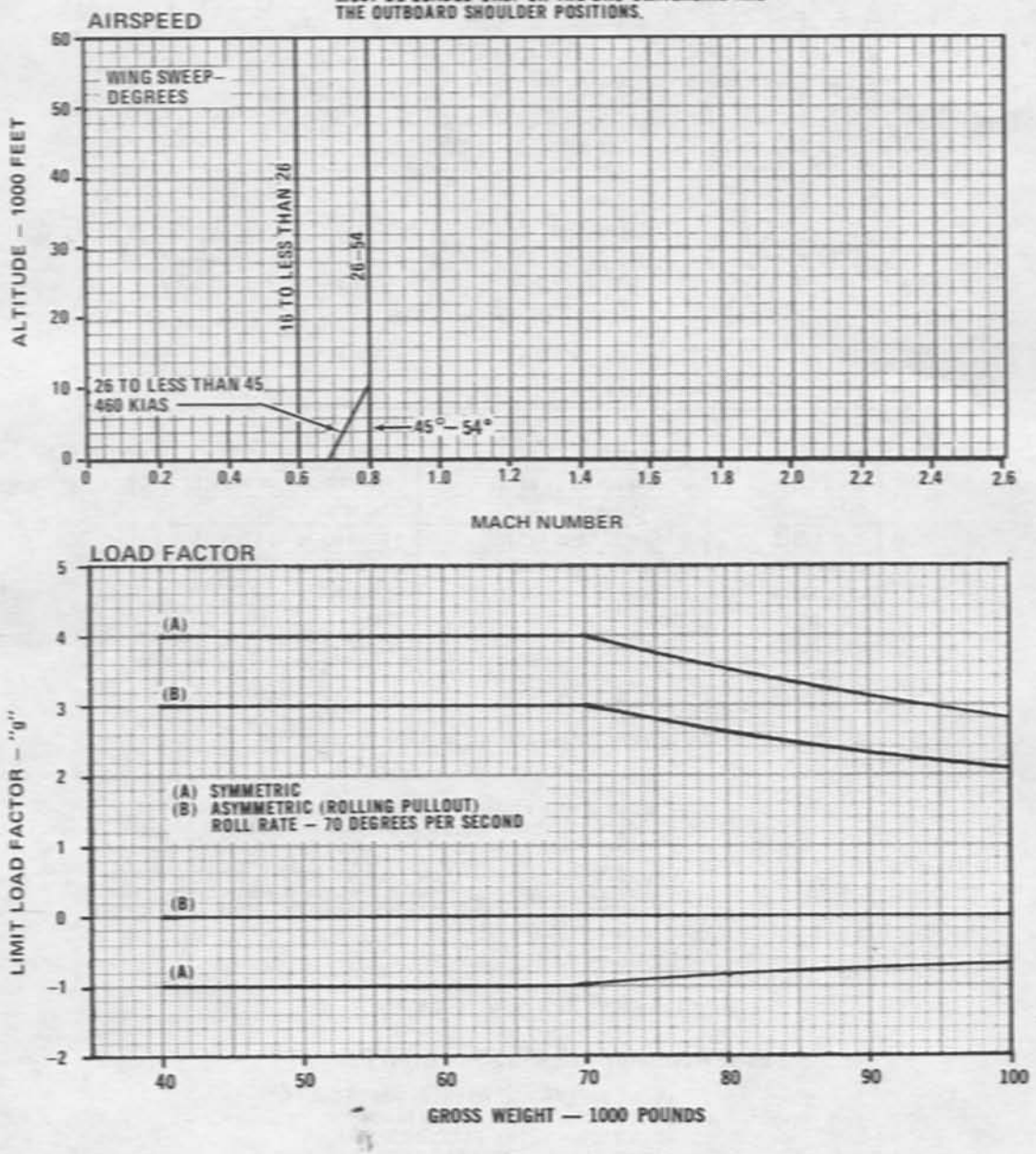
Figure 5-19. (Sheet 3)

Carriage Limits - M - 117w MAU-103A/B, M-117R, D or CBU-24H/B, -29H/B, -49H/B, -52B/B, -58/B or -71/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
4 EACH ON 3, 4, 5 AND 6 (SLANT), OR:
4 EACH ON 4 AND 5 (SLANT), WITH PYLONS ON 3 AND 6

NOTE: FOR THE 4 WEAPONS PER BRU LOADING, THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND THE OUTBOARD SHOULDER POSITIONS.



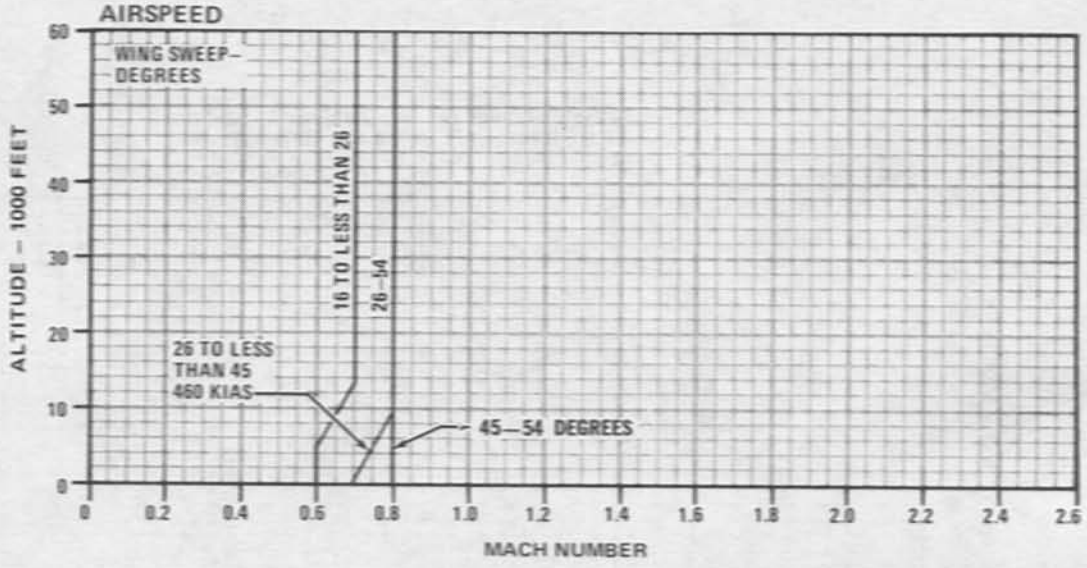
A0000000-E095C

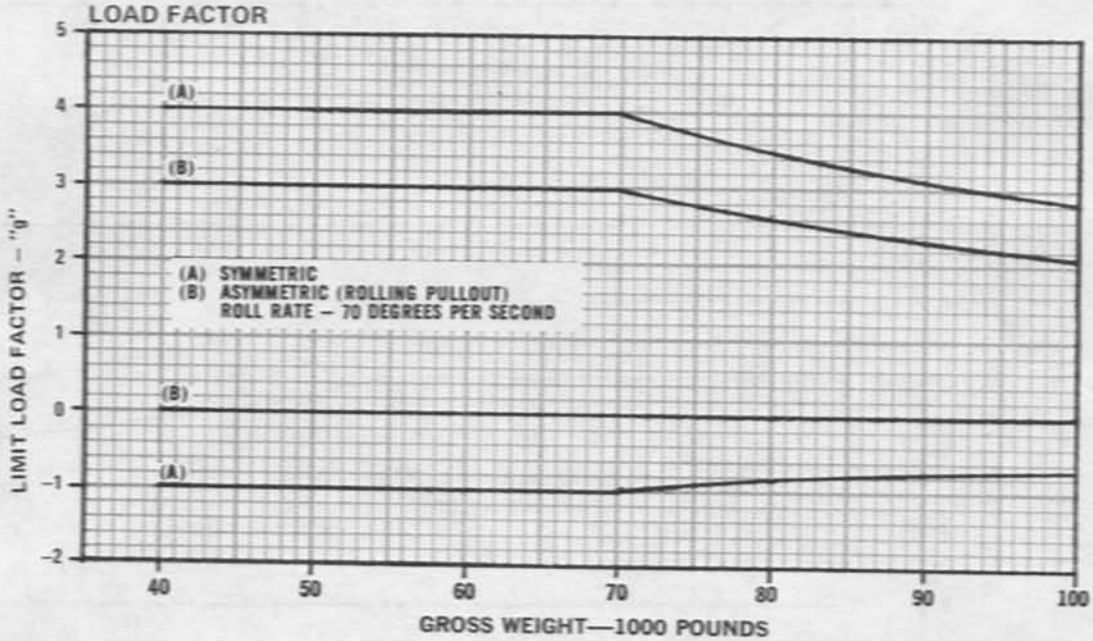
Figure 5-19. (Sheet 4)

Carriage Limits - M - 117w MAU-103A/B

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
6 EACH ON 3, 4, 5 AND 6, OR;
6 EACH ON 4 AND 5 WITH PYLONS ON 3 AND 6





A0000000 - E098C

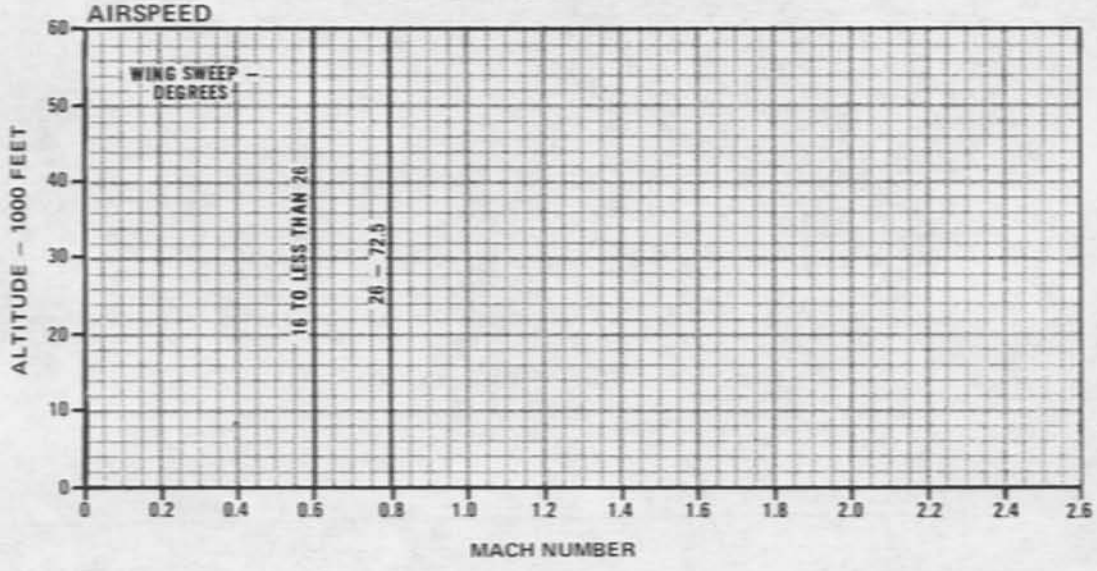
Figure 5-19. (Sheet 5)

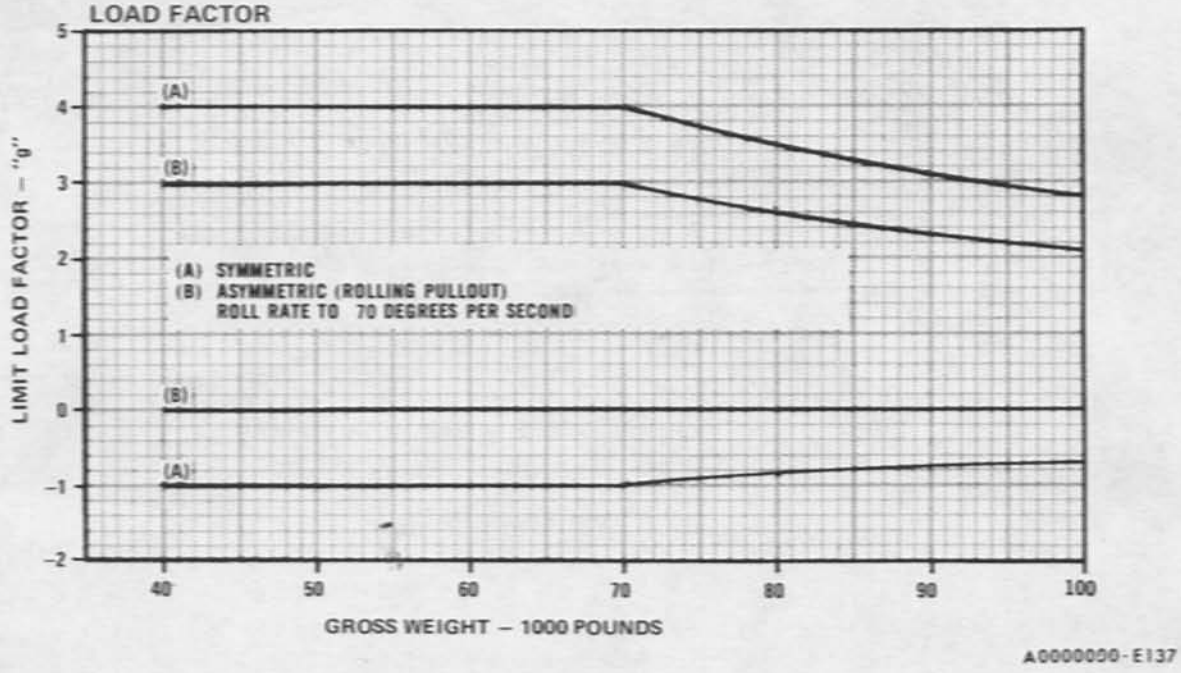
Carriage Limits - M-117R or M-117D

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
4 EACH ON 3 AND 6 (SLANT)
W/O PYLONS ON 4 AND 5.

NOTE: FOR THE 4 WEAPONS PER BRU LOADING, THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND THE OUTBOARD SHOULDER POSITIONS.





A0000000-E137

Figure 5-19. (Sheet 6)

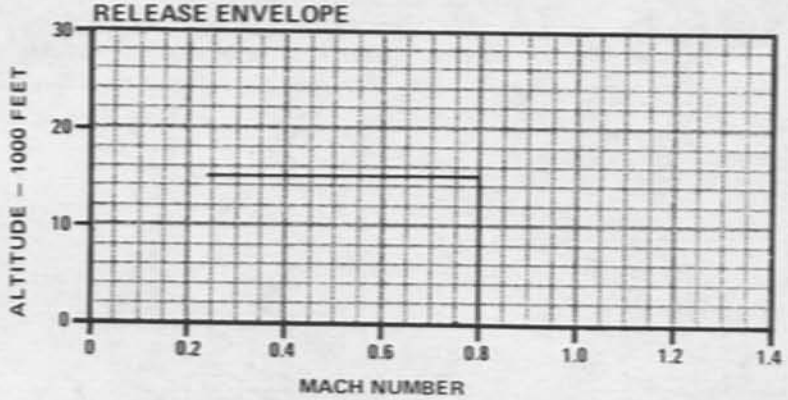
Release and Jettison Limits - M-117R or M-117D

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
4 EACH ON 3 AND 6 (SLANT), W/O PYLONS ON 4 AND 5.

NOTE: FOR THE 4 WEAPONS PER BRU LOADING THE WEAPONS MUST BE LOADED ONLY ON THE BRU CENTERLINE AND OUTBOARD SHOULDER POSITIONS.

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	LEVEL
SPEED BRAKE	RETRACTED
WING SWEEP	26 DEGREES TO 45 DEGREES
ALTITUDE – FEET	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 15 DEGREES
CLIMB ANGLE	0 DEGREES TO 15 DEGREES
ROLL ANGLE	±5 DEGREES
ROLL RATE	ZERO
NORMAL "G"	+0.8 TO +2.0

AUTHORIZED RELEASE MODES.
- RELEASE SINGLES OR PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE PAIRS
- RIPPLE SALVO – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- MINIMUM INTERVALOMETER SETTING IS 250 MILLISECONDS.

NOTE

FOR RIPPLE RELEASES, WEAPON RELEASE BUTTON MUST BE HELD DEPRESSED LONG ENOUGH TO RELEASE ALL STORES ON THE SELECTED STATIONS.

EMERGENCY JETTISON LIMITS:
- WING SWEEP NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

A0000000-E138

Figure 5-19. (Sheet 7)

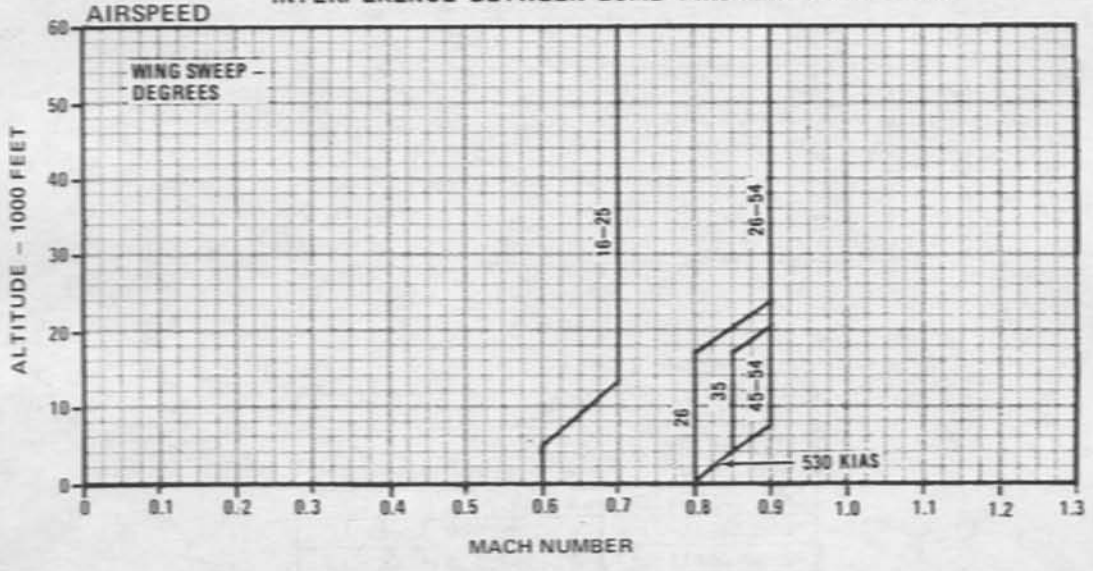
Carriage Limits - M - 118

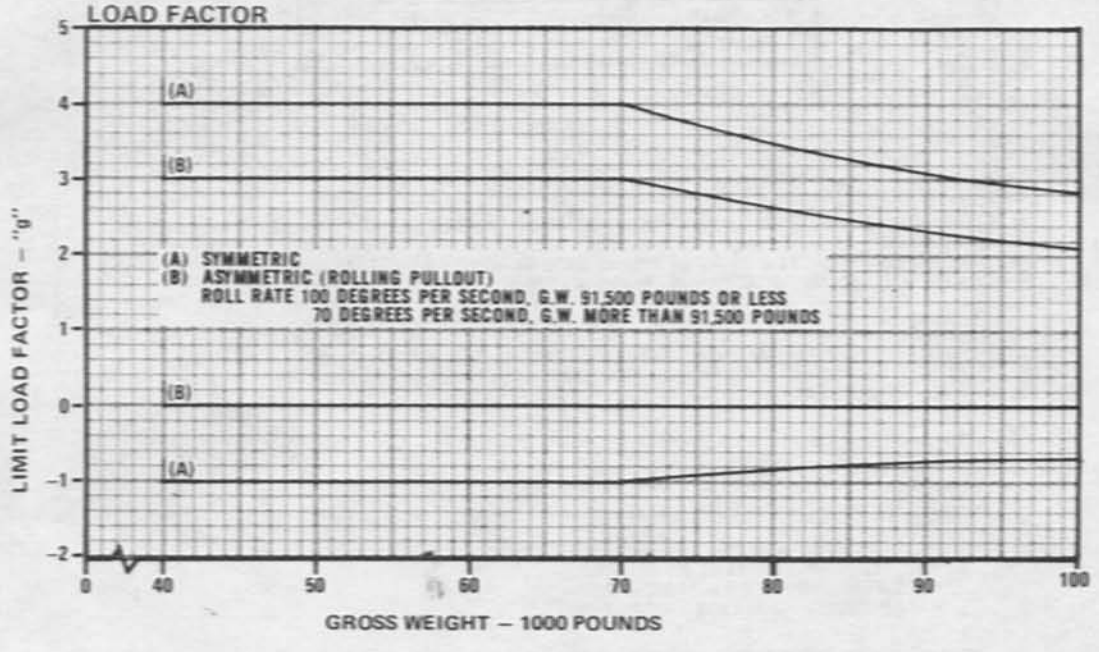
DATE: 30 MARCH 1973

CONFIGURATION:
1 EACH ON 3, 4, 5 AND 6, OR;
★ 1 EACH ON 4 AND 5 AND 3 OR 6, WITH PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, WITH PYLONS ON OTHER PIVOT STATIONS

CAUTION

FLAP EXTENSION BEYOND 25 DEGREES WILL RESULT IN INTERFERENCE BETWEEN BOMB FINS AND WING FLAPS.





A0000000—E099C

Figure 5-20. (Sheet 1)

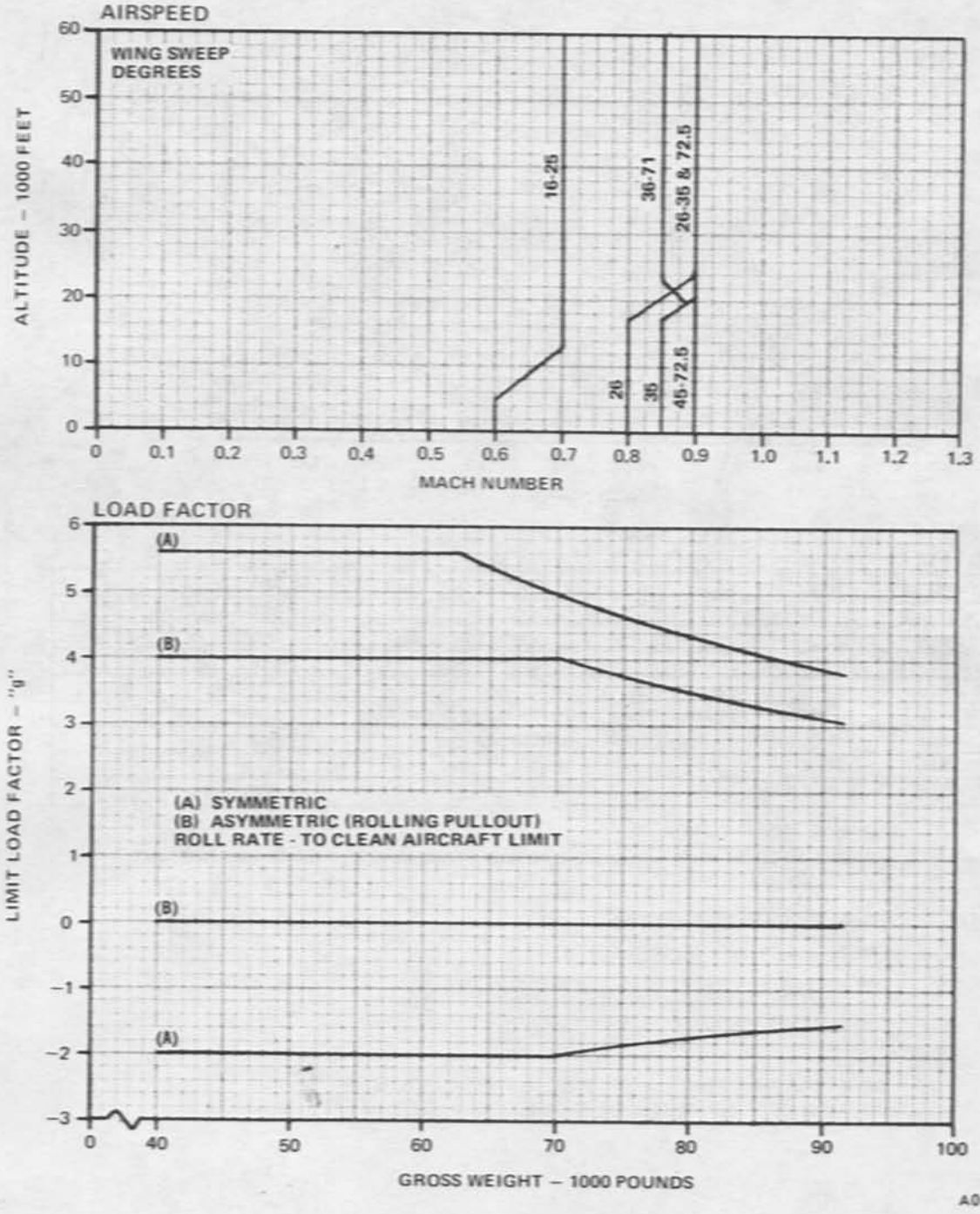
Carriage Limits - M - 118

★ CONFIGURATION:
1 EACH ON 3 AND/OR 6,
W/O PYLONS ON 4 AND 5.

DATE: 30 MARCH 1973

CAUTION

FLAP EXTENSION BEYOND 25 DEGREES WILL RESULT IN INTERFERENCE BETWEEN BOMB FINS AND WING FLAPS.



A0000000-E134A

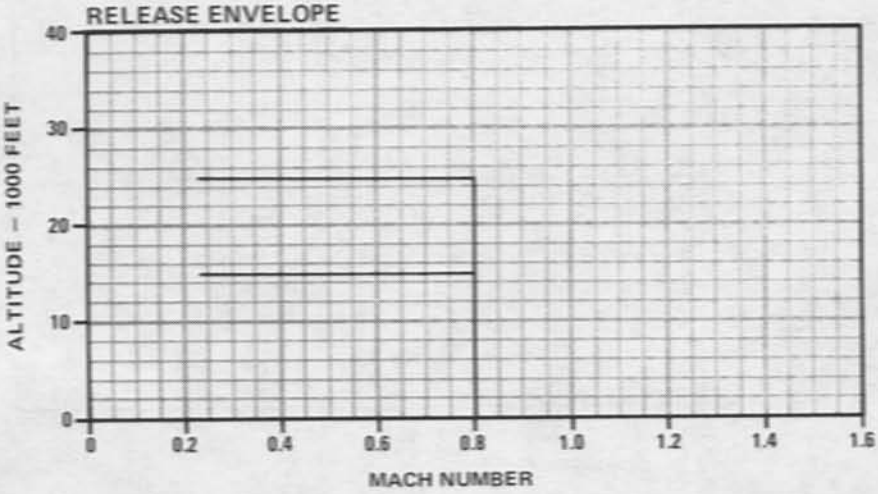
Figure 5-20. (Sheet 2)

Release and Jettison Limits - M - 118

DATE: 30 MARCH 1973

CONFIGURATION:

1 EACH ON 3, 4, 5 AND 6, OR;
★ 1 EACH ON 4 AND 5 AND 3 OR 6, WITH PYLON ON OTHER PIVOT STATION, OR;
1 EACH ON 4 AND/OR 5, WITH PYLONS ON OTHER PIVOT STATIONS, OR;
1 EACH ON 3 AND/OR 6, W/O PYLONS ON 4 AND 5.



RELEASE LIMITATIONS

PARAMETERS	LEVEL	AT ALTITUDE	DIVE
SPEED BRAKE	RETRACTED	RETRACTED	RETRACTED
WING SWEEP	26 DEGREES TO 55 DEGREES	26 DEGREES TO 55 DEGREES	26 DEGREES TO 45 DEGREES
ALTITUDE–FEET	0 TO 15,000	15,000 TO 25,000	0 TO 15,000
DIVE ANGLE	0 DEGREES TO 20 DEGREES	NONE	0 DEGREES TO 20 DEGREES
CLIMB ANGLE	0 DEGREES TO 20 DEGREES	0 DEGREES TO 20 DEGREES	NONE
ROLL ANGLE	±5 DEGREES	±5 DEGREES	±5 DEGREES
ROLL RATE	ZERO	ZERO	ZERO
NORMAL "G"	+0.8 TO +2.0	+0.9 TO +1.1	+0.8 TO +2.0

AUTHORIZES RELEASE MODES:

- RELEASE SINGLE OR PAIRS – SYMMETRIC PAIRS OF PYLONS MUST BE SELECTED SIMULTANEOUSLY.
- RIPPLE SINGLES OR PAIRS
- RIPPLE SALVO

RELEASE OUTBOARD TO INBOARD ONLY

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES
REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

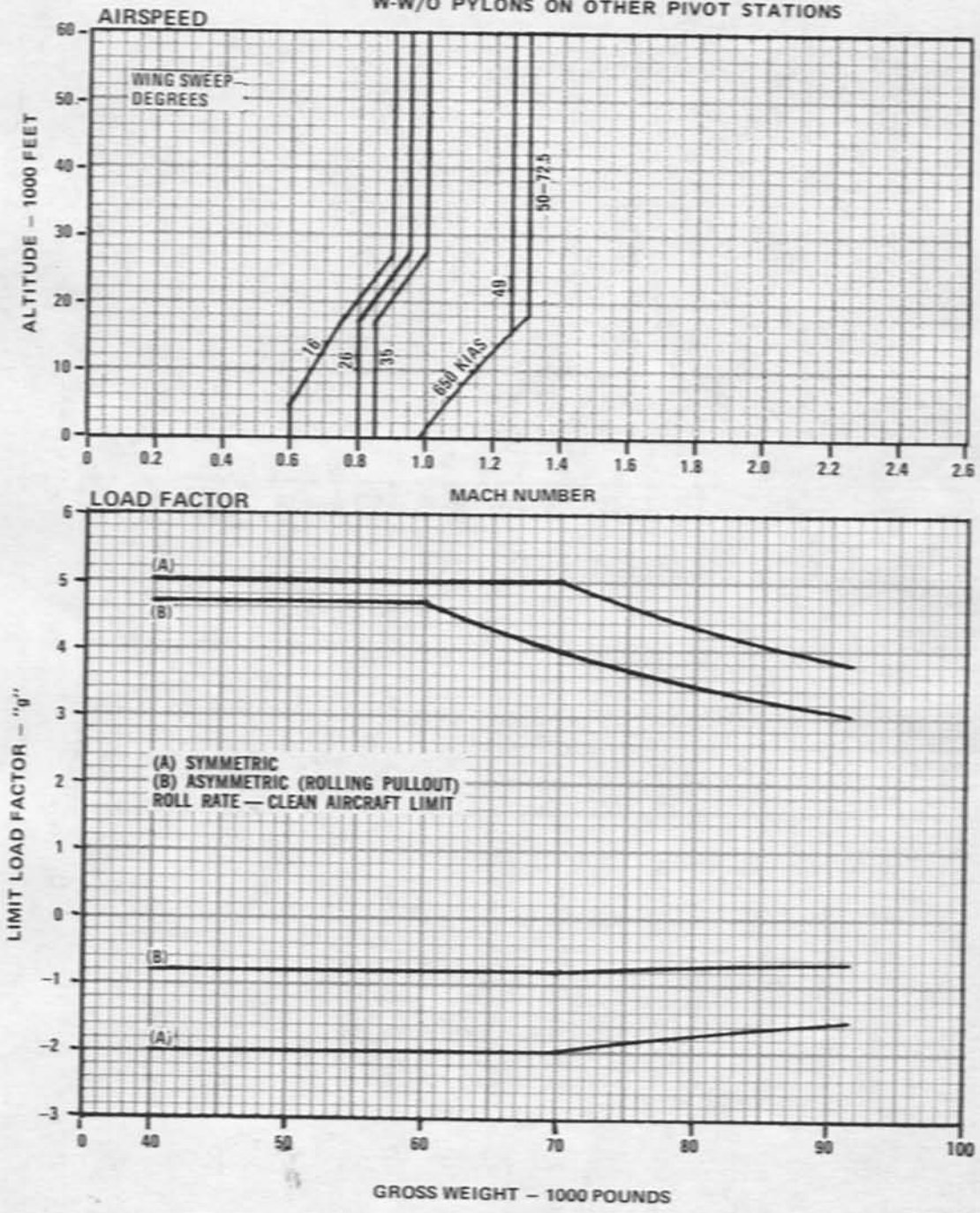
A0000000–E100C

Figure 5-20. (Sheet 3)

Carriage Limits - SUU - 20A/A or SUU - 20A/M or SUU-20B/A

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 4 AND/OR 5, OR,
1 EACH ON 3 AND/OR 6,
W-W/O PYLONS ON OTHER PIVOT STATIONS



A0000000—E091C

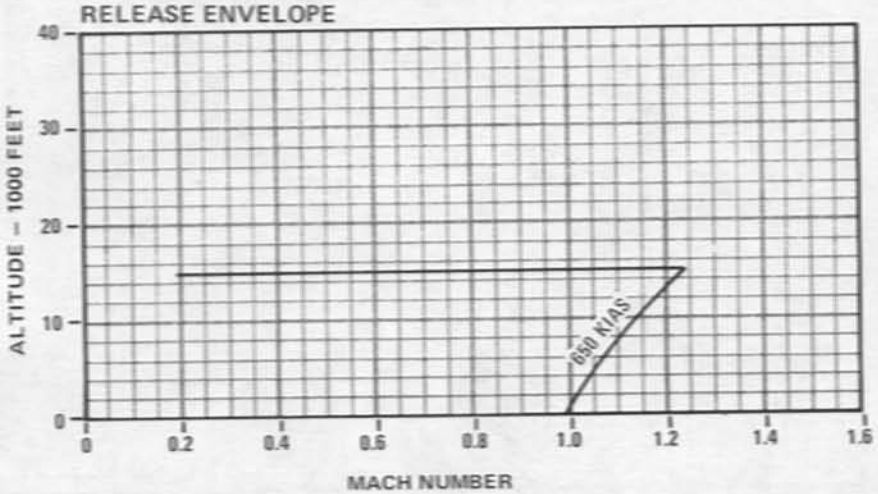
Figure 5-21. (Sheet 1)

Release and Jettison Limits - SUU-20A/A or SUU - 20A/M or SUU-20B/A

DATE: 8 SEPTEMBER 1972

CONFIGURATION:
1 EACH ON 4 AND/OR 5, OR,
1 EACH ON 3 AND/OR 6,
W-W/O PYLONS ON OTHER PIVOT STATIONS

RELEASE ENVELOPE



RELEASE LIMITATIONS

PARAMETERS	2.75" FFAR	MK-106	BDU-33A/B OR BDU-33B/B	BDU-33A/B OR BDU-33B/B (DIVE)	BDU-33A/B OR BDU-33B/B (DIVE)
SPEEDBRAKE	RETRACTED	RETRACTED	RETRACTED	RETRACTED	50°
SPEED OR MACH	0.80	550 KIAS TO 0.95	650 KIAS	550 KIAS TO 0.95	600 KIAS
WING SWEEP	26° TO 45°	26° TO 45°	26° TO 60°	26° TO 45°	26° TO 60°
DIVE ANGLE	0° TO 45°	0° TO 20°	0° TO 20°	0° TO 45°	0° TO 30°
CLIMB ANGLE	0° TO 15° *	0° TO 45°	0° TO 45°	NONE	NONE
ROLL ANGLE	±5°	±5°	±5°	±5°	±5°
ROLL RATE	ZERO	ZERO	ZERO	ZERO	ZERO
NORMAL "G"	+0.7 TO +1.1	+0.7 TO +4.0	+0.7 TO +4.0	+0.7 TO +1.7	+0.7 TO +1.7

* ROCKETS NOT NORMALLY FIRED IN A CLIMB

WARNING

THE BDU-33 CAN BE RELEASED SAFELY IN A DIVE WITH THE SPEEDBRAKE EXTENDED WITHIN THE INDICATED RELEASE ENVELOPE. HOWEVER, A HIGH PROBABILITY OF DAMAGE TO THE EJECTOR DOOR LINKAGE SPRING ASSEMBLY ON THE AIRCRAFT EXISTS DUE TO TURBULENT AIRFLOW WHEN THE SPEEDBRAKE IS EXTENDED. FLIGHT WITH THE SPEEDBRAKE EXTENDED SHOULD NOT EXCEED 1 "G" AND 600 KIAS OR MACH 1.3 (WHICHEVER IS THE MORE RESTRICTIVE) FOR ALL FLIGHT MODES.

EMERGENCY JETTISON LIMITS:

- WING SWEEP – NOT TO EXCEED 26 DEGREES.
- ALTITUDE – 10,000 FEET OR BELOW.
- AIRSPEED – NOT TO EXCEED 250 KIAS.
- FLAPS/SLATS – EXTENDED OR RETRACTED.

FOR EMERGENCY JETTISON PROCEDURES REFER TO APPLICABLE WEAPON DELIVERY MANUAL.

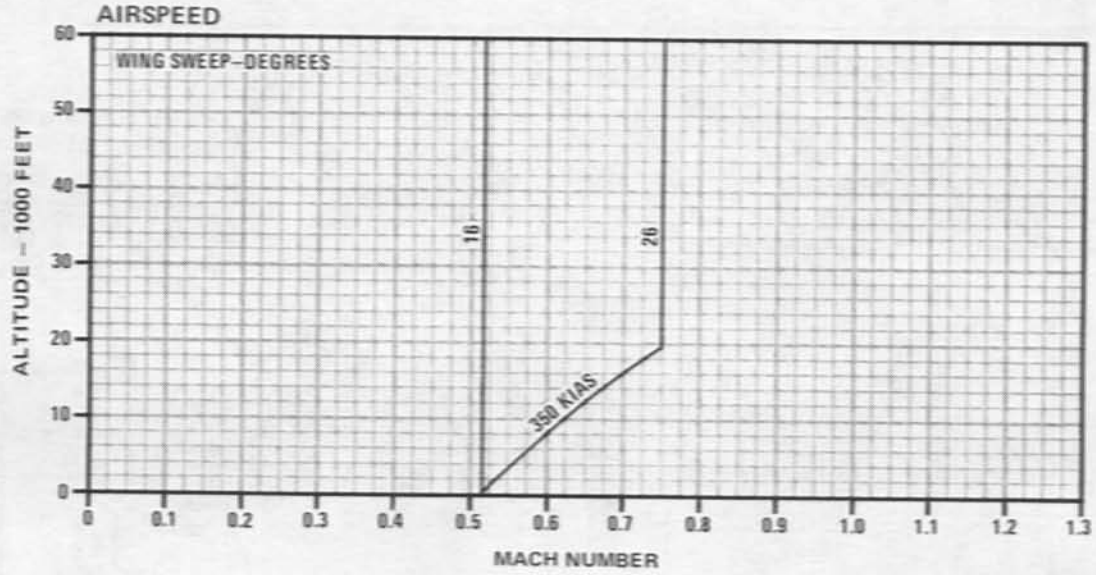
A0000000-E092 C

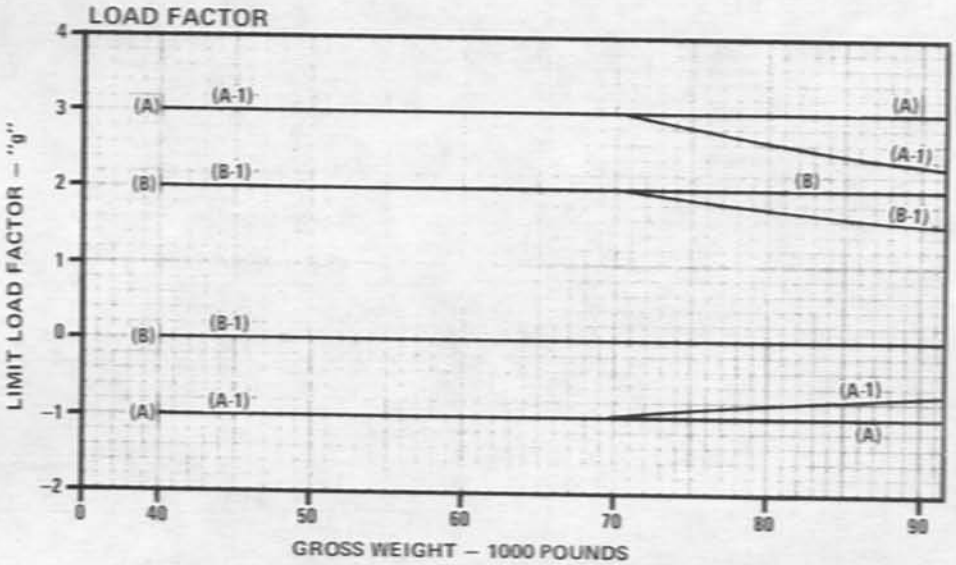
Figure 5-21. (Sheet 2)

Carriage Limits - 600 Gallon Tanks

DATE: 24 APRIL 1970

CONFIGURATION:
1 EACH ON 2 AND 7





(A) SYMMETRIC-WING SWEEP 16 DEGREES, FLAPS AND SLATS EXTENDED OR RETRACTED; WING SWEEP 26 DEGREES FLAPS AND SLATS RETRACTED.

(A-1) SYMMETRIC – WING SWEEP 26 DEGREES, FLAPS AND SLATS EXTENDED.

(B) ASYMMETRIC – WING SWEEP 16 DEGREES, FLAPS AND SLATS EXTENDED OR RETRACTED; WING SWEEP 26 DEGREES FLAPS AND SLATS RETRACTED; ROLL RATE TO 60° /SECOND.

(B-1) ASYMMETRIC – WING SWEEP 26 DEGREES, FLAPS AND SLATS EXTENDED; ROLL RATE TO 60°/SECOND.

A0000000—E118

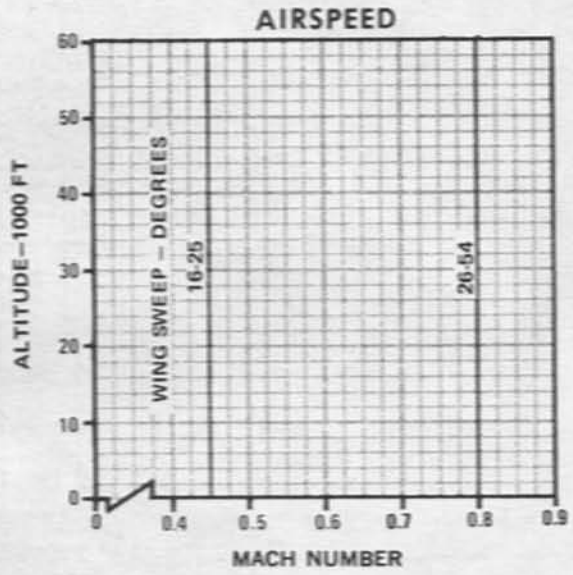
Figure 5-22. (Sheet 1)

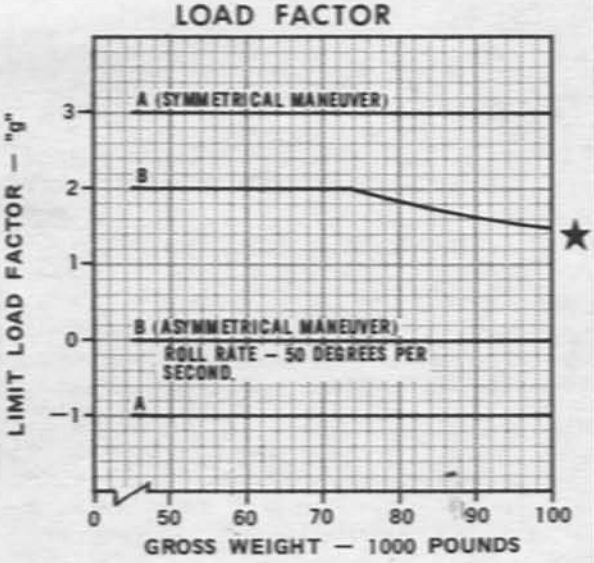
Carriage Limits - Tanks

(With or Without Nuclear Weapons or BDU's)

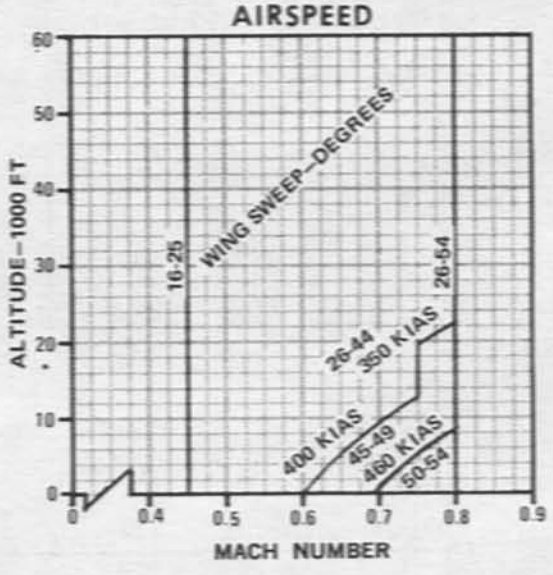
DATE: 30 MARCH 1973

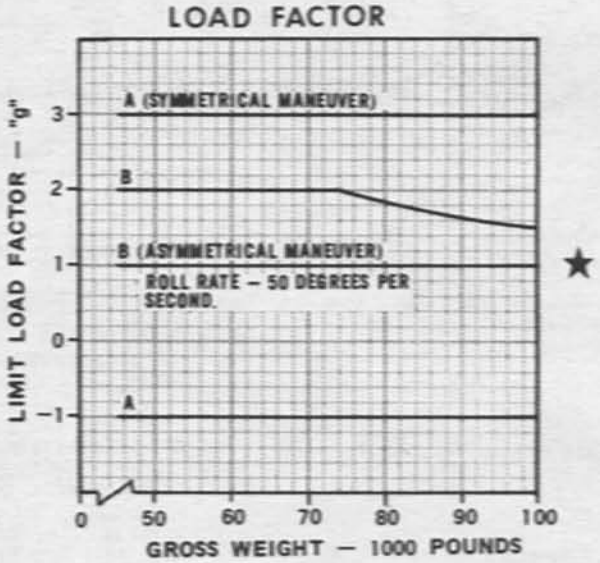
CONFIGURATION:
(2) TANKS–1 EACH ON 3 AND 6.
W/O PYLONS ON 4 AND 5





CONFIGURATION:
(2) TANKS–1 EACH ON 3 AND 6 WITH
(1) B43, 57 OR 61–1 EACH ON 4 OR 5
WITHOUT PYLON ON THE OTHER INBD, STATION





A0000000-E128 C

Figure 5-22. (Sheet 2)

Carriage Limits - Tanks

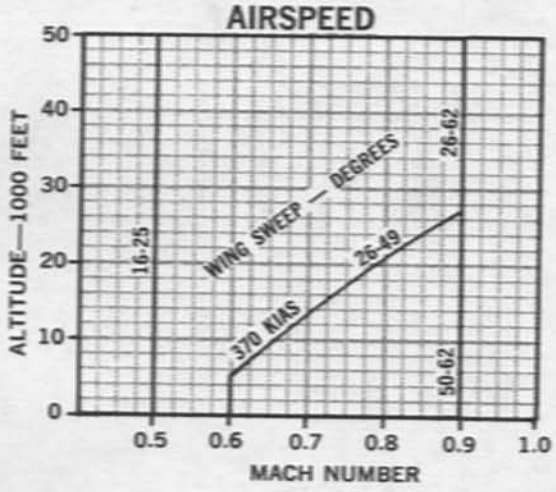
(With or Without Nuclear Weapons or BDU's)

DATE: 31 DECEMBER 1971

NOTE: BUFFET WILL OCCUR ABOVE 0.75 MACH WITH 26 DEGREE WING SWEEP.

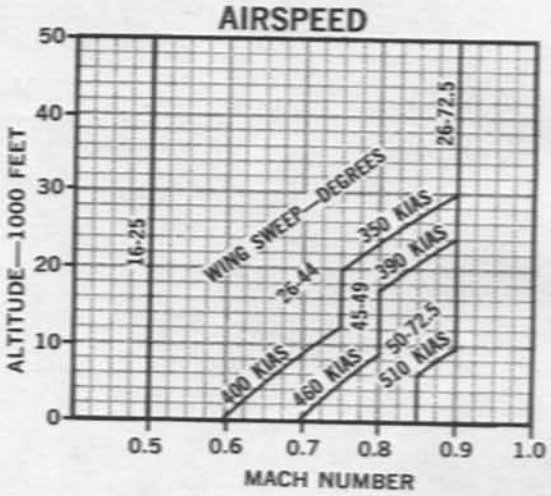
CONFIGURATION:

(4) TANKS—1 EACH ON 3, 4, 5 AND 6; OR
(2) TANKS—1 EACH ON 4 AND 5 WITH PYLONS ON 3 AND 6.

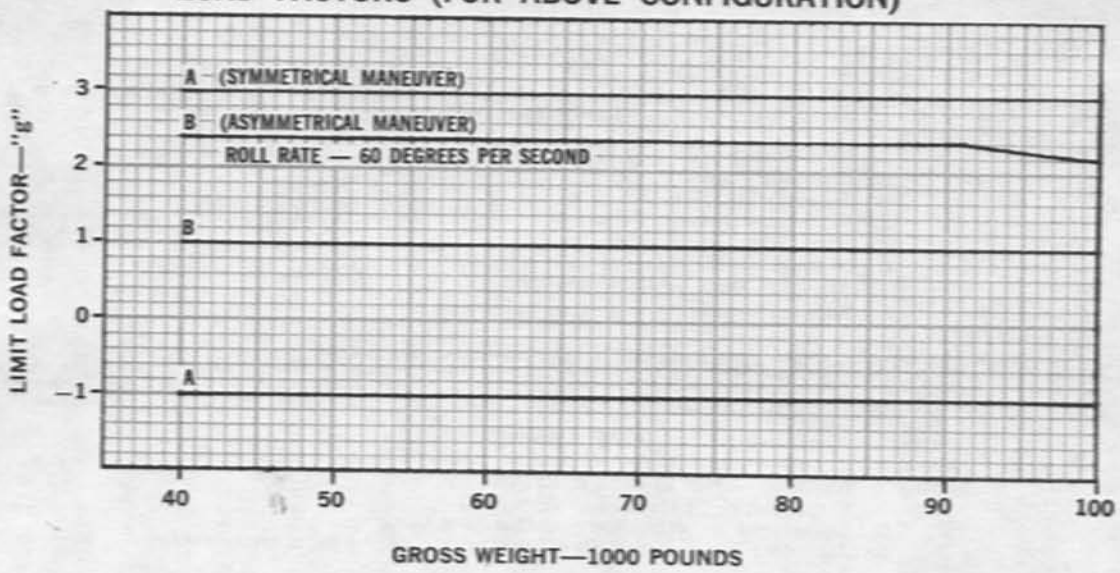


CONFIGURATION:

(2) TANKS—1 EACH ON 3 AND 6 WITH
(2) B 43'S, 57'S OR 61'S—1 EACH ON 4 AND 5.



LOAD FACTORS (FOR ABOVE CONFIGURATION)



A0000000-E129A

Figure 5-22. (Sheet 3)

Release Limits - Tanks

(With or Without Nuclear Weapons or BDU's)

DATE: 8 SEPTEMBER 1972

CONFIGURATION:

LIMIT A

(2) TANKS — 1 EACH ON 2 AND 7.

LIMIT B

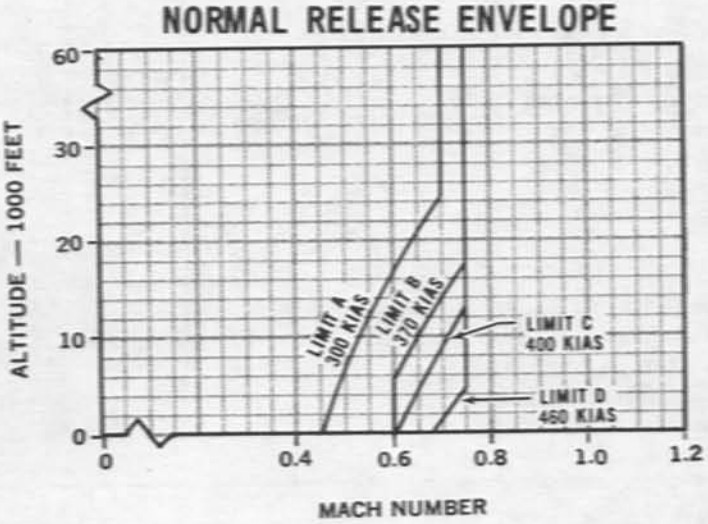
(4) TANKS — 1 EACH ON 3, 4, 5, AND 6; OR
(2) TANKS — 1 EACH ON 4 AND 5 WITH PYLONS ON 3 AND 6.

LIMIT C

(2) TANKS — 1 EACH ON 3 AND 6 WITH
(2) B43's, 57's OR 61's — 1 EACH ON 4 AND 5; OR
(2) TANKS — 1 EACH ON 3 AND 6 WITH
(1) B43, 57 OR 61 — 1 EACH ON 4 OR 5, W/O PYLON ON THE OTHER INBOARD STATION.

LIMIT D

(2) TANKS — 1 EACH ON 3 AND 6, W/O PYLONS ON 4 AND 5.



NORMAL RELEASE CONDITIONS:

Tanks must be released in symmetrical pairs, outboard to inboard with no more than 50 pounds residual fuel remaining. Conditions for normal release are as follows:

1. Wing Sweep—26 degrees.
2. Gear and flaps-up.
3. Acceleration — + 0.8 to + 1.2 "g".
4. Pitch angle — ± 10 degrees.
5. Bank angle — ± 5 degrees.
6. Roll rate — ± 2 degrees per second.
7. Angle-of-attack —10 degrees max.
8. Release mode: Release pairs only.

EMERGENCY JETTISON LIMITS:

Tanks with 1800 pounds or more fuel remaining may be jettisoned under the following conditions:

1. Wing sweep — 16 to 26 degrees.
2. Altitude — 10,000 feet or less.
3. Airspeed — Not to exceed 300 KIAS.
4. Flaps/slats — Extended or retracted.
5. Tanks may be jettisoned simultaneously.
6. Jettison Fixed pylons 0.50 second after tanks, if flaps are retracted.

FOR EMERGENCY JETTISON PROCEDURES REFER TO SECTION III.

A0000000-E130B

Figure 5-22. (Sheet 4)

M61 - A1 Weapon Bay Gun - Limitations

DATE: 6 OCTOBER 1972

CARRIAGE ENVELOPE

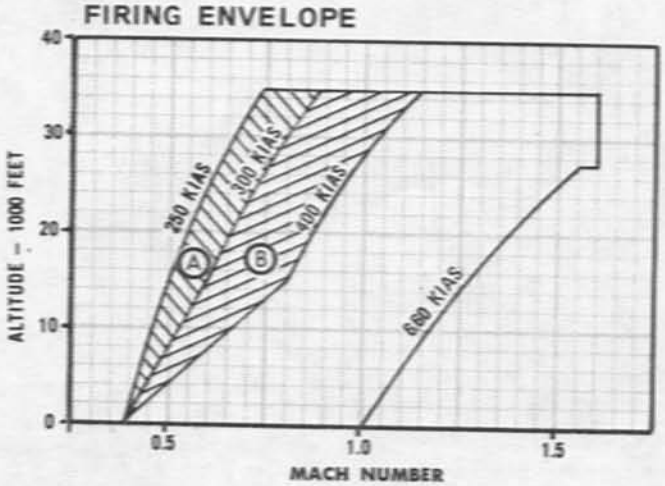
MANEUVER LOAD FACTOR — NO ADDITIONAL AIRCRAFT RESTRICTIONS.
ROLL RATE — NO ADDITIONAL AIRCRAFT RESTRICTIONS.
AIRSPEED — WITH AMMUNITION:

- DO NOT OPEN WEAPON BAY DOOR FOR MORE THAN ONE MIN. AT OR ABOVE TOTAL TEMP OF 75°C.

— WITHOUT AMMUNITION:

- NO ADDITIONAL AIRCRAFT RESTRICTIONS.

FIRING ENVELOPE



WARNING

AREA (A): Area of probable compressor stall during air-to-air firing at all flight conditions.
AREA (B): Area of probable compressor stall during air-to-air firing when operating in afterburner power while in maneuvering flight.

OBSERVE THE FOLLOWING FIRING RESTRICTIONS:

- Do not fire with the weapon bay door open.
- Do not fire gun when negative "g" exceeds minus one "g".
- Do not fire gun above +5.0 "g".
- The following cooling periods must be observed when firing a full drum of ammunition.
  - 225 round bursts — 1 minute between bursts.
  - 400 round bursts — 2.5 minutes between bursts.

A0000000-E120-D

Figure 5-23.

This is the last page of Section V.

SECTION VI

FLIGHT CHARACTERISTICS

Note

The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and, therefore, is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual since it is read directly from the instrument.

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INTRODUCTION.

The flight characteristics information presented in this section is based on quantitative and qualitative data obtained to date. 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 brakeout forces associated with the flight control system enhance ease of handling and maneuverability. 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.

For a detailed description of the flight control system refer to "Flight Control System," Section I.

DEFINITION OF LONGITUDINAL REFERENCE ANGLES.

An illustration describing flight path angle, pitch angle, angle of incidence, angle-of-attack and relative wind is presented in figure 6-1.

ANGLE-OF-ATTACK.

The angle-of-attack indication system provides an indication of the angular position of the wing chord in relation to the aircraft flight path. Angle-of-attack is of primary importance since, for a given aircraft weight and airspeed, sufficient lift can be generated to maintain one "g" flight only at a particular angle-of-attack. That is, lift is a function of airspeed and angle-of-attack. Thus, at one "g" flight if airspeed is held constant, angle-of-attack will remain constant. If airspeed decreases, angle-of-attack must increase if one "g" flight is to be maintained. Conversely, if airspeed increases, angle-of-attack must decrease to maintain one "g" flight. This direct relationship of angle-of-attack and airspeed with lift allows angle-of-attack to be used in place of airspeed. Angle-of-attack can be held constant and calibrated airspeed will remain relatively constant varying in proportion to gross weight but remaining essentially independent of altitude. Further, rate of descent or climb can be controlled by power changes and airspeed will remain constant as long as angle-of-attack remains constant. During normal landings, the recommended approach is 10 de-

Longitudinal Reference Angles

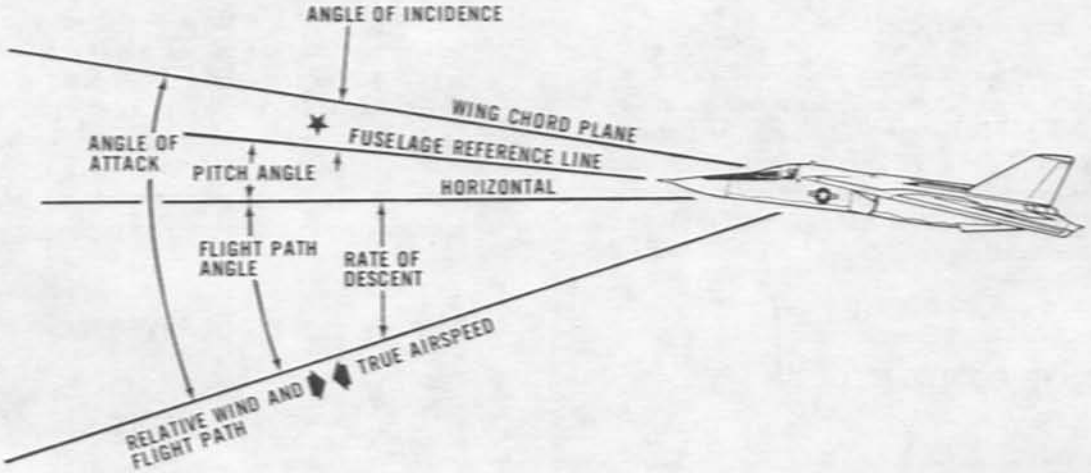

Figure 6-1.

grees angle-of-attack regardless of gross weight. The angle-of-attack indexer is programmed so that the on-speed symbol is lighted in the range of 10 degrees (±1.0°).

LEVEL FLIGHT CHARACTERISTICS.

Refer to Section II for discussion of takeoff and landing characteristics.

SUBSONIC FLIGHT.

Operation of the aircraft at subsonic speeds up to mach 0.80 should normally be accomplished with wings swept between 26 and 50 degrees. Generally, response and damping about all axes in this speed range is considered excellent based on flight experience to date. Rolling maneuvers in the subsonic region (airspeeds greater than 250 KIAS but less than mach 0.80) with wings swept aft of 45 degrees are not recommended due to the fact that the spoilers are locked out aft of this wing sweep. With the spoilers locked out, roll control is significantly reduced and, therefore, aircraft roll performance is reduced. If flight is required with wings swept aft of 45 degrees, uncoordinated rolling maneuvers should not exceed 60 degrees of bank and coordinated rolling maneuvers should not exceed 360 degrees of roll (at maximum roll rate) to prevent excessive sideslip angles from being developed. Excessive sideslip angles tend to reduce the aircraft roll performance and may in some 360 degree rolls reduce the roll rate to values which may appear to the pilot as if the aircraft has ceased rolling. However, all other characteristics of the aircraft are considered good at the aft sweep angles. The angle-of-attack limits presented in Section V should not be exceeded in either 1 "g" or maneuvering flight. Based upon these angle-of-attack limits, minimum airspeeds for 1 "g" and limited maneuvering flight are presented for nominal center-of-gravity positions associated with automatic fuel sequence. (See "Minimum Airspeeds," this section.) The minimum airspeeds will vary as much as one knot from these values for each one percent MAC center-of-gravity deviation from the quoted values. These minimum airspeeds are for operational planning purposes only, and the angle-of-attack limits presented in Section V should not be exceeded in either 1 "g" or maneuvering flight.

WARNING

Under no circumstances should the angle-of-attack limits be exceeded. Possible inadvertent stall and post-stall gyrations could result from exceeding these limits.

TRANSONIC FLIGHT.

During operation of the aircraft at transonic mach numbers (mach 0.80 to 1.1) wing sweep angles of 45 to 72.5 degrees should be utilized. Refer to Section V for flight limitations with external stores. 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 and higher dynamic pressure. During transonic flight above 25,000 feet a relatively small directional trim change may occur 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.

SUPERSONIC FLIGHT.

Flight in the supersonic flight spectrum (mach 1.10 and above) should normally be accomplished with the wings fully swept. Some external store loadings preclude full aft sweep and as such are limited to 54 degrees. Flight can be performed in the supersonic speed range with wing sweep angles as low as 50 degrees; however, such sweep angles are detrimental to optimum performance. Deceleration at supersonic speeds can be greatly enhanced by sweeping the wing forward to obtain increased drag. This allows the pilot to either reduce power to aid deceleration or maintain power for more rapid acceleration should the need arise. During wing sweeping and ensuing deceleration or acceleration, aircraft trim changes will be small and will appear to the pilot principally as attitude changes. Throughout the supersonic flight spectrum covered to date, response and damping characteristics have been good; however the potential of directional instability associated with angle-of-attack in excess of handbook limits still exists.

CAUTION

As the wings are swept forward, exercise caution to avoid exceeding the speed limitations or computed MSMA indications which apply to the forward wing sweep positions, especially wing sweep angles less than 50 degrees. Refer to "Airspeed Limitations" and "Stores Limitations," Section V.

MANEUVERING FLIGHT CHARACTERISTICS.

LONGITUDINAL FLIGHT.

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 14 degrees to preclude the entrance to a stall. Stall is expected to occur at an angle-of-attack of 25 to 28 degrees. The stall warning system will activate at an angle-of-attack of 14 degrees. During flight with the slats and flaps retracted, to preclude entrance to a stall, longitudinal maneuvering should not be allowed to exceed the angle-of-attack limits in Section V.

WARNING

If airspeed decreases during maneuvering flight, the command augmentation feature of the flight control system can produce an increase in angle-of-attack without additional back stick input by the pilot. Angle-of-attack must therefore be monitored and controlled while maneuvering to insure that the limits are not exceeded.

During pullups or turns at high speed with 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. During supersonic flight at altitudes above 30,000 feet with aft wing sweeps, full back longitudinal control maneuvers can result in some stick "talkback" being detected. This characteristic is a result of the pitch damper and mechanical input attaining full noseup surface authority. Excessive rate of longitudinal control application will make this characteristic more apparent; therefore, smooth application of control is recommended. Loss of pitch damping in one direction will result but may be restored by reliev-

ing the back pressure being held. This same characteristic is exhibited at negative load factors for the aft sweep throughout its operational flight envelope.

BUFFET.

Aerodynamic buffet of the airframe is caused by the oscillatory separation and reattachment of the airflow over some portion of the aircraft surface, usually the wing. The separated flow may be due to ordinary stalling over local areas or may be induced by a shock wave caused by local flow reaching sonic velocity. Buffet onset is encountered at moderate to high altitudes in the subsonic to low supersonic speed region. This onset is dependent on flight condition and varies with wing sweep. The data presented herein relative to buffet define the onset (±0.05 "g") only. This onset is not and should not be interpreted as a flight limitation from either structural or operational standpoint. Onset is merely an initial "feel" of buffet and does not define allowable or bearable intensity which must be determined by pilot comfort or other considerations. In the lower wing sweep angles (26°) the intensity increases quite rapidly as load factor or angle-of-attack passes buffet onset conditions; while in the 72.5 degree wing sweep position there is a much slower intensity rise with increasing load factor and the intensity generally does not exceed light buffet (±0.10 "g" to 0.15 "g") at any angle-of-attack up to approximately 20 degrees. Since the altitudes at which buffet occurs are above those for optimum cruise conditions they should be avoided for normal cruise operation. Figure 6-2 presents the angle-of-attack for buffet onset determined from flight test data for the clean configuration. These boundaries are based on ±0.05 "g" buffet intensity.

WING SWEEP/MANEUVERABILITY EFFECTS.

Instantaneous longitudinal maneuver capability for wing sweeps of 26, 45, and 72.5 degrees, clean configuration, and no external stores is presented in figure 6-3. For sustained maneuver load factors, refer to Appendix I. Two typical gross weights are shown: 53,000 pounds and 70,000 pounds. The maneuver capability is based on an angle-of-attack of 15 degrees for wing sweeps of 26 and 45 degrees and an angle-of-attack of 18 degrees for a wing sweep of 72.5 degrees; heavy buffet; and maximum longitudinal control deflection. The heavy buffet line is predicated on extremely limited flight test data. For reference, an estimated maximum afterburner ceiling for 1 "g" flight is presented. Refer to "Airspeed Limitations," Section V, for airspeed limitations.

Angle-of-Attack for Buffet Onset

CONFIGURATION:
GEAR AND FLAPS UP

DATA BASIS: FLIGHT TEST
DATE: 7 MAY 1971

FUEL GRADE: JP-4
ENGINES: TF-30-P-3

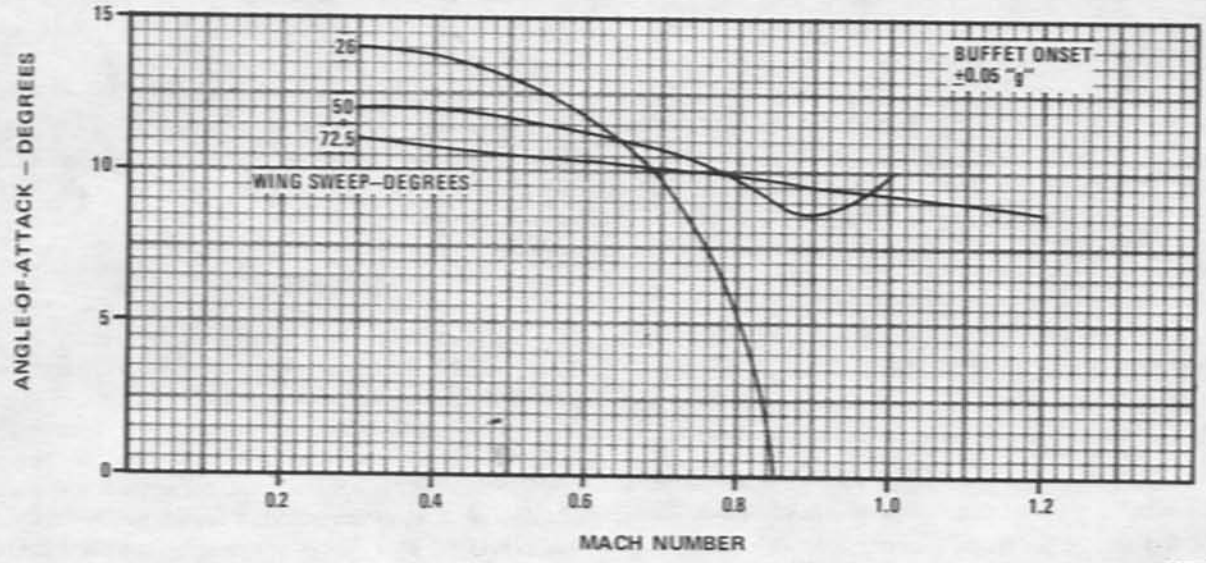

Figure 6-2.

Maneuver Capability

DATA BASIS: ESTIMATED
DATE: 19 MAY 1972

CONFIGURATION:
WING SWEEP–26 DEGREES
GEAR AND FLAPS UP
NO EXTERNAL STORES–
NO WING FUEL

FUEL GRADE: JP–4
ENGINES: TF30 –P–3

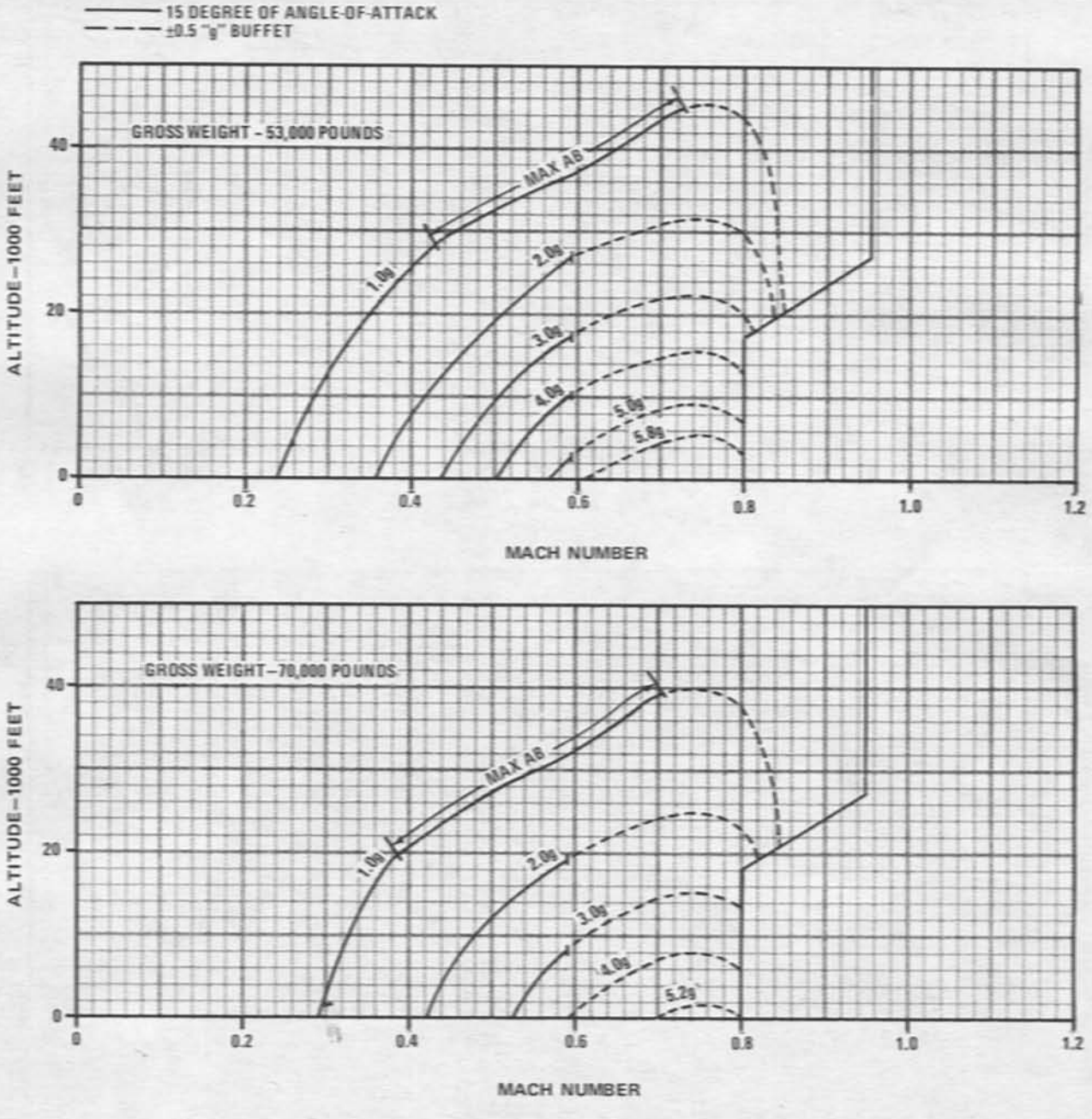

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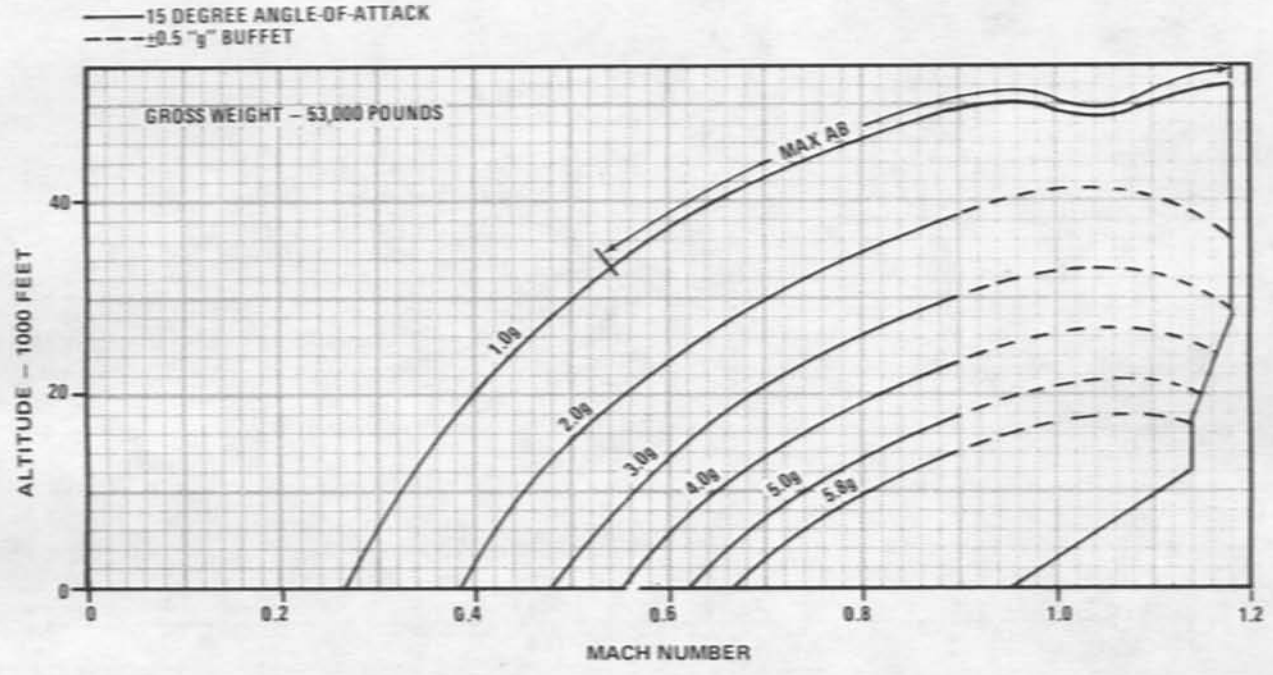
Figure 6-3. (Sheet 1)

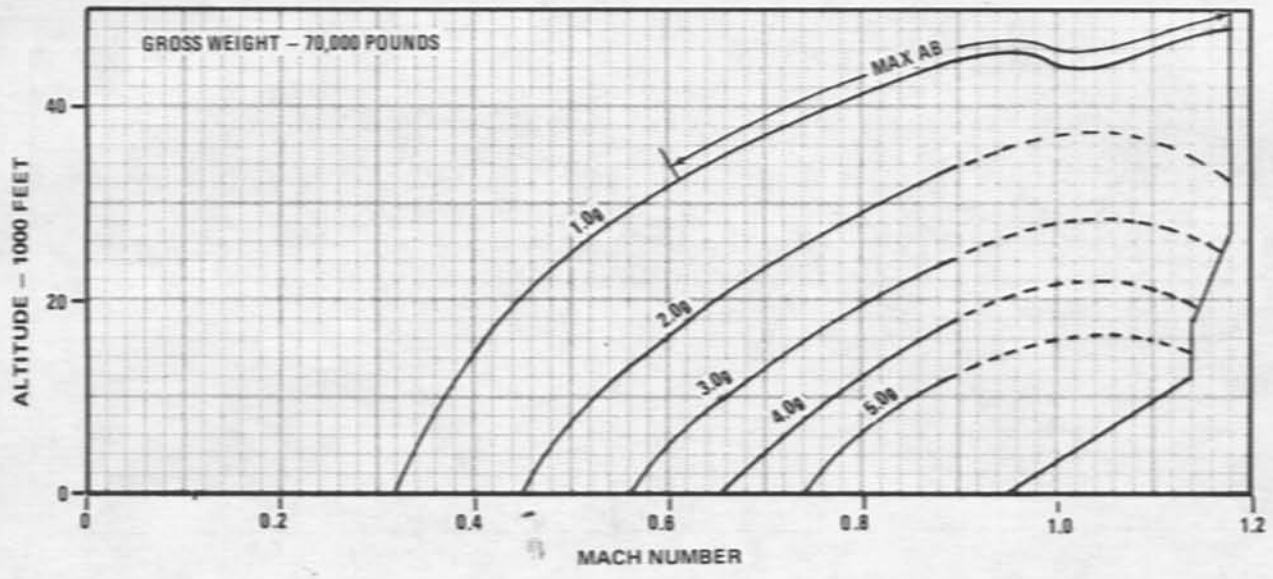
Maneuver Capability

DATA BASIS: ESTIMATED
DATE: 19 MAY 1972

CONFIGURATION:
WING SWEEP—45 DEGREES
GEAR AND FLAPS UP
NO EXTERNAL STORES—
NO WING FUEL

FUEL GRADE: JP—4
ENGINES: TF30—P—3





A0000000—E124A

Figure 6-3. (Sheet 2)

Maneuver Capability

DATA BASIS: ESTIMATED
DATE: 19 MAY 1972

CONFIGURATION:
WING SWEEP–72.5 DEGREES
GEAR AND FLAPS UP
NO EXTERNAL STORES–
NO WING FUEL

FUEL GRADE: JP–4
ENGINES: TF30– P–3

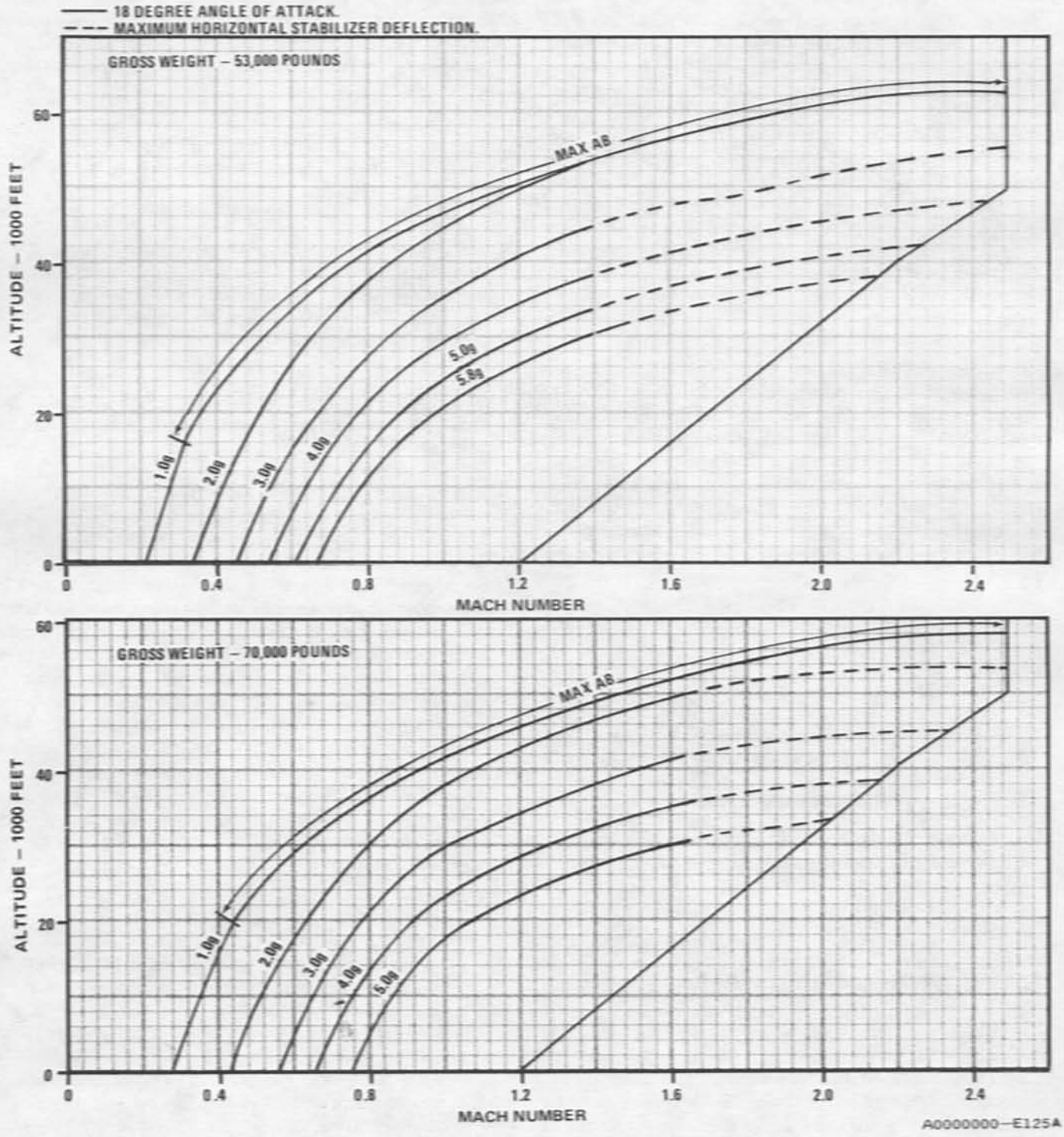


Figure 6-3. (Sheet 3)

WARNING

Flight into heavy buffet is prohibited.

In the mach 0.80 to 1.10 range in maneuvering flight, a wing sweep of 45 degrees is recommended to obtain the best overall buffet free maneuvering margin. Not only will this wing sweep provide a good maneuvering margin but also it is the most aft sweep permissible at which the spoilers are operational for roll control. Figure 6-4 presents, at a typical high subsonic speed (mach 0.80), the angle-of-attack versus wing sweep required to develop various load factors at sea level and 27,500 feet for a high gross weight. It should be noted that at the forward wing sweep of 26 degrees, buffet onset occurs at a relatively low angle-of-attack. As the wing is swept aft, the buffet onset margin improves until the wing is at 45 degrees. Aft of this sweep the buffet onset margin does not increase.

ROLLING FLIGHT.

Clean configuration roll rates up to about 160 degrees per second may be attained with lateral stick forces of fifteen pounds. Normal rolling performance below mach 0.80 decreases significantly when the wings are swept aft of 45 degrees because the spoilers become inoperative. In addition, considerable adverse yaw occurs during low subsonic speed rolling maneuvers, especially when rolls through large bank angle changes at high rates are accomplished. This is particularly apparent when the flaps are deflected and/or the aircraft is operating at relatively high angles of attack (above 10 degrees). This yawing characteristic is manifested by the aircraft nose moving in a direction opposite to the roll.

WARNING

At high speeds during maximum rolling maneuvers, abrupt forward stick motion should not be made to preclude rapid buildup in roll rate due to roll/yaw coupling.

FLIGHT WITH DAMPERS OFF.

The probability of flight without the basic stability augmentation systems in either of the pitch, roll, or yaw channels is extremely remote. Basic redundancy, failure monitoring, and self-test of the system enhance the full time operation of the system. In the event of a flight control system malfunction necessitating turning the pitch, yaw, or roll damper off in flight, the aircraft speed should be reduced to the applicable augmentation off operating limit in Section V and the affected damper turned off. Transonic deceleration should be conducted as rapidly as possible under VFR conditions if practicable. The speed brake should not be used during transonic deceleration and no attempt should be made to reduce any associated small lateral-directional oscillations. Continuing flight should be accomplished with a wing sweep of 26 degrees observing the augmentation off operating limitations for this sweep and landing should be accomplished as soon as practical.

WARNING

During flight with pitch, yaw, or roll damper off, large and/or abrupt stick and/or rudder inputs should be avoided in the damper off axis. Lateral maneuvers should be limited to 60 degree bank angle.

The following discussion is presented to point out those pertinent characteristics of the aircraft that the pilot should know.

SLATS AND FLAPS EXTENDED.

Loss of the pitch damper will result in degraded damping characteristics. As a result, airspeed control on final approach will become more difficult and increased pilot attention to maintaining angle-of-attack will be required. Reduced speed stability will be noticed by the pilot with attendant lower maneuver force gradients. Loss of the yaw damper will result in degrading damping characteristics and loss of the adverse yaw compensation also. As a result, excessively large sideslip angles can be developed during abrupt lateral inputs. Such inputs should be avoided. Loss of roll damper will result in degrading roll damping which is not considered serious. Loss of roll trim will also occur but can be compensated for by trimming the aircraft with the rudder.

Note

In the event that a landing must be accomplished with all dampers off, perform a "straight-in" approach at an angle-of-attack of 10 degrees. Avoid abrupt longitudinal and/or lateral control inputs. Approaches with the pitch damper off will require increased pilot attention to airspeed and angle-of-attack control.

Wing Sweep Maneuverability Effects

DATA BASIS: ESTIMATED
DATE: 30 MARCH 1973

CONFIGURATION:
GEAR AND FLAPS UP
SLATS RETRACTED
NO EXTERNAL STORES

CONDITIONS:
MACH–0.80

FUEL GRADE: JP–4
ENGINES: TF30-P-3

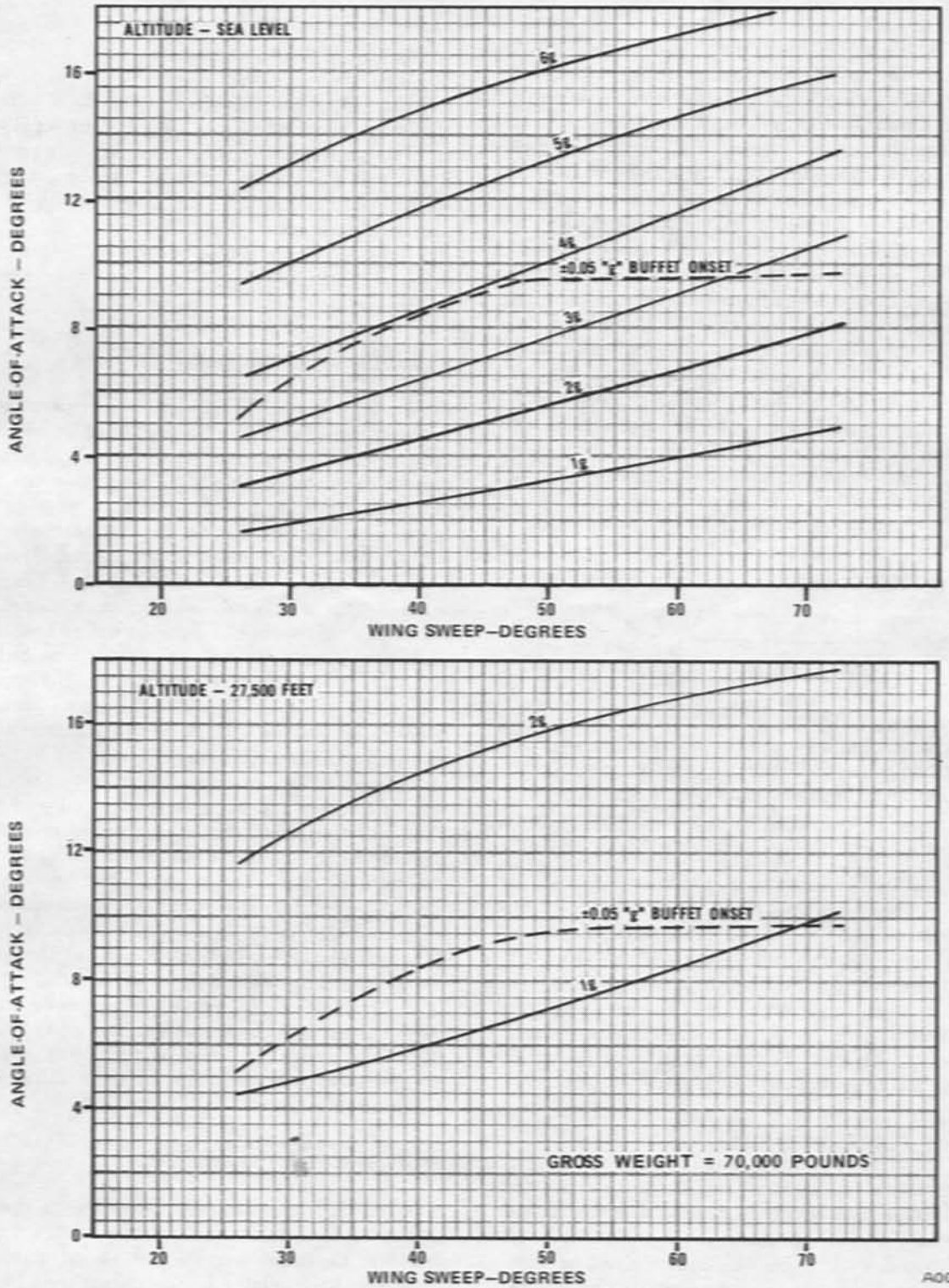


Figure 6-4.

★

SLATS AND FLAPS RETRACTED.

Loss of the pitch damper will result in degrading damping characteristics as well as the loss of the command augmentation system. Much larger variations in stick force per "g" will be reflected to the pilot. Flight in given portions of the operating spectrum is restricted due to low stick force per "g" (less than 3 pounds per "g"). With the combination of low damping, stick force per "g" and short period oscillation characteristics the pilot may be susceptible to pilot induced oscillations. Loss of the yaw damper will result in degraded Dutch roll dynamics, the most significant of these being the reduced damping. This is most pronounced at high supersonic speeds above mach 1.5. Roll inputs should be minimized to preclude excitement of the Dutch roll mode. Attempts to damp the Dutch roll mode through pilot rudder inputs should be minimized to prevent getting in-phase with the oscillations and causing the aircraft to enter a sustained oscillation. Loss of the roll damper will result in degraded damping as well as loss of the command augmentation and roll trim systems. Roll inputs with aft wing sweeps at supersonic speeds should be minimized. Basic aircraft roll damping can be augmented by sweeping the wings forward to 50 degrees at supersonic speeds. Roll trim can be accomplished by using rudder trim.

MINIMUM AIRSPEEDS.

WARNING

- Under no circumstances should the angle-of-attack limits or stall warning activation be exceeded. Possible inadvertent stall and post-stall gyrations will result from exceeding these limits.
- Minimum airspeeds shown in figure 6-5 are presented to show the lowest airspeeds that may be obtained within the current angle-of-attack limits and do not reflect thrust available. In most cases the drag at this minimum airspeed approaches or exceeds thrust available. Rapid decreases in speed and increases in angle-of-attack can result in high sink rates and/or loss of control.

SLATS AND FLAPS EXTENDED.

For the aircraft with the slats and flaps extended, the minimum airspeed is based on a wing angle-of-attack of 14 degrees. Figure 6-5 presents these speeds for 1 "g" flight and for a bank angle of 30 degrees. The indicated minimum airspeeds are representative of normal fuel sequencing within the gross weight range.

Note

At center-of-gravity positions forward of 41 percent with no external stores or forward of 39 percent with external stores and 26 degrees wing sweep, sufficient elevator may not be available to arrest sink rate due to longitudinal control power limiting.

SLATS AND FLAPS RETRACTED.

For the aircraft with the slats and flaps retracted, the minimum airspeed is based on a wing angle-of-attack of 15 degrees for wing sweeps of 16 through 49 degrees and a wing angle-of-attack of 18 degrees with wing sweeps of 50 through 72.5 degrees. Figure 6-5 also presents these speeds for 1 "g" flight and 2 "g" flight (60 degrees bank). These speeds are based on center-of-gravity positions representative of normal fuel sequencing.

THRUSTS REQUIREMENTS.

Particular attention to thrust requirements versus airspeed is essential in this aircraft because of its variable sweep wing, sharp drag rise at high angles-of-attack (typical of aircraft with high wing loading), relatively slow thrust buildup during engine acceleration, and because of the nature of the flight control system. Figure 6-6 shows how thrust required and thrust available change at different airspeeds for a typical wing sweep and gross weight with flaps and slats retracted. Thrust required can be defined as the amount of thrust needed to sustain present airspeed, altitude and "g". The pilot must be aware of the rapid drag rise, or increase in thrust requirement that exists at higher angles-of-attack. This drag rise (the steep slope in the left side of the thrust required curve) which occurs over a very small range of airspeeds, can lead to loss of control unless it is recognized and corrected. The thrust required curve has been drawn as a heavy line to the left of the lowest point and a light line to the right of the lowest point. These two parts of the thrust required curve will be considered separately, because the aircraft behaves differently on each part. The heavy-lined portion is known as "the backside of the thrust required curve" and the light portion as "the frontside of the curve." Changes in thrust, "g", airspeed, gross weight and configuration significantly affect the flying qualities of the F-111, especially at high angles-of-attack. Each of these changes will be discussed separately.

Effect of Thrust Changes.

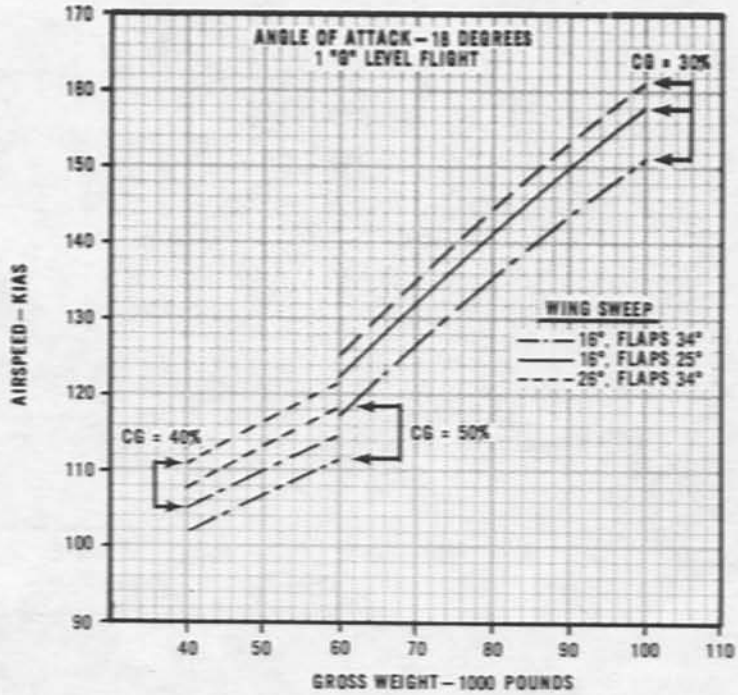
Most of the time, aircraft are flown on the frontside of the curve where it takes more thrust to fly faster and less to fly slower. When thrust reductions are made on the frontside of the curve, the aircraft slows down until it reaches a new stabilized speed at which

Minimum Airspeeds ★

DATA BASIS: ESTIMATED
DATE: 30 MARCH 1973

CONFIGURATION:
GEAR AND FLAPS DOWN
SLATS EXTENDED
NO EXTERNAL STORES

FUEL GRADE: JP-4
ENGINES: TF30-P-3



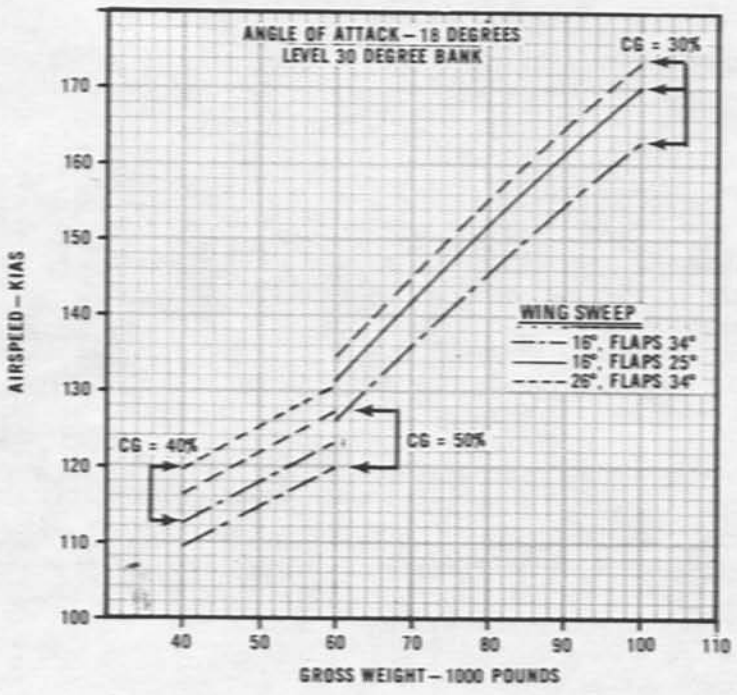


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Figure 6-5. (Sheet 1)

Minimum Airspeeds ★

DATA BASIS: ESTIMATED
DATE: 30 MARCH 1973

GEAR UP AND FLAPS UP
SLATS RETRACTED
NO EXTERNAL STORES

FUEL GRADE: JP-4
ENGINES: TF30-P-3

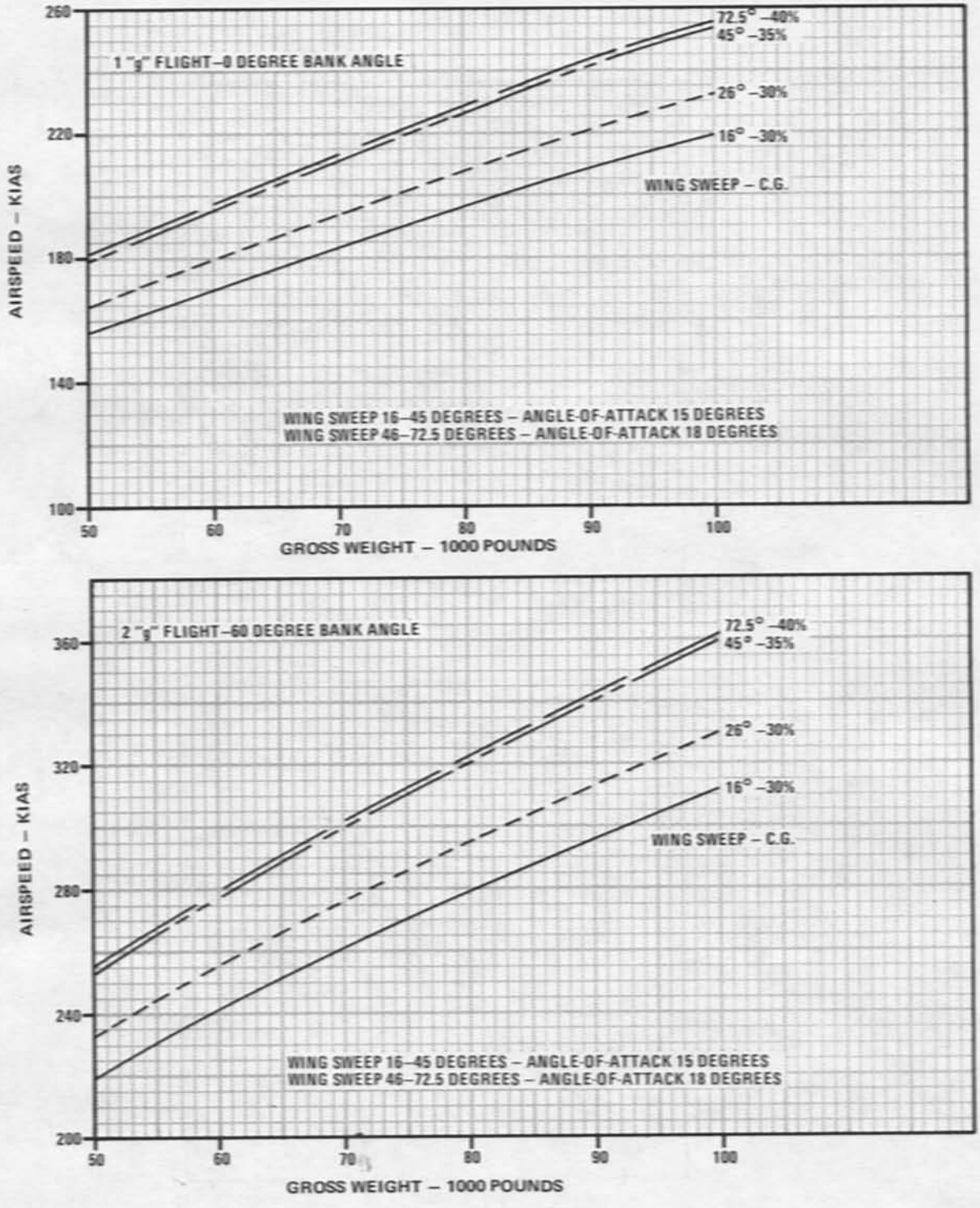

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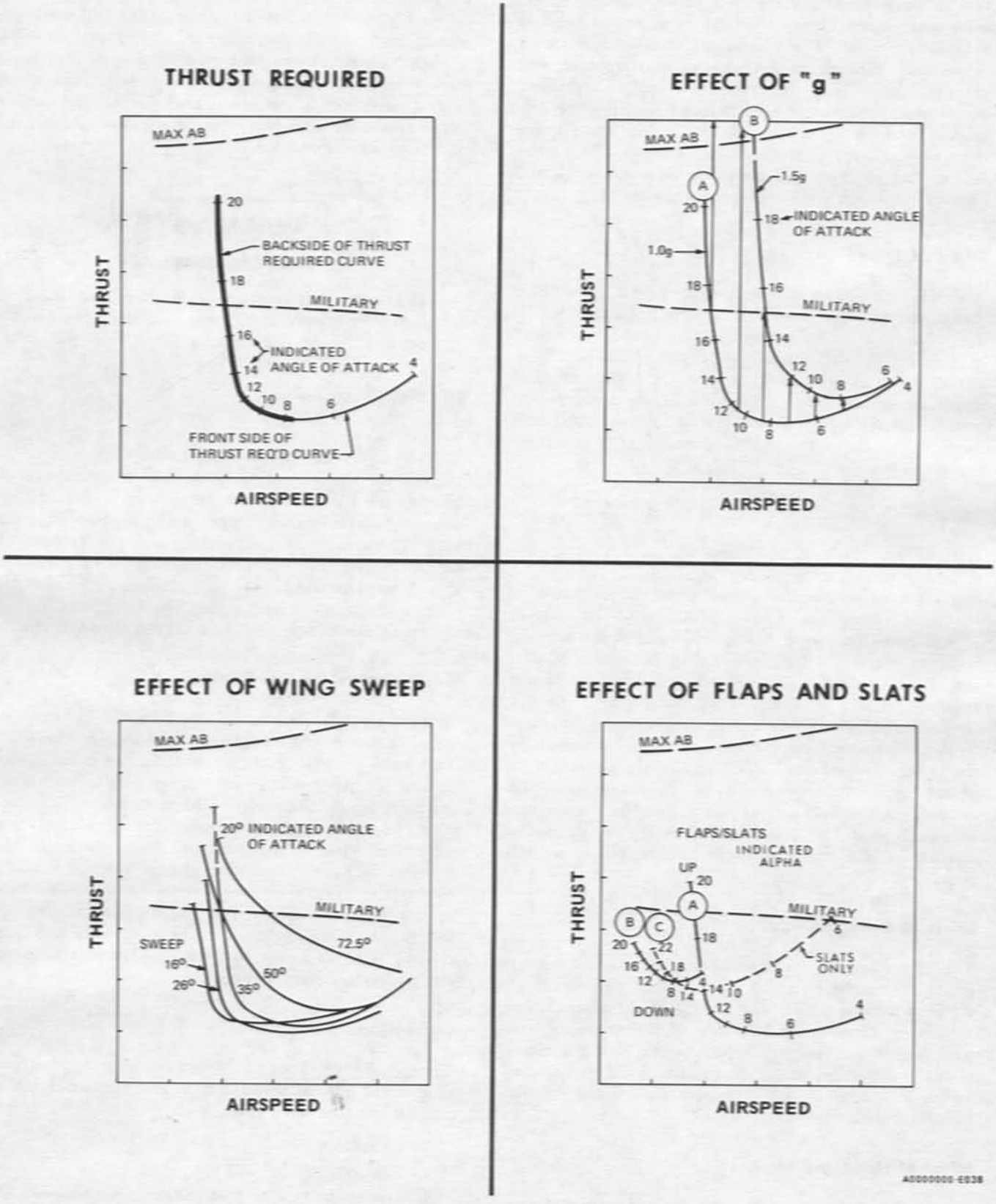
Figure 6-5. (Sheet 2)

Thrust Versus Airspeed



A0000000-E038

Figure 6-6.

thrust required equals the new thrust selected. On the backside of the curve, however, this is not true. Thrust reductions on the backside of the curve are divergent; that is, once thrust is reduced, speed will begin to reduce and, unless a correction is made, never stabilize at a lower speed. The reason may be seen in figure 6-6. Choose a point on the backside of the curve, and imagine the aircraft flying there in 1 "g" stabilized conditions. If thrust is now reduced slightly, the aircraft will begin to slow down, but at the slower speed even more thrust is required, so it slows down even faster. The aircraft will continue to decelerate until control is lost, or until a correction is made.

Effect of Speed Changes.

A similar result can also be produced by a decrease in speed at constant thrust. If, for example, speed falls off slightly due to atmospheric disturbances (gust, turbulence, etc.) during flight on the frontside of the curve, airspeed will eventually rebuild and stabilize at its initial value. On the backside of the curve, thrust will be insufficient at the lower speed, and speed will continue to decrease until a correction is made or control is lost.

Effect of "g".

An understanding of the effect of "g" on thrust required may be obtained by referring to figure 6-6 and considering the following example. Curve A is for 1 "g" flight and Curve B is for 1.5 "g" flight. Pick a point on Curve A and assume that the aircraft is flying there in 1 "g" level flight. Now assume that the aircraft rolls into a level 1.5 "g" turn; the thrust required is now determined by projecting vertically upward to Curve B. At lower airspeeds, the increase in thrust requirements can be very large (as shown by the longer arrows on the left side of the chart). Also, at any speed, increasing the "g" load can place the aircraft on the backside of the curve. At higher airspeeds, higher "g" loads are necessary to place the aircraft on the backside of the curve, but it is still possible.

Effect of Gross Weight.

The effect of gross weight upon thrust requirements is similar to the effect of "g", in that pulling 2 "g" is the same as doubling the weight of the aircraft. For a heavier aircraft, the backside of the curve extends to a higher airspeed; therefore, when flying a heavy aircraft, particular attention must be paid to angle of attack in order to avoid inadvertent flight on the backside of the curve.

Effect of Flaps and Slats.

Figure 6-6 shows three thrust required curves. Curve A is for the aircraft with flaps and slats retracted, Curve B is for the aircraft with flaps and slats extended, and Curve C is for the aircraft with slats only extended. Note that the slope of the backside of the curve is more gradual with flaps and slats extended or with slats only extended, hence, drag and angle-of-attack buildup will be easier to detect and control. Figure 6-6 also shows that if extension of flaps and slats is delayed during decelerating flight, the clean aircraft will reach the steep backside of the power curve at a much higher airspeed than it would if flaps and slats were extended.

WARNING

It is most important to remember that any delay in selection of flaps and slats can be critical during decelerating flight.

Effect of Wing Sweep.

Figure 6-6 shows thrust required curves for 16, 26, 35, 50, and 72.5 degrees wing sweep. Note that although the slope is more gradual at wing sweeps aft of 35°, the backside of the thrust required curve extends to much higher airspeeds. An important effect of wing sweep is that if wings are inadvertently left aft of 26 degrees, flaps and slats cannot be extended. This may place the aircraft at a critical airspeed in the clean configuration, and unless immediate corrections are made, thrust required may exceed thrust available.

Flight Control System Effects.

In most aircraft, deceleration at constant "g" requires either nose-up trim or back pressure on the stick. This trim change or stick force change is an indication to the pilot that speed has been lost. In this aircraft during decelerating flight at constant "g," command augmentation will produce additional elevator deflection with no pilot input. As a result, in this aircraft, stick force and trim change are not available to tell the pilot that speed is being lost. The pilot must refer to his flight instruments, particularly during decelerating flight at lower airspeeds, and must control angle-of-attack to avoid inadvertent flight on the backside of the curve.

Corrections.

There are four types of corrections that can be made to prevent loss of speed due to insufficient thrust on the backside of the curve:

1. Increase thrust.
2. Reduce "g".
3. Lower the nose to trade altitude for airspeed.
4. Change configuration.

It is important to realize that compensation for insufficient thrust must be made immediately or thrust required may quickly exceed maximum thrust available. If this happens, and if the configuration cannot be changed quickly by lowering flaps and slats, only two possible corrections remain: Reduce "g" or decrease altitude. If the aircraft is already at minimum "g" and altitude, no recovery is possible. The key to avoiding inadvertent flight on the backside of the power curve is control of angle-of-attack. By controlling angle-of-attack, the pilot can compensate for variations in wing sweep, "g" loading and gross weight, and can readily maintain a safe margin.

LOW SPEED FLIGHT-FLAPS/SLATS RETRACTED.

Low speed flight with flaps and slats retracted involves a critical angle-of-attack buildup problem. In 1 "g" flight, wings level, 11 degrees indicated angle-of-attack can be maintained with moderate power requirements at 26 degrees wing sweep. (True angle-of-attack in this speed range may be as much as 1.7 degrees greater than indicated.) Actual angle-of-attack errors are as follows:

Mach	*Angle-of-Attack Error (Degrees)*	
	Flaps Up	*Flaps Down*
Less than 0.30	+1.7	0
0.40	+0.4	−1.3
0.45 to 1.25	0	−1.7

Note

Add error to indicate angle-of-attack to obtain true angle-of-attack. Interpolate between mach numbers.

A turn requiring a 30-degree bank will increase indicated angle-of-attack to 15 degrees or greater and require military thrust to maintain speed. Any delay in applying power, or additional flight path disturbance (gusts, additional bank angle, etc.) may result in the loss of several knots airspeed, and full military thrust will not be sufficient to maintain level flight. Maximum afterburner thrust may not be attainable before thrust requirements have reached and exceeded that thrust level (a loss of only 3 to 8 knots is sufficient to cause angle-of-attack to exceed 22 degrees), resulting in a rapid (2-3 seconds) and uncontrollable deceleration into a stall, resulting in a possible post-stall gyration or spin. Rapid rotation to a nosedown attitude can halt this deceleration, but will be extremely dangerous at low altitude. It is imperative that the recommended no flap/slat angle-of-attack for landing approach not be exceeded (11 degrees for wing sweeps 16-45 degrees and 12 degrees for wing sweeps greater than 45 degrees). Maneuvering at angle-of-attack in excess of 10 degrees should be avoided.

STALL/LOSS OF CONTROL CHARACTERISTICS.

The aircraft obtains a large amount of lift from its fuselage and glove, particularly at high angles of attack. During the approach to stall, wing lift decreases rapidly while fuselage and glove lift continue to increase. As a result of this continuing lift, the aircraft does not exhibit the conventional "g" break associated with stall on most other aircraft. Since sufficient longitudinal control power exists at all wing sweeps to pull the aircraft up to angle-of-attack in excess of those where directional stability goes to zero (i.e., zero restoring moment to yaw disturbance), stall angle-of-attack is defined as the angle-of-attack at which directional divergence occurs.

WARNING

- Aircraft drag at high angle-of-attack may exceed total thrust available. This will result in a loss of airspeed, altitude, or "g" capability, and can lead to rapid loss of control unless the crew becomes aware of the situation and takes immediate corrective action. This is a particularly critical condition during 1 "g" flight at minimum altitude, where no corrective action may be possible.
- The pitch command augmentation feature of the flight control system will attempt to maintain the stick commanded level of pitch rate and "g" force independent of airspeed variations. During flight conditions where airspeed is decreasing, the horizontal stabilizer will be commanded to increase angle-of-attack without additional pilot input. Under these conditions, unless the pilot is monitoring and controlling angle-of-attack, command augmentation can cause angle-of-attack to increase until control is lost.

STALL WARNING CHARACTERISTICS.

Test results have shown that no sudden, abrupt motions will occur to warn of stall. As angle-of-attack increases during a typical stall approach, the following stall warning characteristics may occur in the order shown.

1. Artificial Stall Warning System (after T.O. 1F-111-891). This system provides simultaneous actuation of the pedal shaker, a flashing red lamp and a steady tone. System operation is described in Section I.

2. Rudder Pedal Shaker (prior to T.O. 1F-111-891). Although the rudder pedal shaker will be activated, it has not proved to be an effective stall warning device. It may be masked by airframe vibration or buffet, and obviously requires that the pilot's feet be on the pedals.
3. Indicated Angle-of-Attack Will be Above Section V Limit. Stall is an angle-of-attack related event. Stalls can occur at a variety of airspeeds, "g" loadings, gross weights, wing sweeps, attitudes, thrust settings, external store configurations and flap/slat positions, but always occur as a result of excessive angle-of-attack. The aircraft cannot be stalled within angle-of-attack limits. Angle-of-attack indications are reliable up to 22 degrees. If angle-of-attack exceeds 25 degrees between mach 0.4 and 1.25, the CADC caution lamp and the master caution lamp will light.
4. A High Sink Rate Most Evident During 1 "g" Stalls. During flight test 1 "g" stall approaches at MIL thrust, sink rates of 3,000 to 6,000 feet per minute developed. Prestall sink rates were greater at aft wing sweeps. The sink rate may not be noticeable during maneuvering flight.
5. Precise Attitude Control Becomes Difficult. If the pilot is attempting to control attitude precisely, pilot control inputs increase in size and number just before 1 "g" level flight stalls. This will not be a useful stall warning cue during stall when the pilot is not attempting to control attitude precisely.
6. Wing Rock and Degraded Roll Control. A small amount of low amplitude, low frequency wing rock or wing drop may occur. Roll control effectiveness will rapidly degrade as stall angle-of-attack is reached. A continuous lateral stick input may be necessary just to keep from rolling or to continue a desired angle of bank. Roll damper saturation will be indicated by a large increase in lateral stick force.
7. Degraded Directional Stability. If the stall is approached slowly, the nose of the aircraft will gradually and smoothly begin to wander to the left or right. A gradual increase in side forces may be noticed. This will begin a few seconds prior to complete loss of control.

CAUTION

- There is no sudden loss of lift ("g" break) or change in stick force or position associated with aircraft stall. Pre-stall buffet may exist, but it is not a dependable stall warning since it varies for different configurations and its intensity may remain constant with increasing angle-of-attack. Buffet is very light and may not be noticed at aft wing sweeps.
- If the stall is approached rapidly, the natural aerodynamic cues will be effectively nonexistent. During hard maneuvering, angle-of-attack must be monitored and the artificial stall warning must be heeded.
- During maneuvering flight at high angles of attack within limits, large or abrupt control inputs should be avoided as they may cause unintentional angle-of-attack and sideslip excursions and contribute to loss of control.

In all cases and in all configurations, the immediate action which must be taken upon recognition of impending departure is to unload the aircraft and reduce angle-of-attack. Sufficient elevator power is available at all wing sweeps to effect recovery right up to the point of departure. There should be no effort made to counter uncommanded roll or yaw motions with roll control or rudder as these inputs may aggravate the situation. An immediate, forward stick displacement is the best means of lowering the angle-of-attack and recovering a controlled flight condition. Experience has shown that stalls can occur with little warning, and that the motion of the aircraft prior to, during, and following stall can be deceptively smooth and comfortable. The timing of recovery control application is critical. A momentary delay may mean complete loss of control and possibly loss of the aircraft. Stall avoidance is of particular importance, since the chances of recovery from a fully developed out-of-control condition are not good due to large altitude losses.

WARNING

If the preceding stall warning cues are not recognized, and stall is permitted to occur, the critical and immediate action required is to put the stick full forward and centered. Any delay can produce sustained out-of-control flight, from which the chances of recovery are not favorable.

DEPARTURE FROM CONTROLLED FLIGHT.

Departure from controlled flight is the event in the post-stall flight regime which precipitates entry into a post-stall gyration or spin. Departure is the brief aircraft motion which constitutes a transition from a controllable flight condition to complete loss-of-control. Departure is evidenced by a yaw divergence (nose slice) followed by an initial rolling motion in the direction of the yaw. After departure, the motion may continue in a rolling fashion for several rolls (probably the most prevalent form of post-stall gyration) or

the aircraft may directly enter a spin. The most predominant indication of departure is a yaw acceleration. At low airspeeds the departure will be smooth and fairly slow. For high airspeed entries, the departure will be more rapid.

WARNING

The critical and immediate action which must be taken when the pilot realizes that the aircraft has departed controlled flight is to reduce angle-of-attack. The out-of-control recovery procedures must be given time to be effective. Maintain these controls until type of maneuver is identified.

OUT-OF-CONTROL MOTIONS.

Note

The flight characteristics information beyond departure is based upon limited flight test data.

Following a departure from controlled flight, the aircraft may undergo any or all of the following different types of out-of-control motions: post-stall gyrations, upright or inverted spins, and inertia-coupled recovery rolls. Each out-of-control mode has certain characteristics which may enable the pilot to differentiate between them and to take the appropriate corrective action.

Post-Stall Gyration.

A post-stall gyration is uncontrolled motion about one or more aircraft axes following departure. Although the motions differ from the motions occurring at departure, no additional control action is required. Maintain the "Out-of-Control Recovery Procedures", Section III. Although a majority of the gyration occurs at a post-stall angle-of-attack, lower angles may be encountered intermittently in the course of the motion. The spin is differentiated from the post-stall gyration by the spin's predominant yaw rotation at a continuous post-stall angle-of-attack. In effect, the post-stall gyration will be any out-of-control event that is not specifically recognized as a spin or an inertia-coupled roll. The post-stall gyration will probably be of a rolling nature, although the motions may be somewhat random. Its characteristics are uncommanded motions (primarily roll and not yaw), an angle-of-attack indication generally above 25 degrees, and a low airspeed. Because the post-stall gyration will demonstrate primarily a rolling motion, it can easily be confused with the recovery roll (see "Recovery Characteristics", this section). The latter occurs near or within angle-of-attack limits and has its own recovery steps. It is reemphasized that upon the first indication of loss of control, apply the "Out-of-Control Recovery Procedures," Section III.

Spin.

A spin is a continuous uncommanded yaw rotation at angles of attack above stall. The aircraft will enter spins from both upright and inverted conditions. Due to the large rate of altitude loss during an out-of-control situation (18,000 to 24,000 feet per minute), chances of recovery from a fully developed spin are marginal, particularly if the spin is entered at altitudes of 24,000 feet AGL or below. During a fully developed spin, flight control system hydraulic pressure may be lost if the rpm of both engines decreases below 35 percent.

Note

In all out-of-control conditions, one or both engines may stall. Stall will not be recognizable to the pilot as there will be no loud compressor stall. The engine(s) rpm will begin to decrease and TIT will increase due to insufficient airflow. Engine rpm will decrease to about 40 percent if the out-of-control situation persists.

Upright Spin.

Spin entry may occur directly following departure, or from a post-stall gyration. If the spin is entered directly from a high speed departure, the aircraft will initially follow a ballistic trajectory in which the yaw rotation appears to the pilot to be similar to a roll because of the alternating view of the ground and the sky. As the aircraft's forward velocity is reduced, the trajectory will become vertical and the yawing motion will become more evident. During a spin the ground will appear to sweep horizontally across the pilot's field of vision. Angle-of-attack will indicate between 22 and 25 degrees, but may occasionally show erroneous readings as low as 0 degrees during large nose-left sideslip conditions. Airspeed will indicate 140 KIAS or less. The motion will be smooth and constant without buffet. The turn needle will be pegged in the direction of the spin. Upon determining that the aircraft is in a spin, apply "Spin Recovery" procedures. Both full lateral control and forward stick are required for spin recovery. In order to obtain full lateral control deflection, it will be necessary to remove some of the forward stick used during the "Out-of-Control Re-

covery Procedure," Section III. This is because of the pitch-roll mixer limits and authority limits. Also, with the roll damper off, the lateral stick will have to be moved through the detent position to the mechanical stop to obtain full lateral surface deflection. While the aircraft is spinning, normal acceleration will remain relatively constant at approximately 1 "g". As recovery begins, however, "g" will begin to vary between increasing positive and negative values. This rougher, more oscillatory pitching motion of the aircraft should indicate to the crew that recovery is in progress. Shortly thereafter the aircraft may assume a steeper nose-down pitch attitude and the aircraft motion may become primarily rolling rather than yawing. As control is regained, the aircraft will finally respond to the forward stick input by unloading to zero or negative "g". Immediately neutralize rudder and aileron to avoid entering a spin in the opposite direction. Continue to apply forward stick as necessary to maintain approximately zero "g" and zero degrees angle-of-attack. This forward stick should not be removed until dive recovery airspeed (approximately 300 KIAS) is obtained. All large amplitude oscillations should have ceased by this time. Some uncommanded oscillations may still exist as dive recovery speed is reached; however dive recovery should be initiated even if such residual motions exist. Angle-of-attack should be monitored to insure that recovery has occurred. Note that low angle-of-attack alone is insufficient indication of recovery. Both angle-of-attack and airspeed must be checked. Aircraft oscillations may persist for several cycles after control is regained especially if dampers are off. During the recovery process, the aircraft will initially be in a nearly vertical attitude and external visual cues may be confusing. Continual monitoring of angle-of-attack and altitude is necessary. A smooth dive pullout should be commenced at approximately 300 KIAS observing angle-of-attack limits. If control of the aircraft has not been regained by 15,000 feet AGL, eject.

Inverted Spin.

An inverted spin will be very similar in nature to an upright spin except that the crew will be subjected to approximately a negative 1 "g" condition and the angle-of-attack indicator will be pegged at —2 to —3 degrees. Although the inverted condition might generate confusion in identifying the direction of rotation, referring to the turn needle will always indicate the direction of rotation. Immediately upon determining that the aircraft has entered an inverted spin, apply inverted spin recovery procedures. Erroneous angle-of-attack information will be presented on the AMI while the aircraft angle-of-attack is below the probe limit (—2 to —3 degrees). Once the yaw rotation approaches zero and the nose falls through toward the vertical, rudder must be neutralized to avoid spin reversal.

Roll Coupling.

Coupling results when a disturbance about one aircraft axis causes a disturbance about another axis. An example of coupled motion is the disturbance produced by a rudder deflection which produces a combination of yawing and rolling motions. This interaction results from aerodynamic characteristics and is termed aerodynamic coupling. An example of uncoupled motion is the disturbance produced by an elevator deflection during level flight. A pitching motion occurs without disturbance in yaw or roll. A separate type of coupling results from the inertia characteristics of the aircraft. The inertia characteristics of the complete aircraft can be divided into the roll, yaw and pitch inertia, and each inertia is a measure of the resistance to rolling, yawing or pitching acceleration of the aircraft. The aircraft has a roll inertia which is quite small in comparison to the pitch and yaw inertia, that is, its resistance to roll is low. Inertia coupling can be illustrated by considering the mass of the aircraft to be concentrated in two elements, one representing the mass ahead of the cg and one representing the mass behind the cg. If the aircraft rolls about an axis which passes through these two mass concentrations (inertia axis) no inertia coupling would result from the following motion. If the roll axis is inclined with respect to the inertia axis, rotation about the roll axis will produce centrifugal forces and cause either a yawing or a pitching moment. This is inertia coupling. As a result of aerodynamic and inertia coupling, rolling motions can produce a great variety of longitudinal and lateral-directional forces and moments. All aircraft exhibit varying degrees of aerodynamic and inertia coupling. Roll coupling causes no problem if the moments are easily counteracted by the aerodynamic restoring moments. Under certain conditions this aircraft, like most fuselage heavy aircraft (most of the mass concentrated along the longitudinal axis), can be forced into roll coupling. During rolling maneuvers the combination of forward stick and lateral stick in the direction of the roll can produce an uncommanded roll rate increase. Roll rates of up to 200 degrees per second may occur and be sustained. To recover, neutralize controls. Roll rate should begin to decrease immediately. Angle-of-attack may tend to increase as roll rate decreases, and should be controlled by using forward stick as required. During a sustained roll-coupled condition, angle-of-attack will usually be below 20 degrees, and airspeed will usually be between 200 and 350 KIAS. While a spin will appear to be primarily a yawing motion, roll coupling will be similar to a high roll rate aileron roll. To recover, neutralize controls and wait for the high roll rate to subside. Roll rate should begin to decrease immediately, and although uncommanded rolling will continue for 1 or 2 turns, recovery should be complete within 5 to 10 seconds. If uncommanded roll rate has not subsided within 5 to 10 seconds, rudder should be applied opposite the roll direction.

RECOVERY CHARACTERISTICS.

Recovery is defined as the transition from out-of-control conditions to controlled flight. Stall recovery, post-stall gyration recovery, spin recovery and recovery rolls will be discussed separately.

Stall Recovery.

If recovery controls are applied immediately as stall occurs, uncommanded yawing and rolling motions will stop and control will be restored. If the stall is entered from a high rate condition, control will probably be lost. Timing is important. A one-second delay in applying recovery controls may make the difference between immediate recovery and sustained uncontrolled flight. The key cockpit indications of stall recovery are angle-of-attack below 15 degrees and decreasing, and airspeed above 200 KIAS and steadily increasing. When recovery is assured, forward stick deflection may be reduced. Gradual and careful application of back stick may then be used to recover to level flight. Angle-of-attack must be closely monitored during pullout following stall recovery, as it would be easy to reenter the stall.

Post-Stall Gyration Recovery.

Recovery from a post-stall gyration may be recognized by angle-of-attack below 15 degrees and decreasing, and airspeed above 200 KIAS and increasing. In addition, the aircraft will assume a steeper nose-down pitch attitude, and will unload to zero or negative "g". When recovery is assured, gradually and carefully reduce forward stick deflection and, as airspeed continues to increase, commence a recovery to level flight, controlling angle-of-attack within limits.

Spin Recovery.

The key cockpit indications of spin recovery are angle-of-attack below 15 degrees and decreasing, and airspeed above 200 KIAS and steadily increasing. There are also several physical cues which will aid the pilot in correctly assessing recovery from a spin. These include the following:

1. Rougher, more oscillatory motion of the aircraft as yaw rate decreases.
2. Steeper nose-down attitude.
3. Unloading to zero or negative "g" (if the spin is inverted, unloading to zero or positive "g").
4. Normal aircraft response to flight control inputs is regained. Aircraft oscillations may persist briefly after recovery has occurred, however, a cross-check of angle-of-attack and airspeed will confirm that these are temporary recovery oscillations and not out-of-control motions. No attempt to oppose these motions is necessary or should be made. Continue to apply forward stick as necessary to maintain approximately zero "g" and zero degrees angle-of-attack. This forward stick should not be removed until dive recovery airspeed (approximately 300 KIAS) is obtained. All large amplitude oscillations should have ceased by this time. Some uncommanded oscillations may still exist as dive recovery speed is reached; however, dive recovery should be initiated even if such residual motions exist.

Recovery Rolls.

During the recovery phase of a post-stall gyration or spin, the aircraft will experience an uncommanded roll or series of rolls near or below the stall angle-of-attack. These rolling motions could be caused by a control input or roll coupling, and serve as a further indication that the aircraft has recovered. Airspeed will be steadily increasing above 200 knots during these rolls, and angle-of-attack may increase to 15 to 20 degrees. Having verified that the aircraft is not spinning, neutralize roll control and use forward stick as necessary to keep angle-of-attack within limits. As uncommanded rolls stop and airspeed continues to build, the aircraft can be maneuvered to the proper attitude for dive pullout.

ALTITUDE LOSS AND DIVE PULLOUT.

Altitude loss during out-of-control conditions will depend on entry conditions (airspeed, altitude and vertical speed), configuration (gross weight, wing sweep and store loading), type of motion encountered (stall, post-stall gyration, roll coupling or spin), duration of out-of-control flight and pilot technique. If the aircraft is stalled from 1 "g" level flight and recovered without entry into a post-stall gyration or spin, a minimum of 3,000 feet altitude may be required to recover to level flight at the forward wing sweeps. At aft wing sweeps, the altitude lost during recovery to level flight may be doubled. To minimize altitude loss, the wings, if aft of 45 degrees, should be swept forward during dive recovery. If the stall occurs during high speed maneuvering flight, altitude requirements for recovery may be reduced, particularly if the aircraft was in level flight or climbing when the stall occurred. Altitude loss during a post-stall gyration and recovery can vary from 6,000 to 10,000 feet or more, depending upon entry conditions, configuration and maneuver duration. If a spin is encountered, altitude will be lost at the rate of 18,000 to 24,000 feet per minute. During the time required for recovery, a substantial amount of altitude will be lost (a minimum of 24,000 feet). Chances of recovery to level flight from a fully developed spin are therefore marginal, and become increasingly poor for lower altitude entries. Recovery capability will be marginal for any departure from controlled flight occurring

below 6,000 feet AGL, for a post-stall gyration entered from below 10,000 feet AGL and for a spin entered from any altitude, particularly below 24,000 feet AGL. It is not recommended that external stores be jettisoned during an out-of-control situation because of a possible collision of aircraft and stores. Stores may be jettisoned during dive pullout if altitude is critical. Angle-of-attack limits must be observed during recovery to level flight. Dive recovery information can be obtained from the "Dive Recovery" paragraph in this section. During dive recovery pullouts, the flight control system and drag characteristics can easily contribute to an overrotation and lead to another out-of-control condition. The dive pullout should be conducted at no more than 15 degrees angle-of-attack. The wings, if aft of 45 degrees should be swept forward to minimize altitude loss.

DIVE RECOVERY.

This section presents data to determine the altitude lost during recovery from various dive angles. Data are based on the clean aircraft for sweep angle of 26 to 72.5 degrees.

Note

If the dive recovery chart is used for planning dive entries and pullouts, the stall warning system will be activated during pullout.

A dive recovery chart is presented in figure 6-7 and may be used as follows:

Given:

Wing Sweep — 26 degrees
Dive Angle — 30 degrees
Airspeed — 500 KIAS
Ambient Temperature — 0°C
Start Recovery — 6000 feet
Desired Load Factor — 3.0 "g's"
Gross Weight — 60,000 lbs.

Find:

Altitude required to recover

Solution:

Enter figure 6-7 at 500 KIAS A , proceed horizontally to the right to 6000 feet pressure altitude B , move vertically down to 0°C C , proceed horizontally to the right to 30 degrees dive angle D , move vertically upward to the 3.0"g" load factor line E , and project horizontally to the right and read 2000 feet required to recover F . To check the capability of the aircraft to attain the desired load factor within set angle-of-attack limits, enter figure 6-7 at 500 KIAS as before A , proceed horizontally to the left to the 60,000 pound gross weight line G , then project down to read 6.6 "g's" as the load factor H . Thus the desired 3.0"g" pullout can be accomplished without exceeding the 15 degree angle-of-attack limitations on which the 26 degree sweep load factors are based.

DRAG EFFECT ON DIVE RECOVERY.

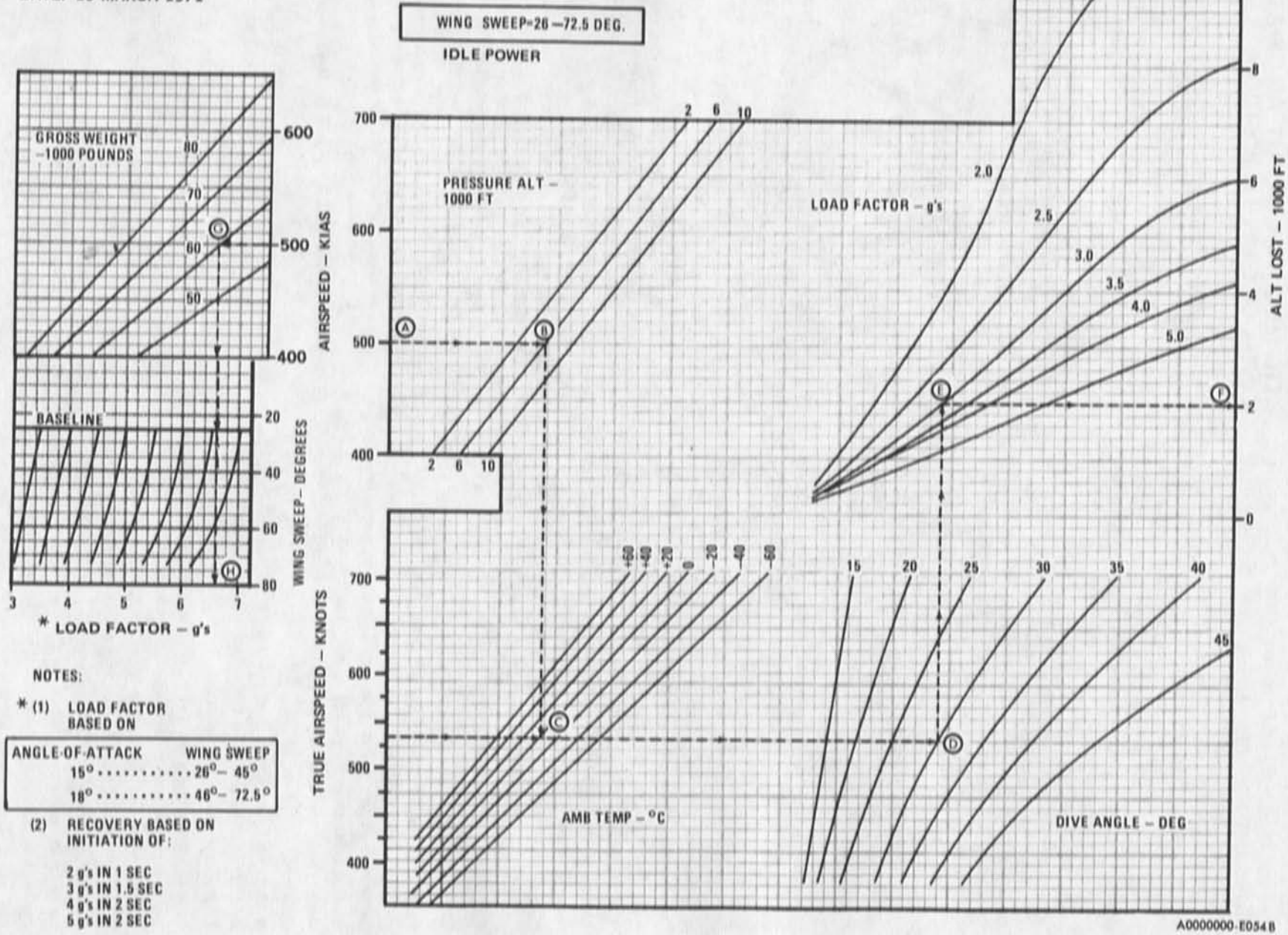
The difference in altitude loss in dive recovery due to weapons drag effect is negligible when compared with altitude to recovery from a dive for a clean aircraft. As drag is increased the altitude to recover is decreased. For example, for a weapons drag index of 200 the altitude to recover will be about 30 feet less than that required for a clean aircraft.

FLIGHT WITH EXTERNAL STORES.

The most predominant effects of the external store loadings, other than performance effects, are the increased weight, inertia and aircraft sensitivity at high subsonic mach numbers (mach greater than 0.80). Abrupt control inputs should be avoided to preclude exceeding structural limitations associated with the various loadings. High roll rates are attainable when full normal lateral stick displacement is used for rolling maneuvers. This is true even with heavy loadings of wing/pylon mounted stores. The use of full normal lateral stick for rolls can produce roll rates in excess of the maximum allowable roll rates which are stated herein for many external store loadings. It is difficult to define an operating procedure that will prevent exceeding the maximum allowable roll rates with external stores installed. These allowable rates are considerably lower than those produced by lateral stick displacement to the force detent position (2.4 inches). The following comments are offered as a guide. Avoid using abrupt lateral stick inputs as a routine flying technique with external stores installed. Abrupt stick input tends to cause high roll acceleration and roll rate. A recommended technique is to pressure the stick to a lateral displacement rather than a forceful displacement. An alternate is to limit lateral stick displacement to ½ that required at the force detent position (1.2 inches). This amount of lateral stick will produce roll rates approximately 50 to 70 degrees per second depending upon the store loading and wing sweep angle. Flight tests have shown that this "½ stick" displacement, abruptly applied (in about 1/10 second), results in 50 to 70 degree per second roll rates in as little as 60 degrees of bank angle change. Slower stick input to the same displacement allows larger changes in bank angle before the 50 to 70 degree per second roll rate is attained. This 50 to 70 degree per second roll rate is equal to or less than the

Dive Recovery (No Safety Factor)

DATA BASIS: ESTIMATED
DATE: 19 MARCH 1971



NOTES:

* (1) LOAD FACTOR BASED ON

ANGLE-OF-ATTACK	WING SWEEP
15°	26° – 45°
18°	46° – 72.5°

(2) RECOVERY BASED ON INITIATION OF:

2 g's IN 1 SEC
3 g's IN 1.5 SEC
4 g's IN 2 SEC
5 g's IN 2 SEC

A0000000-E054B

Figure 6-7.

roll rate limits applicable to most store loadings. It is noted that some store loadings and speed ranges allow clean aircraft roll rates obtainable with detent force displacement of the stick. Weapon separation has very little effect on aircraft response at weapon release. In most cases, the incremental normal acceleration has been less than 0.5 "g". Maximum roll and sideslip response experienced to date has been less than 5.0 degrees of roll and 2.0 degrees of sideslip. During drops conducted with retarded weapons, M-117R and MK-82S, slight aircraft pitching motions have been experienced as the weapons pass under the aircraft horizontal stabilizer. This phenomena is due to the pressure distribution change on the horizontal stabilizer caused by the opening of the retard fins under the stabilizer. Although this light pitching motion poses no flight safety problem, it is a phenomena of which the pilot should be aware. As external stores are released, an asymmetrical store loading requiring more lateral trim than available may be encountered. If such a condition occurs use rudder trim as necessary. The aircraft has the capability of being flown safely and landed with any asymmetric down loading of the authorized store loadings presented in Section V. The asymmetric effects will become more evident as the airspeed is decreased or as the load factor is increased. Proper use of the authorized normal release modes will preclude the most severe asymmetries from occurring. Many asymmetric load possibilities require no additional flight restrictions; and selective jettison, to remove most asymmetric conditions, can be accomplished if desired. Refer to Section V for "Asymmetric Store Load—Flight Limitations" and to Section III for "Landing with Asymmetric Stores."

FLIGHT WITH SPEED BRAKE EXTENDED.

Extension of the speed brake will result in an aircraft noseup trim change. In the speed range (mach 0.80 or less) where the elevator angle for trim with the brake deflected does not exceed four degrees down (series trim limit), the flight control system will retrim the aircraft for the pilot. In the speed range (mach 0.80 to 1.0) where the elevator angle for trim is greater than four degrees down, the pilot will have to augment the series trim with the stick since the series trim is at its authority limit. Extension of the speed brake will also result in aircraft buffet and pulsating motion will be reflected in the LCOS pipper and may cause difficulty in keeping the pipper on the target during speed brake extended operations. Refer to Section V for operating limitations for speed brake extension.

DETERMINATION OF AFT ALLOWABLE CENTER-OF-GRAVITY POSITION.

The aft allowable center-of-gravity positions presented in this section are given to allow calculation of, and verification of the minimum fuel values presented in "Center-of-Gravity Limitations," Section V. In order to calculate the minimum fuel loadings presented in Section V, the user must not only determine the aft allowable center-of-gravity position, but must also have access to the aircraft weight and balance handbook, T.O. 1-1B-40. Figure 6-8, "External Stores and Weapon Loading Configurations," is provided to determine the stores indexes to be used to calculate the aft allowable center-of-gravity position for various store loadings. Figure 6-8 presents a listing of stores presently planned for carriage. Refer to "Stores Limitations," Section V, for a listing of stores authorized for carriage and release. Figure 6-9 presents the aft allowable center-of-gravity positions as a function of wing sweep angle and mach number for all gross weights. The basic limits are presented as center-of-gravity versus wing sweep and mach number. The data for gear and flaps retracted is presented for two mach ranges of 0.80 or greater and mach 0.55 or less. For mach numbers between 0.80 and 0.55, linear interpolation should be used. In addition to the basic limits, increments are provided to account for configuration and/or flight condition changes. The aft allowable center-of-gravity position is the sum of the basic limit plus all of the applicable increments. The longitudinal aft allowable positions for wing sweep angles of 16 through 50 degrees are based on maintaining a one percent static margin with gear and flaps retracted, and zero static margin with gear and flaps extended. The directional stability aft allowable positions for wing sweep angles of 50 through 72.5 degrees are based on maintaining a minimum level of directional stability.

Example: Determine the aft allowable center-of-gravity position for store loading, wing sweep, and airspeed for specific phases of flight with B-43 weapons loaded.

Given: Stations 4 and 5, 3 and 6: B-43 weapons for each of the following:

a. Takeoff: Low speed, flaps and gear down, 16 degree wing sweep, 25 degree flap.
b. Low Speed: Flaps and gear up, 16 degree wing sweep, slats extended, mach less than 0.55
c. Low Speed: Flaps and gear up, 26 degree wing sweep, clean aircraft, mach less than 0.55
d. High speed: 26 degree wing sweep, mach 0.70.
e. High speed: 72.5 degree wing sweep, mach 2.20.
f. Landing: Low speed, flaps and gear down, 26 degree wing sweep.

★External Stores And Weapon Loading Configurations

(For determining the aft allowable center-of-gravity position. Bay loading does not affect the aft allowable center-of-gravity limits, but does affect center-of-gravity location.)

PYLON STATIONS (1)			INDEX		PYLON STATIONS			INDEX	
2 & 7	3 & 6	4 & 5	Long.	Dir. (2)	2 & 7	3 & 6	4 & 5	Long.	Dir.
None	None	None	A	E		2 BLU-27/B	2 BLU-27/B	C	*
	Pylon	Pylon(3)	A	E	2 BLU-27/B	2 BLU-27/B	2 BLU-27/B	D	*
Pylon	Pylon	Pylon	A	*		6 CBU-24		C	*
	AA-37U-15		B	*		6 CBU-24	6 CBU-24	C	*
	Pylon	AIM-9B	B	F		CBU-30/A	CBU-30/A	B	*
	2 AIM-9B	AIM-9B	B	F	CBU-30/A	CBU-30/A	CBU-30/A	D	*
	2 AIM-9B	B-43	B	G		CBU-49B/B		B	*
	600 gal tank	B-43	C	*		Pylon	4 M-117	C	*
	B-43	B-43	B	G		4 M-117		C	*
	2 AIM-9B	B-57	B	F		Pylon	6 M-117	C	*
	600 gal tank	B-57	C	*		6 M-117		C	*
	B-57	B-57	B	F		4 M-117	4 M-117	C	*
	2 AIM-9B	B-61	B	F		6 M-117	4 M-117	C	*
	600 gal tank	B-61	C	*		6 M-117	6 M-117	C	*
	B-61	B-61	B	F		6 M-117A1		C	*
		BDU-8/B	B	G		4 M-117A1	4 M-117A1	C	*
	Pylon	BDU-8/B	B	G		6 M117A1	6 M-117A1	C	*
	BDU-8/B		B	G		6 M117D		C	*
	BDU-8/B	BDU-8/B	B	G		4 M-117R		C	*
		BDU-12/B	B	F		4 M-117R	4 M-117R	C	*
	Pylon	BDU-12/B	B	F		Pylon	M-118	C	*
	BDU-12/B		B	F		M-118		C	*
	BDU-12/B	BDU-12/B	B	F		M-118	M-118	C	*
		BDU-18/B	B	G		6 MK20 MOD 2		C	*
	Pylon	BDU-18/B	B	G					
	BDU-18/B		B	G		6 MK20 MOD 2	6 MK20 MOD 2	C	*
	BDU-18/B	BDU-18/B	B	G					
		BDU-19/B	B	F		6 MK-36		C	*
	BDU-19/B		B	F		4 MK-36	4 MK-36	C	*
	Pylon	BDU-19/B	B	F		6 MK-36	4 MK-36	C	*
	BDU-19/B	BDU-19/B	B	F		Pylon	4 MK-82	C	*
		BDU-38/B	B	F		Pylon	6 MK-82	C	*
	Pylon	BDU-38/B	B	F			6 MK-82	C	*
	BDU-38/B		B	F		6 MK-82		C	*
		BDU-38/B	B	F		4 MK-82	4 MK-82	C	*
	2 BLU-1C/B	2 BLU-1C/B	C	*		6 MK-82	4 MK-82	C	*
2 BLU-1C/B	2 BLU-1C/B	2 BLU-1C/B	D	*		6 MK-82	6 MK-82	C	*

Figure 6-8. (Sheet 1)

External Stores And Weapon Loading Configurations

★

PYLON STATIONS (1)			INDEX		PYLON STATIONS			INDEX	
2 & 7	3 & 6	4 & 5	Long.	Dir.	2 & 7	3 & 6	4 & 5	Long.	Dir.
	Pylon	4 MK-82S	C	*			★SUU-21/A	B	*
	4 MK-82S		C	*		★ SUU-21/A		B	*
	6 MK-82S		C	*		TDU-10/B & AA-37U-15(4)		A	*
	4 MK-82S	4 MK-82S	C	*			TDU-11/B	B	*
	6 MK-82S	4 MK-82S	C	*		Pylon	TDU-11/B	B	*
		MK-84	C	*		TDU-11/B	TDU-11/B	B	*
	Pylon	MK-84	C	*		TMU-28/B		C	*
	MK-84		C	*	TMU-28/B	TMU-28/B		C	*
	MK-84	MK-84	C	*		600 gal tank		C	*
		SUU-20A/A	B	*	600 gal tank			C	*
	SUU-20A/A		B	*		600 gal tank	600 gal tank	C	*

Notes:

1. No stores carried on stations 1 & 8.
2. For aircraft equipped with a forward ECM pod only or both a forward and aft ECM pod on the fuselage, the directional stability aft allowable position must be shifted six percent forward of the values presented for directional indexes E, F, and G. For an aft ECM pod only on the fuselage, use the directional stability aft allowable position as presented.
3. Pylon refers to only the pylon or the pylon with any rack(s) attached.
4. After TDU-10/B is released observe the aft allowable center-of-gravity limits for AA-37U/15.

*Not carried to critical mach/altitude conditions; therefore directional aft allowable position is not applicable.

Figure 6-8. (Sheet 2)

Find: Aft allowable center-of-gravity position for the following:

a. Takeoff.

b. Low speed, flaps and gear up, 16 degree wing sweep, slats extended, mach less than 0.55.

c. Low speed, flaps and gear up, 26 degree wing sweep, clean aircraft, mach less than 0.55.

d. High speed, 26 degree wing sweep, mach 0.70.

e. High speed (mach 2.20), 72.5 degree wing sweep longitudinal and directional aft allowable position.

f. Landing.

Refer to "External Stores and Weapons Loading Configurations," figure 6-8. Locate the B-43 weapons loadings on pylon stations 4 and 5, and 3 and 6. To determine the correct store index to be used for determination of the longitudinal and directional aft allowable center-of-gravity position, read to the right and note:

a. The aft allowable position for longitudinal stability is depicted by longitudinal index "B".

b. The aft allowable position for directional stability is depicted by directional index "G". (An asterisk in this column indicates no directional aft allowable position applicable for that store loading configuration.)

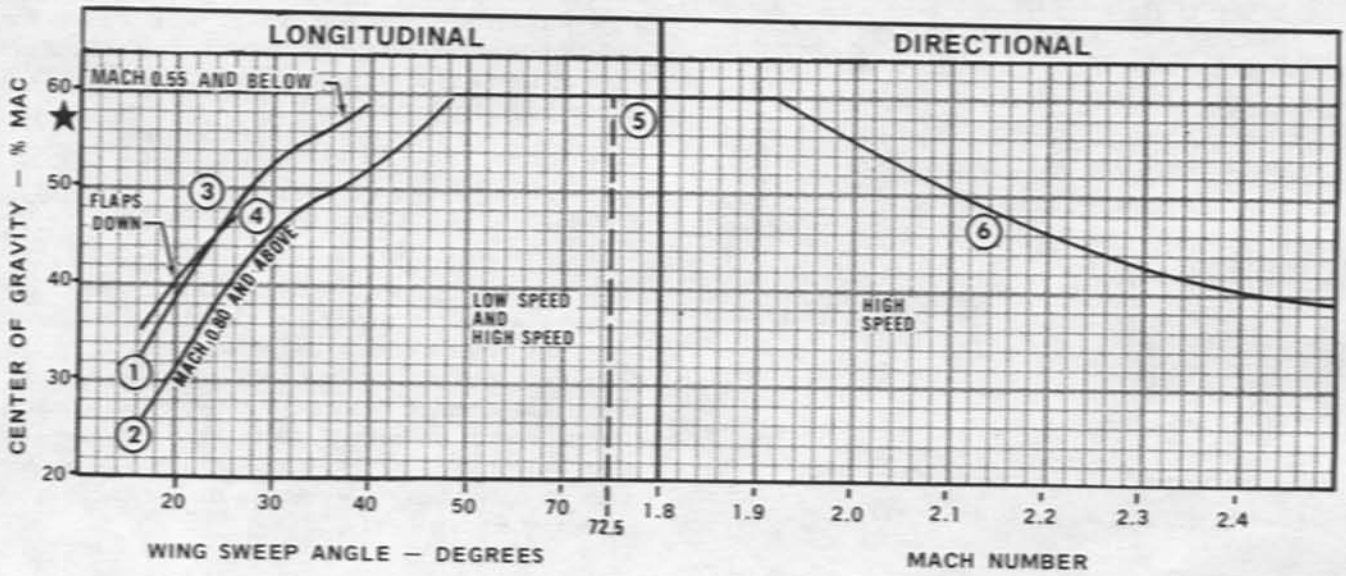
Follow the example lines on figure 6-9, in the appropriate speed regime for the particular configuration specified above to determine the following:

		Aft Allowable Position
a. Takeoff		32.5%
Basic	34.5% ❶	
25° Flap	−1.0%	
B Stores	−1.0%	
	32.5%	
b. Low speed, flaps and gear up, 16 degree wing sweep, slats extended, mach less than 0.55		28.0%
Basic	32.0% ❷	
Slats Extended	−2.0%	
B Stores	−2.0%	
	28.0%	

Aft Allowable Center of Gravity Positions

DATE: 30 MARCH 1973

CONFIGURATION:
NO EXTERNAL STORES



STABILITY INCREMENTS

Note

- Add the sum of all applicable limits below to the limits in the above chart.
- Linear interpolation required between Mach 0.55 and 0.80.

LONGITUDINAL

Flaps Down	
25 degrees flaps	−1.0%
★ A Index stores	0.0%
B Index stores	−1.0%
C Index stores	−2.0%
D Index stores	−3.0%
Flaps Up	
Slats only	−2.0%
A Index stores	0.0%
B Index stores	−2.0%
C Index stores	−3.0%
D Index stores	−6.0%

DIRECTIONAL

Forward ECM Pod	−6.0%

Add the following increments for each 0.1 mach above 1.9.

E Index Stores	0.0%
F Index Stores	−0.5%
G Index Stores	−0.75%

A0000000-E056 D

Figure 6-9.

c. Low speed, flaps and gear up, 26 degrees wing sweep, mach less than 0.55 ..... 46.0%

Basic	48.0% ❺
B Stores	−2.0%
	46.0%

d. High speed, mach 0.70, 26 degree wing sweep 41.8%

*Basic	43.8% ❹
B Stores	−2.0%
	41.8%

*Determine basic by linear interpolation between 0.55 and 0.80 mach.

e. High speed, mach 2.20, 72.5 degree wing sweep 43.75%

(1) Longitudinal

Basic	60.0%
B Stores	−2.0%
	58.0% ❺

(2) Directional

Basic	46.0%
B Stores	−2.25%
	43.75% ❹

Note

The aft allowable position for directional stability is further forward than that for longitudinal stability; therefore it would determine the aft allowable center-of-gravity position for flight in this regime.

f. Landing, 26 degrees wing sweep 46.5%

Basic	47.5%
B Stores	−1.0%
	46.5%

After determining the aft allowable center-of-gravity position, either longitudinal or directional, refer to T.O. 1-1B-40 to determine the gross weight ranges within which the aircraft should be operated to maintain the center-of-gravity limits.

Note

Loadings which result in an aft center-of-gravity in excess of 60 percent MAC can cause the aircraft to tip back when brakes are released with AB power.

For crew module center of gravity limitations, refer to "Center-of-Gravity Limitations", Section V.

CENTER-OF-GRAVITY ENVELOPE.

Figure 6-10 presents typical center-of-gravity envelopes for the aircraft with and without internal stores. Because of the variance in basic weight and center-of-gravity conditions between aircraft, the weight and balance handbook, T.O. 1-1B-40, must be used to determine the actual weight and balance conditions of a specific aircraft. Fuel sequencing (pounds used) for an 8200 pound differential between the forward and aft tanks is identical regardless of stores loading if auto engine feed is selected.

ENGINE STALL CHARACTERISTICS.

Stalls are caused by an aerodynamic disruption of the airflow through the engine resulting in a breakdown of airflow in the engine compressor similar to the disruption in flow encountered during a wing stall. Engine stalls may be classified into two types of stalls: fan stalls and compressor stalls. Fan stalls usually occur when selecting or while in the afterburner range of operation. Compressor stalls are possible at any power setting.

ENGINE OPERATING ENVELOPE.

In order to minimize engine stalls, a mach-altitude engine operating envelope and angle-of-attack envelope for fixed throttle settings are defined in figure 6-11. Engine stalls may occur more frequently at angles-of-attack near those specified in the angle-of-attack envelope or during AB light at altitude. Engine operation at higher mach numbers than those depicted may be accomplished up to the clean aircraft limit, but with increasing probability of engine stall. In order to prevent inlet buzz and/or engine stall, the throttle should not be reduced below MIL power above 1.5 mach. Below 1.5 mach, but above 35,000 feet, the throttle should not be positioned lower than 80 percent RPM (approximately halfway from MIL to IDLE). If inlet duct rumble is encountered when selecting 80 percent RPM, do not retard the throttle further, but advance as necessary to eliminate the inlet duct rumble. Below 1.5 mach and below 35,000 feet, throttle position is unrestricted.

STALL RECOGNITION.

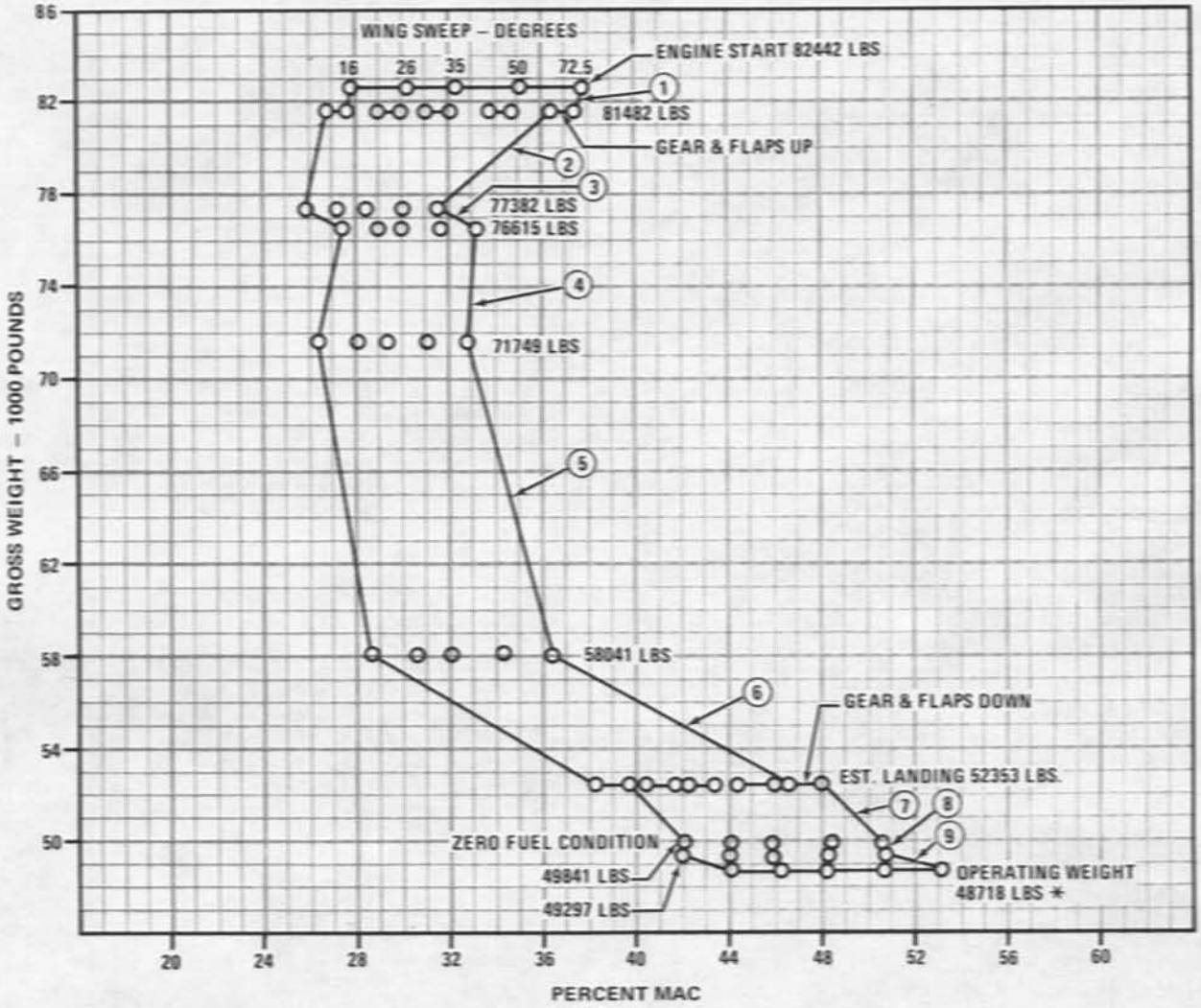
Fan stalls result in an audible "bang" with an almost immediate recovery to military power and in some cases to afterburner power (with the throttle in AB). These stalls occur and recover too quickly to be detected by observing any engine instrument except nozzle position, to determine which engine stalled.

Center of Gravity Envelope (Typical)

DATA BASIS: ESTIMATED
DATE: 6 JUNE 1969

LOADING WITHOUT STORES

FUEL GRADE: JP-4
ENGINES: TF30-P-3



FUEL SEQUENCE (AF/AT) = 8200 LB DIFFERENTIAL	
① WING	–960 (TAXI & TAKEOFF)
② WING	–4100 (WINGS EMPTY)
③ FORWARD	–767 (8200 LB. △ FWD & AFT)
④ FORWARD & AFT	–4866 (A2 EMPTY)
⑤ FORWARD & AFT	–13708 (A1 & A2 EMPTY)
⑥ FORWARD	–5688 (F1 & F2 EMPTY)
⑦ FORWARD (RESERVOIR)	–2512 (RESERVOIR EMPTY)
TOTAL FUEL USED	32601 POUNDS

⑧ EXPENDABLE AMMO (W/B GUN) 544 POUNDS

⑨ NON–EXPENDABLE AMMO (W/B GUN) 577 POUNDS

* FOR OPERATING WEIGHT DETAILS REFER TO "AIRCRAFT WEIGHT," SECTION I.

A0000000–E060

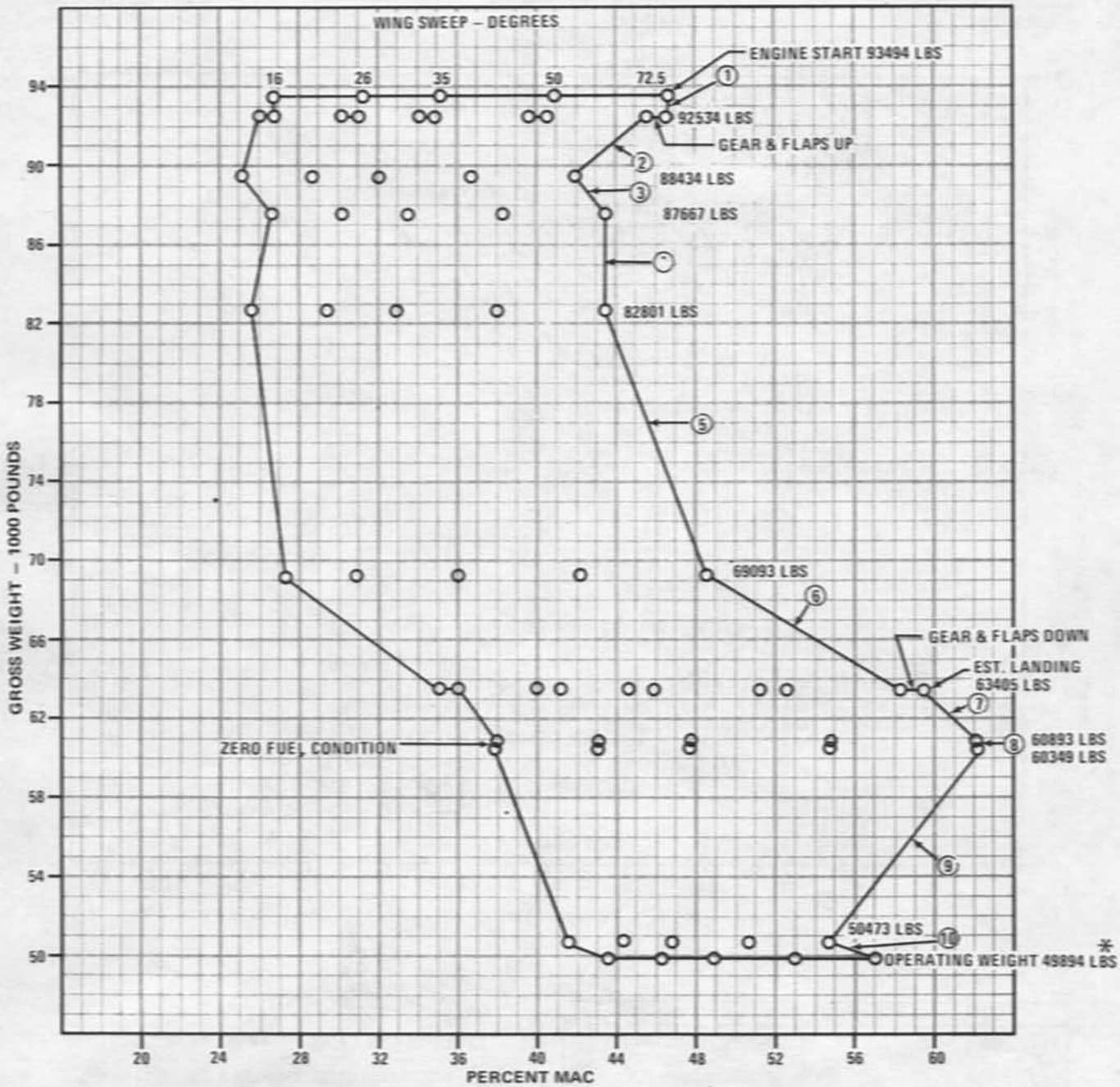
Figure 6-10. (Sheet 1)

Center of Gravity Envelope (Typical)

DATA BASIS: ESTIMATED
DATE: 6 JUNE 1969

LOADING WITH STORES

FUEL GRADE: JP-4
ENGINES: TF30-P-3



FUEL SEQUENCE (AF/AT) = 8200 LB DIFFERENTIAL	
① WING	–960 (TAXI & TAKEOFF)
② WING	–4100 (WINGS EMPTY)
③ FORWARD	–767 (8200 LBS FWD & AFT)
④ FORWARD & AFT	–4866 (A2 EMPTY)
⑤ FORWARD & AFT	–13708 (A1 & A2 EMPTY)
⑥ FORWARD	–5688 (F1 & F2 EMPTY)
⑦ FORWARD (RESERVOIR)	–2512 (RESERVOIR EMPTY)
TOTAL FUEL USED	32601 POUNDS

⑧ EXPENDABLE AMMO (W/B GUN) 544 POUNDS

⑨ M–117 STORES (12) (PYLON STATIONS 3 & 6) 9876 POUNDS

⑩ NON–EXPENDABLE AMMO (W/B GUN) 579 POUNDS

* FOR OPERATING WEIGHT DETAILS REFER TO "AIRCRAFT WEIGHT," SECTION I.

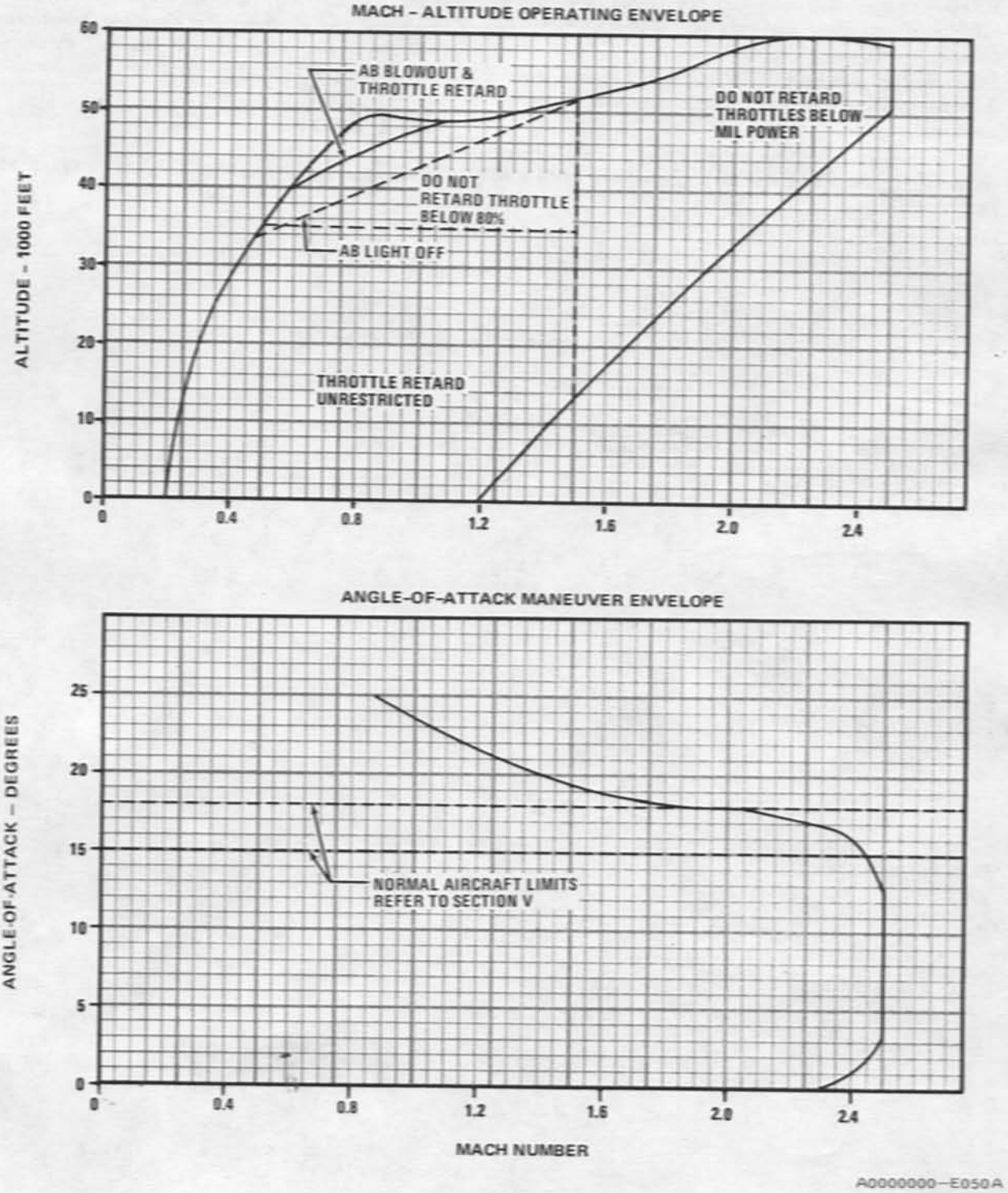
A0000000—E061

Figure 6-10. (Sheet 2)

Engine Operating Envelope

DATE: 24 APRIL 1970

ENGINES: TF30-P-3



A0000000—E050A

Figure 6-11.

The nozzle will end up in a closed position, if recovery is to the military power range, or will be transitioning from closed to open if it recovers to afterburner. Compressor stalls are noted by the audible "bang" and in most cases, at supersonic speeds, are preceded by a period of inlet rumble. The engine instruments react as follows during a compressor stall:

- EPR—quickly drops to 1.0
- RPM—decreases at a moderate rate to below a normal idle speed and then is slowly unstable until stall recovery.
- TIT—flashes to a high TIT followed by an unstable decrease to some base level where it remains unstable until the engine recovers.
- Fuel flow—decreases at a moderate rate towards that required for the RPM and then is unstable until stall recovery.
- Nozzle Position—
  (a) If non-afterburning operation—no change.
  (b) If afterburning operation—closes; sometimes fast, sometimes slow, and sometimes may appear to remain in an intermediate position for a period of time.

The best indicator of a stalled engine is its TIT—it will be significantly and abnormally different from the unstalled engine. The peak-out TIT will depend on airspeed and altitude. For example, the peak-out TIT will be higher at mach 1.2/50,000 feet than at mach 2.0/50,000 feet or at mach 1.2/30,000 feet.

STALL RECOVERY.

For engine stall recovery refer to "Engine Stall", Section III.

This is the last page of Section VI.

SECTION VII

ALL WEATHER OPERATION

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Ice and Rain	7-7
Turbulence and Thunderstorms	7-8
Night Flying	7-9
Cold Weather Procedures	7-10
Hot Weather and Desert Operation	7-11

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

The aircraft is designed to perform operational missions in unfavorable weather conditions. 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. Instrument system coupler mode selector knob—As required.
3. HSI course/heading knob—Set.
4. Attitude indicator—Adjust pitch trim knob to index.
5. Pitot heat—Climatic.

INSTRUMENT TAKEOFF.

An instrument takeoff is accomplished using a combination of visual and instrument references. Procedures are the same as for a normal takeoff. After the brakes are released, use visual references to maintain alignment. As the takeoff progresses, the pilot's crosscheck should transition from outside references to the aircraft instruments. Rate of change is dependent upon how rapidly outside references deteriorate. It is very important that the transition to instrument references be complete before losing outside references. At rotation speed, smoothly rotate the aircraft to increase indicated attitude to 10 degrees above indicated ground static attitude. Crosscheck the vertical velocity indicator and altimeter to insure a positive rate of climb before retracting the gear and flaps/slats.

INSTRUMENT CLIMB.

After liftoff, maintain the 10 degree pitch attitude used for takeoff to obtain a positive increase in both altitude and airspeed and complete desired configuration changes as for a normal takeoff. After establishing climb configuration, control aircraft attitude to maintain a positive increase in airspeed and altitude until attaining desired climb speed. The climb schedule recommended in Appendix I is suitable for instrument flight. A crosscheck of all instruments, particularly the turn indicator, is extremely important during initial climb to verify reliability of all instruments.

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 300 KIAS. Maximum bank angle of 30 degrees is normally used.

JET PENETRATION.

Prior to beginning penetration, check the weather and availability of radar or ILS. If ceiling or visibility is below minimum, make the decision to proceed to an alternate while still at altitude. For maximum range, an idle power descent at 250 KIAS with 26 degree wing sweep and speed brake retracted is recommended. For minimum time in descent, 350 KIAS with 26 degree wing sweep and speed brake extended is recommended. A normal penetration, either TACAN or Enroute, is accomplished at 300 KIAS with 26 degree wing sweep, or forward as required, speed brake extended and approximately 80 percent power. Upon arrival at the initial approach fix (IAF), retard throttles to approximately 80 percent, extend speed brake, lower nose to maintain 300 KIAS, and accomplish the penetration as required. One thousand feet above level-off altitude, retract speed brake, as required, and adjust power as required to maintain desired altitude and an airspeed compatible with aircraft configuration and gross weight (250 KIAS desired). Do not exceed 10 degrees angle of attack during maneuvering flight or aircraft configuration changes. Accomplish the "Landing Pattern" checklist.

INSTRUMENT APPROACHES.

WARNING

High angles-of-attack can occur rapidly with power, bank, and aircraft configuration changes. Therefore, throughout all instrument traffic pattern maneuvering, the angle-of-attack indicator and indexers should be closely monitored and included in the normal instrument crosscheck. Ten degrees angle-of-attack should not be exceeded during pattern maneuvering and aircraft configuration changes.

Note

- Maintain an airspeed compatible with aircraft configuration and gross weight (250 KIAS desired). Accomplish the "Landing Pattern" checklist and maintain a minimum of computed approach speed plus 20 knots (160 KIAS minimum) until rolled out on final.
- Turbulence, gusty winds or other conditions may exist which may induce variations in angle-of-attack or airspeed or will cause excessive sink rates to develop on final approach. The pilot may decrease angle-of-attack to 8 degrees or increase final approach speed 10 knots in such cases to improve aircraft handling characteristics. To avoid undesirable touch down characteristics, this additional airspeed should be dissipated so that an "on-speed" indication exists prior to initiation of flare.

TACAN APPROACH.

A typical Tacan Penetration/Approach is illustrated on figure 7-1. Complete the appropriate checklists prior to the final approach fix. Maintain an airspeed compatible with aircraft configuraton and gross weight (250 KIAS desired). Do not exceed 10 degrees angle-of-attack during maneuvering flight or aircraft configuration changes. Accomplish the "Landing Pattern" checklist. Reduce airspeed to computed approach speed plus 20 knots minimum until aircraft is rolled out on the final approach course. Prior to final approach fix, establish desired angle-of-attack/airspeed. Do not descend below minimum descent altitude (MDA) unless runway is in sight.

Tacan Penetration/Approach (Typical)

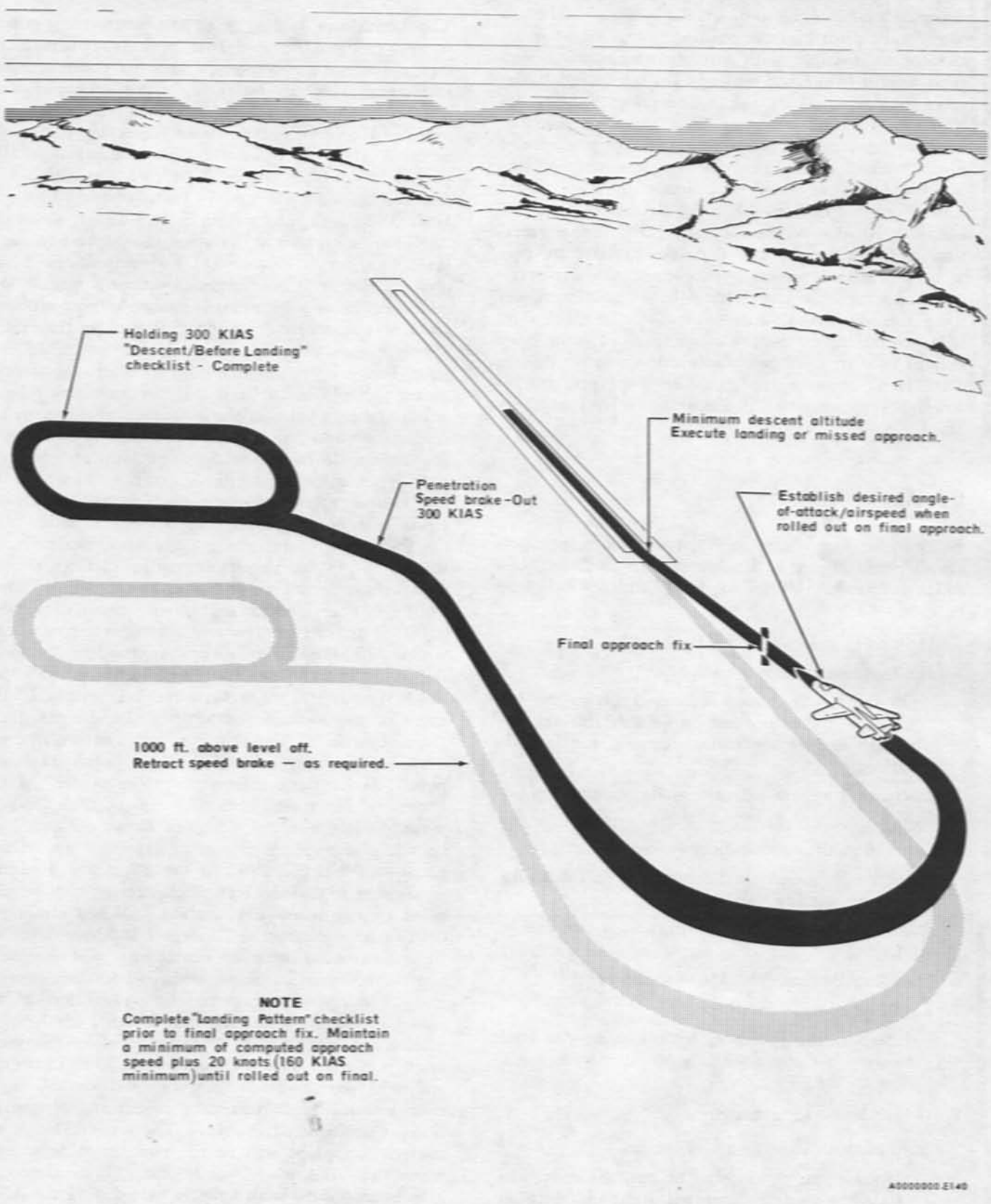

Figure 7-1.

PAR/ASR.

Radar approaches should be flown at 26 degree wing sweep or less (as required by CG for landing) and clean configuration until on the downwind leg. Maintain an airspeed compatible with aircraft configuration and gross weight (250 KIAS desired). Do not exceed 10 degrees angle of attack during maneuvering flight or aircraft configuration changes. After gear and slat/flap extension, reduce airspeed to computed approach speed plus 20 knots minimum. Maintain this airspeed until completion of turn onto final approach. After rolling out on final, establish desired angle-of-attack/airspeed. At the decision height (DH), on a precision approach, make the decision to land or perform the missed approach procedure. On a surveillance approach, do not descend below minimum descent altitude (MDA) unless runway is in sight. The radar altimeter and radar altitude low warning lamp may be used to monitor aircraft altitude in relation to DH/MDA; however, the altimeter is the primary instrument reference for determining arrival at DH/MDA.

ILS APPROACH.

Refer to landing charts in Appendix I for appropriate airspeeds. Figure 7-2 illustrates a typical radar/ILS/AILA approach. ADI and/or LCOS steering bars may be used.

1. ILS power switch—POWER.
2. ILS frequency selector knob—Set.

   Set the ILS frequency selector knob to the frequency of the localizer to be used for the approach and adjust volume control for identification.
3. Instrument system coupler mode selector knob—ILS.
4. Horizontal situation indicator—Set.

   Set the inbound localizer course in the course selector window.
5. Radar altimeter — Set to minimum altitude AGL.
6. "Landing Pattern" checklist—Complete.
7. Localizer course—Intercepted.

   Intercept the localizer, by centering the bank steering bar approximately 10 miles from touchdown.
8. Glide slope—Intercepted.
9. Minimum altitude fly-up—Checked.

   Upon reaching decision height, the radar altitude low warning lamp will light and the pitch steering bar will indicate a fly-up command.

AIRBORNE INSTRUMENT LOW APPROACH (AILA).

The bomb nav system with the instrument system coupler in the AILA mode will provide artificial localizer/glideslope deviation and steering commands for instrument landings on runways not equipped with ground based radio landing aids or as a backup in the event of ILS or GCA radar failure. Steering commands to the synthetic localizer and glideslope beams are displayed on the pitch and bank steering bars of the ADI and LCOS, and course deviation is displayed on the HSI. Glide path information is also displayed on the ADI on the glideslope deviation indicator. The simulated localizer is located 12,000 feet past the touch down point; this will provide steering signals and course deviation which closely resemble those required for ILS. The course deviation displayed on the HSI is a synthetic localizer deviation scaled for ± 2.5 full scale. For accurate glideslope information, the aircraft altitude should be calibrated. If circumstances permit, it is recommended that the altitude calibration be accomplished over the landing runway. The bank steering bar should be followed during localizer intercept only if the aircraft is initially positioned on a 45-degree course intercept line drawn from the aircraft position to a point greater than 5 nautical miles from the touchdown point along a synthetic localizer centerline. When the aircraft is positioned for a normal approach, a 45 degree course intercept will be flown until the lateral deviation becomes equal to a value based on aircraft velocity and a 30 degree bank angle which will allow an asymptotic approach to the localizer beam center. Localizer deviation is provided on the HSI and ADI when the aircraft is within 2.5 degrees of the localizer centerline. The desired glideslope, limited to less than or equal to six degrees, must be manually entered into the bomb nav computers. Glide path information is provided on the glideslope deviation indicator when the glideslope deviation is less than ±0.9 degrees. A two dot deflection on the glideslope deviation indicator on the ADI is equal to 0.7 degrees. When the glideslope deviation diminishes to 0.07 degrees, pitch steering will be supplied by the bomb nav computer. When either the 0.9 degree glideslope or 2.5 degree localizer deviation limit is exceeded the pitch steering bar is stowed until limit conditions are re-established. Localizer steering and glideslope deviation continue regardless of the status of the pitch steering bar. When glideslope deviation becomes equal to 0.07 degrees or less, the bank angle command limit is changed from 30 to 15 degrees and will remain at 15 degrees until an ISC mode change is initiated. Automatic sequencing of destinations is not utilized in AILA. The coordinates of the current steerpoint will be the same as those entered when the AILA mode was selected. Offset aimpoints may be associated with a destination during the AILA mode for precise radar position fixes during letdown.

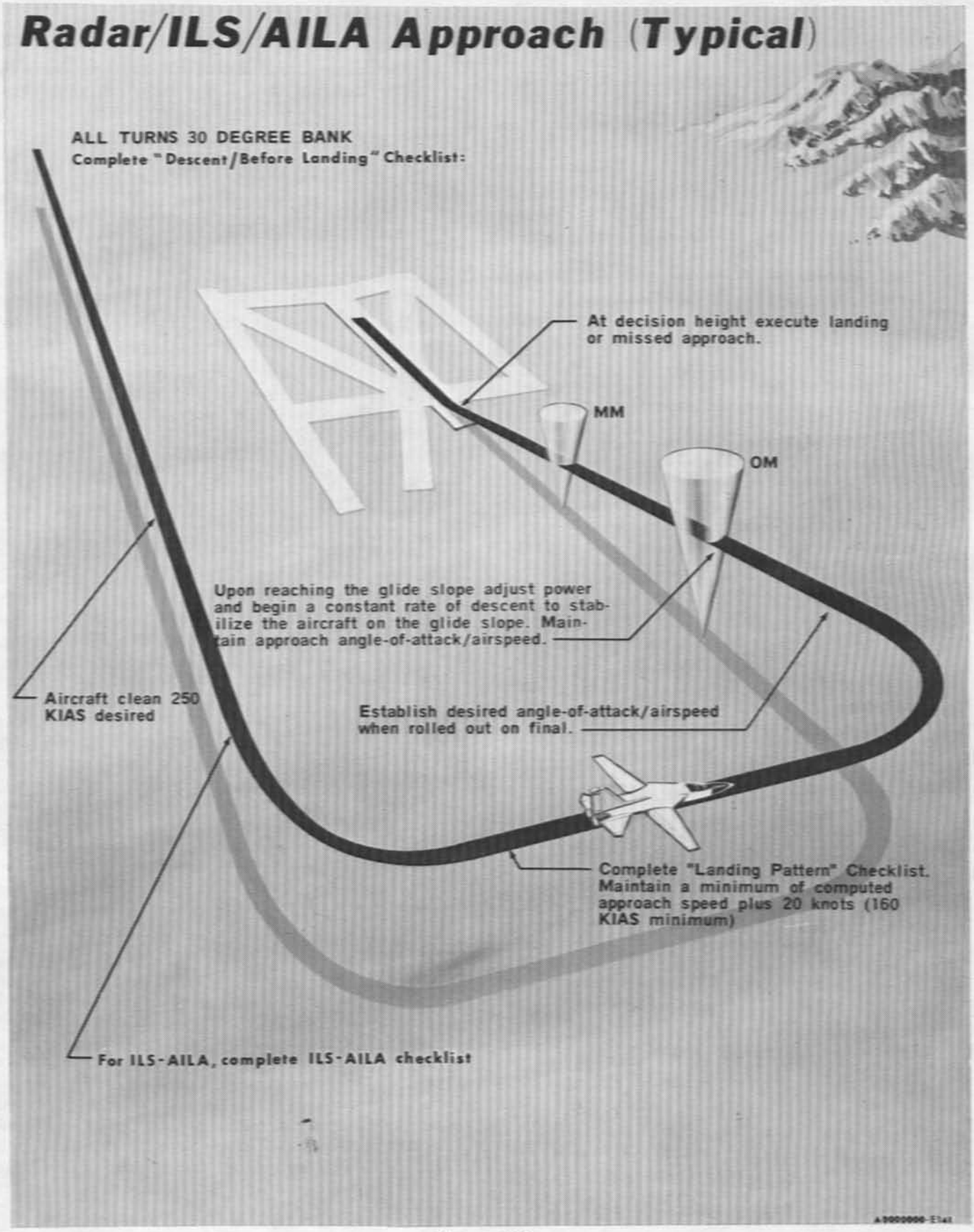
Radar/ILS/AILA Approach (Typical)
ALL TURNS 30 DEGREE BANK
Complete "Descent/Before Landing" Checklist:
At decision height execute landing or missed approach.
MM
OM
Upon reaching the glide slope adjust power and begin a constant rate of descent to stabilize the aircraft on the glide slope. Maintain approach angle-of-attack/airspeed.
Aircraft clean 250 KIAS desired
Establish desired angle-of-attack/airspeed when rolled out on final.
Complete "Landing Pattern" Checklist. Maintain a minimum of computed approach speed plus 20 knots (160 KIAS minimum)
For ILS-AILA, complete ILS-AILA checklist

Figure 7-2.

Minimum altitude must be manually set on the radar altimeter indicator. Figure 7-2 illustrates a typical airborne instrument landing and approach.

1. Bomb nav mode selector knob—SHORT RANGE.
2. Destination position counters—Set.
   Set destination position counters to touchdown coordinates.
3. TARGET fix mode selector button—Depressed.
4. Altitude calibration/alignment—Completed.
   Calibrate altitude over available point closest to the runway with the most accurately known elevation, preferably the landing runway. Set field pressure altitude when using altitude alignment.

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 beyond the desired touchdown point.

5. Fixpoint elevation counter—Set.
   Set fixpoint elevation counter to touchdown point elevation/field pressure altitude.
6. Attack radar mode selector knob—GND AUTO/GND VEL.
   Initial acquisition may be made in GND AUTO or GND VEL. Final approach should be made in GND VEL.
7. Attack radar range selector—Minimum setting.
8. Instrument system coupler mode selector knob—NAV.
9. Aircraft initial position—Established.
10. Glide/Dive angle counter—Set.
    Set the glide/dive angle counter to the approach glide slope.
11. Radar altimeter—On and set.
12. "Landing Pattern" checklist—Complete.
13. Airspeed and altitude—Established.
    Establish airspeed of 160 KIAS and altitude of 1000 feet AGL, or as locally directed.
14. Instrument system coupler mode selector knob—AILA.
15. LCOS mode select knob—COM.
16. Horizontal situation indicator—Set.
    Set course selector window to runway heading.
17. ADI/LCOS bank steering bar—Centered.
    Center the ADI and/or LCOS bank steering bar, and maintain it centered throughout the remainder of the approach.
18. Attack radar range and azimuth cursors—On touchdown point.

WARNING

Cursors should be laid precisely on the touchdown point and should track precisely as approach continues. Correct as required with the tracking handle. If cursor drift cannot be corrected, accomplish go-around.

19. Runway heading—Intercepted.
    Intercept runway heading by maintaining ADI and/or LCOS bank steering bars centered.
20. Glide slope—Intercepted.
    The ADI/LCOS pitch steering bars will come into view at glide slope interception.
21. ADI/LCOS pitch steering bars—Centered.
    Establish descent as required to maintain the pitch steering bars centered.
22. Minimum altitude fly-up—Checked.
    Upon reaching MDA, the radar altitude low warning lamp will light and the pitch steering bar will indicate a fly-up command.

ENROUTE/STRAIGHT-IN APPROACHES.

The "Landing Pattern" checklist will be completed and desired angle-of-attack/airspeed established prior to the final approach fix or glide slope interception. Maintain an airspeed compatible with aircraft configuration and gross weight (250 KIAS desired). Do not exceed 10 degrees angle-of-attack during maneuvering flight conditions or aircraft configuration changes. Accomplish the "Landing Pattern" checklist. After landing configuration is established, maintain a minimum of computed approach speed plus 20 knots (160 KIAS minimum) until on the final approach course, then establish desired angle-of-attack/airspeed.

MISSED APPROACH.

If a missed approach is required, smoothly advance power as necessary to insure an increase in airspeed and altitude. When both are increasing, retract gear and flaps/slats in accordance with normal takeoff procedures. Accomplish "After Takeoff" checks if proceeding to an alternate or "Landing Pattern" checks if another approach is being planned to the same destination.

ICE AND RAIN

The aircraft is equipped with (1) pitot-static, angle-of-attack, angle-of-sideslip, and total temperature probe anti-icing and de-icing, (2) engine anti-icing, and (3) windshield rain removal and defogging provisions. Refer to "Anti-Icing and Defog Systems," Section I. There are no provisions for surface anti-icing. Flight through areas of sustained heavy icing is not recommended. The performance capabilities of the aircraft should be utilized to avoid extreme icing conditions. When moderate to heavy icing is encountered, a change in altitude, course, or airspeed should be made to prevent ice accumulation on the wings and empennage. Engine anti-icing capabilities should be utilized whenever icing conditions are anticipated.

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 aircraft. Either of these conditions tends to reduce total range. Flight can be safely accomplished during light to moderate icing by using normal flight procedures. Rain has little or no appreciable effect on the flight characteristics.

OPERATION IN RAIN OR ICING CONDITIONS.

Icing is possible under the following conditions:

Ground operation.

1. In visible moisture—Temperature between +5 and −15° C.
2. In clear air—Relative humidity above 70 percent and dew point temperature 25 to 35° F.

Inflight operation.

1. In visible moisture—Total temperature of +5° C or below.
2. In clear air—No limitations related to icing.

The aircraft is cleared for flight in icing conditions as follows:

Light Icing Conditions: Aircraft are cleared for all normal service use at all altitudes. The rate of ice accretion in light icing conditions is ordinarily insufficient to make diversionary action necessary. Takeoffs, descents, penetrations, low approaches and landings, when icing conditions exist, should be planned to minimize the occurrence of ice ingestion which may cause engine stalls.

Moderate Icing Conditions: Aircraft operation in moderate icing conditions may be tolerated for short periods of time but some type of evasive action (changing altitude, course, or airspeed) will have to be undertaken to exit the icing condition when ice is observed accumulating on the aircraft structure. Continuous flight in moderate icing conditions during takeoff, approach, or landing should not exceed 5 minutes duration at any airspeed. Cruising flight in moderate icing conditions may be conducted continuously until ice is observed accumulating on the aircraft. The following icing conditions should be avoided:

1. Heavy icing conditions.
2. Icing conditions associated with thunderstorms at airspeeds which produce total temperatures from +5 to −15° C.

GROUND OPERATION.

Operate the aircraft and systems as indicated in the "Cold Weather Procedures" in this section. Rain removal should be used when needed to improve visibility.

CAUTION

Painted areas on runways, taxiways, and ramps are significantly more slippery than non-painted areas. When painted areas are wet, the condition may deteriorate to the extent that the coefficient of friction is negligible. In addition, painted areas sometimes serve as condensation surfaces and it is possible to have wet, frosty, or even icy conditions on these areas when the overall weather condition is dry. When conditions of snow or ice exist, the approach end of the runway is usually more slippery than other areas because of the melting and refreezing of the ice and snow at this point.

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.

WARNING

Engine stalls may be caused by water ingestion if takeoff is attempted with excessive water on the runway.

CRUISE.

Operate the aircraft as necessary to avoid icing conditions whenever possible. When ice is encountered, pitot heat and engine anti-icing should be used. During flight in moderate or heavy rain the windshield selector switch should be placed to the L (left) position only if maximum air flow is required. Selection of the BOTH position will divide the air flow between the two nozzles and reduce rain clearing performance. The BOTH position is recommended only for light rain conditions. 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 for indications of structural deterioration. 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.

CAUTION

To minimize impact damage from rain or sleet, remain below 450 KIAS.

LANDING.

1. Loss of Visibility—Reduced visibility may accompany a power decrease. This is due to a loss of air pressure to the windshield wash/rain removal system, which is directly related to the engine rpm.
   a. Therefore, by selecting L (left) or R (right) rain removal, visibility will be improved.
2. Hydroplane Effect.
   a. Braking—Reduced braking potential may be experienced when landing on wet runways.
   b. Directional control—Nose wheel steering and differential braking may have a reduced effect when landing on wet runways.

TURBULENCE AND THUNDERSTORMS

WARNING

Flight through thunderstorm activity or known severe turbulence is not recommended and should be avoided if at all possible.

The use of attack radar, air to air mode, provides an excellent means of navigation between or around storm cells. If circumstances should force the flight into a zone of severe turbulence or heavy precipitation, establish throttle setting and pitch attitude. An airspeed of 275-300 KIAS and a 26 degree wing sweep is recommended for thunderstorm penetration and for operation in areas of known severe turbulence or heavy precipitation. To reduce vision deterioration from lightning, the high intensity white cockpit lighting should be adjusted to the required level prior to penetration. If flight into an area of heavy precipitation cannot be avoided, the tolerance of the engines to water ingestion can be increased by activating the bleed air systems for the engine/inlet anti-icing system and the windshield rain removal system. Prepare the engines for penetration of heavy precipitation as follows:

1. Engine/inlet anti-icing switch—Man.
2. Windsheld selector switch—Both.
3. Rain removal switch—Rain remove.

Note

- Optimum engine tolerance to water ingestion will be assured by an EPR setting between 1.8 and 1.5. If unusually heavy precipitation is encountered, the airstart button may be held depressed to provide continuous ignition for as long as is considered necessary. After the button is released, ignition will continue to be provided for approximately 55 seconds. If an engine stall or flameout occurs, use the stall recovery and/or airstart procedures contained in Section III.

- When using terrain following radar, the back scatter from drizzle or rain and other forms of precipitation will often be visible on the scope. If the precipitation is so heavy that the operator cannot determine visually where the terrain ends and the precipitation begins, the automatic signal detection circuitry will also be incapable of this discrimination and a climb command will result.
- 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, Section II, when cold weather operation is necessary.

BEFORE ENTERING AIRCRAFT.

WARNING

- All accumulated ice and snow must be removed from the aircraft before flight is attempted. For complete ice and/or snow removal procedures, refer to T.O. 1F-111E-2-1. 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 surfaces or other critical areas where refreezing may cause damage or binding.
- After ground cold soak at minus 50° F, heat the rocket motor compartment with warm air, between 80 to 120° F, for 30 minutes prior to takeoff.

CAUTION

- To avoid damage to aircraft surfaces, do not permit ice to be chipped or scraped away.
- Ensure that engines are free to rotate and that engine lower section moisture is thawed out per T.O. 1F-111E-2-1.

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 aircraft carefully for fuel and hydraulic leaks caused by the contraction of fittings or by shrinkage of packings. Inspect areas behind the aircraft 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 Limitations," Section V.

BEFORE TAXIING.

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 aircraft to insure safe stopping distance and to prevent icing of aircraft surfaces by melted snow and ice in the jet blast of preceding aircraft. Insure that all instruments have warmed sufficiently to insure operation. Check for sluggish instruments during taxiing.

CAUTION

Painted areas on runways, taxiways and ramps are significantly more slippery than 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 runways 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.

Follow normal procedures.

DESCENT.

Follow normal procedures.

LANDING.

Follow normal procedures.

ENGINE SHUTDOWN.

Follow normal procedures.

BEFORE LEAVING AIRCRAFT.

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 aircraft 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 aircraft is not in use.

CAUTION

Do not attempt takeoff or engine operation in a sand storm or dust storm if avoidable.

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 aircraft 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 checks should be 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

High surface temperature conditions often result in unexpected gusts and wind shears near the ground.

ENGINE SHUTDOWN.

Follow normal procedures.

CAUTION

To minimize canopy and windshield damage, consideration should be given to parking the aircraft tail into the wind.

BEFORE LEAVING THE AIRCRAFT.

Leave the canopy open, if the aircraft is to be exposed to direct sunlight at ambient temperatures above 100 degrees Fahrenheit. This will prevent excessive heat buildup within the cockpit. Close the canopy in the evening and any time there is a possibility of blowing sand or dust.

This is the last page of Section VII.

Appendix I
PERFORMANCE DATA

Performance Data is contained in Appendix I of Supplement T.O. 1F-111E-1-1.

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