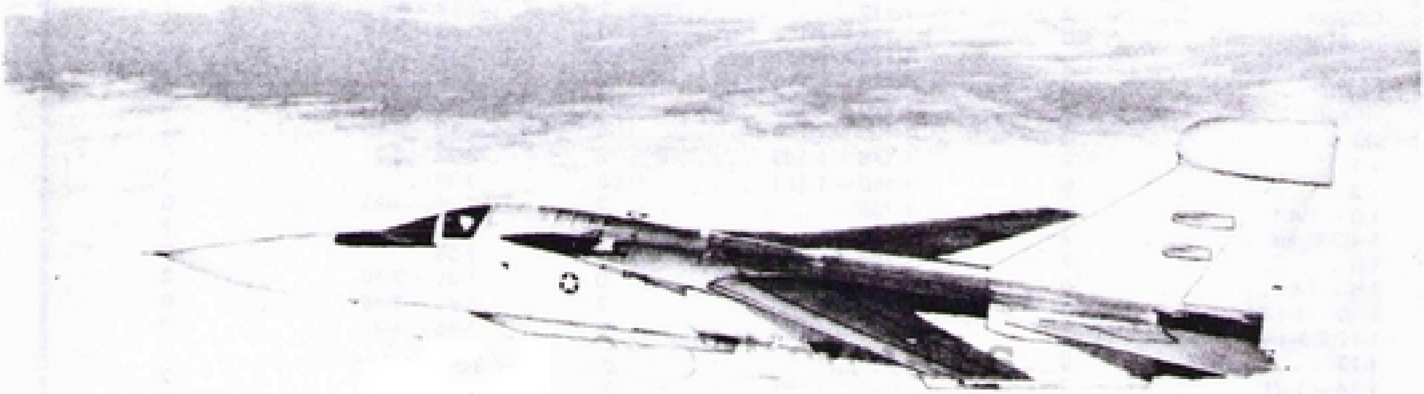
T.O. 1F-111(E)A-1

# FLIGHT MANUAL

USAF SERIES AIRCRAFT

# EF-111A

Contracts: F33657-79-C-0068
F33657-82-C-0165



THIS MANUAL IS INCOMPLETE WITHOUT T.O. 1F-111(E)A-1-1, T.O. 1F-111(E)A-1-2, and T.O. 1F-111(E)A-1-3

This change incorporates Safety Supplements T.O. 1F-111(E)A-1SS-2, -3, -4, and -5.

REFER TO TECHNICAL ORDER INDEX T.O. 0-1-1-4 AND ITS SUPPLEMENTS FOR CURRENT STATUS OF FLIGHT MANUALS, SAFETY SUPPLEMENTS, OPERATIONAL SUPPLEMENTS, AND FLIGHT CREW CHECKLISTS.

**COMMANDERS ARE RESPONSIBLE FOR BRINGING THIS PUBLICATION TO THE ATTENTION OF ALL AIR FORCE PERSONNEL CLEARED FOR OPERATION OF SUBJECT AIRCRAFT**

Published under authority of the Secretary of the Air Force

**16 JANUARY 1984**

CHANGE 2 — 18 OCTOBER 1985

# LIST OF EFFECTIVE PAGES

**INSERT LATEST CHANGED PAGES, DESTROY SUPERSEDED PAGES.**

NOTE: The portion of the text affected by the change is indicated by an "R" in the outer margins of the page. Changes to illustrations are indicated by a miniature star ( ★ ).

Dates of issue for original and changed pages are:

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| 1-95 – 1-96 | 2 | 3-2.2 Blank | 2 | 7-10.2 Blank | 2 |
| 1-97 – 1-106 | 0 | 3-3 – 3-4 | 2 | 7-11 | 2 |

#Zero in this column indicates an original page.

# STATUS PAGE

This page contains a list of the affected Aircrew Flight Manual and related Supplements and Checklists current on the date of this publication. Changes or revisions in production are shown in parentheses.

| AIRCREW FLIGHT MANUAL | DATE | CHANGE NO. AND DATE |
|---|---|---|
| TO 1F-111(E)A-1 | 16 Jan 84 | 2 - 18 Oct 85 |
| TO 1F-111(E)A-1-1 | 30 Aug 82 | 1 - 18 Oct 85 |
| TO 1F-111(E)A-1-2 | 20 Apr 84 | 1 - 18 Oct 85 |
| TO 1F-111(E)A-1-3 | 20 Apr 84 | 1 - 18 Oct 85 |

| FLIGHT CREW CHECKLISTS | DATE | CHANGE NO. AND DATE |
|---|---|---|
| TO 1F-111(E)A-1CL-1 | 16 Jan 84 | 2-18 Oct 85 |

| SAFETY AND OPERATIONAL SUPPLEMENTS | DATE | SUBJECT |
|---|---|---|

| SUPERSEDED/RESCINDED SUPPLEMENTS | DATE | DISPOSITION |
|---|---|---|
| TO 1F-111(E)A-1SS-2 | 18 Apr 85 | Incorporated in Change 2 |
| TO 1F-111(E)A-1SS-3 | 23 May 85 | Incorporated in Change 2 |
| TO 1F-111(E)A-1SS-4 | 29 Jul 85 | Incorporated in Change 2 |
| TO 1F-111(E)A-1SS-5 | 21 Oct 85 | Incorporated in Change 2 |

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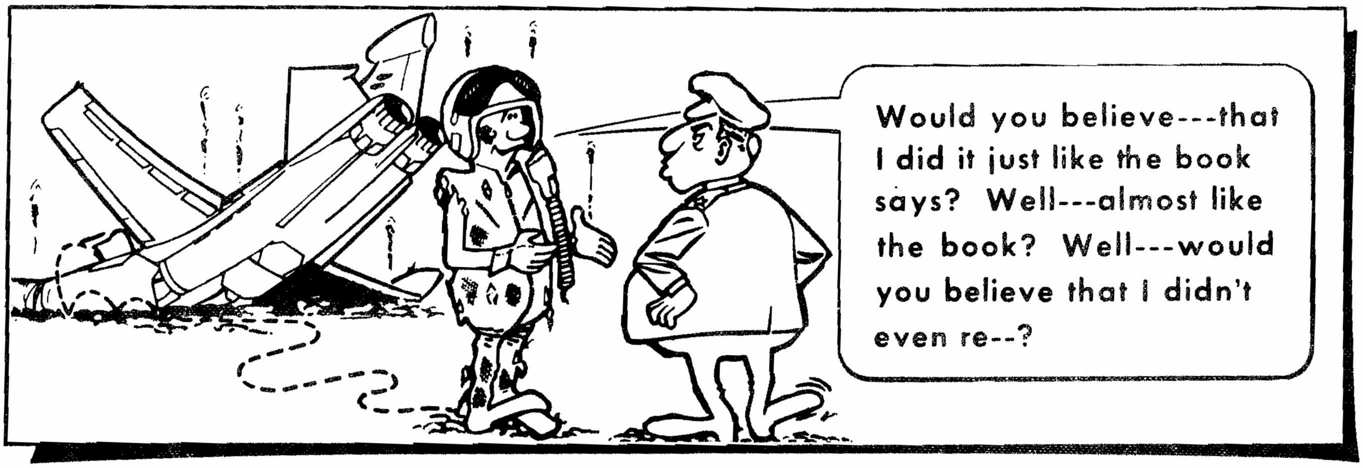



## SCOPE

This manual contains the necessary information for safe and efficient operation of your aircraft. These instructions provide you with a general knowledge of the aircraft and its characteristics and specific normal and emergency operating procedures. Your experience is recognized; therefore, basic flight principles are avoided. Instructions in this manual are prepared to be understandable by the least experienced crew that can be expected to operate the aircraft. This manual provides the best possible operating instructions under most circumstances, but it is not a substitute for sound judgement. 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 for a listing of all current flight manuals, safety supplements, operational supplements, and checklists. Also, check the flight manual title page, the title block of each safety and operational supplement, and all status pages attached to formal safety and operational supplements. Clear up all discrepancies before flight.

## 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 T.O. 1F-111(E)A-1-1.

## SAFETY SUPPLEMENTS

Information involving safety will be promptly forwarded to you in a safety supplement. Urgent information is published in interim safety supplements and transmitted by teletype. Formal supplements are mailed. The supplement title block and status page (published with formal supplements only) should be checked to determine the supplement's effect on the manual and other outstanding supplements.

## 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.

## CHECKLIST

The flight manual contains itemized procedures with necessary amplifications. The checklist contains itemized procedures without the amplification. Primary line items in the flight manual and checklist are identical. If a formal safety or operational supplement affects your checklist, the affected checklist page will be attached to the supplement. Cut it out and insert it in your checklist but never discard the replaced checklist page in case the supplement is rescinded and the page is needed.

## HOW TO GET PERSONAL COPIES

Each flight crewmember is entitled to personal copies of the flight manual, safety supplements, operational supplements, and a checklist. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your publication distribution officer - it is his job to fulfill your T.O. requests. Basically, you must order the required quantities on the appropriate Numerical Index and Requirement Table (NIRT). T.O. 00-5-1 and T.O. 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver the publications to the flight crews immediately upon receipt.

## FLIGHT MANUAL BINDERS

Looseleaf binders and sectionalized tabs are available for use with your manual. They are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part I). 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 could result in personal injury or loss of life if not carefully followed.

**CAUTION**

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

**Note**

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

## USE OF WORDS SHALL, WILL, SHOULD, AND MAY

The words "shall" or "will" are to be used to indicate a mandatory requirement. The word "should" is to be used to indicate a nonmandatory desire or preferred method of accomplishment. The word "may" is used to indicate an acceptable or suggested means of accomplishment.

## 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. 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:

Sacramento Air Logistics Center
McClellan AFB, California 95652
Attn. MMSRB

## SYSTEM LIMITS AND TOLERANCES

In some cases, the limits and tolerances presented in the Flight Manual are not precisely identical to those presented in the system maintenance manuals. The numerical values in the Flight Manual are to be used as operating guides by aircrews.

## GLOSSARY

| | |
|---|---|
| AFC | Automatic Frequency Control |
| AFRS | Auxiliary Flight Reference System |
| AI | Airborne Intercept |
| AILA | Airborne Instrument Low Approach |
| ALTM | Altimeter |
| ALT REF | Altitude Reference |
| AMI | Airspeed Mach Indicator |
| AOA | Angle-of-Attack |
| ATT GYRO | Attitude Gyro |
| AUX ATT | Auxiliary Attitude |

| | |
|---|---|
| AUX NAV | Auxiliary Navigation |
| AVVI | Altitude-Vertical Velocity Indicator |
| AYC | Adverse Yaw Compensation |
| BDHI | Bearing Distance Heading Indicator |
| CADC | Central Air Data Computer |
| CCM | Counter-Counter Measures |
| CIR | Circular |
| CMDS | Countermeasures Dispenser Set |
| CMRS | 'Countermeasures Receiver Set |
| COMP | Compass |
| CRS SEL NAV | Course Select Navigation |
| CVDS | Combustable Vapor Detection System |
| DDI | Digital Display Indicator |
| DEST | Destination |
| DISP | Dispenser |
| FCDS | Flight Control Disconnect Switch |
| FDC | Flight Director Computer |
| FLSC | Flexible Linear Shaped Charge |
| FRL | Fuselage Reference Line |
| FTC | Fast Time Constant |
| GCU | Generator Control Unit |
| GND/GRD | Ground |
| GND MAN | Ground Manual |
| GND VEL | Ground Velocity |
| IMC | Instrument Meteorological Conditions |
| I/P | Identification of Position |
| IR | Infrared |
| IRT | Infrared Threat |
| IRU | Inertial Reference Unit |
| ISC | Instrument System Coupler |
| JETT | Jettison |
| JSS | Jamming Subsystem |
| LARA | Low Altitude Radar Altimeter |
| LAM | Low Altitude Monitor |
| LCCS | Landing Configuration Caution System |
| LSB | Lower Side Band |
| MA | Missile Alert |
| MAC | Mean Aerodynamic Chord |
| MAN CRS | Manual Course |
| MAN FIX | Manual Fix |
| MAN HDG | Manual Heading |
| MFC | Manual Frequency Control |
| ML | Missile Launch |
| MLR | Missile Launch Response |
| MSMA | Maximum Safe Mach Assembly |
| MRT | Modulator-Receiver Transmitter |
| MUX | Multiplex |
| NC | Navigation Computer |
| NRS | Navigation Radar System |
| NWS/AR | Nosewheel Steering/Air Refueling |
| PP/PRES POS | Present Position |
| PPI | Plan Position Indicator |
| RAD | Radiation |
| REC | Receive |
| Rs | Slant Range |
| SCAI | Self-Contained Attitude Indicator |
| SCP | Set Clearance Plane |
| SIF | Selective Identification Frequency |
| SIS | Stall Inhibitor System |
| SLC | Side Lobe Cancellation |
| SMDC | Shielded Mild Detonating Cord |
| SP | Stabilized Platform |
| SPC | Special Purpose Chaff |
| SPS | Self-Protection System |
| STAB AUG | Stability Augmentation |
| STC | Sensitivity Time Control |
| TACAN | Tactical Air Navigation |
| TBC | Trackbreaker Chaff |
| TFR | Terrain Following Radar |
| TIT | Turbine Inlet Temperature |
| TOD | Time of Day |

| | |
|---|---|
| TR | Transformer Rectifier |
| T/R | Transmit/Receive |
| TTI | Total Temperature Indicator |
| TTS | Thermal Transport System |
| TTWS | Terminal Threat Warning System |
| TWS | Track While Scan |
| USB | Upper Sideband |
| VMC | Visual Meteorological Conditions |
| WOD | Word of Day |

## DEFINITIONS:

### ASYMMETRICAL LOADING

Weapon/tank load on any pylon that 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.

### TERMS

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

AC - Tasks accomplished by the left seat crew member.

EWO - Tasks accomplished by the right seat crew member.

BOTH - Tasks accomplished by one crew member and verified by the other.

COMMAND RESPONSE - Those procedures in which one crew member will state the first portion of the checklist statement and the other 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.

## AIRCRAFT DESIGNATION NUMBER AND SERIAL NUMBER CROSS-REFERENCE

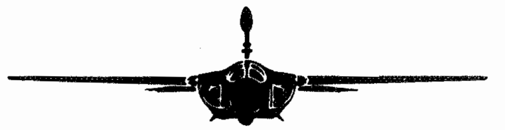
| Designation | Serial No. | Designation | Serial No. |
|---|---|---|---|
| M1 | 66–049 | A22 | 67–041 |
| M2 | 66–041 | A23 | 66–056 |
| A3 | 66–051 | A24 | 66–033 |
| A4 | 66–031 | A25 | 66–039 |
| A5 | 66–020 | A26 | 67–042 |
| A6 | 66–019 | A27 | 66–038 |
| A7 | 66–021 | A28 | 66–028 |
| A8 | 66–027 | A29 | 66–048 |
| A9 | 66–013 | A30 | 66–050 |
| A10 | 66–016 | A31 | 67–037 |
| A11 | 66–018 | A32 | 67–052 |
| A12 | 66–014 | A33 | 66–036 |
| A13 | 66–015 | A34 | 67–048 |
| A14 | 66–044 | A35 | 67–033 |
| A15 | 66–047 | A36 | 66–026 |
| A16 | 67–039 | A37 | 66–055 |
| A17 | 66–023 | A38 | 67–034 |
| A18 | 66–046 | A39 | 67–035 |
| A19 | 67–038 | A40 | 67–032 |
| A20 | 66–035 | A41 | 66–057 |
| A21 | 66–037 | A42 | 66–030 |

| Serial No. | Designation | Serial No. | Designation |
|---|---|---|---|
| 66–013 | A9 | 66–044 | A14 |
| 66–014 | A12 | 66–046 | A18 |
| 66–015 | A13 | 66–047 | A15 |
| 66–016 | A10 | 66–048 | A29 |
| 66–018 | A11 | 66–049 | M1 |
| 66–019 | A6 | 66–050 | A30 |
| 66–020 | A5 | 66–051 | A3 |
| 66–021 | A7 | 66–055 | A37 |
| 66–023 | A17 | 66–056 | A23 |
| 66–026 | A36 | 66–057 | A41 |
| 66–027 | A8 | 67–032 | A40 |
| 66–028 | A28 | 67–033 | A35 |
| 66–030 | A42 | 67–034 | A38 |
| 66–031 | A4 | 67–035 | A39 |
| 66–033 | A24 | 67–037 | A31 |
| 66–035 | A20 | 67–038 | A19 |
| 66–036 | A33 | 67–039 | A16 |
| 66–037 | A21 | 67–041 | A22 |
| 66–038 | A27 | 67–042 | A26 |
| 66–039 | A25 | 67–048 | A34 |
| 66–041 | M2 | 67–052 | A32 |

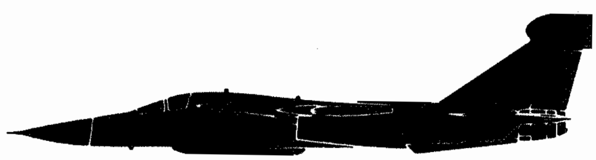
## AIRCRAFT RETROFIT TECHNICAL ORDER INFORMATION

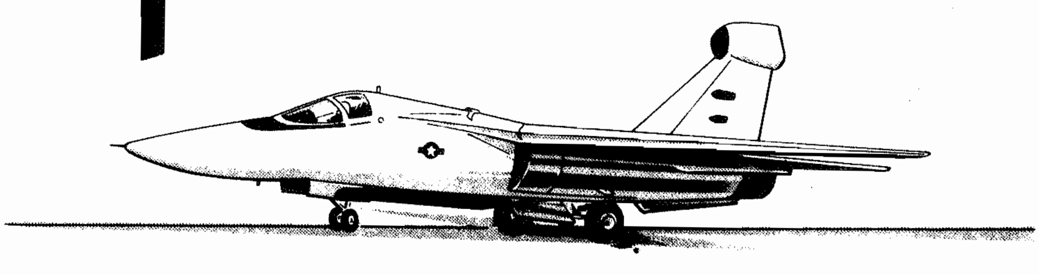
This list includes the applicable TCTO numbers that have been issued up to the date of this publication. Those issued after that date will appear in the next Change/Revision. This is not a complete TCTO listing. Refer to the Basic Index (T.O. 0-1-1-4) for the complete listing of TCTO's which are applicable to these aircraft.

| T.O. No. | Short Title | System/Equipment Affected | Production Effectivity | Retrofit Effectivity |
|---|---|---|---|---|
| 1F-111-1401 | Engine Fire Pushbutton Operation | Engine Fire Detection and Extinguishing System, Section I; Engine Fire Procedures, Section III | None | M1, M2, A3 and On |
| 1F-111(E)A-501 | ILS Control Panel Relocation | Communications Equipment and ILS, Section I; Normal Procedures, Section II | M2, A8 and On | M1, A3 thru A7 |
| 1F-111(E)A-556 | Addition of Blow-In Door for Improved IDG Ground Cooling | Alternating Current Power Supply System, Section I; Normal Procedures, Section II | None | M1, M2, A3 and On |



# The EF-111A





# SECTION I

# DESCRIPTION & OPERATION

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## THE AIRCRAFT

The EF-111A is a two-place (side-by-side), long-range electronic countermeasures aircraft modified by Grumman Aerospace Corporation. It is designed for all-weather supersonic operation at both low and high altitudes. The aircraft mission is to support strike aircraft and ground troops by negating or degrading the effectiveness of enemy electronic weapon systems. An automatic low-altitude terrain-following system enhances penetration capability. Power is provided by two TF-30 axial-flow, dual-compressor turbofan engines 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 from 16 to 72.5 degrees. 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 for 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, which ensures symmetrical main gear operation. The MXU-648 travel pod may be carried on the pivoting pylons. Installation of jettisonable stores is not authorized. A fin fairing at the top of the vertical stabilizer, and a movable equipment pallet in the weapons bay, contain avionic equipment. The fuel system incorporates both in-flight and single-point ground refueling capabilities and gravity refueling capability through filler caps in the top of the wing and fuselage. (See figure FO-1 for aircraft general arrangement and figure FO-2 for crew station general arrangement.)

## AIRCRAFT DIMENSIONS

Length (overall including pitot static boom) - 76 feet, 4.6 inches

Wing span (wings swept) - 31 feet, 11.8 inches

Wing span (wings extended) - 63 feet

Height (to top of vertical tail) - 20 feet, 0.05 inch

## AIRCRAFT WEIGHT

The aircraft operating weight is approximately 56,835 pounds. This weight includes two crewmembers (430 pounds), engine oil (60 pounds), unusable fuel (292 pounds), cooling water (400 pounds), and oxygen (25 pounds). The operating weight also includes ten transmitters and five exciters.

**Note**

For specific aircraft weight, refer to the associated handbook of Weight and Balance Data, T.O. 1-1B-40.

## ENGINES

The aircraft is powered by two interchangeable Pratt and Whitney TF-30 sixteen-stage axial flow turbofan engines equipped with afterburners. 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. An automatically controlled, movable spike is used in each inlet duct to control airflow to the engines. Additional engine inlet air is provided during ground operation and at low airspeeds through openings in the outboard side of each nacelle when the translating cowls are in the extended position. 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 inital 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 ($N_1$) of which three stages are the fan, and 7 stages of the high pressure compressor ($N_2$). 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.
- Translating cowl anti-icing.
- Windshield rain removal.
- Throttle boost.
- Canopy and wing seals.
- Starting opposite engine.
- On the ground engine nacelle ventilation and hydraulic oil, engine oil and constant-speed drive oil cooling.

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 FO-6) 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 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 $N_2$ speed, compressor inlet pressure, compressor inlet temperature, and Mach number which is provided from the central air data computer.

**Note**

> The effect of a CADC Mach failure on the fuel control unit can occur only when the landing gear 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 and may be detected by the compressor bleed valve position indicator not reading NONE at or above military power.

The metering system selects the rate of fuel flow to be supplied to the engine in response to throttle setting. However, metering sections are regulated by the fuel control computing system which monitors the various engine operating parameters. Fuel enters the fuel control through a filter that is provided with a spring-loaded bypass. Fuel metering is accomplished by maintaining a constant pressure across a variable valve area which is controlled by the computing system. The constant pressure is maintained by means of a pressure regulating bypass valve. This valve consists of a servo-operated valve and a spring-loaded valve. Normally, the servo maintains constant valve regulation; but in the event of servo malfunction, the spring valve alone will provide adequate regulation. Deviations from the desired metering pressure are sensed in the valve regulating unit which varies the bypass flow area, thereby restoring the desired pressure by returning excess fuel to the pump inlet.

## ENGINE AFTERBURNERS

The afterburner (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 spring-loaded 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 the 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 FO–6) consists of the following major components: an exhaust nozzle pump, an afterburner fuel pump, an afterburner fuel control unit with integral exhaust nozzle control, an afterburner fuel/oil cooler 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 turbine 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 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 spike unlocks and moves fully aft (approximately 17.5 in.) at 0.44 local mach. Between 0.44 and 0.82 local mach, the spike remains aft in the fully contracted position. Beyond 0.82 local mach, the spike position and diameter are scheduled by the spike controller, based on variation in local mach and diffuser exit mach pressure ratios as sensed by the spike control unit. Signals from the control unit operate hydraulic actuators, powered by the utility hydraulic system, to position the spike fore and aft 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. Use of the emergency pneumatic system forces total reliance on the mechanical locking system. If the mechanical locks break or fail to engage, spike blossoming will occur resulting in engine stalls and decreased performance.

**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 TRANSLATING COWLS

During ground operation and low speed flight, an additional amount of air is required to prevent possible compressor stalls when power settings of over 75 percent are used. This additional air is provided by extension of translating cowls which form the leading edge of each inlet duct. The translating cowl system consists of a hydraulically actuated (utility system) movable cowl on each engine inlet, two flip-flop type cowl position indicators, an amber cowl caution lamp, a red cowl warning lamp, individual switches for each cowl for automatic operation or manual opening, and an additonal switch for operation of a pneumatic back-up system to open both cowls in event of hydraulic system failure. During normal operation the cowls are automatically controlled as a function of speed and altitude.

**Note**

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

## ENGINE VARIABLE EXHAUST NOZZLES

The variable nozzle system incrementally opens and closes the engine exhaust nozzle for afterburner modulation. The control is a hydromechanical computing device that determines and sets the nozzle area required to maintain a desired turbine pressure ratio during afterburner operation. The nozzle position is scheduled by the throttle setting and governed by turbine pressure ratio. The nozzle is closed for all ranges of nonafterburner operation except for ground engine idle at which time it is positioned fully open for minimum thrust. The nozzle closes when the corresponding throttle is advanced 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 engine ignition system provides an ignition source during ground and air starts. Each engine has a dual main ignition system and an automatic restart switch. The system is capable of providing ignition for ground starting or windmill starts during all flight conditions. Advancing the throttle over the OFF ramp provides ignition when the engine start switch is in PNEU or CARTRIDGE. Ignition is cut off when the ground start switch or throttle is in OFF. In the event of a combustion chamber flameout, automatic ignition will be provided to the failed engine, or the airstart button can be depressed to provide ignition to both engines.

## ENGINE STARTING SYSTEM

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. 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 cartridge 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 after the second engine is started, shutting off pneumatic pressure. This will occur at 38 to 41 percent on the left engine and at 45 to 48 percent on the right engine. 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.

## NACELLE VENT/EJECTOR SYSTEM

The nacelle vent consists of an ejector ring, located around the engine afterburner, at the aft end of the nacelle. When the engine compressor bleed air is ejected from the outlet nozzles of the ejector ring, a low pressure area is created between the outside of the engine and the inside of the nacelle. Outside air then enters from the boundary layer entrance, from the nacelle fire access doors, and after T.O. 1F-111(E)A-556, from the IDG supplementary air-oil cooler blow-in door. This air, moving aft through the nacelle cavity around the engine, moves any vapor out through the aft end of the nacelle, thereby reducing the probability of a nacelle fire during ground operation. The engine/IDG air-oil cooler ejector consists of an ejector manifold with a series of ejector nozzles downstream of the engine/IDG air-oil cooler. When engine compressor bleed air is ejected from the ejector nozzles, a low pressure area is created behind the core of the air-oil cooler, resulting in the flow of cooling air through the core. Since nacelle vent ejection and engine/IDG air-oil cooler ejection are not required in flight, the main landing gear safety switch locks out the system when the aircraft weight is not on the main landing gear wheels. With the aircraft weight on the main landing gear wheels, the ejector system is automatically in operation except during engine-start operations.

## ENGINE CONTROLS AND INDICATORS

### Throttles

The throttles (6, figure 1-1) provide thrust setting adjustment for the respective engines. Throttle friction is controlled by means of the friction lever located adjacent to the right throttle. 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, and 10 to 40 pounds in the event pneumatic boost is lost, depending on the friction lever position. The throttles have positions marked OFF, IDLE, MIL, and MAX AB. The throttles must be raised to go into or from the OFF position. When the 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 will automatically fire the left engine starter cartridge. If the ground start switch is in the PNEU position, lifting either throttle 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 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 1-1 for a detail of both the MIL and minimum AB throttle positions. The right throttle includes a microphone switch and a speed brake switch.

### Engine Ground Start Switch

The engine ground start switch (3, figure 1-2) marked PNEU, OFF and CARTRIDGE is solenoid held in the PNEU and CARTRIDGE position and 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 cut-out 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.

**Note**

The engine ground start switch disables auto- R
matic operation of the emergency generator R
during engine start. If the switch comes out of R
CARTRIDGE or PNEU during an engine start R
with no external power connected, hydraulic R
demands of the emergency generator may R
cause rpm hangup. R

### Airstart Button

The airstart pushbutton (5, figure 1-1) marked AIR START provides ignition for air starting the engines. Ignition is provided to both engines when the button is depressed, and continues for approximately 55 seconds after the airstart button is released.

### Ground Ignition Cutoff Switch

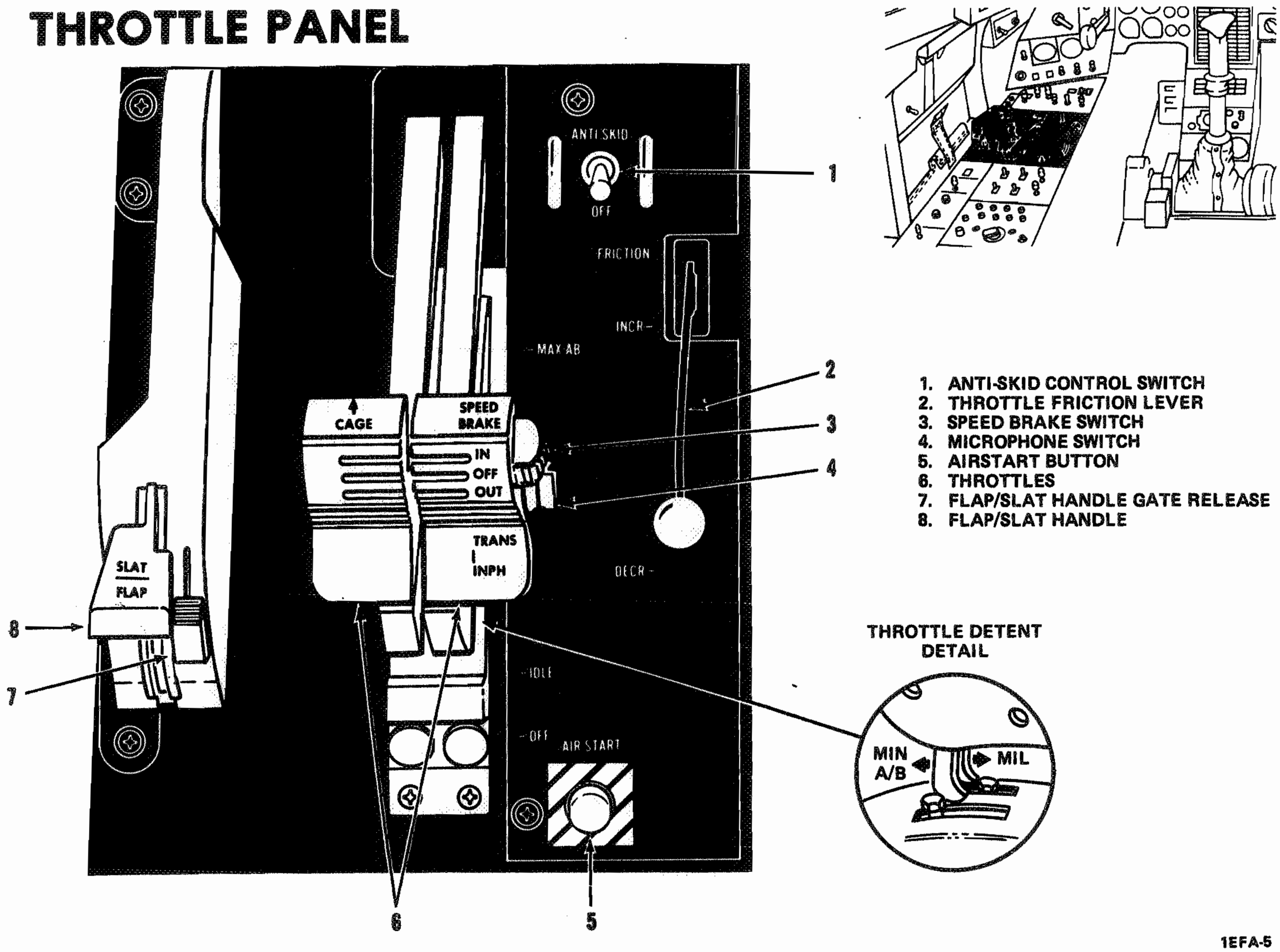
The ground ignition cutoff switch (9, figure 1-20), labeled GRD IGNITION, is marked NORM and OFF. When the switch is in OFF, the engine electrical ignition system is deactivated. When the switch is in NORM, the ignition circuits are activated.

### Translating Cowl Switches

Two translating cowl switches (6, figure 1-2), are lever-lock switches labeled L COWL and R COWL with two positions marked OPEN and AUTO. The OPEN position provides for manually opening the cowls regardless of other conditions. In AUTO, the position of the cowls is automatically controlled as a function of speed and altitude. The cowls will automatically open when either or both of the following conditions occur:

1. Mach is less than 0.44 or

# THROTTLE PANEL




1. ANTI-SKID CONTROL SWITCH
2. THROTTLE FRICTION LEVER
3. SPEED BRAKE SWITCH
4. MICROPHONE SWITCH
5. AIRSTART BUTTON
6. THROTTLES
7. FLAP/SLAT HANDLE GATE RELEASE
8. FLAP/SLAT HANDLE

Figure 1-1.

2. Airspeed is less than 255 KIAS and altitude is less than 9,000 feet.

The cowls will automatically close when either or both of the following conditions occur:

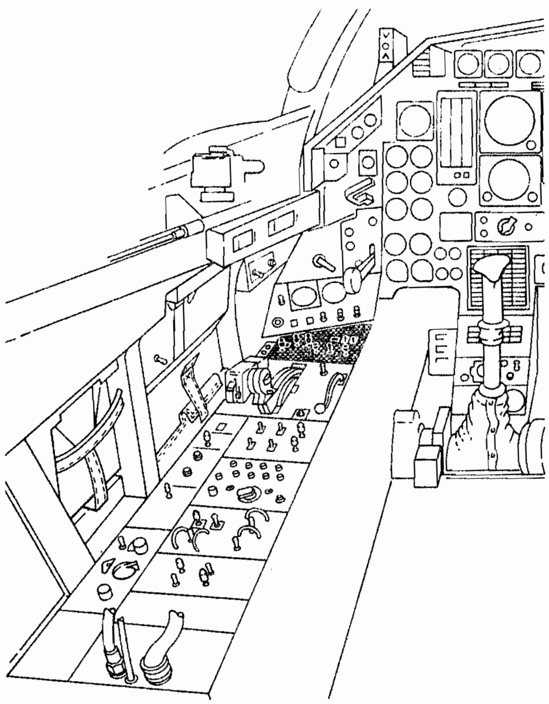
1. Mach is greater than 0.50 and no automatic open signal is present or

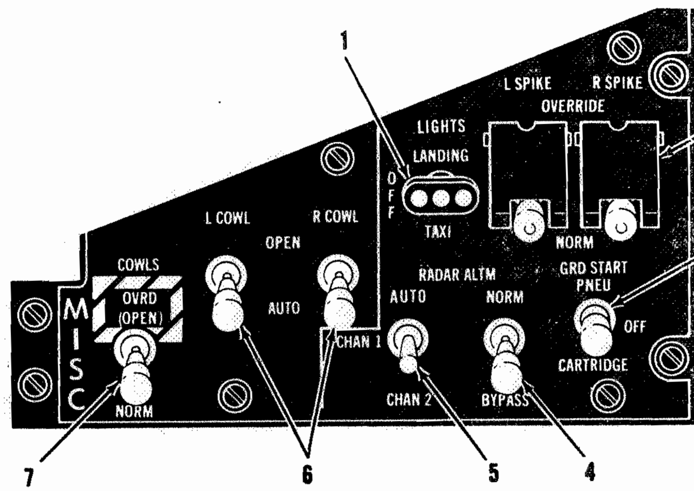
2. Should the primary mach signal source fail to close the cowls at 0.50 Mach they will close on an independent signal from the spike system, before the aircraft exceeds 0.90 Mach.

### Translating Cowl Test Switches

Two translating cowl test switches (6, figure 1-20), are provided for ground checking the operation of the translating cowl system. The switches, labeled L COWL and R COWL, Have two positions marked NORM and HI MACH. The switches are held in the NORM position when the door to the ground check panel is closed. Placing both switches to the HI MACH position simultaneously, with the translating cowl switches in AUTO will close the cowls. When the switches are returned to NORM the cowls will open. the switches are also used in conjunction with the flight control master test switch and CADC test switch for cowl systems checkout.

# MISCELLANEOUS SWITCH PANEL






1. LANDING AND TAXI LIGHT SWITCH
2. SPIKE CONTROL SWITCHES
3. ENGINE GROUND START SWITCH
4. RADAR ALTIMETER BYPASS SWITCH
5. RADAR ALTIMETER CHANNEL SELECTOR SWITCH
6. TRANSLATING COWL SWITCHES
7. TRANSLATING COWL EMERGENCY OVERRIDE SWITCH

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**Figure 1-2.**

### Translating Cowl Emergency Override Switch

The translating cowl emergency override switch (7, figure 1-2) is a lever-lock switch marked OVRD (OPEN) and NORM. With the switch in the NORM position utility hydraulic system pressure is provided to actuate the cowls. When the switch is placed to the OVRD (OPEN) position pneumatic pressure will drive the cowls open. This position is used only in the event of a hydraulic system failure or if the cowls cannot be opened by any other means. The functions of the switch are disarmed at speeds above Mach 0.50.

Note

After override has been used the system must be reset on the ground.

### Mach Trim Test Switch

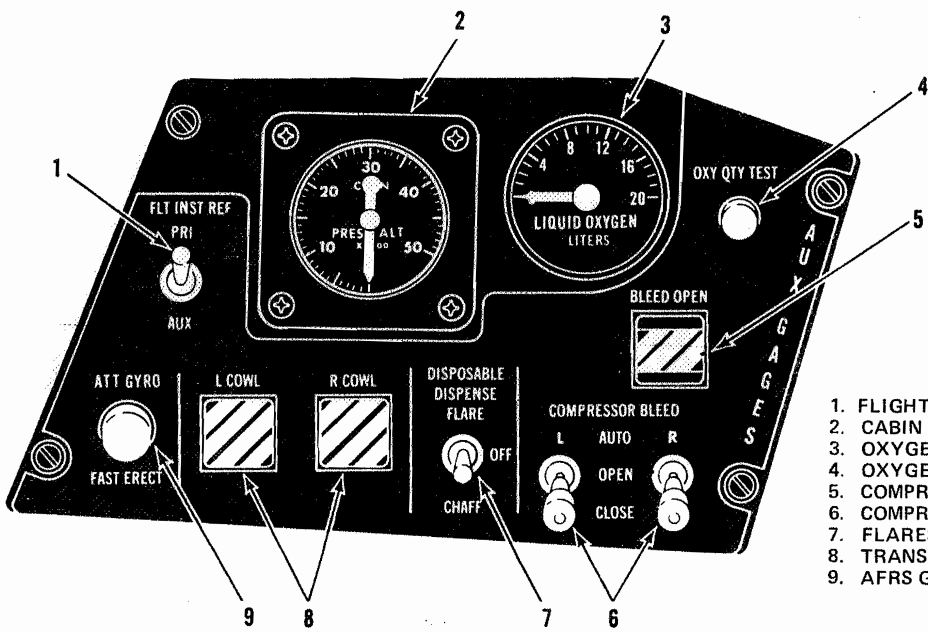
The Mach trim test switch (3, figure 1-20) marked NORM and TEST, is provided for maintenance ground check operation of the engine Mach lever on the fuel control unit.

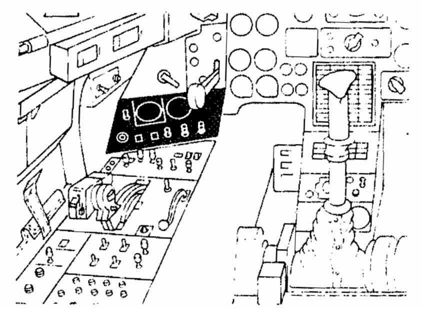
### Compressor Bleed Valve Control Switches

Two compressor bleed valve control switches (6, figure 1-3) are lever-lock type and 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. 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.

# AUXILIARY GAGE PANEL



1. FLIGHT INSTRUMENT REFERENCE SELECT SWITCH
2. CABIN ALTITUDE INDICATOR
3. OXYGEN QUANITY INDICATOR
4. OXYGEN QUANTITY INDICATOR TEST BUTTON
5. COMPRESSOR BLEED VALVE POSITION INDICATOR
6. COMPRESSOR BLEED VALVE CONTROL SWITCHES
7. FLARES/CHAFF DISPENSE SWITCH
8. TRANSLATING COWL POSITION INDICATORS
9. AFRS GYRO FAST ERECT BUTTON

A-1EFA-60

**Figure 1-3.**

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 BOTH, continued operation at the above conditions will probably result in compressor stall.

### Spike Control Switches

Two spike control switches (2, figure 1-2), labeled L SPIKE and R SPIKE, are lever-lock switches with two positions marked OVERRIDE and NORM. In NORM, 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.

### Spike Test Buttons

Two spike test buttons (2, figure 1-20) 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 (20, figure FO–3), 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 1-3) is provided to indicate the commanded positions of each engine compressor bleed valve. When electrical power is off, the indicator shows crosshatched. When electrical 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 (22, figure FO–3) show fuel flow for each engine in pounds per hour. The indicators are calibrated from 0 to 80,000 pph in increments of 2,000 pph. A digital readout of fuel flow to the nearest 100 pph is displayed on the face of the indicator.

**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 (23, figure FO–3) show nozzle position. The indicators are calibrated from 0 (closed) to 10 (fully open).

**Note**

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

### Engine Oil Pressure Indicators

Two engine oil pressure indicators (26, figure FO–3) indicate engine oil pressure. The indicators are calibrated from zero to 100 psi in increments of 5 psi.

### Engine Pressure Ratio Indicators

Two engine pressure ratio (EPR) indicators (24, figure FO–3) 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 on the indicator face is calibrated in 0.01 increments for precise reading. A set button on each indicator moves an index pointer for computed EPR. The digital readout window on the indicator face corresponds to the index pointer settings.

### Engine Turbine Inlet Temperature Indicators

Two engine turbine inlet temperature (TIT) indicators (21, figure FO–3) 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 2 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 battery 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 are located on the main caution lamp panel (figure FO–5). When lighted, the letters L ENG SPIKE and R ENG SPIKE 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 fully expanded position.

### Translating Cowl Position Indicators

Two translating cowl position indicators (8, figure 1-3) are labeled L COWL and R COWL. When the cowls are full open the indicators display the word OPEN. When the cowls are in transit the indicators will show cross hatched. With the cowls closed, the indicators will display the word CLOSED.

### Translating Cowl Caution Lamp

The translating cowl caution lamp (figure FO–5), located on the main caution panel, will light for the following conditions:

On the ground:

- If loss of either automatic open command (Mach or airspeed-altitude) occurs.

- If the flight control master test button is depressed.

In flight:

- If either translating cowl is not fully closed after speed has exceeded Mach 0.50 for 15 seconds.

- At speeds above Mach 1.10 with both translating cowls fully closed, if a loss of automatic close command redundancy from either spike controller occurs.

When the lamp lights the words AUTO COWL are visible.

### Translating Cowl Warning Lamp

A red translating cowl warning lamp (figure FO–5), located on the upper warning and advisory panel, which reads COWL when lighted is provided to warn the crew when either cowl is not fully open and one of the following conditions exists:

1. Mach is below 0.35.
2. Landing gear handle is in the down (DN) position.
3. Mach is below 0.44 for more than 15 seconds.
4. Landing gear emergency release handle has been pulled.

### Engine Oil Hot Caution Lamps

The two engine oil hot caution lamps are located on the main caution lamp panel (figure FO–5). When the oil temperature of either engine exceeds 285°F, the associated lamp will light, and the letters L ENG OIL HOT or R ENG OIL HOT will be visible in the respective lamp.

### Engine Overspeed Caution Lamps

Two amber engine overspeed lamps are located on the main caution lamp panel (figure FO–5). When lighted the letters L ENG OVERSPEED and R ENG OVERSPEED are visible. An engine overspeed lamp will light at excessive $N_1$ compressor speeds. In addition, the lamps will light prior to engine start, provided there is electrical power on the aircraft, and will go out prior to reaching idle rpm.

## 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, coolanol, 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 (17, figure FO–3), labeled L Eng and R Eng, are located on the left instrument panel. Fully depressing either button to the point where it latches will close the engine fuel shutoff valve, coolanol shutoff valve, and the utility and primary hydraulic system shutoff valves to the respective engine. It will also arm the extinguishing agent discharge switch to that engine and shutdown the JSS coolanol pumps. Additionally, a 1/4 inch white band should be visible on the inside of the fire pushbutton housing. Depressing the pushbutton again will disarm the fire extinguisher agent discharge valve; however, if the button was latched or held fully depressed for at least 4 seconds, the fuel, coolanol, and hydraulic shutoff valves will remain closed.

**WARNING**

Caution must be exercised to prevent inadvertently depressing the wrong pushbutton and shutting down the good engine since the coolanol valve and hydraulic shutoff valves cannot be reopened in-flight. In some cases, it may take from 5 to 10 minutes for hydraulic system pressure to decay enough to light the primary and utility hydraulic caution lamps of the engine that has been shut down.

**CAUTION**

Operation of the JSS transmitters with an engine fire pushbutton depressed may cause transmitter overheat.

**Note**

When a fire pushbutton is fully depressed and latched, the pushbutton will be recessed approximately 1/4 inch. The white band may not be visible while the fire warning lamp is lighted due to its tendency to reflect the red glow.

### Agent Discharge/Fire Detect Test Switch

The agent discharge/fire detection test switch (15, figure FO-3), located on the left main instrument panel, is a three position lever-lock 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 Extinguishing System" and "Wheel Well Overheat Detection System," this section.

**Note**

The engine fire extinguishing agent will be depleted after one actuation.

## ENGINE COMPARTMENT OVERHEAT DETECTION SYSTEM

An overheat detection circuit is installed in each engine compartment. Because of the high temperature of the 16-stage bleed air being ducted through the compartment in close proximity to a fuel cell, a potentially hazardous condition would exist if a bleed duct should fail. The indication of a bleed duct failure is provided by caution lamps, labeled L BLEED DUCT and R BLEED DUCT, on the main caution lamp panel (figure FO-5). These lamps should light when the agent discharge/fire detection test switch is held in the FIRE DETECT TEST position.

### Engine Fire Detection System Module Self-Test Button

The engine fire detection system module self test button
R (1, figure 1-20), labeled MODULE SELF TEST, is used to ground check the detection circuits in the event any of the fire and overheat detection systems lamps should fail to come on when the agent discharge/fire detect test switch is positioned to FIRE DETECT TEST.

## 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. During ground operation, the time to accelerate from idle to military power will increase with a rise in ambient temperatures. 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 1-66.

## ENGINE OIL QUANTITY INDICATOR

The engine oil quantity indicator (51, figure FO-3), labeled L and R for the left and right engine, respectively are graduated from 0 to 16 in one quart increments. A pointer for each display provides an indication of the number of quarts of usable oil remaining.

**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 through 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 oil quantity indicator test button (51, figure FO–3), located on the left main instrument panel.
- The oil quantity indicator gives a valid reading only during engine run and up to 15 minutes after engine shutdown, provided electrical power is supplied by external power source or the generator of the other engine. This is due to the fact that oil in the tank seeps into the accessory gear box after engine shutdown.

## ENGINE OIL QUANTITY INDICATOR TEST BUTTON

The engine oil quantity indicator test button (51, figure FO–3), will drive the oil quantity indicators 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.

## OIL LOW CAUTION LAMP

An amber oil low caution lamp (figure FO–5), 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.

## FUEL SUPPLY SYSTEM

The fuel supply system (figure FO–7) consists of a forward and aft fuselage tank, two wing tanks, vent tank, and associated fuel pumps, controls, and indicators. The fuel system employs ten fuel pumps, of which six deliver fuel to the engine and four are used to transfer fuel from the wing tanks to the fuselage tanks. Provisions are made for 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.

## FUEL TANKS

The fuel tanks consist of internal forward and aft fuselage tanks, left and right internal wing tanks, and an integral vent tank in the vertical stabilizer. See figure FO–7 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. The reservoir tank reserves approximately 2,500 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 2,300 ($\pm$235) pounds. The aft tank is divided into bay A-1, incorporating two "saddle" tanks, and bay A–2. All fuel in the external and internal wing 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 transfer fuel into the fuselage.

## FUEL QUANTITY MEASUREMENT SYSTEM

**Note**

> Due to a structural modification in the EF-111A, it is possible to trap 2.7 gallons of fuel in the saddle tanks. Since the fuel quantity measuring system assumes that this small volume of trapped fuel represents an overall fuel level in the aft tanks, an indication of some 150 pounds is added to the fuel quantity. When combined with the $\pm 250$ pound accuracy tolerance of the fuel probes, approximately 400 pounds of fuel may be erroneously indicated when the aft tanks are essentially empty.

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 2,000 pounds. A normal response by the indicators verifies that the indicator circuit is functioning normally.

## FUEL PUMPS

There are ten 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, and 8 and 10 are in the right wing. Pumps 3, 4, 5 and 6 are the primary engine feed pumps, and 1 and 2 are standby engine feed pumps. Number 1 boost pump is a standby pump and operates continuously with the engine feed selector knob 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. The number 2 pump is normally in standby, with the engine feed selector knob in AFT or BOTH, and is energized by pressure sensing. In AUTO, the number 2 pump is energized when the fuselage fuel quantity indicator indicates more than approximately 8500 pound 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 buses:

### AC POWER

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

### DC POWER

| Pumps | Bus |
|---|---|
| 4, 5, 7, and 8 | Essential |
| 1, 2, 3, 6, 9, and 10 | 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 aircraft 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 quantiy indicator is inoperative or malfunctions, refer to "Abnormal Fuel Distribution/Indicator Malfunction," 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 fuel distribution between the fuselage tanks is automatically controlled by the F (forward) and A (aft) pointers on the fuselage fuel quantity indicator to maintain the aircraft center-of-gravity within prescribed limits. 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 8,500 pounds, as is the case when the tanks are fully serviced and until all wing 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 7,900 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

An alternate fuel distribution monitoring system provides a means to indicate an abnormal aft 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 9,000 pounds. Likewise, the two in the aft tank are located at approximately 5,300 pounds and 2,500 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 5,300 pound sensor in the aft tank is covered. Likewise, when the fuel level in the forward tank decreases to a point below the 9,000 pound level, the fuel distribution caution lamp will light if the aft tank level is above the 2,500 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 backup 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 internal wing tanks, it must be transferred to the fuselage tanks. Fuel transfer is controlled by the transfer knob. The fuel level in the fuselage 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 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 internal wing tanks.

Transfer from the internal wing tanks is effected by transfer pumps. When transferring from the 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, 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 lamp 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. 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 handle is up. In this mode, the system is automatically depressurized when the refueling receptacle door is opened or when the gear handle is down. 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 handle 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.

## FUEL DUMP SYSTEM

The fuel dump system provides the capability of jettisoning fuel at a rate of 2,300 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 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.

## 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 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 5,100 to 5,800 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 the receiver pilot or 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 (EBL) 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 hookups 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.

## FUEL SYSTEM CONTROLS AND INDICATORS

### Engine Feed Selector Knob

The engine feed selector knob (5, figure 1-4) 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 - Forward and aft fuselage fuel quantity indicator differential approximately 8,200; 1, 3, 4, 5 and 6 energized; 2 on standby.

Forward and aft fuselage fuel quantity indicator differential approximately 7,900 or less; 1, 3, 4, 5, and 6 energized; 2 on standby.

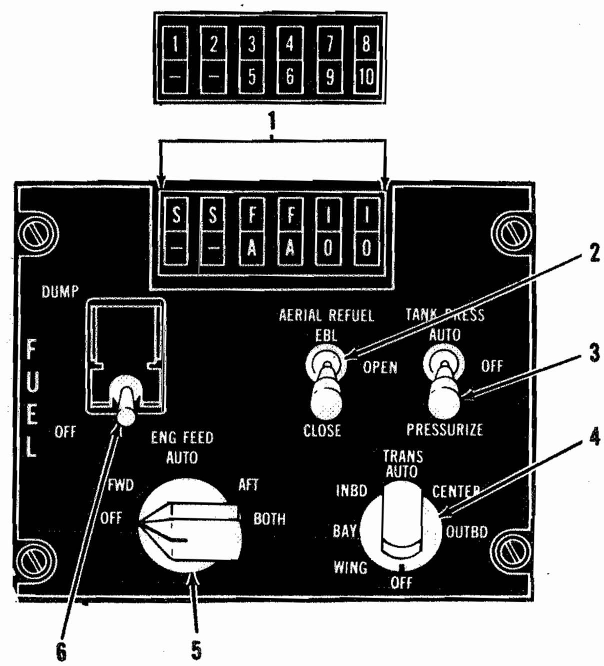
Forward and aft fuselage fuel quantity indicator differential approximately 8,500 or greater; 1, 2, 3, and 4 energized.

AFT - 1, 5 and 6 energized, 2 on standby.

BOTH - 1, 3, 4, 5 and 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.

## 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

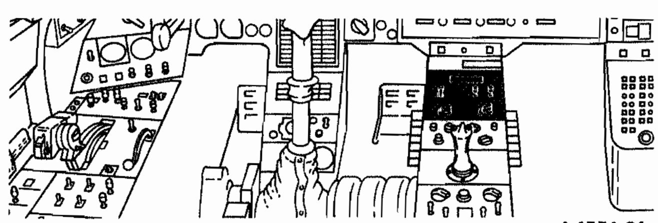


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Figure 1-4.

**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 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.

**Note**

> The knob must be in either the AUTO or BOTH position to enable the functions of the air refueling switch.

#### Fuel Transfer Knob

The fuel transfer knob (4, figure 1-4) 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. The BAY, INBD, CENTER and OUTBD positions are not used. The AUTO position automatically sequences the transfer of fuel from the wing tanks.

**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 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 1-4) is 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 1-4) is a 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 1-4) is a lever-lock switch marked EBL, OPEN and CLOSE. Refer to "Air Refueling," this section, for operation of the air refueling switch.

#### Nosewheel Steering/Air Refuel Button

The nosewheel steering/air refuel button (4, figure 1-16), on the control stick grip, is labeled NWS and A/R DISC. The air refueling function of the button is activated when the aircraft is airborne to provide a means of manually disconnecting the air refueling boom. Depressing the button will interrupt power to the boom latching mechanism causing it to unlatch. For a description of the NWS function of the button, refer to "Nosewheel Steering System," this section.

#### Position Lights/Stores Refuel Battery Power Switch

The position lights/stores refuel battery power switch (5, figure 1-20) is marked POS LIGHTS, NORM and STORES REFUEL. The STORES REFUEL position is not used. 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 (41, figure FO–3) is provided to self-test the fuel quantity indicators. 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

With the engine feed selector switch in any position other than AUTO, the fuel distribution caution lamp will light 10 to 15 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 7,600 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 (47, figure FO–3) 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. The BAY, LI, RI, LC, RC, LO and RO positions are not used. 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 (44, figure FO–3) provides indications of total fuel in all tanks and the fuel remaining in individual wing tanks. The indicator is graduated from zero to 5 (times 1,000 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 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 1,000 pounds could exist.

**Note**

- When the strip lights are turned on, there may be an increase on the fuel totalizer by 300 to 500 pounds.
- Due to a structural modification in the EF-111A, it is possible to trap 2.7 gallons of fuel in the saddle tanks. Since the fuel quantity measuring system assumes that this small volume of trapped fuel represents an overall fuel level in the aft tanks, an indication of some 150 pounds is added to the fuel quantity. When combined with the ±250 pound accuracy tolerance of the fuel probes, approximately 400 pounds of fuel may be erroneously indicated when the aft tanks are essentially empty.

### Fuselage Fuel Quantity Indicator

The fuselage fuel quantity indicator (40, figure FO–3) 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 an operational 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 1,000 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 8,200 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. 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 FO–5), 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 below limits.

### Fuel Low Caution Lamp

The amber fuel low caution lamp (figure FO–5) located on the main caution lamp 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 2,300 ($\pm$235) pounds. Due to the gaging system tolerances, the forward fuel quantity indication will be between 1,700 and 3,000 pounds.

**WARNING**

Zero or negative g operation must be avoided whenever the fuel low caution lamp is lighted. The fuel system can supply fuel to the engines during zero or negative g operation for 10 seconds if the reservoir tank is initially full. There may be no zero or 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 1-4) 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 ($\pm$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 - Not used

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 FO–5), located on the main caution lamp panel, lights when fuel tank air pressure drops below approximately 3.5 ($\pm$0.5) 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.
- The fuel tank pressurization caution lamp will light 30 seconds after the landing gear handle is moved to the UP position, if the tanks have not pressurized by that time, and remain on until the tanks have pressurized. When the tanks are full, the lamp will not normally light, but when the tanks are relatively low on fuel, the lamp will light briefly due to the large volume of air required to pressurize the tanks.

### Fuel Distribution Caution Lamp

The fuel distribution caution lamp (figure FO–5), located on the main caution lamp 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 7,600 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 aft abnormal fuel distribution conditions only. When the lamp lights the letters FUEL DISTRIB are visible.

### Nosewheel Steering/Air Refueling Indicator Lamp

The nosewheel steering/air refueling indicator lamp (figure FO–5), located on the upper warning and advisory 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 the normal operation, except the lamp will go out if the NWS and A/R DISC button is depressed. The lamp will come on when the NWS and A/R DISC button is released, if a disconnect has occurred. For a description of the NWS function of the lamp, refer to "Nosewheel 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 it 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 internal wing tanks.
- If all tanks were fully serviced at takeoff, both engines will be fed from the forward fuselage tank until internal wing tanks are expended and the fuel level in the forward tank is burned down to approximately 8,200 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 7,900 pounds differential the automatic transfer valve will open to allow fuel to be transferred from the aft tank to the forward tank. This will reestablish the 8,200 pound differential.
- If the aft tank is burned down to 8,500 pounds differential, the aft tank pumps are turned off and both engines are fed from the forward tank until the 8,200 pound differential is reestablished.

### 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 8,000 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 to WING to empty the external wing and internal wing tanks. 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 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 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 antisuction 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.

**Note**

> During ground operation with high ambient temperature, idle thrust, and unpressurized fuel tanks, gravity feed operation may result in engine flameout.

### 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 8,500 pounds differential, 6 only)
5. Transfer 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 2,300 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 1,750 pounds per minute if both aft tank pumps are operating or at 1,100 pounds per minute if only one pump is operating. All fuel except that in the reservoir tank (approximately 2,500 pounds) can be dumped.

**WARNING**

- Due to possible fuel dump line damage, fuel should not be dumped after a wheel well hot, bleed air duct failure, or fuselage or engine fire, unless thrust requirements or landing conditions dictate.
- To avoid the possibility of dumped fuel reentering the aircraft and causing a fire hazard, fuel dumping should be accomplished in coordinated flight at approximately 1 "g" with 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 8,200 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 8,200 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 8,200 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.

### Fuel Dumping Without Tank Pressurization

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/AR lamp will light to indicate the receptacle is open and the system is ready to accept the refueling boom.

**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 boom 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 receiver pilot by use of the NWS and A/R DISC 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 boom 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 NWS and A/R DISC 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 boom 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 hookups 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 5,000 pounds. Normal operating boom loads do not exceed 2,300 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 begining 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.

## EXTERNAL STORES JETTISON

**Note**

Installation of jettisonable external stores is not authorized.

### External Stores Jettison Button

The external stores jettison button (14, figure FO–3) is a recessed pushbutton labeled EXT STORES JETTISON. Depressing the button, when the aircraft is on the ground or in flight, will jettison all external stores (drop tanks).

### SELECT STORES Jettison Master Switch

The two-position select stores jettison master switch (61, figure FO–3) is a lever-lock type switch labeled MSTR and permits selective jettisoning of stores when ON is selected.

### SELECT STORES Jettison PUSHBUTTONS

The external stores jettison select pushbuttons (61, figure FO–3) are labeled SEL 4/5 (inboard) and SEL 3/6 (outboard). When the pushbuttons are depressed with the external stores master jettison switch in the ON position, the selected stores will be jettisoned when the weapon release button is depressed.

## ELECTRICAL POWER SUPPLY SYSTEM

The electrical power supply system provides 115 volt, three-phase, 400 cycle ac power and 28 volt dc power. Two integrated drive generator assemblies, one mounted on each engine, supply ac power. Two transformer rectifier units provide 28 volt dc power. (See figures FO–8 and FO–9.)

## ALTERNATING CURRENT POWER SUPPLY SYSTEM

AC power is supplied by two 90 kva generating systems. Each integrated drive generator (IDG) 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 by circulating the oil through the IDG supplementary air-oil cooler and the engine/IDG air-oil cooler. Cooling air for the air-oil coolers is provided by ram air inflight and by nacelle vent ejector air during ground operation. After T.O. 1F-111(E)A-556, additional cooling air is provided during ground operation by outside air entering through a blow-in door and passing through the core of the IDG supplementary air-oil cooler. During afterburner operation, additional cooling is provided by circulating the oil through the AB fuel-oil cooler. The left and right IDG's 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 IDG supplies power to the right main ac bus, and the left IDG to the left main ac bus and the essential bus. Each main ac bus is normally individually powered and isolated from the other. The system provides a bus tie function automatically in the event of IDG failure. If a fault or malfunction occurs causing an undervoltage, overvoltage, underfrequency, or overfrequency, the associated ac generator control unit removes the IDG from the bus. If the malfunction is corrected, the IDG 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 IDG's. The emergency generator is driven by a hydraulic motor which receives power from the utility hydraulic system. During engine cartridge start, the emergency generator will not operate with the engine ground start switch in the CARTRIDGE position. 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 1-5), marked ON, OFF-RSET, and TEST are lever-locked in the ON position and spring-loaded from TEST to OFF-RSET. The ON position supplies 28 volts dc for generator control unit excitation and connects the IDG to its respective bus. The OFF-RSET position disconnects the IDG from its bus, opens the exciter field, and resets the generator control unit. If an IDG is disconnected by overvoltage or undervoltage conditions, an attempt to reset the IDG may be made by cycling the switch to OFF-RSET, then to ON. A successful reset will be indicated by the power flow indicator and the generator caution lamp will go out. In the spring-loaded TEST position, the IDG is energized but not connected to the bus. The TEST position is primarily for maintenance use but can be used in-flight as a troubleshooting procedure to verify IDG output. If a generator caution lamp goes out when the switch is placed in the TEST position, generator voltage and frequency are satisfactory and the distribution system is at fault. If the lamp remains lighted, generator voltage or frequency are at fault.

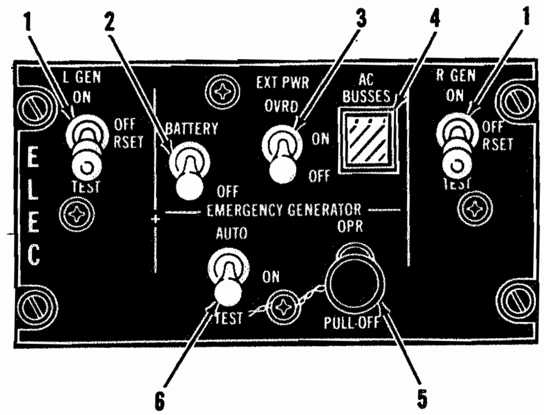
### Electrical Power Flow Indicator

The electrical power flow indicator (4, figure 1-5) is a flip-flop indicator labeled AC BUSSES and displays the various bus configurations. If both buses are receiving power from their respective IDG, the indicator will display NORM. If only one IDG 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 IDG comes on the line, then it will indicate NORM. The indicator will display a crosshatched surface if there is no ac power applied to
R the aircraft or while the emergency generator switch is in TEST.

### Emergency Generator Switch

The emergency generator switch (6, figure 1-5) is 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 will operate, and be connected to the essential ac bus.

## ELECTRICAL CONTROL PANEL




1. GENERATOR SWITCHES
2. BATTERY SWITCH
3. EXTERNAL POWER SWITCH
4. ELECTRICAL POWER FLOW INDICATOR
5. EMERGENCY GENERATOR INDICATOR/CUTOFF PUSHBUTTON
6. EMERGENCY GENERATOR SWITCH

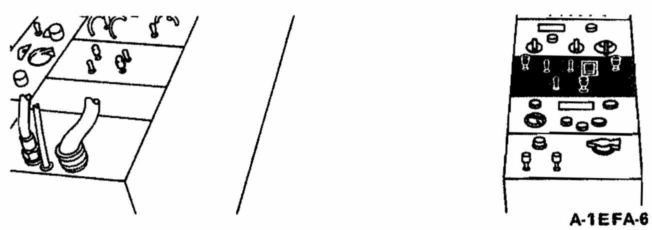


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Figure 1-5.

**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.

**WARNING**

- In flight, avoid selecting the ON position of the emergency generator switch. To do so places an unnecessary drain on the hydraulic system and does not prevent a fast erection cycle of the AFRS if emergency generator power is required.

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 angle-of-attack indexers will remain lighted and, with the instrument test button (7, figure 1-20) depressed, the instrument tapes will drive if both transformer/rectifiers are operational.

### Emergency Generator Indicator/Cutoff Pushbutton

The emergency generator indicator/cutoff pushbutton (5, figure 1-5) 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 IDG's 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 1-5) is 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 IDG will supply total aircraft electrical load, and external power is disconnected from the ac buses. With only the right engine operating, the right main IDG supplies power to the right main ac bus, and extenal power feeds the left main ac and essential buses. A power monitor measures external power voltage, frequency and phase sequence. Should any one of these parameters be out of tolerance, the monitor prevents application of external power. When the switch is in the OVRD position, the monitor circuit is bypassed, allowing external power to be applied to the aircraft buses, even though the voltage, frequency, or (in some cases) phase may be out of tolerance.

**CAUTION**

The OVRD position should not be used unless required by an emergency.

### Generator Caution Lamps

Two generator caution lamps (figure FO-5) are located on the main caution lamp panel. Either lamp will light when its respective IDG is disconnected from its bus and remain lighted until the generator has been reconnected to its bus. When lighted, the letters L GEN are visible in the left lamp and R GEN in the right lamp.

### Overload Indicator Reset Switch

The overload indicator reset (OVLD IND RESET) switch (3, figure 1-6), marked OFF and ON, is lever-locked in the OFF position and spring-loaded from the ON position. When a generator overload condition exists, jammer transmitters will go to standby and both GEN OVLD warning lamps (figure FO-5) will light. When the ON position is selected, jammer transmitter power is again available and the GEN OVLD lamps will go out.

### Generator Power Monitor Selector Switch and Ammeter

The generator power monitor select switch (4, figure 1-6) is marked L GEN, OFF and R GEN. The ammeter marked AMPS AC has a green scale from 0 to 250, a yellow scale from 250 to 315, and a red scale from 315 to 350. When L GEN or R GEN is selected, the respective generator load will be displayed on the ammeter.

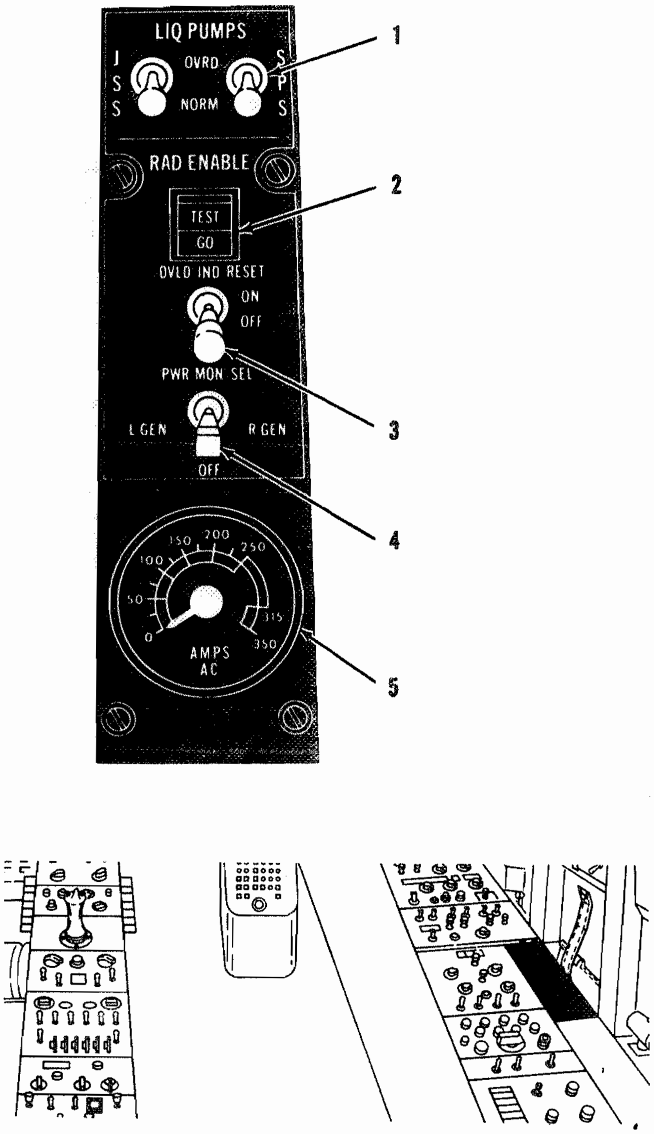
### Generator Overload Warning Lamps

Two generator overload warning lamps (figure FO-5) are located on the JSS/SPS/TTWS caution and warning panel and the upper warning and advisory panel. The letters GEN OVLD will be visible when the load on either generator exceeds 250 A.

## 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 four dc buses; a main dc bus, an essential dc bus, a battery 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

# COOLING AND POWER MONITOR PANEL




1. JSS/SPS LIQUID PUMPS SWITCHES
2. RADIATE ENABLE INDICATOR LAMP
3. OVERLOAD INDICATOR RESET SWITCH
4. GENERATOR POWER MONITOR SELECT SWITCH
5. GENERATOR AMMETER

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Figure 1-6.

crew module on the aft console, (figure 1-7). 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

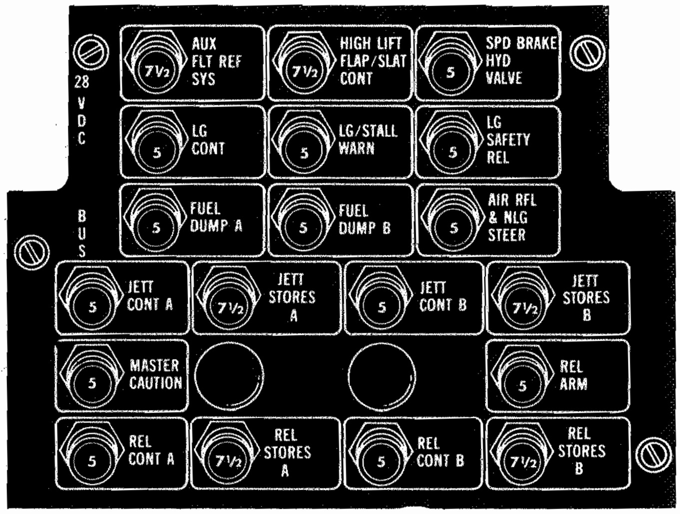
The battery switch (2, figure 1-5) is marked OFF and BATTERY. Positioning the switch to BATTERY connects the engine start bus to the aircraft 24 volt battery, provided the essential dc bus is not energized. If the essential dc bus is energized, the battery is connected to the main dc bus through the 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.

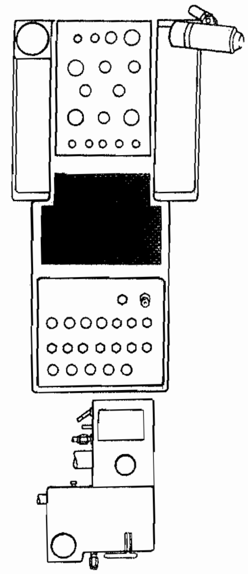
## HYDRAULIC POWER SUPPLY SYSTEM

Hydraulic power is supplied by two independent, parallel hydraulic systems called the primary and utility systems (figure FO–10). Both systems operate simultaneously to supply hydraulic power for the flight controls and wing sweep. If one system should fail, the other 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:

- Nosewheel steering
- Landing gear
- Wheel brakes
- Speed brake
- Flaps/slats
- Rotating glove
- Rudder authority
- Tail bumper
- Emergency electrical generator
- Spikes
- Air refueling system
- Translating cowls
- Ground cooling fans R

# CIRCUIT BREAKER PANEL



| 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, STALL WARNING LAMP, AND TO THE WARNING TONE GENERATOR. |
| LG SAFETY RELAY | PROVIDES POWER TO THE GROUND SAFETY SWITCH CIRCUIT WHICH INHIBITS CERTAIN AIRCRAFT FUNCTIONS. |
| FUEL DUMP A | PROVIDES POWER TO THE 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. |
| AIR RFL & NLG STEER | PROVIDES POWER TO THE AIR REFUELING RECEPTACLE OR NOSE LANDING GEAR STEERING. |
| MASTER CAUTION | PROVIDES POWER TO WARNING, CAUTION AND INDICATOR LAMPS. |

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Figure 1-7.

Hydraulic pressure for each system is supplied by two engine-driven pumps. To assure hydraulic pressure in the event of single engine failure, each engine drives one utility and one primary pump. Pressurized accumulators are installed in the system to supplement engine-driven pump delivery during heavy demand. 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 lamps for each of the four pumps are on the main 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.

## HYDRAULIC PUMPS

Four variable delivery pumps are employed, (consisting of one primary and one utility pump, driven by each engine).

## HYDRAULIC ACCUMULATORS

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 figures 1-64 through 1-66 for servicing data.

## HYDRAULIC FLUID RESERVOIRS

Both primary and utility hydraulic reservoirs use 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 each 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. The reservoir also acts as a surge damper for return line impulse pressures. See figure 1-66 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 400 psi, and cut off all systems except flight controls and wing sweep. The primary system pressure must increase to approximately 1,200 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, nosewheel steering, and ground cooling fan systems when the aircraft is in flight. Isolation takes place immediately after the last of the following two controlling conditions is satisfied:

1. The utility hydraulic isolation override switch is in NORM.
2. The landing gear is up and locked.

## UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH

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

- Landing gear
- Nosewheel steering
- Brakes
- Ground cooling fans

During normal operation positioning the switch to PRESSURIZE supplies utility hydraulic pressure to these systems.

## HYDRAULIC PRESSURE INDICATORS

There are two 0-4,000 psi hydraulic pressure indicators (25, figure FO–3), one each for the utility and primary systems.

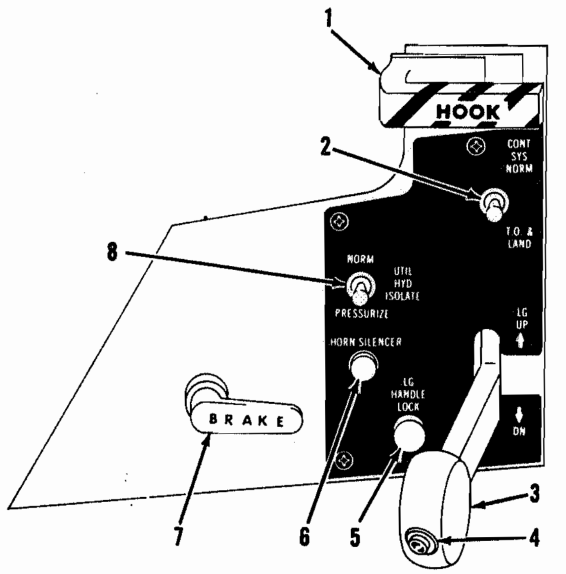
## LOW PRESSURE CAUTION LAMPS

Four low pressure caution lamps, energized by pressure switches in each pump pressure line, are located on the main caution lamp panel (figure FO–5). These lamps light when pump pressure decreases below 500 psi. When lighted, the letters L PRI HYD, L UTIL HYD, R PRI HYD, and R UTIL HYD are visible.

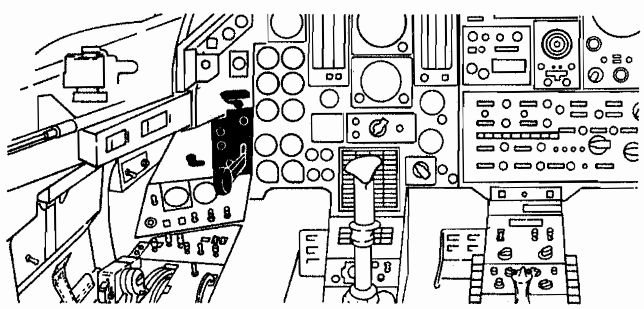
**Note**

It is possible for the utility hydraulic system to go into emergency isolation when the primary system pressure decreases to approximately 500 psi without lighting the primary low pressure caution lamps.

# 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. HORN SILENCER BUTTON
7. AUXILIARY BRAKE HANDLE
8. UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH



A-1EFA-4

**Figure 1-8.**

## HYDRAULIC FLUID OVERHEAT CAUTION LAMPS

Two hydraulic fluid overheat caution lamps, one for each system, are located on the main caution lamp panel (figure FO-5). A lamp lights when the hydraulic fluid temperature of the system exceeds its allowable limits. When lighted, the letters PRI HOT and UTIL HOT are visible.

## PNEUMATIC POWER SUPPLY SYSTEMS

The independent pneumatic power supply systems are 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. Translating cowl emergency extend system.
7. Wheel brakes parking/emergency system.
8. Arresting hook extend system.
9. Tail bumper system.

Two pneumatic reservoirs, one for each hydraulic system reservoir, provide pneumatic pressure for hydraulic system operation. Reservoirs in the main landing gear wheel well provide pneumatic pressure for emergency: (1) landing gear extension, (2) operation of the air refueling system, and (3) spike and translating cowl operation. The overwing fairing actuators are provided pneumatic pressure to hold the fairing firmly against the aircraft 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 figures 1-65 and 1-66.

## LANDING GEAR SYSTEM

The landing gear is tricycle-type, forward retracting, and hydraulically operated. The main landing gear consists of a single common trunnion with two wheels which ensures symmetrical main landing gear operation. Thermal pressure relief plugs in the main landing gear wheels relieve tire pressure in the event of overheated 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 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. One actuator retracts the main landing gear, Two 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 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 nosewheel well doors. One actuator retracts the nose landing gear, one locks the gear in the retracted position, and one locks the gear down. Through linkages, the uplock actuator opens and closes the nose wheel well doors.

## 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 1-9 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 inflight systems.

- Cowl anti-icing.
- Adverse yaw compensation.
- SIS alpha limiter.
- Artificial stall warning.
- Operation of landing gear handle from the down position.
- Secondary alpha/beta probe heater.
- Touchdown skid control.
- Speed brake.
- Jammer radiate.

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

- Nosewheel steering.
- Ground roll spoilers.
- Ground to crew 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.
- TFR and flight control tie-in checks.

### Landing Gear Handle

The landing gear handle (3, figure 1-8) is marked UP and DN. A red landing gear warning lamp is located in the end of the landing gear handle.

**Note**

> Rudder authority is automatically scheduled with the landing gear handle. Refer to "Yaw Channel", this section. Also, the flight control system switch is interlocked through the landing gear handle. Refer to "Flight Control System Switch" this section.

GEAR UP

When the handle is moved to the UP position, the red warning lamp lights and an electrical signal actuates the 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, and the green landing gear position lamp goes out. 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 nosewheel doors close, snubbers mounted on the doors engage the tires to stop nose wheel rotation. The speed brake door actuator extends the main gear forward door. Hydraulic pressure 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 and the main gear position lamp goes out. 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. The red warning lamp goes out.

# GROUND SAFETY LOCKS AND SAFETY PINS

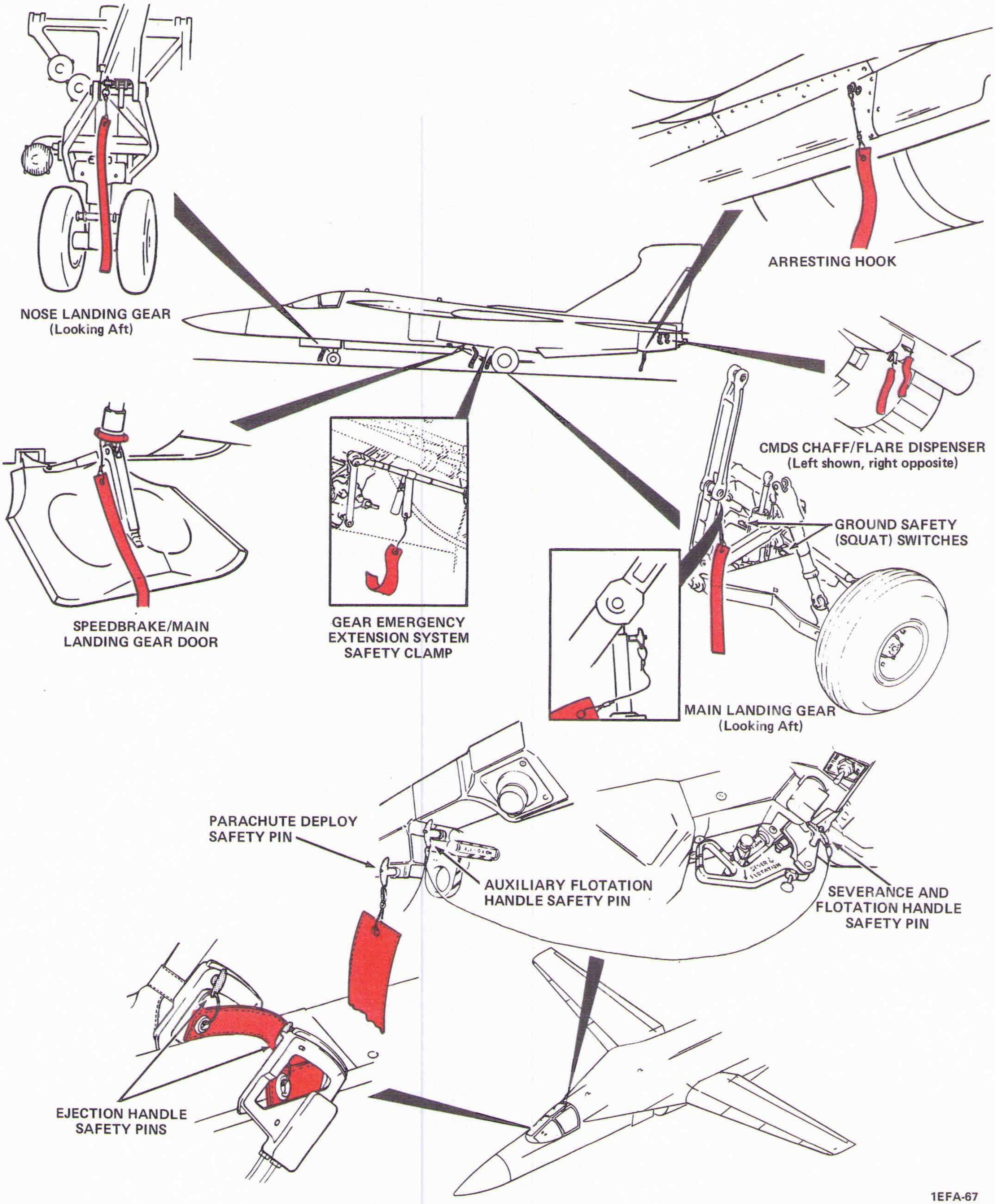



Figure 1-9.

**Note**

> If the warning lamp on the gear handle remains lighted following gear retraction, the landing gear is not fully up and locked, or the speed brake is open. For landing gear retraction malfunction procedures, refer to Section III.

GEAR DOWN

When the handle is moved to the DN position, the red warning lamp lights and an electrical signal actuates a 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 damping of its retract actuator. When the gear is almost extended, the downlock actuator drives it fully extended and locked, lighting the green landing gear position lamp. The speed brake door actuator opens the main gear forward 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, lighting the main gear position lamp. Downlock actuation causes the speed brake door actuator to position the door in the partially retracted (trail) position, causing the red warning lamp in the handle to go out. 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 uplocks 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- enegized, 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 (5, figure 1-8) 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, refer to Section III.

### Landing Gear Emergency Release Handle

The landing gear emergency release handle (62, figure FO-3), labeled ALT GEAR DOWN, 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. Pneumatic pressure will then actuate the nose and main gear downlocks and retract the speed brake door to the trail position. 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 (with the landing gear handle down) after the gear is extended and locked. Should this occur, pushing the ALT GEAR DOWN handle 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.

**Note**

> Once the gear has been extended by the emergency method, it cannot be retracted.

### Landing Gear Warning Horn

The landing gear warning horn provides an intermittent audible signal in the crewmembers' 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 200 $\pm$12 knots.
2. The aircraft is below an altitude of 10,000 $\pm$350 feet.

3. One or both throttles are set below minimum cruise power.

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 warning horn silencer button (6, figure 1-8).

**Landing Gear Position Indicator Lamps**

The landing gear position indicator consists of a platform silhouette of the aircraft with two green indicator lamps positioned to represent the nose and main landing gear (42, figure FO–3). When the landing gear is down and locked, an electrical switch in each downlock actuator is activated and 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 the up position.
- When accompanied by the warning horn, that airspeed is below 200 $\pm$12 knots, the aircraft is below 10,000 $\pm$350 feet, one or both throttles are below minimum cruise power 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 the trail position, the lift cylinder pressure is relieved and the tail bumper is extended by the pneumatic action of the tail bumper dashpot, which functions as the impact shock absorber.

## NOSEWHEEL STEERING SYSTEM

Nosewheel steering provides aircraft directional control during taxiing, takeoff and landing. The system is electrically engaged hydraulically powered and controlled by the rudder pedals. Rudder pedal movement provides a gradually increasing ratio between steering angle and pedal displacement. More pedal movement is needed near neutral for the same wheel movement than is needed near full rudder pedal position. During nose gear retraction, the nosewheels are automatically centered if the rudder pedals are not displaced over 50 percent. Nose gear steering adjustment can be checked by disengaging nosewheel steering while taxiing on a level surface. If a steering transient is observed on reengagement, a misalignment/malfunction is indicated.

**CAUTION**

If a misaligned/malfunctioning steering system is evident, do not take off unless required, and do not retract landing gear.

Maximum rudder pedal deflection steers the nosewheels 40 degrees either side of center with resultant aircraft turning radius as shown on figure 2-3. Nosewheel shimmy damping is accomplished by restricting hydraulic flow within the steering valve. Full rudder authority, and thus full steering authority, is automatically provided whenever the landing gear handle is in the DN position or the rudder authority switch is in the full position. The nosewheel steering system is equipped with a limit switch mounted on the nose landing gear shock strut adapter. When nosewheel 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 nosewheels are returned to the normal steering range, (0° to 40°), controlled steering automatically reengages. Power from engaging steering is furnished from the essential dc bus.

**Note**

- Nosewheel steering will not be available if the landing gear is extended using the landing gear emergency release handle.
- With the air refueling switch in OPEN and the engine feed selector knob in AUTO or BOTH, nosewheel steering will be operative only while the nosewheel steering/air refueling button is depressed.

- With the engine ground start switch in any position except OFF, nosewheel steering will be operative only while the nosewheel steering/air refuel button is depressed.

## NOSEWHEEL STEERING/AIR REFUEL BUTTON

A nosewheel steering/air refuel button (4, figure 1-16), is located on the control stick grip. The button is labeled NWS and A/R DISC. With the weight of the aircraft on the main landing gear, depressing the 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 nosewheel steering will engage and remain engaged as long as the NWS and A/R DISC button is held depressed.

**Note**

> When the nosewheel steering button is depressed and released to disengage the system, a three second time delay is initiated. The system may be reengaged during the three second period by depressing the button, but the holding relay will not be energized during this period.

For a description of the A/R DISC function of the button refer to "Fuel Supply System," this section.

## NOSEWHEEL STEERING/AIR REFUELING INDICATOR LAMP

The green nose wheel steering/air refueling indicator lamp labeled NWS/AR (figure FO-5) will light when the nosewheel steering system is energized. For a description of the air refueling disconnect function of the lamp, refer to "Fuel Supply System," this section.

**Note**

> During ground operation when the air refueling door is open the nosewheel steering/air refueling indicator lamp will light to indicate door position and nosewheel 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 each brake and the other circuit operates the other half. During normal operation of the brakes, pressure is metered to the brakes from 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 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 1,000 psi. At this pressure all normal braking will be lost and the pedals will be fully depressed.

**Note**

> Brake pedals should not be actuated inflight when utility hydraulic pressure is isolated from the brake system as emergency brake accumulators will be depleted. If the utility hydraulic system subsequently fails or becomes isolated, pedal braking may not be available on landing. If brake pedals are actuated inflight during normal utility hydraulic system isolation, placing the utility hydraulic system isolation switch to PRESSURIZE for several seconds will replenish brake accumulators.

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 the gear is retracted as the brakes are isolated from the hydraulic system.

**CAUTION**

Make certain that the aircraft is definitely airborne before retracting the landing gear. Premature movement of the landing gear handle to UP may result in blowing out both main gear tires due to the action of the automatic braking system.

## ANTI-SKID SYSTEM

The anti-skid touchdown control system provides the following functions:

- Anti-skid touchdown control.
- Proportional skid control.
- Locked wheel skid control.
- Anti-skid failure detection.

Anti-skid touchdown 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 knots. The anti-skid system provides fast skid recovery response to obtain minimum stop distance under any condition. This is done by fast and continuous control of brake pressure that will compensate for changes in runway condition, aircraft CG distribution, and input overpressure from the brake pedals. During normal braking, the anti-skid system first reduces pressure from the brake when the onset of a skid is detected. The anti-skid system then meters pressure to the brake at a pressure slightly below that at which the onset of skid occurred. However, if the brake pedal metered pressure is greater, the pressure to the brake is allowed to slowly increase until the onset of the skid repeats. The most efficient braking occurs when the brake pressure for each wheel is just below the skid threshold pressure for that wheel. This pressure is generally exceeded because the pilot cannot sense or detect the onset of a skid.

**CAUTION**

If the same braking technique is used with anti-skid off as anti-skid on, locked wheels and blown tires may result.

Locked wheel skid control is activated above 20 knots 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.

**CAUTION**

If anti-skid protection is lost, after touchdown, the failure detection circuit will automatically return the brake system to manual control and light the anti-skid caution lamp. This situation can occur during hydroplaning and may result in blown tires.

**Note**

The failure detection circuit will automatically return the brake system to manual control in the event of an anti-skid malfunction or essential dc electrical failure. However, a malfunction in the failure detect circuit may not revert the brake system to manual control; if loss of braking occurs during the landing roll, the anti-skid system should be turned off to assure deactivation of the anti-skid system.

### Anti-Skid Control Switch

The anti-skid control switch (1, figure 1-1), is marked ANTI-SKID and OFF. Placing the switch to ANTI-SKID 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 FO–5). The lamp will light when the anti-skid switch is in the ANTI SKID position and a malfunction has caused the anti-skid system to become deenergized. The anti-skid caution lamp will light anytime the landing gear is down and the anti-skid switch is in the OFF position.

**Note**

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

## AUXILIARY BRAKE HANDLE

The auxiliary brake handle (7, figure 1-8) is labeled BRAKE. 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 3,150 psi) is ported directly to the brake cylinders, bypassing the metering valves and the anti-skid valves. It should be noted that 1,750 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**

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

## BRAKE HYDRAULIC HAND PUMP

A hydraulic hand pump (figure FO–10), 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 1-8) 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

An amber caution lamp, labeled HOOK DOWN, is located on the main caution lamp panel (figure FO–5) to indicate hook down position.

# AERODYNAMIC DECELERATION EQUIPMENT

## SPEED BRAKE

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

**Note**

- The utility hydraulic system isolation switch must be in the NORM position for normal 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 Switch

A three-position speed brake switch (3, figure 1-1), marked IN, OFF, and OUT, is thumb actuated and slides forward (IN) and aft (OUT). The switch is detented in all positions. When the switch is positioned to OFF, the speed brake is hydraulically locked in its present position. To maintain a constant load on the door and to ensure minimum drag, the switch must remain in the IN position, except during speed brake operation. The speed brake switch is 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 1-11) is marked 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 26-1/2 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. If the flaps start to extend before the slats are full down during extension or if the slats start to retract before the flaps are full up during retraction, the asymmetry system may close the flap/slat drive shutoff valve and 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.

**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 at the existing position.

## 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 FO–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 flat/slat handle (8, figure 1-1) is labeled UP, SLAT DOWN and FLAP DOWN. A manually operated gate (7, figure 1-1) 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 reversed when the handle is moved from one area to another.

**Note**

Allow the flaps to fully retract or the slats to extend completely before reversing the gate. This will expedite reversing flap/slat travel if an asymmetrical condition occurs.

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. With the landing gear handle down, 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. (Refer to "Flight Control System Switch" in this section.) When the gate is reversed 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. Two detent positions are provided in the flap/slat control mechanism to aid in selecting the 15- degree and 25-degree flap positions. Full down position of the flap/slat handle will provide 34 degrees of flap deflection. 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. R

**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 1-17) is 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 1-17) is marked EXTEND and RETRACT and is spring loaded to the center unmarked OFF position. The switch is provided as an emergency method of operating the main flaps and slats in the event of 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 indicator (19, figure FO–3). The flap position indicator provides flap position in degrees. The slat indicator 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, actuators. The actuator gearboxes are mechanically interconnected to ensure positive synchronization (figure 1-10). This protection can be lost in the event of internal failure of either the gearbox or the actuator. 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. Wing position is controlled by 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 (26-1/2) degree position when either the flap/slat handle is out of the UP position or the flaps and slats are not retracted. 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.

## WING SWEEP CONTROL HANDLE

The wing sweep control handle (5, figure 1-11) is spring-loaded to a stowed position under the canopy sill. 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 1-11), are labeled FIXED STORES and WEAPONS. 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 stored lockout control, when ON, prevents the wing sweep handle from being moved aft past the 26 degree position. The weapons lockout control restricts aft movement of the wing sweep handle to 54 degrees. 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 1-11) 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 (19, figure FO–3). The indicator displays the wing position 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 the outside of the scale is provided to assist in setting the wing sweep handle to the

# WING SWEEP AND PYLON SYSTEM

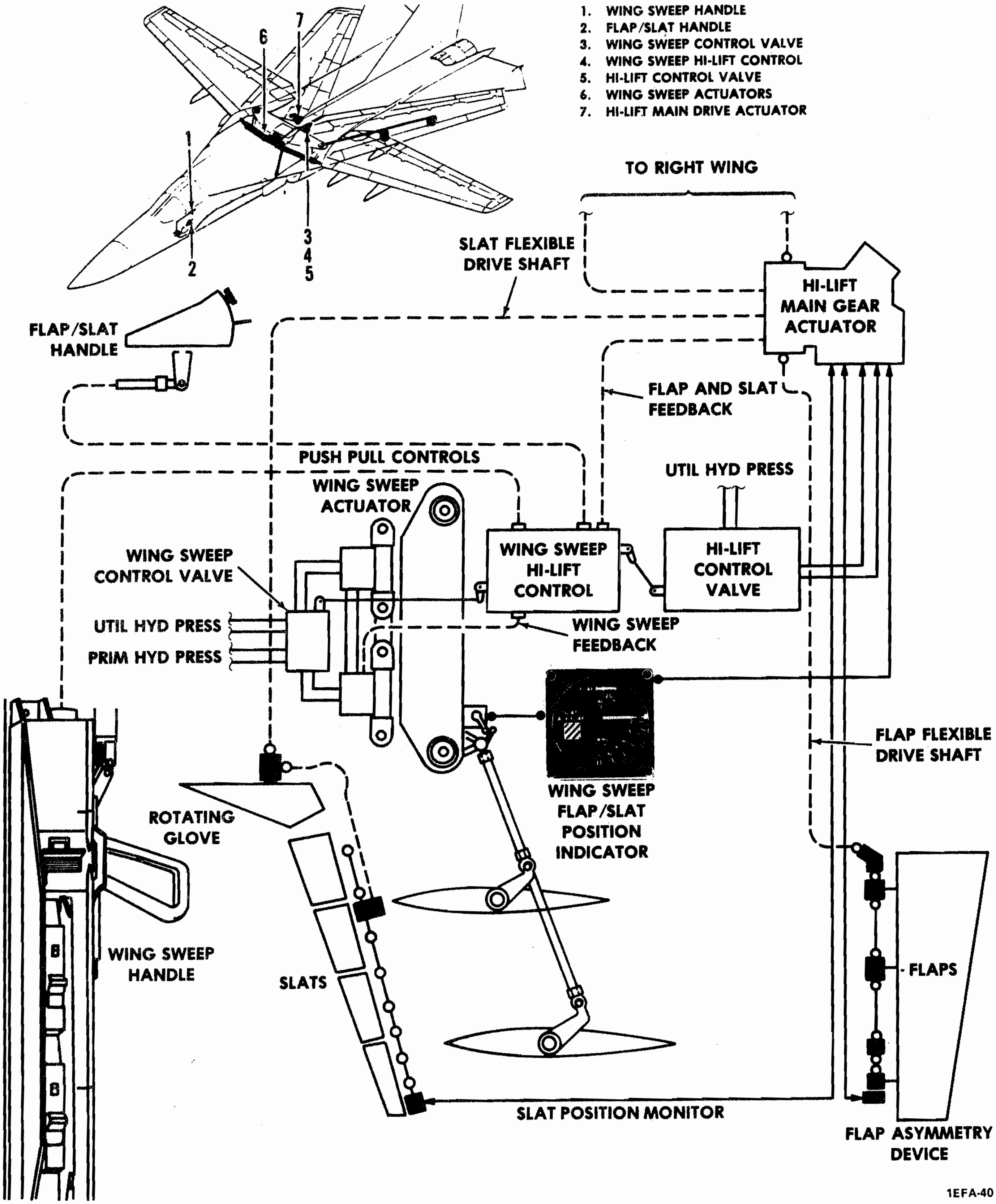



Figure 1-10.

# LEFT SIDEWALL

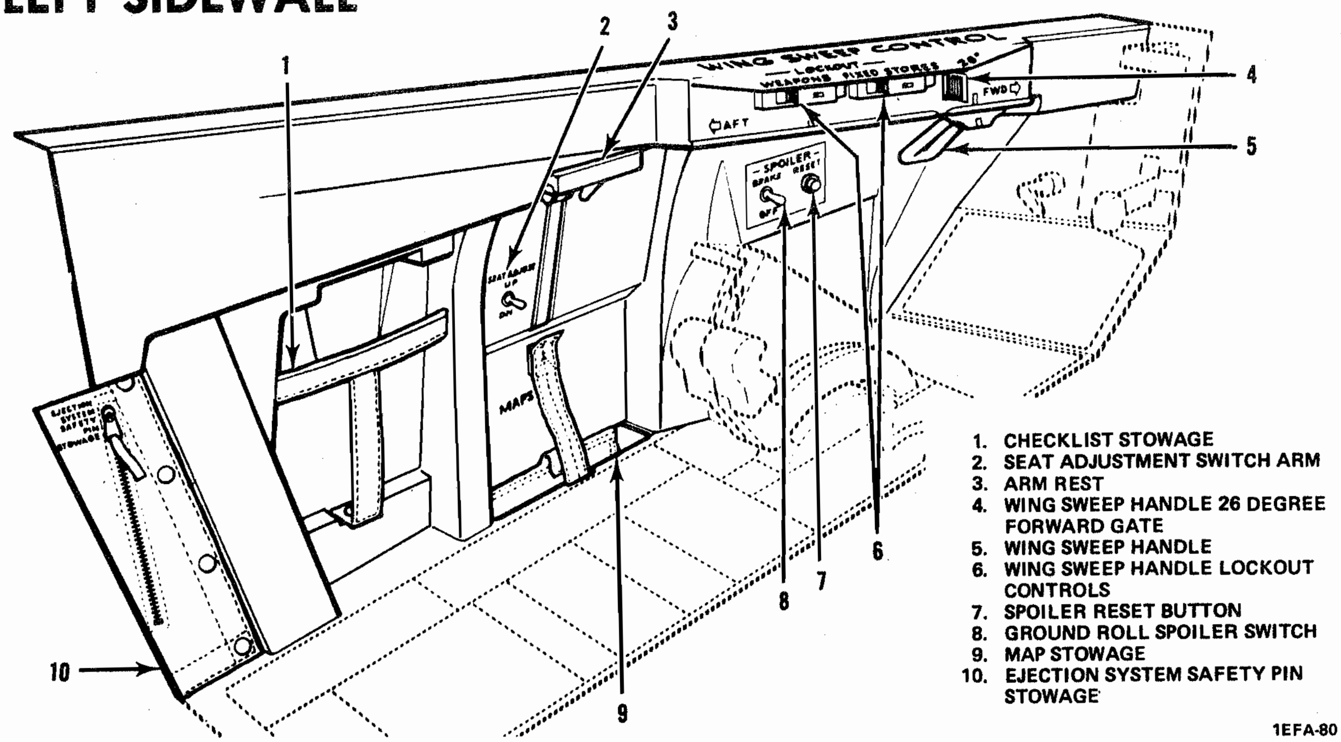



Figure 1-11.

desired position. The position 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.

## ARTIFICIAL STALL WARNING SYSTEM

The artificial stall warning system consists of a rudder pedal shaker (attached to the pilot's left rudder pedal), 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. At all wing sweeps, if the indicated angle-of-attack is greater than 14 degrees, 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, exceeds 18 ($\pm$1). Since 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 ($\pm$1.6) degrees | R |
| Greater than Mach 0.45 but less than Mach 1.25 | 19.7 ($\pm$1.6) degrees | R |
| Greater than Mach 1.40 | 18.8 ($\pm$1.6) degrees | R |

In addition to the above conditions, when the flight control disconnect switch is in the OVRD position and the indicated angle-of-attack is greater than 14 degrees, the stall warning lamp and audible signal will occur. However, the pedal shaker will be inoperative. The stall warning lamp (figure FO-5) is a flashing red lamp located in the left 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 a 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**

- The ground checks only confirm proper activation of the artificial stall warning indications and not the accuracy of system inputs such as angle-of-attack. Erroneous indications will not be obvious until system activation when the aircraft becomes airborne.

- During ground checks, if the flaps are extended 15 degrees or more, the rudder may deflect due to an adverse yaw compensation input when the malfunction and indicator lamp test button is depressed.

## LANDING CONFIGURATION CAUTION SYSTEM

The landing configuration caution system (LCCS) provides an audio and visual indication when the aircraft is not properly configured for takeoff or for low speed, low altitude flight. The visual indication is supplied by an amber caution lamp (figure FO–5) on the left instrument panel. When lighted, the word CONFIG appears on the face of the lamp. The audio indication is supplied by a warning horn that produces a warbling tone in both headsets. The lamp will light and the horn will sound for each of the following conditions:

1. On the ground, when both throttles are advanced beyond minimum cruise power setting and the flaps are less than 15 degrees extended.

2. During flight when the aircraft is below 10,000 feet MSL, with either or both throttles retarded below minimum cruise setting, and:

   - Airspeed is below 300 KIAS and the wings are aft of 26 degrees sweep; or

   - Airspeed is below 220 KIAS and the slats are less than 70 percent extended; or

   - Airspeed is below 200 KIAS and the flaps are less than 15 degrees extended.

The LCCS warning horn can be silenced by depressing the horn silencer button adjacent to the landing gear handle; however, the CONFIG caution lamp will remain on until the aircraft configuration is corrected or the airspeed is increased. Since three warning tones (stall, LCCS, and landing gear) can be present in the headset, the warning systems are designed so that the stall warning tone will override both the LCCS and landing gear warning tones, and the LCCS tone will override the landing gear warning tone. If conditions cause more than one warning tone circuit to be activated at the same time, depressing the horn silencer button once will silence the higher priority tone and the tone next in priority will then sound until the silencer button is depressed again.

## FLIGHT CONTROL SYSTEM

The primary aircraft 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. Depending on wing sweep angle, roll control may be 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 actuators are used to produce control surface movement. The control stick is mechanically and electrically connected to the flight control system. 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 is varied by a self-adaptive system as flight conditions change (gain). Command augmentation, through the pitch and roll dampers, augments the pilot's inputs to provide a near constant relationship between control force and aircraft response throughout the operational envelope. A stall inhibitor system (SIS) and adverse yaw compensation (AYC) improve handing characteristics. In the event of system failures, the flight controls are mechanically connected to hydraulic actuators to ensure aircraft control through all but the very high speed region of the flight envelope. Automatic failure detection and rejection, as well as self-test features, are provided in the pitch, roll, and yaw stability augmentation systems. 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) to maintain the aircraft at a preselected clearance plane above the terrain.

## PITCH AXIS

### Major Components

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 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 FO-11 is a simplified schematic of the mechanical flight control linkage.

PITCH FEEL SPRING

Since stick movement is metering hydraulic fluid rather than physically moving the stabilizers, there is no aerodynamic feedback to the stick. Therefore, a feel spring is used to provide stick centering and feel forces. With both trim actuators centered and the pitch damper centered and OFF, total stick travel from neutral to full aft is approximately 6 1/2 inches, and from neutral to full forward is approximately 4 inches. The force required to move the stick from neutral to full aft ranges from the initial breakout force of 1.7 pounds to 65 pounds. The force required to move the stick from neutral to full forward ranges from 1.7 pounds to 55 pounds.

PARALLEL PITCH TRIM ACTUATOR

The parallel trim actuator relieves pressure on the stick caused by the pitch feel spring by repositioning the center or zero pressure point of that spring. The actuator is normally positioned by the pitch trim button on the control stick and has a range of 10 degrees trailing edge up (TEU) to 8 degrees trailing edge down (TED). (Refer to "Parallel Trim," this section.)

PITCH STICK POSITION TRANSDUCER

This sensor measures the physical position and rate of movement of the stick and changes these measurements into electrical signals, which are sent to the pitch computer.

PITCH COMPUTER

The pitch computer interprets the signal from the stick position transducer and compares it to the actual performance of the aircraft represented by a feedback signal from the aerodynamic sensors. This feedback signal in pitch is a combination of the outputs from the pitch rate gyro and the normal accelerometer (g). If actual aircraft performance is not equal to the command from the stick, the computer sends a signal to the pitch damper demanding more or less horizontal stabilizer deflection. (Refer to "Pitch Computer Logic," this section.) The computer also varies the sensitivity of the command signal to the horizontal stabilizers. (Refer to "Pitch Gain Control," this section.)

PITCH DAMPER SERVO

The pitch damper is an electrohydraulic servo actuator with an authority of $\pm 13$ degrees. The pitch damper servo provides pitch trim, stability augmentation, and command augmentation to the horizontal stabilizers.

SERIES PITCH TRIM ACTUATOR

The series pitch trim actuator operates to maintain full damper authority by automatically keeping the damper centered (nulled). The actuator has a range of 10 degrees TEU to 4 degrees TED. (Refer to "Series Pitch Trim," this section.)

FEEL AND TRIM ASSEMBLY

The feel and trim assembly contains electrical and electromechanical components that manage all, or part of the following functions: pitch parallel trim, pitch series trim, aux pitch trim, roll trim, yaw trim, spoiler system, pitch and roll autopilot interlocks, auto TFR interlocks, TFR fail/fly-up command, takeoff trim, auto flight control system cockpit self-test, pitch and roll control stick steering, AYC, and stall warning logic.

### Pitch/Roll Mixer Assembly

Combined pitch and roll movements of the control stick are transmitted by the linkage of their respective axis 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 FO-11.)

### Horizontal Stabilizer Actuators

The output of the pitch roll mixer is transmitted by two push-pull tubes causing displacement of control valves on the left and right horizontal stabilizer actuators. When the control valves are displaced from neutral, hydraulic fluid is ported to the respective actuator, which results in horizontal stabilizer displacement. 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 degress.

### Power Supply

Power to the pitch damper servo, series trim actuator, parallel trim actuator, stick transducer, pitch computer, feel and trim assembly, and stall inhibitor computer 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.

### Stability Augmentation

Pitch stability augmentation is provided through the pitch damper system to provide aircraft damping and to improve the handling characteristics of the aircraft. The pitch computer differentiates between changes due to aerodynamic disturbances and pilot inputs by comparing aerodynamic feedback to stick position. The aerodynamic feedback signals come from the accelerometers, gyros, and angle-of-attack probes, while the stick position and rate command signals come from the stick position transducer. After comparing these signals, the pitch computer sends an electrical input to the pitch damper to move the control surfaces thus minimizing the effects of aerodynamic disturbances.

### Command Augmentation

The effectiveness of any control surface varies with flight conditions. At low speed and high altitude, several degrees of horizontal stabilizer 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 command augmentation system changes the horizontal stabilizer deflection so that aircraft response ("g" response, or angle-of-attack response at high AOA's) for any given stick input is the same regardless of airspeed or altitude. The command augmentation feature compares the stick position transducer output (degrees/second command) to a feedback signal, which is a combination of the pitch rate gyro output (deg/sec) plus normal accelerometer output ("g" converted to deg/sec). The stick position transducer command is reduced as angle-of-attack increases above eleven degrees angle-of-attack. This feature causes the command augmentation system to gradually change from a "g" command system as angle-of-attack increases. (Refer to "Alpha Limiter," this section.)

**Note**

When the flight control system is in the takeoff and land configuration, the normal accelerometer feedback is removed. Therefore, command augmentation is not available.

During decelerating flight, the command augmentation system drives the pitch damper to increase the horizontal stabilizer deflection required to maintain the commanded "g" without additional stick input. The result is an increase in AOA even though the stick is not moved.

### Stall Inhibitor System

The stall inhibitor system (SIS), through the stability and command augmentation systems, physically limits the maximum attainable angle-of-attack and minimizes sideslip during rolling maneuvers at high angles-of-attack. The system includes a triple-redundant SIS computer, an alpha probe on each side of the nose, and a total aileron transducer. The SIS computer modifies signals to the pitch, roll, and yaw computers. In the pitch axis, an alpha limiter in the SIS computer acts through the pitch damper and series trim systems to provide positive speed stability at angles of attack above 11 degrees and limits the maximum angle-of-attack attainable. (Refer to "Alpha Limiter," this section.) In the roll axis, an unsaturator is provided to minimize the time required for the roll damper to come out of saturation due to large lateral control stick inputs. In the yaw axis, a beta reducer in the SIS computer minimizes sideslip during rolling maneuvers and thus improves rolling performance at angles-of-attack greater than 7 degrees.

**WARNING**

The angle-of-attack limiting is not absolute. Large abrupt aft stick inputs or repetitive push-pull inputs at high angles-of-attack can result in overshoots of angle-of-attack limits and loss of control. Also, large abrupt cross controlling and reversals at near maximum angle-of-attack may force the aircraft into departure from controlled flight.

### Pitch Computer Logic

All pitch command signals enter the pitch computer where they are converted into three signals. (See figure 1-12.) 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 the board's input signal with

# COMPUTER AND DAMPER HYDRAULIC LOGIC

SIGNAL COMPARATOR ACTUALLY MONITORS ALL THREE SIGNAL SELECTOR INPUTS/OUTPUTS

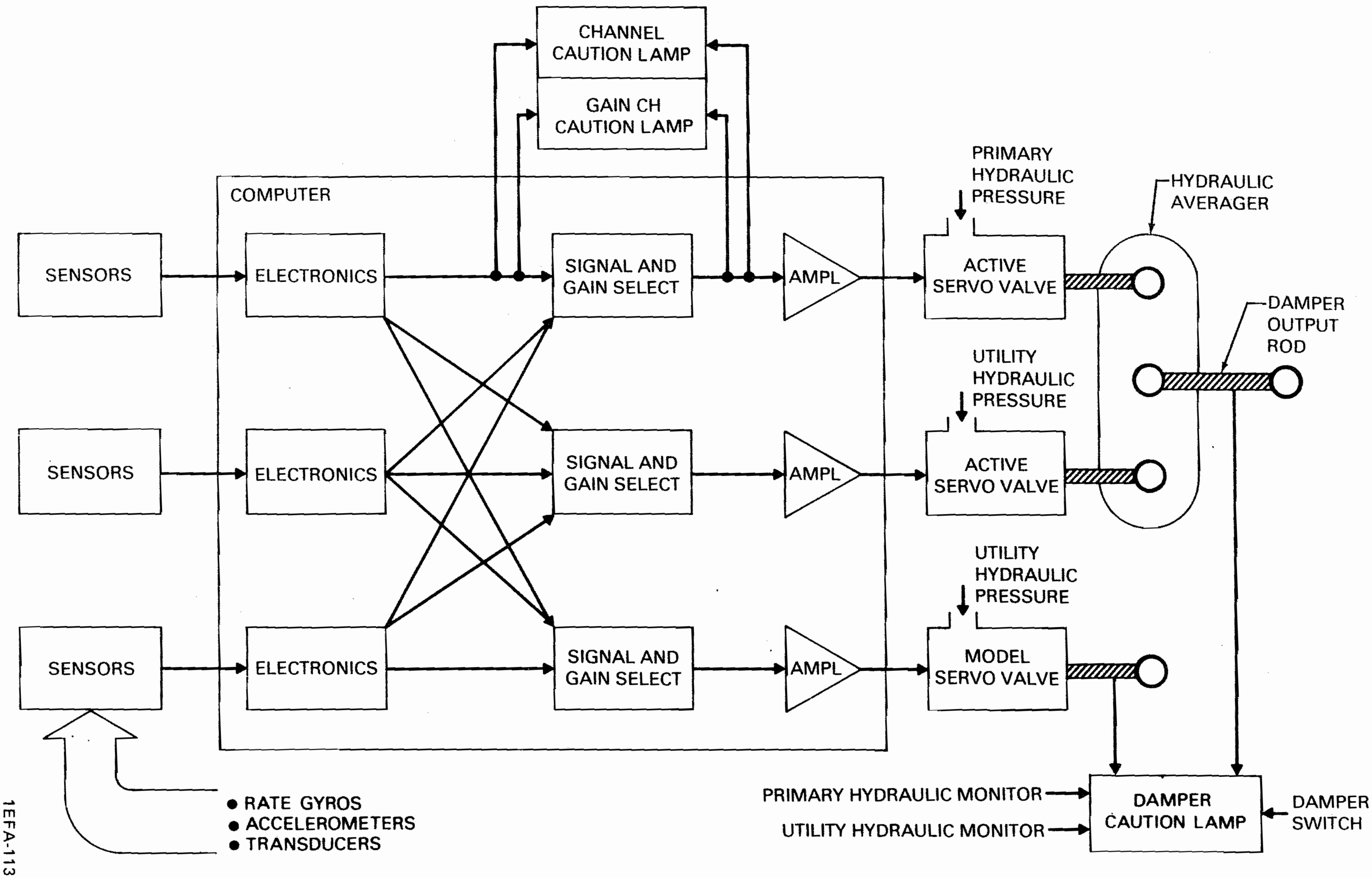



Figure 1-12.

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 may be continued if the pitch channel caution lamp resets. (Refer to caution lamp analysis, Section III.) If the lamp cannot be reset, the first failure was 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 the pitch damper be turned OFF. With the pitch damper OFF, the pitch damper caution lamp will light.

### Signal Amplification

Three servo amplifiers are located downstream of the signal selectors (figure 1-12). Each 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 servo valves supplied by separate hydraulic systems, and a model valve supplied by the utility hydraulic system (figure 1-12).

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.

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.

If hydraulic pressure is not available, the position of the pitch damper servo is indeterminate.

### Pitch Gain Control

The pitch computer varies the command signal to the horizontal stabilizers by continuously monitoring the pitch rate gyro and normal acceleration signals and adjusting the gain of the signal to the damper in response to changing flight conditions. Since the system modifies its gain as a function of its own performance by monitoring aircraft response to changing flight conditions, it is called a self-adaptive gain system. In general the required gain varies inversely with dynamic pressure. Horizontal stabilizer effectiveness is affected by many variables such as Mach number, altitude, wing sweep angle, gross weight, center-of-gravity, and external stores. The damper contribution in response to command augmentation inputs is a function of gain, therefore, a gain increase will compensate for reduced horizontal stabilizer effectiveness by giving more horizontal stabilizer 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 persistance. Also, the gain changer will increase the gain for inputs less than 1.4 cycles per second. 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 to 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.

When the flight control system is in the takeoff and land configuration, the pitch gain is locked at 30 percent.

**Pitch Gain Logic**

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. (Refer to "Pitch Computer Logic," this section.) 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 cannot be reset, a second failure could cause the total gain to go to minimum, maximum, or be unaffected. For this reason, a decrease in airspeed is recommended.

**Pitch Trim System.**

Pitch trim can be effected from one of three inputs; the parallel pitch trim, series pitch trim, and auxiliary pitch trim. 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 operates to maitain full damper authority by automatically keeping the damper centered (nulled). In the autopilot and auto TF modes, there is always a difference between the commanded and actual damper position. Series trim drives to eliminate these steady state errors.

An auxillary pitch trim system is provided as a backup trim system and may be used if a trim failure occurs. The operation of each trim input is determined by: (a) the auxiliary pitch trim switch position, (b) the pitch damper switch position, and (c) the control system configuration. Figure 1-13 summarizes the trim system as a function of these variables.

PARALLEL PITCH TRIM

Parallel pitch trim is designed to relieve pressure on the stick. During normal operation, parallel trim will be at neutral for one "g" flight. The parallel pitch trim actuator is driven from the control stick 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 centered at the takeoff position (zero degrees) by depressing the takeoff trim button. Furthermore, it will center and lock by one of the following means.

- Turning the pitch damper switch to OFF.
- Moving the aux pitch trim switch out of the STICK position.
- Placing the autopilot/damper switch to AUTOPILOT or placing the auto TF switch to AUTO TF.

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/flyup maneuvers.

SERIES PITCH TRIM

The series pitch trim actuator drives the horizontal stabilizers but does not normally drive the control stick. (Refer to "Stick Talk Back," this section.) While the control system is in its normal flight configuration, the series trim will drive in response to a pitch damper position signal. When the pitch damper is displaced, the series trim will drive in a direction to allow the damper to return to neutral. This is called the null mode of operation. The null mode will be in effect unless the pitch damper is turned off, the takeoff and land configuration is established, or auto TF or pitch autopilot is engaged. Displacement of the control stick will command an increase or decrease in aircraft response through the command augmentation feature. If the aircraft response is zero (no pitch rate and one "g" normal acceleration) while the stick is at neutral, then the pitch damper will be at zero, and the series trim actuator will stop. If the horizontal stabilizer 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 horizontal stabilizer required to maintain the aircraft in trim. Because of this action, the stick will be centered for one "g" flight regardless of speed unless 11 degrees angle-of-attack is exceeded. (Refer to "Stall Inhibitor System," this section.)

# 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<br>2. SERIES TRIM ACTUATOR IN DAMPER NULL MODE |
| STICK | DAMPER | TAKEOFF & LAND | 1. STICK TRIM DRIVES PARALLEL PITCH TRIM<br>2. SERIES TRIM LOCKED AT EXISTING POSITION |
| STICK | OFF | CLEAN OR TAKE-OFF & LAND | 1. PARALLEL PITCH TRIM ACTUATOR CENTERS AND LOCKS AT ZERO<br>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<br>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<br>2. SERIES PITCH TRIM ACTUATOR IN DAMPER NULL MODE<br>3. AUXILIARY PITCH TRIM SWITCH DRIVES DAMPER |
| OFF | DAMPER | TAKEOFF & LAND | 1. PARALLEL PITCH TRIM ACTUATOR CENTERS AND LOCKS AT ZERO<br>2. AUXILIARY PITCH TRIM SWITCH DRIVES DAMPER AND ALSO DRIVES THE SERIES PITCH TRIM ACTUATOR |

*FLIGHT CONFIGURATION:
CLEAN – AIRCRAFT NOT IN T.O. & LAND
T.O. & L: (1) SLATS DOWN OR
(2) FLIGHT CONTROL SYSTEM SWITCH–T.O. & LAND AND LANDING GEAR HANDLE DN.

## PITCH TRIM DURING AUTOMATIC CONTROL MODES

| MODE | PITCH TRIM OPERATION |
|---|---|
| AUTO TF | 1. STICK TRIM DEACTIVATED AND PARALLEL PITCH TRIM ACTUATOR CENTERS<br>2. AUXILIARY PITCH TRIM SWITCH MUST NOT BE USED IN THIS MODE<br>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<br>2. AUXILIARY PITCH TRIM SWITCH MUST NOT BE USED IN THESE MODES<br>3. SERIES PITCH TRIM IS DRIVEN BY PITCH AUTOPILOT ERROR SIGNAL |

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★ Figure 1-13.

**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 flight control system switch is in NORM.

The actuator locks at its present position when the control system is switched to the takeoff and land configuration. The series trim drives to its 3.8-degree trailing edge up position when the takeoff trim button is depressed. 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 button. The actuator trim rate is 1.4 degrees per second.

AUXILIARY PITCH TRIM

The auxiliary pitch trim system is operated when the auxiliary pitch trim 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 1-13 defines which of the above 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 damper will be a function of the pitch computer gain.

**Alpha Limiter**

Limitation of angle-of-attack is accomplished by routing the command augmentation and pitch rate gyro signals through the alpha limiter in the SIS computer as shown in figure FO-12. Angle-of-attack signals are used to adjust the gain of these signals. The alpha limiter has no effect on the flight control system at angles-of-attack less than 11 degrees. As angle-of-attack increases above 11 degrees, increased stick force and travel will be required to attain a given angle-of-attack. Thus, for a one "g" slow-down, no stick force will be required up to 11 degrees angle-of-attack. If the slow-down is continued, aft stick force must be held to maintain one "g" flight. The aft stick force required will continue to increase as angle-of-attack increases, and will become more pronounced if 18 degrees angle-of-attack is exceeded. If full aft stick is held, the maximum attainable angle-of-attack will be approximately 20 degrees. This limiting action is caused by a nose down command to the horizontal stabilizers, which causes the aircraft to descend in relation to the power setting. Action of the alpha limiter during maneuvering flight at angles-of-attack greater than 11 degrees allows the pilot to more closely control angle-of-attack by stick force cues rather than constant monitoring of the angle-of-attack indicator. The alpha limiter is active during all flight conditions if the pitch damper is on. It is not active when the pitch damper is off. This function is deactivated when the weight of the aircraft is on the landing gear. Slat extension increases the alpha limiter gain, which improves pitch damping and causes more positive angle-of-attack limiting.

**SIS Computer Logic and Angle-of-Attack Monitoring**

Signals to the pitch, roll, and yaw computers are routed through the SIS computer. In the SIS computer, a probe selector switching logic circuit selects an angle-of-attack signal from one of the two probes. The selector will choose the lowest angle-of-attack value if sideslip is less than 7 degrees. Should a difference of 6 degrees or more exist between the two angle-of-attack probes, and the sideslip angle is less than 7 degrees, the SIS caution lamp will light. If the sideslip angle is greater than 7 degrees, the selector in the SIS computer will select the highest angle-of-attack signal from one of the two angle-of-attack probes, and the SIS caution lamp is inhibited. The probe selector and inhibitor signals are triple redundant. Failure of the selector or inhibitor signals to agree may cause a pitch, roll, or yaw channel caution lamp to light.

**WARNING**

- Lighting of the SIS caution lamp, even though it resets, is a warning to the pilot that he can no longer depend on the alpha limiter to limit angle-of-attack. The angle-of-attack probe may have been damaged or loss of electrical redundancy may have occurred.

- An inoperative alpha probe may result in an uncommanded pitch-down maneuver if sideslip is greater than 7 degrees. If the SIS caution lamp lights, even though it may reset, or if an alpha probe is suspected of being damaged or inoperative, refer to SIS Caution Lamp Analysis. Refer to Section III. Slow the aircraft to damper off limit, turn the pitch damper off, and land as soon as practical using dampers off landing procedures. Refer to Section III.

### Flight Control Automatic Switching

Certain changes occur within the flight control system during different phases of flight. Figure 1-15 describes the configuration for the various conditions.

### Automatic Pitch Control

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

### Pitch Command Limits

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

| INPUT | TEU LIMIT | TED LIMIT |
|---|---|---|
| Control stick | 20 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 |

R

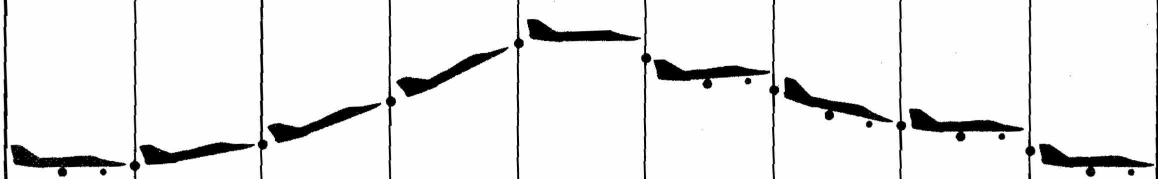
With the pitch damper on, control stick displacement or parallel trim actuator displacement will cause the pitch damper to displace in response to signals from the pitch stick transducer, which usually limits the amount of control stick available. Combined pitch and roll movements of the control stick are transmitted to the pitch/roll mixer assembly where they are converted into left and right horizontal stabilizer actuator command signals. (See figure FO–11.)

**Figure 1-14 deleted.**

# FLIGHT CONTROL AUTOMATIC SWITCHING

CONDITIONS:
- RUDDER AUTHORITY SWITCH - AUTO
- FLIGHT CONTROL SYSTEM SWITCH - NORM
- DAMPER SWITCHES - DAMPER
- FLIGHT CONTROL DISCONNECT SWITCH - NORM

NOTE:
MANUAL FLIGHT CONTROL SWITCHING IS AVAILABLE. REFER TO FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS, THIS SECTION



| FUNCTION | BEFORE TAKEOFF | AIRBORNE LG UP FLAPS >15° | FLAPS <15° | SLATS <70° | CRUISE | LG DOWN | SLATS >70° | FLAPS >15° | DOWN |
|---|---|---|---|---|---|---|---|---|---|
| RUDDER AUTHORITY | FULL | LIMITED | → | → | LIMITED | FULL | → | → | FULL |
| ALPHA LIMITER | OFF | ON | → | → | → | → | → | ON | OFF |
| BETA REDUCER | OFF | OFF | ON | → | → | → | ON | OFF | OFF |
| AYC | OFF | ON | OFF | → | → | → | OFF | ON | OFF |
| PITCH/ROLL GAIN | LOCKED | → | LOCKED | ADAPTIVE | → | ADAPTIVE | LOCKED | → | LOCKED |
| PITCH SERIES TRIM | LOCKED | → | LOCKED | UNLOCKED | → | UNLOCKED | LOCKED | → | LOCKED |
| GAIN LIGHTS | OFF | → | → | → | OFF | ON | OFF | → | OFF |

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Figure 1-15.

## ROLL AXIS

### Major Components

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. These control valves regulate the flow of hydraulic fluid to the actuators, thus causing the horizontal stabilizers to move asymmetrically. Figure FO–11 shows a simplified mechanical schematic of the roll axis linkage and damper system.

ROLL FEEL SPRINGS

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 4-degree roll asymmetry through the pitch/roll mixer to 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 16 degrees of roll asymmetry through the pitch/roll mixer to the horizontal stabilizers. With the roll damper off, the available stick deflection is limited by stick stops located within the cockpit to $\pm 8$ degrees of mechanical command. Figure FO–11 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.

ROLL STICK POSITION TRANSDUCER

This sensor measures the physical position of the stick and changes this measurement into an electrical signal, which is sent to the roll computer and spoiler actuators.

ROLL COMPUTER

The roll computer interprets the signals from the stick position transducer and compares it to the actual performance of the aircraft represented by a feedback signal from the aerodynamic sensors. This feedback signal in roll comes from the roll rate gyros. If actual aircraft performance is not equal to the command from the stick, the computer sends a signal to the roll damper demanding more or less horizontal stabilizer or spoiler deflection. (Refer to "Roll Computer and Damper Hydraulic Logic", this section.) The computer also varies the sensitivity of the command signal to the horizontal stabilizers and spoilers. (Refer to "Roll Gain Control," this section.)

ROLL DAMPER SERVO

The roll damper is an electro-hydraulic servo actuator with an authority of $\pm 6$ degrees. The roll damper servo is used to provide roll trim, stability augmentation, and command augmentation to the horizontal stabilizers. The roll damper servo also provides roll control inputs to the pitch/roll mixer. The roll damper servo is identical to the pitch and yaw damper servos.

PITCH/ROLL MIXER ASSEMBLY

Refer to "Pitch/Roll Mixer," under Pitch Axis, this section.

POWER SUPPLY

The electrical power to the roll damper system is provided through the three computer 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, the roll damper actuator is hydraulically driven to its neutral position, roll trim is removed, and spoiler operation for roll control is not available.

### Stability Augmentation

Roll stability augmentation is provided by redundant roll rate gyros and the roll computer used in conjunction with the roll damper servo to provide aircraft roll damping signals. (See figure FO–13.)

### 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, horizontal stabilizer displacement is proportional to stick displacement. With the roll damper on, the stability augmentation input causes roll damper commands to oppose aircraft toll 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 horizontal stabilizer displacement due to control stick inputs will vary with flight conditions so that variations in the resulting aircraft response will be minimized. The command augmentation signal is routed through the SIS computer. When the roll damper is commanding near its maximum roll, the roll damper unsaturator in the SIS computer causes the damper to become more responsive to reversals in control stick inputs.

### Roll Computer and Damper Hydraulic Logic

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 Computer Logic," "Signal Amplification," and "Pitch Damper Hydraulic Logic," 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. Refer to "Spoilers," this section, for the logic and redundancy features incorporated into the spoiler system.

### Roll Gain Control

The roll computer varies the command signal to the horizontal stabilizers by continuously monitoring the roll rate gyro and adjusting the gain of the signal to the damper in response to changing flight conditions. Since the system modifies its gain as a function of its own performance by monitoring aircraft response to changing flight conditions, 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 cycles per second. 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 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. Certain malfunctions can cause high gains and result in small amplitude roll oscillations. If the gain control circuit does not reduce the gain to the proper value, it can be reset to its minimum value by momentarily cycling the roll damper switch to 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 aircraft response and damping may be noticeably reduced. When in the takeoff and land configuration, the roll gain is locked at maximum.

### Roll Gain Logic

Within the roll computer, three separate circuits compute the required gain. These three circuits provide system redundancy in the same manner as the pitch gain logic. (Refer to "Pitch Gain Logic," this section.)

### 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 mechanical linkage, the control stick does not move as trim is applied. Roll trim is controlled by a trim button located on the control stick. Approximately 8 seconds is required to insert the maximum roll trim command. Roll trim is removed and the roll trim actuator is centered when the roll damper is off.

When the flight control disconnect switch is placed to OVRD, the roll trim actuator inputs to the pitch damper are inhibited. However, the roll trim actuator can still be repositioned by the stick trim button or takeoff trim button. If the flight control disconnect switch is returned to NORM, any previously inhibited roll trim actuator inputs will go to the roll damper.

### Roll Command Limits

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 12 degrees of differential horizontal stabilizer deflection. That is, 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.

### Spoilers

#### SPOILER OPERATION

Each spoiler surface is actuated by a hydraulic servo actuator. The outboard pair of spoiler actuators has extension/retraction pressure supplied by the utility hydraulic system and has lock down pressure supplied by the primary hydraulic system. The inboard pair of spoiler actuators receive extension/retraction 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

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 while airborne. If a spoiler inadvertently extends without being commanded and the aircraft starts a roll, the pilot should 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 control. The spoiler monitor may be reset by depressing the 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. Resetting the spoilers during flight is not recommended, because it is likely that the malfunction will recur and cause another unwanted roll maneuver. 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.

## YAW AXIS

### Major Components

#### MECHANICAL LINKAGE

Rudder pedal movement is transmitted by cables, push-pull tubes and bellcranks to the rudder control valve, which controls the flow of hydraulic fluid from both the primary and utility hydraulic systems to the rudder actuator. Figure FO-14 is a simplified yaw axis mechanical and electrical schematic.

#### YAW VARIABLE FEEL ACTUATOR

The yaw variable feel actuator provides two rudder authorities: full pedal travel of approximately 2.5 inches and $\pm 30$ degrees of rudder deflection or limited authority of approximately 1 inch pedal travel and $\pm 11.25$ degrees of rudder deflection. Full rudder pedal authority is automatically provided when the landing gear handle is placed to DN or when the rudder authority switch is positioned to FULL. Single hydraulic system failure will cause rudder authority to switch to full, refer to "Hydraulic System Failure," Section III.

YAW FEEL SPRING

Rudder feel forces are provided by the yaw feel spring and yaw variable feel actuator. Centering is provided by the yaw feel spring. Rudder feel forces vary from 12 pounds for breakout to 80 pounds for available pedal travel.

YAW TRIM ACTUATOR

Yaw trim is accomplished by an electrically driven actuator, which mechanically positions the rudder linkage. Since the yaw trim actuator is downstream of the yaw feel spring, there is no movement of the rudder pedals as trim is applied. (Refer to "Rudder Trim Switch," this section.)

YAW COMPUTER

The yaw computer receives signals from the lateral accelerometers, yaw rate gyros, adverse yaw compensation (AYC) circuits and SIS computer. When airborne, the yaw computer receives AYC signals and lateral accelerometer signals through the SIS computer if the flaps are extended 15 degrees or more. When the flaps are extended less than 15 degrees, the SIS computer supplies damping signals to the yaw computer and also provides signals to reduce sideslip during roll maneuvers when angle-of-attack exceeds 7 degrees.

YAW DAMPER SERVO

Directional damping is provided by the yaw damper servo. The authority of the yaw damper is +15 degrees. The yaw damper servo is identical to the pitch and roll damper servos.

POWER SUPPLY

Power to the yaw damper is controlled from the computer power switches on the ground check panel.

**Stability Augmentation**

Yaw stability augmentation is provided through a fixed gain redundant yaw damper system with lateral accelerometer and yaw rate signals as inputs to provide continuous yaw damping. To prevent continued opposition to steady state pilot commands, yaw rate input is decreased as a function of time (wash out).

**Adverse Yaw Compensation (AYC)**

AYC is provided to improve turn coordination by augmenting the directional stability with a sideslip to rudder feedback. The SIS computer selects signals from both AOA probes, the beta probe, and roll rate inputs to produce rudder deflection in the direction of roll and proportional to the roll rate. Roll rate feedback is gain adjusted as a function of angle-of-attack. AYC is operational only when the weight is off the main gear and flaps are extended 15 (+3) degrees or more.

AYC FAILURES

AYC failures may result from a sideslip probe or electronic component malfunction. A major input to the AYC is the beta probe, which is located along the aircraft centerline just forward of the nose gear. Should the probe not be free to rotate due to damage, icing, etc., erroneous sideslip information will be fed to the AYC circuits. This erroneous information can cause significant rudder inputs during takeoff or landing. The AYC will not become engaged if the probe senses that sideslip is greater than 6 degrees, thus preventing erroneous rudder inputs. If this condition occurs, the pitch and roll gain changer lamps will be lighted when airborne while the landing gear handle is down.

**Note**

> Once the AYC is engaged, it will not subsequently disengage, even if the probe senses sideslip greater than 6 degrees. To terminate AYC, the flight control disconnect switch must be placed to OVRD.

AYC can be checked on the ground by depressing the damper servo and flight control master test buttons while the sideslip probe is positioned by the ground observer. If an electronic failure is present at lift-off, the yaw channel caution lamp will light but can be reset after flap retraction. It will again light during flap extension and cannot be reset until weight is on the main gear or the flaps are retracted. Small rudder transients at lift-off or upon airborne flap extension may occur. These transients are due to improper nulling of the AYC circuits.

**Beta Reducer**

Sideslip reduction is accomplished by the beta reducer function of the SIS computer. The beta reducer uses angle-of-attack data (See "SIS Probe Selector Logic," this section.) and inputs from the yaw rate gyros, roll rate gyros, lateral accelerometers, and total aileron transducer. Beta probe data is not used to compute beta reduction. These inputs provided to the SIS computer improve rolling performance by reducing sideslip at angles-of-attack above 7 degrees. The beta reducer gains are zero until angle-of-attack exceeds 7 degrees and the gains are maximum at 16 degrees angle-of-attack.

### Yaw Axis Automatic Switching

See figure 1-15 for yaw axis automatic switching.

## FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS

### Control Stick

The control stick grip (figure 1-16) contains a trim button, a reference engage button, a nose wheel steering/air refueling button, a weapons release button, a gun trigger, and an autopilot release lever. The control stick also serves as a means of actuating the crew module bilge/flotation bag inflation pump. (Refer to "Crew Module Escape System," this section).

### Rudder Pedals

The rudder pedals control the rudder, nosewheel steering, and operate their respective wheel brakes.

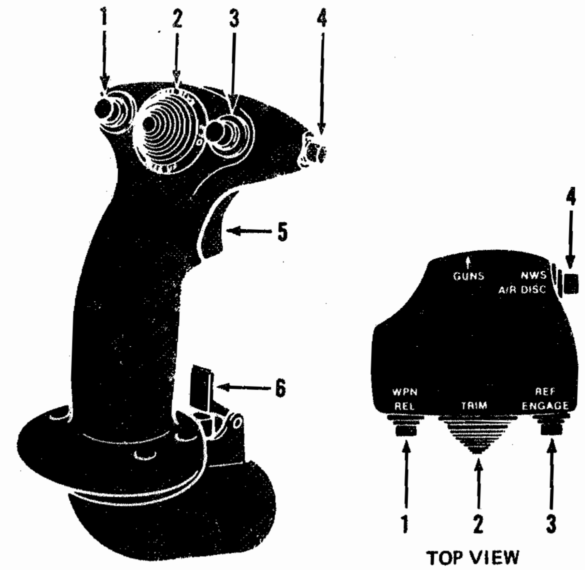
### Trim Button

A trim button (46, figure FO–3) on the control stick grip is 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 positions the horizontal stabilizer surfaces symmetrically, while moving the button LWD or RWD positions the horizontal stabilizer surfaces asymmetrically. Refer to "Pitch Trim System" and "Roll Trim," this section.

### Auxiliary Pitch Trim Switch

The auxiliary pitch trim switch (4, figure 1-17), is marked STICK, NOSE DN, NOSE UP, and OFF. When the switch is in the STICK position, pitch trim can be commanded by the trim button on the control stick. When the auxiliary pitch trim switch is placed to the center (OFF) position, the control stick trim buttons is deactivated in pitch and the 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 1-13.

## CONTROL STICK




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

* NOT USED

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Figure 1-16.

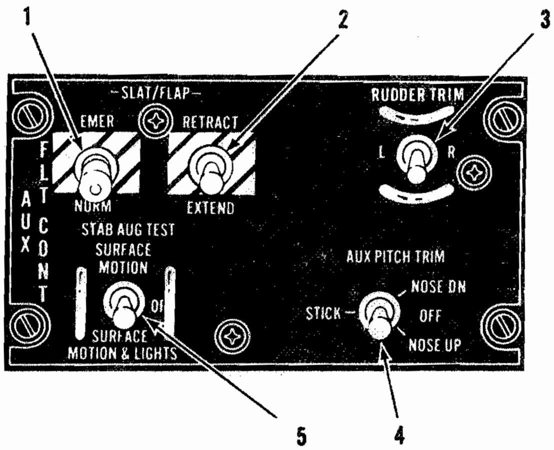
### Rudder Trim Switch

A rudder trim switch (3, figure 1-17) marked L and R, 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 or a maximum deflection of 10.25 degrees is obtained. The trim rate is 2.3 degrees per second.

### Takeoff Trim Button

When the takeoff trim button (46, figure FO–3) 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 a 3.8 degrees trailing edge up. The button also functions during airborne operation.

## 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

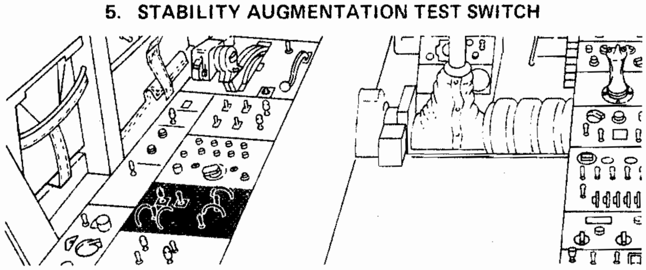



**Figure 1-17.**

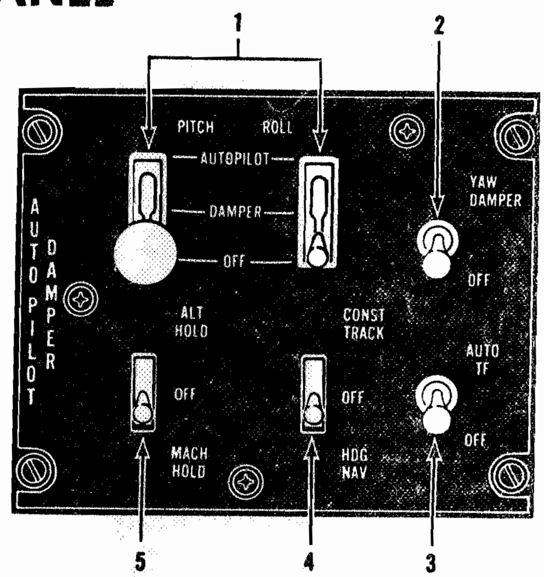
### Autopilot/Damper Switches

The pitch and roll autopilot damper switches (1, figure 1-18) are three-position switches marked AUTOPILOT, DAMPER and OFF. These switches are solenoid held in the AUTOPILOT position and are spring-loaded to the DAMPER position. The yaw damper switch (2, figure 1-18) is a two-position switch marked DAMPER and OFF. The three switches are lever locked in the OFF position. The top 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 damper on. Placing either the pitch or roll switch to AUTOPILOT will engage autopilot attitude stabilization. Placing a switch to OFF disengages the damper and causes the damper caution lamp to light. Refer to "Autopilot System," this section.

### Damper Reset Button

The damper reset button (3, figure 1-19) is a pushbutton labeled DAMPER RESET. When the button is depressed, the pitch, roll and yaw damper caution lamps and their channel caution lamps on the main caution lamp panel will go out. Also, the dampers and their 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, and the SIS caution lamp.

## 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

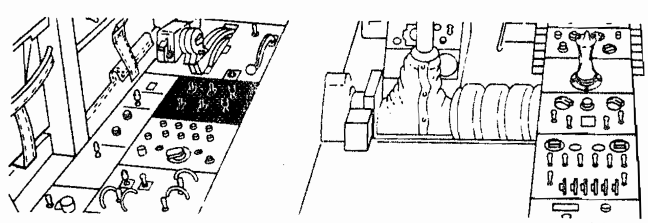
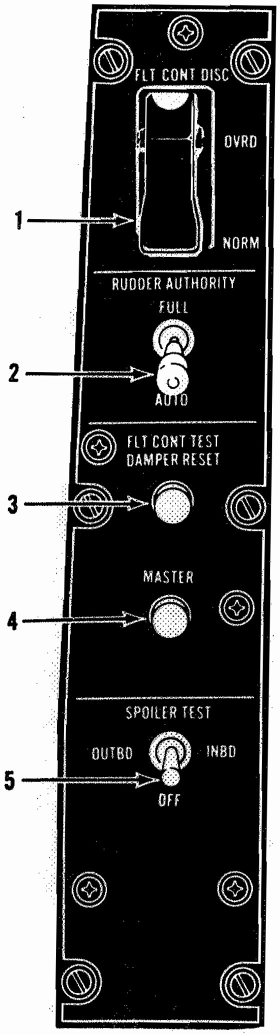



**Figure 1-18.**

# FLIGHT CONTROL TEST PANEL




1. FLIGHT CONTROL DISCONNECT SWITCH
2. RUDDER AUTHORITY SWITCH
3. DAMPER RESET BUTTON
4. FLIGHT CONTROL MASTER TEST BUTTON
5. SPOILER TEST SWITCH

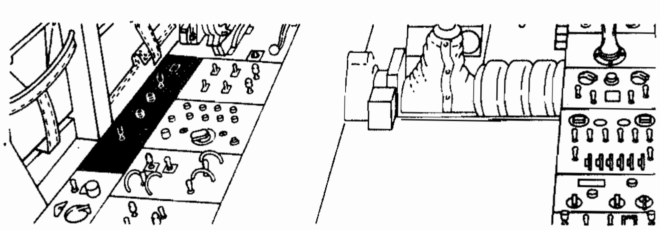


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**Figure 1-19.**

### Rudder Authority Switch

The rudder authority switch (2, figure 1-19) is marked FULL and AUTO. Placing the switch in the AUTO position causes rudder authority to be controlled by the landing gear handle. When the rudder authority switch is in the FULL position, full rudder authority is available regardless of the position of the landing gear handle.

### Flight Control System Switch

The flight control system switch, located on landing gear control panel (12, figure FO–3) is marked NORM and T.O. & LAND. In the NORM position, slat extension determines the configuration of the flight control system. The T.O. & LAND position is used in the event of a system malfunction to override the slat position signal and place the flight control system in the takeoff and landing configuration. The flight control system switch is interlocked with the landing gear handle so that the switch position has no effect unless the landing gear handle is down. When the flight control system switch is placed to T.O. & LAND with the gear handle down or when the slats are extended beyond 70%, the following takeoff and landing functions are automatically provided: the TFR signals are locked out, the pitch and roll computer gains are locked, pitch accelerometer signals are locked out, and the pitch series trim actuator is locked at its present position. The flight control system switch has no effect on rudder authority or adverse yaw compensation.

**Note**

> The 70 percent slat switches are located on both the right and left slats. The right switch provides data for automatic flight control system switch functions. Both switches are used in the flap/slat asymmetry monitoring circuit.

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. 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 flight control system switch to T.O. & LAND with the landing gear handle down, will place the control system switch in the takeoff and land configuration. The gain changer lamps will go out unless a malfunction is present or unless the flight control disconnect switch is in the OVRD. Failure of the lamps to go out with the flight control disconnect switch in the OVRD position is caused by inhibiting adverse yaw compensation. 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.

**Flight Control Disconnect Switch**

The flight control disconnect switch (1, figure 1-19) is 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. Auxiliary pitch-trim inputs to the pitch damper
4. TFR climb/dive commands to the pitch damper
5. Adverse yaw compensation, and pedal shaker commands.

Positioning the flight control disconnect switch to OVRD will light the roll and pitch gain changer caution lamps, provided the control system is in the takeoff and land configuration. These lamps light due to the override action of the switch on the adverse yaw compensation and pedal shaker system.

Also, the reference not engaged caution lamp will light when the switch is placed to OVRD. 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 indicated angle-of-attack exceeds 14 degrees. Functions not affected by the flight control disconnect switch are:

1. Stability augmentation
2. Series and parallel trim while the auxiliary pitch trim is in STICK
3. TFR fail fly-up maneuver.
4. TFR climb/dive commands to the pitch series trim actuation.

**Flight Control Master Test Button**

Depressing the flight control master test button (4, figure 1-19) accomplishes the following:

1. Provides power to the flight control test switches and buttons on the ground check panel.
2. Provides power to the CADC test switch on the ground check panel.
3. Provides power to the stability augmentation test switch on the auxiliary flight control panel.
4. Provides power to the spoiler test switch on the flight control switch panel.
5. Lights the AUTO COWL caution lamp.
6. Unlocks series trim, causing possible horizontal stabilizer drift.
7. When used in conjunction with the damper servo button, enables SIS & AYC ground checks.

**Stability Augmentation Test Switch**

The stability augmentation test switch (5, figure 1-17) is marked SURFACE MOTION, SURFACE MOTION & LIGHTS and is spring loaded OFF. When used in conjunction with the flight control master test button, this switch provides a means of ground checking the stability augmentation system.

**Spoiler Reset Button**

The SPOILER RESET button (7, figure 1-11) resets the spoiler monitor when a malfunction has caused a pair of spoilers to be voted out and locked down.

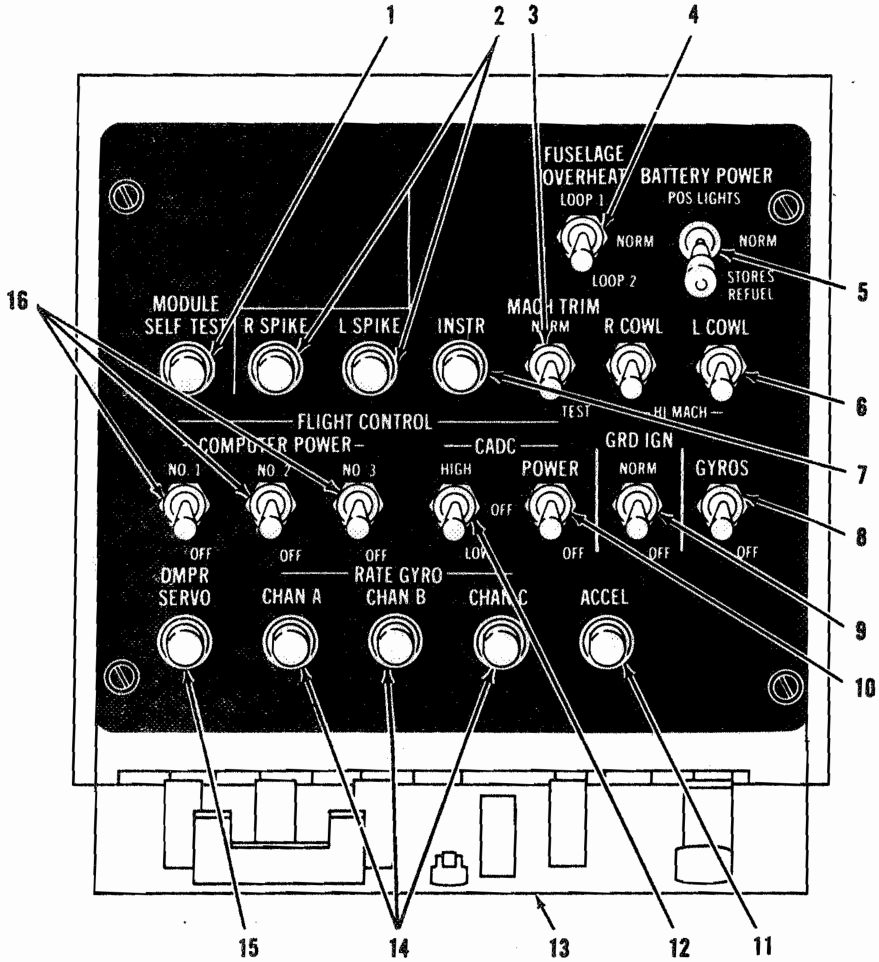
**Spoiler Test Switch**

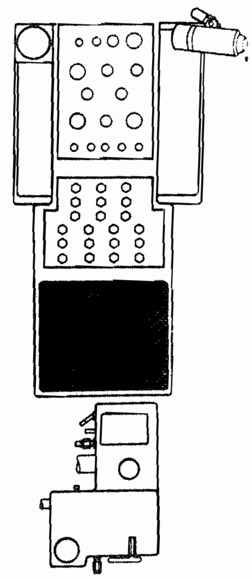
The spoiler test switch (5, figure 1-19) is marked OUTBD, OFF and INBD. The switch is spring-loaded to OFF. The switch is used in conjunction with the flight control master test button to ground check the operation of the spoilers. With the switch in OUTBD, depressing the master test button will cause the outboard pair of spoilers to extend momentarily and then be locked down by the spoiler monitor, 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.

**Accelerometer Test Button**

The accelerometer test button (11, figure 1-20) is used for maintenance ground check.

# GROUND CHECK PANEL



1. ENGINE FIRE DETECTION SYSTEM MODULE SELF TEST BUTTON
2. SPIKE TEST BUTTONS (2)
3. MACH TRIM TEST SWITCH
4. FUSELAGE OVERHEAT TEST SWITCH
5. POSITION LIGHTS/STORES REFUEL BATTERY POWER SWITCH
6. TRANSLATING COWL TEST SWITCHES
7. INSTRUMENT TEST BUTTON
8. AFRS POWER SWITCH
9. GROUND IGNITION CUTOFF SWITCH
10. CADC POWER SWITCH
11. ACCELEROMETER TEST BUTTON
12. CADC TEST SWITCH
13. GROUND CHECK PANEL ACCESS DOOR
14. RATE GYRO TEST BUTTONS (3)
15. DAMPER SERVO BUTTON
16. COMPUTER POWER SWITCHES (3)

1EFA-54

Figure 1-20.

### Computer Power Switches

The computer power switches (16, figure 1-20) are labeled NO 1, NO 2, and NO 3. 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 switches from the essential bus. When the switches are ON, and electrical power is on the aircraft, 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 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 1-20) is used for maintenance ground checks and in conjunction with the flight control master test button, to perform ground tests of the SIS and AYC.

### Rate Gyro Test Buttons

The three rate gyro test buttons (14, figure 1-20) are used for maintenance ground checks.

### Control Surface Position Indicator

The control surface position indicator (27, figure FO–3) has three separate sets of indicators which provide indications of the positions of the spoilers, rudder, and 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, graduated in 5 degree increments. The trailing edge position of the horizontal stabilizers is indicated by two pointers, marked L and R, on a scale, 30 degrees up and 20 degrees down, 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. Indicator tolerance is $\pm 2$ degrees.

### Takeoff Trim Indicator Lamp

A takeoff trim indicator lamp (figure FO–5) 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 FO–5). 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. Refer to "Pitch Computer Logic," this section. 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 and/or loss of the TF fail/fly up capability. For these reasons, TF operation is not recommended and autopilot operation should be monitored closely if a nonresettable yaw channel lamp is lighted.

### Pitch, Roll, 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 FO–5). Lighting of any one of the lamps indicates that a malfuncion 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. Refer to "Damper Hydraulic Logic," this section.

### Spoiler Caution Lamp

The spoiler caution lamp, located on the main caution lamp panel (figure FO–5), is provided to indicate when a malfunction in the spoiler circuitry has occurred causing a symmetric pair of flight control spoilers to be locked down. 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 FO–5). 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. Depressing the damper reset button will reset the lamp for a temporary error. Lighting of both lamps simultaneously will normally indicate a disagreement between the position of the flight control system switch, the position of the slats and the configuration of the flight control system. This will occur while airborne when the landing gear handle is in the DN position and the flight control system is not in the takeoff and land configuration. This will also occur when the landing gear handle is in the UP position and the flight control system is in the takeoff and land configuration when not commanded. When the flight control system is in the takeoff and land configuration, both gain changer lamps will be lighted and the AYC will not engage until the sideslip probe senses a sideslip angle less than 6 degrees.

### Rudder Authority Caution Lamp

The rudder authority caution lamp (figure FO–5) is located on the main caution lamp panel. Lighting of the lamp indicates the rudder authority actuator is not in the position commanded by the position of the landing gear handle. The lamp will light when the landing gear handle is down if the rudder does not switch to full authority, or when the landing gear handle is up if the rudder does not switch to limited authority. The caution lamp circuit is equipped with a 5 second time delay relay to prevent nuisance activations of the lamp during normal cycling of the system. If the lamp blinks during configuration changes, a malfunction is indicated.

### Stall Inhibitor System (SIS) Caution Lamp

The SIS caution lamp, located on the left status indicator (figure FO–5) will light to indicate failures in the angle-of-attack data utilized by the alpha limiter and beta reducer portions of the SIS. Should the physical difference between the left and right angle-of-attack transmitters be greater than the equivalent of 6 degrees angle-of-attack, the lamp will light, provided the side slip is less than 7 degrees. In this case, a stuck or binding probe is indicated. The lamp will also light if the angle-of-attack signal being used by one of the three SIS branches differs by more than 6 degrees from the signal being used by the other two. This would indicate a loss in redundancy for the alpha limiter and beta reducer. The pilot may not be able to distinguish between a damaged probe problem and an electronic failure; therefore, SIS should be considered unreliable. The SIS caution lamp is inhibited when weight is on both wheels, except during the pre-taxi SIS checks. The lamp may light when electrical power is first applied to the system. This condition is normal, provided the lamp is resettable.

**Note**

The SIS caution lamp is inhibited when angle of sideslip from the Beta probe is greater than 7 degrees.

## FLIGHT CONTROL SYSTEM OPERATION CHARACTERISTICS

### 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, 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 FO–12. 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. Since pitch and roll inputs travel through the pitch/roll mixer assembly, a high frequency input from the roll gyros may result in stick talk back in the pitch axis. In this case, turning the roll damper off may stop the stick talk back. The output rods from the pitch/roll mixer assembly control the left and right horizontal stabilizer actuators (refer to figure FO–12). 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.

## PRETAXI 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. Fore and aft motion of the control stick with the pitch damper on may 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 Normal Procedures, 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 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 Normal Procedures, Section II. The damper reset button is used to reset all lamps after the surface motion and lights check is complete.

### Stall Inhibitor System Check

The purpose of the stall inhibitor system check is to ensure that the alpha limiter and beta reducer circuits in the SIS computer are responding to angle-of-attack and control stick inputs without any malfunctions. The check is performed with the flight control master test button and the damper servo button depressed and held while the angle-of-attack probes are held in the required positions. The absence of caution lamps is verified when both probe slots are up. Surface motion and no caution lamps are verified when both probe slots are down.

The AYC check provides a method to check the AYC system on the ground. The flaps must be extended greater than 18 degrees to ensure AYC operation. The test is accomplished by depressing and holding the damper servo and flight control master test buttons while the ground observer positions the sideslip probe. The rudder should not respond until the probe is moved to near its neutral position. Once the probe is within this range, rudder deflection will respond to probe movement. The yaw channel caution lamp should not come on during the AYC check.

### 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.

## 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 1-21) and computes command signals to the pitch and roll dampers. 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 inertial 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 caution 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 forward or aft to 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 not engaged, caution 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 engaged caution lamp will go out. If an auto pilot submode is selected, the reference engage button must be depressed momentarily to engage submodes and turn off the reference not engaged caution lamp. The attitude limits are $\pm 30$ degrees in pitch and $\pm 60$ 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 $\pm 30$ 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

#### Mode Selection

The autopilot Mode switches are solenoid held to the selected mode position. They may be turned off by manually repositioning the switches to OFF or by momentarily depressing the autopilot release lever on the control stick. If a mode other than basic attitude stabilization is selected, the reference engage button on the 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 inertial 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, 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.

# AUTOPILOT-SUBSYSTEMS TIE-IN

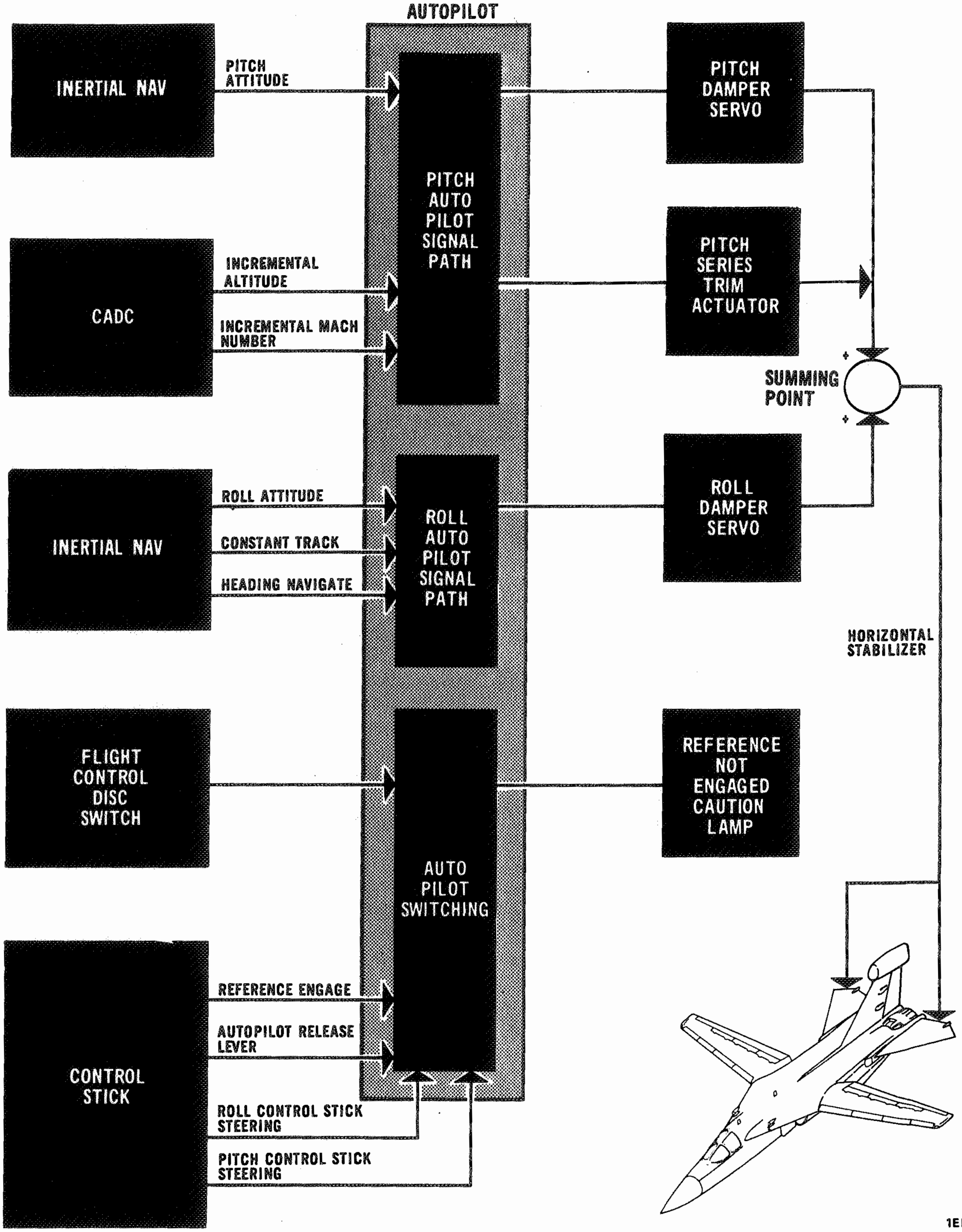



Figure 1-21.

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

Mach hold 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 2,000 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**

- Do not use the pitch autopilot while the aircraft is in takeoff and land configuration (slats down or control system switch in T.O. & LAND with gear handle down). To do so may cause pitch transients when the autopilot is disengaged. R R R R
- Do not use the autopilot in the Mach or altitude hold mode during operation in the transonic flight region between 0.92 and 1.1 Mach.

The control stick will be centered and the pitch functions of the stick trim button will be inoperative when the 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 degrees, the roll synchronizer signal will cause the aircraft to roll to wings level; if the roll angle is greater than 3.5 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 (HDG NAV) is selected, the aircraft will steer to the NC destination for all ISC modes except manual course. If manual course is selected on the ISC, heading navigation will maintain the course manually selected on the HSI. For a discussion of the symbology and displays in each of the steering modes refer to "Instruments," this section, and figure FO–15. In the yaw computer the error signal from the inertial 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 ±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 a 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 inertial nav system to maintain the ground track existing at the time the mode was engaged.

## 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 $\pm30$ degrees or roll attitude exceeds $\pm60$ 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 the 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 the 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 staiblized, the autopilot will hold the aircraft course within $\pm1$ degree of the steering error received from the inertial nav system. Depressing the autopilot release lever on the control stick will return the constant track/heading nav selector switch to OFF, and the roll autopilot switch to DAMPER.

**Note**

If heading navigate submode is used, caution should be exercised for the following reasons:

- Roll rate capability is less than 8 degrees per second, which is too slow 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.

### 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 $\pm30$ 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 indictions.

**WARNING**

The roll autopilot authority may be insufficient to hold the wings level with a large asymmetric wing loading. In this condition, the aircraft may roll off the referenced heading and altitude without disengaging the submodes or lighting the reference not engaged caution lamp.

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 $\pm$60 feet or speed within $\pm$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. Inertial 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.

## CONTROLS AND INDICATORS

### Constant Track/Heading Nav Mode Selector Switch

The constant track/heading nav mode selector switch (4, figure 1-18) is marked CONST TRACK, OFF and HDG NAV. The switch is solenoid held 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 1-18) is marked ALT HOLD, OFF, and MACH HOLD. The switch is solenoid held to ALT HOLD or MACH HOLD and is spring-loaded to OFF.

### Reference Engage Button

A reference engage button (3, figure 1-16), marked REF ENGAGE, is located on the control stick grip. When any autopilot mode is selected, other than attitude stabilization, the button must be momentarily depressed before the mode will engage.

### Autopilot Release Lever

The autopilot release lever (6, figure 1-16) permits the pilot to disengage all 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. When the aircraft is being flown on TFR, the TFR commands to the flight controls can be interrupted by depressing and holding the autopilot release lever. If in auto TF operation, the reference not engaged caution lamp will light. The commands will resume when the lever is released unless a mode other than TF is selected.

### Reference Not Engaged Caution Lamp

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

1. The autopilot/damper switches are in the AUTOPILOT position and control stick steering is being used.
2. 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.

3. Either TFR channel mode selector knob is in the TF position and the auto TF switch is OFF.
4. The auto TF switch is in AUTO TF and neither TFR channel mode selector knob is in the TF position.
5. Autopilot release lever is depressed during auto TF operation.

**Note**

For TF modes, the reference not engaged caution lamp will light when the TF mode on the TFR is selected and the auto TF mode is not controlling the aircraft.

6. The flight control disconnect switch is placed to the OVRD 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 raw data consists of 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 CADC is equipped with a failure monitoring system. Should a computing function fail, a caution lamp on the main caution lamp panel will light. If a failure occurs 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 goods 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 respectively. 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. Inertial nav system (pressure altitude, pressure altitude rate, true airspeed).
10. True airspeed indicator (true airspeed).
11. Terrain following radar (true wing angle-of-attack, true airspeed, and angle-of-attack failure monitor).
12. Angle-of-attack indexer (true wing angle-of-attack).
13. Environmental control (indicated airpseed, true air temperature).
14. Flight director (pressure altitude and pressure altitude failure monitor).
15. Landing gear warning (indicated airspeed, pressure altitude).
16. Marker beacon (pressure altitude).
17. Stall warning system (true wing angle-of-attack).
18. Automatic translating cowl system (Mach number and Mach failure monitor).

## CADC POWER SWITCH

The CADC power switch (10, figure 1-20) is marked POWER and OFF. When the switch is in OFF, 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, power is supplied to the CADC and the maximum safe Mach assembly.

## CADC TEST SWITCH

The CADC test switch (12, figure 1-20) is marked HIGH, OFF, and LOW. 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 and LOW positions of the switch are used in conjunction with ground maintenance tests.

**Note**

When the CADC test switch is used, the inertial 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.

## CADC CAUTION LAMP

The CADC caution lamp (figure FO–5), 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 CADC 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. A power failure to the MSMA will cause the CADC 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 attitude director indicator (ADI) whenever the flight instrument reference select switch is in the AUX position or in event of malfunction of the inertial 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 inertial 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 high 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 the 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. Unreliable AFRS attitude and gyro fast erection are indicated by the auxiliary attitude (AUX ATT) caution lamp 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.

## CONTROLS AND INDICATORS

### Flight Instrument Reference Select Switch

The flight instrument reference select switch (1, figure 1-3) is marked PRI and AUX. Placing the switch to the PRI (primary) position supplies pitch, roll and heading information from the inertial nav system to the following subsystems.

- Autopilot
- Attitude Director Indicator
- Horizontal Situation Indicator
- Flight Director Computer
- Terrain Following Radar
- Nav Radar

Placing the switch to the AUX position supplies pitch, roll and heading information from the AFRS to all the above subsystem except autopilot and the nav radar. The nav radar gets roll information only. Selecting AUX will also light the primary attitude caution lamp. If the inertial nav system is operating normally, selecting AUX will cause the nav 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 AFRS power switch (8, figure 1-20) is marked GYROS and OFF. Placing the switch to GYROS supplies power to the AFRS, the BDHI and the turn and slip indicator on the ADI. Placing the switch to OFF de-energizes these components.

### AFRS Gyro Fast Erect Button

The AFRS gyro fast erect button (9, figure 1-3) 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 indicator returns to normal. Whenever the fast erect button is depressed, the displacement gyroscope erects at a rate of approximately 12 degrees per minute.

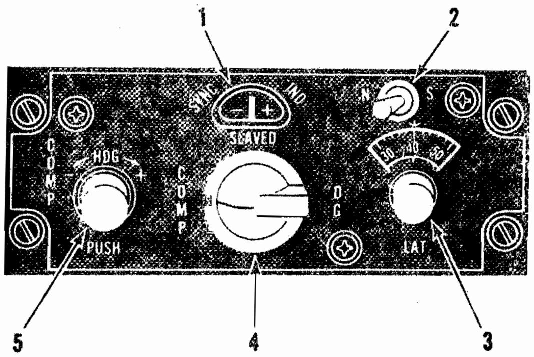
**Note**

The button should be used only when the aircraft is in level flight to prevent inducing errors into the system. The AUX ATT caution lamp will be lighted while the button is depressed.

### Compass Mode Selector Knob

The compass mode selector knob (4, figure 1-22) is used to select the mode of operation of the AFRS 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

## COMPASS CONTROL PANEL




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

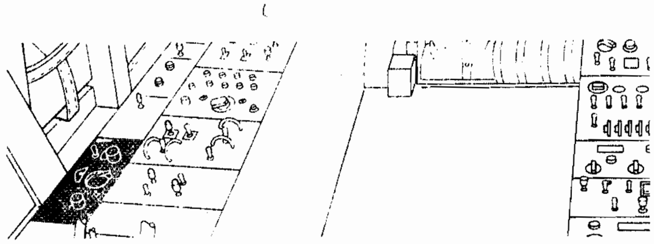



**Figure 1-22.**

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 AFRS.

**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 (3, figure 1-22) 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 the directional gyro mode and improves heading accuracy when operating in the slaved mode for latitudes up to 65 degrees.

### Heading Set Knob

The heading set knob (5, figure 1-22) 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.

### Hemisphere Selector Switch

The hemisphere selector switch (2, figure 1-22) is 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 (1, figure 1-22) indicates whether or not the AFRS gyro and remote compass are synchronized, when operating in the slaves 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 deactivated when operating in the directional gyro or compass mode.

### Auxiliary Attitude (AUX ATT) Caution Lamp

The auxiliary attitude caution lamp (figure FO–5), located on the main caution lamp panel, will light if attitude information from the AFRS becomes unreliable. The lamp will also light during initial erection and when the fast erect button is depressed. When lighted, the amber letters AUX ATT are visible. Should the lamp light and remain on, the flight instrument reference select switch should be positioned to the PRI position.

**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. 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 AFRS 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. Aircraft acceleration/ deceleration reduces the gyro pitch erection rate to one-fourth of the normal erection rate. 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 ensure 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 ensure 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.

If the ADI is receiving AFRS attitude information, the self-contained attitude indicator, 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 resynchronize 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), the crew module q sensor, and the translating cowl redundant pressure 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.

**Note**

> Differences will exist between the primary and secondary instrument readings because the standby instrument readings 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 is located on the crew compartment aft bulkhead (16, figure FO–2). The indicator 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 degrees 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 degree C.

### Total Temperature Caution Lamp

The total temperature caution lamp, located on the lower caution lamp panel (figure FO–5), 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 (figure FO–5), located on the left 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 (28, figure FO–3) provides a digital readout of true airspeed. The instrument displays true airspeed on a servo-driven 4-digit counter within the range of 0-1,750 knots. The indicator is operated by electrical signals from the CADC. The true airspeed indicator is not reliable when the CADC caution lamp is lighted.

## STANDBY INSTRUMENTS

The standby instruments include the airspeed indicator, altimeter, vertical velocity indicator, magnetic compass, self-contained 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 (34, figure FO–3) 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 (53, figure FO–3) 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.
- Aircrews should not attempt to calibrate the standby altimeter to correct for an out-of-tolerance condition.

### Vertical Velocity Indicator

The vertical velocity indicator (37, figure FO–3) provides rate of climb and descent information. The instrument operates on static pressure from the pitot-static system.

### Magnetic Compass

The magnetic compass (55, figure FO–3) provides magnetic heading. A deviation correction card for the compass is located below the center of the glare shield.

### Self-Contained Attitude Indicator (SCAI)

The SCAI (36, figure FO–3) provides backup attitude information. The indicator, which is equipped with an integral vertical gyro, displays pitch and roll information on an attitude sphere in relation to a miniature aircraft. In the event of electrical power failure, an OFF warning flag will appear on the upper left face of the indicator. A knob in the lower right corner of the indicator is provided for pitch adjustment and gyro caging. When the cage knob is pulled out and rotated fully clockwise, the SCAI will remain caged and the off flag will remain in view even after the knob is released.

**Note**

If loss of electrical power occurs, the SCAI will display pitch and roll information for a minimum of 9 minutes to within 6 degrees of vertical.

### Bearing-Distance-Heading Indicator

The bearing-distance-heading indicator (27, figure FO–4) 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. 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. For JSS operation refer to Classified Supplement T.O. 1F-111(E)A-1-3.

## 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
- Inertial nav system
- Nav radar system
- Radar altimeter

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 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.

## CADC Angle-of-Attack Correction

The angle-of-attack indicating system consists of a conical probe transmitter located on the left side of the aircraft fuselage, an angle-of-attack correction cam in the CADC, two angle-of-attack indexers, and a vertical scale readout on the airspeed Mach indicator. Two identical angle-of-attack probes are installed, one on each side of the fuselage. The right angle-of-attack probe is connected directly to the SIS computer, but is not connected to the CADC and therefore does not affect the angle-of-attack indicators. In flight, the left 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 Mach 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 Mach 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 Mach 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 probes. 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 $\alpha/\beta$ 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.

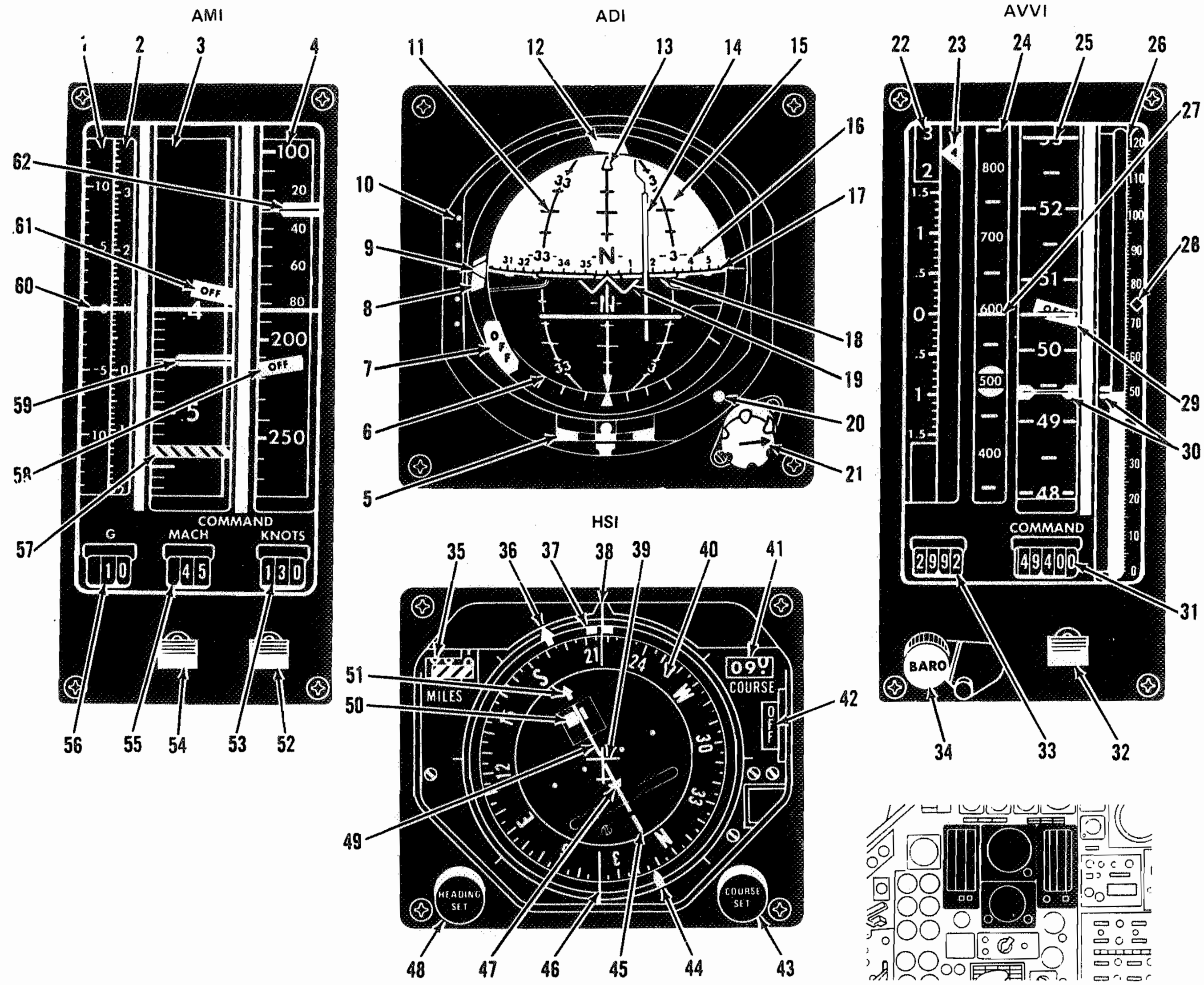
ANGLE-OF-ATTACK INDEXERS.

There are two angle-of-attack indexers (18, figure FO–3 and 23, figure FO–4) each consisting of three 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 index 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.

## Airspeed Mach Indicator (AMI)

The AMI (figure 1-23), 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

# 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 (INOPERATIVE)
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. TACAN RANGE INDICATOR AND WARNING FLAG
36. TACAN BEARING POINTER
37. HEADING MARKER
38. UPPER LUBBER LINE
39. AIRCRAFT SYMBOL
40. COMPASS CARD
41. COURSE SELECTOR WINDOW
42. POWER OFF WARNING FLAG
43. COURSE SET KNOB
44. BEARING POINTER TAIL
45. COURSE ARROW TAIL
46. LOWER LUBBER LINE
47. TO-FROM INDICATOR
48. HEADING SET KNOB
49. COURSE DEVIATION INDICATOR
50. COURSE WARNING FLAG
51. COURSE ARROW HEAD
52. COMMAND AIRSPEED SLEWING SWITCH
53. COMMAND AIRSPEED READOUT WINDOW
54. COMMAND MACH SLEWING SWITCH
55. COMMAND MACH READOUT WINDOW
56. ACCELERATION READOUT WINDOW
57. MAXIMUM ALLOWABLE MACH MARKER
58. AIRSPEED WARNING FLAG
59. COMMAND MACH MARKER
60. FIXED INDEX LINES
61. POWER OFF WARNING FLAG
62. COMMAND AIRSPEED MARKER

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Figure 1-23.

airspeed markers are located on the bottom of the indicator. Signals for operation of the various scales are provided from the 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 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. 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 be reliable as soon as the aircraft angle-of-attack is reduced to 22 degrees or below. If a flight condition is encountered in which the angle-of-attack indicator is near its limits 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 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 indicates true Mach number which is shown on a moving scale and is read against the fixed index. The scale is calibrated in hundredths and shows numbers in tenths from 0.4 through 3.5. At speeds below Mach 0.4, the scale will continue to read 0.4. The moving scale is operated by electrical signals from the CADC. A command Mach marker and command Mach readout window indicate manually selected command Mach. The command Mach marker remains at the top or bottom of the display column until the selected command Mach comes into view on the Mach scale, at which time it will synchronize and move with the scale. The selected 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 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 1,000 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 indicate 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 with 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 (AVVI)**

The AVVI (figure 1-23) 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 5,000 feet.

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

VERTICAL VELOCITY INDICATOR

The vertical velocity indicator indicates climb or dive velocities from 0 to 1,500 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 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 −1,000 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 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 (ADI)**

The ADI (figure 1-23), 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 inertial nav system. Attitude data to the ADI is received directly from either the inertial 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 inertial 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 the SCAI 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 self-contained attitude indicator should be used.

**WARNING**

Frequent cross checks between the ADI, the SCAI 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.

**Horizontal Situation Indicator (HSI)**

The HSI (Figure 1-23) 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 inertial nav system or auxiliary flight refrence 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 coarse 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.)

**Instrument System Coupler Pitch Steering Mode Switch**

The ISC pitch steering mode switch (43, figure FO-3) 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 ISC 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). The ALT REF position is compatible with all positions of the ISC 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 bar on the ADI. The TF position is compatible with all positions of the ISC mode selector knob except ILS, AILA and AIR/AIR.

**Note**

Altitude reference submode limits are $\pm 500$ 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 the ALT REF position if either TFR channel is in the TF mode and is operating normally.

**Instrument System Coupler Mode Selector Knob**

The ISC mode selector knob (43, figure FO-3, has nine positions 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 provde the following functions: For ISC modes versus instrument indications refer to figure FO-15. Refer to figure FO-16 for ADI and HSI steering indication and instrument warning flag analysis.

- In the OFF position, the steering bars and OFF flags are biased out-of-view on the ADI leaving attitude and heading displays.
- The ILS 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 bar on the ADI and course deviation information is displayed on the course deviation indicator of the HSI. Glide slope deviation is displayed on the glide slope deviation indicator on the ADI. Pitch steering commands are displayed on the pitch steering bar on the ADI 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 bar on the ADI.

**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 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 FO-16.
- If a localizer deviation of more than two dots on the course deviation indicator occurs when the pitch steering bar on the ADI is in view, it will drive out of view until a correction is made to bring the localizer deviation back within the two dot limit.

With the radar altimeter operating and set for a minimum altitude penetration, the pitch steering bar on the ADI 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 recentered 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 5,000 feet absolute altitude the radar altitude low warning lamp will momentarily light and the pitch steering bar on the ADI will momentarily indicate a fly-up command when the aircraft descends through 5,000 feet.

- The AILA 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 inertial nav system in conjunction with the nav 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 and HSI as the ILS position. For AILA procedures refer to Section VII.

**Note**

During AILA approaches, when the inertial nav system is furnishing a simulated localizer, the pitch steering bar will remain in view even when the aircraft deviates more than $\pm 2$ dots from the simulated localizer.

- The TACAN 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 bar on the ADI 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 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 HSI course selector window. This establishes a course signal to the inertial nav system where a course error signal is developed. The inertial 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 computer course to destination. These two signals are combined with aircraft roll to provide steering commands to the bank steering bar on the ADI 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 bank steering bar. 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. The 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 position provides computed course information from the inertial nav system when it is used in one of two modes of operation. When the inertial nav mode selector knob is in either the GREAT CIRCLE or SHORT RANGE positions, computed course steering commands to a destination set into the inertial nav system are displayed by the bank steering bar on the ADI 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 the NAV mode. The course arrow and course selector window display current magnetic ground track from the inertial 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 position provides the capability of flying a manually selected course instead of an inertial 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 inertial nav system and an error signal is provided to display course steering commands on the bank steering bar on the ADI 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 inertial 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 bar on the ADI. Any deviation from this heading will generate a steering command on the bank steering bank on the ADI. If the inertial 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 ISC mode selector knob in the OFF, ILS, AILA, TACAN, CRS SEL NAV, NAV, MAN CRS or MAN HDG positions and with the ISC pitch steering mode switch at OFF, a fly-up command will be displayed on the pitch steering bar on the ADI and the radar altitude low warning lamp will light when the aircraft 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 aircraft climbs back through the altitude index setting. If a descent is begun from above 5,000 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 bar on the ADI will momentarily indicate a fly-up command when the aircraft descends through 5,000 feet.

- The AIR/AIR position provides the steering capability to a target being tracked by the nav radar system. In this mode the HSI heading marker is driven by a bearing signal from the nav radar and provides a signal to indicate the necesary steering commands on the bank steering bar on the ADI to steer the aircraft to the target. The pitch steering bar on the ADI 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 1-20) is provided for ground checking and troubleshooting of the integrated flight instruments, ISC, 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 ISC 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 FO–5.

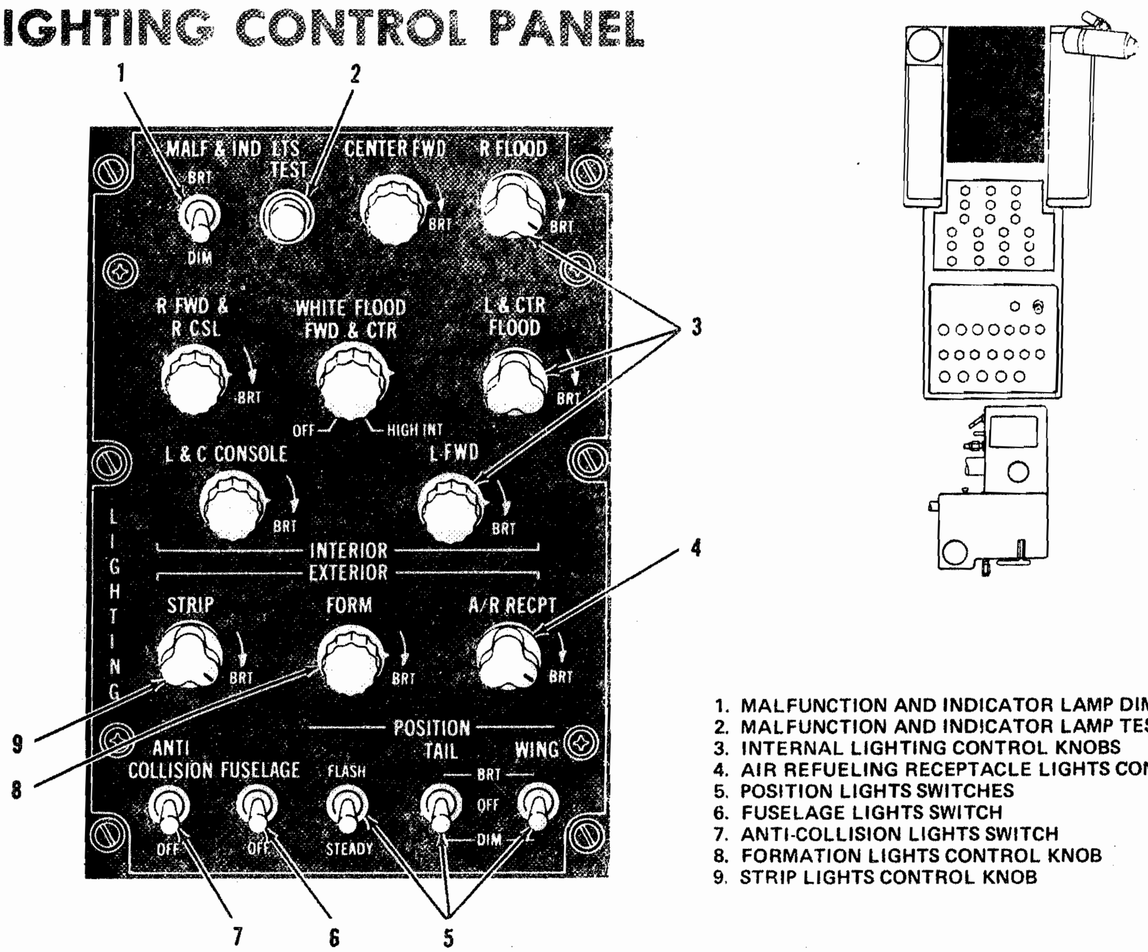
## MASTER CAUTION LAMP

The master caution lamp (figure FO–5) will light to alert the crew that a malfunction exists when any of the individual caution lamps on the main and lower caution lamp panels light. 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 (1, figure 1-24) is a three-position switched marked BRT (bright) and DIM and is spring-loaded to an unmarked center position. When the L FWD internal lighting control knob is out of the off detent, this switch controls the light

# LIGHTING CONTROL PANEL




1. MALFUNCTION AND INDICATOR LAMP DIMMING SWITCH
2. MALFUNCTION AND INDICATOR LAMP TEST BUTTON
3. INTERNAL LIGHTING CONTROL KNOBS
4. AIR REFUELING RECEPTACLE LIGHTS CONTROL KNOB
5. POSITION LIGHTS SWITCHES
6. FUSELAGE LIGHTS SWITCH
7. ANTI-COLLISION LIGHTS SWITCH
8. FORMATION LIGHTS CONTROL KNOB
9. STRIP LIGHTS CONTROL KNOB

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Figure 1-24.

intensity 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 individually tested and dimmed.

   a. TFR R & L Channel Fail Indicator Lamps.

   b. IFF Panel Reply & Test Caution Lamps.

   c. CMRS Indicator Lamp.

   d. Angle-of-Attack Indexer Lamps. (Dimming Only)

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

### Malfunction and Indicator Lamp Test Button

The malfunction and indicator lamp test button (2, figure 1-24) 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, and (4) the CMRS ready/test indicator lamp. The malfunction and indicator lamp test button is also used to ground check the stall warning system.

**Note**

Depressing the malfunction and indicator lamp test button, with the flaps extended 15 degrees or more, 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, a taxi light and electroluminescent strip lights (refer to figure FO–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 amber 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. Electroluminescent strip lighting is provided to augment the formation lights. The strip lights are light emitting panels approximately 4 by 36 inches in size. The strip lights are green and are located on the aircraft as follows (figure FO–1): one light on each wing tip consisting of two light emitting panels, one each on the upper and lower wing tip surfaces; one light located forward and one light located aft on each side of the fuselage; one light located on each side of the vertical stabilizer; and two lights located on top of the fuselage, one aft of each glove.

**CAUTION**

The electroluminescent strip lights should not be operated in sunlight or during supersonic flight. Sustained operation in bright sunlight or at high ambient temperatures during supersonic flight may permanently damage the strip lights or significantly reduce operating service life.

Three anti-collision/fuselage lights, one located on top and two 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 (5, figure 1-24) 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.

#### Position Lights/Stores Refuel Battery Power Switch

The position lights/stores refuel battery power switch (5, figure 1-20) 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 Control Knob

The formation lights control knob (8, figure 1-24) labeled FORM, controls the intensity of the formation lights. The full counterclockwise position of the knob turns the lights off. As the knob is turned clockwise, the intensity of the lights varies from off to full brightness.

#### Strip Lights Control Knob

The strip lights control knob (9, figure 1-24), labeled STRIP, provides selection of the desired intensity of the strip lights. The full counterclockwise position of the knob turns the lights off. Rotating the knob clockwise varies the intensity of the lights from off to full brightness.

**Note**

When the strip lights are turned on, there may be an increase on the fuel totalizer by 300 to 500 pounds.

#### Anti-Collision Lights Switch

The anti-collision lights switch (7, figure 1-24) is a two position switch marked ANTI COLLISION, and OFF.

Placing the switch to ANTI COLLISION 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.

### Fuselage Lights Switch

The fuselage lights switch (6, figure 1-24) is marked FUSELAGE and OFF. Placing the switch to FUSELAGE 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 (4, figure 1-24) is labeled A/R RECPT. 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.

### Landing and Taxi Lights Switch

The landing and taxi lights switch (1, figure 1-2) is marked LANDING, OFF and TAXI. If the switch is left in either the LANDING or TAXI position, a switch on the nose gear down lock will turn them off when the gear is retracted.

## 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 four circuits, each with control knob, for the left instrument panel, center instrument panel, right instrument panel and right console, and left and center console. They are powered by 115 volt ac power from the right main ac bus through two circuit breaker panels (figure 1-25) one on the crew compartment aft bulkhead and one below the right console. The flood lights consist of left, center and red flood lights and high intensity white flood lights at various locations around the cockpit. Red flood lights provide cockpit lighting in the event the instrument panel and console lights fail. Control of the left and center red flood lights is combined into a single control knob, and the right red flood lights are controlled by an individual knob. The left and center instrument panel white flood lights provide secondary lighting during conditions in which the air crew has no need for dark adaptation and high intensity white flood lights prevent temporary blindness from lightning when flying in adverse 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 (19, figure FO–2 and 5, figure 1-26), 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

Internal lighting control knobs (3, figure 1-24), control the various internal lighting circuits. The full counterclockwise position of each knob turns the lights off. As the knobs are turned clockwise, the intensity of the lights vary R
from off to full bright. Four of the knobs control the red instrument panel and console lighting. Knobs are labeled and control the respective circuits as follows:

L FWD - Left instrument panel. Controls lamp dimming switch when out of the off detent.

CENTER FWD - Center instrument panel.

R FWD & R CSL - Right instrument panel and right console.

L & CTR CONSOLE - Left and center consoles.

The red flood lights are controlled by two knobs marked L & CTR FLOOD and R FLOOD for the left, center and right flood lights. A single knob marked WHITE FLOOD FWD & CTR 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 knob 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.

# INTERNAL LIGHTING CIRCUIT BREAKER PANELS

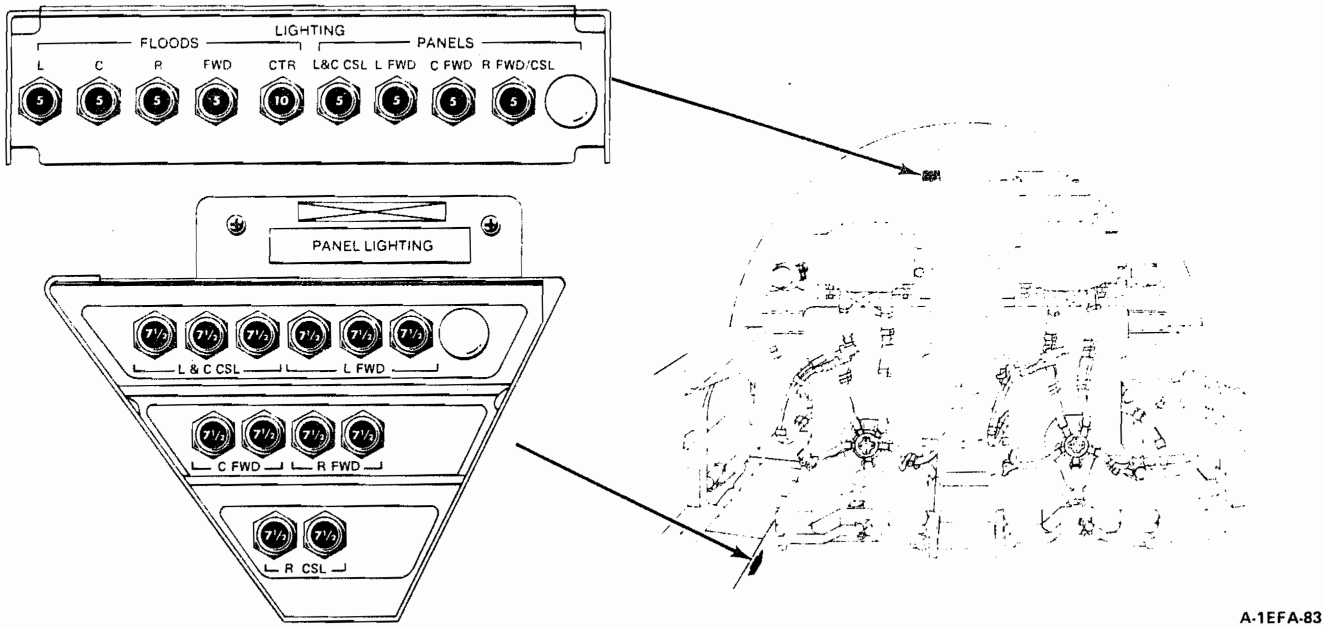


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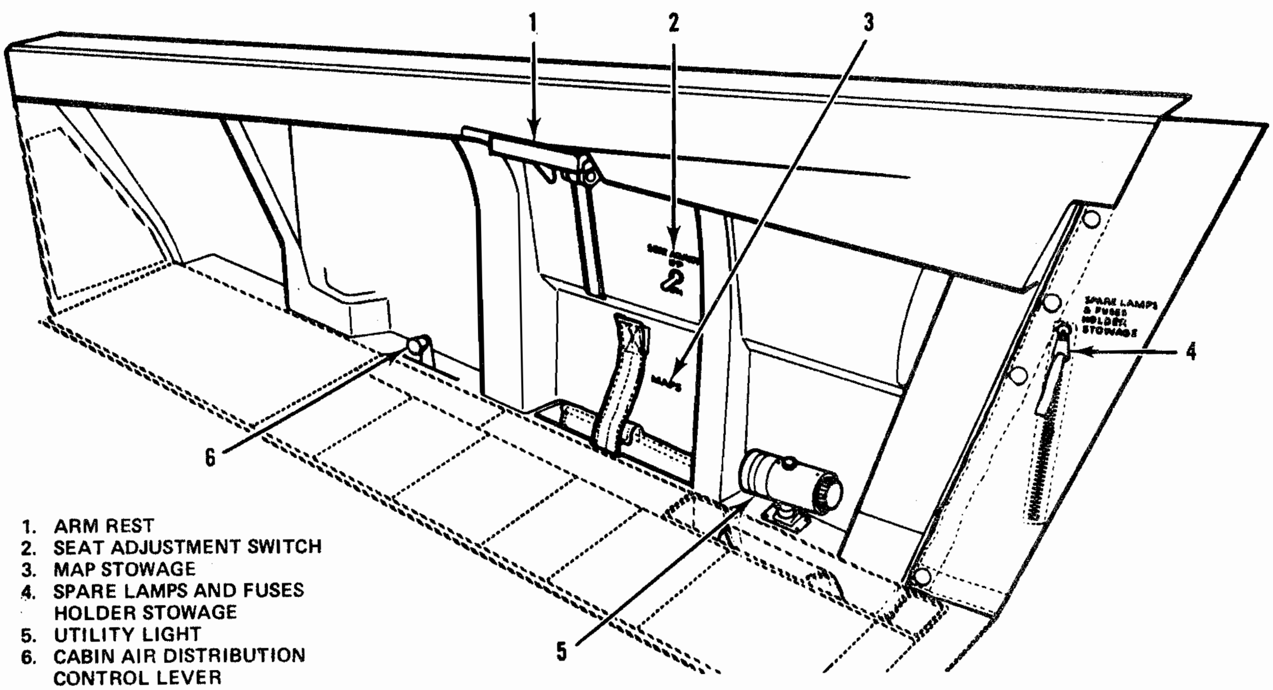
**Figure 1-25.**

# RIGHT SIDEWALL




1. ARM REST
2. SEAT ADJUSTMENT SWITCH
3. MAP STOWAGE
4. SPARE LAMPS AND FUSES HOLDER STOWAGE
5. UTILITY LIGHT
6. CABIN AIR DISTRIBUTION CONTROL LEVER

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**Figure 1-26.**

## INTERNAL CANOPY LATCH HANDLES

Two canopy latch handles are located on the inside lower horizontal frame member of each canopy hatch (2, figure FO–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 the 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 mechanically linked to a flush external canopy latch handle located outside of each hatch. Inflation of the canopy pressurization seal is automatically operated by closure of the canopy hatch. The actuator mounted on the hatch lower surface depresses a plunger in the canopy sill to inflate the seals and turn off the canopy unlock warning lamp.

## CANOPY EXTERNAL LATCH HANDLES

Two flush mounted canopy external latch handles (figure 3-2) located on the lower horizontal frame member of each canopy hatch are mechanically linked to respective internal handles. 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

The red canopy unlock warning lamp located on the upper warning and advisory lamp panel (figure FO–5) will light when either hatch is not locked. When lighted the word CANOPY is visible.

## ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (figure FO–17) provides 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, weapons bay, anti-icing and defog systems, windshield rain removal system, anti-"g" suits, and pneumatic pressure for throttle boost and fuel tank pressurization. In addition, the system includes a thermal transport section, which through liquid cooling loops, removes waste heat from the JSS and SPS.

## 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, fin pod, and to the electronic equipment in the forward electronics bay which requires a controlled environment for efficient operation. See figure FO–17. 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 the primary air-to-air heat exchanger, where it is cooled by ram air. The air is then routed to the compressor where it is compressed to a higher pressure. From the compressor it flows through the secondary air-to-air heat exchanger where it is again cooled by ram air. 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 flow modulating valve. It also controls the setting of the cockpit hot air modulating 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 water tank vent located on the lower side of the fuselage between the weapons bay and the main landing gear door. This is a normal condition and should be no cause for concern.

During ground operation, cooling of engine bleed and compressor discharge air in the primary and secondary heat exchangers is automatically provided by two hydraulically-driven fans. When the pressure in the turbine inlet duct reaches a preset value, a solenoid valve opens to provide utility hydraulic system pressure to drive the fans.

### Cabin Air Distribution Control Lever

A cabin air distribution control lever (6, figure 1-26) 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**

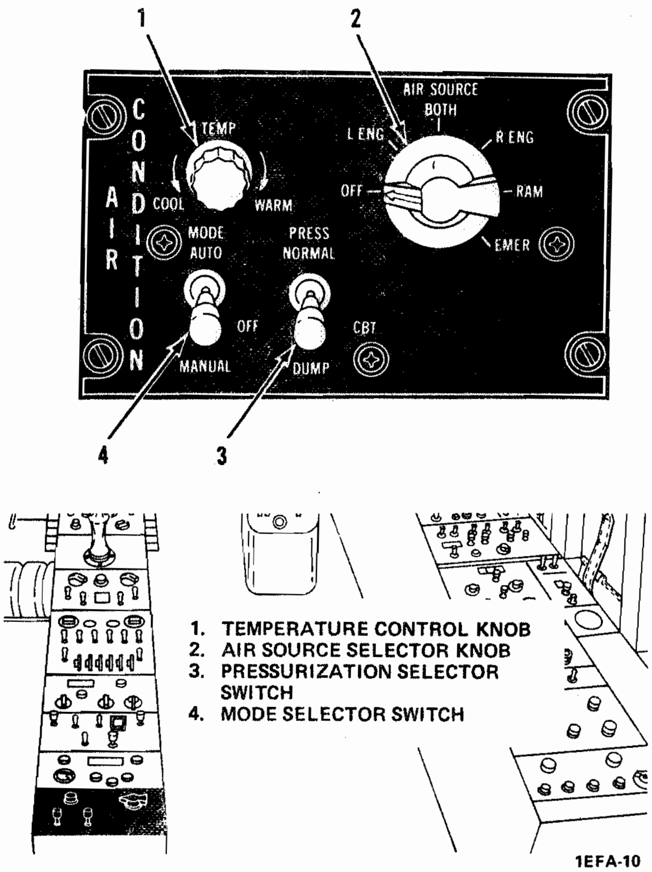
- During operation at high power settings, and when airflow to the cabin is high, moisture may be sprayed on the windshield and in the crew members' face. Moving the lever toward the aft position will decrease this possibility.
- Prior to conducting a rapid descent in a tropical environment, the cabin 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 1-27) 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 turbine flow modulating valve will be open, but there will not be any flow. The cabin hot air valve will be modulating, but there will not be any flow. The primary 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 turbine flow modulating and shutoff valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The primary 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 flow modulating and shutoff valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The primary 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 turbine flow modulating and shutoff valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The primary pressure regulating and shutoff valve will be regulating.
- In the RAM position, the engine bleed air check and shutoff valves are open. The turbine flow modulating and shutoff valve is closed (no flow). The cabin hot air modulating and shutoff valve is modulating in response to the positon of the temperature control knob. The primary pressure regulating and shutoff valve is open.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 rain removal, throttle boost, anti-"g" suits, canopy seals, and weapons bay pallet seals.

# AIR CONDITIONING CONTROL PANEL




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

**Figure 1-27.**

- In the EMER position the left and right engine bleed air check and shutoff valves are closed. The turbine flow modulating shutoff valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating (no flow). The primary pressure regulating and shutoff valve is closed. The emergency 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 rain removal, throttle boost, anti-g suits or for electronic equipment. Caution lamps associated with these systems may come on shortly after selection of the EMER position.

**Note**

Placing the ground start switch to PNEU or CARTRIDGE for an airstart will open the bleed air check and shutoff valves regardless of the position of the air source selector knob.

### Air Conditioning System Mode Selector Switch

The mode selector switch (4, figure 1-27) is a three-position lever-lock toggle switch marked AUTO, OFF, and MANUAL. 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 go to the full cool position except as limited by the flow control unit.

### Temperature Control Knob

The temperature control knob (1, figure 1-27) is marked COOL and 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 40 degrees F or above 95 degrees 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 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 cabin is supplied with warm or cool air corresponding to the position of the temperature control knob.

**Forward Equipment Hot Caution Lamp**

The amber equipment hot caution lamp, marked FWD EQUIP HOT, is located on the main caution lamp panel (figure FO-5). 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.

- ALQ–137 (SPS)
- Nav Radar
- TFR
- ALR–62 (TTWS)
- HF Radio (transmit)
- UHF Radio (transmit)
- TACAN
- Inertial Nav
- Radar Altimeter
- HF Radio (receive)
- IFF
- UHF Radio (receive)
- ALR–23 (CMRS)

**Note**

With air conditioning on, to minimize water consumption and to ensure minimum required airflow to equipment, the following procedures must be observed:

- During ground operation, at higher ambient temperatures, canopy position will affect air conditioning requirements.
  - OAT ≤70°F - Equipment operation is unrestricted.
  - OAT ≥70°F - ALQ–137 (SPS) is limited to 5 minutes or less regardless of canopy position. The remaining equipment is limited to 30 minutes or less with the canopy closed and is unrestricted with the canopy open.
- During ground operation, with a tailwind or single air source selected, operate engines at 70% RPM. If forward equipment hot caution lamp lights, refer to "Caution Lamp Analysis" Section III.
- During approach, ensure JSS power and ALQ–137 (SPS) power are off.
- Do not operate the rain removal and anti-icing systems simultaneously for more than 15 total minutes on any one mission.

## PRESSURIZATION SYSTEM

Pressurization of the cockpit, throttle boost, wing seals, weapons bay door 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 8,000 feet, the pressure regulating valve automatically maintains an unpressurized condition in the cockpit regardless of the schedule selected. When the aircraft is above 8,000 feet, the pressure schedule is maintained by controlling the outflow of incoming ventilation air. A cabin pressure safety valve located at the rear of the cockpit will relieve pressure any time the cockpit pressure exceeds limits. 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 1-28 for cockpit pressure schedule for normal and combat conditions.

# CABIN PRESSURE SCHEDULE

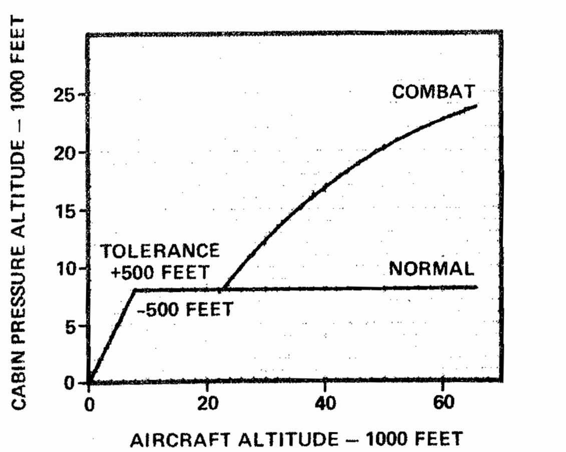



Figure 1-28.

**CAUTION**

A priority valve is incorporated into the system which ensures 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 the priority valve may cause the 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 (3, figure 1-27) 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 8,000 foot cabin altitude from 8,000 feet up to the operational ceiling of the aircraft. In the CBT (combat) position, the cockpit maintains an 8,000 foot cabin altitude from 8,000 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 (2, figure 1-3) is provided to monitor cabin altitude.

### Pressurization Caution Lamp

An amber pressurization caution lamp marked CABIN PRESS is located on the main caution lamp panel (figure FO–5). 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 (figure FO–5) marked CABIN PRESS is located on the upper warning and advisory 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 lower caution lamp panel (figure FO–5). The lamp will light when the supply pressure to the pressurized electronic equipment (TFR and ECM equipment) drops below limits.

## ANTI-ICING AND DEFOG SYSTEMS

### 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 anti-icing/windshield rain removal 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/translating cowl, and leading edge of the auxiliary cowl (aft of translating cowl). The engine inlet system uses regulated compressor bleed air from the 16th stage of each engine.

**Note**

- Do not operate the rain removal and anti-icing systems simultaneously for more than 15 total minutes on any one mission.
- Idle RPM will not 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.

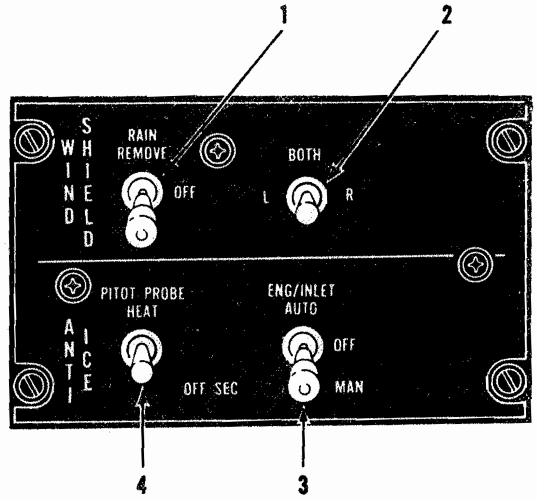
### Engine/Inlet Anti-Icing Switch

The engine/inlet anti-icing switch (3, figure 1-29) 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 energizs 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.

**Note**

Operation with engine/inlet anti-icing switch in MANUAL for extended periods may result in an increase of boil-off rate of environmental system cooling water. This in turn could cause a loss in cabin pressurization, cooling capacity and illumination of the forward equipment hot light due to system shutdown caused by a dry water tank. The switch should be kept in the AUTO mode unless the AUTO mode is known to be malfunctioning. Refer to "Icing," Section III, for malfunctions.

## ANTI-ICING/WINDSHIELD RAIN REMOVAL CONTROL PANEL (TYPICAL)




1. WINDSHIELD RAIN REMOVAL SELECTOR SWITCH
2. WINDSHIELD SELECTOR SWITCH
3. ENGINE/INLET ANTI-ICING SWITCH
4. PITOT/PROBE HEATER SWITCH

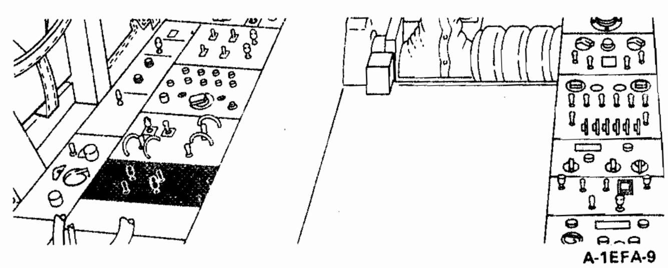



Figure 1-29.

### Engine Icing Caution Lamp

The engine icing caution lamp, located on the lower caution lamp panel (figure FO–5), will light when the electronic ice detector senses an icing condition. While the icing condition exists, the caution lamp will display the word ICING and 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 cease.

### Inlet Hot Caution Lamp

The inlet hot caution lamp, located on the lower caution lamp panel (figure FO–5), provides an indication that the temperature of anti-icing bleed air to the auxiliary cowls has exceeded limits. 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 to the probe heater is controlled by a pitot/probe heater switch located on the anti-icing/windshield rain removal control panel. An alpha/beta probe caution lamp, located on the lower caution lamp panel (figure FO–5), 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 1-29) 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 probe heater is off.
- In the HEAT position, power is provided to the primary alpha/beta heaters, pitot-static heater and the total temperature probe heater.

On takeoff the squat switch on the landing gear activates to arm the secondary heater circuits in the two angle-of-attack probes and side-slip angle probe. During flight if the primary heaters in either the angle-of-attack or side-slip angle 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 side-slip 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 CADC caution lamp may light and remain lighted until total temperature drops below 50 degrees C.

The switch controls 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) probes and side-slip angle (beta) probe.

### Alpha/Beta Probe Heat Caution Lamp

The alpha/beta probe heat caution lamp is located on the lower caution lamp panel (figure FO–5). When the lamp lights, the legend $\alpha/\beta$ PROBE HEAT is displayed. The lamp provides indications that the angle-of-attack (two) 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 one or more 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 one or more probes are not functioning.
- With the pitot/probe heater switch in the OFF/SEC position, indicates the secondary heater elements in one or more 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 RAIN REMOVAL SYSTEM

The windshield rain removal system is provided to keep both the windshields clear of ice, and rain. Compressor bleed air is directed over the outside of the windshields by a fixed area nozzle.

### Windshield Rain Removal Selector Switch

The windshield rain removal selector switch (1, figure 1-29) is marked RAIN REMOVE, and OFF. The switch is locked out of RAIN REMOVE and 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 selected windshield(s). Under conditions of heavy rain, the system may not supply sufficient hot air to clear both windshields simultaneously. Therefore, more desirable results may be obtained by selecting only one windshield for rain removal, directing all available air to the selected windshield.

### Windshield Selector Switch

The windshield selector switch (2, figure 1-29), marked L (left), R (right), and BOTH, determines the windshield(s) to receive rain removal air. For optimum performance of the rain removal system, operate left side only. Selection of the BOTH position will decrease airflow on each windshield.

### Windshield Hot Caution Lamp

The windshield hot caution lamp, located on the lower caution lamp panel (figure FO–5) indicates rain removal high temperature. An overheat switch, installed in the rain removal air supply duct upstream of the shutoff valve, will close when the air temperature exceeds limits. 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.

**CAUTION**

The windscreen itself does not contain overheat sensors. Normal rain removal temperatures, if left on in excess of operating limits, will damage the windscreen without lighting the windscreen hot caution lamp.

## THERMAL TRANSPORT SYSTEM

The thermal transport system (TTS) provides cooling for the JSS transmitters located in the weapons bay, and the SPS aft high band receiver and power amplifier located in the left and right speed bumps. The TTS (figure FO–18) consists of a self-contained coolant system for the JSS and one for the SPS.

### JSS Coolant System

The JSS coolant system (figure FO–18) circulates liquid coolant through 10 transmitters in the weapons bay. Basic components consist of coolant pumps, shut-off valves, diverter valves, ram air heat exchangers, coolant-to-fuel heat exchangers, solenoid valves, and a coolant accumulator. The coolant is circulated by two pumps in parallel. When coolant temperature is below approximately 80°F, diverter valves cause the coolant to bypass the heat exchangers providing rapid transmitter warm-up during cold weather operation. When coolant temperature is above approximately 90°F, all the coolant is diverted to the ram air heat exchangers and after exiting, approximately half the flow goes through the JSS coolant-to-fuel heat exchangers. If fuel temperature through the JSS coolant-to-fuel heat exchangers is below 38°F and ram air temperature is below 115°F, coolant flow bypasses the JSS coolant-to-fuel heat exchangers. When ram air temperature exceeds 200°F, the ram air doors close and coolant flow is routed through the JSS coolant-to-fuel heat exchangers. Depressing an engine fire pushbutton warning lamp will close the respective engine coolant shut-off valve and shutdown the coolant pumps. Depressing the button again will turn the pumps back on. The shut-off valve cannot be reopened in flight. Depressing the fuselage fire pushbutton warning lamp will shut down the coolanol pumps and JSS. Depressing the button again will turn on the coolanol pumps and JSS.

### SPS Coolant System

The SPS coolant system (figure FO–18) circulates liquid coolant through the aft SPS high band receiver in the left speed bump and the aft SPS high band power amplifier in the right speed bump. Basic components consist of a reservoir, coolant pump, and SPS coolant to fuel heat exchanger. Coolant from the reservoir is circulated by the pump to the SPS coolant-to-fuel heat exchanger in the lower left fuel trap tank portion of the reservoir tank and then to the SPS high band power amplifier and receiver.

### JSS Ram Doors Override Switch

The JSS ram doors override switch (51, figure FO–3) is marked CL, AUTO and OPN. When in the CL or OPN, the ram air doors are either fully closed or fully open. In the AUTO position operation of the ram air doors is automatic, depending on jammer status and ram air temperature as follows:

| Jammer Mode | Ram Air Temperature | Door Position |
|---|---|---|
| — | > 200°F | Closed |
| OFF | — | Closed |
| Standby | < 200°F | Intermediate |
| Radiate | < 200°F | Open |

**Note**

Ensure JSS ram doors override switch is not in the OPN position for Mach No. above 1.7.

### JSS Ram Doors Caution Lamp

The JSS ram doors caution lamp on the lower caution lamp panel (figure FO–5) will light when door position does not agree with the control logic determined by the JSS ram doors override switch position.

### Heat Exchanger Override Switch

The heat exchanger override switch (51, figure FO–3) is marked AUTO and OFF. In the AUTO position coolant flow through the JSS coolant-to-fuel heat exchangers is controlled automatically. In the OFF position, coolant flow to the JSS coolant-to-fuel heat exchangers is shut off.

### Fuel Hot Caution Lamps

Two fuel hot caution lamps on the lower caution lamp panel (figure FO–5) are marked L FUEL HOT and R FUEL HOT. The lamps will light when fuel exiting the applicable JSS coolant-to-fuel heat exchanger reaches 153°F.

### JSS/SPS Liquid Pump Switches

The JSS and SPS liquid pump switches (1, figure 1-6) are marked NORM and OVRD. In the NORM position, the pumps are turned on automatically when their respective system is activated. Placing the switch to OVRD overrides the automatic feature causing the pumps to operate.

### JSS Pump Overheat Caution Lamp

The JSS pump overheat (PMP OVHT) caution lamp on the JSS/SPS/TTWS caution and warning lamp panel (figure FO–5) will light when either pump temperature reaches 300°F.

### JSS Liquid Temperature/Pressure Caution Lamp

The JSS liquid temperature/pressure (LIQ T/P) caution lamp on the JSS/SPS/TTWS caution and warning lamp panel (figure FO–5) will light when liquid coolant temperature prior to entering the weapons bay reaches 185°F or pump outlet pressure falls below 20 to 22 psi.

### SPS A-HI Hot Caution Lamp

The SPS A-HI hot (A-HI HOT) caution lamp on the JSS/SPS/TTWS caution and warning lamp panel (figure FO–5) will light when coolant pressure falls below 18 to 22 psi.

## FUSELAGE FIRE DETECTION AND EXTINGUISHING SYSTEM

The 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 nosewheel well area.

### FUSELAGE FIRE PUSHBUTTON WARNING LAMP

The fuselage fire pushbutton warning lamp (16, figure FO–3), located on the left 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, and shut down the weapons bay coolanol pumps and JSS. 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 and turn on the coolanol pumps and JSS. 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

Momentarily positioning the agent discharge/fire detect test switch (15, figure FO–3) to the 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 (4, figure 1-20) is 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

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, the plumbing crossover areas aft of the main landing gear bulkhead, the weapons bay routing tunnel and near the ac power panel, 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.

**Note**

The engine bleed air check and shutoff valves are electrically controlled and pneumatically operated. Each valve is controlled by 28 volt dc power from the essential bus and is protected by a circuit breaker located in the right forward equipment bay. Loss of electrical power to a valve with sixteenth stage bleed air present will cause the valve to open.

### AGENT DISCHARGE/FIRE DETECT TEST SWITCH

Holding the agent discharge/fire detect test switch (15, figure FO–3) 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 FO–5), located on the main caution lamp panel, provides an indication of an overheat condition in the main wheel well, the plumbing crossover areas aft of the landing gear bulkhead, and the weapons bay routing tunnel and ac power panel area, in event of a rupture in the engine bleed air lines. The words WHEEL WELL 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 supply consists of 10 liters. The normal oxygen system consists of a 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 crewmember's torso harness. A three 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 1-30. Refer to figure 1-66 for oxygen system servicing.

#### Oxygen Control Knob

The oxygen control knob, located on the oxygen regulator (9, figure 1-31) 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 at cabin altitudes from 0 to 17,500 feet. At cabin altitudes above 17,500 feet, 100 percent oxygen is supplied and pressure breathing will commence at approximately 35,000 feet. 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 (1, figure 1-31) are provided to control flow of oxygen from the supply system to the oxygen regulator. When in the ON position, oxygen is supplied from the converter or, if activated, the emergency oxygen system to the regulator; when in OFF, oxygen flow is shut off at the control valve in the oxygen-suit control panel.

**WARNING**

With the 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.

**CAUTION**

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

#### Oxygen Quantity Indicator

The oxygen quantity indicator (3, figure 1-3) is graduated from 0 to 20 liters in increments of one liter to indicate the quantity of liquid oxygen in the converter. When fully serviced the indicator will read 10 liters. In the event of power failure, the indicator pointer will freeze.

#### Oxygen Quantity Indicator Test Button

The oxygen quantity indicator test button (4, figure 1-3), when depressed will move the indicator pointer to zero 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 lower caution lamp panel (figure FO-5) will light when the oxygen quantity indicator indicates 2 liters or less or when oxygen system pressure is below limits. 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 and the master caution lamp will light.

# OXYGEN DURATION

WITH REGULATOR AT 100 PERCENT OR EMER POSITION.

| CABIN ALTITUDE | CONSUMPTION 2 MEN cu. ft./hr. | DURATION – HOURS* | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 35,000 | 9.8 | 31.2 | 28.1 | 25.0 | 21.8 | 18.7 | 15.6 | 12.5 | 9.4 | 6.2 | 3.1 |
| 30,000 | 13.4 | 22.8 | 20.5 | 18.3 | 16.0 | 13.7 | 11.4 | 9.1 | 6.8 | 4.6 | 2.3 |
| 28,000 | 15.0 | 20.4 | 18.4 | 16.3 | 14.3 | 12.2 | 10.2 | 8.2 | 6.1 | 4.1 | 2.0 |
| 26,000 | 16.0 | 19.1 | 17.2 | 15.3 | 13.4 | 11.5 | 9.6 | 7.6 | 5.7 | 3.8 | 1.9 |
| 24,000 | 18.52 | 16.5 | 14.9 | 13.2 | 11.6 | 9.9 | 8.3 | 6.6 | 5.0 | 3.3 | 1.6 |
| 22,000 | 20.76 | 14.7 | 13.3 | 11.8 | 10.3 | 8.8 | 7.4 | 5.9 | 4.4 | 2.9 | 1.5 |
| 20,000 | 23.0 | 13.3 | 12.0 | 10.6 | 9.3 | 8.0 | 6.6 | 5.3 | 4.0 | 2.7 | 1.3 |
| 18,000 | 25.24 | 12.1 | 10.9 | 9.7 | 8.5 | 7.3 | 6.1 | 4.8 | 3.6 | 2.4 | 1.2 |
| 16,000 | 27.43 | 11.1 | 10.0 | 8.9 | 7.8 | 6.7 | 5.6 | 4.4 | 3.3 | 2.2 | 1.1 |
| 14,000 | 30.0 | 10.2 | 9.2 | 8.2 | 7.1 | 6.1 | 5.1 | 4.1 | 3.1 | 2.0 | 1.0 |
| 12,000 | 32.8 | 9.3 | 8.4 | 7.5 | 6.5 | 5.6 | 4.7 | 3.7 | 2.8 | 1.9 | 0.9 |
| 10,000 | 35.6 | 8.6 | 7.7 | 6.9 | 6.0 | 5.1 | 4.3 | 3.4 | 2.6 | 1.7 | 0.8 |
| 8,000 | 39.4 | 7.8 | 7.0 | 6.2 | 5.4 | 4.7 | 3.9 | 3.1 | 2.3 | 1.5 | 0.7 |
| Sea Level | 55.6 | 5.5 | 4.9 | 4.4 | 3.8 | 3.3 | 2.7 | 2.2 | 1.6 | 1.1 | 0.5 |
| AVAILABLE OXYGEN | LITERS (LIQUID) | 10 | 9 | 8 | 7 | 6 | 5 | 4 | 3 | 2 | 1 |
| | Cu. Ft. GAS | 306.0 | 275.4 | 244.8 | 214.2 | 183.6 | 153.0 | 122.4 | 91.8 | 61.2 | 30.6 |

WITH REGULATOR IN NORM POSITION.

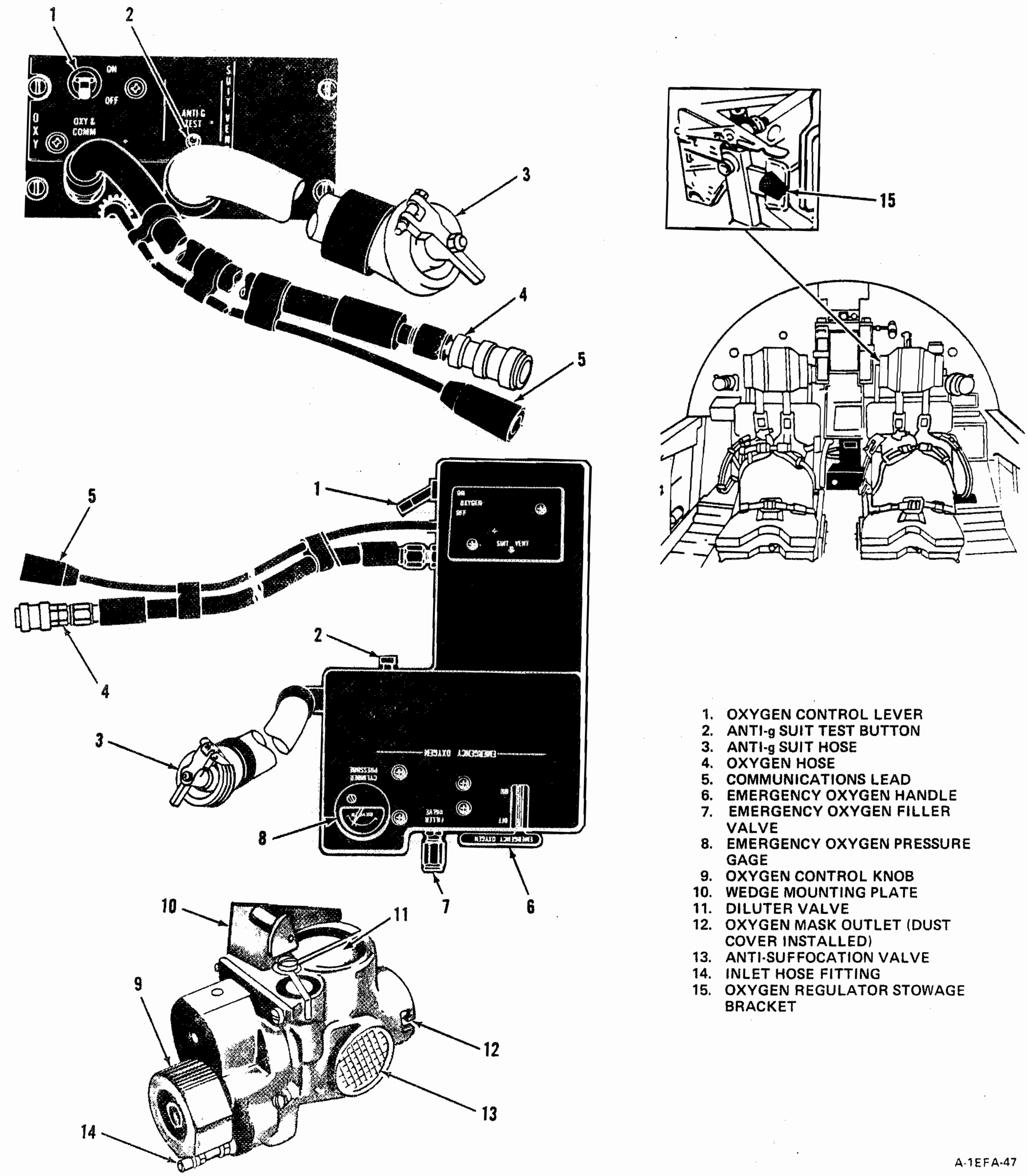
| CABIN ALTITUDE | CONSUMPTION 2 MEN cu. ft./hr. | DURATION – HOURS* | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 35,000 | 9.8 | 31.2 | 28.1 | 25.0 | 21.8 | 18.7 | 15.6 | 12.5 | 9.4 | 6.2 | 3.1 |
| 30,000 | 13.2 | 23.2 | 20.9 | 18.6 | 16.2 | 13.9 | 11.6 | 9.3 | 7.0 | 4.6 | 2.3 |
| 25,000 | 14.0 | 21.9 | 19.7 | 17.5 | 15.3 | 13.1 | 10.9 | 8.7 | 6.6 | 4.4 | 2.2 |
| 20,000 | 12.4 | 24.7 | 22.2 | 19.8 | 17.3 | 14.8 | 12.3 | 9.9 | 7.4 | 4.9 | 2.5 |
| 15,000 | 10.2 | 30.0 | 27.0 | 24.0 | 21.0 | 18.0 | 15.0 | 12.0 | 9.0 | 6.0 | 3.0 |
| 10,000 | 10.2 | 30.0 | 27.0 | 24.0 | 21.0 | 18.0 | 15.0 | 12.0 | 9.0 | 6.0 | 3.0 |
| 8,000 | 10.2 | 30.0 | 27.0 | 24.0 | 21.0 | 18.0 | 15.0 | 12.0 | 9.0 | 6.0 | 3.0 |
| 5,000 | 10.2 | 30.0 | 27.0 | 24.0 | 21.0 | 18.0 | 15.0 | 12.0 | 9.0 | 6.0 | 3.0 |
| Sea Level | 10.2 | 30.0 | 27.0 | 24.0 | 21.0 | 18.0 | 15.0 | 12.0 | 9.0 | 6.0 | 3.0 |
| AVAILABLE OXYGEN | LITERS (LIQUID) | 10 | 9 | 8 | 7 | 6 | 5 | 4 | 3 | 2 | 1 |
| | Cu. Ft. GAS | 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.

1EFA-52

Figure 1-30.

# OXYGEN-SUIT CONTROL PANEL AND DILUTER DEMAND OXYGEN REGULATOR



1. OXYGEN CONTROL LEVER
2. ANTI-g SUIT TEST BUTTON
3. ANTI-g SUIT HOSE
4. OXYGEN HOSE
5. COMMUNICATIONS LEAD
6. EMERGENCY OXYGEN HANDLE
7. EMERGENCY OXYGEN FILLER VALVE
8. EMERGENCY OXYGEN PRESSURE GAGE
9. OXYGEN CONTROL KNOB
10. WEDGE MOUNTING PLATE
11. DILUTER VALVE
12. OXYGEN MASK OUTLET (DUST COVER INSTALLED)
13. ANTI-SUFFOCATION VALVE
14. INLET HOSE FITTING
15. OXYGEN REGULATOR STOWAGE BRACKET

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Figure 1-31.

## 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 automatically during ejection or manually by a handle. This system also 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. It is then routed into the normal oxygen system and is controlled in the normal manner. Sufficient emergency oxygen is available for 10 minutes duration at 27,000 feet cabin altitude.

**Note**

EMER or 100% must be selected on the oxygen control knob (9, figure 1-31) to receive pure oxygen to the oxygen mask. Selecting NORM on the oxygen control knob supplies a proportionally diluted air/oxygen mixture.

### Emergency Oxygen Handle

The green emergency oxygen handle (6, figure 1-31) is used to manually activate the emergency oxygen system in the event automatic activation fails during ejection. Also, in event of failure of the normal oxygen system, this handle is used to activate the 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 (8, figure 1-31) indicates the pressure in the emergency oxygen bottles. The gage is marked REFILL in the red region and FULL in the black region.

## CREW MODULE ESCAPE SYSTEM

The crew module (figure FO–19) forms an integral portion of the forward fuselage and encompasses the pressurized cabin and forward portion of the wing glove. The system protects the occupants from environmental hazards on either land or water and provides manual underwater escape capabilities.

**WARNING**

The removal or addition of components in the crew module will change the center-of-gravity of the module and adversely affect its stability on ejection.

## CREW MODULE SEATS

The crew module seats (figure FO–19) 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 inertia reel to be unlocked. 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 harnesses to attach the oxygen regulator. When unlocked, the inertia reel, located behind the head rest, allows the shoulder straps to extend or retract 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 EWO's 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.

**WARNING**

No additional seat cushions have been authorized for use in this aircraft. The use of unauthorized seat cushions can increase the susceptibility to spinal injury during ejection.

## EJECTION EQUIPMENT.

The ejection equipment consists of 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 parachute 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 crewmembers 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 (±30) 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 (±30) knots by severance of the rocket upper nozzle diaphragm. The increase in exhaust area results in reduced lower nozzle thrust (about 9,000 pounds at the lower nozzle and 7,000 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.

### 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 a 1-second time delay initiator and blocks propagation to the rocket motor upper nozzle when aircraft speed is less than 300 (±30) knots. When aircraft speed is greater than 300 (±30) 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 the 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 shorlty 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. 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.

**WARNING**

> Failure to depress the lock release button on the canopy center beam handles prior to exerting "pull" pressure may prevent movement of the handles.

## 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 EWO's seat behind the back seat cushion (figure FO–19).

**WARNING**

> When gaining access to the survival compartment, crewmembers should make sure that feet are not under the seat pan and hands are clear and above seat sides. The 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 pilot's head rest. The kit is readily accessible to the crew, if the situation demands that the crew abandon the module immediately after landing.

## CREW MODULE EJECTION SEQUENCE

**WARNING**

> 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 avilable if required. After a delay of 0.35 second to allow for powered repositioning of crewmembers, 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. Crewmembers 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. The catapult spring may cause a banging noise on the back of the module after deployment. 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. The crew module usually rolls after ejection, with the roll rate dependent upon airspeed and the severity of roll-producing factors such as lateral c.g. eccentricity and ejection during side slip. Rolls generally degenerate into an upright "Dutch roll," which is stopped by deployment of the recovery parachute, at which time a rapid change in crew module attitude occurs. 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 (±30) knots and altitudes below 15,000 feet, the recovery parachute is deployed after a 1.0 second delay. At speeds above 300 (±30) 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 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, 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 (to sea level) 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. Crew members may expect moderate landing impact decelerations for the nominal weight crew module.

## DITCHING ESCAPE SEQUENCE

If the aircraft is ditched, crew module severence 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 FO–19) 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 the left and right sidewall (2, figures 1-11 and 1-26). 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. The seat has a maximum vertical travel of 5 inches. The headrest does not move with vertical seat adjustment.

**CAUTION**

> Placement of objects under the EWO's seat may cause damage to the fire warning system control boxes, which are located on the floor, just forward of the right seat.

### Headrest Adjustment Lever

A headrest adjustment lever (6, figure FO–19), 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.

**CAUTION**

> Placement of hard objects behind the seat back may cause damage to the seat back and the adjustment mechanism.

### Inertia Reel Control Handle

The inertia reel control handle (7, figure FO–19) is provided to lock or unlock the inertia reel.

### Ejection Handles

Two ejection handles (21, figure FO–19), 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 the handle is depressed the handle is released and may be pulled out. Pulling the handle out approximately 1/2 inch will fire the initiator to start the ejection sequence.

**Note**

> The ejection handle safety pins provided must be installed with the T-handles inboard, as shown in figure 1-9, to preclude interference of the pins with seat adjustment.

### Recovery Parachute Deploy Handle

The ring-shaped recovery parachute deploy handle (24, figure FO–19), 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 FO–19), 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 FO–19), 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 FO–19), 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.

**WARNING**

- Do not actuate the severance and flotation handle prior to impact. To do so may result in severe post-landing gyrations and rupture of the bags.
- Do not actuate the severance and flotation handle when personnel are within 50 feet of the crew module because of explosive severance of the metal covers.

### Bilge/Flotation Bag Inflation Pump

The bilge/flotation bag inflation pump (figure 1-32), provides simultaneous inflation of the flotation bags and pumping of water overboard. The pump is operated by

# BILGE/FLOTATION BAG INFLATION PUMP

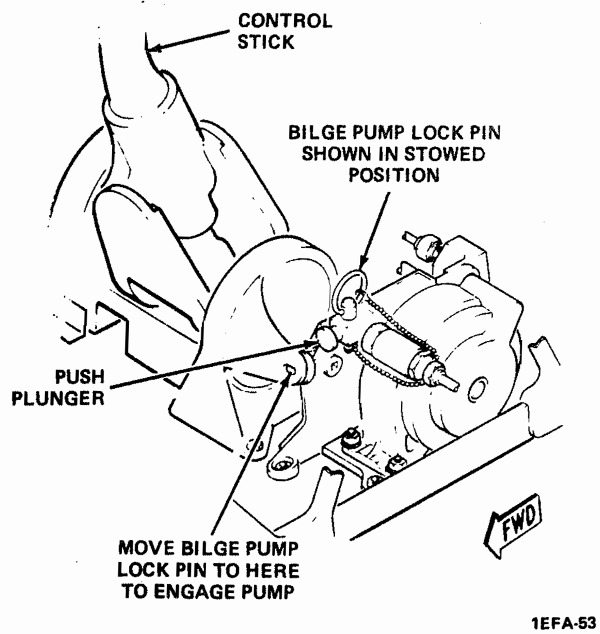



Figure 1-32.

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**

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.

**CAUTION**

Failure to push in the plunger may result in damage to the pump and render it inoperative.

### Chaff Dispenser Control Lever

The chaff dispenser control lever (14, figure FO-19) 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, if installed. 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.

- When the lever is in the ON position, the radio beacon set will be automatically activated as the crew module ejects.

## COMMUNICATIONS EQUIPMENT

For a listing and function of communications equipment see figure 1-33.

### UHF COMMAND RADIO (AN/ARC-164).

The UHF command radio provides two-way communications, automatic direction finding (ADF) in conjunction with the AN/ARA-50 and has a jam resistant capability. 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 1-34 for antenna locations. There are 7,000 channels available in 0.025 megahertz increments in the frequency range from 225.000 to 399.975 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 BDHI and audio to the interphone when operating in the ADF mode. Refer to Section IV for ADF operating procedures. The guard receiver monitors the guard frequency of 243.0 megahertz when the BOTH function is selected. The control panel allows selection of 20 preset channels and manual selection of any frequency in the frequency range of the radio. The upper and lower antenna 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 operating in ADF, the receiver is connected to the ADF loop antenna. Operation of the jam resistant radio in the normal mode is the same; however, operation in the active jam resistant mode requires precision time synchronization, common frequency switching pattern, common frequency hop rate, and a common active net. The operator must program the correct preset channel, time of day (TOD), word of day (WOD), and active net.

#### Time of Day (TOD)

The time of day function allows all radios in synchronization with each other to simultaneously frequency hop for uninterrupted communication. The radio will automatically accept the first TOD signal after power is applied to the UHF radio or the first TOD signal received within one minute of momentarily selecting T on the A-3-2-T knob (11, figure 1-35). TOD may be obtained from a signal transmitted automatically every 10 seconds on a predesignated frequency or from a signal transmitted manually by another suitably equipped UHF radio. TOD synchronization is required for communication in the A mode. Improper synchronization will prohibit proper communication. Reception of garbled communication in the A mode indicates that the receiving radio and the transmitting radio are not in synchronization.

**Note**

In a multiple radio environment, the radio receiving all transmissions garbled is the radio that is out of sync.

#### Word-of-the-Day (WOD)

The word of day function programs the radio with frequency hopping pattern and rate. The WOD program is normally entered prior to takeoff; however, it can be entered during flight. The WOD varies in length and will

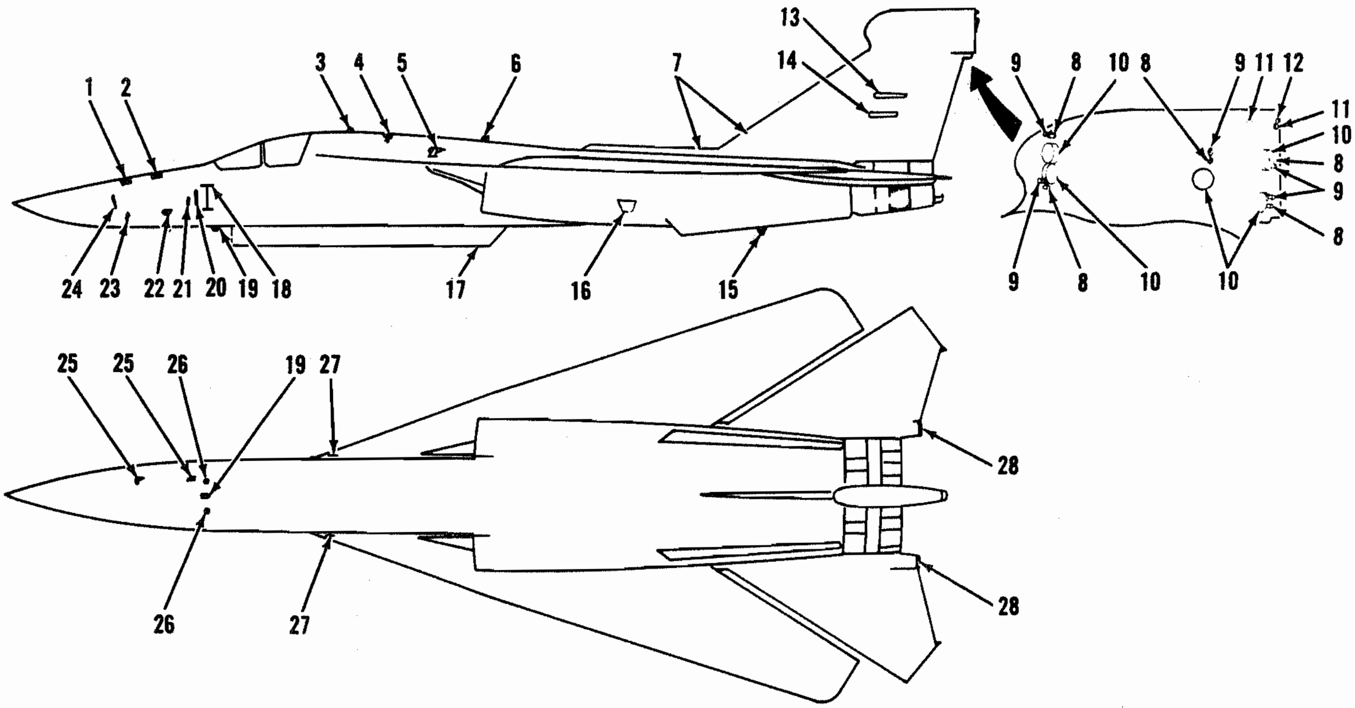
# COMMUNICATIONS AND AVIONICS EQUIPMENT

| Type | Designation | Function | Crew Station | Range |
|---|---|---|---|---|
| UHF Radio | AN/ARC–164 | Air-to-air and air-to-ground voice communication | Both | Line-of-sight |
| UHF ADF | AN/ARA–50 | Provides bearing information to selected UHF transmitters | Both | Line-of-sight |
| HF Radio | AN/ARC–112 | Air-to-air and air-to-ground long range voice communications | 1 Both<br>2 EWO | 5000 miles |
| Radio Beacon Set | AN/URT–27 or –33 | Provides a tone signal for rescue aircraft to home on | Pilot | Line-of-sight |
| Interphone | AN/AIC–25 | Interphone between crew members and monitoring of all communications facilities | Both | |
| IFF (AIMS) | AN/APX–64 | Provides coded IFF replies to an interrogating ground radar station | Both | Line-of-sight |
| TACAN | AN/ARN–118(V) | 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 | 1 Pilot<br>2 Both | Localizer 45 nm Glide slope 25 nm |
| Radar Altimeter | — | Provides precise altitude measurements from 0 to 5,000 feet | Pilot | 0-5000 feet |
| Terrain Following Radar | AN/APQ–110 | Provides all weather, low altitude terrain following, obstacle avoidance and blind letdown capability | Both | Line-of-sight up to 15 miles |
| Inertial Navigation System | AN/AJQ–20 | Provides integrated navigation capabilities in conjunction with other systems in the aircraft | Both | |
| Nav Radar | AN/APQ–160 | All weather navigation and fixtaking | Both | Line-of-sight up to 160 miles |

1 Prior to T.O. 1F-111(E)A-501
2 After T.O. 1F-111(E)A-501

**Figure 1-33.**

# ANTENNA LOCATIONS

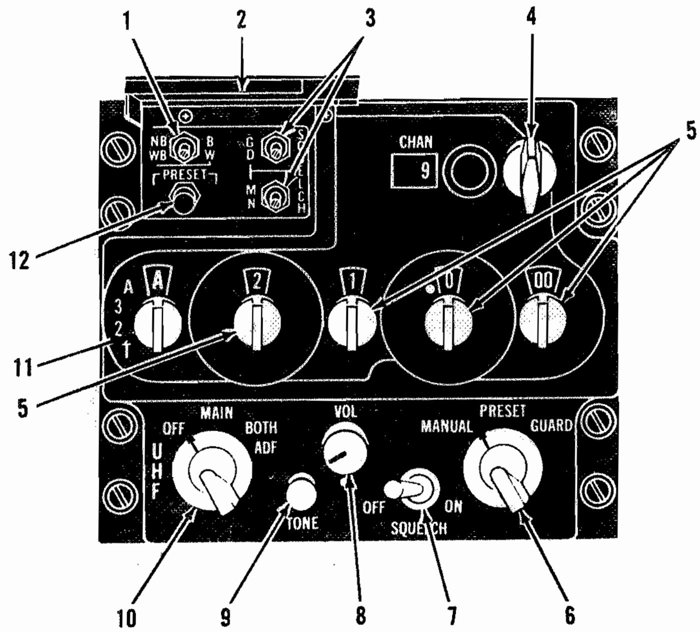



1. GLIDE SLOPE
2. ADF
3. IFF (UPPER) AND UHF DATA LINK
4. RADIO BEACON SET
5. ALQ-137 LOW, MID, HIGH REC AND ALR-62 FWD REC
6. UHF NO. 1 AND TACAN UPPER
7. HF
8. ALQ-99 BAND 9
9. ALQ-99 BAND 8
10. ALQ-99 BANDS 4, 5/6 AND 7
11. ALR-62 AFT REC
12. ALQ-137 LOW AND MID REC
13. ALQ-99 BAND 1 (2)
14. ALQ-99 BAND 2 (2)
15. IFF LOWER
16. ALQ-99 BANDS 1 AND 2 (2)
17. ALQ-99 BANDS 4, 5/6, 7, 8 AND 9
18. LOCALIZER (2)
19. UHF NO. 2 AND TACAN LOWER
20. ALQ-137 LOW BAND XMTR
21. ALQ-137 MID BAND XMTR
22. ALQ-137 OMNI MID BAND XMTR
23. TFR (2)
24. NAV RADAR
25. RADAR ALTIMETER
26. ALQ-137 OMNI LOW AND MID BAND XMTR
27. ALQ-137 HIGH BAND XMTR
28. ALQ-137 HIGH BAND REC AND XMTR

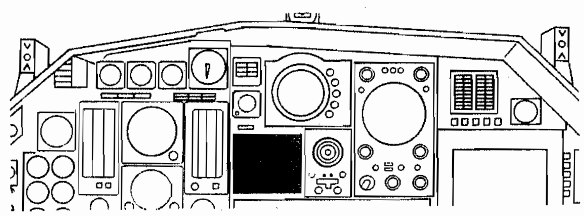
1EFA-99

Figure 1-34.

# UHF RADIO CONTROL PANEL (TYPICAL)




1. BAND WIDTH SELECTOR
2. HINGED ACCESS DOOR/PRESET CHANNEL FREQUENCY CARD
3. SQUELCH ADJUSTMENT
4. PRESET CHANNEL SELECTOR KNOB
5. MANUAL FREQUENCY SELECTOR KNOBS
6. MODE SELECTOR KNOB
7. SQUELCH CONTROL SWITCH
8. VOLUME CONTROL KNOB
9. TONE BUTTON
10. FUNCTION SELECTOR KNOB
11. A-3-2-T KNOB
12. CHANNEL PRESET PUSHBUTTON



A-1EFA-19

**Figure 1-35.**

R require one to six channels. For a WOD frequency entry,
R start with channel 15 or first WOD frequency, using the
R same method used in entering preset frequencies in the
R normal mode. This procedure is repeated for channels
R 16, 17, 18, 19, and 20 in that order, as necessary. Cycle
R out of channel 20, then back to channel 20. Successively
R select each preset channel used for WOD storage. After
the channel 15 entry, or end of WOD, a double tone signal will be heard indicating that the radio has accepted all WOD entries. If the radio is turned off or if channel 20 is selected it is necessary to initiate another transfer of the WOD into memory, starting with channel 20 and rotating through the remaining channels until a double tone signal is heard. A double tone signal indicates that WOD transfer is complete.

Note

> The radio will not function in the A mode if WOD is not in memory. If the A mode is selected without WOD in memory a steady tone will be heard in the head set.

Channels containing WOD elements cannot be used in the normal preset function. WOD channels can be selected and transmissions made but they will transmit on the last frequency seen prior to selecting the WOD channel.

### Net Number

The net number programs the entry point in the WOD frequency hopping pattern allowing for multiple radio net operation using a common WOD and TOD. Selecting A on the A-3-2-T knob puts the radio in the active mode and programs the radio to use the next three numbers following the A. These three digits are defined as the net number. The radio will function in the A mode when WOD and TOD are programmed, manual or preset selected, and appropriate net number set in the three digits following the A. If WOD is not in memory or TOD has not been programmed a tone signal will be heard in the headset.

### UHF Radio Function Selector Knob

The UHF radio function selector knob (10, figure 1-35) is marked OFF, MAIN, BOTH, and ADF. Rotating the knob to MAIN activates the receiver-transmitter unit for normal transmission and reception on the channel selected; the guard receiver is inoperative. Rotating the knob to BOTH also activates the receiver-transmitter unit for normal use and in addition activates the guard receiver to allow monitoring guard frequency. In ADF, the receiver is switched to the ADF loop antenna and bearing information and audio are supplied to the BDHI and interphone respectively, provided BDHI mode selector switch is in NAV. Audio range is reduced considerably when operating on the ADF antenna; it may be necessary to return the knob to MAIN or BOTH for better reception. If the microphone 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 Mode Selector Knob

The mode selector knob (6, figure 1-35) is marked MANUAL, PRESET and GUARD. MANUAL is used for frequencies that are selected by the manual frequency selector knobs. PRESET is used when selecting one of the 20 preset frequencies. This position is also used when programming the 20 preset channels. GUARD 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 (4, figure 1-35) permits selection of one of 20 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. The number of the channel selected is displayed in a window beside the knob. Frequencies for each channel are written on the channel frequency log. Frequencies of the preset channels can be changed during flight.

### UHF Radio Manual Frequency Selector Knobs

Five manual frequency selector knobs (5 and 11, figure 1-35) allow manual selection of frequencies. Manual frequency selection can be made in steps of 0.025 megahertz from 225.000 to 399.975 megahertz. The first, second, and third knobs (from the left) select the hundreds, tens and units digits respectively of the desired frequency. The fourth knob selects the tenths digits (0 through .9) and the fifth knob selects the hundredths and thousandths digits (00, 25, 50, or 75). The digits selected by each knob are displayed in a window above the knob. The left knob (11, figure 1-35) is marked A,3,2,T. The A position selects the active jam resistant mode, 2 and 3 select the 100's digit of the desired frequency in the normal mode, and T is a momentary spring-loaded position, which enables the radio to receive a new time of day (TOD). The A and T positions override the hundreds digit in both manual and preset operation. The second, third, and fourth knobs (from the left) are used to select the net number for operation in the active jam resistant mode.

### UHF Radio Volume Control Knob

The volume control knob (8, figure 1-35) is provided to set UHF receiver volume.

### UHF Squelch Switch

The squelch switch (7, figure 1-35) is marked OFF and ON. Placing the switch to OFF turns off the squelch. Placing the switch to ON turns the squelch on. If UHF reception is breaking up and intermittent, switching to OFF may restore reception.

### UHF Radio Tone Button

The tone button (9, figure 1-35) is activated when the UHF radio is in operation. Depressing the tone button will interrupt reception and transmit a continuous 1020 Hz tone signal on the selected frequency when operating in the normal mode (TOD clock not in operation).

When TOD clock is in operation, depressing the tone button will activate a 1667 Hz signal momentarily, followed by a steady 1020 Hz tone signal.

### UHF Radio Channel Preset Pushbutton

The channel preset pushbutton (12, figure 1-35) is used to set or change preset channel frequencies. The button is only effective when the mode selector knob is in the PRESET position and the function selector knob is in the

R MAIN or BOTH position. 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 preset channel. The button is located behind the preset channel frequency log (2, figure 1-35).

### UHF Band Width Selector

The band width selector (1, figure 1-35) marked BW is located behind the preset channel frequency log and has two positions marked NB (narrow band) and WB (wide band). The switch must always be in the WB position until such time as NB operation is authorized by USAF directive.

### UHF Squelch Adjustment

The two squelch adjustments (3, figure 1-35), located behind the preset channel frequency log and labeled GD (guard) SQUELCH and MN (main) SQUELCH, are currently inoperative.

### Transmitter Selector Knob

A transmitter selector knob (3, figure 1-39), labled HF, UHF, and INT, is located on each interphone control panel to select either the HF radio, UHF radio, or interphone for transmission.

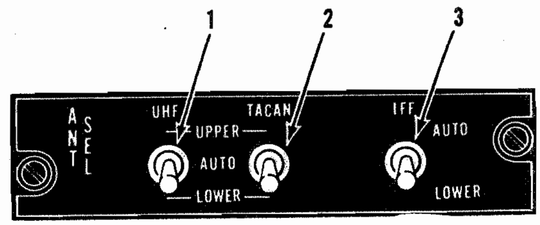
### Microphone Switches

A three-position, pivot-type microphone switch, marked TRANS and INPH with an unmarked off position, is located on the right throttle (1, figure 1-1). 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 intephone panel. The EWO is provided with foot switches for these functions. The left switch marked MIKE is for transmit and the right marked ICS is for interphone. The microphone switches are spring-loaded to the off position.

### UHF Radio Antenna Selector Switch

The three-position antenna selector switch (1, figure 1-36) 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 LOWER or UPPER controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

# ANTENNA SELECT PANEL




1. UHF RADIO ANTENNA SELECTOR SWITCH
2. TACAN ANTENNA SELECTOR SWITCH
3. IFF ANTENNA SELECTOR SWITCH

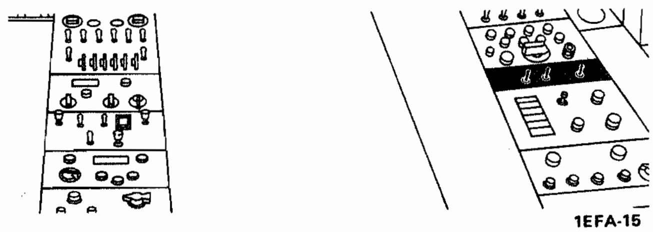



Figure 1-36.

## HF RADIO (AN/ARC-112)

The HF radio provides long range high frequency single side-band communications. The radio operates in three modes: USB, LSB, and AM (upper side band with carrier for receivers without single side band reception capability). There are 28,000 channels available in 1 kilohertz increments in the frequency range of 2,000 through 29,999 kilohertz. Components of the system include a receiver-transmitter (RT) unit, amplifier power supply, antenna, antenna coupler, antenna coupler control and control panel. The RT unit, amplifier power supply and antenna coupler control are located in the right forward electronic bay. The antenna coupler is located in the aft fuselage below the antenna which is a part of the vertical stabilizer and dorsal fin. Refer to figure 1-34 for antenna location. The system incorporates self test features for maintenance troubleshooting. Refer to Section IV for HF radio operation.

### HF Radio Mode Selector Knob

The HF radio mode selector knob (5, figure 1-37) has six positions marked OFF, USB, LSB and AM with two TEST positions marked RCVR EXC and XMIT. Placing the knob

# HF RADIO CONTROL PANEL

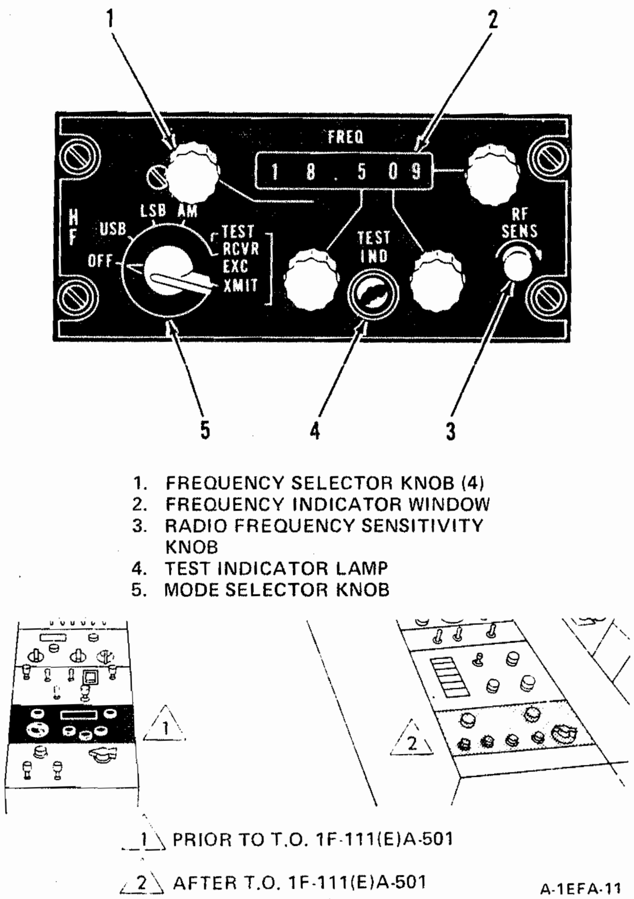



**Figure 1-37.**

to USB or LSB provides upper or lower side-band transmission and reception respectively. The AM position provides upper side-band transmission with carrier for reception by receivers without single side-band capability. The TEST positions provide self test features used for maintenance troubleshooting.

### HF Radio Frequency Selector Knobs

Four HF radio frequency selector knobs (1, figure 1-37) provide a means of setting desired frequencies. Each knob has an indicator line drawn to the window(s) it controls.

### HF Radio Frequency Indicator Window

The HF radio frequency indicator window (2, figure 1-37) displays five digits indicating the frequency selected for transmission and receiving. Each window has an indicator line drawn to its corresponding frequency selector knob.

### HF Radio Frequency Sensitivity Knob

The radio frequency sensitivity knob (3, figure 1-37) labeled RF SENS, provides an adjustment for receiver sensitivity.

### HF Radio Test Indicator Lamp

The green test indicator lamp (4, figure 1-37) provides both self test and malfunction indications. The lamp is used in conjunction with the test positions for system self-test during maintenance troubleshooting. During normal operation, a flashing lamp will indicate a system malfunction.

## RADIO BEACON SET

A radio beacon set (figure 1-38), located behind the pilot's seat, 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. When the on-off switch is placed to ON and the chaff control lever (14, figure FO–19) is ON, the radio will be automatically actuated whenever ejection occurs.

## INTERPHONE (AN/AIC–25)

The interphone provides the following functions: Communications between crewmembers and between crewmembers and ground crew; individual monitoring and volume control for UHF radio, HF radio, TACAN, ILS, TFR aural commands, TTWS, JSS audio and hot mic and call capability. Two identical interphone control panels (figure 1-39) are provided for the crew members. Interphone stations for ground crew operation are located in the nose wheel well, main landing gear well and ground power receptacle. Power is applied to the interphone whenever power is on the aircraft.

# RADIO BEACON SET

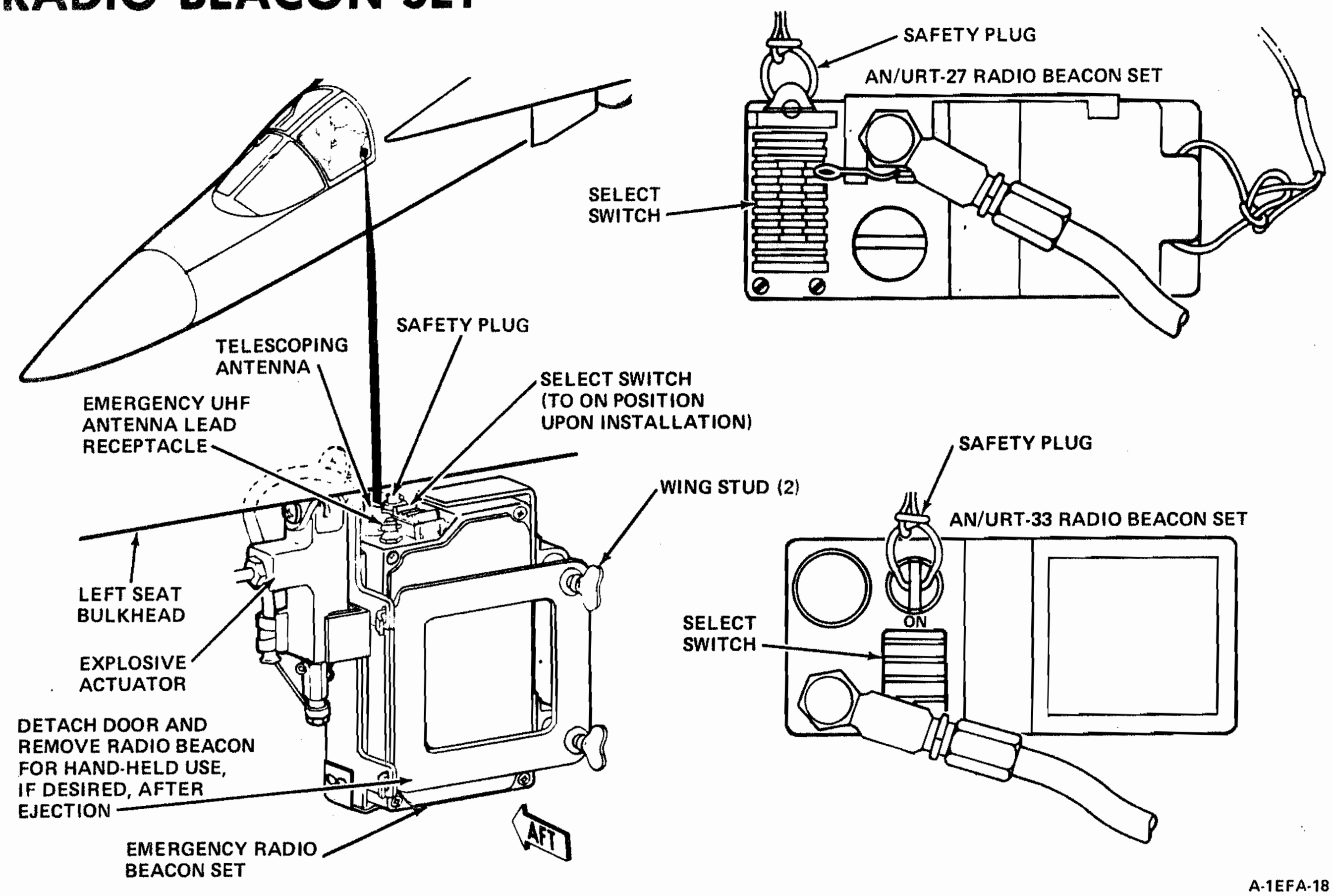



**Figure 1-38.**

### Monitor Knobs

Eight push-pull monitor knobs (1, figure 1-39) are marked and monitor the functions as follows:

| | |
|---|---|
| INT | Interphone |
| UHF | UHF Command Radio |
| HF | HF Radio |
| ILS | ILS and Localizer, and TFR Aural Command |
| TACAN | TACAN Identification |
| RHAWS | ALR–62 (TTWS) |
| MISSILE | JSS Audio |
| HOT MIC LISTEN | Hot Mic Reception |

Other signals fed to the interphone panel are the landing gear warning signal, the stall warning signal, and the LCCS signal. The monitor knobs are pulled out to turn on and pushed in to turn off. Each knob (except HOT MIC TALK) may be rotated for volume control.

### Master Volume Control Knob

A master volume control knob (4, figure 1-39) controls the volume of all 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 Button

A push-pull (Hot MIC) hot microphone talk button (5, figure 1-39) provides a continually operating microphone when it is pulled. When this button is pulled, the crewmember can transmit without using the microphone switch; however, the crewmember at each station must have his hot mic listen monitor knob pulled to receive the transmission.

# INTERPHONE CONTROL PANEL

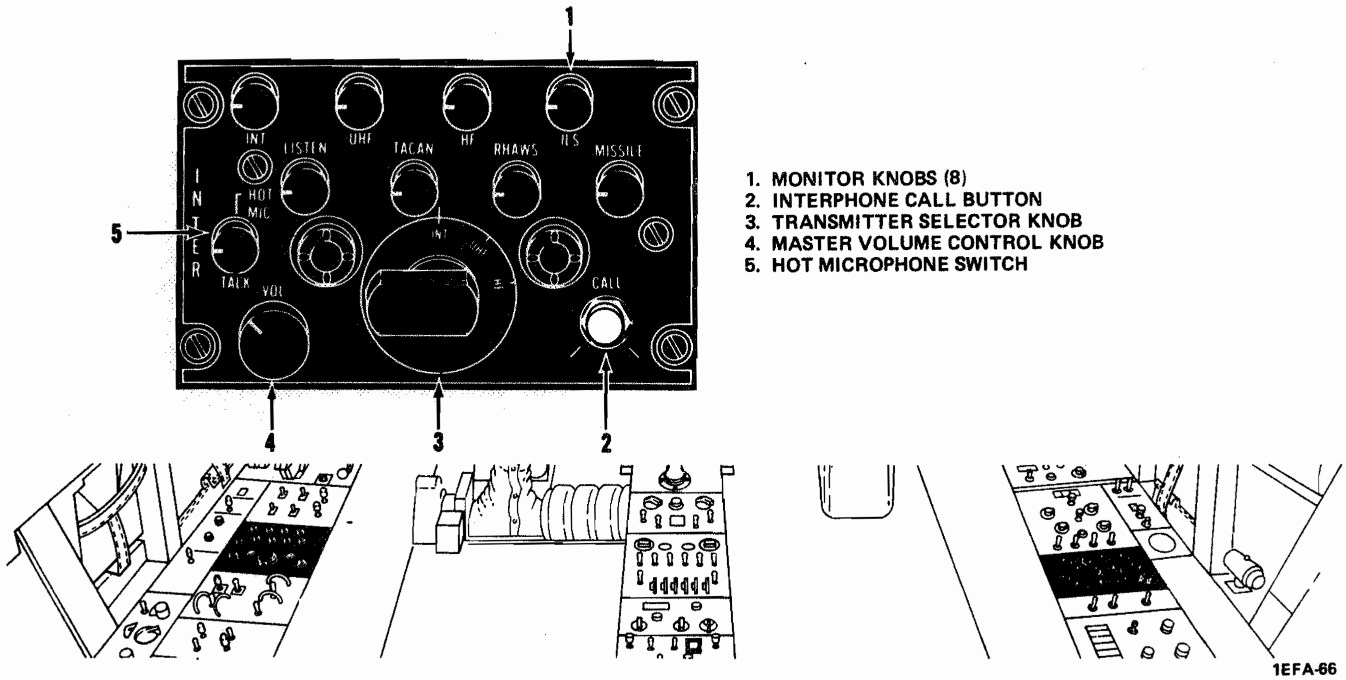



Figure 1-39.

### 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 1-39) permits either crewmember to call the other crewmember 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 knobs or transmitter selector knob at either station.

### Transmitter Selector Knob

A three-position transmitter selector knob (3, figure 1-39) is marked UHF, HF and INT. In HF or UHF only the radio transmitter selected will be keyed when the pilot's microphone switch is moved to the TRANS position or the EWO's MIC footswitch is depressed. 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. The INT position of the transmitter selector switch has no operational function.

### Exterior Interphone Stations

Exterior interphone stations in the nosewheel well, the ground power receptacle, 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.

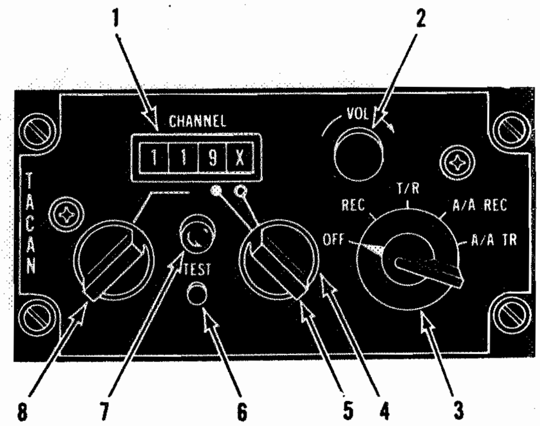
## TACTICAL AIR NAVIGATION SYSTEM (AN/ARN-118(V))

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 in each mode (X or Y). 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 1-34), 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, unless paired with a suitable bearing transmitter type set. The TACAN works in conjunction with the ISC, the BDHI, the HSI, and ADI, and through the interphone control panel for audio output. When TACAN is selected on the ISC, all heading information is supplied by the AFRS. For all other ISC selections (except AIR/AIR), TACAN bearing information is still provided by the AFRS, but all other integrated flight instrument system heading data is supplied by the inertial nav system. The TACAN control panel (figure 1-40) is located on the center console. Refer to Section IV for TACAN operation procedures.

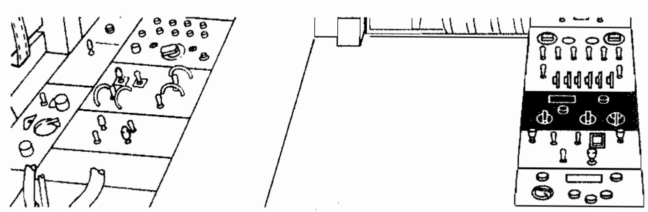
## TACAN FUNCTION SELECTOR KNOB

The function selector knob (3, figure 1-40) is marked OFF, REC, T/R, A/A REC, and A/A TR. In the REC position, the set will receive bearing and audio identity signals only, range information will not be displayed because the TACAN transmitter is not on. In the T/R position, both the receiver and the transmitter are operative, the system will receive and display both range and bearing to the station being interrogated, and radio identity signals are fed into the interphone system. When A/A REC is selected, the TACAN system receives and measures bearing to a suitably equipped, cooperating aircraft and calculates the relative bearing to that aircraft. When A/A TR is selected, the TACAN system interrogates a suitably equipped, cooperating aircraft and receives and calculates the slant range and relative bearing to that aircraft. If the interrogated aircraft is not equipped with bearing producing equipment, only slant range is calculated. To operate in the air-to-air mode, the channels selected in both aircraft 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. The AN/ARN–118V transmits range only, not bearing.

## TACAN CONTROL PANEL




1. CHANNEL DIGITAL DISPLAY
2. VOLUME CONTROL KNOB
3. FUNCTION SELECTOR KNOB
4. X-Y CHANNEL SELECTOR KNOB
5. UNITS CHANNEL SELECTOR KNOB
6. PRESS-TO-TEST BUTTON
7. TEST INDICATOR LAMP
8. HUNDRED-TENS CHANNEL SELECTOR KNOB



1EFA-12

**Figure 1-40.**

**Note**

- As many as five aircraft can lock on to a "parent" transmitter in the air-to-air mode. However, the radius of operation for all aircraft involved will be limited to a distance equal to four times the distance between the transmitter aircraft and the nearest receiver.
- Interference between the IFF and TACAN channels 1 thru 9, 63 thru 72, and 126 can prevent air-to-air TACAN lock-on. Avoid using these channels in the air-to-air modes.

## TACAN CHANNEL SELECTOR

The channel selector located on the TACAN control panel, consists of three separate knobs (4, 5, 8, figure 1-40). The hundred-tens knob and the units knob are used to select the TACAN channel. The X-Y channel selector knob provides for the selection of the X or Y mode of operation. The channel digital display (1, figure 1-40) shows the channel selected with the three channel selector knobs.

## TACAN VOLUME CONTROL KNOB

A volume control knob (2, figure 1-40) controls the volume of the audio identity code.

## TACAN ANTENNA SELECTOR SWITCH

The TACAN antenna selector switch (2, figure 1-36) is marked UPPER, AUTO, and LOWER. Placing the switch to AUTO causes the antenna selector to control the antenna switching relay to select the correct antenna. Placing the switch to UPPER or LOWER controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

## TACAN SELF TEST

The press-to-test button (6, figure 1-40) is used to manually initiate TACAN self-test. The test indicator lamp (7, figure 1-40) lights when a malfunction occurs during automatic or manual self-test. The manual self-test is initiated by depressing the press-to-test button; automatic self-test is initiated when the received signal becomes unreliable or is lost.

## INSTRUMENT LANDING SYSTEM (AN/ARN-58)

The instrument landing system (ILS) provides the capability of making instrument approaches to runways equipped with ILS. 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 ISC for display on the ADI and HSI. 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 marker beacon lamp. Power is applied to the marker beacon receiver whenever power is on the aircraft. Refer to "Instrument Flight Procedures," Section VII for instrument landing system operating procedures. Refer to figure FO–15 for ILS indications in the various knob position.

## ILS FREQUENCY SELECTOR KNOB

The frequency selector knob (2, figure 1-41) allows individual selection of 20 ILS channels ranging in localizer frequencies from 108.1 to 111.9 megahertz in 0.2 megahertz increments. The knob has a detent position for each channel. Each localizer frequency selected is automatically paired with a glide slope frequency. 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 1-41) is marked POWER and OFF. When the switch is placed to POWER, power is applied to the localizer and glide slope receivers.

## ILS VOLUME CONTROL KNOB

The volume control knob (4, figure 1-41) adjusts the volume of the localizer station identification signal.

## MARKER BEACON LAMP

The marker beacon lamp (figure FO–5) provides a visual station signal when the aircraft is over a marker beacon facility. When lighted, the words MARKER BEACON are displayed in green.

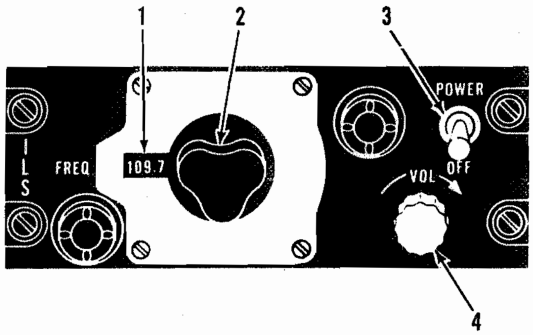
**Note**

> The marker beacon lamp may blink during HF radio transmission on some frequencies due to electromagnetic interference. This is normal.

## IFF SYSTEM (AN/APX-64)

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 R

# ILS CONTROL PANEL




1. FREQUENCY WINDOW
2. FREQUENCY SELECTOR KNOB
3. POWER SWITCH
4. VOLUME CONTROL KNOB

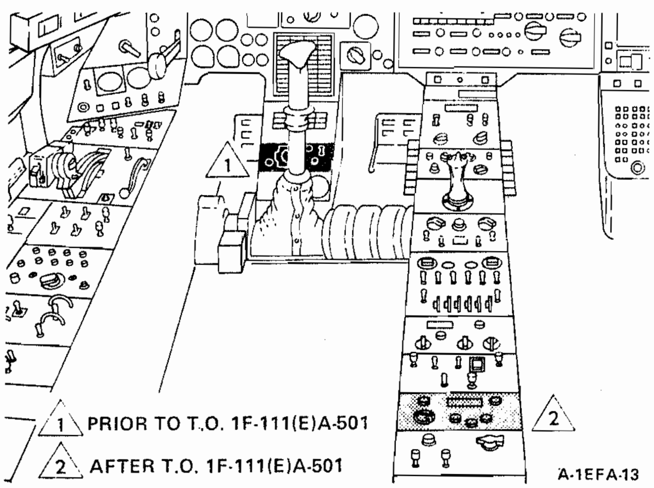



**Figure 1-41.**

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 1-34 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. Mode 2 and 4 code settings are set into the 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. Mode C provides altitude information from the CADC to the ground in 100 foot increments.

**Note**

With a known or suspected CADC malfunction, the mode C output should be verified (if possible) and turned off, if necessary, as this could result in the transmission of erroneous altitude information.

An optional low sensitivity setting provision restricts sensitivity so that replies are made only to local interrogations.

## IFF MASTER CONTROL KNOB

The five-position IFF master control knob (4, figure 1-42) is marked OFF, STBY, LOW, NORM and EMER. The OFF position removes power from the set and also zeroizes mode 4 code settings. In STBY, the equipment is turned on and warmed up but will not transmit. In LOW, only local (strong) interrogations are recognized and answered. With NORM selected full range recognition and reply occurs. 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.

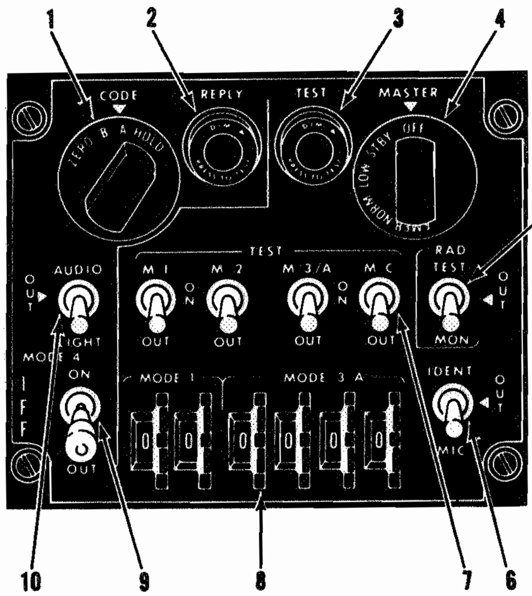
## IDENTIFICATION-OF-POSITION SWITCH

The identification-of-position (I/P) switch (6, figure 1-42) is 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 in reply to a mode 1 or 3/A interrogation as long as a microphone switch is closed 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 in the neutral (off) or interphone position, transmission of the I/P pulse groups will be withheld. Placing the switch to the OUT position prevents transmission of I/P groups.

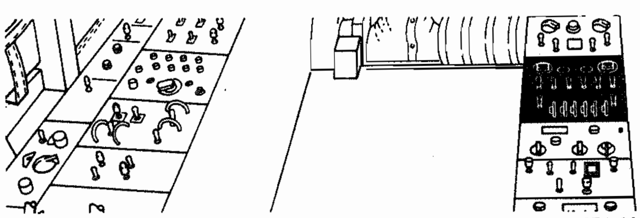
## IFF ANTENNA SELECTOR SWITCH

The two-position antenna selector switch (3, figure 1-36) 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

# 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



1EFA-14

Figure 1-42.

lower antenna to provide thorough antenna pattern coverage. When the antenna selector switch is placed to LOWER, the lower antenna will be used to receive and reply to interrogation signals.

## IFF MODE SELECT/TEST SWITCHES

Four mode select/test switches (7, figure 1-42) 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. IFF modes 1, 2, 3/A and C can be tested by holding the associated mode select/test switch in TEST while the RAD test/monitor switch is in OUT or MON and the MASTER control knob is in LOW or NORM. If the test lamp lights while the mode select/test switch is held in TEST, the related mode is functioning properly.

**Note**

> When any of the four mode select/test switches is placed to ON or TEST, and the rad test/ monitor switch is in MON, the test lamp may light, either continuously or periodically, due to external interrogations. This also validates operation of the mode selected.

## RAD TEST/MONITOR SWITCH

The three position rad test/monitor switch (5, figure 1-42) is used for control of the radiation test and monitor provisions. The switch has three positions marked RAD TEST, OUT and MON and is spring-loaded from the RAD TEST to the OUT position. With the switch in MON, the test lamp will light when external or self-test interrogations are received and answered. The RAD TEST position is used by maintenance personnel for a more thorough self-test of the system. When the switch is placed to OUT, the radiation test and monitor circuits are inoperative.

## CODE SELECTOR WHEELS

Two sets of thumb actuated code selector wheels (8, figure 1-42) 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.

## MODE 4 CONTROL SWITCH

The mode 4 control switch (9, figure 1-42) is marked ON and OUT. Mode 4 operation is enabled by placing the switch to ON with the master control knob in LOW, NORM, or EMER. 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 1-42) is marked ZERO, A, B, and HOLD. The knob must be pulled out before it can be moved to the ZERO position, 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 (figure 1-42) is in any position except OFF. Both codes will be automatically zeroized when the IFF is turned off after landing. However, positioning the knob momentarily to the spring-loaded HOLD position, after landing and waiting a minimum of 15 seconds before placing the control switch to OFF, will retain both codes with power off. The HOLD function is operative only when the landing gear handle is in the DN position, and when desired, should be selected after landing and prior to turning off system power.

## MODE 4 MONITOR CONTROL SWITCH

The mode 4 monitor control switch (10, figure 1-42) is 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 lighting of the reply lamp on the IFF control 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 1-42) 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 lower caution lamp panel (figure FO-5). 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 1-42) will light when a mode select/test switch is held in TEST, to indicate the related mode is operable. When any of the mode select/test switches are placed to ON, and the rad test/monitor switch is in MON, the lamp may light because of external interrogations.

## INERTIAL-NAVIGATION SYSTEM (AN/AJQ-20A)

The inertial-navigation system is a self-contained dead reckoning analog inertial system. The system consists of a stabilized platform (SP), a navigation computer (NC), and a remotely located flux valve.

The system provides the following functions:

- Computed aircraft position in latitude and longitude.
- Range and bearing to target or destination for navigation steering.
- 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, ADI and HSI for navigation.
- Constant ground track steering signals to the autopilot.
- Drift angle.
- Slant range and bearing to a fixpoint for nav radar crosshair laying.
- Provisions for tracking no-show radar targets by offset radar sighting.
- Position correction via radar fix-taking.
- Present position and true heading.
- Pushbutton overfly fix-taking capability for present position correction.
- Altitude calibration by use of nav 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 dives.
- A backup capability in case of stabilized platform failure.
- Simple self-test features to isolate system troubles to the stabilized platform or navigation 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 groundspeed. 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 the 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 1/4 the time required for gyrocompass alignment. 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 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 inertial 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 analog data from the NC corresponding to aircraft latitude. R

## NAVIGATION COMPUTER

The navigation computer (NC), located in the center instrument panel, is a self-contained navigation steering and radar sighting 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 airspeed and pressure altitude from the CADC and magnetic heading from the system flux valve. The NC provides all the computing, control, and display functions for the inertial 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 AFRS to substitute for SP data.

## CONTROLS AND INDICATORS

### Inertial Nav Mode Selector Knob

The ten-position inertial nav mode selector knob (13, figure 1-43) is labeled MODE SEL. The knob controls warm-up, power turn on, and system operating modes. The HEAT and ALIGN positions set the system up for operation. The GREAT CIRCLE through SHORT RANGE 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. This knob has detents at all positions and requires a pullout to rotate from ALIGN to any normal navigation position. The knob markings and functions are as follows:

1. OFF - All power off.
2. HEAT - All power off except for inertial platform heater power - provided the platform alignment control knob is in any position except OFF/AUX NAV.
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. 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. 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 - Not operational.
7. AUTO BOMB - Not operational.
8. TRAIL BOMB/MAN BAL - Not used.
9. RANGE BOMB/MAN BAL - Not used.
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.

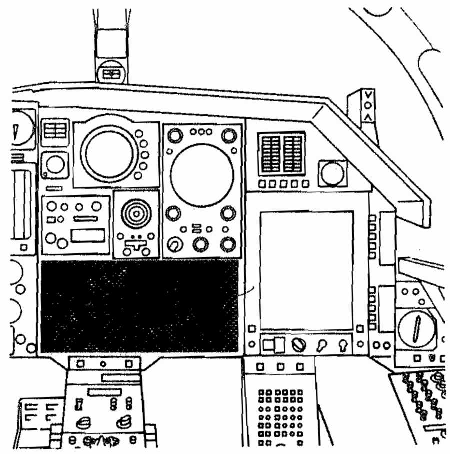
### Inertial Nav Fix Mode Selector Buttons

Only one of the nine inertial nav fix mode selector pushbuttons (20, figure 1-43) can be depressed at any given time. With each new mode selection, the preceding mode will disengage. In addition, a bar on the panel below the engaged button will become visible, indicating that the button has been depressed and the indicated mode selected. To disengage an operating mode without engaging another mode, lightly depress any other button. 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 values, then both counters drive together indicating present position until another fix mode selector pushbutton is depressed. Also deactivates the target bearing, slant range, course angle, and distance to destination servos.
2. TARGET - Selects position set in the destination position counters as the radar sighting point and the navigation 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 destination.
4. MAN FIX - Applies energizing voltages to the present position hold and present position correction buttons. Removes nav 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 knobs.
5. DEST STORAGE 1, 2, and 3 - Slaves destination shafts to the stored 1, 2, or 3 destination. Removes nav 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.

# INERTIAL 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 GROUND-SPEED COUNTERS (2)
10. GO LAMP
11. ALTITUDE/TEST SELECTOR KNOB
12. TRAIL/RANGE COUNTER
13. 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. NAV FIX MODE SELECTOR BUTTONS
21. OFFSET RANGE AND OFFSET BEARING COUNTERS (2)
22. DESTINATION POSITION COUNTERS



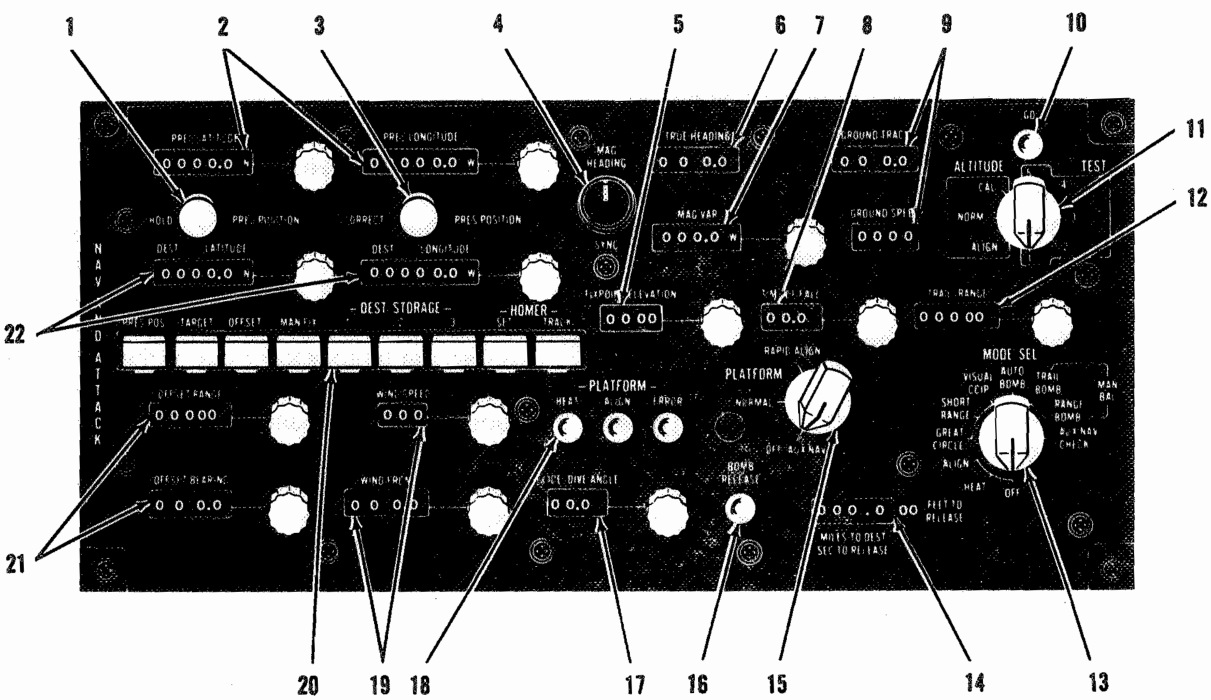



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Figure 1-43.

**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.

### Present Position Correction Button

The present position correction button (3, figure 1-43) labeled CORRECT PRES POSITION 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 1-43) labeled HOLD PRES POSITION, 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 resetting 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 1-43) marked OFF/AUX NAV, NORMAL and RAPID ALIGN, must be pulled out before it can be rotated from any position. The knob controls the alignment modes of the SP when the inertial nav mode selector knob is positioned to ALIGN. In the OFF/AUX NAV position, the SP is completely de-energized. In the NORMAL position, the SP may be aligned two ways, (1) gyrocompass or (2) alignment to preset magnetic variation. Gyrocompass alignment is the normal alignment, and is much more precise. In gyrocompass alignment, stabilized platform goes through two modes of azimuth alignment: (1) Alignment to true north as defined by the inertial 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 inertial nav mode selector knob to a navigate mode after the SP 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 has previously been stored in the system and the aircraft not moved. In this mode, the SP will align to the stored heading and will be ready for operation in less than 1/4 the time required for gyrocompass alignment.

### Altitude/Test Selector Knob

The altitude/test selector knob (11, figure 1-43) labeled ALTITUDE TEST, 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 nav 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 inoperative.

**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 1-43) are labeled HEAT, ALIGN and ERROR. The amber heat indicator lamp provides a monitor of the platform heat signal from the SP. When the system is activated, the lamp will light until the gyroscope temperature reaches the required operating level. 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 inertial nav mode selector knob to a nav mode, platform alignment will be degraded in proportion to aircraft movement, thereby rendering the SP 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 inertial 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 inertial nav platform alignment control knob is in the OFF/AUX NAV position. The lamp will light, should the SP turn itself off, when the inertial 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 FO–5), will light when either attitude or heading information from the SP is interrupted or becomes unreliable due to a platform malfunction. The lamp will light: (1) Any time the inertial nav system is in an align mode. (2) Any time the inertial 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 inertial 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 1-43) is used to monitor the navigation computer self-test circuits. The lamp will light when the altitude/test selector knob in placed to TEST positions 1, 2, or 3, provided the proper settings have been set on the panel, and the inertial 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 inertial nav system is operational.

### Present Position Counters

Two present position counters (2, figure 1-43) labeled PRES LATITUDE and PRES LONGITUDE, display the geodetic latitude and longitude utilized as the aircraft position coordinates. The counters are continously and automatically up-dated by inputs of true north and east velocity components from the SP 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 handset 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 nav radar tracking handle.

### Fixpoint Elevation Counter

The fixpoint elevation counter (5, figure 1-43) 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 TARGET selector button is 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 or radar cursor laying functions. Two destination position counters (22, figure 1-43), labeled DEST LATITUDE and DEST LONGITUDE, display destination position in latitude and longitude and 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 nav 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 these counters should be set to the nearest 0.025 minutes.

### True Heading Counter

The true heading counter (6, figure 1-43) is labeled TRUE HEADING. Except in self-test, the counter continuously displays the computer aircraft heading relative to true north derived from (1) inputs of true heading from the SP during all normal navigation modes, and (2) magnetic heading input from the AFRS and handset magnetic variation during all auxiliary navigation modes.

### Magnetic Variation Counter

The magnetic variation counter (7, figure 1-43) labeled MAG VAR, 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 update the navigation computer true heading.

### Magnetic Heading Synchronization Indicator

The magnetic heading synchronization indicator (4, figure 1-43) 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 inertial 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 gyrocompass 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 1-43) are labeled GROUNDTRACK and GROUNDSPEED. Except 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 SP during all normal navigation modes, and (2) airspeed input from the CADC, 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 1-43) 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 SP and airspeed from the CADC. 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 256 knots and wind direction is 0 to 360 degrees.

### Glide/Dive Angle Counter

The glide/dive angle counter (17, figure 1-43) 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 the AILA mode. The resultant steering information is displayed on the ADI. The glide/dive angle counter limits are from 00.0 to 99.9 degrees.

**Note**

Forcing glide/dive angle counter past lower (00.0) or upper (99.9) limits could result in misalignment of counter and possible failure of glide/dive related functions.

### Offset Range and Offset Bearing Counters

The offset range and offset bearing counters (21, figure 1-43) labeled OFFSET RANGE and OFFSET BEARING, are handset by the operator. The maximum offset range that can be set is 99,990 feet. 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 1-43) 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 inertial nav mode selector knob is positioned to 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).

### Trail/Range Counter

The trail/range counter (12, figure 1-43) is operational but not used.

### Time of Fall Counter

The time of fall counter (8, figure 1-43) is operational but not used.

### Bomb Release Lamp

The bomb release lamp (16, figure 1-43) is operational but not used.

## PRINCIPLES OF OPERATION.

The inertial-navigation system generally utilizes electromechanical analogue computing circuits to provide continuous solutions to navigation problems, as represented by operator inserted destination/target position data and derived or sensed data for aircraft position attitude and velocity. (Refer to figure 1-44.) 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. The inertial nav flux valve is the system's primary magnetic heading reference. A more detailed discussion of each unit and its interface with other units and systems is given below. For inertial navigation system operating procedures, refer to Section IV.

### Stabilized Platform (SP)

COMPONENTS

The SP 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. 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

# INERTIAL NAV — SUBSYSTEM TIE-INS

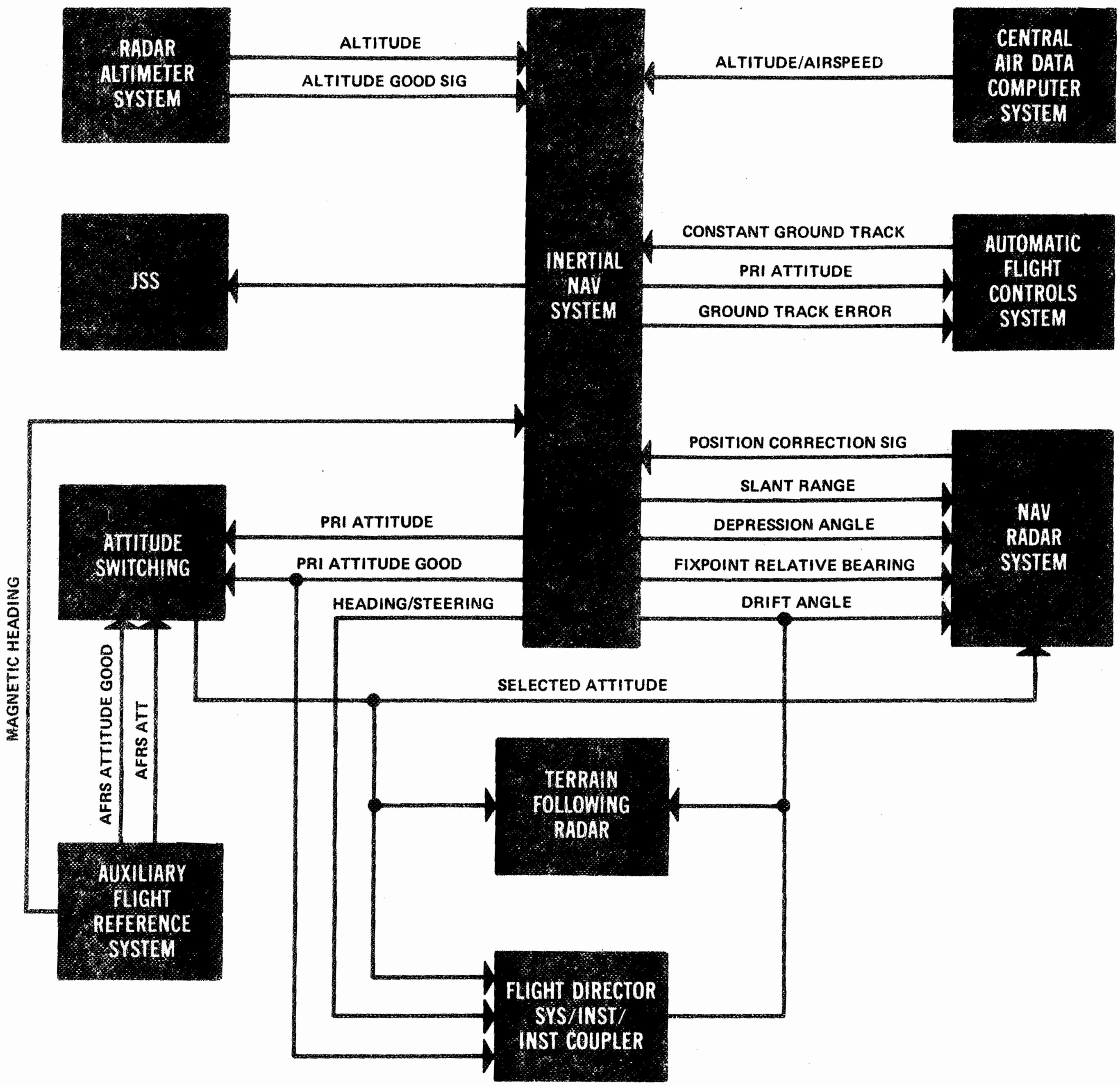



Figure 1-44.

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 transmit 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 requires 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 the 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 inertial nav mode selector knob on the NC, and by self-contained automatic circuits. The previously noted temperature control circuit is activated when the inertial 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 inertial 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 inertial 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 inertial nav mode selector knob 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 inertial nav mode selector knob to HEAT, at least momentarily, to activate the SP, before going to ALIGN. This step is not necessary if inertial 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 inertial 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 inertial 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 transmission to the flight director 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 light 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 inertial nav mode selector knob 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 inertial 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 inertial 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 inertial 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 inertial nav mode select knob is placed to ALIGN, provided that the platform alignment control knob is in NORMAL or RAPID ALIGN and the inertial nav mode selector knob has been cycled through the HEAT position (not necessary if setting SP to rapid align mode after landing if inertial 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 inertial 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.

LEVELLING

When the 30 second time period has expired, with the inertial 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 accuracy 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 inertial 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 inertial nav mode selector knob is in ALIGN, the operator may then set the inertial 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 inertial 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 elapsed 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 inertial 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 inertial 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 inertial 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 inertial 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 5-3/4 minutes. 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. 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 inertial 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 inertial 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 main 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 inertial 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 nav radar antenna pitch control circuit, the ADI pitch and roll 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 nav 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. In the coarse align and levelling phases of alignment, the NC uses this data to devlop 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 servo-mechanism, 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 whenever 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 inertial 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 where 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 the present position latitude or destination latitude counter and another is used for either the 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 inertial 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 positon data point. As controlled by the inertial 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 inertial nav mode selector knob is in GREAT CIRCLE, spherical trigonometric functions are provided; in all other positions of the inertial 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.

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 ensure 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 nav radar ground velocity or ground auto mode cursor drift.

6. Unequal nav 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.

**WARNING**

With one or more of the above indications; disengage/do not engage autopilot, pullup 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.

## NAV RADAR (AN/APQ-160)

The nav radar provides all weather navigation air-to-ground and air-to-air tracking 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/IR indictor panel, a nav radar control panel, and a tracking control handle, all located for operation by either crewmember.

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 inertial nav system. The antenna may be positioned in elevation (tilt) within ±30 degrees of the horizontal by a depression angle signal provided by the inertial nav system or by manual operation of a tilt control. For location of the antenna, see figure 1-34. 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 inertial 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. This 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/IR indicator 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 nav radar, generates range marks for the radar scope display, provides automatic angle tracking of air targets, generates precision range and azimuth cursors, supplies the receiver with signals for automatic gain control, and monitors radar operation for inflight malfunction detection and isolation of the malfunction during self-test operation. The radar/IR indicator panel provides range data, a radar scope display, and tuning controls for the radar scope. Power, mode, and function controls on the nav radar control panel cause corresponding signals to be sent to the units of the nav radar. The tracking control handle is used to control antenna tilt in the air mode of operation, change antenna scanned sector from $\pm 45$ degrees to $\pm 10$ degrees, provide rapid slewing of the range cursor, and position the azimuth and range cursors for fix-taking and target tracking. Self-test features incorporated into the radar are used for preflight, inflight, and maintenance malfunction analysis and troubleshooting. The system has three ground modes of operation (ground manual, ground automatic, and ground velocity) and one air mode (air). Tie-ins between the nav radar and other aircraft systems are shown in figure 1-49.

## NAV 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 elevation and 90 degrees in azimuth. The pointing elevation of the antenna, to a maximum of $\pm 30$ degrees with reference to the horizon until 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 azimuth 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 both elevation tracking lamps (arrows) lighting. The range to the target is displayed on the range readout display on the radar/IR indicator scope. 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 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.

## NAV RADAR GROUND MODES

Three ground modes of operation are provided primarily to enable target identification for radar navigation and radar fixtaking. The use of these ground modes, in conjunction with other avionics systems, also enables the radar to be employed for airborne instrument low approach (AILA) landings and altitude calibrations. 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 maximum crosshair range is 124 NM. 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 inertial 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 nav 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 inertial nav system supplies a drift angle (difference between the true heading and ground track counters on the navigation computer) to the nav radar. To keep the sweep in view on the radar scope, the drift angle should be limited to an angle of between $\pm 45$ degrees. To connect the drift signal from the inertial nav system to the nav 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 primarily when failure occurs in the other two ground modes which renders them unusable. In ground manual the antenna depression angle is controlled manually by use of the antenna tilt control knob located on the radar/IR indicator panel. This permits the crewmember to position the fan beam to scan terrain closer in or farther out from the aircraft for optimization of the display. 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. The crewmember 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 radar scope display.

### Ground Automatic and Ground Velocity Modes

The ground auto and ground velocity modes are used primarily during fixtaking. The depression angle of the antenna is adjusted automatically (with the beta switch in NORM) and the crosshairs are positioned by signals from the inertial nav system. Angular corrections to the depression angle may be made for display refinement by manually adjusting the antenna tilt control knob on the radar/IR indicator panel. The navigation computer in the inertial 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 nav 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 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 inertial 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 inertial 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**

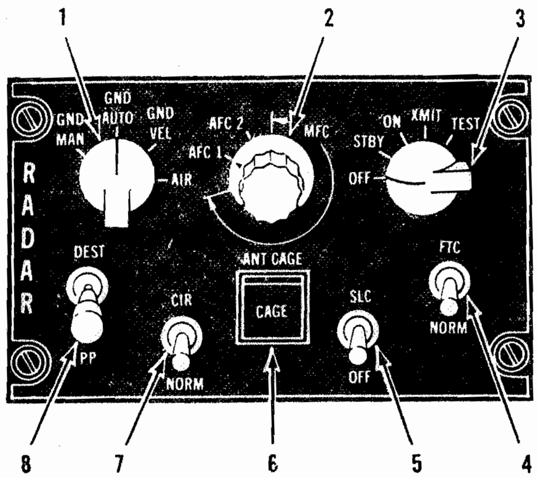
Do not leave the radar in GND VEL/GND AUTO and narrow scan after the turn point/target has passed aft of the radar scan limits. The antenna will oscillate against the limiting stop and may result in reduced life of the antenna azimuth drive.

## CONTROLS AND INDICATORS

### Nav Radar Function Selector Knob

The nav radar function selector knob (3, figure 1-45) is marked OFF, STBY, ON, XMIT and TEST. In the OFF position the entire system is deenergized. Placing the switch

# NAV 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

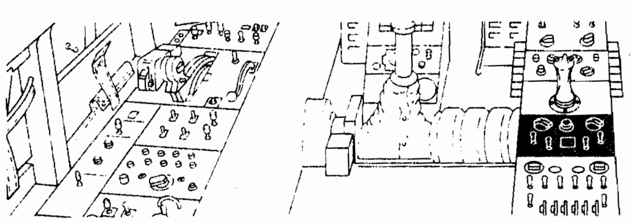



**Figure 1-45.**

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 cursor range display will illuminate when 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 troubleshooting and ground maintenance.

**WARNING**

In the TEST position the nav radar is transmitting. The danger area must be clear prior to using the TEST mode.

### Nav Radar Mode Selector Knob

The nav radar mode selector knob (1, figure 1-45) is marked GND MAN (ground manual), GND AUTO (ground automatic), GND VEL (ground velocity), and AIR. In GND MAN, 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 GND AUTO, the cursors and antenna tilt are automatically positioned by signals from the inertial nav system. The tracking control handle is used to correct the inertial nav system present position and destination counters and the tilt control knob is used to refine the scope display. GND VEL is the same as GND AUTO, 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 AIR, the antenna is programmed from 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 nav radar mode selector knob allow 1 second for each mode. Rapid cycling of the mode selector knob through two or more 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 nav 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.

### Nav Radar Frequency Control Knob

The frequency control knob (2, figure 1-45) is marked AFC 1 (automatic frequency control), AFC 2, and MFC (manual frequency control). In AFC 1, 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, in this position, provide immunity to many types of jamming and improves stability and legibility of the PPI display. In AFC 2, the receiver operates in automatic frequency control and the transmitter is manually tuned using the transmitter tune control knob, located on the radar/IR indicator 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 midband 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.

**Nav Radar Fast Time Constant Switch**

The fast time constant switch (4, figure 1-45) marked FTC (fast time constant) and NORM, provides a means of selecting the desired receiver time response characteristics. This function is used to break 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 to FTC highlights the leading edge of targets, blanks 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 NORM, anti-jamming capabilities are inoperative.

**Note**

FTC should be used in the air mode of operation when obvious jamming signals are present on the nav radar display or to reduce the effect of ground clutter during target detection.

**Nav Radar Side Lobe Cancellation Switch**

The side lobe cancellation (SLC) switch (5, figure 1-45) is 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 1-45) is marked CIR (circular) and NORM. With the switch in NORM, 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 1-45) is marked DEST and PP. The switch position determines which inertial nav system counters are updated in GND AUTO and GND VEL modes.

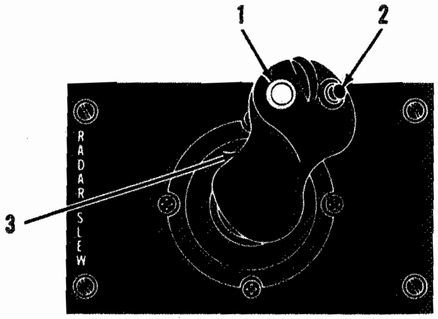
**Nav Radar Tracking Control Handle**

The tracking control handle (figure 1-46) contains a blade-type enable switch (3, figure 1-46) that 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. For tracking handle operation see figure 1-50.

**Nav Radar Range Search (Rs) Button**

The range search button (2, figure 1-46) 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 1-46) 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 Rs button overrides range lock (if established) and permits the range 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 searching will resume from the point at which the range cursor was positioned by slewing.

# NAV RADAR CONTROL HANDLE




1. SECTOR SWITCH
2. $R_S$ BUTTON
3. ENABLE SWITCH

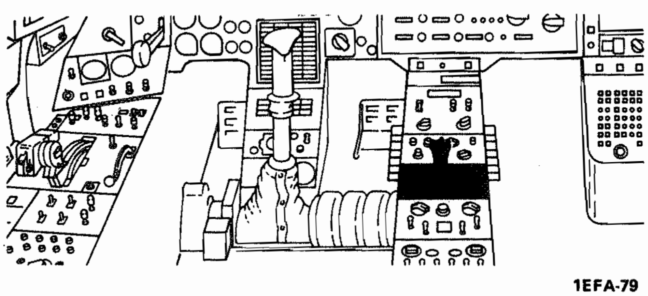



Figure 1-46.

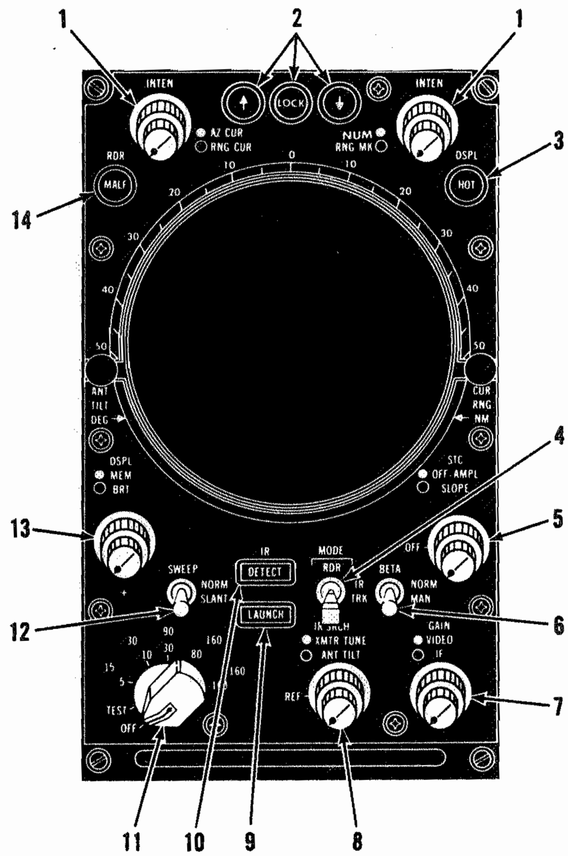
## Nav Radar Sector Switch

The sector switch (1, figure 1-46) labeled SECTOR, 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 aircraft. 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.

## Gain Control Knobs

Two concentric control knobs, labeled GAIN (7, figure 1-47), permit adjustment of receiver gain. The inner knob is marked IF (intermediate frequency), and the outer knob is marked VIDEO. The IF gain control permits adjustment of

# RADAR/IR INDICATOR




1. SCOPE INTENSITY CONTROLS
2. STATUS LAMPS
3. RADAR/IR INDICATOR OVERHEAT LAMP
4. MODE SWITCH
5. SENSITIVITY TIME CONTROL
6. BETA SWITCH
7. GAIN CONTROL KNOBS
8. XMTR TUNE/ANTENNA TILT CONTROL KNOBS
9. IR LAUNCH LAMP
10. IR DETECT LAMP
11. PWR/RNG SELECTOR SWITCH
12. SWEEP CONTROL SWITCH
13. SCOPE DISPLAY CONTROLS
14. RADAR MALFUNCTION LAMP

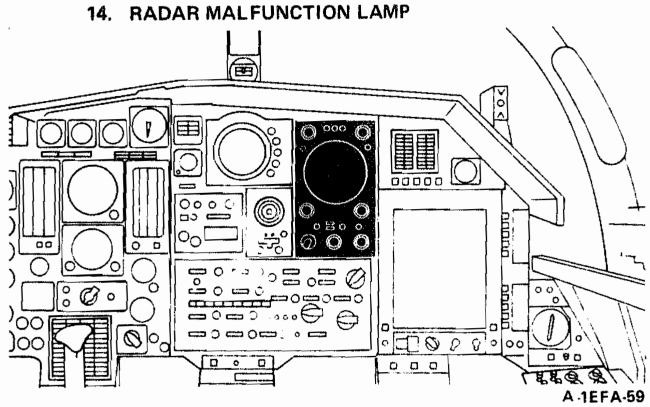



Figure 1-47.

receiver gain when operating in ground modes. This control determines maximum usable sensitivity of the receiver circuits and functions primarily as a brightness control. To set IF gain, the video gain control must be advanced to a position to allow noise video to be painted on the scope. With the video control in about the mid-range position, advance the IF gain control clockwise until the presentation shows a snowy noise appearance, then 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 IF gain control knob should be normally near top of its range but may have to be decreased as a point target is approached in order to reduce background clutter. The video gain control knob will increase the amplitude of the video signal supplied to the nav radar scope when it is turned clockwise. The video control determines the brightness of target returns as opposed to the display BRT control which sets the overall baseline brightness of the scope. To set the video control properly, advance the function knot 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.

### Antenna Tilt Control Knob

The antenna tilt control knob (8, figure 1-47) provides a means of manually adjusting antenna tilt position when operating in the ground modes and 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 inertial nav system and the knob is used to refine this position. The knob has a detent marked REF, 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) operating the antenna can only be physically adjusted to −10 degrees (pitch plus tilt) to prevent interference with the TFR. 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 uniform 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 5,000 feet, the sighting angle should be adjusted with the tilt control for approximately 0 degrees. Antenna tilt position is displayed on the nav radar scope in 5-degree increments from zero to ±30 degrees. In GND MAN, GND AUTO, and GND VEL, with the beta switch in the MAN position, the sighting angle of the antenna as commanded by the antenna tilt control knob is displayed. In GND AUTO and GND VEL, with the beta switch in NORM, the sighting angle displayed is that commanded by the inertial 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. The antenna is pitch stabilized; therefore, the elevation boresight zero reference is parallel with the horizon.

**Note**

> The antenna tilt display tolerance about zero degrees is ±1.5 degrees (full scale ±3.0 degrees). Once airborne and at 15,000 feet AGL, the operator can check the relative accuracy of the display in the air mode by decreasing antenna tilt from 0 degrees until the first ground return coincides with the 20 mile range mark (30 mile range scale). Antenna tilt display should read between −4 and −7.5 degrees.

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 maximum 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 display 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 display. 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 display as a readout device, with the individual parameters selected by use of the range selector knob and the SLC switch. Figure 1-48 shows the command sequence required to display the desired parameter on the tilt and the desired readings to indicate a GO condition.

**Transmitter Tuning Control Knob**

The transmitter tuning control knob (XMTR TUNE, 8, figure 1-47) allows continuous tuning of the transmitter over its entire frequency range, when the frequency control knob is in the AFC 2 position.

**Scope Display Controls**

Two concentric control knobs, labeled DSPL (13, figure 1-47), permit adjustment of scope intensity and persistence. The outer knob, marked BRT, 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 ON. With the IF gain and video controls fully ccw, advance the knob clockwise (increasing brightness) to a point where the sweep is barely visible. 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, 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. The inner knob, marked MEM (memory) will vary scope persistence from 0 to approximately 20 seconds.

**Numeral/Range Marks Intensity Control Knobs**

Two concentric control knobs, labeled INTEN (1, figure 1-47), provide an adjustment of numerals and range marks intensity. The inner knob is marked NUM (numerals) and the outer knob is marked RNG MK. 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.

**Range and Azimuth Cursor Intensity Control Knobs**

Two concentric control knobs, labeled INTEN (1, figure 1-47), provide an adjustment of range and azimuth cursor intensity. The knobs are individually marked RNG CUR and AZ CUR. Adjustment of the range and azimuth cursor intensity to the lowest usable value will allow more precise placement of the cursors over the target. The range cursor knob also controls the intensity of the antenna tilt angle and target slant range displays.

**Nav Radar Sweep Control Switch**

The sweep control switch (12, figure 1-47), marked NORM and SLANT, 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. In the air mode, the switch is inoperative and the sweep is automatically set to slant.

**Nav Radar Beta Switch**

The beta switch (6, figure 1-47), marked NORM and MAN (manual), 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 inertial 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.

**Nav Radar Mode Switch**

The mode switch (4, figure 1-47), marked RDR, IR TRK and IR SRCH is used for selecting radar or infrared display modes. In the RDR position a radar display is presented on the nav radar scope and the infrared threat indicator lamps are enabled. In the IR TRK position, an infrared display is presented, the nav radar is placed in the track mode and the infrared threat indicator lamps are enabled. In IR SRCH, an infrared display is presented, a track-break signal places the nav radar in the search mode and the infrared threat indicator lamps are disabled.

**IR Detect and Launch Lamps**

The IR DETECT and IR LAUNCH lamps (9 and 10, figure 1-47) are enabled when the nav radar mode switch is in either the RDR or IR TRK positions.

## MRT Parameter Checks

| Power/Range Selector Knob | SLC Switch | Parameter | Antenna Tilt Display 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 1-48.**

**Sensitivity Time Control Knobs**

Two concentric sensitivity time control (STC) knobs (5, figure 1-47) provide a means of equalizing radar intensity over the entire scope display when operating in the ground modes at low altitude. The inner knob labeled OFF/AMPL, 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 outer 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 inoperative in the AIR mode. With STC on, an altitude compensation signal is provided to maintain uniform 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 aircraft than for a condition with the toe 80 miles ahead. As an example of practical use of the STC function, assume a low level run on a target just inside the 30-mile range. Ground returns

# NAV RADAR — SUBSYSTEM TIE-INS

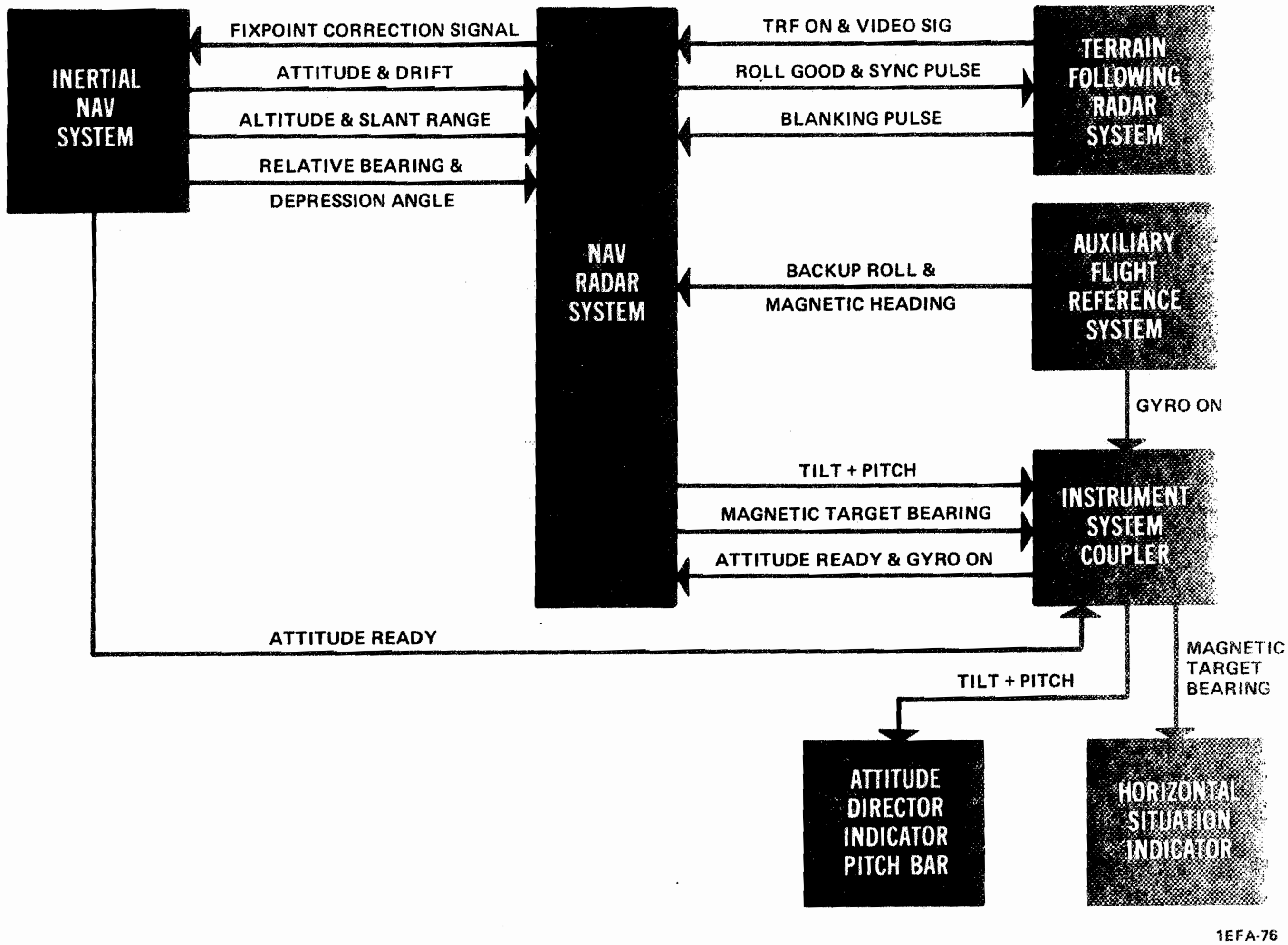



**Figure 1-49.**

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 low level 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 run because antenna sighting angle will not change appreciably as range decreases. On medium to high altitude radar 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 inner STC knob labeled OFF/AMPL.

### Power/Range Selector Knob

The power/range (PWR/RNG) selector knob (11, figure 1-47) is marked OFF, TEST, 15/5, 30/10, 90/30, 160/80 and 160/160. The TEST position energizes a self-test circuit and presents a self-test display on the nav radar scope. The inner number represents the range displayed on the scope for all modes of operation. In the ground velocity mode, the outer number represents the maximum offset distance from the aircraft to the scope center.

### Nav Radar System Malfunction Lamp

An amber radar system malfunction lamp (14, figure 1-47), labeled RDR, will not light when the function

# TRACKING HANDLE OPERATION

| Nav 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 navigation fix bearing<br>Azimuth cursor controlled by aircraft navigational computer unit | Provides correction voltage and aircraft navigational computer unit for updating navigational fix range<br>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)<br>Azimuth cursor controlled by aircraft navigational computer unit | Provides correction voltage to aircraft navigational computer unit for updating target spot range (range cursor)<br>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)<br>Slews display left or right (in azimuth)<br>Fixpoint and cursors at center of radar scope<br>Display ground stabilized<br>Sweep origin offset to aircraft position (off radar scope) | Provides correction voltage to aircraft navigational computer unit for updating navigational fix range (range cursor)<br>Slews display out or in (range)<br>Fixpoint and cursors at center of radar scope<br>Display ground stabilized<br>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)<br>Slews display left or right (in azimuth)<br>Target and cursors at center of radar scope<br>Display ground stabilized<br>Sweep origin offset to aircraft position (off scope) | Provides correction voltage to aircraft navigational computer unit for updating target spot range (range cursor)<br>Slews display out or in (in range)<br>Target and cursors at center of radar scope<br>Display ground stabilized<br>Sweep origin offset to aircraft position (off scope) |
| Air | Slews azimuth cursor left and right | Controls antenna tilt. Controls range cursor if Rs button is depressed |

**Figure 1-50.**

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, and (4) when the nav radar frequency control knob is turned to MFC for manual tuning of the antenna receiver. When lighted the letters MALF are visible.

**Note**

- The nav radar system malfunction lamp will light any time the combined antenna pitch plus tilt angle exceeded the vertical limits. Vertical limits with TFR on are −10 to +30 degrees. It will be normal for the nav radar system malfunction lamp to light intermittently during TFR operation.
- The radar malfunction lamp will light whenever the roll limits of 45 degrees are exeeded.
- If the system is not usable the function selector knob should be placed to STBY to stow the antenna.

### Nav Radar Range Lock Indicator Lamp

A green range lock indicator lamp (2, figure 1-47), labeled LOCK, will light in the AIR MODE with a target LOCK-ON.

### Radar/IR Indicator Overheat Lamp

The radar/IR indicator overheat (3, figure 1-47), labeled DSPL, will light when radar/IR temperature limits are exceeded. When lighted the word HOT is visible.

### Antenna Cage Pushbutton Indicator Lamp

The antenna cage pushbutton indicator lamp (6, figure 1-45) is labeled ANT CAGE and, when 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, roll and drift by primary signals from the inertial nav system. In the event of absence of the primary signal, due to a failure in the inertial 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 inertial 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. Caging the antenna with the NRS in ground manual will remove radar scope presentation drift stabilization and true heading will be presented down the zero bezel. Caging the antenna with the NRS in ground auto/ground velocity will not remove drift stabilization.

**Note**

When the antenna is caged in roll, the terrain following radar will generate a fly-up signal.

### Nav Radar Scope

The radar scope (figure 1-47) 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 aircraft position on the scope is at the bottom in vertical alignment with the center of the scope, except in ground velocity mode. 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 five degree increments with each 10 degrees marked to show azimuth displacement up to 50 degrees either side of the aircraft heading or groundtrack. 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 groundtrack. After lock-on in the air mode, two arrows at the top of the bezel indicate target position relative to antenna scan. 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 fixtaking 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. 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 in the scope panel. Antenna tilt angle is displayed at the lower left of the display. Target slant range, up to a distance of 130 nautical miles, is displayed at the lower right of the scope display.

## ELECTRONIC COUNTER-COUNTERMEASURES (ECCM) FEATURES

The nav 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 nav 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 lob 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 nav radar control panel 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 in the nav radar control panel. 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 nav 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 nav 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/IR indicator displays aircraft heading at zero degrees azimuth and the range sweep depicts linear slant range instead of the normal ground map-like presentation (ground automatic/velocity modes are not affected in this way). In the event of failure in the ground auto or ground velocity modes, the nav radar should be operated in ground manual mode. In ground manual mode, the nav radar operates independently of the inertial navigation system and the radar/IR indicator display is read directly by the operator. If the transmitter frequency agility should fail in AFC 1, the nav 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 nav radar should be operated in MFC. In MFC, the transmitter operates at a midband fixed frequency, while the receiver is tuned manually.

## RADAR ALTIMETER SYSTEM

The multiplexed 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 5,000 feet is read on the radar altimeter indicator. 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. 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 $\pm 20°$.

**WARNING**

If bank or pitch limitations are exceeded, radar altimeter indications are unreliable.

The system incorporates a self-test feature for checking reliability. Refer to figure 1-34 for antenna location.

The multiplexed radar altimeter system is composed of two receiver-transmitter units, two antennas (one to transmit and one to receive), a radar altitude indicator, a calibrator unit, a low altitude monitor unit, a LARA multiplex unit, a multiplex switching unit, warning and caution indicator lamps, a mode selector switch, and a bypass switch. The multiplexed radar altimeter provides the following:

- Simultaneous operation of both radar altimeter channels.
- Continuous comparison of radar altimeter range outputs from both channels, when auto multiplex mode selected.
- Automatic single channel selection for malfunctions resulting in loss of track in one channel, when auto multiplex mode selected.
- Manually selectable single channel operation.
- Complete system self-test from a single control.
- A LARA multiplexer caution lamp which identifies abnormal multiplexer conditions.
- A radar altitude low warning lamp which identifies less than desired terrain clearance and low altitude monitor detected failures.

The auto mode is the primary mode of operation for the multiplexed radar altimeter system. When auto is selected on the radar altimeter channel selector switch, both RT units are placed into operation. These are synchronized by circuits within the LARA multiplex unit such that the transmitter outputs/receiver inputs for each RT unit are alternately connected to the antennas. This switching is accomplished without any reduction in the data rate (pulse repetition frequency) for either RT unit. Under normal operating conditions, both RT units will acquire an "intrack" condition. The LARA multiplex unit selects the altitude signals from both RT units and performs a comparison of these two signals. These two signals must compare to within 12.5 percent. If the comparator is out of these limits, the in track (data good) discrete signal is interrupted to all the interface avionics and the radar altimeter indicator, and the LARA MUX caution lamp is turned on if the TFR is in TF, SIT, or GM. Additionally, a fail fly-up maneuver will occur in the TF mode. Single channel operation is selected by positioning the channel selector switch to either CHAN 1 or CHAN 2. Selection of single channel operation places the other channel in electrical standby. The output signals of the selected channel are routed through the multiplex unit for use by the interfacing avionic systems. The comparator function is inhibited during single channel operation and the LARA MUX caution lamp will remain on if the TFR is in TF, SIT, or GM.

## CONTROLS AND INDICATORS

### Radar Altimeter Channel Selector Switch

The radar altimeter channel selector switch (5, figure 1-2), labeled RADAR ALTM, has three positions marked AUTO, CHAN 1 and CHAN 2. The primary mode of operation is selected by positioning this switch to AUTO, which is the multiplexed mode of operation. The selection of either CHAN 1 or CHAN 2 provides for single channel operation of the radar altimeter system.

### Radar Altimeter Bypass Switch

The radar altimeter bypass switch (4, figure 1-2) is marked NORMAL and BYPASS. Placing the switch to BYPASS when above 5,000 feet over the terrain provides a signal to the TFR to permit auto TF letdowns. As 5,000 feet AGL is passed during descent, the switch will go to NORMAL. When the switch is in the NORMAL position, auto TF letdowns from below 5,000 feet AGL may be accomplished. Placing the switch to BYPASS during terrain following operations below 5,000 feet AGL will cause a TF fail, a fly-up maneuver, and light the radar altitude low warning lamp, unless the actual AGL altitude is within 10 percent of the 1,875 foot pseudo radar altitude from the bypass switch.

### Radar Altimeter Indicator

The radar altimeter indicator (38, figure FO-3) provides absolute altitude indications from 0 to 5,000 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 1,000, and 500 feet from 1,000 to 5,000.

**WARNING**

If power is lost to the system the OFF warning flag will appear on the dial and the pointer will remain at the last indication.

### Radar Altimeter Control Knob

The radar altimeter control knob is located on the lower right of the altimeter and serves three functions; as an on-off control, sets the minimum altitude index pointer, and tests 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 300 (±15) feet if the receiver-transmitter unit is operating.

The self-test feature may be used at any time and at any altitude below approximately 38,000 (±5,000) feet.

**WARNING**

Initiation of the self-test during TF operations at the 200- or 300-foot set clearances while the radar altimeter signal to the TFR is controlling the aircraft may result in a dive command from the TFR. In this case, the 68 percent fly-up will not be operable.

**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.

**Note**

Performing a self-test of the radar altimeter during terrain following at 500, 750, or 1,000 feet set clearances will cause a fly-up, since the false altitude (300 ±15 feet) that the radar altimeter locks on during self-test is less than 68 percent of these clearances. If the radar altimeter signals are controlling the aircraft and self-test is performed at 400 feet set clearance, a climb command will result, and at the 200 or 300 feet set clearances, a dive may result. The 68 percent fly-up will be inhibited during radar altimeter self-test at the 200, 300, and 400 feet set clearances.

### Radar Altitude Low Warning Lamp

The radar altitude low warning lamp (figure FO-5) located on the left 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. This lamp is also used in conjunction with the low altitude monitor. The monitor will cause this lamp to light for any of the following conditions.

1. The radar altitude data to the radar altimeter indicator differs from that being supplied to the TFR by 10 percent or more.
2. The aircraft has penetrated below 68 percent of selected set clearance.
3. The set clearance circuitry in the low altitude monitor has failed to a value less than 120 feet.
4. The low altitude monitor has failed.

When the warning lamp lights due to any of these conditions, a fail fly-up maneuver will occur. A fail fly-up will not occur if the lamp lights due to indicated radar altitude being less than that set by the index pointer.

### Radar Altitude LARA MUX Caution Lamp

The radar altitude low altitude radar altimeter multiplexer caution lamp (figure FO-5) is located on the left instrument panel. The lamp displays the letters LARA MUX when lighted. It will light for any of the following conditions:

1. Single LARA channel operation is selected and TFR is in TF, SIT, or GM.
2. LARA multiplexer unit self-test with AUTO selected.
3. The radar altitude data from both receiver-transmitter units exceed the comparison limit of 12.5 (±2.5) percent in AUTO and TFR is in TF, SIT, or GM.
4. Either one or both receiver-transmitter units fail to acquire or lose their "in-track" condition in AUTO and TFR is in TF, SIT, or GM.

### LARA Multiplex Unit

The LARA Multiplex Unit provides the following functions:

1. Compares the radar altimeter data from both channels to insure that these data do not differ by more than 12.5 (±2.5) percent. The comparator is only enabled when AUTO is selected on the radar altimeter channel selector switch. If the comparison exceeds the comparator limits, the radar altimeter intrack (data good) discrete is interrupted and the LARA MUX caution lamp comes on. (LARA MUX caution lamp is inhibited unless TFR is in TF, SIT, or GM, except during self-test.) If terrain following operation is being performed when the data good discrete is interrupted, a TFR system fail fly-up will also be initiated.
2. Provides the synchronization and switch control required to permit simultaneous operation of both receiver-transmitter units utilizing a single pair of antennas.
3. Provides the logic circuits required for the automatic channel selection when one channel loses track when the channel selector switch is in AUTO.

## AUTOMATIC MODE OPERATION

During TF operation, with the radar altimeter channel selector switch in the AUTO position, a fail fly-up will be initiated and the LARA MUX caution lamp will light if the LARA multiplex unit detects a difference between LARA channels exceeding 12.5 percent, or if both LARA channels lose track. When a single channel loses track, the LARA multiplex unit automatically selects single channel operation and lights the LARA MUX caution lamp. In this case, a fail fly-up will not be generated since a good LARA channel remains locked on. Single channel operation, whether manually or automatically selected, does not affect fly-up protection of the TFR. If the remaining LARA channel subsequently loses track, a fail fly-up will occur. If both channels subsequently regain track, the LARA MUX caution lamp will go out, and all multiplex functions will be restored.

**WARNING**

If neither LARA channel locks on following descent through 5,000 feet AGL, the LARA MUX caution lamp will remain lighted, but no fail fly-up will occur since the radar altimeter bypass switch will remain in BYPASS.

**Note**

During AUTO TF descents, if the LARA MUX caution lamp remains lighted after descending through 5,000 feet AGL, a fail fly-up may or may not be present. If a fail fly-up is present, a 12.5 percent comparison failure has been detected. If no fail fly-up is present and a LARA channel is locked on, single channel operation has been automatically selected.

## TERRAIN FOLLOWING RADAR AN/APQ-110

The terrain following radar (TFR) provides low altitude terrain following, obstacle avoidance, and blind letdown capability. The TFR consists of left and right antenna receivers, synchronizer transmitters, power supplies and computers in a dual channel configuration, a radar scope panel and a control panel. Each channel may be operated independently of the other in any one of three modes: terrain following (TF), situation display (SIT), or ground mapping (GM). During auto or manual TF turning flight maneuvers, the TFR antenna is positioned to look along the anticipated flight path, and the climb/dive commands are compensated for the aircraft being in a roll attitude. The TFR receives inputs from the radar altimeter, nav radar, inertial nav systems, flight controls system, and central air data computer. Refer to figure 1-51 for TFR system tie-ins with other weapon system avionic equipment.

### PRIMARY ATTITUDE REFERENCE

The TFR is dependent upon the inertial nav system for roll, pitch, ground speed, and drift signals. Since all sources of possible error are not continuously tested, the

# TERRAIN FOLLOWING RADAR– SUBSYSTEM TIE-INS

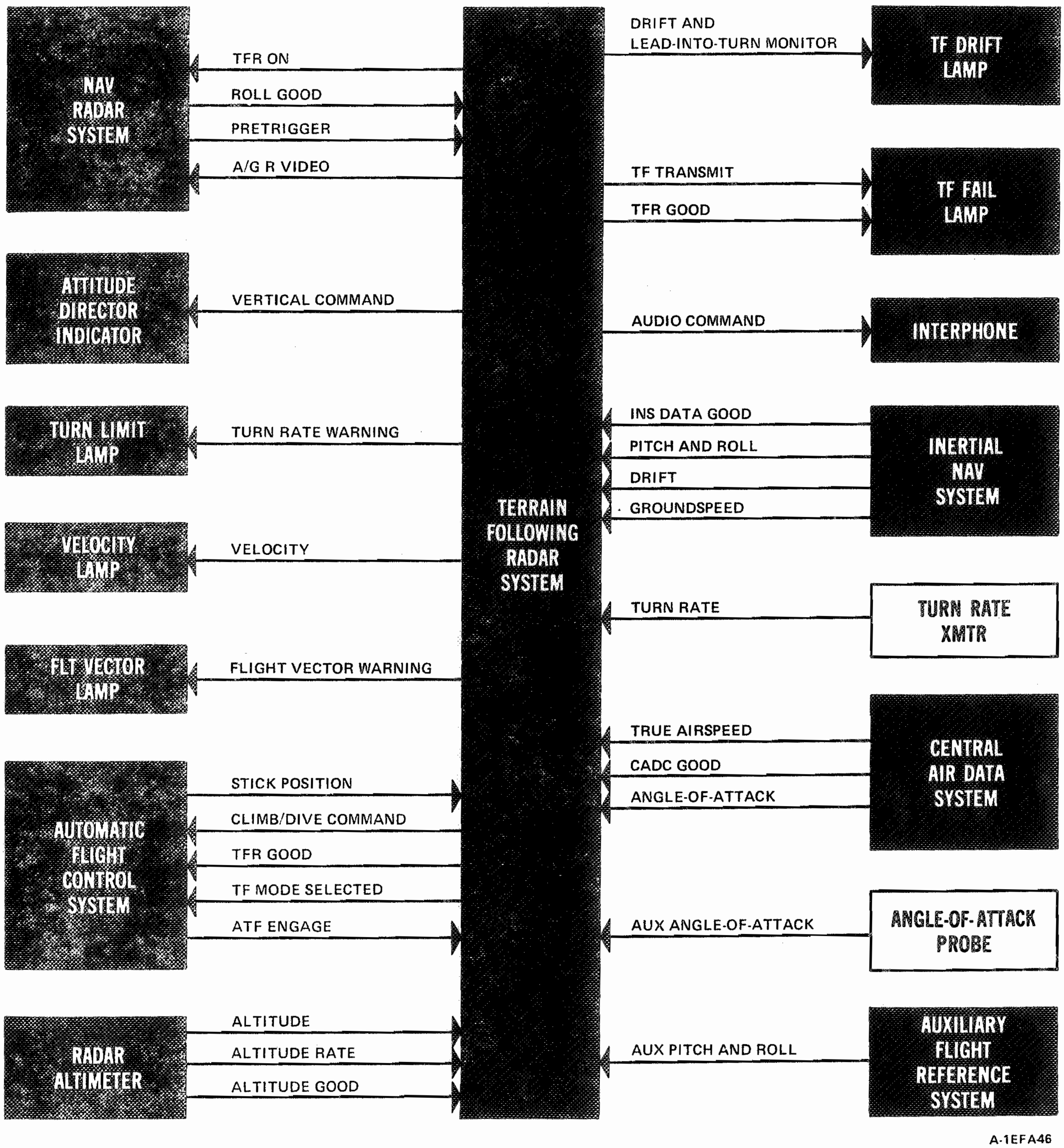



Figure 1-51.

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 inertial nav system. If an abnormal condition is observed, TF flight should be terminated.

## AUXILIARY ATTITUDE REFERENCE

The auxiliary flight reference system (AFRS) provides pitch and roll signals to the TFR when the inertial navigation system is inoperative. During terrain following flight on the AFRS, primary attitude reference will not be available for a comparative check to detect errors or failures of critical inputs to the TFR systems. When using the auxiliary reference, the primary attitude and flight vector caution lamps will be lighted indicating that the primary flight vector is not being checked for accuracy. Refer to figure 1-52.

**WARNING**

Terrain following flight should be limited to day VMC when the AFRS is furnishing attitude signals to the aircraft subsystems.

## 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 ADI. In the manual mode, the set clearance can be maintained by flying pitch steering commands on the ADI. In the automatic mode, the climb and dive signals are coupled into the pitch channel of the flight control sytem. Refer to figure 1-53. An aural climb/dive signal is provided in addition to the pitch steering commands. This aural signal consists of a 2,500 hz frequency tone for climb and a 500 hz frequency tone for dive at 20 pulses per second per g. On/off and volume control for the aural command is provided by the ILS monitor knob on each interphone control panel.

The TF mode can also be used to make auto or manual TF letdowns to a preselected terrain clearance. When using this capability, descent can be made manually using the pitch steering commands on the ADI, or automatically by placing the auto TF switch to the AUTO TF position. The descent is limited to a 12 degree dive. Only one channel at a time can be operated in TF mode. If both channels are placed to TF the second channel placed to TF will go to a standby condition as a backup and will automatically take over should the operating channel fail. The TFR provides a fail-safe fly-up capability for internally detected malfunctions.

Refer to figure 1-51 for information pertaining to tie-in of the TFR with the flight control system. In the TF mode, the antenna is roll and drift stabilized. Antenna scan is vertical and the scope display is in the form of a non-linear E type presentation. A zero command line, displayed on the scope, provides a terrain clearance reference. The slope of the zero command line will vary with the speed of the aircraft, terrain clearance setting and the type of

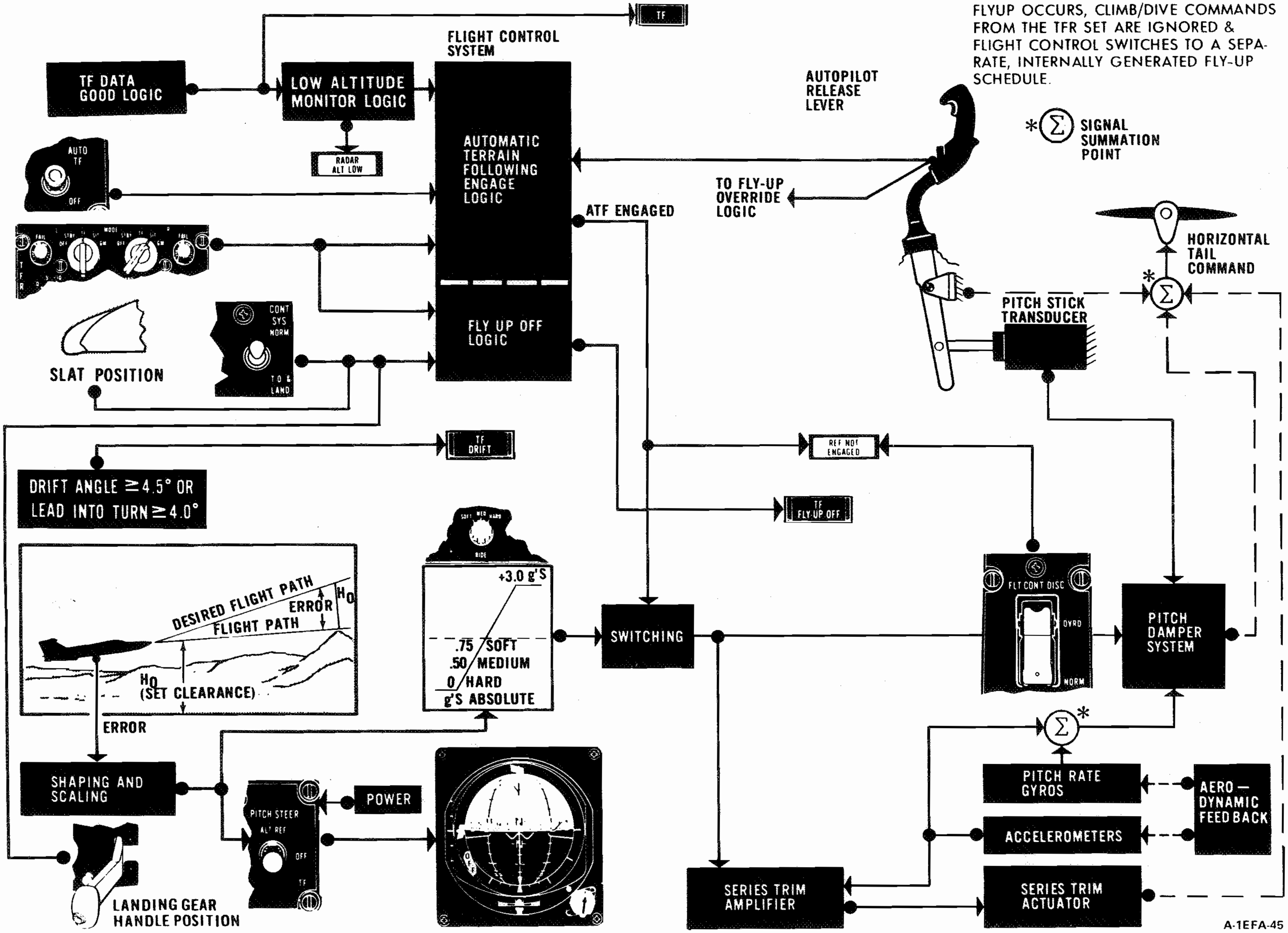
## TFR Warning & Caution Lamp Indications

| Data Signals | | | | |
|---|---|---|---|---|
| Roll | LARA | CADC | Inertial | Lamps Light |
| Fail | Good | Good | Good | L&R Fail Caution Lamps, TF Warning Lamp, Reference not engaged |
| Good | Fail | Good | Good | L&R Fail Caution Lamps, TF Warning Lamp, Reference not engaged |
| Good | Good | Fail | Good | FLT Vector Caution Lamp |
| Good | Good | Good | Fail | FLT Vector Caution Lamp |
| Good | Good | Fail | Fail | L&R Fail Caution Lamps, TF Warning Lamp, Reference not engaged |
| Note: A LAM only detected failure will result in the illumination of just the radar altitude low and reference not engaged caution lamps. | | | | |

**Figure 1-52.**

# AUTO TF FLIGHT CONTROL SCHEMATIC

NOTE:
ATF MODE SHOWN ENGAGED. WHEN FLYUP OCCURS, CLIMB/DIVE COMMANDS FROM THE TFR SET ARE IGNORED & FLIGHT CONTROL SWITCHES TO A SEPARATE, INTERNALLY GENERATED FLY-UP SCHEDULE.




★ Figure 1-53.

ride selected. Range displays on the scope are from left to right on a nonlinear scale so that ranges up to two miles are displayed over three-fourths of the scope and the remaining one-fourth of the scope displays returns up to ten miles. Elevation of returns along the ground track are displayed vertically on the scope. The maximum computing range of the TFR computer is 36,000 feet. The minimum computing range of the TFR system is 1250 feet in all set clearance place.

## GROUND MAPPING (GM) MODE

The GM mode provides a scope presentation of the terrain that is ahead of the aircraft. Antenna tilt can be adjusted for best picture. This mode is used primarily for navigation. The antenna scan is 30° in azimuth either side of ground track. In this mode, the antenna is pitch, roll, and drift stabilized. PPI presentation is displayed with linear range graduations. The GM mode presentation will provide a radar altimeter override function display.

## SITUATION (SIT) MODE

This mode of operation is used in conjunction with TF mode for obstacle avoidance. The antenna scan, stabilization and scope display are the same as GM. Antenna tilt cannot be adjusted. Returns of the terrain that are at or above the aircraft altitude are displayed on the radar scope. Since this mode does not provide a margin of vertical clearance, the scope display should only be used as reference for avoiding obstacles and not for overflying obstacles.

## TFR CONTROL AND INDICATORS

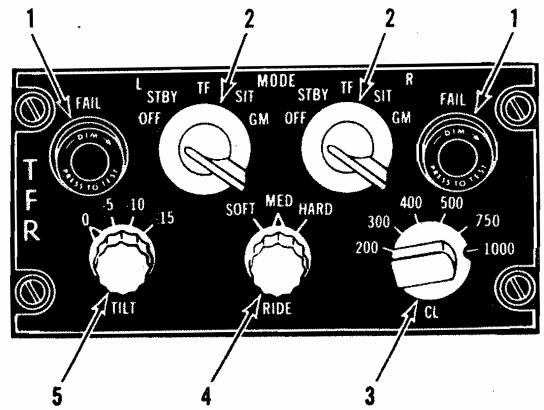
### TFR Channel Mode Selector Knobs

Two five-position rotary channel mode selector knobs (2, figure 1-54) 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 STBY position, power is applied to the channel for warmup and antenna stabilization. The TF, SIT and GM positions provide terrain following, situation display or ground mapping modes of operation, respectively.

**CAUTION**

> If both channels are not placed in STBY or above when transmitting on the TFR or nav radar, the crystal diodes in the receiver of the channel in OFF may be degraded or ruined, making the receiver less sensitive or inoperative.

# TFR CONTROL PANEL




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

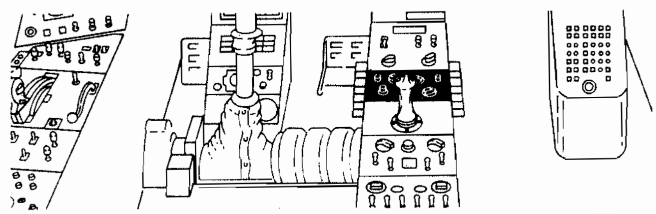


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**Figure 1-54.**

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 switch 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 failures 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. When channel switching occurs over water, it is normal for the second channel to initially fly the aircraft above the set clearance with a gradual decrease in the command until the set clearance is re-established. This condition is caused by the standby channel having a full climb present from the altimeter override command which receives zero altitude while in standby. During any automatic switchover, the TF channel fail caution lamp of the channel, which caused the switch, will light. If the failure which caused the automatic switching is no longer present, the lamp will remain lighted until subsequent auto switchover occurs. The lamp may also be extinguished by manually recycling the channel.

Note

For normal operations, one channel should be in TF and the other in SIT or GM. Crews will not rely on automatic switchover.

**Terrain Clearance Knob**

The terrain clearance knob (3, figure 1-54), has six positions marked 200, 300, 400, 500, 750 and 1,000. The 500 position provides a weather mode marked (WX) which limits the area from which radar returns are processed.

**WARNING**

When the 500-foot position is selected, the TFR does not command on targets beyond 15,000 feet or above the aircraft fuselage center line, and will not anticipate the terrain early enough to provide a command to prevent terrain impact if 0.85 MACH is exceeded.

Note

A malfunction in the terrain clearance knob may cause the TF computers to fly at a set clearance plane other than the one selected. If this happens there are no fail-warning lamps or fly-ups, because the system does not recognize a failure. The zero command line is a possible way of determining if the TF computer is using a set clearance plane other than the one selected.

The TFR in hard ride should maintain flight within the following tolerances.

| Selected Clearance | Terrain Clearance Min | Terrain Clearance Max |
|---|---|---|
| 200 | 170 | 300 |
| 300 | 260 | 425 |
| 400 | 350 | 550 |
| 500 | 440 | 650 |
| 750 | 675 | 950 |
| 1000 | 900 | 1200 |

The LARA altitude, when cresting peaks, will usually be slightly less than the stabilized, level terrain clearances. The above tolerances are not directly applicable to TF flight over rugged terrain; however, terrain clearances when cresting peaks should not consistently exceed these tolerances. Clearances may be slightly higher in SOFT or MED ride.

**Ride Control Knob**

The ride control knob (4, figure 1-54) is 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 1-54) is used to position antenna tilt between zero and −15 degrees for the best ground return when operating in the GM mode. The knob has antenna tilt angles of 0, −5, −10, and −15 marked for reference only.

**Range Selector Knob**

The range selector knob (5, figure 1-55), 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.

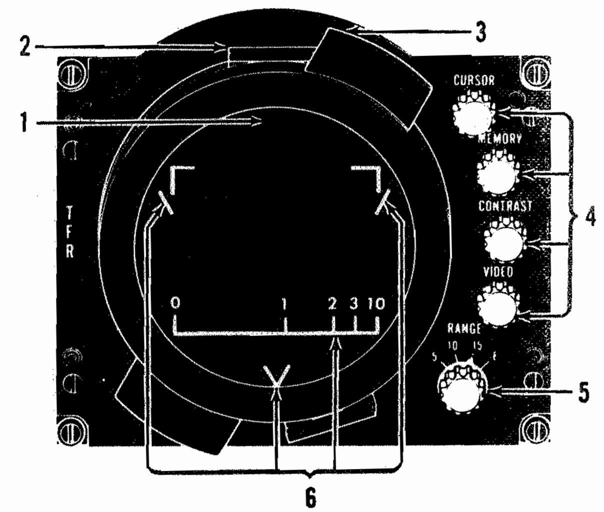
**TFR Scope Tuning Control Knobs**

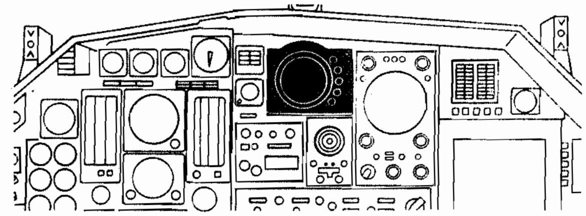
Four radar scope tuning control knobs (4, figure 1-55) are labeled CURSOR, MEMORY, CONTRAST and VIDEO from top to bottom. The cursor knob adjusts the brilliance of the range cursors. The memory knob increases or decreases scope storage retention time. The contrast knob adjusts scope contrast for optimum viewing. The video control adjusts the video return brightness to desired level. For TFR scope tuning procedures, see TFR ground checks, Section II.

**TFR Scope**

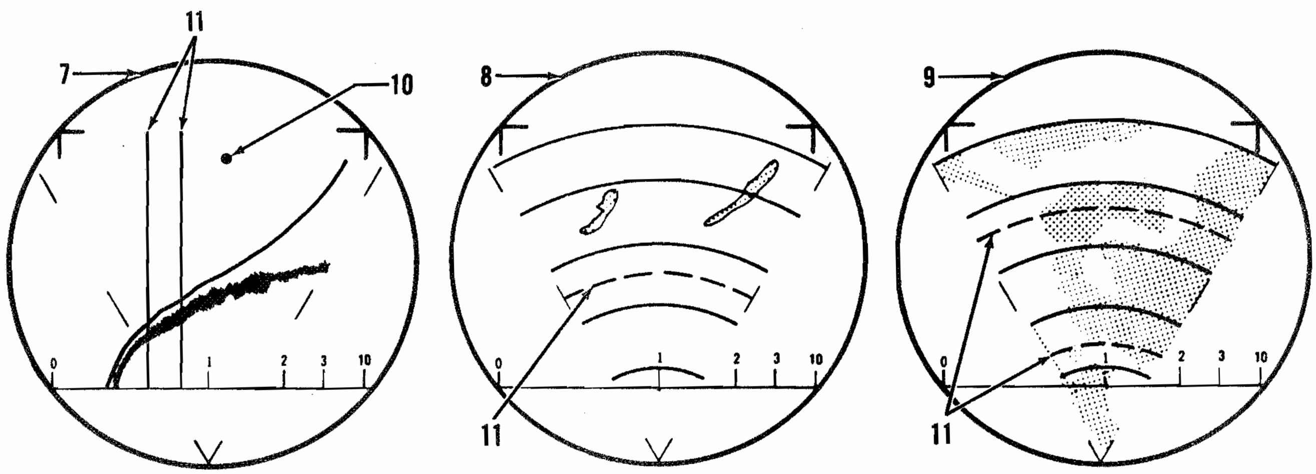
The TFR scope (1, figure 1-55) provides a direct viewing presentation of either an E scan (vertical) display when in TF mode, or a sector PPI (azimuth scan) display when operating in SIT or GM modes. The E-scope display is a square pattern representing the vertical scan pattern of the TFR radar. The scan is 8 degrees above, to 32 degrees below, the fuselage reference line. The square presentation results from expanding the point of the antenna scan wedge at the left edge of the scope, representing the position of the aircraft, into a square. This results in the fuselage reference line becoming a horizontal line approximately one fifth the way down from the top of the scope. The bottom edge (range) of the scope is not linear. This range is an E squared scale representing 10 miles slant range in front of the aircraft. One mile is halfway across

# TFR SCOPE PANEL AND PRESENTATIONS



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
11. RADAR ALTIMETER OVERRIDE FUNCTION DISPLAYS




## Test Patterns

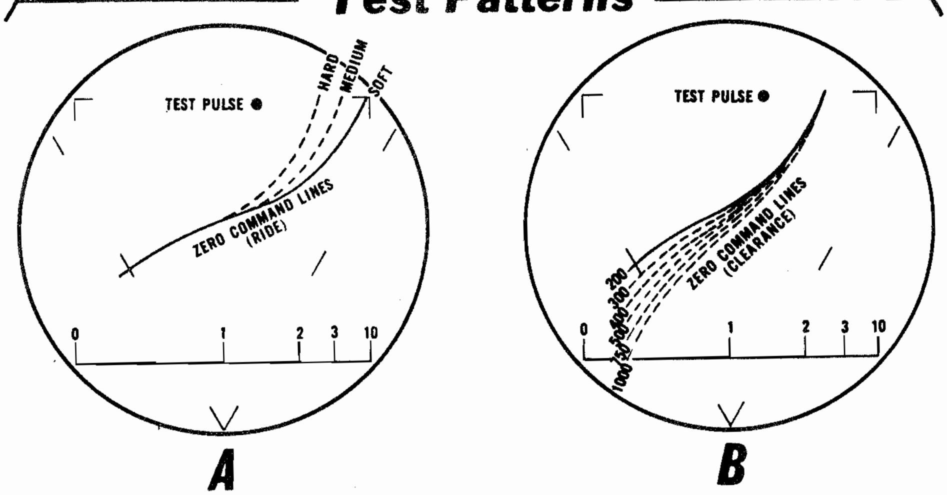



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Figure 1-55.

the scope; two miles is halfway from the center to the right edge, etc. This provides an expanded presentation of objects close to the aircraft. The end result makes the commanded flight path an S shaped line, called the Zero Command Line (ZCL). The relationship of this line to various clearance settings and ride control selections are depicted by figure 1-55. It should be noted that the ZCL is a much simplified graphic display of the results of the complicated equation being solved by the TFR computer.

**Note**

At clearance settings of 1,000 and 750 feet, the start of the TFR ZCL may have a short vertical line extending downward. Terrain video may penetrate through the vertical portion when flying close to the selected clearance at very close range (3/4 mile). Also, below 750 feet clearance, there may be a short gap in the ZCL near the center of the presentation that can change with ride control, selected clearance or airspeed. These are normal conditions on the E-scope presentation and will not affect system performance.

The TFR scope presents a distinct display when radar altimeter inputs are being used to compute climb/dive commands. When operating in the TF mode, two vertical bars are present on the E-scope display when in the radar altimeter override function (11, figure 1-55). The vertical bars bracket the 3,500 to 4,500 foot range of the altimeter override gate. If SIT or GM mode is selected on one channel, with TF on the other channel, and the TF scope range switch is positioned to 5, 10, or 15, dashed range marks are displayed at 2.5 NM and 7.5 NM on the PPI scope when in the override function (11, figure 1-55). A self-test pulse (10, figure 1-55) 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.

### Auto Terrain Following Switch

The auto terrain following (AUTO TF) switch (3, figure 1-18) is a two position lever lock switch marked AUTO TF and OFF. The switch is locked in OFF and must be pulled out to move from OFF to AUTO TF. When the switch is in OFF, and either TFR channel mode selector knob is in TF, the aircraft must be flown manually using the pitch steering commands on the ADI to hold the terrain clearance selected on the TFR terrain clearance knob. With the switch in OFF, the reference not engaged caution lamp will remain on. When the switch is placed to AUTO TF and either TFR channel mode selector knob is in TF, signals from the TFR will control the pitch damper and series trim to automatically hold the aircraft on the terrain clearance setting selected by the TFR terrain clearance knob. With the switch in AUTO TF, the reference not engaged caution lamp will go out.

**Note**

- Selecting AUTO TF without at least one TFR channel in the TF mode will cause abnormal series trim operation and will light the fly-up off caution lamp and the reference not engaged caution lamp.
- If the pitch autopilot/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.

When AUTO TF is selected and the autopilot release lever is not held, the control stick will be centered. If auto TF mode is controlling the aircraft, as indicated by the reference not engaged caution lamp, the pitch trim function of the stick trim button will be inoperative.

### Autopilot Release Lever

When flying auto or manual TF, the autopilot release lever (6, figure 1-16) may be used for overriding fly-up maneuvers caused 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 and the reference not engaged caution lamp will light. The TF failure warning lamp will not go out when the autopilot release lever is depressed. The radar altitude low warning lamp, if lighted because of a failure sensed by the low altitude monitor 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.

### Instrument System Coupler Mode Selector Knob

The ISC mode selector knob (figure FO-3) has nine marked positions. TFR operation 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 T.F. position. The knob should be placed in the NAV position for TFR operation. For a description of the other functions of the knob, refer to "Instruments," this section.

### Instrument System Coupler Pitch Steering Mode Switch

The ISC pitch steering mode switch (figure FO-3) is marked ALT REF (altitude reference), OFF, and TF (terrain following). Placing the switch to TF couples the manual commands from the TFR to the pitch steering bars on the 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 ILS, AILA, or AIR/AIR. It is recommended that the switch be placed in TF when performing auto TF so 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.

**Note**

Placing either TFR channel mode selector knob to TF, without placing either the ISC pitch steering switch to TF, or the auto terrain following switch to TF may result in a flashing self test pulse and intermittent fly-up commands.

### Flight Instrument Reference Select Switch

The flight instrument reference select switch (1, figure 1-3) is marked PRI and AUX. Normal TFR operations should be performed with this switch in the PRI position. Selecting the AUX position while the inertial nav system is operating normally will cause the nav radar antenna to cage and the TFR to fail. Should the inertial nav stabilized platform system fail or be turned off, the switchover 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.

## TFR SYSTEM WARNING/CAUTION LAMPS

### TF Failure Warning Lamp

A red TF failure warning lamp (figure FO-5), located on the left main instrument panel, provides a redundant indication of TFR channel malfunctions. If each channel is being operated in a different mode, the lamp will light when the channel in TF mode malfunctions. If both channels are in TF mode, the lamp will momentarily light when the channel takes over. Should the backup channel in turn fail, the lamp will light and remain on. The lamp will not light for a LAM only detected failure.

### TFR Channel Fail Caution Lamps

Two amber channel failure caution lamps (figure FO-5), 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 switch in TF, SIT, or GM position indicates a malfunction in that channel. The lamps will also light when that channel is in TFR, the aircraft is above 5000 feet AGL, and the radar altimeter bypass switch is in the NORMAL position. These lamps will not light for a LAM only detected failure.

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, or switching into or out of the 500 setting. This is a normal indication.
- The lamps will also blink rapidly when more than 0.5 "g's" are commanded if the instrument system coupler pitch steering mode switch is not in the TF position when operating in the manual TF mode.

### Reference Not Engaged Caution Lamp

The reference not engaged caution lamp (figure FO–5), located on the right status indicator of the left main instrument panel, will light under the following conditions:

1. Either TFR channel mode selector knob is in the TF position and the auto TF switch is OFF.
2. The auto TF switch is in AUTO TF and neither TFR channel mode selector knob is in the TF position.
3. The flight control disconnect switch is placed to the override (OVRD) position.
4. During any fail safe fly-up due to loss of the TFR data good signal.
5. Auto pilot release lever depressed during AUTO TF.

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

**WARNING**

Any time the reference not engaged lamp lights during TF operation, immediate action must be taken to put the lamp out or terminate auto TF.

**Note**

While operating in auto TF, the reference not engaged caution lamp can come on due to conditions not associated with TFR operation. For example, when operating in auto TF with an autopilot submode engaged, moving or bumping the control stick will disengage the submode 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 detailed description of the lamp and its functions with other systems, refer to "Autopilot System," this section.

### TF Fly-up Off Caution Lamp

The TF fly-up off caution lamp, located on the main caution lamp panel (figure FO–5), provides an indication that fly-up protection is not available. After TF mode selection, the letters TF FLY-UP OFF are visible on the face of the lamp when it is lighted indicating the fly-up maneuver is inhibited.

Arming of the fly-up circuit will result in the TF fly-up off lamp going out. The fly-up circuit can be armed by one 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 lights) will result in a fly-up maneuver.

When airborne, and with the L or R TFR channel in TF, R
the lamp will light when the flight control system switch is R
in T.O. & LAND with the gear handle down, or when the R
slats are extended. Certain power failures within the TFR R
system will also cause the lamp to light. The lamp will also light if the auto TF switch is placed to 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.

**WARNING**

If the autopilot release lever is depressed during manual or auto TF operation, the TF fly-up off caution lamp will not light and fly-up protection is not available.

### TF Drift Caution Lamp

The TF drift caution lamp located on the main caution lamp panel (figure FO–5) provides an indication that the drift signal is greater than 4.5 degrees, or the computed lead into turn signal comparison exceeds 4 degrees. When the lamp is lighted, the letters TF DRIFT are visible.

**WARNING**

Do not fly night or IMC TFR if the TF DRIFT caution lamp is lighted even though drift accuracy checks within limits. A lead-into-turn malfunction may preclude safe TF operation.

### Velocity Caution Lamp

The velocity caution lamp located on the main caution lamp panel (figure FO–5) indicates the TFR is computing with a mach 0.8 airspeed. The lamp will light when the true airspeed and inertial groundspeed differ by 130 knots. The letters VELOCITY are visible on the lamp when it is lighted. Terrain following flight shall be restricted to a velocity range of 0.7 mach to 0.9 mach when the lamp is lighted. Should the comparison return to its normal level, the lamp will go out and operation will be normal.

**WARNING**

Airspeed should be held between mach 0.7 and 0.9 when the velocity caution lamp is on. If these speeds are exceeded, the TFR will not properly anticipate terrain clearance commands, and the aircraft may fly into the ground.

**Note**

The velocity caution lamp will light during TFR ground checks.

### Flight Vector Caution Lamp

The flight vector caution lamp located on the main caution panel (figure FO-5) provides an indication that the TFR is no longer making a comparison check of the aircraft flight vector. This lamp will be lighted for any of the three following conditions.

1. CADC data good signal to the TFR is missing.
2. INS data good signal to the TFR is missing.
3. INS synchro reference signal to the TFR is missing.

**Note**

Terrain following flight must not be attempted with this lamp lighted if the CADS lamp is also lighted or there are no other associated lamps lighted.

### Turn Limit Caution Lamp

The turn limit caution lamp located on the main caution lamp panel (figure FO-5) provides an indication when lighted that either or both of the following conditions exist:

1. The aircraft is executing a turn in which the heading rate exceeds the maximum safe value (2.0 degrees per second) for which the TFR antenna lead-into-turn stabilization is reliable.
2. The aircraft is in a roll attitude and the computed climb command exceeds 5.2 "g's" absolute.

**Note**

During TF flight with a failed turn needle, there may be no warning lamp indication when exceeding TFR turn rate limits.

The lamp should be monitored in turns. When the lamp lights, the bank angle should be reduced to put the lamp out. When lighted, the letters TURN LIMIT are visible on the lamp.

### Low Altitude Monitor

The low altitude monitor (LAM) provides redundant 68 percent fly-up protection by interrupting the TFR data good signal to the aircraft flight control system if a detected failure occurs. The monitoring function of the LAM is enabled only when one or both TFR channels are in a TF mode and the radar altimeter is locked on. The LAM:

1. Checks that the two altitude signal outputs from the LARA multiplexer unit compare within 10 percent of each other.
2. Detects malfunctions in the LAM computed set clearance plane circuitry, where LAM computed set clearance is 120 feet or less.
3. Detects a failure of the bypass switch to properly switch from bypass to normal (actual radar altitude and pseudo altitude differ by greater than 10 percent.)

LAM detected malfunctions will initiate a fail-safe, flyup maneuver.

**Note**

- It is possible for a specific low altitude monitor failure to cause an inadvertent fail/fly-up in conjunction with all the following conditions:
- The TF mode is not selected on either TFR channel.
- Radar altitude low warning lamp on.
- The flight control system is not in takeoff and land configuration.
- The flight control system "fly-up" logic circuit is armed by momentarily depressing the autopilot release lever.

If this malfunction occurs, holding the auto-pilot release lever depressed will override the fly-up. The radar altimeter should then be turned off for the remainder of the flight. Turning the TFR to OFF will not override the inadvertent fly-up.

Note

When a LAM only induced fail safe fly-up maneuver occurs, the radar altitude low and reference not engaged warning lamps will light. Other associated TFR caution or warning lamps will not light and the pitch steering bars may not indicate a fail climb.

## TERRAIN FOLLOWING MISSION PLANNING

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 preflight planning must be accomplished to assure safe low level operation. 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. Determine the initial maximum allowable angle-of-attack and corresponding minimum airspeed for the desired wing sweep from the "Angle-of-Attack and Airspeed VS Wing Sweep for TF Operation" chart in Appendix I, Part 12. With this or higher airspeed, calculate cruise performance for the desired radius of action to assure an adequate fuel reserve. See Appendix I, Part 12, for guidance in airspeed selection. The maximum allowable angle-of-attack shown on the referenced chart is provided for the purpose of defining the minimum airspeed at which a TF failure (fly-up maneuver) may be experienced without exceeding the angle-of-attack limits presented in Section V. A fly-up occurring at minimum airspeed must be terminated within 2 seconds to avoid excessive airspeed loss and angles-of-attack in excess of limits.

**WARNING**

Under no circumstances should airspeed be allowed to decrease below minimum allowable airspeed during TF operations. To do so may result in exceeding angle-of-attack limits in event of a TF fly-up.

3. Next, carefully select the route for TFR operation. The auto TF letdown point should be chosen so that a letdown into rugged mountainous terrain is avoided, if possible. A letdown over water should be sufficiently far from the coast so that the aircraft can be stabilized at TF altitude prior to the transition from water to land. 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 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.

6. Route turning points need special consideration during TFR operations because of the limitations of the TFR system during turns. During normal TF operations, limit bank angle to 30 degrees. If mission requirements dictate, a maximum bank angle of 40 degrees may be used. If 40 degrees of bank angle is used, then a minimum of 460 KTAS must be maintained during the turns. The turn limit caution lamp will come on when the aircraft exceeds 2 degrees per second turn rate. If this caution lamp comes on, reduce the turn rate or

roll angle until it goes out. If it does not go out, go to the MEA. In the event of a TF fail in a bank, go to zero bank when terrain permits.

7. The terrain clearance setting will normally be determined by the following considerations:

   a. Day or night, VMC or IMC.

   b. Terrain profile and differential altitude.

   c. Height of trees or man-made obstacles.

   d. Mission requirements, i.e. training or combat.

8. Preflight planning must include consideration of enroute weather. If heavy rain or thunderstorm activity is forecast, TFR operation may not be possible.

## OPERATIONAL CONSIDERATIONS

### Critical TF Considerations

Both crewmembers must continuously monitor all available information and communicate essential data to each other during TF flight. Numerous indicators are available, such as E-scope, NRS, LARA, AVVI, warning and caution lamps, visual cues, etc., and should be monitored by both crew members because they are critical safety items.

Careful monitoring of the E-scope while performing AUTO TF is essential to ensure proper TFR/Flight controls tie-in. The crew should be especially watchful for the following indications:

- Exceeding set clearance plane limits
- Ballooning over ridges or hills
- Short duration flyups
- Degraded (slow) response to climb/dive commands
- Terrain penetrating the ZCL
- Failure of the pitch steering bar to center in response to TFR climb/dive commands

**WARNING**

- If any malfunction occurs that distracts either crew member from monitoring TFR performance during night or IMC, climb to MEA immediately.
- Do not fly with the autopilot release lever held depressed even for a few seconds at TFR altitudes. No fail safe fly-up maneuver is available.
- Anytime the aircraft descends below 68 percent of the set clearance plane while flying TF and a flyup is not initiated, terminate all TF operations.
- The E-scope should be monitored to verify video returns and to check for obstacles. If video is lost over terrain that should provide video returns, discontinue all TF operations.
- If the TF warning lamp, TF channel fail lamp, or the RADAR ALT LOW warning lamp lights with the radar altimeter pointer above the set index, the flight control system should be commanding a fail safe climb. If not, manually initiate a climb, and terminate all TF operations.
- Certain malfunctions of the TF computer can result in sufficient radar video to inhibit LARA override operation but insufficient returns to compute climb/dive commands. This condition may manifest itself in weak TF video displays to the crew.
- Any time the reference not engaged lamp lights during AUTO TF operation, immediate action must be taken to put the lamp out or terminate auto TF.

**Note**

While operating in auto TF, the reference not engaged caution lamp can come on due to conditions not associated with TFR operation. For example, when operating in auto TF with an autopilot submode engaged, moving or bumping the control stick will disengage the submode 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.

### Operating Configurations

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 obstacle avoidance capabilities of the system. 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 IMC 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.

**WARNING**

During operation over mountainous terrain with hard ride selected, the negative g's encountered during pushover maneuvers may cause the fuel low caution lamp to light. Prolonged pushovers could result in engine flame-out due to fuel starvation. Selecting medium or soft ride will reduce the possibility of this occurrence.

It is recommended that the index pointer in the radar altimeter be set to 10 percent below the clearance to be flown. If the low altitude warning lamp lights, closely monitor the TFR system and if descent continues, immediately take control of the aircraft.

**WARNING**

If a recurring undershoot of more than 10 percent of the set clearance is experienced during night/IMC, immediately go to the next higher clearance setting that will remain within limits or try the other TFR channel. Do not operate on a given channel at or below a set clearance where the 10 percent undershoot has occurred.

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 there is no forward video. This is known as radar altimeter override.

**WARNING**

Terrain following flight should not be attempted when there is an altitude error indicated on the radar altimeter indicator.

## AUTO TF DESCENT

It is recommended that an initial set clearance of 1,000 feet be used for all auto TF descents. Refer to figure 1-56.

When initiating an AUTO TF descent in the TF mode from above 5,000 feet AGL, the radar altimeter will not be locked on and will drop out its data good signal to the TFR. To prevent this signal from generating a flyup, the altimeter bypass switch must be positioned to BYPASS to give the TFR a pseudo altitude and data good signal. In BYPASS, the radar altimeter override function provides the TF dive commands. The TFR scope display must be monitored to ensure proper override function display prior to the descent. The radar altimeter override indication will be present at the 1,000 foot clearance plane setting and may be present at the 750 foot clearance plane setting. At lower clearance plane settings, the LARA override indication will not be present due to the pitch limiter restricting the dive angle to 12 degrees. Upon passing through 5,000 feet AGL, the radar altimeter should lock on, the bypass switch will return to NORM, and LARA altitude and data good signals will be supplied to the TFR and the radar altimeter override function display will disappear.

If the descent is being made to 1,000 feet set clearance, the dive angle will be limited to 10 degrees until the radar altimeter locks on. At this time, the dive angle will increase to 12 degrees. At other set clearances, the dive limits are 12 degrees above and below 5,000 feet. At 1,000 feet set clearance plane, a climb command should be indicated on the ADI command bar at approximately 2,000 feet AGL.

**WARNING**

The minimum altitude to start the pullout on an auto TF descent is 1700 feet AGL.

When a descent is made from high altitude in the vicinity of mountainous or rugged terrain, the clearance may suddenly increase beyond 5000 feet AGL and a fail/fly-up signal will result. Should this occur, immediately reposition the bypass switch to BYPASS and resume the descent. Monitor the radar scope to ensure radar altimeter override function display reappears.

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. Interrupt the auto TF letdown in time to level off at MEA. After the aircraft is established

# TYPICAL AUTO TF LET-DOWN PROFILE

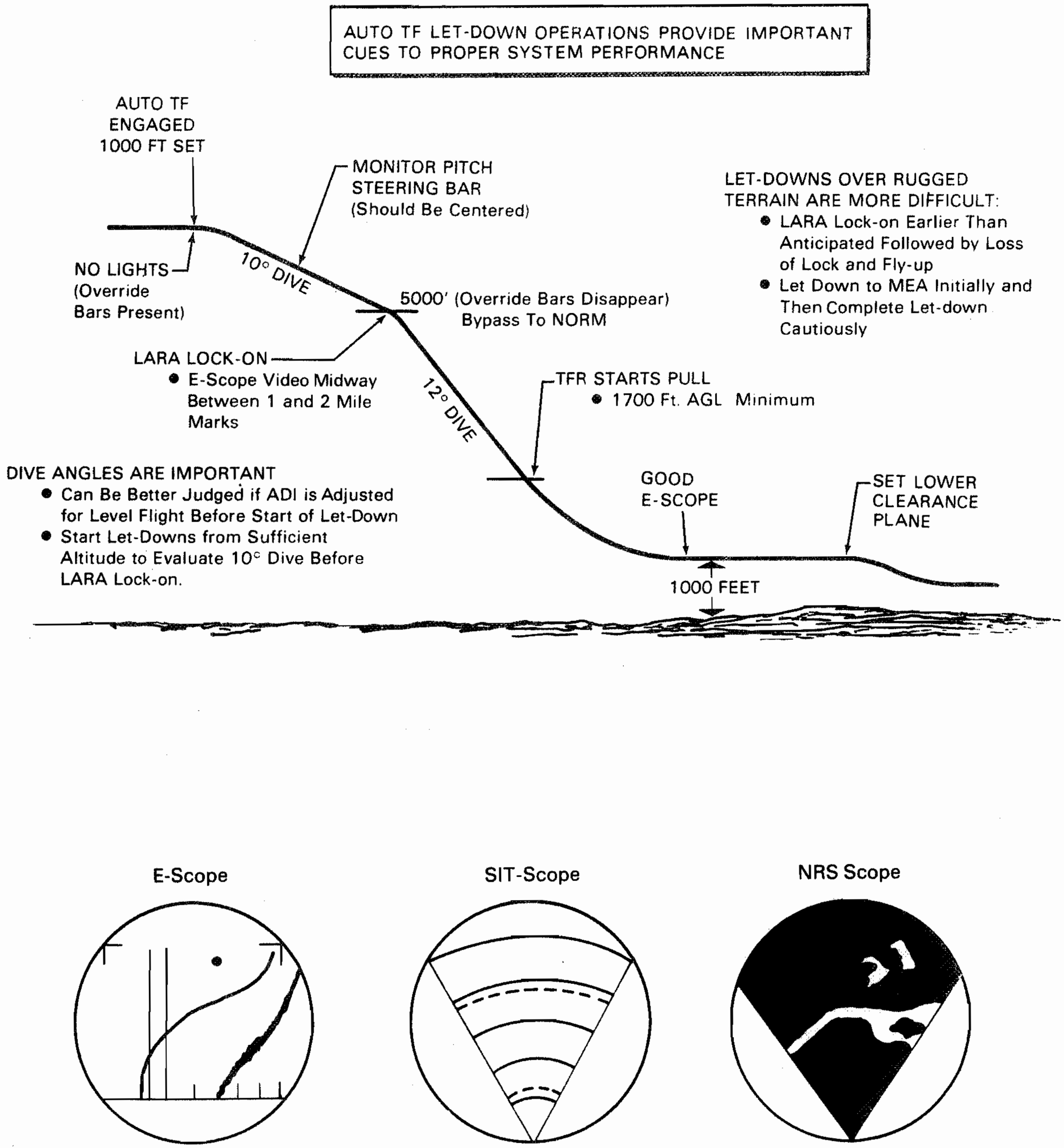


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**Figure 1-56.**

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 reinitiated. Aircrews should be aware that while flying at MEA, the 5,000 foot limit of the radar altimeter may be exceeded resulting in a TF fail-safe flyup. After leveling off at 1,000 feet, the desired operating clearance can be selected.

**WARNING**

When an auto TF letdown is made, the radar altimeter must be monitored to assure that it locks on after passing through 5,000 feet AGL.

**Note**

- Auto TF letdowns started at an altitude that does not allow dive angle to reach 10 ($\pm$1.0 degree) prior to LARA ring-in (5,000 feet AGL), does not verify proper radar altimeter operation.
- If the letdown 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. The altimeter override function begins to control the level off commands at approximately 2,000 feet AGL as the override command becomes more positive than the 12 degree limiter. The override function display will be presented on the radar scope as the radar altimeter override function initiates the letdown recovery.

During auto TF descent (figure 1-56), the terrain trace will be observed to move across the E-scope until reaching a point where it is approximately parallel to the ZCL. As the bottom of the terrain trace reaches a point half-way between the 1 and 2 mile points, the LARA should lockon and begin to provide actual terrain clearance information to the TF. As long as the aircraft remains above the terrain, no video should be observed on the SIT display, if selected. Although accomplishing a blind descent with the SIT display selected is not recommended, failures of the TF system may require manual letdowns to MEA with SIT selected. Any terrain showing on the SIT display indicates terrain that is above the FRL. As long as the aircraft remains above the terrain, no significant shadowing should be seen on the NRS. The LARA override warning bars will be visible during descent prior to LARA lockon in 1000 SCP and may be visible in 750 SCP. At lower set clearance plane settings, the override bars will not be visible.

**Radar Altimeter Override Function**

The radar altimeter override function provides the TFR with climb/dive commands based upon radar altimeter inputs. These commands are proportional to the difference between the actual radar altitude and the desired altitude when the terrain following radar return is unreliable or non-existant. The TFR switches to this function if no radar return is detected at a range of 3,500 feet to 4,500 feet (altimeter override gate) and the radar altimeter pitch angle command is more positive than the terrain following radar pitch angle command. Upon detecting radar returns in the altimeter override gate, the TFR switches back to climb/dive commands generated by the terrain following radar. A video display is presented on the TFR scope when the altimeter override function is computing climb/dive commands. In the TF mode with the E display being monitored, two vertical bars are displayed at the range of the altimeter override gate. In SIT or GM mode with TF selected on the other channel, dashed range marks are displayed at 2.5 and 7.5 nautical miles on the PPI presentation with either 10- or 15-mile range selected. In SIT or GM mode, with the 5-mile range selected, a dashed range mark of 2.5 nautical miles is present when in override. During normal TF operations, the radar altimeter override function displays may be present intermittently when cresting mountain peaks or overflying terrain which presents minimal radar return. When over calm water or flat terrain such as dry lake beds, dry wheat fields, smooth sand, or snow-covered areas, the override display will be present at all times until an area of sufficient radar return is intercepted and the climb/dive command generation shifts to the forward looking terrain following radar. This may occur with no radar return present in the LARA override gate as in the case of approaching higher terrain while still over an area of no radar return. The more positive command from the TFR, resulting from the higher terrain, will provide the climb/dive command, and the LARA override function display will not be present. Self-test circuitry monitors the override function. If the function fails to enable when forward video is absent in the override gate or if the override function enables properly but the pitch command fails to the maximum dive, a TFR system failure is declared. The TF data good signal to the flight control system is removed, and a TF fail/fly-up is initiated.

**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. This is due to the series trim driving to an improper position.

**Aircraft Trim**

When the auto TF switch is placed to the AUTO TF position, and the autopilot release lever is not held, the control stick will be centered. When the auto TF mode is controlling the aircraft, the pitch trim function of the stick trim button is inoperative. Auxiliary pitch trim should not be used during auto TF operation.

**WARNING**

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

**Low Altitude Fly-Up**

A TF fail fly-up will result in a +3.0 "g" command when the TF data good signal to the flight control system is removed.

**Note**

The TFR is limited to a maximum (+3.0 "g") command maneuver. However, the initial aircraft response to the fly-up command may be as much as +3.8 absolute "g"s.

The fly-up command is sent to the pitch damper and series trim with the AUTO TF switch in either AUTO TF or OFF. The ADI and aural tone will indicate a fly-up command. Additionally, the TF channel fail caution lamp of the channel in use, radar altitude low warning lamp, TF failure warning lamp, and the reference not engaged caution lamp will be lighted.

**Note**

The above lamp indications will not all be present for a LAM only induced fly-up.

In all fly-up cases the pitch trim function of the stick trim button is disabled and the control stick is centered. The fly-up circuitry is disenabled in the takeoff and land configuration; however, audio and light indications will still be present. A TF fly-up may be caused by one or more of the following conditions:

1. The aircraft descends below 68 percent of the selected clearance plane setting
2. An operating TF channel fails
3. The TFR detects an internal malfunction
4. The TFR detects a loss of one or more data good signals from the NRS, CADC, or radar altimeter. Refer to figure 1-52 to determine TFR system indications of loss of data good signals
5. The TFR computer detects an excessive error in input data
6. A detected malfunction in the radar altimeter override function

**Note**

A TF fail fly-up may occur if the aircraft attitude exceeds 20 degrees of pitch or 45 degrees of roll due to the radar altimeter breaking lock and/or roll gimbal limits being exceeded. R R

7. A malfunction detected by the low altitude monitor
8. Failure of the LARA interference blanking pulse with the ECM midband on.

**Note**

Electromagnetic interference from HF radio transmission on certain frequencies may cause a fly-up maneuver when operating 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, the terrain should be cleared visually, or the aircraft climbed to MEA.

Should a TF fail or a low altitude monitor fail occur and a fail-safe fly-up maneuver be initiated, the maneuver should be overridden by depressing the autopilot release lever 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. Before releasing the lever, the pilot should check that the channel failure caution lamp on the TFR control panel for the channel in the TF mode, is not lighted and that the radar altitude low warning lamp is not lighted. If either lamp is still on, releasing the autopilot release lever will allow the fly-up maneuver to be resumed.

**WARNING**

If the condition that has 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.

**Note**

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. If the fly-up continues, turn the radar altimeter off.

The TF fail safe fly-up signal is limited to 3.0 "g's" absolute (2.0 "g's" incremental) and is not roll compensated; that is, you must roll out of a bank to get all the "g's" in vertical. The fly-up can instantaneously give up to 3.8 "g's" absolute (2.8 "g's" incremental) due to a 40 percent over response, but will stabilize at 3.0 "g's" absolute. This fail safe fly-up signal always commands 3.0 "g's" through the top of the aircraft, regardless of its attitude. The TF fail safe fly-up signal and TFR climb commands are two different/separate electrical inputs; when a fail safe fly-up occurs, climb/dive commands are ignored and the flight control system switches to a separate, internally generated, fly-up signal.

**Note**

If the TFR is commanding a climb of more than 3.0 "g's" and a failure occurs, the fly-up command will decrease the commanded "g" to 3.0 "g's" absolute. This reduction could result in contact with the ground.

**Turning During TF Flight**

The TFR computes lead-into-turn for turn rates up to and including 2 degrees per second. The lead-into-turn function displaces the TFR antenna in the direction of the turn, up to a maximum of 4 degrees in azimuth. The four degree azimuth limit ensures that terrain data along the airplane instantaneous horizontal flight vector remains within the TFR radiated beam, and permits the maximum look-ahead (anticipation) for obstacles in the predicted horizontal flight vector. However, when turning toward tall, sheer, man made objects (such as towers) the detection time afforded by the lead-into-turn function could be insufficient to command the airplane over the obstacle.

**WARNING**

- During normal TF Operation limit turning flight bank angles to 30 degrees or less. If mission requirements dictate bank angles greater than 30 degrees, a maximum bank angle of 40 degrees at a minimum of 460 KTAS may be used.
- If bank angles of greater than 30 degrees are to be accomplished, flight planning shall confirm the absence of tall, sheer, man made objects in or near the turning flight path.
- If the turn "g" limit caution lamp comes on during a turn, reduce the turn rate or bank angle until it goes out.
- If a TF fail/fly-up occurs in a bank, go to zero bank when terrain permits.

**Note**

A turn rate of 2 degrees per second is achieved at the following combinations of bank angle and airspeed.

| BANK ANGLE | KTAS |
|---|---|
| 30° | 320 |
| 32° | 340 |
| 34° | 370 |
| 35° | 390 |
| 36° | 400 |
| 38° | 430 |

**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$0.5) degrees or greater. This antenna displacement can be caused by a malfunction in the TFR, an erroneous drift signal from the inertial nav system, or actual aircraft drift of 4.5 ($\pm$0.5) degrees or greater. 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 Check," Section II. The lamp may come on momentarily during turning maneuvers. This is a normal condition. If the lamp is on steady prior to or during an AUTO descent, 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 an AUTO descent 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.

**Auxiliary Navigation Modes**

Terrain following flight in the auxiliary navigation modes should be limited to day VMC 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 ensure that excessive drift errors do not build up.

**Changing Terrain Clearance Settings**

Rapid switching beyond more than one incremental set clearance may cause the aircraft to detect a 68 percent violation and assume a climb angle greater than 20 degrees, thus causing the radar altimeter to lose range lock-on. As a result, the aircraft will not level off at the desired clearance setting and the fly-up must be overridden by manual control. It is, therefore, recommended that the aircraft be allowed to stabilize at each incremental setting.

**Effect of Precipitation**

If heavy rain or thunderstorm activity is encountered, TFR operation may not be possible. Use of nav radar to vector around weather cells is recommended and the 500 set clearance will provide more capability in dense clouds or rainfall. 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 rain returns approach the ZCL a climb command can be expected. When a climb command occurs, do not allow the TFR to exceed a 20-degree climb attitude or exceed the angle-of-attack/airspeed limits. 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.

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. The TFR scope display must be monitored to determine radar altimeter override operation.

**WARNING**

If E-scope blanking due to weather conditions is observed or suspected, an immediate climb must be initiated to MEA.

**Flight Over Wooded Terrain**

During flight over sparsely wooded terrain, dead or defoliated trees, the TFR may not command sufficiently on the back scatter from the trees to maintain the selected clearance over the tops of the trees thus the tops of isolated trees may project above the 68 percent fly-up threshold at the lowest clearance.

Over densely wooded areas, the radar altimeter will tend to measure aircraft clearance alternately between the ground and the tops of the trees, thus the indicated radar altitude may become very erratic and may generate a number of short duration fly-ups. In addition, when LARA is in auto mode, intermittent lighting of the LARA multiplexer caution lamp may occur with intermittent fail/fly-up due to exceeding comparator limits.

**Flight Over Sloping Terrain**

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/degree 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.

**Flight Over Mountainous Terrain**

When flying auto TF over steep mountainous terrain, a fly-up command without a failure indication may occur shortly after cresting a peak which has a large extended slope with a small hill on the back side. This fly-up is an inherent characteristic of the terrain following radar system and the frequency and severity is a function of the terrain, set clearance, airspeed, ride setting and aircraft gross weight. Fly-ups for a duration of up to 4 seconds may be experienced with angle-of-attack approaching 15 degrees with buffet onset. Flyups of this nature can be minimized, if desired, by performing flight over rough mountainous terrain at a higher set clearance and/or selection of medium or soft ride.

**Flight Over Sand or Snow Covered Areas**

When flying over sand or snow covered areas, there will be little energy returned to the TFR. The system may revert to the LARA override mode providing safe flight only if the ground does not rise rapidly.

**Use of The 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 and processed.

**WARNING**

When the video on the TFR scope becomes weak or barely visible, the TF system may revert to LARA override. If under these conditions the pilot cannot assure that the surface is water or other low reflective, smooth terrain, an immediate climb must be initiated.

Figures 1-57 through 1-62 provides examples of E-scope displays associated with flight over various type of terrain features. Also included are sample displays from the NRS and SIT Scope that would be associated with these terrain features. Due to possible failures of the forward looking TF radar, it is absolutely imperative that coordination be constant between the EWO and pilot during night or IMC TF flight. Such coordination should relate the information presented by the various radar displays and actual aircraft performance.

As the aircraft approaches an area of low reflectivity (Figure 1-57), such as an area of sand or a lake or river, a blank spot corresponding to the range of the no-show area will be noted on the E-scope display. The display should show a vertical area of missing returns as indicated at A in example 1. With no terrain above the FRL, no returns should be present on the SIT display, if selected, as shown in example 1. The NRS will show the shape of the no show area as indicated at A in example 3. As the no show area is approached, the system may revert to the LARA override mode if no positive climb commands are being received from the forward video. This will be indicated by the LARA override display as shown in examples 4 and 5.

Mountainous terrain features, (figure 1-58), that are below the aircraft altitude will show on the E-scope as a blank spot in the terrain trace. The terrain trace should curve upward somewhat just prior to the area of no return. This shadowing blank area will differ from the no show area caused by a flat area of low reflectivity by the lack of missing vertical return area. If the terrain feature does not extend above the FRL, no display should be present on the SIT scope. On the NRS, shadowing should be present and should decrease in size as the aircraft approaches the obstacle. As long as the object is below the aircraft altitude, terrain returns should be present beyond the shadow.

As the aircraft approaches an area of terrain that is above the aircraft altitude (figure 1-59), the E-scope will show a return that curves upward toward the ZCL with no returns behind it as shown in example 1. As long as the aircraft remains below the object, no returns will be seen behind it. The closest edge of the object will be displayed on the SIT scope (example 2) and shadowing will extend to the edge of the NRS display as indicated in example 3. As the aircraft climbs above the obstacle, terrain will begin to paint behind it on the E-scope and NRS (examples 4 and 6). The object will disappear from the SIT display (example 5), if selected. The system may display LARA override bars during the pushover until terrain begins to cause a climb command.

# RADAR DISPLAYS APPROACHING A NO-SHOW AREA OVER LEVEL TERRAIN

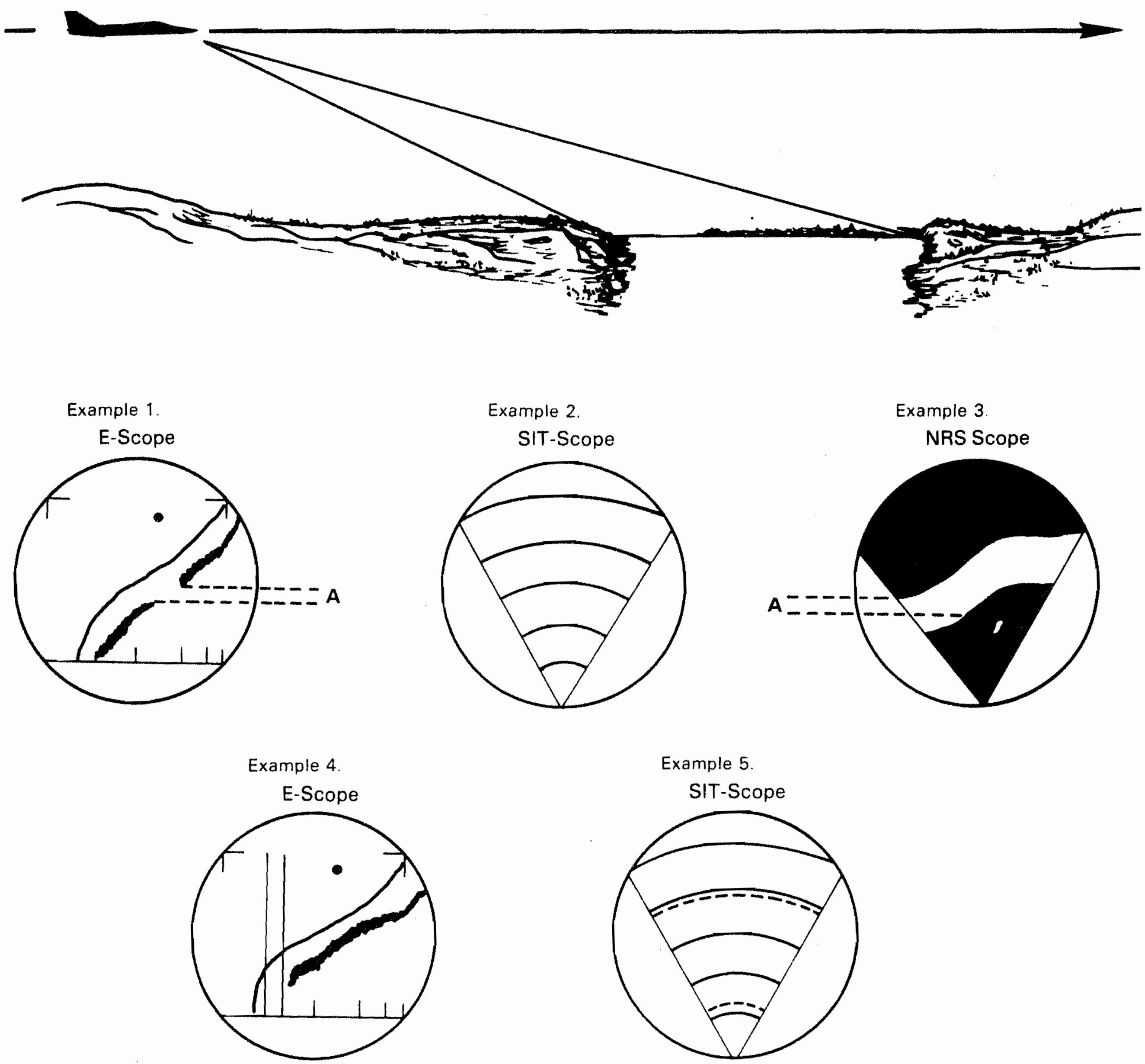


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**Figure 1-57.**

# RADAR DISPLAYS APPROACHING A HILL BELOW THE AIRCRAFT FLIGHTPATH

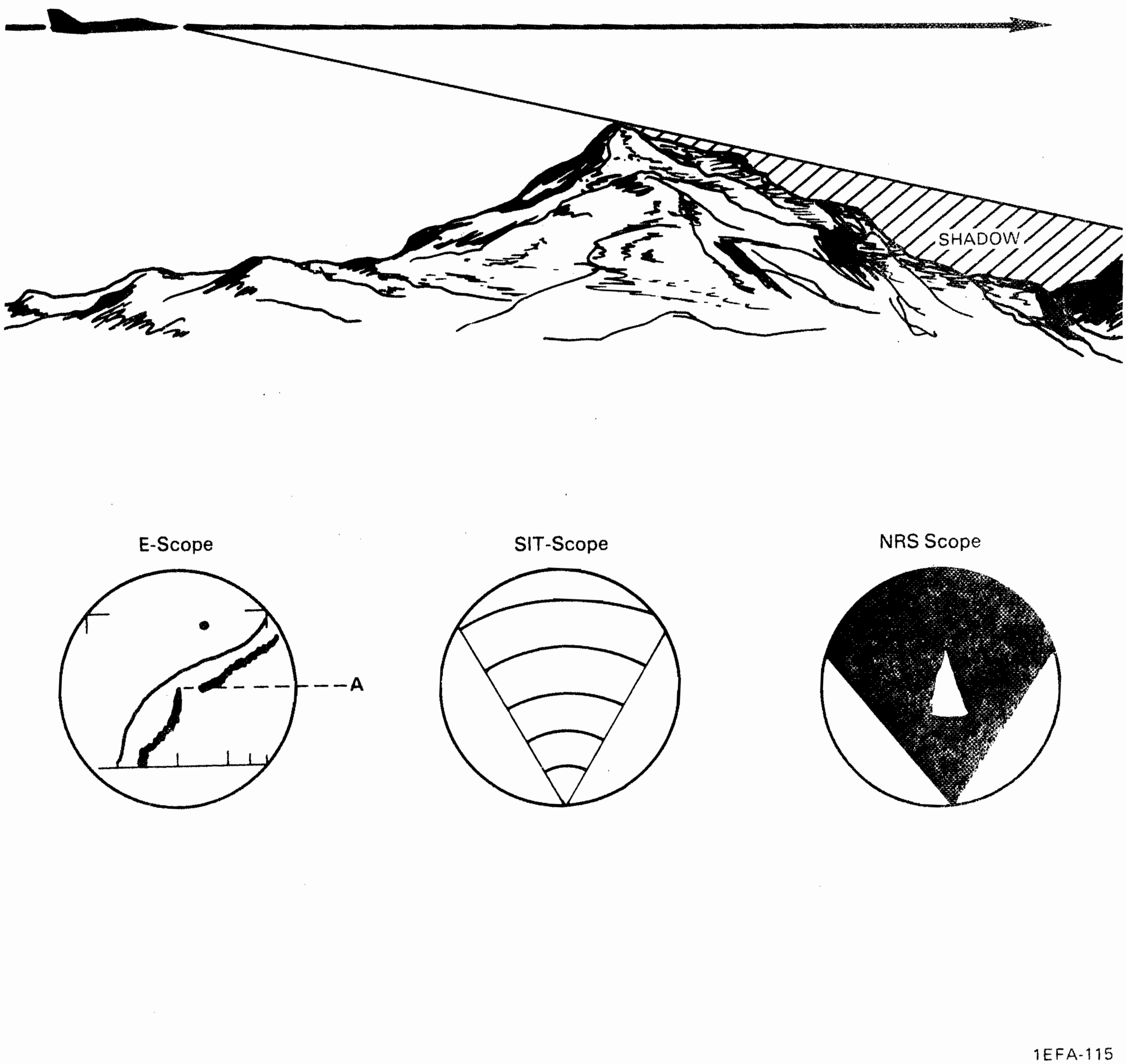


Figure 1-58.

# RADAR DISPLAYS APPROACHING MOUNTAINOUS TERRAIN ABOVE THE AIRCRAFT ALTITUDE

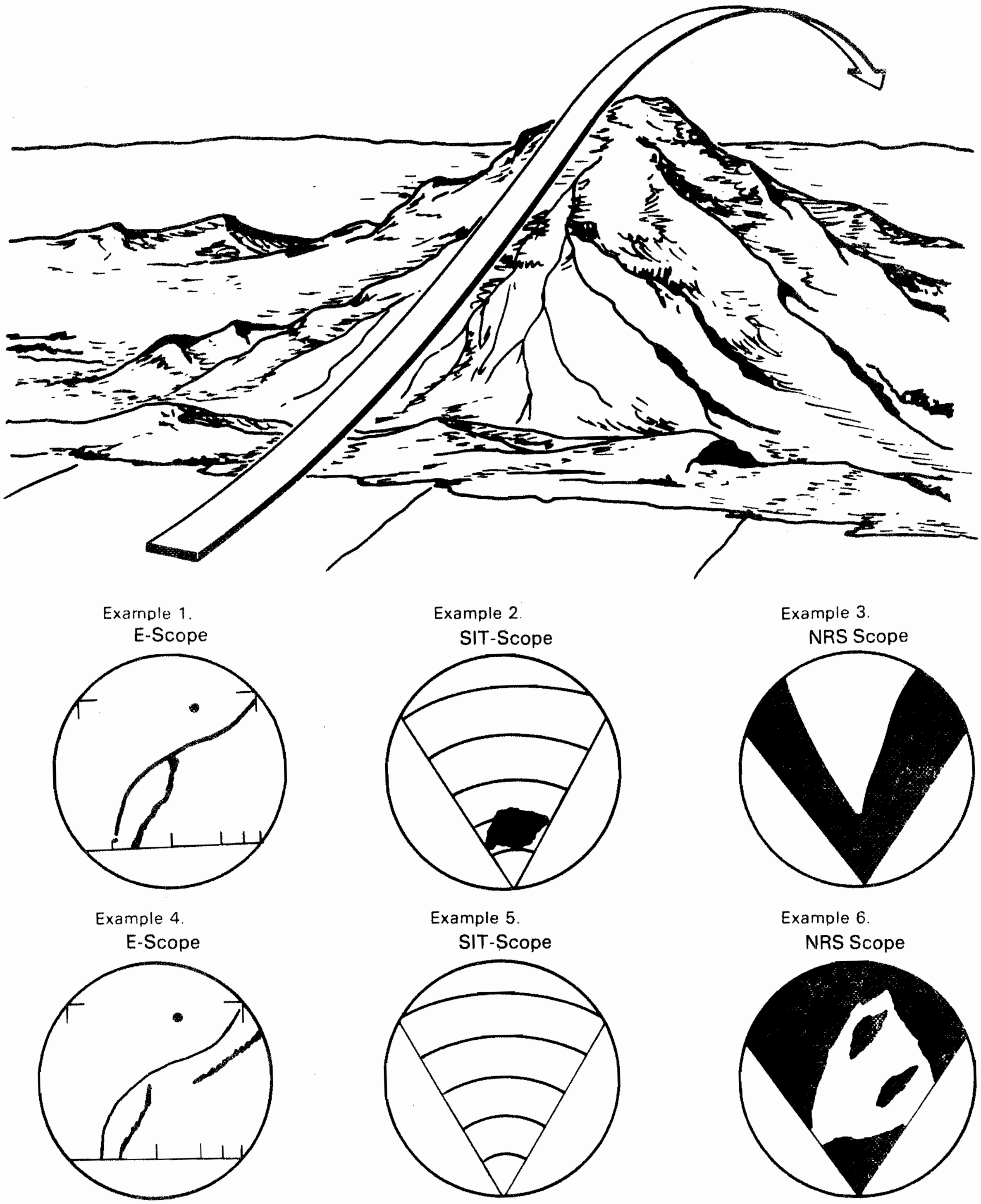


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**Figure 1-59.**

An object of low radar reflectivity (figure 1-60), such as a sand dune or snow covered mountain, creates a significant hazard to TF flight since the TF system may not command on it and direct a climb. Such terrain should be treated with extreme caution. If the object is below the aircraft altitude, such as situation A, the E-scope trace will show a vertical area of no return similar to a no-show flat terrain feature. This is an extreme case as usually there will be an area at the base of the object that provides returns. No return will be seen on the SIT display and some shadowing will be seen on the NRS, although the object itself may not provide any returns.

If the height of the object is above the aircraft altitude such as situation B, there will be no terrain trace beyond it on the E-scope. It still may not show on the SIT display due to its low reflectivity. However, there would be a shadowing effect that extends to the maximum range of the NRS scope. Any situation that causes a blanking of a portion of the E-scope should be treated with extreme caution, and consideration should be given to a climb to MEA prior to reaching the object.

**Note**

> Dependent on the slope and height of the terrain, the LARA may not command a TF fail fly-up sufficiently early to clear the terrain.

Approaching landfall from overwater (figure 1-61) requires close coordination between the pilot and EWO. While over water, the system will be in the LARA override mode (indicated by the LARA override display). Approaching landfall the crew must determine that the system has terminated LARA override mode and is responding to terrain returns. As the aircraft approaches the shore line, a terrain trace should begin to appear at the right edge of the E-scope and move toward the left. The NRS scope will begin to show terrain at the far edge of the scope and it will move toward the vertex. The SIT scope will display any terrain that is above the FRL.

It is recommended that an overwater TF letdown be accomplished a sufficient distance from landfall to allow an opportunity to determine correct system operation prior to encountering high terrain. When an auto TF letdown is accomplished over water, the crew should expect a more abrupt than normal rotation to level flight due to the fact that the LARA is providing the only information used in the level off command.

The most difficult situation to evaluate using the various radar displays is where the aircraft is approaching terrain higher than the aircraft, with still higher terrain beyond. The presence of terrain beyond a particular obstacle can give a false sense of security to the unalert crew. In the situation depicted in figure 1-62, the aircraft is aproaching three hills, all of which are above the aircraft altitude. The E-scope display may be similar to one with a series of hills below the aircraft altitude. The SIT scope will show the first object above the FRL and subsequent objects that are not in shadow. However, in mountainous terrain, the SIT display is not recommended. The NRS would show a shadowing effect but the shadows may not extend to the maximum range of the scope. Generally, the shadows could be expected to increase in size if the objects are above the aircraft altitude and decrease in size if they are below the aircraft. Crews must be alert for this type of situation during mission planning and be prepared to climb to MEA if there is any doubt of being able to evaluate TF system performance in flight.

## TFR 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 1-63 for malfunction analysis of TFR system malfunctions. The fail conditions noted in figure 1-63 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 degree 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 nav 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-scope for evidence of the test 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-scope. If the intermittent condition is present in

# RADAR DISPLAYS APPROACHING A SAND DUNE OR OTHER LOW REFLECTIVITY OBSTACLE

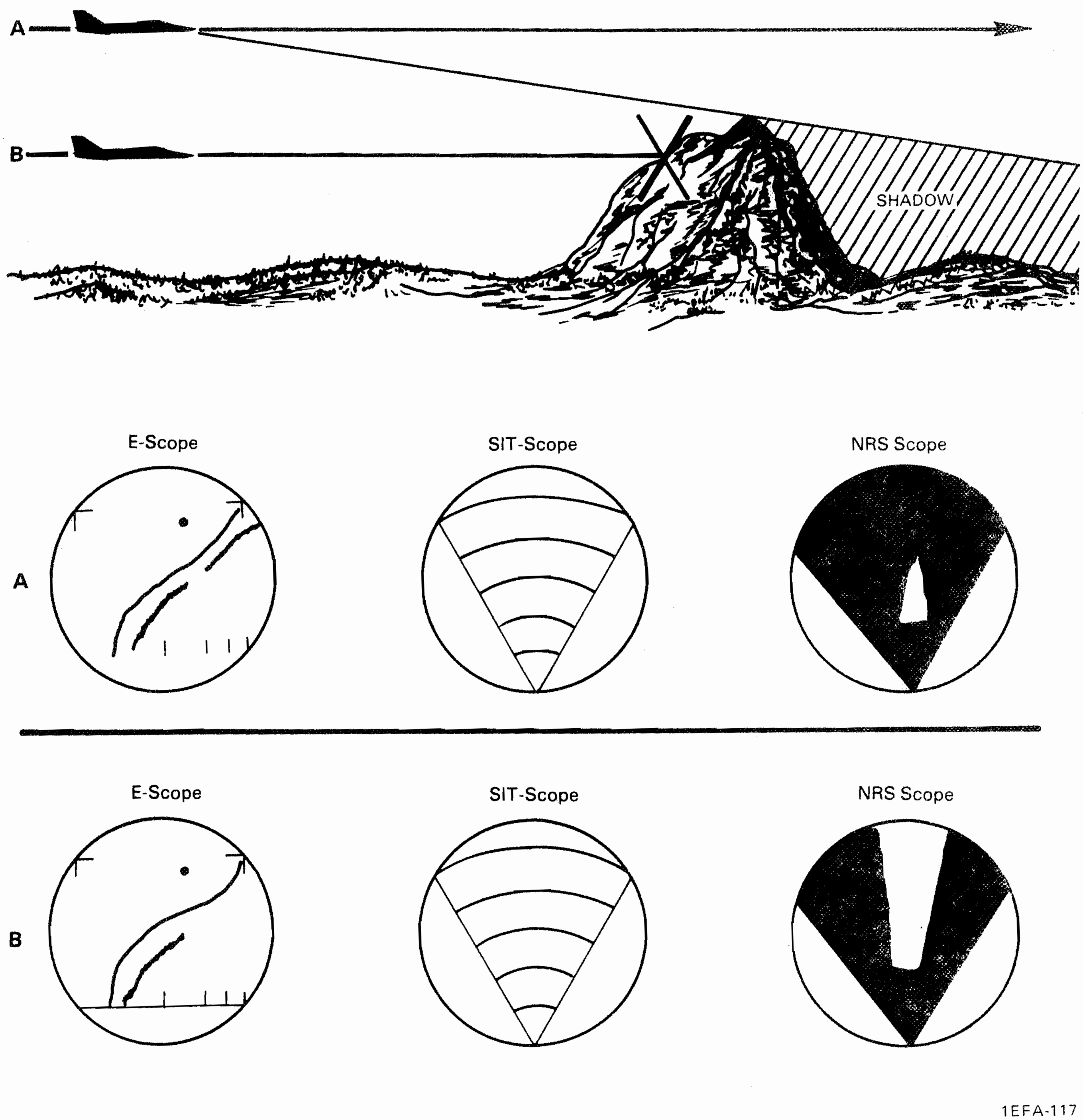
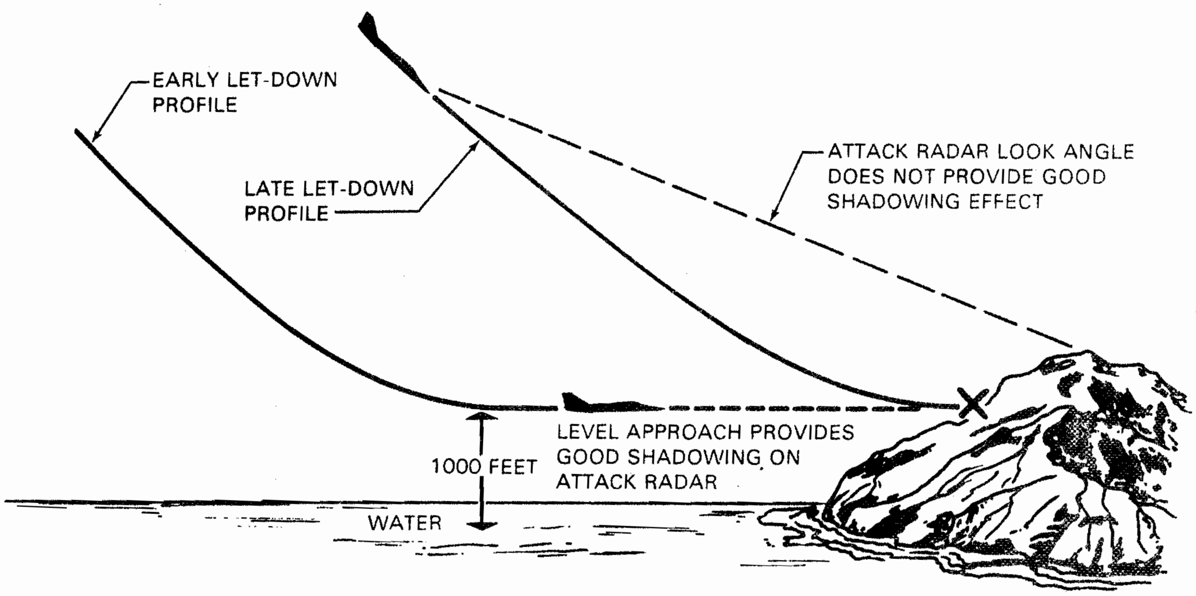



Figure 1-60.

# RADAR DISPLAYS APPROACHING LANDFALL

## PLAN EARLY LET-DOWN FOR OVERWATER ENTRY




- COMPLETION OF LET-DOWN WELL BEFORE LANDFALL GIVES MORE TIME TO ASSESS TERRAIN PROFILE AND YOUR LOCATION

E-Scope | SIT-Scope | NRS Scope

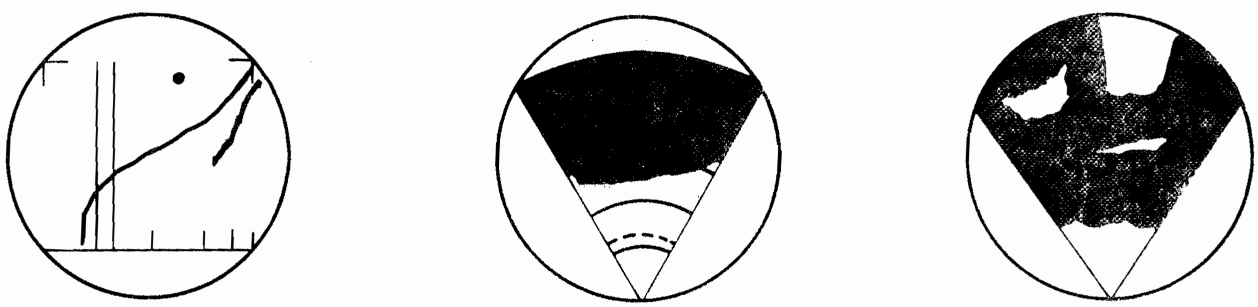


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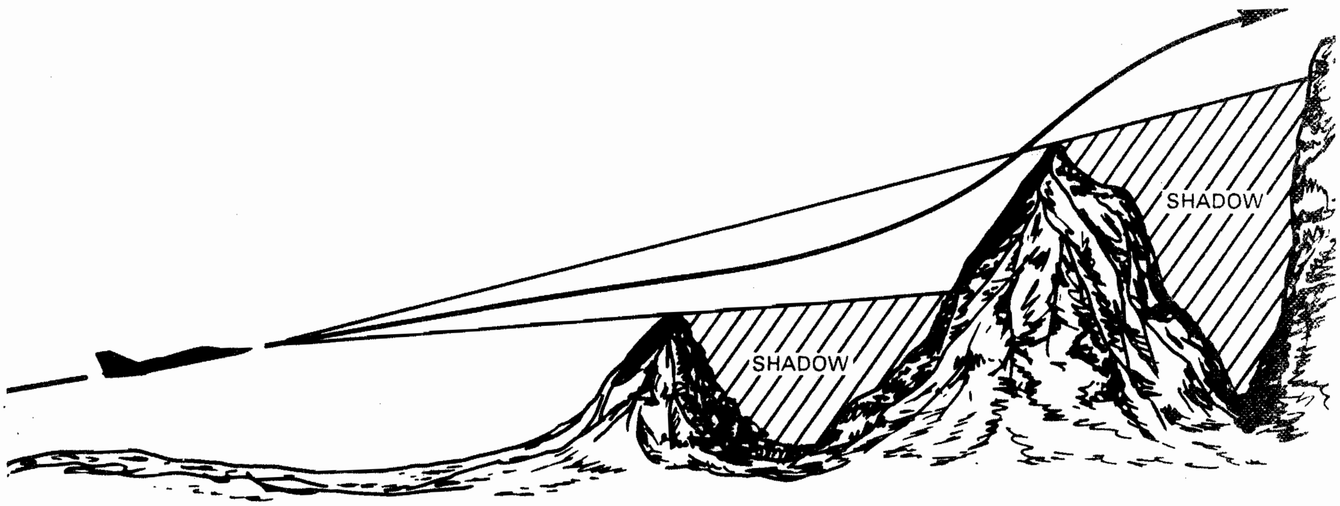
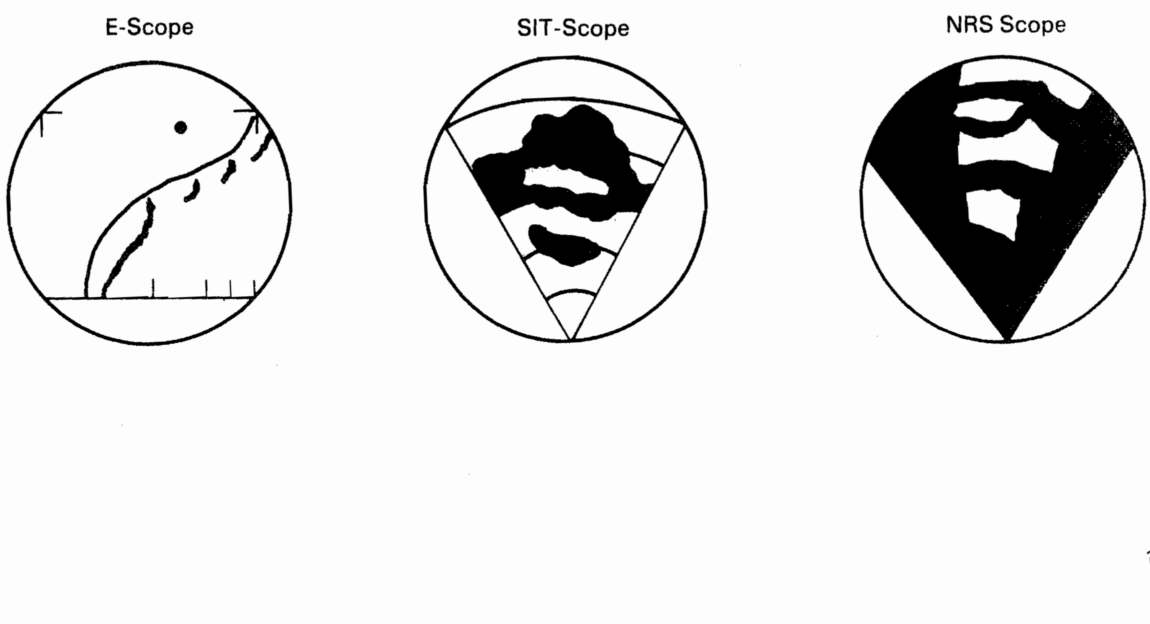
Figure 1-61.

# RADAR DISPLAYS APPROACHING PROGRESSIVELY RISING TERRAIN




1EFA-119

Figure 1-62.

# 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 the data good signal from the nav radar roll pedestal. | Check nav radar malfunction lamp and/or antenna cage pushbutton indicator lamp lighted. If roll pedestal is not stabilized, scope displays for nav radar and TFR in SIT or GM will be blank on one side of scan. If antenna cannot be uncaged, check primary attitude/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 only. | 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. If the following does not correct the problem, terminate TF operation. If the channel selector switch is in AUTO and LARA multiplexer caution lamp is out, terminate TF operation. If the channel selector switch is in AUTO and the LARA multiplexer caution lamp is on, select channel operation. If the channel selector switch is in CHAN 1 or 2 (LARA multiplexer lamp normally on), select other channel. |
| | Loss of data good signal from radar altimeter. | If channel selector switch is in AUTO and LARA multiplexer caution lamp is on, select CHAN 1 or 2. |
| LARA multiplexer caution lamp lighted. | Altitude $>$5,000 AGL, in TFR, SIT, or GM. | Normal condition. Lamp will remain on until LARA lock-on occurs. |
| | Altitude $<$5000 AGL, TFR in TF, SIT, or GM, and LARA channel selector switch in AUTO; loss of LARA lock-on of either radar altimeter channel or altitude data from both channels exceeds comparison limit of 12.5 ($\pm$2.5) percent. | Select CHAN 1 or 2 and perform LARA self-test to determine which is operable. |

Figure 1-63 (Sheet 1)

# TFR Malfunction Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | Altitude <5000 AGL, TFR in TF, SIT, or GM, and LARA channel selector switch in CHAN 1 or 2. | Lamp will normally be on to indicate no LARA multiplex capability. |
| 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.<br><br>Malfunction in synchronizer transmitter.<br><br>Malfunction in amplifier power supply or antenna assembly.<br><br>Aircraft below 68 percent terrain clearance setting.<br><br>Malfunction in TFR radar altimeter override function circuitry. | Check E-scope display for test pulse at top end of scan.<br><br>Check E-scope display for discontinuities in the zero command line or complete lack of display.<br><br>Check E-scope display for proper scan.<br><br>Compare radar altimeter indication with terrain clearance setting.<br><br>Use opposite TFR channel for all of above conditions.<br><br>Check E-scope display for two bar radar altimeter override indication with discontinuity of forward video in the first mile (3,500 to 4,500 feet).<br><br>Select other TFR channel if altimeter override indication is not present. |
| 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. |

**Figure 1-63 (Sheet 2)**

## TFR Malfunction Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| In TF mode, radar altitude low warning lamp is lighted but indicated radar altitude is higher than index pointer setting. | Discrepancy between the two altitude signal outputs of radar altimeter channel. | Select opposite radar altimeter channel. |
| | Malfunction in low altitude monitor. | Terminate TF operations. |
| | Aircraft is below 68% of selected clearance. | Terminate TF operations. |

**Figure 1-63 (Sheet 3)**

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 Section I.

   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.

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 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.

a. Probable Cause: 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. Probable Cause: Certain malfunctions in the TFR computer may allow cross-talk from the nav radar transmit pulse to trigger the command circuits.

Check: Turn the nav radar out of the XMIT mode briefly to see if the erratic commands are corrected.

Corrective Action: Use opposite TFR channel.

c. Probable Cause: 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 pitch steering bar 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. Probable Cause: Two or more aircraft flying in formation 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. Probable Cause: 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 "Terrain Following Mission Planning," this section.

## JSS/ECM SYSTEMS

For all information pertaining to these systems, refer to supplements, T.O. 1F-111(E)A-1-2, -3. Refer to Section II for turn-on procedures.

## 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 (6, figure FO-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 pushbutton labeled CURTAIN RELEASE must be depressed to release the curtain for retraction. A decal located adjacent to the curtain release button contains instructions for extending or stowing the curtain.

**Forward Panel**

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 crewmember 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 (3, figure 1-31). Pressure for the anti-G suit is supplied from the engine compressor section. A test button (2, figure 1-31) marked ANTI-G 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 (4, figure FO–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 (9, figure 1-11 and 3, figure 1-26). 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 and a data stowage bag aft of the digital display indicator control. 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. The data stowage bag contains a flap with a velcro fastener.

## CHART STOWAGE

Two chart stowage compartments are located on each side of the lighting control panel (15 and 21, figure FO–2). 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 1-11), located at the aft end of the left sidewall, is provided for stowing the ejection system safety pins.

## SPARE LAMP HOLDER STOWAGE

A stowage compartment (4, figure 1-26), located at the aft end of the right sidewall, is provided for stowing spare light bulbs.

## CHECKLIST STOWAGE

A space for stowing the checklist is provided on the left sidewall (1, figure 1-11). A nylon strap retains the checklist in place.

## FOOD STOWAGE COMPARTMENT

A food stowage compartment (23, figure FO–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 (24, figure FO–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.

## CORRECTION CARD HOLDERS

Card holders are provided under the glareshield for EPR setting and compass correction. Each card holder is attached by spring tensioned hinges riveted to the glareshield. The card holders are pulled out into position for reading purposes and spring back against the lower side of glareshield when released.

## LIQUID CONTAINERS

Two insulated liquid containers (13, figure FO-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.

# SERVICING DATA

For servicing information refer to figures 1-64 through 1-66.

# TIRE INFLATION PRESSURES

| Aircraft Gross Weight | Inflation Pressure (PSI) MLG Tire Size 47X18-18 | Inflation Pressure (PSI) NLG Tire 22X6.6 10-20 Ply Rating | Inflation Pressure (PSI) NLG Tire 21X7.25 10-20 Ply Rating |
|---|---|---|---|
| 72,000 & below | 135-145 psi | 215-225 psi | 245-255 psi |
| 72,000 to 84,000 | 155-165 psi | 255-265 psi | 265-275 psi |
| 84,000 to max gross weight | 175-185 psi | 275-285 psi | 285-295 psi |

NOTES:

1. Over inflation is permissible for all gross weights, but is not to exceed maximum specified above.
2. Difference in inflation pressure between the two MLG tires or between the two NLG tires must not exceed 5 psi.

**Figure 1-64.**

# PNEUMATIC PRESSURES

| Temperature (Degrees F) | Wheel Brakes (PSI) | Autopilot Damper Servo/Horizontal Stabilizer Accumulators (PSI) | Overwing Fairing (PSI) | Spike/Inflight Refuel/Cowls/ Landing Gear Emergency Extension Reservoirs (PSI) |
|---|---|---|---|---|
| Above 100 | 850 to 900 | 1,500 to 1,600 | 1,900 to 2,000 | 3,250 to 3,500 |
| 50 to 100 | 750 to 850 | 1,350 to 1,500 | 1,700 to 1,900 | 2,850 to 3,250 |
| 10 to 50 | 700 to 750 | 1,200 to 1,350 | 1,550 to 1,700 | 2,650 to 2,850 |
| −60 to 10 | 600 to 700 | 1,000 to 1,200 | 1,200 to 1,550 | 2,450 to 2,650 |

**Figure 1-65.**

# SERVICING DIAGRAM

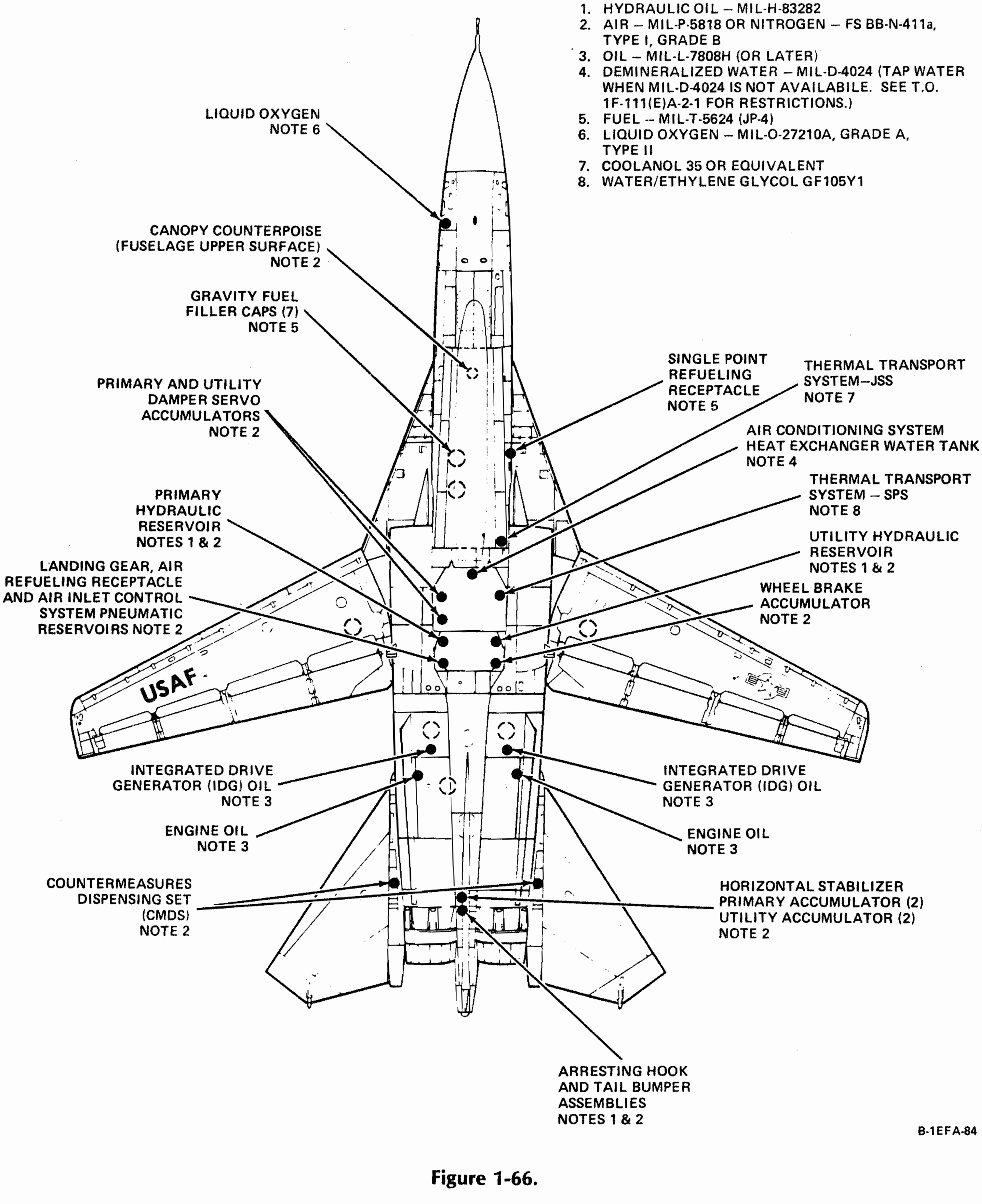



**Figure 1-66.**

This is the last page of Section I.

# SECTION II

# NORMAL PROCEDURES

## TABLE OF CONTENTS



## 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-111(E)A-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.

## CHECKLISTS

This Flight Manual contains only amplified procedures. Flight Crew Checklist T.O. 1F-111(E)A-1CL-1 is issued as a separate document.

**Note**

It is the responsibility of the aircraft commander to assure that all checklist procedures are complied with.

## PREFLIGHT CHECK (AC)

### BEFORE EXTERIOR INSPECTION (AC)

1. AFTO Form 781 - Checked.
2. Ejection handle safety pins (2) - Installed.
3. Canopy center beam safety pins (3) - Installed.
4. Bilge pump lockpin - Stowed.

5. Quick rescue kit - Sealed and stowed.

6. Thermal curtains - Stowed. (If applicable)

7. Emergency oxygen bottle pressure - Check, 1,400-2,500 psi.

8. All circuit breakers - In.

9. Ground check panel - Check.

- Computer power switches (3) - On.
- Central Air data computer power switch - POWER.
- Ground ignition cutoff switch - NORM.
- Gyros power switch - GYROS.
- Translating cowl test switches - NORM.
- Mach trim test switch - NORM.

10. Lighting control panel - Set. (If desired)

11. Radio beacon set - ON or as applicable.

To prevent inadvertant actuation of the radio beacon, insure the safety plug/lanyard is properly installed.

12. Auxiliary brake handle - Pull. (If previous landing not within two hours)

## EXTERIOR INSPECTION (AC/EWO)

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. Following the route shown in figure 2-1, check all surfaces for any type of damage; signs of fuel, oil, hydraulic or other fluid leaks that may have developed since the preflight inspection. Check all access doors and covers for security and all protective covers removed. Check for missing fasteners. Check pylons and stores secure.

**Note**

The stores station inspection will be accomplished concurrently with the aircraft exterior inspection.

1. Left forward area.

a. Static ports.

Insure that static ports are clean and unobstructed.

b. Angle-of-attack and sideslip probes.

Check for damage and freedom of movement.

c. ECM antennas.

d. Pitot boom.

Check for damage, openings unobstructed.

2. Nosewheel well.

a. Landing and taxi lights.

Check for security, lenses intact.

b. Fuselage fire bottle - 350 to 650 psi.

c. Impact attenuation valve port.

Insure clear of obstructions.

d. Nose gear ground safety pin - Installed.

**CAUTION**

Inspect nose gear carefully and ensure that no binding exists against the pin. If pressure or binding on the pin is encountered, notify maintenance. Do not attempt to remove the pin.

e. Tires, wheels, and strut.

f. Uplock roller.

Check for freedom of movement.

3. Right forward area.

a. ECM antennas.

b. Angle-of-attack probe.

c. Total temperature probe.

# EXTERIOR INSPECTION

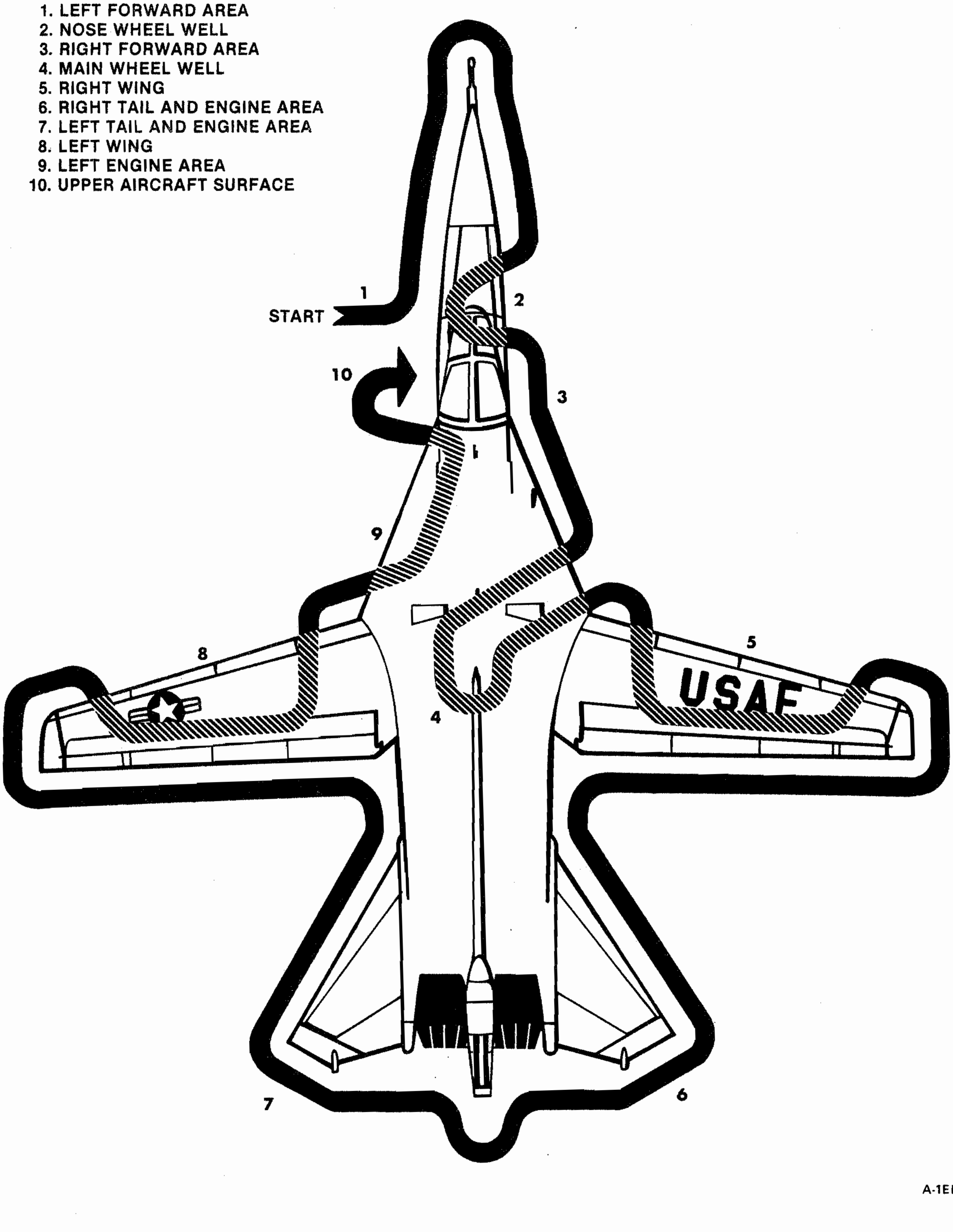



Figure 2-1.

d. Static port.

e. Local Mach probe.

f. Spike.

Full forward and contracted.

g. Intake.

- Check general condition and free of foreign objects.
- Cowls - Check.

h. Weapons bay pallet and radome latches. Check fastened, secured and safetied.

4. Main wheel well.

**Note**

Refer to "Pneumatic Pressures," Section I, for extreme ambient temperature conditions (below 10°F or above 100°F).

a. Two damper servo accumulators - 1,200 to 1,500 psi.

b. One overwing fairing accumulator - 1,550 to 1,900 psi.

c. Five emergency accumulators - 2,650 to 3,250 psi.

d. Primary and utility hydraulic reservoir accumulators - 300 to 500 psi.

e. JSS coolant accumulator gas volume - 40 to 60%. (Per temperature decal)

f. Gear pin - Removed.

g. Uplock assembly - Properly positioned.

Ensure that the red tip of the hook assembly is aft of the uplock roller guide.

h. Two brake accumulators - 700 to 800 psi.

Check brake pucks flush to verify the brake accumulators have been bled.

i. Gear emergency extension system safety clamp - Removed.

**CAUTION**

The landing gear emergency system may be actuated by contact with the lever actuator located in the main wheel well on the left side.

j. Hydraulic pump handle - Stowed, pin installed.

k. SPS (ALQ–137) coolant reservoir level - Check. (Minimum one inch from bottom of sight gage)

l. Starter cartridge case cover - Secure.

m. Speedbrake ground safety lock - Installed.

**Note**

For normal operations, the speedbrake/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.

n. Fuel shutoff valves - Open (red lever right).

o. Water filler pipe cover - Secure.

p. Air refueling system accumulator - 2,650 to 3,250 psi.

q. Right wheel, brake, tire and strut.

5. Right wing.

a. Slats, flaps, rollers and drive asembly. (If down)

b. Flap vane actuator levers - Cocked. (If down)

6. Right tail and engine areas.

a. Low band antenna.

b. Aft gear door.

c. Fire extinguisher pressure - 450 to 650 psi.

d. Oil cap.

e. Wing seal.

f. Strake.

c. Fire extinguisher pressure - 450 to 650 psi.

d. Oil cap.

e. Wing seal.

f. Strake.

g. Chaff/flare safety blocks - Removed. (If applicable)

h. Afterburner section and engine.

i. Tail bumper.

j. Tail hook pin - Removed.

k. Horizontal Stabilizer - Checked.

Lift up on the trailing edges of the horizontal stabilizer and check for less than 0.25 inch freeplay. If an unlocked stabilizer is encountered, the freeplay check should be reaccomplished with the control stick held forward during the check.

7. Left tail and engine areas.

a. Accumulator access door - Secure.

b. Afterburner section and engine.

c. Strake.

d. Wing seal.

e. Oil cap.

f. Low band antenna.

g. Chaff/flare safety blocks - Removed. (If applicable)

h. Left wheel, brake, tire and strut.

8. Left wing.

a. Slats, flaps, rollers, drive assembly, and guide vanes. (If down)

b. Flap vane actuator levers - Cocked. (If down)

9. Left engine area.

a. Intake.

- Check general condition and free of foreign objects.
- Cowls.

b. Spike.

Full forward and contracted.

c. Local Mach probe.

d. Weapons bay radome - Secure. (All fasteners installed)

10. Aircraft upper surface.

a. Fuel caps - Properly seated.

b. Forward SPS antennas.

## INTERIOR INSPECTION (AC)

### Power Off (AC)

1. Harness and leads - Attach and check.

- Ensure that the yoke of the restraint harness is adjusted firmly against the neck with head against headrest and sitting errect 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.

**WARNING**

Failure to ensure lap and shoulder harnesses are snug may result in delayed activation of critical switches/handles during out-of-control maneuvers. It may also result in spinal injury during ejection.

- 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. (As applicable)

2. Seat pan - Check for freedom of movement over full range of travel.

**Note**

If the seat pan does not adjust freely fore and aft, it may not be possible to assume an optimum position for ground impact following an ejection.

3. Oxygen system and personal equipment - Connected and checked.

**CAUTION**

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

- Oxygen control lever - OFF then ON.

  Check that 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 - 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.

- Oxygen control knob - NORM.

  Inhale and check movement of the diluter valve through the screen in the top of the regulator.

4. Auxiliary flight reference system compass mode selector knob - SLAVED and set present latitude.
5. Pitot/probe heater switch - OFF/SEC.
6. Engine/inlet anti-icing switch - AUTO.

**Note**

Operation with engine/inlet anti-icing switch in MANUAL for extended periods may result in an increased boil-off rate of environmental system cooling water and result in a loss of pressurization and cooling capacity after system shutdown due to a dry water tank.

7. Windshield rain removal selector switch - OFF.
8. Auxiliary pitch trim switch - STICK.
9. Flap/slat switch - NORM.
10. Interphone panel - Set.
11. Rudder authority switch -AUTO.
12. Flight control disconnect switch - NORM. (Cover down)
13. Autopilot damper panel switches - OFF.
14. Throttles - OFF.
15. Speed brake switch - IN.
16. Flap/slat handle - Corresponds with surface position.
17. Wing sweep handle - Corresponds with wing position.
18. Anti-skid switch - OFF.
19. Ground roll spoiler switch - OFF.
20. Translating cowl emergency override switch - NORM.
21. Translating cowl switches (2) - AUTO.
22. Ground start switch - OFF.
23. Spike control switches (2) - NORM.
24. Landing/taxi lights switch - OFF.
25. Flight instrument reference select switch - PRI.
26. Compressor bleed valve control switches (2) - AUTO.
27. Disposable dispense switch - OFF.
28. Landing gear handle - DN.
29. Utility hydraulic system isolation switch - NORM.
30. Control system switch - NORM.
31. Arresting hook handle - In.
32. Radar altimeter control knob - Off.
33. Instrument system coupler mode select knob - As required.
34. ILS power switch - OFF. (Prior to T.O. 1F-111(E)A-501)
35. CMRS - Check: (After T.O. 1F-111(E)A-501)
    a. Function selector knob - OFF.
    b. Sensitivity control knob - AUTO.

36. Ram air doors override switch - AUTO.

37. Heat exchangers override switch - AUTO.

38. UHF radio - OFF.

39. Select stores jettison master switch - OFF.

40. Landing gear emergency release handle - In.

41. 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.

42. TFR channel mode selector knobs (2) - OFF.

43. Nav radar function selector knob - OFF.

44. IFF master control knob - OFF.

45. TACAN - OFF.

46. Electrical control panel - Set.

    a. Generator switches (2) - OFF.

    b. Battery switch - OFF.

    c. External power switch - OFF.

    d. Emergency generator switch - AUTO.

    e. Emergency generator indicator/cutoff push-button - In. (Safetied)

47. HF mode selector knob - OFF. (Prior to T.O. 1F-111(E)A-501)

48. ILS power switch - OFF. (After T.O. 1F-111(E)A-501)

49. Air conditioning control panel - Set.

    a. Temperature control knob - As desired. (Mid-range with canopy open, above 70°F)

    b. Air source selector knob - BOTH.

    c. Mode selector switch - AUTO. (MANUAL with canopy open, above 70°F)

    d. Pressurization selector switch - NORMAL.

**Power On (AC)**

1. Battery switch - BATTERY.

   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 electrical 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.

**CAUTION**

The OVRD position should not be selected unless required by an emergency.

**Note**

If external power is not obtained when ON is selected, the external power source should be replaced or a battery start made.

4. UHF radio - BOTH and set.

5. Caution lamps - Check.

   a. Following lamps will be lighted:

      PRI ATT/HDG

      PITCH, ROLL, YAW DAMPER

      ANTI-SKID

$\alpha/\beta$ PROBE HEAT

L & R PRI HYD

L & R UTIL HYD

L & R ENG OVERSPEED

L & R GEN

b. Following lamps may be lighted:

CANOPY

PITCH, ROLL, GAIN CHANGER;

PITCH, ROLL, YAW CHANNEL, SIS (Must reset)

AUX ATT (Until initial erection)

L & R FUEL PRESS

LOW EQUIP PRESS (High field elevations)

If the caution lamps in a. are not lighted or lamps other than in b. are lighted, a malfunction may be 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.

c. The FWD EQUIP HOT lamp will light within 3 minutes after power on if cooling is not available.

6. Seat and headrest - Adjusted.

7. 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.

**CAUTION**

Extended operation of the electroluminescent strip lights in bright sunlight may permanently damage or significantly reduce the operating service life of the lights.

8. Malfunction and indicator lamps - Check.

- 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, check warning horn silence button operation.
- Malfunction and indicator lamps test button - Release.

9. AFRS synchronization indicator - Nulled.

10. Anti-skid switch - ANTI SKID for approximately 5 seconds then OFF.

Check that anti-skid caution lamp goes out and stays out until the switch is turned OFF. Illumination of this lamp while the switch is on indicates a malfunction.

11. Bleed indicator - Check. (Pneumatic start only)

Move both throttles to MIL and check bleed indicator reads NONE. Return throttles to cutoff and check indicator reads BOTH. Force required to move throttles should not exceed 16 pounds with friction lever in DECR position.

12. 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.

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, both engine fire warning lamps, and the L and R bleed duct fail caution lamps are lighted.

14. Landing gear position indicator lamps - Checked.

15. Oil quantity indicators - Check, 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.

16. 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).

17. 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 (±1,000 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 the 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 - 2,000 (±400) pounds.
- Select tank - 2,000 (±100) pounds.
- Total fuel - 2,000 (±1,250) 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 10 to 15 seconds. R
- Fuel quantity indicator test button - Release.
- Fuel distribution caution lamp - Goes out within 15 seconds. R

18. Engine feed selector knob - AUTO.

    Select AUTO when the forward tank pointer is approximately 2,000 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 taxi.

- 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.

19. Fuel transfer knob - As required.

    Fuel pump low pressure indicator lamps 7 through 10 should light and go out unless the tank is empty and then the lamp should remain lighted.

20. FWD EQUIP HOT lamp - Checked.

## ENGINE STARTING MALFUNCTIONS

### 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 visible fuel or vapors are found the engine must be cleared using the "Engine Clearing" procedure. If no fuel is visible a second start may be attempted.

### 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.

### STARTING ENGINES (AC)

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.

**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. Engine clearing will reduce the temperature.

**Note**

When using the MA-1A starter cart, left engine starting capability is marginal.

1. Check ground observer - Ready for engine start; fire guard posted and engine areas clear.

2. External air - Applied. (GO) (If required)

3. Engine ground start switch - PNEU or CARTRIDGE. (As applicable)

**Note**

During an engine start where external power is used, the ground start switch will be electrically held in the CARTRIDGE or PNEU position until manually moved to OFF. If battery power is used for the start, the ground start switch will automatically move to OFF when the applicable generator is brought on the line. Failure of the switch to remain in the CARTRIDGE or PNEU position throughout the start sequence may result in hung rpm due to the hydraulic demands of the emergency generator. Dim illumination or flashing of master caution panel lamps will normally occur when the switch does not remain in CARTRIDGE or PNEU during the engine start sequence. The switch must be repositioned to CARTRIDGE or PNEU to preclude rpm hang during engine start.

4. 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.

   c. Applicable hydraulic caution lamps - Out.

**CAUTION**

If hydraulic pressure reads zero, the engine should be shut down immediately to prevent further damage to the hydraulic pumps.

5. 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.

6. Engine instruments - Check.

   a. Fuel flow - 1,100 pph maximum.

   b. Turbine inlet temperature indicator - 710 degrees C maximum.

   c. Idle rpm - 57 to 69 percent.

   d. Hydraulic pressure indicators - 2,950-3,250 psi, applicable caution lamps out.

   e. Idle oil pressure - 35 to 50 psi.

   f. Nozzle position - Open.

7. Engine ground start switch - OFF. (If required)

**Note**

- Cooling air will not be available if the switch is in any position other than OFF.
- During second engine start, check that the switch moves to OFF prior to reaching 50 percent engine rpm.

8. Engine overspeed caution lamp - Out.

9. Generator switch - ON. Check caution lamp out.

**Note**

If the generator caution lamp remains lighted, place the generator switch to OFF-RSET, then to ON.

10. 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.

11. Nacelle vent ejector system - Check. (GO)

After engine start, the ground crew will observe the nacelle vent/fire access doors and, after T.O. 1F-111(E)A-556, the IDG supplementary air-oil cooler blow-in door, to insure that the doors are open, indicating nacelle vent ejector system operation. The ground observer will feel for air being ejected by the nacelle vent ejector system by placing a hand behind the ram air/ejector air exit. If no airflow is felt, the nacelle vent ejector system or the landing gear ground safety (squat) switch has malfunctioned.

**WARNING**

Do not move any flight controls until nacelle vent ejector system check is completed and the ground observer is clear of the horizontal stabilizer.

12. Nose gear ground safety pin - Removed and displayed. (GO)

13. Speed brake ground safety lock - Removed. (GO)

14. External power switch - OFF.

15. External power and air - Disconnected, if crossbleed is used. (GO)

**Note**

If the GO is unable to remove the starter hose, temporarily place the air source selector knob to the OFF position to relieve trapped air pressure in the hose.

16. Power on checks - Complete. (If battery start is accomplished)

17. Air refueling receptacle and light - Check. (If required) (GO)

18. Remaining engine - Start. (Repeat steps 1 thru 11)

**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.

19. External air - Disconnected. (If required)

20. Power flow indicator - NORM.

21. 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 TR units by noting that the angle-of-attack indexer lamps light, and AMI and AVVI tapes move when the instrument test button is depressed. 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.

22. Oil quantity - Check minimum of 8 quarts.

If quantity is below 8 quarts, a visual check of the oil quantity in the tanks is recommended.

## AFTER ENGINE START (AC)

**Note**

- During equipment check/prolonged engine operation at idle power, the engine oil will be scavenged every 10 minutes by operating the engines at 75-79 percent rpm for 15 seconds.
- With a tailwind or single air source selected, sustained static idle may not provide sufficient IDG cooling. In this event, recommend the aircraft be turned or operate at 70 percent rpm, as practicable.
- If forward equipment hot caution lamp lights, refer to Caution Lamp Analysis, Section III.

1. Radar altimeter - Set 100 feet.
2. TTWS power/test switch - ON.

3. Radar/IR indicator panel power/range selector switch - TEST.

R 4. L and R TFR channels - STBY.

5. Nav radar function selector knob - STBY.

6. IFF - STBY.

7. TACAN - T/R and set.

8. ILS power switch - power and set.

9. HF radio - Set. (If required) (Prior to T.O. 1F-111(E)A-501)

**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.

10. Wing sweep handle - Set for takeoff.

11. Wing sweep handle lockout controls - ON. (If applicable)

12. Ground crew check flight controls - Clear. (GO)

13. 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 stick laterally to the detent to check 4 degrees horizontal stabilizer asymmetry then to full lateral stick position to check 16 degrees horizontal stabilizer asymmetry. Spoilers should be fully extended at the detent.
- With the control stick full aft, check 24 degrees TEU, then check 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, check 10 degrees TED, 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.
- Move rudder 10 degrees and then slowly (minimum of 3 seconds) return rudder to neutral. Remove feet from pedals and advise the GO, who will note neutral position and observe that the rudder does not oscillate. Repeat in opposite direction. GO will assure that neutral position agrees within one inch. If oscillations are observed or the rudder does not return to center within one inch, the aircraft will not be flown.

**WARNING**

Malfunction of a hydraulic servo actuator control valve may cause an uncommanded full control surface deflection. Aircraft response cannot be predicted in this situation and control inputs may not correct the uncommanded deflection.

e. Series trim - Checked.

Ensure auxiliary pitch trim switch is in STICK; actuate stick trim button to nose up, check that control surface moves to 10° ($\pm$2°) TEU;

actuate stick trim button to nose down; check that control surface moves to 4° (±2°) TED. Control stick should not move.

f. Damper switches (3) - DAMPER.

g. Damper reset button - Momentarily depressed. (If necessary)

Check that the pitch, roll, and yaw damper caution lamps go out.

h. Flight controls - Check.

Move stick and rudder to ensure damper and channel lamps do not light.

**CAUTION**

With dampers on, lateral stick force in excess of that required for full flight control deflection could damage the flight control mechanical linkage. Do not attempt to force the stick past the force detent.

i. Trim - Checked.

Set T.O. trim with auxiliary pitch trim switch in STICK, actuate stick trim button to NOSE DOWN, NOSE UP, RWD, LWD, and rudder trim left and right; check control surfaces give proper response to trim inputs. With nose up pitch trim set, move auxiliary pitch trim switch to OFF, check the stick returns to neutral and the stabilizers drive down. Actuate stick trim button to NOSE DOWN and NOSE UP, and check for no movement of the stick or stabilizers. Move auxiliary pitch trim switch to NOSE DOWN, then NOSE UP; check control surfaces travel in response to switch positions and that control stick does not move. Return the auxiliary pitch trim switch to STICK.

14. 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.

15. 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.

16. 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.

17. 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 a minimum of 8 degrees right of center, then 8 degrees left of center.
  - Left horizontal stabilizer drives down.
  - Right horizontal stabilizer drives down causing a LWD indication on the surface position indicator.
  - 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.

**Note**

- In high wind conditions the horizontal stabilizers may not drive to the normal position during the surface motion check due to flight control gains being driven to a low value.
- The wing spoilers may momentarily extend then retract during surface motion or surface motion and lights test. If the spoilers remain up, a malfunction is indicated.

18. Flap/slat handle - Extend slats.
19. Surface motion and lights test - Complete. (GO optional)
    - Stability augmentation test switch - SURFACE MOTION & LIGHTS and hold until next step is completed.
    - Flight control master test button - Depress and hold for the following checks:
        - Rudder drives right, then centers.
        - Left horizontal stabilizer drives down.
        - Right horizontal stabilizer drives down causing a LWD indication on the surface position indicator.
        - Pitch, roll, and yaw damper, channel, and pitch and roll gain changer caution lamps light (8).
    - Master test button - Release.
        - Rudder drives left then returns to neutral.
20. Damper reset button - Depress momentarily.
21. Auto TF switch - OFF.
22. All flight controls caution lamps - Out.
23. SIS and AYC - Check.
    a. Takeoff trim - Set.
    b. Left and right alpha probe slots - Full down and hold; beta probe centered.

       Angle-of-attack indicator indicates 25 degree angle-of-attack.
    c. Damper servo button - Depress and hold.
    d. Master test button - Depress and hold.
        (1) Control surfaces - Check trailing edge down. (Checks alpha limiter).

        After initial nose down input, both stabilizers may drift together in pitch.

        (2) All flight control caution lamps - Check out. (Pitch and roll gain changer lamps may light. If these lamps light, have the (GO) re-center the beta probe; the lamps should go out).

**Note**

- If the SIS caution lamp lights, instruct the (GO) to reset the probes and repeat the check.
- If the beta probe is not centered within $\pm 6$ degrees, the pitch and roll gain changer lamps will light.

    e. Control stick - Move laterally to left detent and hold. Check control surfaces indicate LWD and left rudder. (Checks beta reducer)
    f. Probes, buttons, and stick - Release.

**Note**

Disregard horizontal stabilizer deflection in the following checks.

    g. Flaps - Set to 18 degrees or more.

       Flaps must be extended 18 degrees or more to ensure AYC operation.
    h. Takeoff trim - Set.

i. Left and right alpha probe slots - Full up and hold; beta probe slots - Full left and hold. (GO) Angle-of-Attack indicator indicates less than zero degree angle-of-attack.

j. Damper servo button - Depress and hold.

k. Master test button - Depress and hold.

(1) Both horizontal stabilizers may drift together in pitch.

No sudden movement is allowed. (Checks right alpha probe connected).

(2) All flight control caution lamps - Check out. (Pitch and roll gain changer lamps will light.

**Note**

The SIS lamp may light if the left and right alpha probes differ by more than six degrees. If this occurs, have the (GO) reset both alpha probes to approximately the same position and repeat the check. If alpha probes are within six degrees of each other, the SIS lamp should not light.

l. Beta probe - Rotate slowly from full left to full right, then center. Rudder kicks left slightly, then moves from left to right as the beta probe is rotated through neutral. Pitch and roll gain changer caution lamps go out when the rudder begins to displace. Rudder will follow beta probe to center. (Checks AYC)

m. All flight control caution lamps - Check out.

n. Probes and buttons - Release.

24. Stall warning system - Check. (GO)

- Pitot/probe heater switch - OFF/SEC.

**WARNING**

If pitot/probe heater switch has been in HEAT position, residual heat in probe may be sufficient to cause injury to ground personnel.

- Angle-of-attack probe slots - Full up. (Less than zero degrees angle-of-attack value)

- Malfunction and indicator lamps test button - Depress and hold.

- Warning horn silence button - Depress.

- Angle-of-attack probe slots - Rotate slowly to full down. (25 degree angle-of-attack value)

- With malfunction and indicator lamps test button depressed, stall warning lamp should flash throughout test. The audible warning and pedal shaker should activate as angle-of-attack passes 18 degrees at all wing sweeps.

**Note**

During ground checks with the flaps extended 15 degrees or more, depressing the malfunction and indicator lamp test button may cause the rudder to deflect due to an adverse yaw compensation input.

- Malfunction and indicator lamps test button - Release.

**Note**

When lamps test button is released, yaw channel caution lamp may remain lighted; reset to put lamp out.

25. Pitot heat, angle-of-attack and sideslip probes - Check. (GO) (If required)

26. 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.

27. 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.

28. Autopilot - Checked. (Optional)

- Pitch autopilot/damper switch - AUTOPILOT.

  Check that pitch inputs light the reference not engaged lamp with the pitch autopilot engaged. Engage a submode of the pitch autopilot, depress the autopilot release lever and check that the submode returns to off and the pitch autopilot/damper switch returns to DAMPER.

- Roll autopilot/damper switch - AUTOPILOT.

  Check that roll inputs light the reference not engaged lamp with the roll autopilot engaged. Engage a submode of the roll autopilot, depress the autopilot release lever and check that the submode returns to off and the roll autopilot/damper switch returns to DAMPER.

- Pitch and roll autopilot/damper switches - AUTOPILOT.
- Engage submodes - Check all submodes engaged.
- Autopilot release lever - Depressed/released.

  Check that pitch and roll autopilot/damper switches move to the DAMPER position.

29. Radar altimeter - Checked.

  a. Radar altimeter channel selector switch - AUTO.

  b. Depress and hold radar altimeter control knob for five seconds minimum.

     (1) LARA multiplexer caution lamp - On, approximately five seconds, then out.

     (2) RADAR ALT LOW warning lamp (on left instrument panel) and low altitude warning lamp (on LARA indicator) - Out.

     (3) Radar altimeter indicator - Checked.

     - Pointer positioned to approximately 9:00 o'clock position.
     - Pointer indicates zero altitude for next 1.5 seconds.
     - Pointer indicates 300 ($\pm$15) after the above is completed.

**Note**

If any of the above observations are incorrect, select single channel operation and repeat the self-test sequence. If single channel operation is selected, the LARA multiplexer lamp will remain on during both self-test and normal operation if the TFR is in TF, SIT, or GM. However, the other indications will be as described above.

30. TFR/FLIGHT CONTROL GROUND CHECK:

**Note**

- This check will be accomplished if the flight will include TFR operation.
- This check must be accomplished on the ground to obtain proper lamp indications.
- When switching channels or changing clearance plane settings, a momentary TF fail may occur.
- The radar altimeter override function displays may be present on the E-scope during ground checks.

a. Flight controls - Clear. (GO)

b. Flaps/slats - Retracted.

c. Antenna cage indicator lamp - Out.

d. Radar altimeter channel selector - CHAN 1 or 2.

LARA multiplexer caution lamp will be lighted when either CHAN 1 or 2 is selected and TFR is in TF, SIT, or GM.

e. INS mode selector knob - Check in ALIGN. R

f. Altitude/test selector knob - Test 2.

g. TFR/FLIGHT CONTROL tie-in check - Complete.

**WARNING**

Do not transmit with the TFR if personnel or equipment are within the dangerous radar emission area. See figure 2-2.

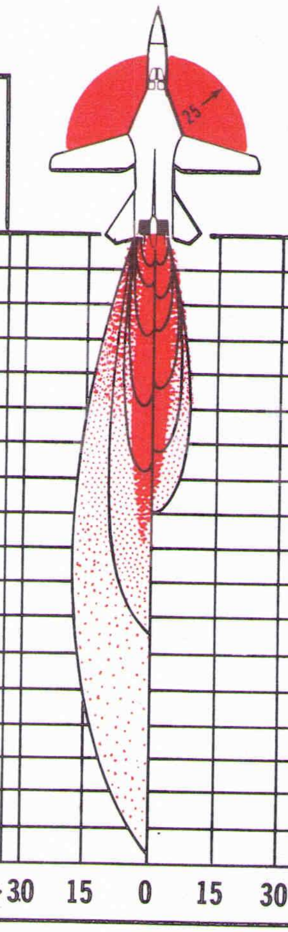
(1) Terrain clearance knob - 400.

(2) Ride control knob - Soft.

# DANGER AREAS

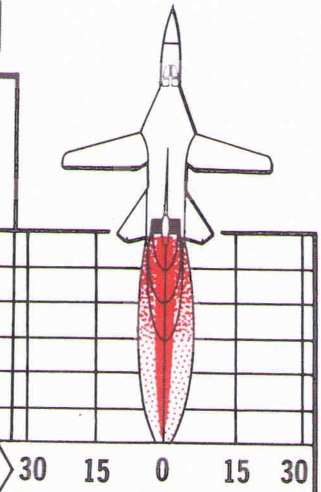
ENGINES: TF30-P-3
DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

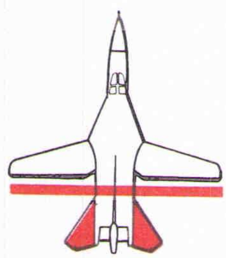
| DISTANCE FROM JET NOZZLE (FEET) | MAX AB THRUST TEMPERATURE (FAHRENHEIT) | MAX AB THRUST VELOCITY (MPH) | MILITARY THRUST TEMPERATURE (FAHRENHEIT) | MILITARY THRUST VELOCITY (MPH) |
|---|---|---|---|---|
| 20 | 2000° | 1750 | 500° | 1100 |
| 40 | | | 300° | 455 |
| 60 | 600° | 410 | 200° | 210 |
| 80 | | | | |
| 100 | 400° | 200 | 150° | 105 |
| 120 | | | | |
| 140 | 300° | 125 | 125° | 70 |
| 160 | | | 110° | 50 |
| 180 | | | | |
| 200 | | | | |
| 220 | 200° | 55 | | |
| 240 | | | | |
| 260 | | | | |
| 280 | | | | |
| 300 | | | | |
| 320 | | | | |
| 340 | | | | |
| 360 | 150° | 30 | | |



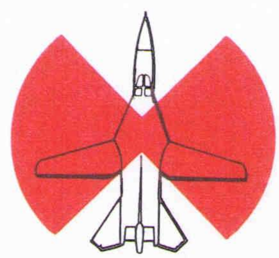

IDLE THRUST

| DISTANCE FROM JET NOZZLE (FEET) | TEMPERATURE (FAHRENHEIT) | VELOCITY (MPH) |
|---|---|---|
| 30 | 150° | 70 |
| 40 | | |
| 50 | 125° | 40 |
| 60 | | |
| 70 | | |
| 80 | | |



ROTATING PLANES OF ENGINE TURBINES AND DANGER AREAS FOR FLIGHT CONTROLS



TIRE AVOIDANCE

## WARNING

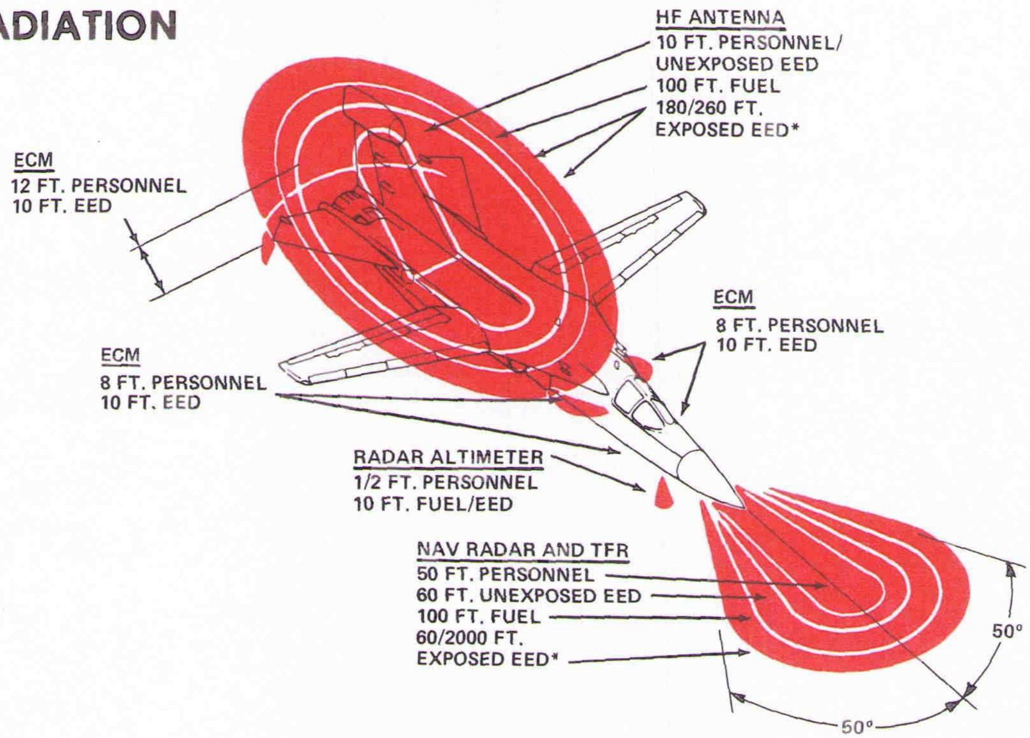
- AT HIGH THRUST SETTINGS, THE DANGER AREA AROUND THE INTAKE DUCTS MAY EXTEND AS FAR AS FOUR FEET AFT OF THE TRANSLATING COWL.
- EAR PROTECTION MUST BE WORN WHEN IN THE PROXIMITY OF AN AIRCRAFT WITH AN OPERATING ENGINE.
- DO NOT ACTUATE FLIGHT CONTROLS WITH PERSONNEL IN CLOSE PROXIMITY OF THE HORIZONTAL STABILIZERS.
- PRIOR TO ANY MAINTENANCE OR SAFETY PIN INSTALLATION FORWARD OF THE MAIN GEAR WELL, SHUT DOWN THE ENGINE ADJACENT TO THE WORK AREA.

IF LANDINGS ARE MADE WHICH FOR SOME REASON REQUIRE MAXIMUM BRAKING TO STOP THE AIRCRAFT, AVOID AREA EXTENDING TO AT LEAST 300 FEET IN A 45 DEGREE CONE AROUND AXLE ON BOTH SIDES OF WHEEL FOR AT LEAST 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.

B-1EFA-69-1

Figure 2-2. (Sheet 1)

# DANGER AREAS — CONT
## RADIATION




SAFE DISTANCE FROM ANTENNAS

| TRANSMITTER | PERSONNEL HEALTH HAZARD | EXPOSED EED* 20MM AMMO OR CBW TYPE INITIATORS | EXPOSED EED* ALL OTHER EED | UNEXPOSED EED | FUEL |
|---|---|---|---|---|---|
| NAV RADAR | 48 ft | 1780 ft | 50 ft | 50 ft | 100 ft |
| TFR | 12 ft | 320 ft | 15 ft | 10 ft | 16 ft |
| HF RADIO | 10 ft | 260 ft | 180 ft | 10 ft | 100 ft |
| ECM | 8 ft fwd/12 ft aft | 10 ft | 10 ft | 10 ft | 10 ft |
| LARA (RADAR ALTIMETER) | 1/2 ft | 10 ft | 10 ft | 10 ft | 10 ft |
| BOTH NAV RADAR & TFR | 50 ft | 2000 ft | 60 ft | 60 ft | 100 ft |

*EXPOSED EED MEANS NO METALLIC COVERING. FOR EED OTHER THAN 20MM AMMO OR INITIATORS USING CARBON BRIDGE WIRES (CBW) USE ALL OTHER EED DISTANCES.

JSS TRANSMITTER DANGER AREAS (FEET)

| BAND | PERSONNEL ONE TRANS. OPERATING | EED-ONE TRANS. OPERATING UNEXPOSED | EED-ONE TRANS. OPERATING EXPOSED** |
|---|---|---|---|
| 1 | 7 | 85 | 300 |
| 2 | 14 | 85 | 135 |
| 4 | 16 | 35 | 100 |
| 5/6 | 16 | 25 | 55 |
| 7 | 24 | 10 | 25 |
| 8 | 66 | 10 | 25 |
| 9 | 66 | 10 | 25 |

** EXPOSED EED MEANS NO METALLIC COVERING.

**WARNING**

THESE DISTANCES ARE THE RADII OF A SPHERE MEASURED FROM THE WEAPONS BAY RADOME TO THE OBSERVER.

**NOTE**

- THESE ARE WORST CASE MINIMUM SAFE DISTANCES MEASURED FROM THE MAIN BEAM SOURCE ANTENNA.
- THERE IS NO INFLIGHT FUEL HAZARD.
- DISTANCES ARE GIVEN FOR ONE TRANSMITTER OPERATING ON A GIVEN BAND. IF MORE THAN ONE TRANSMITTER IS OPERATING, TAKE THE SQUARE ROOT OF THE SUM OF THE SQUARES AT THE GIVEN DISTANCES BY BAND. EXAMPLE: IF BAND 4 AND BAND 7 ARE OPERATING.

$$\sqrt{(16)^2 + (24)^2} = 29 \text{ FEET}$$

WORST CASE FOR MULTIPLE TRANSMITTER OPERATION IS:

| | |
|---|---|
| PERSONNEL | – 210 FT. |
| EED (UNEXPOSED) | – 160 FT. |
| EED (EXPOSED) | – 510 FT. |

C-1EFA-69-2

★ **Figure 2-2.(Sheet 2)**

(3) TFR channel mode selector knobs - L TF, R SIT.

The following lamps will be lighted:

FLT VECTOR

INS mode selector must be in ALIGN.

VELOCITY -

Altitude/test selector knob must be in Test 2.

TF FLY-UP OFF -

The TF fly-up caution lamp is lighted because the fly-up circuit is not armed.

TF failure warning -

The TF failure warning lamp is lighted because the aircraft is below 68 percent of set clearance.

REF NOT ENGAGED -

The reference not engaged caution lamp is lighted because the TF system is in the manual TF configuration.

LARA MUX -

The lamp is lighted because single channel operation is selected.

RADAR ALT LOW -

TFR channel fail caution -

The channel fail caution lamp of the channel in TF should be lighted and the lamp of the channel in SIT should be out.

(4) FLT VECTOR - Check.

(a) Flight control master test button - Depress.

(b) CADC test switch - LOW.

(c) CADS caution lamp - On (Lamp on due to CADC test).

(d) FLT VECTOR caution lamp - Out.

(e) Flight control master test button and CADC test switch - Release.

(f) FLT VECTOR caution lamp - On.

(g) CADS caution lamp - Out.

(5) ISC pitch steering mode switch - TF.

During ground checks, if the TF steering bar is not selected, the TF failure warning lamp may flash at one cycle per second.

(6) Control system switch - TO & LAND

(7) Takeoff trim-Set

Removes a possible climb command indication due to stick position.

(8) Left alpha probe - Set 2 to 6 degrees.

(9) Radar altimeter control knob - Depress and hold thru step (20).

(a) The following lamps will go out:

RADAR ALT LOW -

The radar altitude low lamp going out verifies the operational status of the low altitude monitor (LAM) system. If the lamp remains lighted, the LAM has detected a fault in the monitor checks or the LAM is not operational.

TF failure warning -

TF FLY-UP OFF -

This indicates the fly-up system is armed. It will remain armed until both channels are returned to STBY. The horizontal stabilizers do not move.

**Note**

If the TF FLY-UP OFF caution lamp illuminates during any of the following steps, a malfunction is indicated.

TFR channel fail caution

(b) Drift lamp - Check

Inst Test button - Depress.

- TFR channel fail caution lamps - Out.
- TF failure warning lamp - Out.
- TF DRIFT lamp - On.

(c) Pitch steering bar and aural command - Positive Climb.

**Note**

- If a positive climb is not indicated, the TFR Video check procedure (this section) must be accomplished prior to takeoff.
- R If forward video (video inside one mile and
R above the ZCL) is providing a positive
R climb indication, the left alpha probe does
R not need to be set 13 to 15 degrees. Pro-
R ceed to step (11).

(10) Left alpha probe - Set 13 to 15 degrees.

(a) Pitch steering bar and aural command - Indicate positive climb.

If a positive climb command is not indicated the remainder of this check is not valid.

(11) Trim nose down for 1 second - Horizontal stabilizers will drive down while trim button is held and the control stick will move slightly forward.

(12) Auto TF switch - AUTO TF.

(a) Horizontal stabilizers - Drive rapidly TEU.

When this climb command occurs, the pitch damper will drive the stabilizers rapidly to approximately 10 degrees TEU.

(b) Control Stick - Drives slightly aft.

(c) REF NOT ENGAGED caution lamp - Out.

(13) Control system switch - NORM.

(a) Horizontal stabilizers - Drive slowly TEU.

The series trim will drive slowly TEU at 1.4 degrees per second until maximum deflection is reached.

(14) Autopilot release lever - Depress then release.

To ensure the normal TFR climb command R
can be removed, check the autopilot release lever. While the lever is depressed, the horizontal stabilizers will drive rapidly down and the REF NOT ENGAGED lamp will light. When released, the horizontal stabilizers will drive rapidly TEU and the REF NOT ENGAGED lamp will go out.

(15) Auto TF switch - OFF.

(a) Horizontal stabilizers - Drive rapidly down.

After the initial rapid damper action, the series trim may command further TED drift.

(b) REF NOT ENGAGED caution lamp - Lighted.

(16) Takeoff trim - Set.

Set takeoff trim to confirm series trim operation in the following steps. Perform the next step immediately after releasing the takeoff trim button.

(17) Radar altimeter bypass switch - BYPASS and hold.

This is an operational check of the LAM bypass circuit. The following TF failure indications are due to loss of the LARA data good signal to the TFR computer.

(a) Horizontal stabilizers - Drive rapidly TEU then slowly TEU.

**Note**

The pitch damper will drive the stabilizers up very rapidly to approximately 14 degrees TEU. Immediately following this initial movement, the series trim will continue to drive TEU at a more gradual rate (1.4 degrees per second) until maximum deflection is reached. The aircrew must check for the initial rapid deflection from the pitch damper followed by the slower deflection from the series trim. This will ensure rapid flight control response to a TFR fly-up command.

(b) Pitch steering bar and aural command - Maximum climb.

**Note**

The pitch steering bar and aural command will already be indicating a climb command. However, the induced fail condition of this test will provide a maximum climb indication for the pitch steering bar, and aural command.

(c) The following lamps will be lighted:

TF failure warning

RADAR ALT LOW

TFR channel fail caution.

(18) Radar Altimeter bypass switch - Release to NORMAL.

(a) Horizontal stabilizers - Drive rapidly down.

After initial stabilizer movement the stabilizer may drift TED due to series trim.

(b) The following lamps will go out:

TF failure warning

RADAR ALT LOW

TFR channel fail caution.

(19) Terrain clearance knob - Set 500 feet.

This check verifies the TF computer and LAM detection of the 68 percent set clearance fail fly-up command.

(a) Horizontal stabilizers - drive rapidly TEU.

(b) The following lamps will be lighted.

TF failure warning

RADAR ALT LOW

TFR channel fail caution.

(20) Radar altimeter control knob - Release.

(21) Autopilot release lever - Depress, then release.

To ensure the fail fly-up command can be removed, check the autopilot release lever. While the lever is depressed, the horizontal stabilizers will drive rapidly down. When released, the horizontal stabilizers will drive rapidly TEU.

(22) E-scope - Checked.

Rotate all TFR scope tuning knobs (4) fully CCW, then adjust the contrast control until a thin vertical line along the right side of the E-scope is discernible. This adjustment assures that the scope will display the video returns received. For best video presentations, minimum memory should be used. Rotate the video knob clockwise until the self-test pulse is visible (adjust for optimum video display when at low altitude).

(23) Self-test pulse - Checked.

(24) 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 - Check.

Rotate thru each position. Check the zero command line position for proper movement and a smooth curve for all clearance settings.

(25) SIT and GM modes - Check.

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. With channel R
in GM, select 5, 10, 15, and E, checking R
range and cursor as above. R

(26) TFR channel mode selector knobs - STBY.

Confirm the horizontal stabilizers move TED. If the horizontal stabilizers do not move TED the TFR commands to the flight controls are still present.

h. Repeat TFR/FLIGHT CONTROL tie-in check with TFR and radar altimeter channels reversed. Video should be approximately the same on both channels.

i. Radar altimeter channel selector - AUTO; LARA multiplexer caution lamp out.

j. Altitude/test selector knob - NORM.

31. Ground check panel door - Closed.

32. Nav radar - Tune. (If not accomplished by EWO)

**WARNING**

Do not place nav radar to XMIT in parking area unless the area ahead of the aircraft has been checked and cleared of all personnel.

- Antenna cage indicator lamp - Out.
- Antenna tilt - +30°.
- Function selector knob - ON.
- Mode selector knob - GND AUTO or GND VEL.
- Antenna tilt - ±2°.
- Mode selector knob - GND MAN.
- Brightness knob - CW until sweep is just visible.
- Video gain knob - Midrange.
- IF gain knob - CW until snow appears, then CCW until snow disappears.
- Bezel/range marks - Check.
- Range/azimuth cursors - Check.
- Function selector knob - XMIT. To check for returns.
- Function selector knob - STBY.

33. TTWS - Check. (Optional) R

34. CMRS - Check. (Optional) (After T.O. 1F-111(E)A-501) R

35. Takeoff trim - Set. (GO)

After the takeoff trim lamp lights, check the control surface position indicator to ensure that the horizontal stabilizer and rudder are set for takeoff and have the ground observer verify.

**Note**

Any time the flight control master test button is used for a system check, the series trim is unlocked allowing drift of the horizontal stabilizer. Takeoff trim must be reset any time the flight control master test button has been depressed.

36. Anti-g suit operation - Check. (If applicable)

37. Attitude indicator - Check and set.

38. SCAI - Caged and set.

## PREFLIGHT CHECK (EWO)

### BEFORE EXTERIOR INSPECTION (EWO)

1. Ejection handle safety pins (2) - Installed.
2. Canopy center beam safety pins (3) - Installed.
3. Survival equipment compartment covers (2) - 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.
6. Thermal curtains - Stowed. (If applicable)

### INTERIOR INSPECTION (EWO)

#### Power Off (EWO)

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.

**WARNING**

Failure to ensure lap and shoulder harnesses are snug may result in delayed activation of critical switches/handles during out-of-control maneuvers. It may also result in spinal injury during ejection.

- 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. (As applicable)

2. Seat pan - Check for freedom of movement over full range of travel.

**Note**

- If the seat pan does not adjust freely fore and aft, it may not be possible to assume an optimum position for ground impact following an ejection.
- Storage of items under the right seat is prohibited. Contact with the fire warning relay boxes may cause false fire indications in flight.

3. Oxygen system and personal equipment - Connected and checked.

**CAUTION**

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

- 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 - 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.

- Oxygen control knob - NORM.

  Inhale and check movement of diluter valve through screen in the top of the regulator.

4. CMRS - Check. (Prior to T.O. 1F-111(E)A-501)
   a. Function selector knob - OFF.
   b. Sensitivity control knob - AUTO.
5. HF mode selector knob - OFF. (After T.O. 1F-111(E)A-501)
6. CMDS - Check.
   a. Arming switch - SAFE.
   b. Mode selector knobs (3) - OFF.
7. Antenna select switches (3) - AUTO.
8. Interphone panel - Set.
9. Radiate enable overload indicator switch - OFF.
10. Liquid pump switches (2) - NORM.
11. SPS control panel - Set.
    a. Mode control switches (4) - OFF.
    b. Spill over switches (4) - ON.

12. Receiver control panel switches - Set.
    a. Mode switches (6) - SCAN.
    b. Attenuation switches (4) - 0.
13. Jammer control panel switches - Set.
    a. Master radiate switch - OFF.
    b. Mode switches (10) - AUTO.
    c. Jammer radiate switches (10) - OFF.
14. JSS modes panel switches - Set.
    a. JSS power switch - OFF.
    b. Inhibit switch - NORM.
    c. Test selector switch - SYS BIT.
    d. Computer switch - NORM.
    e. Sector switch - DIR.
    f. UC COV disp switch - OFF.
15. Heading source selector switch - PRI.
16. BDHI mode selector switch - JSS.
17. DDI mode selector switch - DATA DISP.
18. INS - Set.
    a. Mode selector knob - OFF.
    b. Platform alignment knob - OFF.
    c. Magnetic variation counter - Set.
    d. Fixpoint elevation counter - Set.

R 19. TTWS - Set.
    a. Power/test switch - OFF.
    b. Altitude select switch - AUTO.
    c. Mode select knob - DISP.
20. Radar/IR indicator power/range select knob - OFF.
21. Clock - Set.
22. Report ready for electrical power.

**Power On (EWO)**

1. INS.
    a. Mode selector knob - ALIGN.
    b. Platform alignment knob - OFF.
    c. Man fix pushbutton switch - Depressed.
    d. Present position - Set. (As required)
    e. Mode selector knob - HEAT.

**Note**

When ambient temperature exceeds 110 degrees F, do not leave inertial nav mode selector knob in HEAT for more than 5 minutes without external air conditioning.

    f. Platform alignment knob - NORMAL.
    g. Heat indicator lamp - Lighted.
    h. Altitude/test selector knob - NORM.
2. Jammer sequencer check - Complete.
3. Report ready for engine start.

## AFTER ENGINE START (EWO)

**Note**

- At OAT above 70°F, with canopy open or closed, limit ground operation of ALQ–137 (SPS) to 5 minutes or less. Remaining equipment is limited to 30 minutes with canopy closed and is unrestricted with canopy open.
- At OAT below 70°F equipment operation is unrestricted.
- 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. INS mode selector knob - ALIGN.
2. Inertial nav destination storage (3) - Set. (If applicable)
3. CMRS - Check. (Optional) (Prior to T.O. 1F-111(E)A-501) R

4. HF radio - Set. (If required) (After T.O. 1F-111(E)A-501)

**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.

R 5. SPS Mode switches (4) - REC.

**WARNING**

Ensure weapons bay has been ventilated for 5 minutes prior to application of JSS power, due to possible combustible fluid leaks.

6. JSS power switch - ON. (After 5 minutes ECS ventilation of weapons bay.)

   a. Enter access code.

   b. Clear alerts from edit line.

   c. Receiver calibrate mode - AUTO.

7. Verify mission lists.

8. JSS short pulse width pushbutton lamps (5) - OUT.

9. Jammer configuration - Check and record on jammer status panel.

10. JSS initialization - Complete. (Refer to T.O. 1F-111(E)A-1-2)

    a. Attitude limit lamp - Out.

    b. DDG BIT - Accomplish.

    c. Verify all major and minor DDI displays and tableaus.

11. JSS BIT checks - Complete.

R 12. SPS BITS - Complete. (Refer to T.O. 1F-111(E)A-1-2)

**WARNING**

Prior to selecting ON or TEST, advise all ground personnel to remain a minimum of 4 feet from the aircraft. The TEST position is the same as the ON position until the BIT START button is depressed. Initiate the self test immediately after selecting TEST.

13. CMDS - Check. (Refer to T.O. 1F-111(E)A-1-2) R

14. CMRS - Check. (Prior to T.O. 1F-111(E)A-501) R
(Refer to T.O. 1F-111(E)A-1-2)

15. Anti-g suit - Checked. (If applicable)

16. TTWS - Check. (If not accomplished by AC) (Re- R
fer to T.O. 1F-111(E)A-1-2)

17. Nav radar - Tune. (If not accomplished by AC)

**WARNING**

Do not place nav radar to XMIT in parking area unless the area ahead of the aircraft has been checked and cleared of all personnel.

- Antenna cage indicator lamp - Out.
- Antenna tilt - +30°.
- Function selector knob - ON.
- Mode selector knob - GND AUTO or GND VEL.
- Antenna tilt - ±2°.
- Mode selector knob - GND MAN.
- Brightness knob - CW until sweep is just visible.
- Video gain knob - Midrange.
- If gain knob - CW until snow appears, then CCW until snow disappears.
- Bezel/range marks - Check.
- Range/azimuth cursors - Check.
- Function selector knob - Xmit (Check for returns)
- Function selector knob - STBY.

## BEFORE TAXIING (BOTH)

1. TACAN/ILS/IFF - Checked.

   IFF check:

   - Master control knob - LOW.
   - Rad test/monitor switch - OUT.
   - Mode select/test switch - Hold to TEST momentarily and check that test lamp lights. Repeat for each mode.

   **Note**

   When any of the four mode select/test switches are placed to ON or TEST and the rad test/monitor switch is in MON, the test lamp may light, either continuously or periodically, due to external interrogations. This is also a valid check for the mode selected.

   - Master control knob - STBY.

2. INS:

   a. Verify aligned.

   b. Inertial nav mode selector knob - GREAT CIRCLE or above.

3. Rain removal system/bleed air shutoff system - Check.

   - Air source selector knob - OFF.
   - Rain removal switch - RAIN REMOVE.
   - Power - Set 75-80% RPM.

   Check for rain removal airflow.

   **WARNING**

   Airflow indicates failure of a bleed air shutoff valve. Do not fly the aircraft.

   - Air source selector knob - BOTH.

   Air should flow through the rain removal system.

- Rain removal switch - OFF.

**CAUTION**

Failure to confirm that rain removal airflow has ceased may result in windscreen damage.

4. Wing sweep, slats, and flaps - Set for taxi or takeoff.

   When required to taxi with slats up or wings aft of 26 degrees:

   a. Slats - Retracted.

   b. Wing sweep - As required.

**CAUTION**

- Do not exceed tanks/stores wing sweep limitations.
- Coordinate with GO prior to sweeping wings to insure area is clear of obstructions.

   c. Nose strut extension - Checked. (GO)

**CAUTION**

At light gross weights or with external stores, sweeping the wings full aft may establish an aft center of gravity condition resulting in full nose strut extension and free castering of the nosewheel.

5. Altitude calibration - Complete.

**Note**

If erratic operation of the LARA (in auto mode) is being experienced, select the best single channel (CHAN 1 or CHAN 2) for altitude calibration.

6. Heading crosscheck - Complete.

   Primary and AFRS heading indications should agree within 4 degrees. The standby magnetic compass should agree within 7 degrees.

**Note**

Local magnetic interference may influence this check.

7. Altimeters - Set.

8. EPR's - Set.

9. Hydraulic pressure - Checked.

   Check for 2,950 to 3,250 psi indication.

10. Ground communications cord - Disconnect. (GO)

11. Chocks - Removed. (GO)

12. Auxiliary brake handle - In.

13. Nosewheel steering - Engaged.

## TAXIING (BOTH)

**Note**

If turn radius is exceeded, range switch will automatically disengage controlled steering from rudder pedals and NWS/AR lamp will go out. Turning radius is presented in figure 2-3.

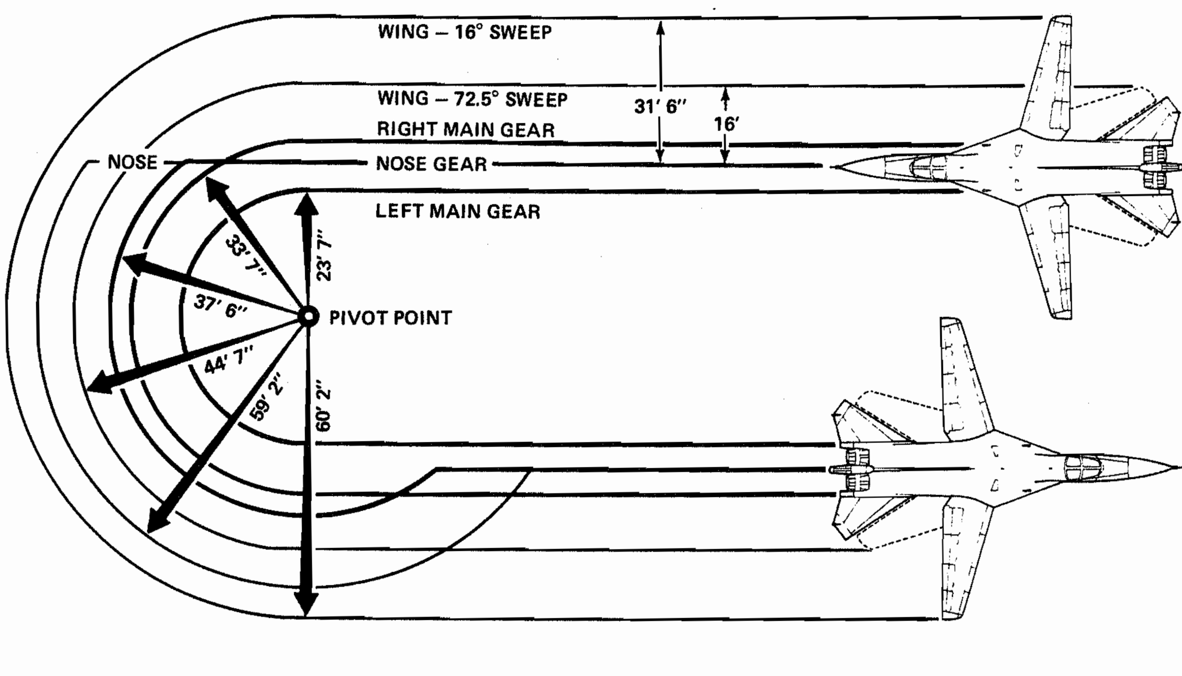
1. Brakes - Checked.

   Depress brake pedals and check for proper braking.

2. Flight instruments - Checked.

   Check the flight instruments for proper operation during taxi.

# TURNING RADIUS




★ Figure 2-3.

## BEFORE TAKEOFF (COMMAND RESPONSE)

1. Crew module safety pins (5) - Removed.
2. Rudder authority switch - AUTO.
3. Wing sweep, flaps, and slats - Set for takeoff.

   Check the wing sweep flap/slat position indicator for selected wing, flap, and slat settings.
4. Fixed stores lockout control - ON.
5. Ground roll spoiler switch - BRAKE.
6. Speed brake switch - IN.
7. Translating cowl switches - AUTO.
8. Translating cowl indicators - Checked, OPEN.
9. Control system switch - NORM.
10. Takeoff trim - Set.

    Depress takeoff trim button and hold until takeoff trim lamp lights. Check surface position indicator reads 3.8° (±2°) TEU.
11. Fuel quantity and distribution - Checked.
12. Fuel panel - Checked.
13. TFR video check - Completed. (If required)
14. Circuit breaker panels (3) - Checked. R
15. Canopies - Closed and latched, lock tab flush, handle checked, unlock warning lamp out, and altimeters steady.

    Snap the spring-loaded latch handle lock tab into the locked (flush) position and pull on the latch handle to check that it is locked.

**WARNING**

When canopy is closed and locked, check to insure that the AVVI and standby altimeter do not jump. If they do, it indicates a leak in the static source and a static system pressure leak check should be accomplished before flight. Airspeed, Mach and altitude information will be unreliable. Normally CADS caution lamp or OFF flags will not be visible.

16. Helmet visors - Lowered. (As practical)

17. Warning and caution lamps - Checked.

    Check that all warning lamps are out and that caution lamps are compatible with mission. The $\alpha/\beta$ probe heat and anti-skid caution lamps will be lighted if the switches are left in the OFF position.

18. Takeoff brief - Completed.

## TAKEOFF

### RUNWAY CHECK (BOTH)

1. Anti-collision light - ANTI COLLISION, position lights to STEADY.

2. IFF - As required.

3. Nav radar video - Checked.

4. Takeoff trim - Checked.

**WARNING**

A possible malfunction is indicated if trim has drifted. 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.

5. Anti-skid switch - ANTI SKID, caution lamp out.

6. Pitot/probe heater switch - HEAT.

7. Flight instruments - Checked.

**WARNING**

Do not take off if the airspeed Mach indicator reads greater than 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.

**Note**

Check to ensure that attitude information is correct in PRI and AUX. Insidious failure of AFRS attitude reference may not light the AFRS caution lamp. Subsequent failure of primary attitude reference in critical IMC takeoff environments may result in severe disorientation and difficult transition to the SCAI.

8. Throttles - MIL. (Check bleed valve indicator - NONE.)

**WARNING**

- If the bleed valve indicator reads other than NONE, hold the Mach trim test switch to the test position for 5 seconds or less. If the bleed valve indicator still reads other than NONE after this action, a possible malfunction of the main fuel control is indicated and maintenance will be required before flight.

- If the engines have not been operated in the past several days, a slight amount of cockpit smoke or fumes may be present momentarily on initial run-up. This discrepancy must be documented in AFTO Form 781 after the flight. The smoke must be slight, and must dissipate immediately or a serious engine malfunction may exist and takeoff must not be attempted.

9. Throttles - MAX AB. (Check engine instruments)

**WARNING**

EPR is the primary means of evaluating engine thrust developed. An EPR reading outside the limits shown in figure 2-4, indicates an engine malfunction which should be evaluated further prior to takeoff.

**Note**

Releasing brakes after MIN AB is achieved greatly reduces landing gear axle fatigue damage.

## EPR Check Values

**CONDITIONS: MAXIMUM A/B**
**AIR SOURCE SELECTOR KNOB - OPPOSITE ENGINE OR OFF**
**FLAPS/SLATS - EXTENDED**
R **ENGINE/INLET ANTI-ICE SWITCH - OFF**
**ENGINES: TF30–P–3**

| OAT °F (°C) | Minimum EPR* |
|---|---|
| −10 (−23) & Below | 1.97 |
| 0 (−18) | 1.97 |
| 10 (−12) | 1.97 |
| 20 (−7) | 1.97 |
| 30 (−1) | 1.97 |
| 40 (4) | 1.95 |
| 50 (10) | 1.92 |
| 60 (16) | 1.89 |
| 70 (21) | 1.87 |
| 80 (27) | 1.86 |
| 90 (32) | 1.84 |
| 100 (38) | 1.81 |
| 110 (43) | 1.78 |
| 120 (49) | 1.75 |

Note: For maximum EPR add 0.20 to min. EPR shown.

**★ Figure 2-4.**

## 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 nosewheel steering or rudder as required. Differential wheel braking will extend takeoff roll. Nosewheel 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.

**Note**

- For center-of-gravity locations forward of 25 percent MAC, a correction to takeoff speed is necessary to assure proper rotation beginning 15 KIAS below takeoff speed. Takeoff speed should be increased 2 KIAS for each 1 percent center-of-gravity forward of 25 percent MAC. The resulting correction should be compared to any required Delta VSE and the larger of the two values used as Delta $V_{SE}$.

- 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.
- Adequate longitudinal control may be available to lift the nosewheel 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.
- Any back stick pressure prior to computed rotation speed can increase takeoff roll.

Immediately after nosewheel 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**

Maneuvering flight at angles-of-attack greater than 10 degrees should be avoided during the takeoff phase.

For typical takeoff, see figure 2-5. Refer to the Performance Appendix, T.O. 1F-111(E)A-1-1, for takeoff data.

**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 runway.

## CROSSWIND TAKEOFF

Under crosswind conditions, the aircraft tends to weather-vane into the wind. The weather-vaning tendency can be easily controlled with nosewheel 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.

**Note**

- 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.
- 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 a thrust reduction occurs after selecting gear up, do not lower the gear. Lowering the gear will not regain the lost thrust.
- If it is necessary to depress the landing gear handle lock release button to move the handle to the UP position, refer to Section III.

**Note**

The fuel tank pressurization caution lamp may light 30 seconds after the landing gear handle is moved to the UP position and remain lighted until the tanks are pressurized.

# TAKEOFF (TYPICAL)

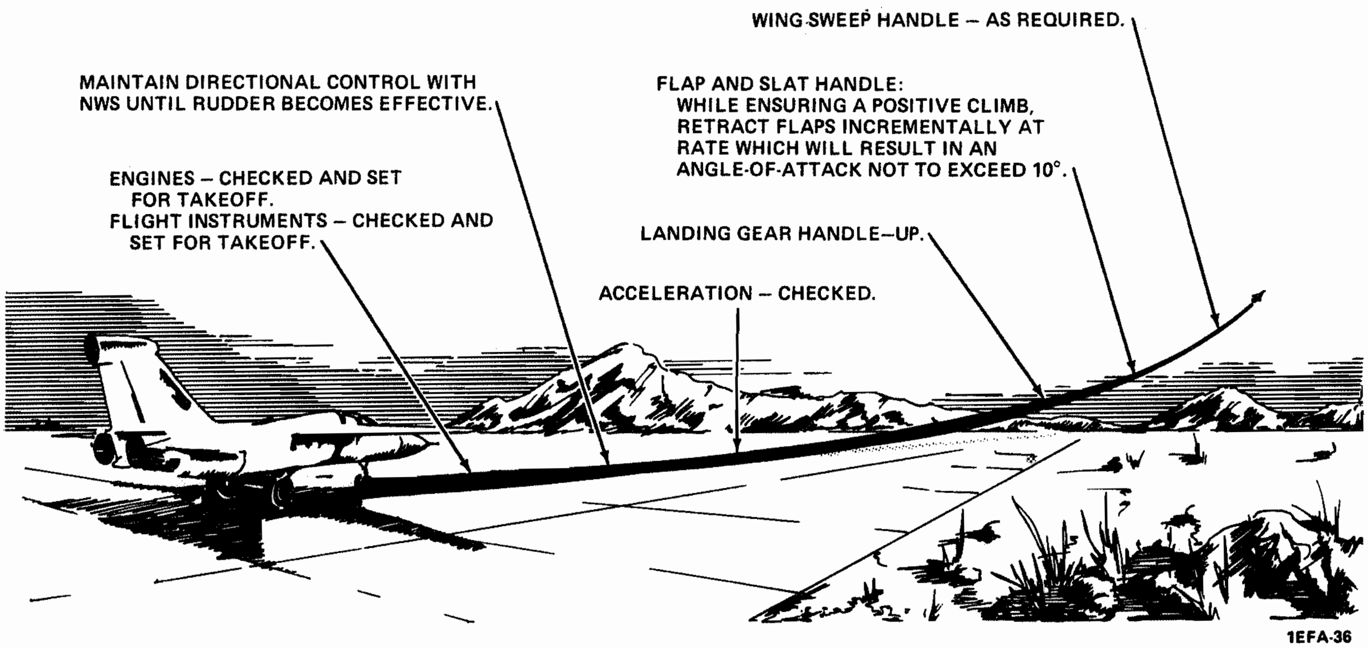



**Figure 2-5.**

2. Flap/slat handle:

   a. Flaps - While ensuring a positive climb, 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. An attempt to retract the slats by either the normal or emergency method may cause the asymmetry device to lock the flaps and slats preventing any further flap/slat movement. Sufficient lateral control may not be available to counter an asymmetrical flight condition. Refer to "Flap/Slat Retraction Malfunction," Section III.

**CAUTION**

If the flaps stop at an intermediate position during retraction, 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 that a landing be 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 futher damage or undesirable flap movement. Refer to "Flap/Slat Retraction Malfunction", Section III.

**Note**

Do not move the flap/slat handle gate release until ready for the next phase of configuration. If an emergency develops this will allow for immediate reversal of the flap/slat handle and prevent slat actuation prior to complete flap retraction.

b. Slats - UP, and verified.

Retract slats after verifying flaps are full up. Check that slat indicator displays UP.

**Note**

Maintain 1 g until slats/flaps are fully retracted.

3. Wing sweep handle - As required.

4. Throttles - As required.

For military power climb, reduce throttles to MIL when climb speed is attained.

5. Translating cowls - Check closed at Mach 0.50.

## CLIMB/CRUISE (BOTH)

When a climb above 10,000 feet MSL is anticipated, this checklist should be accomplished during the climb, and again after level off, once cruise power is set.

**WARNING**

Do not radiate JSS transmitters when cockpit of adjacent aircraft is within 60 feet of transmitters.

Refer to the Performance Appendix for climb/cruise operating data, and to Section I for complete fuel system operation.

1. Engine instruments - Checked.

2. Fuel - Checked.

Check the fuel quantity indicators for normal fuel usage. Check fuel transfer knob as required.

3. Oxygen and cabin altitude - Checked.

4. Altimeters - As required.

5. Wing sweep - As required.

R 6. Engine/Inlet anti-ice switch - As required.

7. Circuit breakers - Checked.

8. Inhibit switch - As required.

9. Jammer turn-on.

Complete the following steps for each jammer turned on:

a. Jammer radiate switches (10) - STBY.

b. Jammer reset pushbutton - Depress.

**Note**

- The TMP lamps on the jammer status panel will be on during the 3 to 5 minutes required for warmup.
- Ensure JSS ram doors override switch is not in the OPN position for Mach no. above 1.7.

c. Mode switches (10) - MAN TRUE. (If jammer BIT is to be performed)

10. JSS jammer BIT - Complete and record. (Optional)

11. Mode switches (10) - AUTO.

12. JSS - As desired.

13. SPS - As desired.

## TFR INFLIGHT CHECK (BOTH)

The following procedure includes a LARA command and fly-up check. It will be accomplished prior to using a TFR channel for TFR mode operation and as close to TFR operation check as practical. These checks must be initiated at an altitude above 5,000 feet AGL to provide proper indications.

1. Nav system drift accuracy - Checked.

**WARNING**

Inaccurate drift angle or lead-into-turn information to the TFR could result in the aircraft flying into vertical obstacles since the antenna search pattern will not be along the ground track being flown.

**Note**

Carefully check the accuracy of the drift information from the NC with the nav radar by setting ground manual mode, antenna uncaged, and azimuth cursor on the zero bezel, then note that the radar return track is parallel to the azimuth cursor. The TF DRIFT caution lamp should be lighted during the following checks if drift exceeds system limits.

2. Wing sweep - As required.

3. Radar Altimeter:

   a. Index pointer - 200.

   b. Altitude - Above 5,000 feet AGL for remainder of check.

4. TFR channel mode selector knobs - STBY.

5. Radar altimeter channel selector switch - CHAN 1 or 2.

6. Radar altimeter bypass switch - NORMAL.

7. TFR ride control knob - MED.

8. Terrain clearance knob - 300.

9. TFR channel mode selector knobs - One in TF; other in SIT.

10. Pitch steering switch - TF.

11. Lamp indications - Check.

   a. TFR channel fail caution lamp - Lighted.

   b. TF FLY-UP OFF caution lamp - Lighted unless autopilot release lever depressed.

   c. TF failure warning lamp - Lighted.

   d. REF NOT ENGAGED caution lamp - Lighted.

   e. RADAR ALT LOW warning lamp - Out.

   f. LARA MUX caution lamp - Lighted.

   g. FLT VECTOR caution lamp - Out.

**WARNING**

Inaccurate flight vector information to the TFR could result in the aircraft flying into the ground since the climb/dive command computation will be incorrect. Do not fly night or IMC TFR with the flight vector lamp lighted.

   h. VELOCITY caution lamp - Out.

   i. TF DRIFT caution lamp - Checked.

12. Radar altimeter bypass switch - BYPASS.

   a. The following lights will go out:

      TF failure warning

      TF FLY-UP OFF caution

      TFR channel fail caution.

   b. Pitch steering bar and aural command - Indicate dive.

13. Radar altimeter control knob - Depress and hold thru step 19.

   a. Radar altimeter bypass switch - Return to NORMAL.

   b. Radar altimeter indicator - 300 ($\pm$15).

   c. RADAR ALT LOW warning lamp - Out.

   d. Pitch steering bar and aural command - Approximately nulled. (If in level flight)

**WARNING**

If the pitch steering bar and aural command are not approximately nulled while in level flight, do not fly TF on that channel. A dive command indicates a malfunction in the LARA override commands and flight into an area where loss of forward video occurs could result in a dive below set clearance plane.

**Note**

Due to LARA and TFR tolerances, an absolute null of the displayed manual command cannot be predicted. However, if there is erratic movement or a steady displacement of $\pm 0.5$ g ($\pm 0.5$ inch) on the pitch steering bar, do not fly night or IMC TF.

14. Auto TF switch - AUTO TF.

    a. Aircraft nulls pitch steering bar. (Approximately level flight)

**Note**

The aircraft may establish a climb or dive due to LARA and TFR tolerances. The climb or dive established will be proportional to the pitch steering bar displacement. If the command is not initially nulled, the rate of climb/descent will continue to increase as a function of time after the pitch steering bar has nulled.

    b. REF NOT ENGAGED caution lamp - Out.

15. Terrain clearance knob - 400.

    a. Pitch steering bar and aural command - Climb command.

    b. Aircraft establishes climb.

**Note**

The occurrence of momentary TF fails when changing set clearances (up or down) can be minimized by selecting the desired set clearance as the TFR antenna sweeps up the scope.

16. Autopilot release lever - Hold depressed.

    REF NOT ENGAGED - Lighted.

17. Auto TF switch - OFF.

18. Terrain clearance knob - 500.

    a. The following lamps will be lighted:

       TFR channel fail caution

       RADAR ALT LOW warning

       TF failure warning.

    b. Pitch steering bar and aural command - Maximum climb.

19. Autopilot release lever - Release momentarily, then depress and hold.

    a. Fly-up maneuver - Initiated.

**WARNING**

If the fly-up capability is not operational in a TF or LARA channel, do not use that channel for TF operation.

20. Radar altimeter control knob - Release.

21. Radar altimeter bypass switch - BYPASS.

22. Autopilot release lever - Release

23. Terrain clearance knob - 1,000 feet SCP.

24. Range selector knob - E position.

25. E-scope - Checked and adjusted.

**WARNING**

If E-scope presentation is unusable for any reason, do not fly TF during night or IMC.

    a. Self-test pulse - Checked.

**WARNING**

If the self-test pulse is absent, do not fly TF during night or IMC.

    b. Radar altimeter override indication - Checked.

       Range selector knob - 15, then 10, then 5, then E. Check radar altimeter override indication at each position.

26. Reaccomplish steps 4 thru 25 with TF and LARA channels reversed.

Note

If a LARA channel is known to be inoperative, use the good channel for both TFR channel checks.

After Channel Checks Are Complete:

27. Radar altimeter channel selector switch - AUTO.

28. TFR mode selector switches - As desired.

## AUTO TF DESCENT (BOTH)

This procedure is required prior to all TFR letdowns. If TFR channel is in TF, begin with step 4.

1. Radar altimeter bypass switch - BYPASS. (If above 5,000 feet AGL)

2. Radar altimeter channel selector switch - Verify
R position.

R Select AUTO if both channels are operating nor-
R mally; select an operational channel if a channel
R does not test correctly.

3. TFR channel mode selector knobs - One in TF; other in SIT.

   LARA MUX caution lamp will go out if below 5,000 feet AGL.

4. Pitch steering switch - TF.

5. Range selector knob - E position.

   If above 5,000 feet AGL, the radar altimeter override function indication will be displayed prior to radar altimeter lock-on.

6. Altimeter - Set.

7. Radar altimeter index pointer - 900.

8. Terrain clearance knob - 1000.

9. Descent data - Confirm.

   a. Computed altitude (MSL) for radar altimeter lock-on. (If required)

   b. Computed level-off altitude (MSL).

10. AMI and AVVI command markers - Set. (As desired)

11. Helmet visors - Lowered

12. Wing sweep - As required.

13. Roll autopilot and heading nav sub-mode - As desired.

14. Auto TF switch - AUTO TF and check:

   a. Aircraft response.

   Check that 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 ($\pm$1.0) degrees with 1,000 feet clearance plane selected.

WARNING

If the flight controls do not center the pitch steering bar or fail to push over properly, a flight control malfunction is indicated. Do not continue the TF descent because aircraft response/pull out altitude may be compromised.

Note

- The 10 degree dive angle is generated through the LARA override circuit of the TF computer. Failure of the aircraft to establish a 10 degree dive (8,000-10,000 ft/min rate of descent) indicates a faulty LARA override circuit or flight control system.
- 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 1,000 feet above MEA. At this time, he should decrease the dive angle and level off at MEA. The letdown to 1000 feet, and subsequent lower settings, can be resumed when the pitch steering bar indicates a dive command as the rain return disappears from the E-scope presentation.

   b. REF NOT ENGAGED caution lamp - Out.

   c. Pitch steering bar and aural command - Nulled.

d. E-scope - Check ground returns.

**WARNING**

- If the E-scope presentation is unusable for any reason, TF operation should be terminated on that channel.

- Both crew members must monitor aircraft terrain clearance during TF flight. The E-scope and Nav radar are equally important. Due to the relative limited range of the E-scope, the Nav radar must be monitored closely. Corrective action must be taken if discrepancies between the E-scope and the Nav radar are detected. During periods when forward video is lost (due to low surface reflectivity) aircraft altitude will be monitored by using outside visual reference (if possible) and AVVI altitude interpretation. Do not TF over rolling no-show terrain because the aircraft will undershoot set clearance by 87 feet/degree of upslope and aircraft ground impact may occur even with a 68 percent fly-up maneuver. The crew should be especially watchful for unexplained descents while over water or level terrain.

**Note**

During AUTO TF letdowns over water, the two bar radar altimeter override indication is present while in bypass. The override indication will disappear at LARA lock on and reappear at the start of the pullout. During letdowns over land, the same indications are present except the radar altimeter override bars will not reappear at pullout due to TF forward video commanding the pullout. Prior to lock on over land or water, with the radar altimeter bypass switch in BYPASS, the indication will be present with the 1,000 feet SCP selected. The indication may be present with 750 feet SCP selected but will not be present at any other SCP.

15. 5,000 foot check (If applicable):

a. Radar altimeter - Locked on.

**WARNING**

- Do not continue TF letdown if proper LARA lock-on is not achieved. Failure of the LARA to lock-on at proper lock-on altitude will result in loss of LARA override operation and 68 percent violation detection capabilities.

- At LARA lock-on, the ground returns at the bottom of the E-Scope display should be half way between the one and two mile range marks. If the ground returns pass the one mile mark and the LARA has not locked-on, a LARA malfunction should be suspected and an immediate climb to MEA should be accomplished.

b. Radar altimeter bypass switch - Returns to NORMAL.

c. Dive angle - Increase to approximately 12 degrees. (2° pitch change)

d. Pitch steering bar and aural command - Nulled.

e. LARA override indication - Checked, not present.

**Note**

The E-scope radar altimeter override indication will disappear when the minus 12 degree pitch limiter commands aircraft dive angle.

f. LARA multiplexer caution lamp - OUT. (LARA R
channel selector switch in AUTO) R

g. Radar altimeter bypass switch - BYPASS momentarily. (If necessary)

**Note**

This step should be accomplished if the TFR channel was returned to STBY after the TFR inflight check. The presence of a momentary flyup confirms that flyup protection is re-armed and available.

16. Level-off check:

a. Aircraft begins rotation toward level flight by approximately 2,000 feet AGL. (1,700 feet AGL absolute minimum.)

b. Pitch steering bar and aural command - Indicate a climb.

c. Radar altimeter/AVVI - Indicate level off.

d. Clearance - 900 to 1,200 feet AGL.

**WARNING**

- If a recurring undershoot of more than 10 percent of the set clearance is experienced during night/IMC, immediately go to the next higher clearance setting that will remain within limits, or try the other TFR channel. Do not operate on a given channel at or below a set clearance where the 10 percent undershoot has occurred.
- Terrain following flight should not be attempted when there is an altitude error indicated on the radar altimeter indicator.

e. TF DRIFT caution lamp - Out.

**WARNING**

Do not fly night or IMC TFR if the TF DRIFT caution lamp is lighted even though drift accuracy checks within limits. A lead-into-turn malfunction may preclude safe TF operation.

**Note**

The TF DRIFT caution lamp will often be lighted by crosswinds at higher altitudes. A descent to MEA or an increase in airspeed may extinguish the lamp.

f. Airspeed/angle-of-attack - Within limits for TF operation. (See Part 12, T.O. 1F-111(E)A-1-1.)

**WARNING**

Under no circumstances should airspeed be allowed to decrease below minimum allowable airspeed during TF operations. To do so may result in exceeding angle-of-attack limits in event of a TF fly-up.

g. Set clearance plane - As desired.

**Note**

- When switching between channels, or when selecting lower clearances, a momentary TF fail and fly-up maneuver may occur.
- The radar altimeter index pointer should be set at the lower limit (approximately 10 percent below the set clearance plane) so the radar altimeter low warning lamp will light to indicate an unsatisfactory or degrading TF system.
- When flying at either 1,000 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 zero to 3/4 mile. At lower set clearances (200, 300, and 400), a separation of approximately 1/4 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 pitch steering bar and flight control system.

## DESCENT CHECK (COMMAND RESPONSE)

1. Wing sweep handle and lockout controls - As required.
2. Radar altimeter - Set.
3. Altimeters - Set.
4. Master radiate switch - OFF.
5. Inhibit switch - SAFE.
6. Jammer radiate switches (10) - STBY then OFF. (Allow 1 minute in standby for cooling before going to off).
7. JSS BIT - Complete.
8. JSS power switch - OFF.
9. SPS mode switches (4) - OFF.
10. TTWS power/test switch - OFF.
11. ILS - POWER and set. (If required)

## BEFORE LANDING (COMMAND RESPONSE)

1. CMDS - SAFE and OFF.
2. TFR mode selector knobs (2) - As required.
3. Ground roll spoiler switch - BRAKE.
4. Anti-skid switch - ANTI SKID.
5. Landing and taxi lights switch - LANDING.
6. Autopilot/damper switches - DAMPER.

## LANDING PATTERN (BOTH)

**Note**

During approach, ensure JSS and SPS are off.

This check will be accomplished prior to each approach.

1. Fuel quantity and distribution - Check.

   Check fuselage indicator totals against totalizer reading (±1,000) 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 "Abnormal Fuel Distribution/Indicator Malfunction", Section III.

2. 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. Refer to "Wing Sweep for Landing" charts, Section VI.
   - The wings must be at 26 degrees or less to allow flap/slat extension.

3. Backup approach speed - Compute. (Figure 2-6)
4. Radar altimeter - Set. (As desired)

   The radar altimeter should be used only as a cross reference to detect errors in the pressure altimeters.

   **WARNING**

   Do not use the radar altimeter to determine DH/MDA since, under certain terrain conditions, this could cause descent below the published minimum altitude.

5. Altimeters - Set.
6. Landing gear handle - DN and check.

   Extend the landing gear after airspeed is below 295 KIAS. Check that warning lamp in landing gear handle is out, landing gear position indicator lamps are lighted, and hydraulic pressure is normal.

   **WARNING**

   - After placing the landing gear handle to DN, selection of the slats/flaps during decelerating flight should not be delayed and extension of slats should be accomplished while landing gear is in the extend cycle. Any delay 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 landing gear warning horn setting, 200 (±12) KIAS, exercise caution to insure the landing gear is down and locked.

# APPROACH SPEED

DATA BASIS: ESTIMATED
DATE: 30 AUGUST 1982

CONFIGURATION:
FULL FLAPS AND SLATS
16 TO 26 DEGREES WING
SWEEP

CONDITIONS:
INDEXER = 10 DEGREES
WING ANGLE OF ATTACK,
C.G. AT MOST FORWARD
ALLOWABLE POSITION

FUEL GRADE: JP-4
ENGINES: TF30-P-3

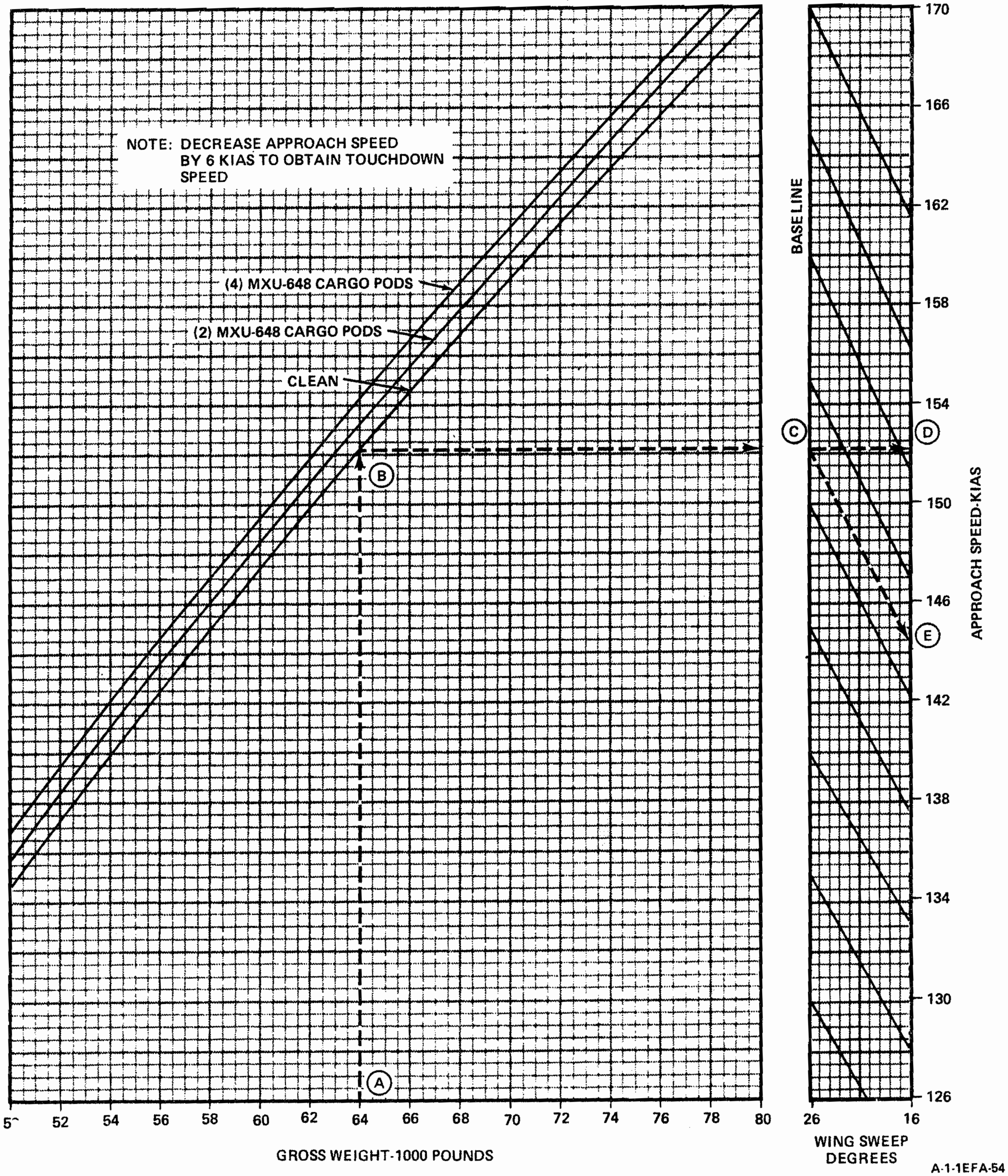



Figure 2-6.

**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.

- There may be a slight delay between the time the slats extend, and the gain changer caution lamps go out if the aircraft is experiencing sideslip.

7. 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. The gain changer lamps provide an indication of slat position. After the landing gear handle is placed down, the gain changer lamps will go out if the slats extend 70 percent or more. The gain changer lamps will remain lighted if the flight control disconnect switch is in OVRD, or if there is a malfunction in the adverse yaw compensation circuits. In this case, the lamp will not go out when the control system switch is placed to takeoff and land. If the gain changer lamps remain lighted and the 70-percent 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 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 "Landing With Flap and Slat Malfunctions," Section III.

**Note**

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

- Airloads may prevent full slat extension at airspeed approaching the slat limit speed; however, as airspeed is reduced resultant lowering of airloads will allow full slat extension.

- Maintain 1 g until slats/flaps are extended to the desired position.

- Do not move the flap/slat handle gate release until ready for the next phase of configuration. If an emergency develops this will allow for immediate reversal of the flap/slat handle and prevent flap actuation prior to complete slat actuation.

8. 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 "Landing With Flap and Slat Malfunctions," Section III.)

**CAUTION**

If the flaps stop at an intermediate position during 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 that a landing be 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 or undesirable flap movement.

9. Ground roll spoiler switch - BRAKE.
10. Translating cowls - Check open at Mach 0.44.
11. Elevator position - Checked. (At 10 degrees angle-of-attack)
    a. 16-degree wing - 4 to 10 degrees trailing edge up.
    b. 26-degree wing - 2 to 10 degrees trailing edge up.

    For wing sweeps between 26 and 16 degrees, 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.

**WARNING**

This check must be accomplished at 1 g and 10 degrees angle-of-attack. If elevator position is out of limits, an abnormal cg exists. Wing sweep must be changed to correct this condition before landing.

**Note**

- Wing sweep selected for landing will depend on gross weight and runway conditions as long as horizontal stabilizer position limits in the landing configuration are maintained.
- The cg horizontal stabilizer position range will provide safe operation for all landing wing sweeps and store loadings.

## LANDING

See figure 2-7 for a typical landing pattern. Brakes should be used as required compatible with runway available. Engage nosewheel steering as required for landing roll. For Landing Data, refer to Performance Appendix.

**Note**

When practical, use the full length of the runway to reduce brake heating and wear.

## 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 2-7. 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 aircaft to approximately 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 computed approach speed plus 20 KIAS (160 KIAS minimum). During the base turn, gradually reduce airspeed to an on-speed indication. Roll out on final approach approximately 300 feet AGL and approximately one mile from the end of the runway. 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. As the aircraft enters ground effect (approximately 30 to 50 feet above the ground), the nose will tend to drop and additional back stick should be applied to maintain a constant pitch attitude. Do not attempt to flare but maintain the approach attitude until touchdown. Reduce the power to idle at touchdown. Allow the aircraft to settle on the main gear and the ground roll spoilers to extend. The aircraft nose will normally tend to fall through at touchdown. This nose R
down rotation is easily controlled and the nosewheel should be flown smoothly to the runway. 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 nosewheel at speeds below approximately 90 KIAS. Directional control can be maintained with the rudder and differential braking down to the lower speed regions. The rudder loses effectiveness below approximately 50 KIAS and differential braking and nosewheel steering should then be used for directional control. Nosewheel steering should be engaged at minimum speeds with the rudder pedals at or near neutral. Brakes can be used as required throughout the landing roll.

# LANDING PATTERN (TYPICAL)

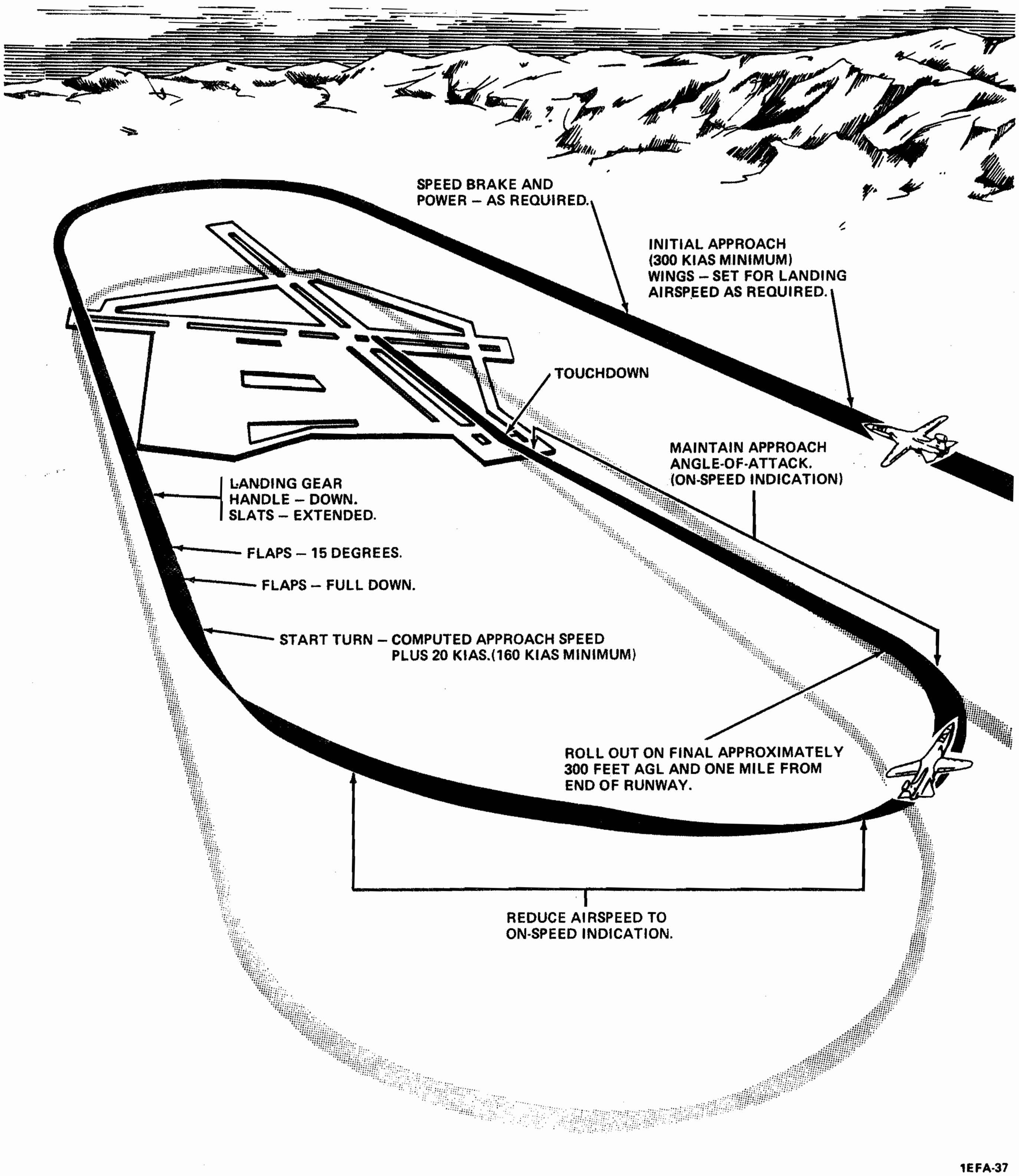



**Figure 2-7.**

**WARNING**

During the landing phase, 14 degrees angle-of-attack should not be exceeded. Inadvertent stall and post-stall gyrations may result from exceeding these limits.

**CAUTION**

Rapid or abrupt lateral or longitudinal stick motions can cause momentary increase in rate of sink and therefore should be avoided.

Turbulence, gusty wind or wind shear conditions may induce variations in angle-of-attack or airspeed and may 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 touchdown characteristics and susceptibility to a pilot induced oscillation, this additional airspeed should be dissipated so that an on-speed indication exists prior to touchdown.

**WARNING**

- Any delay in initiating a go-around from a pilot induced oscillation may result in loss of aircraft control.
- During a hard landing or pilot induced oscillation, the landing gear shock absorbing system may mask landing gear structural damage. After a pilot induced oscillation or aborted hard landing, the gear should be left extended and visually checked.

## SHORT FIELD LANDING

Fly a normal on-speed approach. Touchdown should be as close to the end of the runway as possible. After the spoilers have extended and the nosewheel is firmly on the runway, apply maximum anti-skid 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 nosewheel does not rise from the runway. The stick can be brought to the full aft position without unsticking the nosewheel at speeds below approximately 90 KIAS.

- If a runway cable is available, be prepared to extend the hook if the aircraft cannot be stopped prior to the end of the runway.
- At light gross weights, the anti-skid system cycling will be extreme and continue throughout the ground roll until just before the aircraft is stopped. At heavier gross weights, little anti-skid 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 anti-skid braking applied.
- Maximum braking should be released, if practical, at approximately 25 knots to prevent brakes from fusing.

## 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 aircraft 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 aircraft 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 aircraft with recommended tire pressure and 0.1 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 (nosewheel) 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 aircraft is subjected to hydroplaning to any degree, directional control becomes difficult. Under total dynamic hydroplaining conditions, nosewheel 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 0.08 inch of water. New tires tend to release hydro-dynamic pressures and will require in excess of 0.2 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 above and directional control of the aircraft 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 the 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 cable engagement.

**CAUTION**

If anti-skid protection is lost after touchdown, the failure detection circuit will automatically return the brake system to manual control and light the anti-skid caution lamp. This situation can occur during hydroplaning and may result in blown tires.

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. Maximum aerodynamic braking is obtained through use of the ground roll spoilers and the horizontal tail. The spoilers should be extended and the stick eased aft as brakes are applied. Exercise caution to ensure that the nosewheel does not rise from the runway. The stick can be brought to the full aft positon without unsticking the nose wheel at speeds below approximately 90 KIAS. To avoid inhibiting wheel spinup, and to improve wet runway wheel cornering capability, ensure 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.

**Note**

On slippery runways, wheel cornering capability is reduced by braking action, which may increase susceptibility to skidding. If skidding is encountered during braking, release brake pressure until directional control is assured.

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 nosewheel steering may be used to maintain directional control. Nosewheel 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 cable 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 component 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. Refer to "Crosswind Takeoff and Landing Limits," Section V, for crosswind landing limits.

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 he aircraft has slowed to taxi speed and just prior to turning off the runway.

## 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 ensure 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 nosewheel steering may be used to maintain directional control; however, nosewheel 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 nosewheel steering is engaged, inputs should be kept small as steering effectiveness diminishes rapidly with nosewheel 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 aircraft.

## TOUCH AND GO LANDING

Touch and go landings should be accomplished using the same technique as presented in the Normal Landing and Normal Takeoff procedures this section. After touchdown, power should be reduced to IDLE to allow the ground roll spoilers to extend and the aircraft to decelerate. Directional control should be maintained with rudder pedals while the throttles are advanced to MIL or AB power, as required. The nose should be lowered at least slightly from the on-speed landing attitude. The nosewheel may be flown smoothly to the runway. Either technique will ensure that the spoilers remain extended and the main gear struts remain compressed during the landing phase. During the acceleration or takeoff phase, check engine instruments and caution lamps for normal indications. At 10 knots below the approach speed, rotate to the normal takeoff attitude.

**Note**

Failure to lower the nose slightly during ground roll portion of a touch and go landing may prevent compression of the main landing gear struts. In this situation, the ground roll spoilers will not extend, a pilot induced oscillation may result, and directional control may be more difficult.

## NO FLAP/SLAT TOUCH AND GO LANDING

Touch and go landings with slats and flaps retracted should be made using the "Slats UP (Gain Changer Lamps On), Flaps Five Degrees Or Less Landing" procedures and techniques in Section III. After touchdown, smoothly advance the throttle to MIL or AB power, as required. Do not allow the nose gear to contact the runway above nose gear tire limits. Check engine instruments and caution lamps for normal indications. Care must be taken to prevent the tail section from contacting the runway because of the increased elevator effectiveness in this configuration.

## GO-AROUND

The decision to go-around should be made as early as possible. When wind shear, turbulence, or pilot inputs result in an unsatisfactory descent rate or landing condition, maintain a landing/go-around attitude and immediately apply power as necessary. AOA should be increased to 14 degrees, if necessary, to prevent a hard landing. Refer to figure 6-9 for AOA relationships and figure 6-2 for stall warning safety margin. Improper control inputs near the ground may result in a hard landing or pilot induced oscillation. A pilot induced oscillation may occur after a hard landing. If a pilot induced oscillation occurs, hold the stick slightly aft of center and immediately advance power to MAX AB.

**WARNING**

- Any delay in initiating a go-around from a pilot induced oscillation may result in loss of aircraft control.
- During a hard landing or pilot induced oscillation, the landing gear shock absorbing system may mask landing gear structural damage. After a pilot induced oscillation or aborted hard landing, the gear should be left extended and visually checked.

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 2-8.) 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.

# GO-AROUND (TYPICAL)

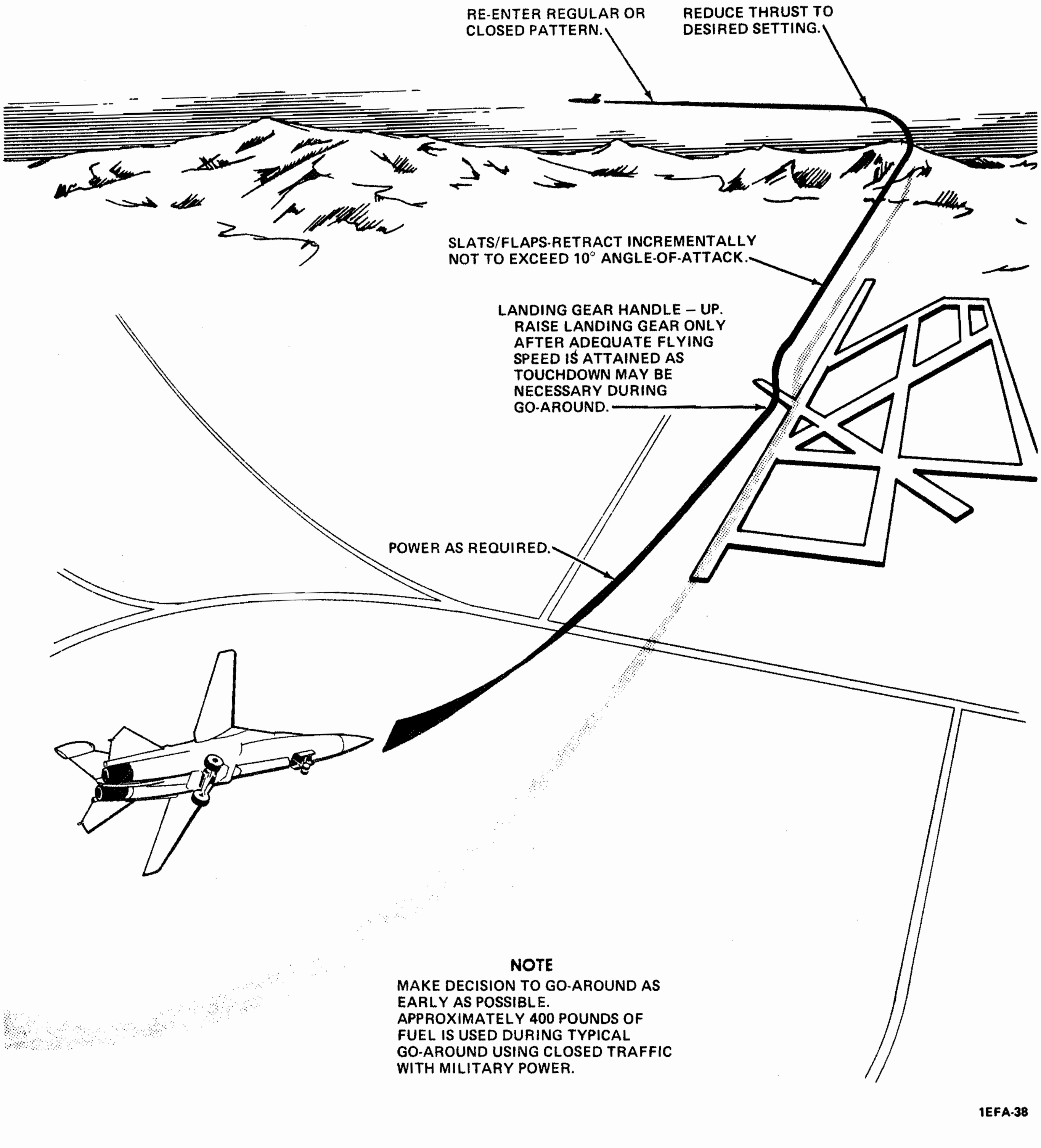



Figure 2-8.

## BEFORE CLEARING RUNWAY (COMMAND RESPONSE)

1. Nav radar function selector knob - STBY.

2. TFR channels (2) - STBY.

**WARNING**

The NRS/TFR must not be left in the transmit position after landing due to the possible danger to personnel in front of the aircraft.

3. Nosewheel steeering - As required.

4. Anti-skid switch - OFF. (Approximately 20 knots)

## AFTER LANDING (BOTH)

**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.)

1. Rain removal system - OFF.

**CAUTION**

Failure to confirm rain removal airflow has ceased may result in windscreeen damage.

2. Crew module safety pins (5) - Inserted.

**Note**

The ejection handle safety pins provided must be inserted from center console outboard to preclude interference of the pins with seat adjustment.

3. IFF mode 4 control knob - HOLD. (If required)

4. IFF master control knob - OFF.

5. Position and landing lights - as required.

   - Landing lights - OFF or TAXI. (If necessary)
   - Anti-Collision light - OFF.
   - Position lights - FLASH.

6. Pitot/probe heater switch - OFF/SEC.

7. All nonessential equipment - OFF.

8. Ground roll spoiler switch - OFF.

9. When required to taxi with slats up or wings aft of 26 degrees:

   a. Flaps/slats - Retracted.

   b. Wing sweep - As required.

10. Applicable throttle - 75 percent rpm, then OFF. (If required)

    Prior to shutdown, operate the engine at 75 percent rpm for 15 seconds to scavenge oil.

**WARNING**

If taxiing with a single engine, wing sweep handle should not be actuated. Momentary loss of utility hydraulic pressure will result.

## 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. Inertial nav computer - Set. (If required)

   - Present position latitude and longitude - Corrected.
   - Rapid alignment. (If applicable)

5. Inertial nav mode selector knob - OFF.

6. Nav radar - OFF.

7. TFR - OFF.

8. Takeoff trim -Set.

9. Pitch, roll and yaw damper switches - OFF.

10. Flight control computer power switches (3) - OFF.

11. Horizontal tail surfaces - Check. (GO)

**WARNING**

Oscillations of the horizontal tail surfaces indicate a malfunction of the horizontal tail servo-actuator valves.

12. CADC, GND IGN, and GYROS power - OFF.

13. Nacelle vent ejector system - Check. (GO)

Prior to each engine shutdown, the ground ob- R
server will feel for air being ejected by the na- R
celle vent ejector system by placing a hand be- R
hind the ram air/ejector air exit. If no air flow is R
felt, the nacelle vent ejector system or the land- R
ing gear safety (squat) switch has malfunctioned. R

**WARNING**

Do not move any flight controls until check is R
complete and ground observer is clear of the R
horizontal stabilizer. R

14. Applicable throttle - 75 percent rpm, then OFF.

Prior to shutdown, operate the engine at 75 percent rpm for 15 seconds to scavenge oil.

15. Nose gear ground safety pin - Installed. (GO)

16. Hydraulic pressure - Check.

Check for 2,950 to 3,250 psi indication and that the applicable hydraulic low pressure caution lamps light.

17. Air refueling receptacle - Open. (If applicable)

The air refueling receptacle will be opened to permit postflight inspection following inflight refueling operations.

18. Seats - Full up.

19. UHF radio - OFF.

**CAUTION**

Failure to turn off the UHF radio prior to interruption of electrical power will reduce equipment life.

20. Nacelle vent ejector system - Check. (GO)

21. Remaining throttle - 75 percent rpm, then OFF.

Prior to shutdown, operate the engine at 75 percent rpm for 15 seconds to scavenge oil.

22. Translating cowl test switches - HI MACH.

Check cowls closed.

23. 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.

24. Horizontal stabilizer droop check - Complete.

To verify the integrity of the stabilizer servo actuator control valves, the horizontal stabilizers must move to a minimum trailing edge down position of 12 inches.

**WARNING**

- A malfunction of a hydraulic servo actuator control valve may cause an uncommanded full control surface deflection. Aircraft response cannot be predicted in this situation, and control inputs may not correct the uncommanded deflection.
- Crew movement of the control stick will invalidate the horizontal tail droop check.

**Note**

- To obtain a valid droop check, the engines must have been run a minimum of 5 minutes to warm system fluid to normal operating temperature.
- Droop will normally occur within 90 seconds of throttle cutoff. If droop does not occur within 90 seconds, a ground observer should push up momentarily on the trailing edge of the stabilizer. The stabilizers may be pushed separately. If the stabilizers droop on release of the force, the check is valid.

25. Battery switch - OFF.

**CAUTION**

To prevent possible erroneous landing gear retraction commands, including speed brake movements, do not turn the battery switch OFF until the horizontal tail droop check is completed.

26. All switches and controls - OFF, NORMAL or SAFE.
27. Oxygen control lever(s) - OFF.

**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.

28. Oxygen control knob(s) - EMER.
29. Oxygen control knob(s) - NORM.
30. Oxygen regulator dust cover(s) - Installed.

## EXTERIOR POSTFLIGHT

Check condition of equipment and the aircraft in general for leaks, secure panels, and missing/loose fasteners.

Specifically check:

1. Forward SPS antennas.
2. Wing glove antennas.
3. Weapons bay radome.
4. Engine intake.
5. ECS inlets and exhausts.
6. Low band blade antennas.
7. Flap vanes.
8. Speed bumps.
9. Low band antennas on fin.
10. Fin cap radomes.
11. Engines (aft end).

## SCRAMBLE

The following procedures assume that the actions below have been completed prior to the aircraft being placed on an alert status:

- Complete preflight inspection to include a power-on cockpit inspection and an operational check of the engines, flight controls/damper system, spoiler monitor and ground roll spoilers.
- INS aligned and placed in RAPID ALIGN.
- Aircraft is cocked for scramble per local policy and instructions.

If the above actions are not completed prior to scramble, normal procedures should be used.

### BEFORE ENTERING COCKPIT (BOTH)

1. Ground safing devices - Removed. (As appropriate)

### BEFORE STARTING ENGINES (BOTH)

1. Ladders and intake covers - Checked removed.
2. Oxygen - On.

### ENGINE START (AC)

1. Battery switch - BATTERY.
2. Engine ground start switch - PNEU or CARTRIDGE. (As applicable)
3. Applicable engine throttle - Lift to start position.
4. Engine throttle - IDLE.
5. Engine instruments - Check.
6. Generator switch - ON.
7. Nose gear ground safety pin - Removed and displayed. (GO)
8. Speed brake ground lock - Removed. (GO) (If installed)
9. External power and air - Disconnected. (GO) (If required)
10. Remaining engine - Start. (Repeat steps 2 thru 6.)

## AFTER ENGINE START (EWO)

1. Aircraft on internal power.
2. Inertial nav mode selector knob - ALIGN.
3. Crew module safety pins (5) - Removed.
4. Inertial nav mode selector knob - GREAT CIRCLE or above. (After flashing ALIGN lamp)
5. Platform alignment control knob - NORM.

## BEFORE TAXIING (BOTH)

1. Chocks - Removed.
2. Auxiliary brake handle - IN.
3. Nosewheel steering - Engaged.

## TAXIING (BOTH)

1. Brakes - Checked.
2. Flight Instruments - Checked and set.

## BEFORE TAKEOFF (COMMAND RESPONSE)

1. Wing sweep, flaps and slats - Set for takeoff.
2. Ground roll spoiler switch - BRAKE.
3. Translating cowl switches - AUTO.
4. Translating cowl indicators - Checked, OPEN.
5. Control system switch - NORM.
6. Takeoff trim - Checked.
7. Canopies - Closed and latched, lock tab flush, unlock warning lamp out and altimeters steady.

## ADDITIONAL ITEMS, NOT ESSENTIAL FOR LAUNCH

8. AFRS synchronization indicator - Nulled.
9. Fuel panel - Checked.
10. IFF - As required.
11. HF radio - Set. (As required)
12. Recheck all switch positions.

## RUNWAY CHECK (COMMAND RESPONSE)

1. Anti-skid - ANTI SKID, caution lamp out.
2. Throttles - MAX AB. (Check engine instruments and bleed valve indicator - NONE.)

## TFR VIDEO CHECK

**Note**

This check confirms TFR climb command computations from forward video. The aircrew may have to taxi the aircraft to a location where forward video is present.

1. Terrain clearance knob - 400.
2. Ride control knob - SOFT.
3. TFR channel mode selector knobs - L TF, R STBY.
4. ISC pitch steering mode switch - TF.
5. E-scope video - Ground returns within 1 NM and above the ZCL.
6. Takeoff trim - Set.
7. NCU-Altitude/test selector knob - TEST 2.
8. Indicated AOA - 2 to 6 degrees.
9. Radar altimeter control knob - Depress and hold.
   a. VELOCITY caution lamp - Lighted.
   b. TF failure warning lamp - Out.
   c. TFR channel fail lamp - Out.
   d. Pitch steering bar and aural command - Indicate climb.
10. Radar altimeter control knob - Release.
11. Repeat steps 3 through 9 with TFR channels reversed.
12. TFR channel mode selector knobs - STBY.
13. Altitude/test selector knob - NORM.
14. Takeoff trim - Set.

This is the last page of Section II.

# SECTION III

# EMERGENCY PROCEDURES

## TABLE OF CONTENTS

This section contains procedures to be followed in the event of an emergency condition. These procedures will ensure 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.

**WARNING**

All odors not identifiable by the flight crew shall be considered toxic. Immediately go on 100 percent oxygen. Properly vent the aircraft and land as soon as practical. Do not take off when unidentified odors are detected.

**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 practical, maintain this configuration for landing.
- The canopy hatches should remain closed during all emergencies that could result in a crash or fire such as crash landings, aborted takeoffs, and cable 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.
- It is impossible to establish a predetermined set of instructions that would provide a ready-made decision applicable to all situtaitons. The emergency conditions combined with the pilot's analysis of the condition 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, lighting, aircraft gross weight and command guidance.

### LAND AS SOON AS PRACTICAL:

Emergency conditions that are less urgent, and although the mission is to be terminated, the degree of the emergency is such that an immediate landing at the nearest suitable field is not necessary.

## CAUTION LAMP ANALYSIS

See figure 3-1 for analysis and suggested corrective action to be taken whenever a caution lamp is lighted. The caution lamp analysis tables deal with the analysis and corrective action for individual caution lamps. The simultaneous lighting of several caution lamps may be caused by a different problem than the individual caution lights would indicate. For example, simultaneous lighting of pitch, roll, and yaw damper lamps usually results from a hydraulic malfunction rather than individual damper problems. Lighting of PRI ATT/HDG, AUX ATT, SIS plus damper, channel and gain changer lamps is normally indicative of an electrical system malfunction. Lighting of several seemingly unrelated caution lamps may be due to wiring damage from a bleed air duct rupture. Therefore, when several caution lamps light simultaneously or in rapid succession, aircrews should recognize the possibility of hydraulic, electrical, engine, and/or bleed air duct malfunctions. R

# INDEX – CAUTION LAMP ANALYSIS

## Caution Lamp Analysis

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| $\alpha/\beta$ PROBE HEAT (This lamp is disabled above Mach 1.10) | On the Ground:<br><br>1. Pitot/probe heater switch OFF/SEC.<br><br>2. Primary heater in angle-of-sideslip or angle-of-attack probe overheated. | <br><br>1. Momentarily place the heater switch to HEAT to verify that heaters are functioning as indicated by the lamp going out.<br><br>2. Place switch to OFF/SEC and allow probe to cool. Lamp will remain lighted due to being in OFF/SEC. |
| | In Flight:<br><br>One heater in the left angle-of-attack probe, the right angle-of-attack probe, or the side-slip probe is inoperative. | Place Pitot/Probe Heater switch to OFF/SEC to ensure $\alpha/\beta$ Probe Heat caution lamp goes out and then return switch to HEAT to provide power to the Total Temperature Probe Heater. When operating in icing conditions, periodically place the Pitot/Probe Heater switch to OFF/SEC to check secondary system operation.<br><br>If the lamp stays on in OFF/SEC, and icing conditions exist, consider the AOA indicator erroneous and the SIS Alpha Limiter inoperative. $\alpha/\beta$ Probe inputs to the TFR, SIS, and AYC may be erroneous. Slow to Damper Off Limits; place the Pitch Damper Off to prevent a SIS induced pitchdown. Place the Flight Control Disconnect switch to OVRD prior to extending the Flaps 15° or more. Land as soon as possible using "Dampers Off Landing" procedures, this section. |
| 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. |

★ **Figure 3-1. (Sheet 1)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| AUTO COWL | On the ground: | |
| | Loss of redundancy in either open or close command sources. | Investigate and correct before takeoff. |
| | Inflight: | |
| | 1. Inflight below Mach 0.44: A close command exists and a second failure could close both cowls. | 1. To continue acceleration, the light must go out at 0.50 ($\pm$0.03) Mach; otherwise observe cowl open limits. When decelerating below 0.44 Mach, place cowl control switches to OPEN position and verify that both cowls are open.<br><br>If either or both cowls do not show OPEN, place the translating cowl override switch to OVRD.<br><br>If above procedure fails to open both cowls, refer to "Translating Cowls Closed Landing," this section. |
| | 2. Inflight above Mach 0.50: One or both cowls not fully closed. | 2. Do not exceed Mach 0.90 or 415 KIAS. |
| | 3. Inflight above Mach 1.10: Auto open lockout has not occurred, and a second failure could cause both cowls to open. | 3. Reduce speed to below Mach 0.90 or 415 KIAS. |
| AUX ATT | 1. AFRS attitude info unreliable. | 1. Verify flight instrument reference select switch is in PRI. Verify that the AFRS circuit breaker is set whenever the auxiliary attitude caution lamp goes out after being lighted. Failure of the I/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. |

**Figure 3-1. (Sheet 2)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| L BLEED DUCT FAIL<br><br>R BLEED DUCT FAIL | Affected bleed air duct ruptured. | On the ground: Shutdown the affected engine. Turn the air source selector knob to OFF and shutdown the remaining engine as soon as possible.<br><br>In flight: Refer to "Engine Bleed Air Duct Failure/Oil Hot Lamp", this section. |
| CABIN PRESS | Cabin altitude above 10,000 feet. (This condition is normal if pressurization selector switch is in COMBAT and flight is above 26,000 feet.) | Don oxygen mask. Check oxygen equipment. Assure oxygen is on. Check that pressurization selector switch is in NORM. Descend to 25,000 feet or below before continuing flight. |
| CADS | One of CADC monitors indicates malfunction. Also indicates loss of power to MSMA. | Leave gear down if failure occurs with gear down. 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. The data good signal to TFR may be lost. Improper scheduling of fuel control, spike caution lamps, gear warning, stall warning, and inertial nav computers may occur. Do not use IFF mode C. Refer to "Landing with CADC/Pitot-Static System Malfunction," this section. |
| PITCH CHANNEL AND/OR ROLL CHANNEL AND/OR YAW CHANNEL (Yaw channel with flaps retracted) | 1. Less than three lighted: One of the triple redundant channels is in error, or computer power supply failure has occurred. | 1. Depress the damper reset button momentarily. If the lamp does not go out, turn the affected damper off within damper off limits and land as soon as practical using "Dampers Off Landing" procedures, this section. If the lamp goes out, continue normal operation; however, if the pitch channel lamp was reset, induce a positive pitch change to assure that the lamp does not come on again. If it does, reset the lamp and do not fly manual or auto TF. |

Figure 3-1. (Sheet 3)

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | 2. If all three caution lamps are lighted, one power source and computer redundancy are lost. | 2. Momentarily depress the damper reset button. If all three caution lamps reset, continue the mission. If any one or two of the lamps remain lighted, proceed with step 1 above. If none of the caution lamps reset, consideration should be given to maintaining airspeed within dampers off limits, turning dampers off, and landing as soon as practical using "Dampers Off Landing" procedures, this section. In strong gusty wind conditions, it may be desirable to leave the dampers on. |
| YAW CHANNEL (With flaps extended) | Possible single failure of one or more of the redundant AYC signals. | 1. Leave flaps extended.<br><br>2. Damper reset button - Depress.<br><br>a. If lamp goes out, induce slight yaw maneuver.<br><br>(1) If lamp remains out, continue normal operation.<br><br>(2) If lamp relights, continue with next step.<br><br>b. If lamp does not go out, place flight control disconnect switch (FCDS) to OVRD and depress the damper reset button.<br><br>(1) If lamp does not go out, turn yaw damper off, return FCDS to NORM, and land as soon as practical using "Dampers Off Landing" procedure, this section.<br><br>(2) If lamp goes out, normal operation may be continued. After the flaps are retracted, the FCDS may be placed to NORM; however, return the FCDS to OVRD prior to lowering the flaps. Land from a straight in approach following normal landing procedures. |

**Figure 3-1. (Sheet 4)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| ROLL DAMPER<br>OR<br>PITCH DAMPER<br>OR<br>YAW DAMPER | One of the triple redundant commands to a damper servo is in error. | Depress damper reset button momentarily; if lamp does not go out, reduce speed to the applicable damper off limits, turn affected damper off, avoid abrupt inputs, limit lateral maneuvers to 60 degrees bank angle, and land as soon as practical using "Dampers Off Landing" procedures, this section. The alpha limiter will be inoperative if the pitch damper is inoperative.<br><br>For a roll damper or yaw damper that will reset, continue normal operation.<br><br>For a pitch damper that will reset, verify that the lamp does not come on during an intentionally induced pitch change at MEA or above. If pitch damper lamp lights during the induced pitch change or will not go out, do not fly manual or auto TF, turn the pitch damper off within damper off limits, and land as soon as practical using "Dampers Off Landing" procedures, this section. |
| 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. Do not depress the damper reset button unless the affected hydraulic 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 | On the Ground (With Slats Up): LG SAFETY RELAY circuit breaker open.<br><br>In Flight:<br><br>1. Flight control disconnect switch is in OVRD or gear handle DN but slats <70%. | Reset circuit breaker. If lamps remain lighted, squat switch failure may be cause.<br><br><br><br>1. Normal conditions no corrective action required. AYC is deactivated. If slats remain <70% refer to "Landing With Flap and Slat malfunction" this section. |

**Figure 3-1. (Sheet 5)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | 2. Landing gear handle DN but slats >70% but flight control system not in takeoff and land configuration or beta probe senses sideslip >6 degrees or malfunction in the AYC circuit.<br><br>3. Landing gear handle up, slats <70%, but flight control system still in takeoff and land configuration. | 2. Place flight control system switch to T.O. & LAND to override the automatic switching. If lamps remain lighted, place flight control disconnect switch to OVRD before lowering flaps. AYC will not be available. Avoid late runway alignment.<br><br>3. Do not fly TFR. Land as soon as practical. |
| PITCH OR ROLL GAIN CHANGER | One of the redundant roll or pitch gain changers is in error. | Depress damper reset button momentarily. If lamp goes out, continue normal operation. If lamp does not go out, 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 scheduled by the landing gear handle. | 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. |
| SIS | 1. One of the two alpha probes has failed.<br><br>2. SIS computer has a single failure. | Do not depend on alpha limiter to provide stick force cues of angle-of-attack even through the lamp resets. Restrict maneuvering and monitor angle-of-attack indicator or indexer during maneuvers.<br><br>Slow the aircraft to damper off limit, turn pitch damper off, and land as soon as practical using "Dampers Off Landing" procedures, this section. |
| FUEL DISTRIB | Fuel distribution out of limits or the primary or alternate fuel distribution monitoring system has malfunctioned. | Refer to "Abnormal Fuel Indicator/Distribution Malfunction," this section. |

★ **Figure 3-1. (Sheet 6)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| L FUEL HOT<br>R FUEL HOT | JSS coolant temperature too high for heat exchanger. | Place heat exchanger override switch to OFF. If lamp is still lighted, place all jammer radiate switches to STBY. After one minute place all jammer radiate switches to OFF. |
| FUEL LOW | Usable fuel in fuselage reservoir tank is 2,300 (±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<br>R FUEL PRESS | Affected fuel manifold pressure is low, improper engine feed selection, or boost pump malfunction. | Check eng feed selector knob, fuel tank pressurizaiton 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. |
| 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.<br><br>If fuel quantity and/or distribution problems exist, refer to "Fuel System Malfunctions," this section. |
| FWD EQUIP HOT | On the Ground:<br><br>Low airflow at low engine power settings. | <br><br>Increase one engine to 78% rpm. Lamp should go out within 30 seconds. If lamp is still lighted, turn ALQ–137 (SPS) off. If it is necessary to exceed 78% rpm, a system discrepancy is indicated. Allow up to 90 seconds with the lamp lighted, then turn off remaining non-essential equipment. |

**Figure 3-1. (Sheet 7)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | In Flight:<br><br>In Flight Preceded by a Reduction in Cockpit Airflow and Throttle Boost:<br><br>1. Ruptured service bleed air duct. | <br><br><br><br>1. Place air source selector knob to OFF or EMER and open speed brake to prevent added heat damage and cool MLG wheel well. Turn off non-essential equipment. |
| | In Flight Preceded by a Reduction in Cockpit Airflow Only:<br><br>2. 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; may also occur due to a failed temperature sensor).<br><br>3. Cooling turbine failure or ruptured duct down stream of cooling turbine; rapid reduction of airflow and cooling will be noted with possible accompanying noises. This should occur prior to fwd equip hot lamp light.<br><br>4. Low air flow due to extended operation of engine/inlet anti-icing switch in MANUAL. Lighting of the lamp may be accompanied by rapid loss of cabin pressurization and airflow. | <br><br>2. Place air source selector knob to RAM and observe flight limitations. Allow up to 90 seconds with lamp lighted then go to BOTH. If lamp does not go out within 10 seconds, return to RAM and turn off nonessential equipment.<br><br>3. Place air source selector knob to RAM and observe flight limits. Turn off all nonessential equipment.<br><br>4. Select AUTO on engine/inlet anti-icing switch or OFF if the AUTO mode is known to be malfunctioning. Once normal cooling is restored to the forward section, cockpit air flow will be restored. If cooling air is not restored, place air source selector knob to RAM. |
| LOW EQUIP PRESS | Pressure to forward equipment bay pressurized components is below limits. Prior to engine start, lamp may be lighted at high field elevations. | Turn TFR to standby and ECM off if above 15,000 feet altitude to prevent equipment damage. |

**Figure 3-1. (Sheet 8)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| L GEN<br>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 cable. Hook cannot be retracted in flight. |
| PRI HOT<br>UTIL HOT | Indicated hydraulic system fluid temperature is above 230°F (110°C). This is an indication of impending hydraulic system failure. | On the Ground:<br><br>Shut down the engines as soon as possible.<br><br>In Flight:<br><br>Place heat exchanger override switch to OFF. If lamp remains lighted, reduce demand on the hydraulic system. Reduce speed and land as soon as possible. |
| L PRI HYD<br>R PRI HYD | Pressure output of the indicated primary hydraulic pump is below 500 psi. | On the Ground:<br><br>Shut down the affected engine as soon as possible.<br><br>In Flight:<br><br>Monitor hydraulic pressure. If it is normal, land as soon as practical. If abnormal pressure, refer to "Hydraulic System Failure," this section. Damper oper will not be affected. |
| L UTIL HYD<br>R UTIL HYD | Pressure output of the indicated utility hydraulic pump is below 500 psi. | On the Ground:<br><br>Shut down the affected engine as soon as possible. |

★Figure 3-1. (Sheet 9)

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | | In Flight:<br><br>Monitor hydraulic pressure. If it is normal, land as soon as practical. If abnormal pressure, refer to "Hydraulic System Failure," this section. Damper oper will not be affected. |
| ICING | Ground Operation:<br><br>1. Icing condition sensed by ice detector. | <br><br>1. Do not take off until the lamp goes out. Select MAN on engine/inlet anti-icing switch for all ground operations before and after flight. After takeoff and clear of icing conditions, return switch to AUTO. Have the engine cowls and translating cowls checked for ice build-up prior to takeoff and after landing. |
| | 2. Malfunction of ice detection system. | 2. If icing conditions are not present, turn anti-icing system off. |

★ **Figure 3-1. (Sheet 9.1)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| | Inflight Operation: | |
| | 1. Icing conditions. | 1. Maintain a minimum of 80% RPM until clear of icing conditions. |
| | 2. Icing condition sensed by ice detector. | 2. 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 8,000 feet there will be a 300 to 500 foot fluctuation in cabin pressure when cycling the switch. There will also be a noticeable increase in EPR when system is cycled to OFF. If not, go to MAN. Lamp will remain lighted until 60 seconds after icing condition ceases. Turn engine anti-icing switch to OFF after clear of icing conditions. |
| | 3. Malfunction of ice detection system. | 3. 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.<br><br>2. Take action to obtain IFF identification on other modes. |
| INLET HOT | Anti-icing temperature excessive. | Shut off engine inlet anti-icing. Lamp should go out. If not, slow aircraft to reduce total temperature. |
| L ENG OIL HOT<br>R ENG OIL HOT | Excessive oil temperature due to engine bleed air duct failure, abnormal oil pressure, or inadequate oil cooling. | On the Ground:<br><br>Move throttle to IDLE, allow up to two minutes with lamp lighted to have ground observer determine if nacelle vent ejector system is operating. Then shut down engine.<br><br>In Flight:<br><br>Refer to "Engine Bleed Air Duct Failure," this section. R |

★ **Figure 3-1. (Sheet 10)**

## Caution Lamp Analysis (cont)

| | Indication | Probable Cause | Corrective Action |
|---|---|---|---|
| | OIL LOW | Indicated oil level in either engine down to 4 quarts. | Test the oil quantity indicator. If the pointer drives below zero, the indicator has malfunctioned. If the indicator tests properly, monitor oil pressure closely. If oil pressure begins to drop, refer to "Oil System Malfunctions", this section. |
| R R | L ENG OVERSPEED<br>R ENG OVERSPEED | Excessive low pressure compressor rpm. (As a self-test feature, lamp is lighted when eng is below idle rpm.) | Retard throttle of affected engine to IDLE, land as soon as practical using "Single Engine Landing" procedures, this section. The affected engine can be used, if necessary. |
| | 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 | 1. The flight instrument reference selector switch in AUX or inertial nav mode selector knob in OFF, HEAT, ALIGN or an AUX NAV CHECK mode.<br><br>2. Certain failures of the SP.<br><br>3. Certain failures of the INC. | 1. None required.<br><br>2. None required.<br><br>3. Place the flight inst reference selector switch to AUX. This will cage the nav radar antenna and cause a TF fly-up.<br><br>If the AFRS circuit breaker subsequently trips, the caution lamps will go out and the displays will revert to the erroneous primary reference. |
| R | L ENG SPIKE<br>R ENG SPIKE | Mach 0.35 or below, and the affected spike has not contracted or is not full forward. | Position both spike control switches to OVERRIDE. Do not attempt to return to NORM position after the spike control switch has been placed to OVERRIDE. Refer to "Spike System Failure" this section. |
| | SPOILER | One pair of spoilers has been voted out and locked down. | Maintain positive control of aircraft attitude and decelerate to safe speed. Do not attempt to reset spoiler since the malfunction will likely recur and cause another unwanted roll maneuver. The roll rate capability and spoiler brake effectiveness during landing will be reduced. |

★**Figure 3-1. (Sheet 11)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| TF FLY-UP OFF | TF fly-up is not available due to one of the following conditions:<br><br>1. Auto TF switch is in AUTO TF but a TFR channel is not in TF.<br><br>2. A TFR channel is in TF and a TF fly-up is not available because:<br><br>a. Control system in T.O. & LAND.<br><br>b. Fly-up circuit not armed. | Check switch positions. If lamp persists, do not fly manual or auto TF. |
| TF DRIFT | 1. Lead-into-turn comparison exceeds 4.0 degrees.<br><br>2. INS drift is 4.5 degrees or greater. | Reduce bank angle. Fly the aircraft to MEA. (If above, do not descend below MEA). Perform and verify the drift angle accuracy check. If actual drift angle is greater than 4.5 degrees, restrict TFR operation to day/VMC. If less than 4.5 degrees, terminate TFR operation on channel in use. |
| TURN LIMIT | The aircraft is executing a turn in which the heading rate exceeds a value of 2 degrees per second or the climb command exceeds 5.2 g's. | Reduce bank angle/climb command until lamp goes out. If lamp remains lighted, consider the condition as a TFR failure. |
| FLT VECTOR | 1. Malfunction in CADC.<br><br>2. Malfunction in INS.<br><br>3. Undetermined. | 1. If CADS caution lamp is also lighted, terminate TFR operation.<br><br>2. If PRI ATT/HDG caution lamp is also lighted, restrict TFR opertion to day/VMC.<br><br>3. If CADS and PRI ATT/HDG caution lamps are both out, terminate TFR operation. |
| VELOCITY | Indicates groundspeed input to the TFR differs from true airspeed by 130 knots or more. | Change indicated Mach to between 0.70 and 0.90. |
| 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. |

**Figure 3-1. (Sheet 12)**

## Caution Lamp Analysis (cont)

| Indication | Probable Cause | Corrective Action |
|---|---|---|
| WHEEL WELL HOT | Wheel well, weapons bay routing tunnel, and/or ac power panel area overheat condition. (Possible rupture of engine bleed air duct.) | Refer to "Wheel Well Overheat" procedures, this section. |
| WINDSHIELD HOT | Rain removal air exceeds 410°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 and observe "Ram or Emer Mode Flight Limits," Section V. |
| JSS RAM DOORS | One or both ram air doors failed to open or close. | 1. Place ram doors override switch to OPN. Maintain Mach at or below 1.7. Monitor JSS LIQ T/P, L FUEL HOT, and R FUEL HOT caution lights. |

**Figure 3-1. (Sheet 13)**

# GROUND OPERATION EMERGENCIES

## 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. Figure 3-2 depicts emergency entrance to cockpit. If time permits, accomplish the following:

1. Throttles - OFF.
2. Fire pushbuttons - Depress.

R Fuel shutoff valves require up to 4 seconds to close. Premature removal of electrical power may prevent fuel shutoff valve closure.

3. Auxiliary brake handle - Pull.
4. Battery switch - OFF.

## BRAKE MALFUNCTIONS

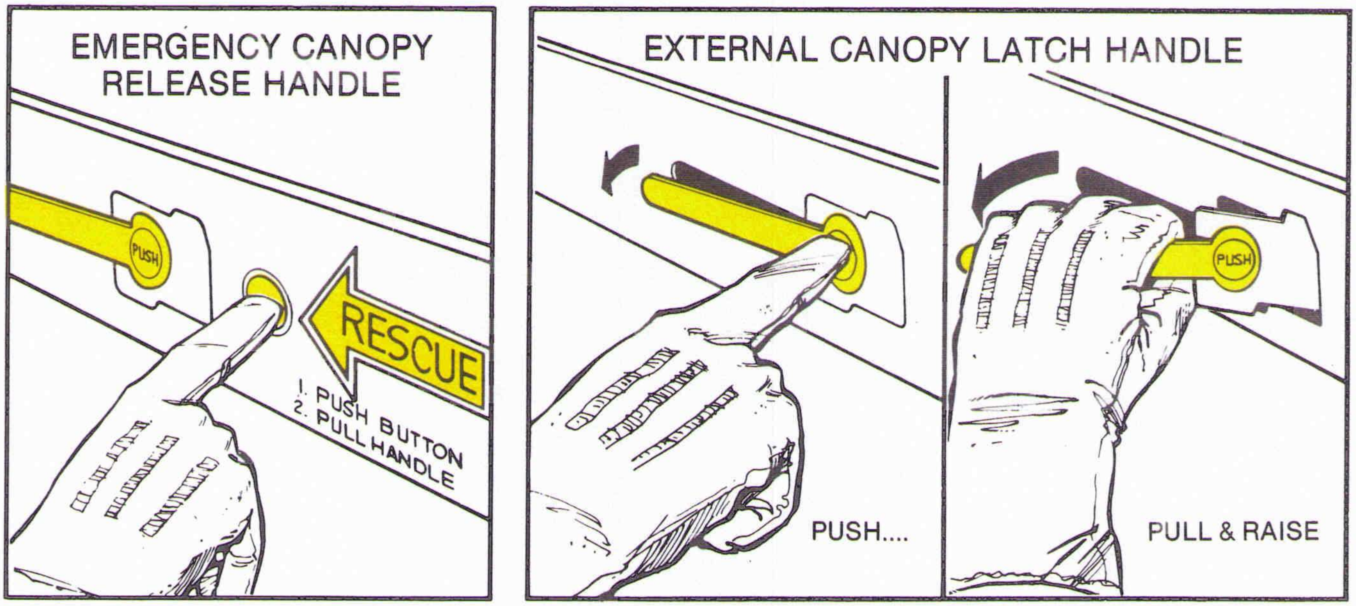
In the event normal pedal braking is not available, several methods may be used to obtain braking and/or control aircraft speed to prevent damage to the aircraft and other equipment. The following are suggested solutions. However, the sequence will be dictated by the situation.

1. Anti-skid - OFF.

   Ensure wheel brakes are released prior to turning the anti-skid switch off.

2. Thrust reduced.
3. Cable engagement.
4. Auxiliary brake handle - Pull.

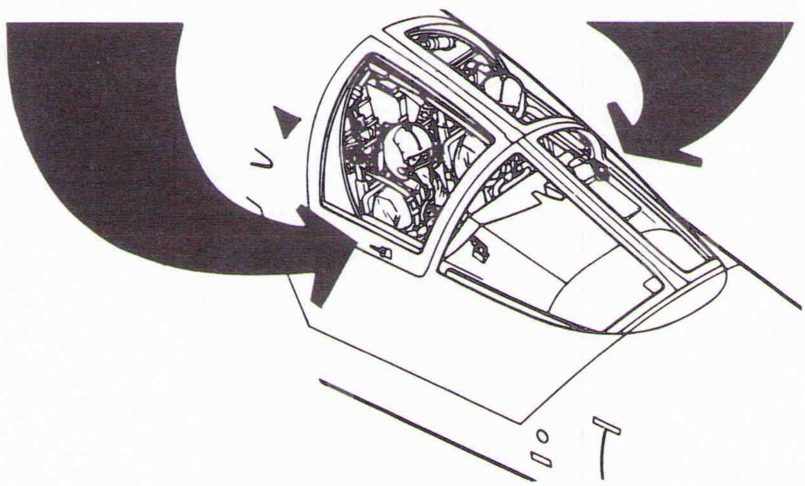
# EMERGENCY ENTRANCE




1. Push plunger to unlock internal handle.

2. Push in on external handle to extend.

3. Grasp handle to to raise hatch.



1EFA-56

Figure 3-2.

## AIRCRAFT/ENGINE FIRE

An aircraft/engine fire on the ground may be indicated by a lighted fire lamp or may be reported by ground personnel. Aircrew actions will be dictated by the nature and severity of the fire, aircraft movement, and the aircraft position in relation to other aircraft or hazardous areas such as fuel pits. The following procedure should be accomplished if time and circumstances permit.

1. Throttles - OFF.

**WARNING**

Time permitting, every effort should be made to shutdown the engines with the throttles due to the possibility of fire damage preventing closing of the fuel shutoff valves.

2. Fire pushbutton - Depress.
3. Agent discharge switch - UP.

**WARNING**

The fire extinguishing agent is available for one actuation only. Depressing the engine fire pushbutton the second time will disarm the fire extinguisher agent discharge valve. R

4. Auxiliary brake handle - Pull.
5. Battery switch - OFF.
6. Exit the aircraft.

In the event of fire engulfment, helmet visors should be down, mask in place, and the end of the oxygen hose placed inside the flight suit to prevent inhalation of fire.

# TAKEOFF EMERGENCIES

## ABORT/CABLE ENGAGEMENT

If an emergency ocurs during takeoff which requires an abort/cable engagement, the most important aircrew consideration is getting the aircraft stopped in the remaining runway. Subsequent aircrew actions will then be dictated by the nature of the emergency which necessitated the abort, such as engine fire. If an abort/cable engagement is necessary during takeoff, accomplish the following:

1. **THROTTLES - IDLE.**
2. **HOOK - EXTEND.** (If required)
3. Shoulder harness - Locked.

**If Fire/Fire Warning Lamp Exists:**

4. Throttle - OFF.
5. Fire pushbutton - Depress.
6. Agent discharge switch - Up.

**If Fire Is Confirmed and Continues:**

7. Auxiliary brake handle - Pull.
8. Remaining throttle - OFF.
9. Battery switch - OFF.
10. Exit the aircraft.

## 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 cable engagement. The chances for a successful cable 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 cable 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. 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 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 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.

## CABLE ENGAGEMENT

**WARNING**

Use of the MA–1 and MA–1A barrier/cable with this aircraft has not been tested, therefore, results of their engagement cannot be accurately predicted.

On center engagements of the BAK–9, BAK–12, extended runout BAK–12, dual BAK–12, and BAK–13 cable 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. Cable 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 cables, 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 cable engagements:

- Disengage nosewheel steering prior to cable contact.
- Do not attempt to correct yaw or roll tendencies during the arrestment until the aircraft is slowed sufficiently to ensure aircraft control.
- Apply light braking at the end of the arrestment when possible to minimize roll back without causing the aircraft to pitch up.

## 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, do not move the throttle. Refer to "Afterburner Failure" this section.

## ENGINE FAILURE DURING TAKEOFF

**If the Decision Is Made to Stop, Refer to "Abort/Cable Engagement" Procedures, This Section.**

**If Takeoff Is Continued:**

1. Throttles - MAX AB.
2. Landing gear handle - UP. (When safely airborne)

**Note**

A normal reduction of rudder authority will occur when the landing gear handle is placed in the UP position. This may be felt as a kickback on the rudder if more than 7.5 degrees rudder deflection is being held.

3. Air source selector knob - OFF or EMER. (As required)

**Note**

- Significant thrust is gained with the air source selector in OFF or EMER.
- With the air source selector in OFF or EMER, no servo air will be available for throttle boost or fuel tank pressurization. Loss of throttle boost will require a significant increase in force required to move the throttle (approximately 40 pounds with maximum throttle friction. Lack of tank pressurization will degrade fuel dump.

4. Flap/slat handle - As required.
   a. Flap/slat retraction - After close-in obstacles are cleared, establish an attitude that will clear terrain 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.

5. Air source selector knob - As required.
6. Refer to "Engine Emergencies" (Inflight), this section. R R

## 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, establish an attitude that will clear terrain and retract the flaps/slats at a rate to maintain 8.5 degrees angle-of-attack. Best single engine climb speed will equate to 8.5 degrees angle-of-attack.

## ENGINE FIRE DURING TAKEOFF

With any engine fire indication, immediate and positive steps must be taken to shut off fuel/hydraulic supply and extinguish the fire. The following procedure contains the most desirable sequence of steps for most engine fire situations. It assumes that the engine indicating fire can be identified promptly, without distracting the aircrew from the essential task of maintaining aircraft control. Under these conditions, closing the throttle prior to depressing the fire pushbutton will reduce engine rotation faster, will decrease the possibility of further damage in case of mechanical failure, and remove sources of re-ignition quicker. Under some conditions it may be difficult for the aircrew to determine which engine is indicating fire. If any doubt exists, the lighted fire pushbutton should be depressed immediately and the corresponding throttle closed when conditions permit. In either case it is essential that the engine be shut down (throttle in cut-off and fire pushbutton depressed) prior to activating the agent discharge system. It is essential that the fire pushbutton be fully depressed so it latches. If it fails to latch, the agent discharge switch will not be armed, but the fuel shutoff valve will remain closed if the button was fully depressed for at least 4 seconds. The fuel flow indicator on the affected engine may not provide a reliable indication that the fuel shutoff valve has closed. Failure to observe this sequence may result in re-ignition of the fire and subsequent loss of the aircraft. Certain engine malfunctions can cause the throttle to freeze. If this occurs, immediately proceed with the remaining checklist steps. Continued attempts to retard the throttle will result in a critical delay in engine shutdown. In this situation, delay placing the agent discharge switch up until rpm has started to decrease. This should aid in removing sources of re-ignition.

**WARNING**

- Delay in accomplishing the critical steps of the following procedure may allow the fire to damage critical systems with subsequent loss of aircraft control.
- Use caution to prevent inadvertently retarding the wrong throttle or depressing the wrong pushbutton and shutting down the good engine.
- With the fire pushbutton depressed and latched, depressing the fire pushbutton a second time will unlatch the button and disarm the fire extinguishing agent discharge valve. If the fire pushbutton does not latch, hold it fully depressed for at least 4 seconds to close the fuel shutoff valve and until the agent discharge switch is positioned to AGENT DISCH.
- The fire extinguishing agent is available for one actuation only. If the agent discharge is activated before fuel is isolated from the engine, it is probable that the fire will be reignited. If this occurs, no further means exists to extinguish the fire.
- If the fire lamp goes out before any of the following steps are accomplished, or if the fire lamp goes out before the procedure is completed, it is imperative that the remaining steps be accomplished to insure that the fire has been extinguished and the possibility of it reoccurring is reduced.

It is possible for a catastrophic engine failure to rupture the 16th-stage bleed duct in the main wheel well. Hot air escaping from the ruptured bleed duct can cause considerable damage before the WHEEL WELL HOT caution lamp lights. Indications of bleed duct failure are heavy acrid smoke in the cockpit or various caution/warning lamps lighting that are unrelated to the engine fire.

**If The Decision Is Made To Stop, Refer To "Abort/Cable Engagement" Procedures, This Section.**

**If Takeoff Is Continued:**

1. **THROTTLE - OFF.**
2. **FIRE PUSHBUTTON - DEPRESS.**
3. **AGENT DISCHARGE SWITCH - UP.**

   In some cases, it may take from 5 to 10 minutes for hydraulic system pressure to decay enough to light the primary and utility hydraulic caution lamps of the engine that has been shut down.

4. Landing gear handle - UP, when safely airborne.

**Note**

A normal reduction of rudder authority will be felt when the landing gear handle is placed in the UP position. This may be felt as a kickback on the rudder if more than 7.5 degrees rudder deflection is being held.

5. Air source selector knob - OFF or EMER. (As required)

**WARNING**

If the engine fire is accompanied by smoke in the cockpit or caution/warning lamps lighting that are unrelated to the engine fire, turn the air source selector knob to OFF or EMER.

CAUTION

If the air source selector knob is repositioned after initial selection, the shutoff valve may fail to an open position.

Note

- Significant thrust is gained with the air source selector in OFF or EMER.
- With the air source selector in OFF or EMER, no servo air will be available for throttle boost or fuel tank pressurization. Loss of throttle boost will require a significant increase in force required to move the throttle (approximately 40 pounds with maximum throttle friction). Lack of tank pressurizaton will degrade fuel dump.

6. Flap/slat handle - As required.

   Flaps/slats retraction - After close-in obstacles are cleared, establish an attitude that will clear terrain 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.

7. If fire is confirmed and continues - Eject.

**If Fire Is Extinguished:**

8. Land as soon as possible, using "Single Engine Landing" procedures, this section.

## FLAP/SLAT RETRACTION MALFUNCTION

CAUTION

If the flaps stop at an intermediate position during retraction, or if the slats will not retract after flap retraction, 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 be made with the existing flap setting, provided landing conditions are acceptable (RCR, ceiling, etc.).

Note

- If an asymmetric flap condition exists, do not attempt to raise the slats using either the normal or emergency mode. Initial slat movement may cause the asymmetry device to lock the flaps and slats preventing any further flap/slat movement.
- If an asymmetric slat condition exists, do not attempt to lower the flaps using either the normal or emergency mode. Initial flap movement may cause the asymmetry device to lock the flaps and slats preventing any further flap/slat movement.

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 Flap and Slat Malfunctions," this section.

**If a Rolloff Condition Does Not Exist (Make No Further Actuations):**

1. Flap/slat switch - EMER.

CAUTION

Placing the flap/slat switch to EMER will relieve hydraulic pressure to the flap motor and could prevent further damage.

2. Refer to "Landing With Flap and Slat Malfunctions," this section.

**If a Rolloff Condition Exists:**

Failure in the slat/flap drive system can result in asymmetric flap/slat configuration which will cause the aircraft to roll. If flap asymmetry is allowed to become large, roll control will be difficult and may require full lateral stick deflection aided with rudder deflection to prevent rolloff. The best corrective action is to stop flap retraction with the flap handle and seek a handle position which will minimize roll control requirement.

1. Apply lateral stick as necessary and follow with rudder for additional roll control if needed.
2. Flap/slat handle - Return toward previous position until rolloff stops. Make no further actuations.

Note

If an asymmetrical slat condition exists, do not position the flap/slat handle below the SLAT DOWN position. Otherwise, upon initial flap travel, the asymmetry device will prevent any further flap or slat movement. This is also true of emergency extension.

3. Flap/slat switch - EMER.

CAUTION

Placing the flap/slat switch to EMER will relieve hydraulic pressure to the flap motor and could prevent further damage or undesirable flap movement.

4. Wing sweep - 26 degrees if roll requirements cannot be reduced and aircraft control is marginal.

5. Refer to "Landing With Flap and Slat Malfunctions," this section.

## LANDING GEAR HANDLE LOCKED IN DOWN POSITION

Failure of the landing gear handle lock to release may be caused by a gear lock solenoid malfunction, an open landing gear safety relay circuit breaker, or dual failure of the landing gear ground safety (squat) switches.

WARNING

With a dual failure of the landing gear ground safety (squat) switches, the SIS alpha limiter and artificial stall warning device will be inoperative, and the ground roll spoilers will be armed inflight.

1. Landing gear handle lock release button - Depress and raise gear handle.

**If the Landing Gear Retracts:**

2. Upon mission completion, land using normal procedures.

**If the Landing Gear Does Not Retract:**

3. Landing gear safety relay circuit breaker - Reset.

**If the Landing Gear Retracts:**

4. Monitor circuit breakers and, upon mission completion, land using normal procedures.

**If the Landing Gear Does Not Retract:**

5. Landing gear handle - DOWN.

6. Ground roll spoiler switch - OFF.

7. Landing gear safety relay circuit breaker - Pull.

All inflight functions are restored except that the landing gear will not retract, the ground roll spoilers are armed and the yaw channel caution lamp will light with 6 degrees of rudder deflection.

Note

The lamp will reset, but will relight if 6 degrees of rudder deflection is exceeded.

8. Land as soon as practical using "Landing With Landing Gear Ground Safety (Squat) Switch Failure" procedure, this section.

## LANDING GEAR RETRACTION MALFUNCTION

Note R

With the gear handle up and the gear handle R
warning lamp out, a nose or main gear posi- R
tion indicator lamp which remains on may be R
caused by frozen moisture in the gear position R
sensing switch. The mission may be contin- R
ued if a visual check confirms the gear are re- R
tracted. If the nose gear position lamp is on, R
ensure the landing/taxi light is turned off as it R
is also controlled by the nose gear position R
sensing switch. R

**If the Landing Gear Is Not Up and Locked (Red Warning Lamp On) 15 Seconds After the Landing Gear Handle Is Positioned to Up:** R

CAUTION

Do not place the utility system isolation switch to pressurize or recycle the gear until the position of the nose gear can be determined. A recycle with the nose gear not properly up and locked could cause damage to the nose gear, nosewheel steering, or the aircraft.

1. Speed Brake switch - Recycle to IN.

   Recycle speed brake switch to correct possible out-of-sequence condition.

2. Ground roll spoiler switch - OFF.

**WARNING**

A single landing gear squat switch failure may prevent landing gear retraction and arm the ground roll spoilers in flight.

3. Landing gear circuit breakers - Check.

4. Speed brake hydraulic valve circuit breaker - Pulled; then reset.

R **If Landing Gear Is Down and Locked (Two Green Lamps On):**

**Note**

The landing gear handle warning lamp will be on if the landing gear is not up and locked when the landing gear handle is in the UP position.

5. Land as soon as practical using "Landing with Landing Gear Ground Safety (Squat) Switch Failure" procedure, this section.

R **If Landing Gear Is Not Down and Locked (One or Both Green Lamps Out):**

6. Ground roll spoiler switch - BRAKE.

7. Obtain visual check if possible. If nose gear is not fully closed or if visual check is not possible, refer to "Nose Gear Retraction Failure Landing," this section.

**If Nose Gear Door Is Visually Checked Closed:**

8. Landing gear handle - DN.

   a. If cockpit indications are not normal, refer to "Gear Malfunctions," this section.

   b. If cockpit indications are normal, obtain a visual check of the landing gear doors, struts, steering linkages, and tires for proper extension, alignment, and security. A malsequence can result in damage to the nosewheel steering linkage. If any mechanical gear and/or steering abnormalities exist or are suspected, pull the emergency gear extension handle and consider an approach end cable engagement.

## TIRE FAILURE/FLAT MAIN STRUT DURING TAKEOFF

Directional control is not difficult with a blown tire if nosewheel steering and differential braking are used properly. The aircraft will lean significantly to the side of the blown tire and external stores may contact the ground. The brake on the good tire should be used normally. Do not lock the brake on the wheel with the blown tire. The first indication of a flat main gear strut will be a wing drop on the side of the flat strut. Directional control will be less difficult than with a blown tire and stores ground clearance will not present a problem.

**Note**

Nosewheel steering should be used to aid in directional control.

**If Decision Is Made to Stop, Refer to "Abort/Cable Engagement" Procedures, This Section.**

**If Takeoff Is Continued:**

1. Do not retract landing gear.

**CAUTION**

If gear is retracted with a blown tire, possible damage to the wheel well area may occur.

2. Instruments - Check.

   Check hydraulic, fuel and engine instruments to determine possible damage resulting from the disintegrated tire.

3. Refer to "Blown Tire/Flat Main Gear Strut/Wheel Failure Landing", this section.

## MISALIGNED PYLON/STORES

**WARNING**

This information is based on limited experience. Use caution to insure proper flight parameters (angle-of-attack, elevator position, etc.) are maintained at all times.

It is possible for the pivot pylons to be misinstalled so that the gears are improperly engaged. This will result in failure of the pylon and associated stores to remain parallel to the aircraft centerline in flight. Although the pylon/stores may rotate either toward or away from the fuselage, the failures experienced have usually resulted in the pylon pointing away from the fuselage on takeoff. It is possible for the malfunctioning pylon and associated stores to contact the fuselage or adjacent stores. Do not sweep the wings aft. When external stores are being carried on the affected pylon, the control difficulty experienced will depend on the size and number of stores present. If only an empty pylon is affected, control difficulties will be negligible. Expect slab split to increase as airspeed increases. Slab split will be approximately 4 degrees at normal landing speeds with gear, slats, and full flaps extended, and should impose no difficulty in landing. A controllability check is recommended prior to landing.

# INFLIGHT EMERGENCIES

## 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 2,000 feet above the terrain whenever possible.

**WARNING**

Do not delay ejection below 2,000 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 2,000 feet above the terrain.

If aircraft control is lost at low altitude, the minimum altitude required for ejection will increase dramatically due to the rapid changes in aircraft attitude. Therefore, the decision to eject must not be delayed. The EWO must be prepared to initiate ejection in situations where the pilot is attempting to maintain/retain aircraft control. To enhance survival opportunity, the crew must be familiar with the terrain elevation and general topography.

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 "Crew Module Ejection Sequence" and "Minimum Terrain Clearance For Ejection," see figures 3-3 and 3-4.

# CREW MODULE EJECTION SEQUENCE
## (Typical Low Speed Ejection)

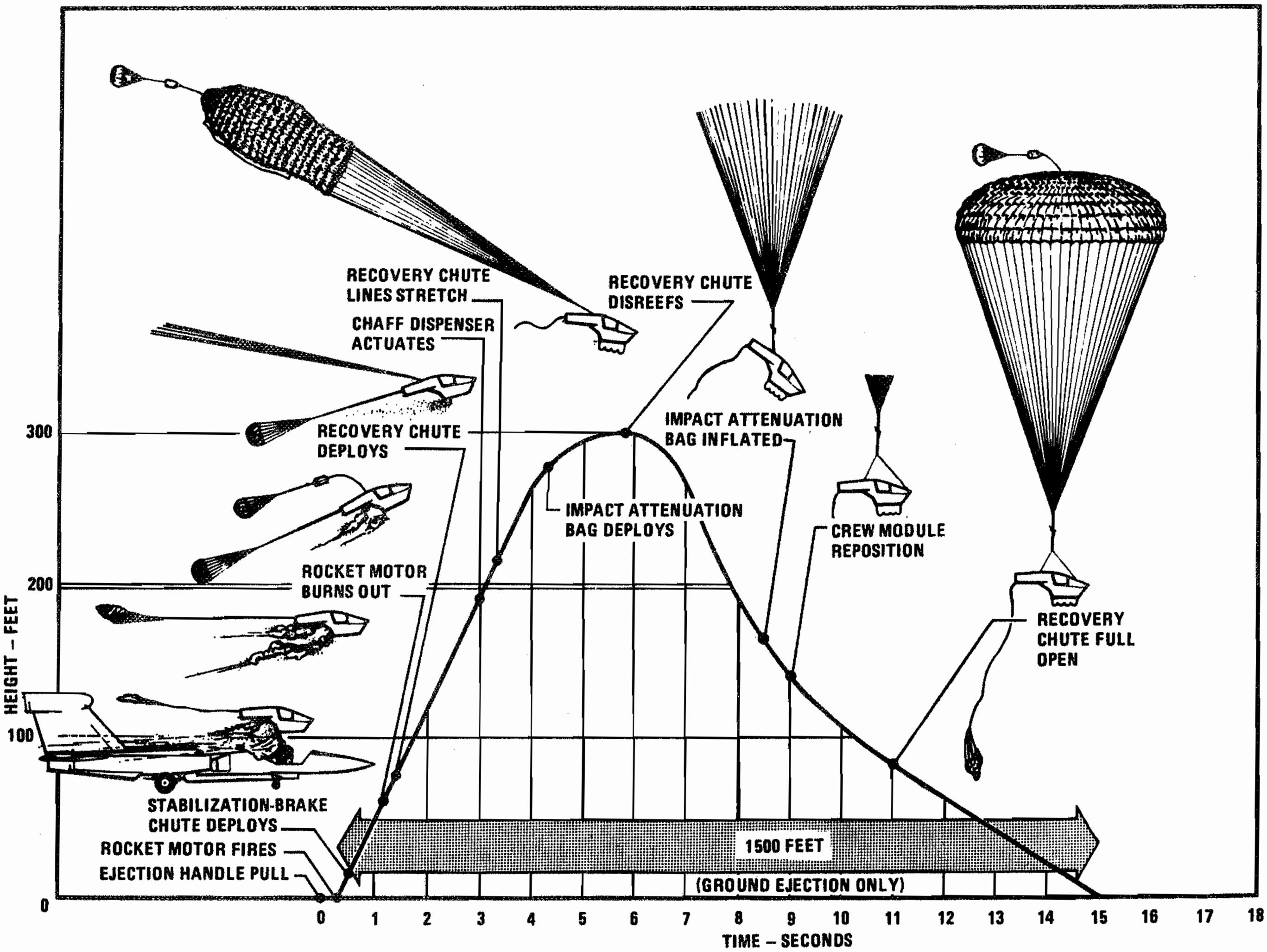



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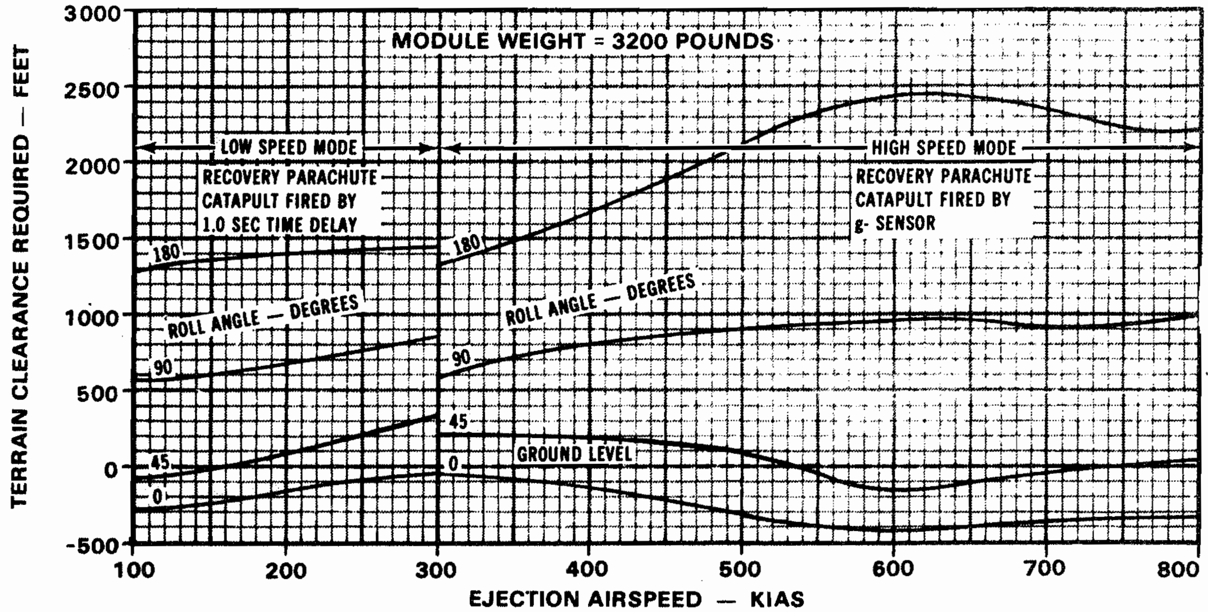
Figure 3-3.

# MINIMUM TERRAIN CLEARANCE FOR EJECTION

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

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.

**BANKED LEVEL FLIGHT**




**DIVE—WINGS LEVEL**

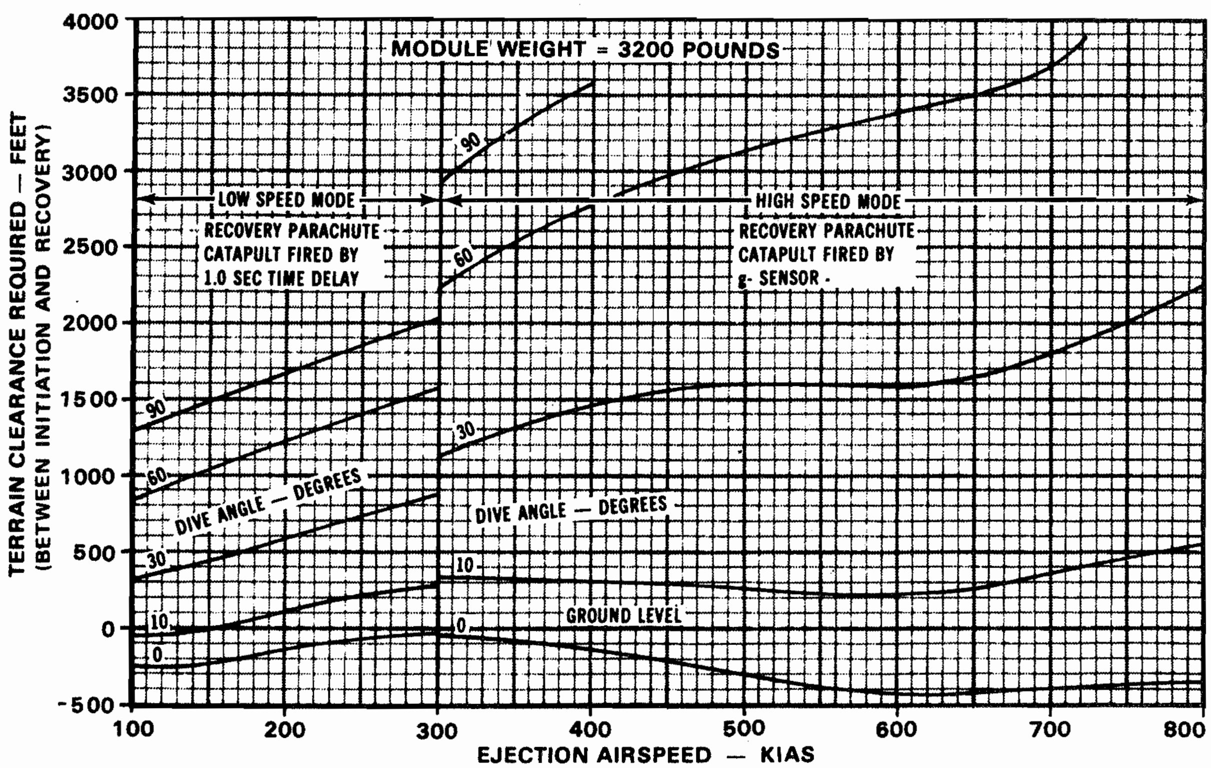



Figure 3-4.

WARNING

- Ejection is not recommended at zero altitude with less than 50 KIAS.
- 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. Advise crew member of situation.
2. Reduce airspeed. (As practical)
3. Transmit MAYDAY. (Give position)
4. IFF master control knob - EMER.
5. Oxygen mask and fittings - On and checked.
6. Oxygen - 100 percent. (EMER, if required)

WARNING

Wear the mask and use 100 percent oxygen throughout the ejection sequence.

**Note**

EMER or 100 percent must be selected on the oxygen control knob to receive pure oxygen to the oxygen mask. Selecting NORM on the oxygen control knob supplies a proportionally diluted air/oxygen mixture.

7. Inertia reel control - LOCKED. (Handle up)
8. Chaff dispenser control handle - ON. (OFF for tactical considerations only)

## EJECTION

Time permitting, assume ejection posture: Sit erect with head against headrest, buttocks against seat back with feet on rudder pedals (pilot) or footrests (EWO) and hands in lap.

WARNING

Do not change seat and/or headrest positions prior to ejection.

1. **EJECTION HANDLE - SQUEEZE AND PULL.**

WARNING

Expect a 0.35 of a second delay before the rocket fires. This delay allows the inertia reels to retract and lock the crew members in the upright position.

## DURING DESCENT

WARNING

If time permits no other action, assume optimum landing posture.

1. Seat pan - Adjust forward to optimum position.

   To optimize upper torso restraint, move the seat pan as far forward as possible without causing leg contact with the instrument panel. Keep buttocks firmly against the seat back.

2. Parachute deploy handle (when below 15,000 feet) - Pull. (If required)

**WARNING**

- The standby altimeter will be unreliable after ejection.
- Ejection above 15,000 feet: Terrain permitting, do not pull parachute deploy handle until below 15,000 feet to prevent parachute failure and/or premature oxygen depletion.
- Ejection below 15,000 feet: Do not attempt to "beat the system." Allow at least five seconds before pulling the parachute deploy handle to ensure adequate aircraft clearance and module deceleration.

3. Oxygen mask and fittings - On and checked.
4. Oxygen - 100 percent. (EMER, if required)

**Note**

EMER or 100 percent must be selected on the oxygen control knob to receive pure oxygen to the oxygen mask. Selecting NORM on the oxygen control knob supplies a proportionally diluted air/oxygen mixture.

5. Emergency oxygen handle - Pull. (If required)

**WARNING**

Wear the mask and use 100 percent oxygen until after landing and toxic gases have been vented from the module.

6. Assume optimum landing posture (figure 3-5).

Hold head against headrest with back against seat back, feet on rudder pedals or foot rests, and arms extended and braced with hands against upper legs.

**WARNING**

- Do not pull the severance and flotation handle. Activation prior to impact will cause flotation/self-righting bag rupturing and severe postlanding gyrations.
- Do not pull the parachute release handle. Activation would release the parachute and the module would then free fall.
- Canopy hatches must be closed and locked prior to impact. (If descent time permits, they may be temporarily opened after the parachute has deployed to vent smoke and fumes from the module.)
- Ensure that restraint harness is tight and confirm that shoulder harness is locked. Attempt manual locking only if automatic locking has not occurred.

## AFTER LANDING

**Note**

The right self-righting bag will start to inflate approximately 75 seconds after the severance and flotation handle is pulled. When the right hatch is not opened prior to inflation, the bag will overlap the hatch and must be deflated to fully open the hatch. Egress can be accomplished by opening the hatch partially and puncturing the bag or by exiting through the left hatch.

### Ground Landing

**WARNING**

Failure to depress the lock release buttons on the canopy center beam handles prior to exerting "pull" pressure may prevent movement of the handles.

1. Severance and flotation handle - Pull. (If required)

# Recommended Body Position For Ground Impact

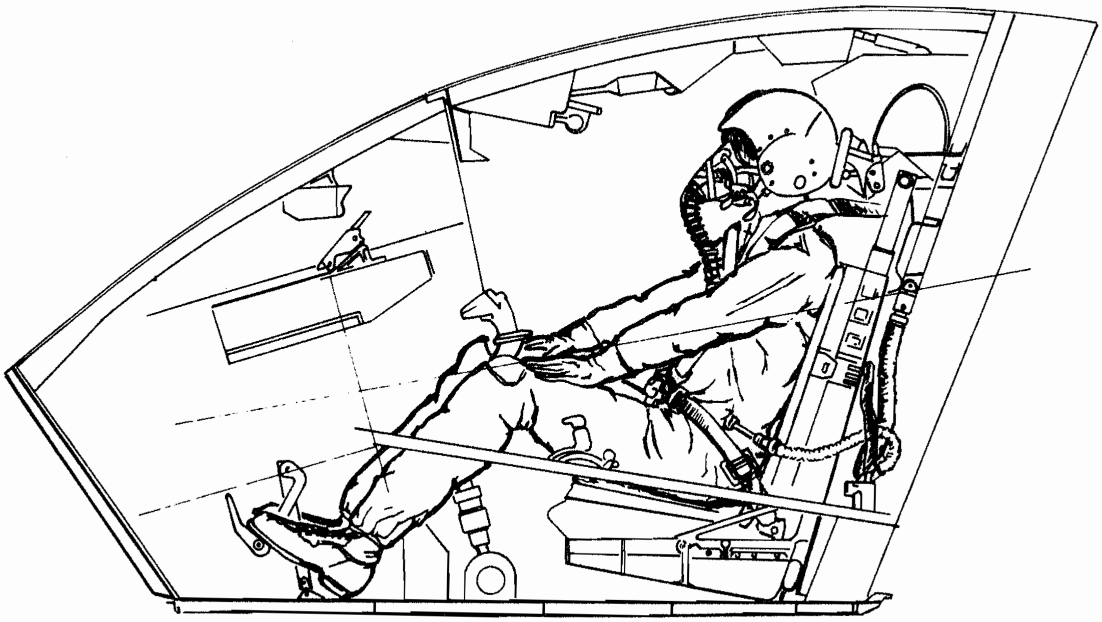


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Figure 3-5.

**WARNING**

Do not actuate the severance and flotation handle when personnel are within 50 feet of the crew module because of explosive severance of metal covers.

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 8,000 feet in an ejected crew module it is possible that atmospheric pressure differential on the canopy hatches will prevent them from being opened. To eliminate this pressure differential, deflate the canopy seals by either using the canopy quick-disconnect coupling located on the canopy center beam or remove the caps from the snorkel/ventilation ports and push in on the exposed tube. This should be done on the ground only after the module has come to rest.

**If Crew Module Lands Inverted:**

4. Inertia reel control handle - Cycle to relieve tension and loosen shoulder straps and lap belts as far as possible.

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**

Depending on sea conditions and wind, the capsule may roll inverted upon impact. Regardless of capsule attitude, the severance and flotation and parachute release handles should be pulled after landing. Pulling the severance and flotation handle will inflate the aft flotation bags and the left self-righting bag. Approximately 75 seconds after the handle is pulled the right self-righting bag will inflate. The capsule will then roll slowly to an upright attitude.

**WARNING**

If the capsule rolls inverted and flooding does not occur, the crew must remain strapped in until the module self-rights (due to the criticality of module center-of-gravity considerations).

Canopy hatches should be open to initially vent the capsule. If it becomes necessary to remain in the capsule for an extended period of time, both canopies should be opened once every three hours for ventilation. The bilge pump should be operated if the capsule begins to take on water. If the capsule takes on large amounts of water or if the capsule flotation is in question, pull the auxiliary flotation handle. The auxiliary flotation bag will inflate giving the capsule a slight nose high attitude plus additional flotation.

If the structural integrity of the crew module has been compromised, as in the case of glass panel failure due to bird strike, capsule flooding is probable. Self righting should occur approximately 75 seconds after actuation of the severance and flotation handle and the capsule will float despite its flooded condition. During the 75 second delay or if self righting malfunctions, it may become necessary for the crew to egress a submerged and flooded capsule. As soon as the module has rotated so it is in the upright attitude, the auxiliary flotation handle should be pulled. Even if the nose of the module is submerged, the auxiliary flotation system will raise the nose of the module above the water. The aft flotation bags along with the auxiliary flotation bag will float a flooded module.

**WARNING**

Failure to depress the lock release button on the canopy center beam handles prior to exerting "pull" pressure may prevent movement of the handles.

1. Severance and flotation handle - Pull.

2. Parachute release handle - Pull.

**WARNING**

Failure to release the parachute immediately after pulling the severance and flotation handle could inhibit the module from self righting.

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)

**CAUTION**

Failure to push in the plunger may result in damage to the pump system and render it inoperative.

5. Auxiliary Flotation handle - Pull. (If required)

**WARNING**

Activation of the auxiliary flotation handle prior to module self righting will inhibit the module from fully self righting.

**Note**

Survival two way radio antenna must be projected out of the module in order to function properly.

## 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. If pitot-static system speed inputs are erroneous and are less than 300 ($\pm$30) 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.

**WARNING**

- If ejection occurs in the low speed mode at actual aircraft speeds greater than 300 knots, high spinal loading, serious structural damage, and recovery parachute failure may occur.

- 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 270 knots. At aircraft gross weights up to 100,000 pounds and with the wings at 26 degrees, a 10 degree or greater angle-of-attack in one g stabilized flight will ensure that the airspeed is below 270 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.

If pitot-static system speed inputs are erroneous and are greater than 300 ($\pm$30) 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.

## ALTERNATE ESCAPE PROCEDURES

**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 rocket motor will not 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. Second ejection handle - Squeeze and pull.
2. Slow aircraft to below 300 KIAS and descend to below 20,000 feet if possible.
3. Inertial reel control handle - LOCKED.
4. While upright with the stick forward to induce negative "g's", pull the severance and flotation handle.

**WARNING**

Operation of the severance and flotation handle will unguard the parachute release handle. Pulling the parachute release handle will explosively release the recovery parachute and the module will free fall.

5. Passing 15,000 - Pull parachute deploy handle.
6. Refer to "Ejection, During Descent" procedures, this section.

**WARNING**

Pulling the severance and flotation handle does not disable the rocket motor; it may still fire. Consider the rocket armed when egressing the module.

**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.

## GLASS PANEL FAILURE OR UNLOCKED CANOPY INDICATION

In the event of a bird strike, the windshield may be cracked, but a catastrophic failure of the windshield will probably not occur. If penetration of the windshield does not occur upon initial bird impact, subsequent aerodynamic and structural loads will not cause further failure and the windshield will continue to provide protection from wind blast; however, cabin pressurization may be lost. After a bird strike, aerodynamic flight loads should be reduced by depressurizing the cabin and reducing airspeeds. An airspeed of 200 to 300 knots and an altitude above 7000 feet are recommended. Complete 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. The aircraft commander should level the wings, reduce airspeed as rapidly as possible consistent with aircraft control, and climb/descend to a safe flight level. Retention of the helmet is a primary consideration, since its loss intensifies the severity of the situation. If the crew member is sitting low and out of the wind blast, every effort should be made to stay there. Experience has shown that a crew member exposed to direct wind blast for even a few seconds suffers a high probability of helmet loss and severe disorientation.

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 - 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 practical.

## ENGINE EMERGENCIES

### AFTERBURNER FAILURE

An afterburner blowout will be indicated by a significant decrease in fuel flow and the nozzle indicator returning to zero. It may be accompanied by a momentary audible "bang" similar to an engine stall. If the TIT goes out of limits or the audible "bang" persists, refer to "Engine Stall" procedures, this section. Immediately after an afterburner blowout, the fuel control will schedule a fuel flow and rpm reduction to compensate for engine internal pressure changes. During this rpm reduction, the automatic restart ignition is inhibited. When rpm has stabilized, which may require as much as 20 seconds, afterburner operation may be regained by cycling the throttle to MIL and back to AB.

**Note**

Afterburner blowouts and associated rpm reduction may occur as the throttle is retarded to MIL. A normal afterburner termination does not cause an rpm reduction.

### DOUBLE ENGINE FAILURE

R **Note**

R Maintain airspeed of approximately 270
R KIAS. This speed will assure sufficient rpm for
R both a successful airstart and adequate hy-
R draulic pressure to maintain the aircraft in a
R controlled descent.

1. Battery switch - Verify on.
2. Airstart button - Depress.
3. Throttles - As required.
4. Fuel panel - Checked.
5. If airstart cannot be accomplished - Eject.

### 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. If excessive engine vibration is present, fire pushbutton - Depress.

**WARNING**

- Excessive engine vibration may cause rupture of the fuel and hydraulic lines where they connect to the engine with resultant fire and fuel loss hazards. Use of the fire pushbutton will minimize these hazards by isolating the engine from the fuel, hydraulic and coolanol lines.
- Use caution to prevent inadvertently depressing the wrong pushbutton and shutting down the good engine. When a fire R
  pushbutton is fully depressed and latched, R
  or if it does not latch, but was held in for at R
  least 4 seconds, the applicable fuel shutoff R
  valve, primary and utility hydraulic shutoff R
  valves, and the JSS coolant shutoff valve R
  will remain closed and cannot be reopened R
  in flight. R

3. Air source selector knob - OFF or EMER. (If required)

**WARNING**

If the engine failure is accompanied by smoke in the cockpit or various caution/warning lamps lighting that are unrelated to the engine failure, turn the air source selector knob to OFF or EMER.

**CAUTION**

If the air source selector knob is repositioned after initial selection, the shutoff valve may fail to an open position.

**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.

4. Land as soon as possible using "Single Engine Landing" procedures, this section.

## SINGLE ENGINE FLIGHT CHARACTERISTICS

Single engine flight characteristics are essentially the same as 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 (clean aircraft with failed engine windmilling) or 255 KIAS (clean aircraft with locked rotor), with 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.

## ENGINE STALL

Refer to Section VI for detailed discussion of the characteristics of engine stalls. It should be noted that mechanical vibration is not a stall characteristic. In this situation, the procedure for Single Engine Failure, Mechanical Failure, this section should be followed. If an engine stalls and the engine must be shut down to clear the stall, the aircraft should be landed as soon as practical unless mission requirements dictate otherwise. In the event of engine stall on one or both engines, proceed as follows:

**WARNING**

If both engines are stalled, one engine must be shutdown immediately to provide maximum potential for a successful airstart.

1. Throttle of affected engine(s) - Retard smoothly to IDLE. (Conditions permitting)

2. Unload aircraft (0.5 to 1.0g).

**Note**

Unloading the aircraft provides a more normal engine inlet airflow.

3. When RPM stabilizers - Set power as desired.

**Note**

In the event that an afterburner blowout occurs following an engine stall, if the RPM recovers, afterburner operation can be regained after the RPM has stabilized by cycling the throttle to MIL and back into afterburner.

### If Stall Does Not Clear or RPM Does Not Respond to Throttle Movement:

4. Shut down stalled engine (only one). Note that RPM is decreasing.

5. Perform airstart and set power as desired.

### If Stall Is Not Cleared By Above Procedure:

6. Shut down stalled engine a second time. Note that RPM is decreasing.

7. Decelerate to below Mach 0.90 or 415 KIAS, whichever is less.

8. Spike control switch - OVERRIDE.

Place the affected engine spike control switch to OVERRIDE. The stall may have resulted from an improperly positioned spike.

**Note**

Momentary lighting of either spike caution lamp indicates spike movement to the full forward and contracted position. Once override has been selected, normal spike operation will be inhibited for the duration of the flight, and airspeed is restricted to mach 0.90 or 415 KIAS, whichever is less.

9. Perform airstart and set power as desired.

**If Stall Is Not Cleared By Above Procedure:**

10. Refer to "Single Engine Failure, Mechanical Failure," this section.

## AIRSTART

Satisfactory airstarts may be accomplished throughout the flight envelope. Airstarts initiated as soon as possible will assure the best possible conditions for restart. Crossbleed may be required to obtain a restart from low compressor RPM. Approximately 17 percent RPM is required for ram airstarts.

**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.
- If the throttle has been retarded below idle, selecting CARTRIDGE on the engine ground start switch provides an alternate source of ignition for the left engine.
- A hot start may occur if restart is attempted with TIT above 400 degrees centigrade. In this event, retard throttle of the affected engine to OFF and allow TIT to decrease below 400 degrees before attempting another restart.
- The time required to achieve an airstart using JP–8 fuel may be slightly longer than experienced using JP–4. In addition, a higher airspeed or descent to a lower altitude may be required. Restart should be initiated before compressor RPM decay or combustion section cooldown.

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. Generator switch of affected engine - OFF/RSET.
4. Affected engine rpm - Checked.
   a. If rpm is 17 percent or above, proceed to step 6.
   b. If rpm is below 17 percent, proceed as follows:
5. Engine ground start switch - PNEU.

Crossbleed air is available regardless of position of air source selector knob.

6. Throttle of affected engine - Lift.
7. Throttle of affected engine - Idle. (At or above 17 percent rpm)
8. Airstart ignition button - Depress momentarily. (Check for relight within 20 seconds.)

**If Engine Starts:**

9. Pitch damper switch - OFF. (Within damper off region)
10. Generator switch of affected engine(s) - ON.

If the generator caution lamp remains lighted, place generator switch to OFF/RSET, then ON and check that the lamp goes out and the power flow indicator reads NORM.

11. Pitch damper switch - DAMPER.

**If Engine Does Not Start:**

12. Throttle of affected engine - OFF.
13. Engine ground start switch - OFF.

14. Attempt additional airstarts - As desired.

15. Land as soon as possible using "Single Engine Landing" procedures, this section.

## OIL SYSTEM MALFUNCTIONS

An oil system malfunction on either engine is recognized by a change in oil pressure, fluctuating oil pressure, complete loss of oil pressure, or excessive oil temperature. In general, it is advisable to land the aircraft as soon as possible when the oil pressure below 15 psi or when the oil temperature is excessive, to minimize the possibility of damage to the engine. If thrust is critical, the engine may be utilized as long as it continues to produce power.

### OIL PRESSURE BELOW 15 PSI

1. Throttle of affected engine - OFF. (If flight conditions permit)

**CAUTION**

If oil pressure goes to below 15 psi and it is necessary to keep the engine operating to sustain flight, engine seizure can be expected.

2. Land as soon as possible, using "Single Engine Landing" procedures, this section.

### OIL PRESSURE BETWEEN 15 AND 40 PSI (EXCEPT AT IDLE)

1. Throttle of affected engine - IDLE.

2. Monitor oil pressure.

3. Land as soon as practical, using "Single Engine Landing" procedure, this section.

### OIL PRESSURE ABOVE 50 PSI

1. Throttle of affected engine - Retard.

2. If oil pressure can be maintained below 50 psi, continue to operate the engine at a reduced power setting and land as soon as practical.

3. If oil pressure cannot be reduced below 50 psi, retard throttle of the affected engine to idle and land as soon as practical, using "Single Engine Landing" procedure, this section.

## 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

1. Relax force on throttle for 20-30 seconds.

2. Attempt slight throttle advance; then apply force to retard throttle.

**WARNING**

Care should be used when abnormal force is needed to retard throttle(s) in order to avoid inadvertantly shutting down the engine(s).

**Note**

Repeating the above procedure several times may free the frozen throttle.

3. If the frozen throttle is freed, it should be retarded to idle and no further attempts made to move the throttle unless thrust is critical.

4. Land as soon as practical using "Single Engine Landing" procedures, this section.

### ENGINE SHUTDOWN WITH A FROZEN THROTTLE

If a throttle is frozen, aircrew actions will be dictated by runway length, aircraft gross weight, fuel remaining and power setting of the frozen throttle. If the throttle is frozen at a setting which will make airspeed control during landing difficult, or compromise stopping distance after landing, the affected engine should be shut down by using the fire pushbutton and the aircraft landed using "Single Engine Landing" procedures, this section.

**WARNING**

- If the decision is made to shut down the affected engine, depress the fire pushbutton prior to starting final approach.
- If the throttle is frozen above the idle position, ground roll spoilers will not be available on landing roll. Angle-of-attack should be controlled closely on final approach to ensure that the aircraft is landed on-speed. Failure to observe this precaution may result in the aircraft becoming airborne after touchdown, a pilot induced oscillation, and loss of aircraft control.

**Note**

If landing is made with both engines operating and stopping distance is marginal, the throttle on the good engine may be retarded to OFF after landing.

1. Fire pushbutton - Depress.

**Note**

- Consideration should be made to configure the aircraft prior to shutting down the engine.
- The engine may be shut down from any power setting.
- At low rpm the engine may continue to run for over 30 seconds after the fire pushbutton is depressed.

2. Land as soon as possible using "Single Engine Landing" procedures, this section.

## 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. Failure of the spike system to maintain the full forward and contracted position at speeds below the transonic region may be indicated by the applicable spike caution lamp and/or a significant loss of thrust from the affected engine. A chase aircraft may confirm a failure at airspeeds below the transonic region.

### Suspected Spike Failure (Supersonic)

1. Airspeed - Reduce until inlet buzz or compressor stall disappears.

### Suspected Spike Failure (Subsonic)

1. Spike control switches (left and right) - OVERRIDE. (Below Mach 0.9)

R **Note**

- Momentary lighting of either spike caution lamp indicates spike movement to the full forward and contracted position. Once override has been selected, normal spike operation will be inhibited for the duration of the flight, and airspeed is restricted to mach 0.90 or 415 KIAS, whichever is less. R
- Do not attempt to return the spike control switches to NORM after they have been placed to OVERRIDE.

2. Airspeed - Maintain Mach 0.9 or below.

## TRANSLATING COWL WARNING LAMP LIGHTED

Lighting of this lamp indicates that either cowl is not fully open and one of the following conditions exist:

- Mach below 0.35.
- Mach below 0.44 for 15 seconds.
- Landing gear handle is down.
- Landing gear emergency release handle is pulled.

1. Translating cowl switch(es) - OPEN.

**If Cowl(s) Fail to Open:**

2. Translating cowl emergency override switch - OVRD. (OPEN)

**CAUTION**

Do not reposition the translating cowl emergency override switch to NORM if OVRD (OPEN) has been selected at airspeeds less then Mach 0.50. The cowl actuating system will not automatically revert to normal hydraulic operation subsequent to actuation of the emergency pneumatic mode. Repositioning of the override switch to NORM will result in venting of the applied pneumatic pressure and loss of positive restraint holding the cowls in the open position, if the cowls have been pneumatically actuated. The presence of this condition during landing approach could result in inadvertent closure of the cowls.

**Note**

The function OVRD (OPEN) switch is disarmed at speeds above Mach 0.50.

3. Translating cowl position indicators - OPEN.
4. If cowl(s) still do not open, follow the procedures for "Translating Cowl(s) Closed Landing," this section.

# ICING

## 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**

R 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 extending the flaps 15 degrees or more. Do not use angle-of-attack indexers.

## ENGINE INLET ICING

If icing conditions are encountered and the icing lamp has not lighted, assume ice detector failure and select MAN on engine/inlet anti-icing switch until clear of icing condition.

# FIRE DURING FLIGHT

## SMOKE AND FUME ELIMINATION

**WARNING**

Smoke from the air-conditioning vents may indicate an impending catastrophic engine failure. Such failures have resulted in aircraft losses. If the source of the smoke can be isolated to an engine, that engine should be shut down as soon as possible.

1. Oxygen - 100 percent.
2. Oil quantity and pressure indicators - Check.

   A low oil quantity or abnormal oil pressure indicates the probable source of engine smoke.

3. Air source selector knob - L ENG, R ENG.

**Note**

Attempt to determine if an engine is the source by selecting each engine on the air source selector knob and waiting for 30 seconds to see if the smoke dissipates. Do not change engine settings during this check, since increased power settings may prevent further smoke from entering the 16th stage bleed air system and mask a serious engine malfunction.

**If the Source Is Isolated to an Engine:**

4. Shut down the engine if flight conditions permit. If not, leave the engine in idle.

**Note**

The probability of catastrophic engine failure is reduced at idle power rpm.

5. Land as soon as possible using "Single Engine Landing" procedures, this section.

**If the Source Cannot be Isolated to an Engine:**

6. Air source selector knob - OFF or EMER. (As applicable)

   For supersonic flight with high total temperature indications, place the air source selector knob to OFF, decelerate and descend to decrease aircraft skin temperature, and establish an altitude at which cabin pressure is not required. If temperature and/or altitude is not a consideration, place the knob to EMER. Refer to "Ram or Emer Mode Flight Limits," Section V.

7. Non-Essential electrical equipment - Off.

   If smoke dissipates after turning off electrical equipment, leave equipment off unless needed for flight. If necessary, turn equipment on one system at a time and check for smoke.

8. Land as soon as practical.

## ENGINE FIRE DURING FLIGHT

With any engine fire indication, immediate and positive steps must be taken to shut off fuel, hydraulic, and coolanol supply and extinguish the fire. The following procedure contains the most desirable sequence of steps for most engine fire situations. It assumes that the engine indicating fire can be identified promptly, without distracting the aircrew from the essential task of maintaining aircraft control. Under these conditions, closing the throttle prior to depressing the fire pushbutton will reduce engine rotation faster, will decrease the possibility of further damage in case of mechanical failure, and remove sources of re-ignition quicker. Certain engine mechanical failures could result in turbine fan blades cutting hydraulic lines. If this occurs, do not delay pushing the fire pushbutton after closing the throttle. A delay could result in the hydraulic pump depleting the system in a matter of seconds. Under some conditions it may be difficult for the aircrew to determine which engine is indicating fire. If any doubt exists, the lighted fire pushbutton should be depressed immediately and the corresponding throttle closed when conditions permit. In either case it is essential that the engine be shut down (throttle in cutoff and fire pushbutton depressed) prior to activating the agent discharge system. It is essential that the fire pushbutton be fully depressed so it latches. If it fails to latch, the agent discharge switch will not be armed, but the fuel shutoff valve will remain closed if the button was fully depressed for at least 4 seconds. The fuel flow indicator on the affected engine may not provide a reliable indication that the fuel shutoff valve has closed. Failure to observe this sequence may result in re-ignition of the fire and subsequent loss of the aircraft.

Certain engine malfunctions can cause the throttles to freeze. If this occurs, immediately proceed with the remaining checklist steps. Continued attempts to retard the throttle will result in a critical delay in engine shutdown. In this situation, delay placing the agent discharge switch up until rpm has started to decrease. This should aid in removing sources of re-ignition.

**WARNING**

- Delay in accomplishing the critical steps of the following procedure may allow the fire to damage critical systems with subsequent loss of aircraft control.
- Use caution to prevent inadvertently retarding the wrong throttle or depressing the wrong pushbutton and shutting down the good engine.
- With the fire pushbutton depressed and latched, depressing the fire pushbutton a second time will unlatch the button and disarm the fire extinguishing agent discharge valve. If the fire pushbutton does not latch, hold it fully depressed for at least 4 seconds to close the fuel shutoff valve and until the agent discharge switch is positioned to AGENT DISCH.
- The fire extinguishing agent is available for one actuation only. If the agent discharge is activated before the throttle is closed, it is probable that the fire will be re-ignited by the engine. If this occurs, no further means exists to extinguish the fire.
- If the fire lamp goes out before any of the following steps are accomplished, or if the fire lamp goes out before the procedure is completed, it is imperative that the remaining steps be accomplished to ensure that the fire has been extinguished and the possibility of it reoccurring is reduced.

It is possible for a catastrophic engine failure to rupture the 16th stage bleed duct in the main wheel well. Hot air escaping from the ruptured bleed duct can cause considerable damage before the WHEEL WELL HOT caution lamp comes on. Indications of bleed duct failure are heavy acrid smoke in the cockpit or various caution/warning lamps lighting that are unrelated to the engine fire.

1. **THROTTLE - OFF.**
2. **FIRE PUSHBUTTON - DEPRESS.**

**CAUTION**

Operation of the JSS transmitters with an engine fire pushbutton depressed may cause transmitter overheat.

3. **AGENT DISCHARGE SWITCH - UP.**

In some cases it may take from 5 to 10 minutes for hydraulic system pressure to decay enough to light the primary and utility hydraulic caution lamps of the engine that has been shut down.

4. Air Source selector knob - OFF or EMER. (If required)

**WARNING**

If the engine fire is accompanied by smoke in the cockpit or various caution/warning lamps lighting that are unrelated to the engine fire, turn the air source selector knob to OFF or EMER.

**CAUTION**

If the air source selector knob is repositioned after initial selection, the shutoff valve may fail to an open position.

**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 air scoop will result in excessive cabin temperature.

5. If fire is confirmed and continues - Eject.

**If Fire Is Extinguished:**

6. Land as soon as possible, using "Single Engine Landing" procedures, this section.

## FUSELAGE FIRE DURING FLIGHT.

A fuselage fire or equipment overheat condition may cause the fuselage fire warning lamp to light.

1. **FIRE PUSHBUTTON - DEPRESS.**
2. **AGENT DISCHARGE SWITCH - UP.**

**Note**

The fuselage fire warning lamp may not go out immediately after discharging the fire extinguishing agent.

3. Air source selector knob - OFF or EMER.

**Note**

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. JSS liquid pumps switch - NORM.
5. JSS power switch - OFF.
6. Master radiate - OFF.
7. Jammer radiate switches (10) - OFF.
8. If fire is confirmed and continues - Eject.
9. If fire ceases - Land as soon as possible.

## ENGINE BLEED AIR DUCT FAILURE/OIL HOT LAMP

The aircraft has a detection system that should cause the L or R bleed duct fail caution lamp to light if a bleed air duct rupture occurs in the engine compartment. There are situations where the oil hot lamp is the only indication of an engine bleed air duct failure. If either the L/R oil hot or L/R bleed duct fail caution lamps light, proceed as follows:

1. **THROTTLE - IDLE** (On affected engine)
2. **AIR SOURCE SELECTOR KNOB - OFF or EMER.** (As applicable)
For supersonic flight with high total temperature indications, place the selector knob to OFF, decelerate and descend to decrease total temperature indication, and establish an altitude where cabin pressure is not required. If temperature and/or altitude is not a consideration, place the knob to EMER.

**CAUTION**

If the air source selector knob is repositioned after initial selection, the shutoff valve may fail to an open position.

**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.

3. Leave the affected engine in IDLE and land as soon as possible using "Single Engine Landing" procedures, this section. The affected engine may be used only if required to sustain flight. If only an oil hot lamp is lighted, with other indications of an oil system malfunction, refer to "Oil System Malfunctions," this section.

**WARNING**

If an actual bleed duct failure exists, power settings above idle may cause extensive fire/heat damage.

## WHEEL WELL OVERHEAT

The most probable cause of a wheel well overheat condition is 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 crew corrective 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. If the caution lamp goes out, a visual inspection should be made of the wheel well and the surrounding area (by chase aircraft or tower fly-by) and the aircraft should be landed as soon as possible. If the lamp does not go out, leave the landing gear down and land as soon as possible. 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 temperature and/or altitude is not a consideration, place the knob to EMER.

**CAUTION**

If the air source selector knob is repositioned after initial selection, the shutoff valve may fail to an open position.

**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. **SPEED BRAKE - EXTEND.**

3. Landing gear - Extend. (If conditions permit)

The landing gear should be extended as soon as possible if aircraft range/performance is not a factor. Lowering the landing gear will continue to provide ventilation in the wheel well area while inducing less drag than a fully extended speed brake. If range/performance is a factor, the landing gear may be left in the up position and drag reduced by partially retracting the speed brake.

**WARNING**

If a wheel well overheat is confirmed or suspected, the landing gear should be extended as soon as airspeed permits. Failure to extend the landing gear may result in a tire fire/explosion which may result in loss of the aircraft.

**If The Lamp Goes Out:**

4. Leave the landing gear down if situation permits and land as soon as possible.

5. An approach end cable engagement should be considered due to possible main landing gear damage.

**If The Lamp Does Not Go Out:**

6. Throttle - Idle on affected engine.

   a. With the landing gear down, retard the throttles to idle one at a time and monitor the wheel well hot caution lamp. If lamp does not go out in 3 to 5 seconds, try the other throttle. If lamp goes out with one of the throttles in idle, leave that throttle in idle and land the aircraft using "Single Engine Landing" procedures, this section. The engine bleed air check and shutoff valves are electrically controlled and pneumatically operated. In the event of damage to the electrical system, the circuit breakers may open, causing one or both of the valves to reopen. Reducing power on the affected engine may reduce the airflow sufficiently to prevent further damage.

7. Leave landing gear down and land as soon as possible.

8. An approach end cable engagement should be considered due to possible main landing gear damage.

## AIRCRAFT CONTROL RECOVERY PROCEDURES

**Note**

It may not be necessary to accomplish the complete procedure to regain aircraft control.

### UNSCHEDULED PITCH MANEUVER

This procedure should alleviate an unscheduled pitch maneuver or correct a failure of the aircraft to respond to a pitch input. An unscheduled pitch maneuver may or may not be abrupt. It will be characterized by inability to hold a constant pitch attitude or angle-of-attack. Under most conditions, an unscheduled pitch maneuver can be controlled by stick inputs.

**WARNING**

- Maximum effort by the pilot may be required to move the stick to the desired position if a galled hydraulic servo actuator control valve caused the unscheduled maneuver. If a galled control valve is suspected, minimize control inputs after recovery from the unscheduled maneuver.
- If control cannot be regained by 15,000 feet AGL, eject.

**Note**

Accomplish the first 3 steps immediately and simultaneously.

1. Stick - Control pitch attitude. (Use maximum stick force if required)
2. Autopilot release lever - Depress and hold.
3. Throttles - As required.

   Unscheduled pitch maneuvers that increase pitch attitude will normally require MIL or MAX AB power to maintain airspeed. A decrease in pitch attitude may require a power reduction.

4. Stick trim - As required.
5. Aux pitch trim switch - OFF.

**Note**

If a parallel trim problem exists, it may take up to 5 seconds for the parallel trim to drive to neutral.

6. Aux pitch trim switch - NOSE UP or NOSE DN. (As required)
7. Flight control disconnect switch - OVRD.
8. If maneuver is present and persists, pitch damper - OFF. (Within pitch damper off limits, if practical)

   Be prepared to adjust pitch inputs. Slippage of the wedge blocks located in the flight control/crew module disconnect system will allow up to two inches of control stick movement prior to the aircraft responding in the pitch axis.

9. If controllability does not improve, pitch damper - DAMPER.
10. Perform aircraft controllability check.

11. Land as soon as possible from a straight-in approach. Use "Dampers Off Landing" procedures with any or all of the dampers off. Consider an approach end cable engagement due to the possibility of uncommanded control surface movement during touchdown.

If the flight control disconnect switch is in OVRD, pitch and roll gain changer lamps will remain lighted even though slats are extended and AYC will be inoperative.

## UNSCHEDULED ROLL/YAW MANEUVER

This procedure should alleviate an unscheduled roll/yaw maneuver, or correct a failure of the aircraft to respond to roll/rudder inputs. An unscheduled roll/yaw maneuver may or may not be abrupt. It will be characterized by inability to hold constant heading, lateral acceleration build-up and increasing lateral input to maintain wings level. Under most conditions, an unscheduled roll/yaw maneuver can be controlled by a combination of rudder and lateral control inputs. Use all cockpit indications (visual and physical cues, turn and slip indicator, heading indicator and control surface position indicator) to determine if the unscheduled maneuver was caused by a malfunction in the roll or yaw control axis. Failures in the wing sweep system causing a sweep asymmetry can also result in an unscheduled roll maneuver which may or may not be abrupt and may follow a wing sweep change command. The aircraft will roll towards the wing having the greatest sweep. Small wing sweep asymmetries can be controlled with lateral control inputs, but the amount of control required will increase with g's. For this reason, use of rudder to sideslip so as to reduce the effective asymmetry is the best technique. Under some failure conditions of the sweep mechanism, the wing sweep handle can be used to reduce the asymmetry and hence reduce the roll-off tendency. Apply forward sweep handle for left roll-off, aft for right roll-off.

**WARNING**

- When recovering from an unscheduled roll/yaw maneuver, monitor angle-of-attack and do not exceed allowable limits.
- Abrupt forward stick movement during rolling maneuvers below 20 degrees angle-of-attack may result in uncommanded roll rate increases in excess of 200 degrees per second (roll coupling).
- If control cannot be regained by 15,000 feet AGL, eject.

**Note**

Accomplish the first 4 steps immediately and simultaneously.

1. Stick - Unload the aircraft. (As required)

   Move the stick as necessary to maintain angle-of-attack within limits. Lateral inputs must be kept to a minimum since they will be detrimental if the aircraft is roll-coupled or stalled.

2. Autopilot release lever - Depress and hold.

3. Rudder - Opposite roll/yaw.

**WARNING**

Immediately apply rudder to center the ball on the turn and slip indicator. This may require full rudder displacement. Avoid inducing large side slip angles while attempting to regain control.

4. Throttles - MIL or MAX AB.

   An increase in power setting is normally required since abrupt roll/yaw maneuvers cause a rapid reduction in airspeed.

**If AOA is Above 20 degrees Refer to "Stall Prevention/Recovery Procedure," This Section**

**If AOA is 20 degrees or less:**

5. Stick - Against Roll. (Use maximum stick force if required)

   Use lateral control to stop the roll rate and resume the desired attitude. This may require full stick displacement.

6. Flight control disconnect switch - OVRD.

   Be prepared to adjust control inputs.

7. Wing sweep - Forward to obtain spoilers, within airspeed limits, unless asymmetric wing sweep is suspected.

   If sweep asymmetry is not suspected, sweep the wings forward of 45 degrees to provide spoilers to assist in controlling the roll or yaw maneuver. Between 50 and 45 degrees, sweep the wings slowly in anticipation of a possible abrupt lateral transient when the spoilers are activated, especially if a large stick displacement is being held to counteract the maneuver.

   If roll-off occurred following a wing sweep change command, return the wing sweep control handle toward its original position until balance is restored. If wing drift is suspected, place the wing sweep control handle full forward for left roll-off of full aft for right roll-off.

8. Rudder authority switch - FULL. (If required)

   Be prepared to adjust pedal inputs.

9. Rudder trim - As required.

10. Perform aircraft controllability check.

11. Land as soon as possible from a straight-in approach. Consider an approach end cable engagement due to the possibility of uncommanded control surface movement during touchdown.

   If wing sweep asymmetry has occurred and one wing has been found to drift relative to the other, it is probable that the drift will continue until full forward or full aft wing sweep is reached. In this event, attempt to control the wings such that full forward or full aft wing sweep is reached before attempting to land.

   If flight control disconnect switch is in OVRD, pitch and roll gain changer lamps will remain lighted even though slats are extended and AYC will be inoperative.

## STALL PREVENTION/RECOVERY PROCEDURE (AOA ABOVE STALL WARNING LIMITS)

**WARNING**

The immediate action that must be taken when an aircrew realizes that the aircraft has exceeded angle-of-attack limits is to reduce AOA to below 14 degrees. Positive forward control stick pressure must be applied. Merely releasing the back pressure will not necessarily lower the AOA due to command augmentation.

Detailed stall 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. 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. During maneuvering flight, the departure may appear as a failure of the aircraft to respond to a change in roll input.

1. **STICK - FORWARD AND CENTERED.**

   Move the stick smoothly forward to reduce AOA below 14 degrees. Full forward stick will not normally be required.

   **WARNING**

   Abrupt forward stick movement during rolling maneuvers below 20 degrees angle-of-attack may result in uncommanded roll rate increases in excess of 200 degrees per second (roll coupling).

2. Autopilot release lever - Depress and hold.

   **Note**

   The autopilot release lever should be depressed simultaneously with the forward stick movement.

3. Rudder - Neutral.

**WARNING**

The forward stick and neutral rudder recovery inputs must be given time to be effective. Maintain these inputs until a spin or recovery is identified.

**If Aircraft Control Is Lost (AOA Above 20°):**

4. Roll damper switch - OFF.

**WARNING**

Action of the roll damper after departure can delay recovery of aircraft control.

5. 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 14 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 22 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. It may be necessary to obtain an airspeed of as much as 300 KIAS to recover a stalled engine.
- High descent rates (up to 20,000 feet per minute) may exist during stalls/spins.
- If control cannot be regained by 15,000 feet AGL, eject.
- If the aircraft departs controlled flight at altitudes less than 15,000 feet AGL, sufficient altitude may not be available to attempt recovery and then eject. As the terrain clearance decreases, time available to make and execute an ejection decision decreases. At low altitudes, immediate ejection may be the only alternative once the aircraft has departed controlled flight.

**Note**

If the first five steps of this procedure do not produce a recovery, the aircraft is probably spinning. An upright spin is indicated by angles-of-attack above 22 degrees but erroneous low angle-of-attack indications may occasionally appear. Inverted spins will indicate a minus 2 or 3 degrees angle-of-attack. A developed spin is characterized by the ground sweeping horizontally across the aircrew's forward field of vision, low airspeed (140 KIAS or lower), and full turn needle displacement in the direction of rotation.

**If Spinning (AOA Above 22° or Below −2°):**

6. Stick - Full with turn needle and forward. (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.

7. Rudder authority switch - FULL.

8. Rudder - Full opposite turn needle.

**When Aircraft Rotation Stops:**

9. Stick - Forward and centered. (Pitch and roll control centered, if inverted)

**WARNING**

Abrupt forward stick movement during rolling maneuvers below 20 degrees angle-of-attack may result in uncommanded roll rate increase in excess of 200 degrees per second (roll coupling).

10. Rudder - Neutral.

**WARNING**

- Immediately after rotation stops, the aircraft will unload and negative g's may be encountered. This unloading 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 14 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 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 controls is a building airspeed above 200 KIAS and an angle-of-attack well below stall. Failure to neutralize roll and yaw controls following spin recovery will result in an excessive altitude loss.

- Do not exceed allowable angle-of-attack limits during recovery.

- If control cannot be regained by 15,000 feet AGL, eject.

**If Rolling Rapidly During Stall/Spin Recovery**

**(Above 200 KIAS and AOA below 20°):**

11. Stick - Neutral.

    Inertial coupling may cause uncommanded roll rates in excess of 200° per second if roll input is not centered.

12. Rudder - Opposite Roll. (As required)

    Application of opposite rudder will reduce roll rate. As the roll rate slows, neutralize rudder.

**If Recovery Is Effected:**

13. Roll damper switch - DAMPER.

14. Throttles - As required.

15. Air start button - Depress.

16. Rudder authority switch - AUTO. R

17. 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.

## HARD-OVER RUDDER

Hard-over rudder deflection can occur as a result of aft fuselage fires or failure of the rudder linkage resulting in full right (30°) rudder deflection. If hard-over rudder is the result of an aft fuselage fire, it may be preceded or accompanied by erroneous readings from the surface position indicators, aft tank fuel quantity gauge and fuel totalizer due to wiring fire damage; therefore, if a fire is suspected or confirmed in the aft fuselage area, the crew should be watchful for the above indications. Hard-over rudder may be recognized by abrupt, uncommanded nose right yawing and right roll. There will be no response to rudder pedal input even through rudder pedal force and travel will appear normal. Roll control is the critical problem associated with hard-over rudder requiring left wing down to maintain a constant heading. Fire damage to, or failure of a hydraulic system will reduce roll control effectiveness; therefore, hydraulic demands should be minimized if possible. If hard-over rudder is experienced, the aircraft should be immediately climbed to a safe ejection altitude. If the flaps and slats are extended, they should not be retracted as this will reduce the amount of lift that can be spoiled and consequently reduce roll control effectiveness. With flaps and slats retracted, maintain an airspeed of 250-300 KIAS and slowly sweep the wings forward to allow operation of both inboard and outboard spoilers. Approximately 15 to 20 degrees left bank will be required to maintain heading. Use asymmetric thrust to reduce sideslip. Eject under controlled conditions, if possible.

**WARNING**

- Do not attempt to land with a hard-over rudder.
- Maintain airspeed above 250 KIAS to ensure adequate roll control.

## LOW FREQUENCY OSCILLATIONS IN PITCH OR ROLL AXIS

Although the pitch and roll gain changers are redundant, certain malfunctions may occur that 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.

2. If oscillations persist, turn damper off and land as soon as pracitcal using "Dampers Off Landing" procedures, this section.

## 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. If a malfunction occurs, proceed as follows:

1. Autopilot/damper switches - DAMPER.
2. If disengagement has not occurred, flight control disconnect switch - OVRD.
3. Land as soon as practical using "Dampers Off Landing" procedures, this section.

   If the flight control disconnect switch is in OVRD, pitch and roll gain changer lamps will remain lighted even though slats are extended and AYC will be inoperative.

## LIGHTNING STRIKE/STATIC DISCHARGE

Due to extreme pilot disorientation following a severe lightning strike, aircraft control is the immediate and primary concern. If flying at low altitude, a climb should be initiated as soon as aircraft control is attained. If extreme conditions such as severe turbulence require damper operation to maintain safe aircraft control, then dampers should be turned off as soon as conditions permit.

1. Pitch, roll and yaw damper switches - OFF.
2. Slow to damper off operating range.
3. Land as soon as practical using "Dampers Off Landing" procedures, this Section.

## FUEL SYSTEM MALFUNCTIONS

### EXCESSIVE FUEL DEPLETION PROCEDURE

Some fuel system failures can result in fuel 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 rate is known or suspected proceed as follows:

**WARNING**

Due to the fire hazard from fuel impinging on the fuselage, afterburner thrust should not be used during or following excessive fuel depletion.

1. Throttles - Set minimum power practical.

2. Fuel tank pressurization selector switch - OFF.

3. Fuel transfer knob - OFF.

   Terminate all transfer from 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 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 reduce to approximately a 20,000 pound maximum and select the desired tank(s) manually. (Do not use AUTO transfer.)

4. Fuel flowmeters - Checked.

   a. Compare left and right flowmeters.

   b. Check each engine against normal flow rate for flight conditions.

   c. Place throttles to IDLE one engine at a time and check for excessive fuel flow.

   d. Check engine response to throttle movement.

   e. If indicated flow is not considered excessive proceed to step 7.

**If Indicated Flow Is Excessive:**

5. Throttle of affected engine - OFF.

6. Fire pushbutton - Depress. (Proceed to step 10.)

**If Indicated Flow Is Not Excessive:**

7. Altitude - Descend below 30,000 feet.

8. Engine feed selector knob - OFF (or AFT if the forward tank is rapidly depleting).

9. 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 "Abnormal Fuel Distribution/Indicator Malfunction," this section.

10. Speed brake - Momentarily extend, then retract.

    Following an excessive fuel depletion in flight, the wheel well should be ventilated of fuel and vapor.

11. Land as soon as possible. Use "Single Engine Landing" procedures, this seciton, if applicable.

### ABNORMAL FUEL DISTRIBUTION/INDICATOR 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 8,200 lb. fuel differential between the forward and aft fuselage tank.

1. Engine feed selector knob - AFT.

   a. Fuel dump circuit breakers - Check in.

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 8,200 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 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 8,000 pounds prior to landing.

   c. If all indicators test good and the tanks are feeding normally, the fuel distribution caution lamp indication was caused by a failure of the alternate fuel distribution monitoring system. Auto feed may be resumed.

   d. If all indicators test good and an abnormal fuel distribution exists, the caution lamp indication was caused by a failure of the primary fuel distribution monitoring system. Operate the fuel system manually to maintain at least 8,200 pounds differential.

**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 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.
- 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. Land as soon as possible. Refer to "Landing With Aft Abnormal Fuel Distribution," this section.

**If Elevator Position Is More Than 2 Degrees Up:**

6. Engine feed selector knob - FWD.

7. Refer to "Landing With Forward Abnormal Fuel Distribution," this section.

## 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 ensure 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 discrepanacy is noted, continue misison. If any portion of this check reveals a loss of fuel, consult "Excessive Fuel Depletion Procedure," this section.

## 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 pressurizaiton.

2. Max AB - Zero to 27,000 feet altitude, zero to 300 KIAS without fuel tank pressurization.

3. Max AB - zero to 30,000 feet altitude, 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/Indicator Malfunction," this section.

## SINGLE AFT BOOST PUMP FAILURE

The following procedure may be used to expedite fuel transfer from the aft tank in the event that one aft boost pump fails. It will momentarily cause a forward center-of-gravity condition. This is acceptable, however, and will reduce the chances of trapping aft fuel, and thus causing a permanent aft center-of-gravity condition, should the remaining aft boost pump also fail.

1. Engine feed selector knob - AFT.

2. Fuel dump A circuit breaker - Pulled.

3. Fuel dump switch - DUMP. (If forward tank is less than full)

4. Fuel dump switch - OFF. (When aft tank is empty or forward tank is full)

Steps 3 and 4 can be repeated as often as practical.

5. Fuel dump A circuit breaker - IN.

6. Engine feed selector knob - As required.

# 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 not 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. 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 to moderate shifting of the flight controls. This may also be accompanied by a stick movement and will usually be felt as an airframe disturbance. The size and duration of the flight control transients may be more severe depending on the nature of the failure and may result in a sudden roll and lighting of the pitch and roll gain changer caution lamps, the pitch, roll, and yaw channel caution lamps, and the auxiliary attitudes caution lamp.

- In the event of a single generator failure, if an overload condition exists, jammer transmitters will go to STBY and the GEN OVLD lamp will light. Jammer radiate switches should be placed in STBY, and jammer transmitters returned to operation one at a time, monitoring the AC loadmeter to prevent overloading the remaining generator.

1. Electrical control panel - Check.

   Check electrical control panel for proper position of switches and that the power flow indicator reads TIE.

2. 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 malfunctioning generator.

3. Applicable generator switch - OFF-RSET, then ON.

   (Attempts to reset generator may be repeated if desired.)

**If Power Flow Indicator Reads NORM and the Generator Caution Lamp Is Out:**

4. Pitch damper switch - DAMPER.

**If Power Flow Indicator Reads TIE:**

5. Generator switch - OFF-RSET.
6. Pitch damper switch - DAMPER.
7. Land as soon as practical.

## GENERATOR OVERLOAD

**If Generator Overload Caution Lamps Are Lighted:**

1. Jammer radiate switches - STBY.
2. Generator select switch - Select operating generator.
3. Overload indicator reset switch - ON, then OFF.
4. Jammer radiate switches - RAD. (As required by threat priorities but not to exceed 250A load.)

**If required Generator Overload Exceeds 250A:**

5. Master radiate switch - OVRD.

**CAUTION**

Generator loads may not exceed 250A for more than 5 minutes.

**Note**

If generator load exceeds 315A, all transmitter loads will be shed automatically.

6. Additional jammer radiate switches - RAD. (As required by threat priorities but not to exceed 315A.)

## DOUBLE GENERATOR FAILURE

### Generator Recovery

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, and the angle-of-attack tape will be inoperative. Refer to figures FO-8 and FO-9 for list of equipment that is powered by the essential buses.

**WARNING**

- 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, and by the OFF flag on 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.

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.

4. Assure sufficient hydraulic pressure for emergency generator operation:

   a. If operating single engine, maintain a minimum of 90 percent rpm and 350 KIAS or less. Do not open or close speed brake. Sweep wings slowly (one degree per second) to 26 degrees.

   b. If both engines are operating, maintain a minimum of 90 percent rpm for wing sweep operation, landing gear extension, or speed brake retractions. Do not re-open speed brake.

**WARNING**

Flight control damper transients may be experienced if hydraulic demands cause an interruption of the emergency generator power.

5. Pitch damper switch - OFF. (Within damper 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.

6. Generator switches (individually) - OFF-RSET, then ON.

   Individually place the generator switches to OFF-RSET then to ON.

**If Power Flow Indicator Reads TIE:**

7. Generator switch (malfunctioning generator) - OFF-RSET.

8. Pitch damper switch - DAMPER.

9. Emergency generator switch - AUTO.

**If Power Flow Indicator Reads NORM:**

10. Pitch damper switch - DAMPER.

11. Emergency generator switch - AUTO.

**If Power Flow Indicator Reads EMER:**

12. Refer to figures FO–8 and FO–9 for essential equipment available.

13. Land as soon as possible using the following fuel system operation/dump procedures.

## FUEL SYSTEM OPERATION ON EMERGENCY GENERATOR

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. 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 operator depending on fuel distribution. If the fuel differential is greater than 8,500 pounds, number 4 pump will supply fuel to the engines until the differential reduces to less than 8,250 pounds. Number 5 pump will then supply fuel to the engines until the differential again increases to 8,500 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 then 7,900 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 gravity feed. In order to transfer fuel from the wing tanks the engine feed selector switch must be turned OFF and the fuel transfer switch placed to WING or AUTO. This will result in the engines being fed by gravity 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 gravity 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 8,200 ($\pm$400) pounds fuel differential.

2. Fuel transfer knob - OFF.

3. Fuel tank pressurization selector switch - PRESSURIZE.

### Wing 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.

## FUEL DUMP ON EMERGENCY GENERATOR

When operating on emergency power, fuel may be dumped only by using the following procedures:

1. Establish an airspeed not greater than 350 KIAS or Mach 0.75, whichever is less, and maintain 1 "g" flight.

2. Position wings slowly to 26 degrees.

3. Engine feed selector knob - AUTO.

4. Fuel dump switch - DUMP.

5. Fuel distribution - Maintain 2,000 to 8,000 pounds differential.

   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.

**To Transfer Wing Tank Fuel, Perform the Following:**

6. Fuel dump switch - OFF.

7. Engine feed selector knob - OFF.

8. Fuel transfer knob - As required.

9. To resume dump procedures, return to step 3.

**When Dumping Is Complete:**

10. Fuel transfer knob - OFF.

11. Fuel dump switch - OFF.

12. Engine feed selector knob - As required.

13. Land as soon as possible.

## PARTIAL LOSS OF AC BUS POWER

Failure of a generator is normally accompanied by lighting of the applicable generator caution lamp. However, certain generator control unit (GCU)/generator failures could occur which would de-excite the applicable generator and not remove it from its buses. In either case, the result would be the loss of, or degraded power to, the applicable main and/or essential ac buses, no generator caution lamp, normal power flow indications, and miscellaneous unrelated caution lamps. All dc buses remain powered.

**WARNING**

> Thorough analysis of the lighted caution lamps and inoperable equipment is critical. These electrical system failure modes may have similar indications to hydraulic system failure or certain overheat situations.

The severity of the electrical system malfunction will depend on whether the failure occurs in the left or right GCU/generator. Loss of power to the left main and essential ac buses can occur due to failure(s) within the left generator control unit. This failure will likely result in an unscheduled maneuver as the flight control computers and dampers lose power. Flight control damper, channel and gain changer caution lamps will light. In addition, those instruments and systems which normally receive power from these buses will be inoperative. Off flags in the flight instruments will be readily apparent. Fuel quantity, fuel flow, and hydraulic pressure indications will freeze and oil pressure, nozzle position, and EPR gauges will be inoperative. The weapons and nav computers, CADC, INS, AFRS, TFR, navigational radar, TACAN, and air-to-ground IFF will also be affected. The left generator caution lamp may not light and the power flow indicator will display NORM. It should be noted that a similar failure involving the right generator and right main ac buses will be much less severe than those associated with the left main and essential ac buses. The following are considerations to be applied as necessary:

- If the damper lamps are lighted, the damper switches should be turned off before attempting to restore electrical power.

- Check each generator output on the generator ammeter (power monitor panel). If one generator output indicates zero, turn the respective generator switch to OFF/RSET. Turning the affected generator switch to the OFF/RSET position may allow the operative generator to power the buses.

- If the left GCU/generator has a malfunction, turning the emergency generator switch on should power all systems on the essential ac bus. Refer to figure FO–8 for essential equipment available.

- Power may be restored to the right main bus only by insuring the left generator has picked up the load by an indication of TIE on the generator control panel.

## 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.

- Airspeed should be maintained within "damper off operating limits" as stability augmentation is not available.

- 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.

- If the wing sweep handle is moved to the 16 degree detent, the handle will lock at that position and the wings cannot be moved aft. The wing sweep position indicator will be inoperative.

- Slats and flaps can be extended using normal extension procedures. The flap/slat position indicator is inoperative.
- The landing gear must be extended using emergency extension procedures. No gear down indication will be available.
- Roll response will be degraded due to the loss of spoilers.
- Anti-skid and ground roll spoilers will not be available on landing. Manual braking above 100 knots may cause the wheels to skid due to a low weight on wheels condition.
- If battery power is available, as indicated by the TIT indicator off flags being out of view, then airstart, TIT indicators, utility lights, fire pushbuttons, and agent discharge are operational; however the engine fire detect system will be inoperative.
- The standby altimeter, airspeed indicator, and vertical velocity indicator, the standby compass, and the tachometers will be operational; all other flight instruments will be inoperative.
- The self-contained attitude indicator (SCAI) will display reliable attitude information for nine minutes after the OFF flag is in view.
- Estimate fuel consumption as closely as possible to aid in setting wing sweep for landing.
- Radio communication may be attempted with the emergency radio contained in the quick rescue kit.

## 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.
3. Airspeed - 260 to 460 KIAS.
4. Air source selector knob - EMER.

**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:**

5. Air Source selector knob - OFF.

**If Cabin Overheat Continues:**

6. Left engine - Reduce to IDLE power.

**If Cabin Overheat Persists:**

7. Left engine - Power as required.

8. Right engine - Reduce to IDLE power.

**WARNING**

Extended delay in reducing high cabin temperature can cause excessive aircrew dehydration and possible incapacitation.

**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)

9. Canopy - Open one or both. (Be prepared for wind blast)

10. Land as soon as possible if overheat condition persists or canopy is opened.

## 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. Air source selector knob - RAM. (If required)

**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.

## 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 landing gear handle is placed in the down position. 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 hydrualic 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, and speed brake will be completely inoperative. Back-up systems are provided to operate the spikes, landing gear extension, flaps and slats, translating cowls, wheel brakes and air refueling system.

1. Wing sweep handle - As required (1 degree/second).

   Maintain wing sweep position compatible with airspeed and sweep wings to 26 degrees when at appropriate airspeed. Minimize flight control movement during wing sweep.

**WARNING**

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. Do not depress the damper reset button unless the affected hydraulic 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.

## EMERGENCY EXTENSION OF FLAPS AND SLATS

Due to the time required for emergency flap/slat extension, adequate time must be allotted during the approach. If a published instrument penetration is to be accomplished, the aircraft may be fully configured prior to the IAF and a penetration made using a minimum of 20 knots above computed final approach speed. The wings must be at 26 degrees or forward prior to attempting emergency extension.

1. Rudder authority switch - FULL.
2. High lift flap/slat control circuit breaker - In.
3. Reduce airspeed to 180 KIAS or 10 degrees angle-of-attack, whichever is higher.
4. Flap and slat switch - EMER.
5. Emergency flap and slat switch - EXTEND and hold. (Emergency extension requires approximately 60 seconds)

**WARNING**

Make a positive check that all slats are extended by visual observations 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.

**CAUTION**

Extreme care must be exercised when operating the slats/flaps with the emergency electric motor. Ensure that SLAT/FLAP RETRACT-EXTEND switch is released immediately when full retract or extend position is reached. Holding the switch in position after the slats and flaps have reached full position can cause damage to the electric motor and the flex drive shafts.

**Note**

- If flaps and slats continue to drive to the up position during or after emergency extension the cause may be a failure of the isolation valve or emergency extension switch. Positioning of the flap/slat handle to DOWN will allow the emergency extension system to drive the flaps and slats down.
- If the decision is made to land with the flaps/slats in a position that does not agree with the flap/slat handle, a malfunction in the flap/slat control system may cause the flaps/slats to drive to the position commanded by the flap/slat handle.

6. Flap/slat handle - Down, to agree with selected flap/slat position.

## SPEED BRAKE MALFUNCTIONS

### Speed Brake Fails to Extend:

1. Utility hydraulic isolation switch - NORM.
2. Speed brake hydraulic valve circuit breaker - Reset.

### Speed Brake Fails to Retract:

3. Speed brake hydraulic valve circuit breaker - Pull out.
4. Before extending gear - Push circuit breaker in.

# LANDING EMERGENCIES

## APPROACH END CABLE ENGAGEMENT

Approach end arrestments are considered practicable and should be attempted when directional control and stopping distance are questionable or when a malfunction presents a threat to directional control and there is sufficient runway in front of the cable on which to land and lower the nosewheel prior to cable contact. Consideration should also be given to the engaging speed limits to prevent structural failure of the arresting cable or the aircraft.

**Note**

- Fly a straight-in approach when possible to ensure an accurate touchdown point on the runway. Considerations should also be given to actions taken if engagement is missed; i.e., go-around and cable engagement on the other end of the runway.
- R A BAK–12/14 can require up to 7.5 seconds from activation to become fully up and locked. Controller reaction time may increase this time.

1. Fuel - Dump. (As required)
2. "Descent Check" and "Before Landing" checklists - Complete.
3. Loose equipment - Stow.
4. Throttle friction - Reasonably tight.

   Deceleration forces could cause throttles to be thrown forward if not tight.
5. 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.
6. Fuel distribution and quantity - Check.
7. Wing sweep handle - Set for landing.
8. Backup approach speed - Compute.
9. Translating cowls - Check open at mach 0.44.
10. Elevator position - Check.
11. Landing gear handle - DN and checked.
12. Flap/slat handle - As required.

**Note**

A full flap single engine approach may be flown with full flaps to decrease cable engagement speed or stopping distance; however, if cable is missed, go-around capability may be marginal until flaps are raised to 25 degrees.

13. Elevator position - Check on final approach.
14. Shoulder harness - Locked.
15. Touch down in center of runway at least 400 feet short of cable and ensure strut compression prior to arrestment.
16. Lower nose immediately.

    Do not make an attempt to steer aircraft to center of cable.
17. Throttles - IDLE.

    Reduce power to idle at touchdown to insure spoiler brake operation.

18. Engage cable with brakes off.

    Off-center engagement may cause the aircraft to veer off course to the off-center side of the runway. Cable 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 ensure aircraft control. Roll back may occur, depending on type of cable 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 roll-back without causing the aircraft to pitch up.

19. NRS and TFR - STBY.

20. Keep engine running until crash crews arrive and signal for engine shutdown.

21. 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/FLAT GEAR STRUT/WHEEL FAILURE LANDING

Approach end cable engagement with a blown tire/flat main gear strut is recommended, but consideration must be given to cable availability and type, runway, weather, and aircraft conditions. Consideration should also be given to actions taken if engagement is missed. If cable engagement is to be made, refer to "Approach End Cable Engagement," this section.

### MAIN GEAR

1. Fuel - Dump. (If required)

2. "Descent Check," "Before Landing," and "Landing Pattern" checklists - Completed.

3. Fly a straight-in approach.

4. Touch down on side of runway opposite the blown tire.

5. Lower nose and use nosewheel 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 the blown tire.

6. Hook - Extend. (If required)

### NOSE GEAR

1. Fuel - Dump. (If required)

2. "Descent Check," "Before Landing," and "Landing Pattern" checklists - Completed.

3. Ground roll spoiler brake switch - OFF.

4. Fly a straight-in approach.

5. Delay lowering nose to runway until just prior to losing flight control effectiveness. Then, apply aft stick for aerodynamic braking effect but keep nosewheel on runway.

6. Use differential braking as required for directional control.

## CONTROLLABILITY CHECK

During any inflight or landing emergency when structural damage or any other failure is known or suspected that may adversely affect aircraft handling characteristics, a controllability check should be performed. As a minimum, the following should be considered:

- Proceed to safe altitude and area (minimum 5,000 feet AGL).

- Adjust gross weight and cg as required.

- Determine optimum configuration available for landing.

- Determine if adequate control is available for landing and/or approach end cable engagement.

**WARNING**

Angle-of-attack should be controlled on final approach to ensure that the aircraft is landed as close to on-speed as possible. A fast landing may result in the aircraft becoming airborne after touchdown, a pilot induced oscillation, and loss of aircraft control.

## DAMPERS OFF LANDING

If a landing is to be made with any or all of the dampers OFF proceed as follows:

1. Fly a straight-in approach at 10 degrees angle-of-attack.

2. Avoid large or abrupt inputs. In crosswinds with yaw damper inoperative, make a wings level crabbed touchdown.

   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 aircraft 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.

3. Descent Check, "Before Landing," and "Landing Pattern" checklists - Completed.

## 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 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.

## FUEL ABNORMAL 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 1 "g" flight, with the speedbrake, flaps, slats, and gear retracted, sweep wings forward until 26 degrees wing sweep or until the stabilizer indication is 1 degree down elevator, whichever is reached first.

**If 26 Degrees Wing Sweep Is Reached and 1 Degree Down Not Exceeded:**

2. "Descent Check", "Before Landing," and "Landing Pattern" checklists - Completed.

3. Make a normal straight-in approach.

**WARNING**

Do not sweep wings forward of 24 degrees in attempting to extend flaps if the stabilizer indication is one degree down. Do not allow average elevator position of more than one degree down because sufficient aircraft nose down elevator authority may not be available to maintain control of the aircraft.

**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 "Slats Up (Gain Changer Lamps On), Flaps 5 Degrees Or Less Landing," this section.

### LANDING WITH FORWARD ABNORMAL FUEL DISTRIBUTION

1. If time permits, dump or burn fuel until the elevator position indicator (EPI) indicates less than two degrees trailing edge up during one "g" flight with speed brake, flaps and slats retracted. If the EPI is inoperative, continue fuel dump/burn until the aft tank is empty and the forward tank indicates 8,000 pounds. Land from a straight-in approach, using normal procedures, Section II.

**If Landing With Forward Abnormal Fuel Distribution Is Required:**

2. "Descent Check," "Before Landing," and "Landing Pattern" checklists - Completed.

3. Slow the aircraft until 10 degrees trailing edge up or until final approach angle-of-attack is reached, whichever occurs first.

4. Do not allow airspeed to decrease below this value until after touchdown.

## GEAR MALFUNCTIONS

### NOSE GEAR RETRACTION FAILURE LANDING

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. "Descent Check" and "Before Landing" checklists - Completed.

2. Landing gear handle - DN. (Do not recycle the landing gear handle)

3. Flaps/Slats - Extend.

**After The Landing Gear Is Fully Extended:**

4. Landing gear emergency release handle - Pull.

   This will remove all hydraulic pressure to the nosewheel steering system and will allow the nose gear to align with the runway.

5. Landing pattern checklist - Complete.

6. Consider cable engagement. Refer to "Approach End Cable Engagement," this section.

## SAFE GEAR INDICATION (TWO GREEN LAMPS ON) WITH WARNING LAMP LIGHTED (RED LAMP ON) OR GEAR WARNING HORN ON

If the landing gear handle warning lamp is lighted or the landing gear warning horn is audible, with nose and main gear down and locked, the speed brake may be mispositioned.

**Note**

If the landing gear warning horn is on and the landing gear handle warning lamp is not lighted, the malfunction and indicator lamp test button should be depressed to check the status of the warning lamp.

1. Speed brake hydraulic valve circuit breaker - Pull. (Do not reset after landing)

**CAUTION**

If the speedbrake goes to trail, do not reset the circuit breaker after landing. To do so may cause the speedbrake to extend and contact the ground.

**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.

**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, nosewheel steering will be inoperative and the nosewheels may be cocked up to 90 degrees. During landing roll the nosewheels will align and present no directional control problem.

## UNSAFE GEAR INDICATION

Landing gear unsafe (not down and locked) is indicated by either or both green landing gear position indicator lamps not being lighted after gear down selection. However, failure of one or both landing gear position indicator lamps to light, together with normal landing gear handle warning lamp operation (red lamp on when gear handle is lowered followed by the red lamp going out) 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 "Safe Gear Indication (Two Green Lamps On) With Warning Lamp Lighted (Red Lamp On) or Gear Warning Horn On," this section.

1. Malfunction and indicator lamps - Checked.

2. Circuit breakers in - Checked.

   - Landing gear control
   - Landing gear warning
   - Speed brake hydraulic valve

3. Utility hydraulic isolation switch - PRESSURIZE.

**If Gear Still Indicates Unsafe:**

4. Obtain visual gear check; if main and nose gear appear to be properly extended, refer to "Emergency Gear Extension," this section.

**Note**

If the green nose gear position indicator lamp went out normally upon gear retraction, the landing/taxi light is a valid indication of safe nose gear.

**If Either Nose Or Main Gear Is Not Extended, Or Visual Check Was Not Possible:**

5. Landing gear - Recycle.

   The landing gear handle may be recycled more than one time if desired; however, it is not recommended that the handle be rapidly recycled. Allow sufficient time for the normal sequence of events to occur after each movement of the handle.

   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 "Emergency Gear Extension," this section.

## EMERGENCY GEAR EXTENSION

1. Landing gear handle - DN.
2. Slats and flaps - Extend.
3. Reduce speed to 160 KIAS or computed approach speed, whichever is higher.
4. 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 take up to 10 minutes.

**Note**

After the landing gear emergency release handle is pulled, nosewheel steering will be inoperative and the nosewheels may be cocked up to 90 degrees. During landing roll the nosewheels will again align and present no directional control problem.

5. Landing gear indicator lamps - Check.

   a. If landing gear is down and locked (two green lamps on) with warning lamp out (red lamp out), proceed to step 9.

   b. If landing gear is down and locked (two green lamps on) with warning lamp lighted (red lamp on), proceed to step 10.

6. If the landing gear is not down and locked (one or both green lamps out) after 10 minutes, landing gear emergency release handle - In.

   Failure of the main landing gear door/main landing gear sequence link may result in the inability to extend the main landing gear by either normal or emergency means. In this situation, pulling the landing gear emergency release handle will unlock the main landing gear uplocks and open the forward main landing gear door; but the door will immediately go to, and be pneumatically held in the trial position, thus preventing landing gear extension. This configuration can be checked visually. The nose gear will be fully extended, the aft main landing gear door will be protruding below the fuselage and the main landing gear wheels will be resting on the forward main landing gear door. Pushing the landing gear emergency release handle back in will deplete the pneumatic pressure and enhance the possibility of completing gear extension.

   a. Alternately impose a negative 1.0 "g" and a positive 2.0 "g" load on the aircraft and check for gear down indication.

**CAUTION**

To avoid possible damage to the gear mechanism, do not exceed 2.0 positive "g" during extension attempts.

   b. Slow the aircraft to minimum practical airspeed to reduce air load on the forward main landing gear door.

   c. Landing gear emergency release handle - Pull.

      Some pneumatic pressure should still be available. Pulling the landing gear emergency release handle a second time will apply this pressure to the landing gear downlocks, and also may prevent the speed brake from extending and dragging on the runway during landing roll.

7. Landing gear indicator lamps - Check.

   a. If landing gear is down and locked (two green lamps on) with warning lamp out (red lamp out), proceed to step 9.

   b. If landing gear is down and locked (two green lamps on with warning lamp lighted (red lamp on), proceed to step 10.

8. If the landing gear is not down and locked (one or both green lamps out), obtain a visual check of gear position. If gear appears down, refer to "Unsafe Gear Indication Landing", this section. If a visual check is not possible, or the gear does not appear down, refer to "Retracted Nose/ Main Gear Landing", this section.

9. Consider cable engagement. Refer to "Approach End Cable Engagement," this section.

**If the Landing Gear Handle Warning Lamp Remains Lighted With Gear Down and Locked:**

10. 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 may allow the door to extend and drag runway.

## UNSAFE GEAR INDICATION LANDING

1. Fuel - Dump. (If required)

2. "Descent Check" and "Before Landing" checklists - Complete.

3. Consider cable engagement. Refer to "Approach End Cable Engagement," this section.

**If Approach End Cable Engagement Not Used:**

4. If nose gear is unsafe, ground roll spoiler switch - OFF.

5. Shoulder harness - Locked.

6. Landing pattern checklist - Complete.

7. Fly straight-in approach at 10 degrees angle-of-attack.

8. Stop aircraft on the runway and insert landing gear pins.

**Note**

Touchdown 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:**

9. After nose is on the runway, throttles - OFF.

10. Fire pushbuttons - Depress.

11. Battery switch - OFF.

12. Exit the aircraft.

## RETRACTED NOSE/MAIN GEAR LANDING

Approach end cable engagement with nose, main or both gear retracted is recommended but must take into consideration the cable 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 cable procedures based on the nature of the gear problem. To provide for positive cable engagement, approach the runway with slats down, flaps full down, and the most forward wing sweep permitted by the c.g., using 10 degrees angle-of-attack. Plan to stabilize flight conditions at normal gear down height at least 400 feet short of the cable. After the aircraft has engaged the cable 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 throttles and fire pushbuttons as this will shut off hydraulic fuel and coolanol lines and lessen chance for fire from fuel drainage, hydraulic or coolanol fluid leakage.

1. Fuel - Dump. (If required)
2. Wing sweep - Most forward permitted by C.G.
3. "Descent Check" and "Before Landing" checklists - Complete.
4. Backup approach airspeed - Compute.
5. Translating cowls - Check open at mach 0.44.
6. Air source selector knob - EMER.
7. Flaps/slats - Down.
8. Ground roll spoiler switch - OFF.
9. Fuel panel switches - OFF.
10. All nonessential equipment - OFF.
    a. JSS/SPS liquid pumps switches - NORM.
    b. Master radiate switch - RAD.
    c. Inhibit switch - NORM.

**Note**

Position of switches removes 28 V dc and 115 V ac power power source to weapons bay.

11. Recommend approach end cable engagement. Refer to "Approach End Cable Engagement", this section.
12. Loose equipment - Stow.
13. Shoulder harness - Lock.
14. Throttle friction - Reasonably tight.
15. Fly straight-in approach at 10 degrees angle-of-attack.
16. Elevator position - Check. (At 10 degrees angle-of-attack)
    a. Wings at 16 degrees - 4 to 10 degrees TEU.
    b. Wings at 26 degrees - 2 to 10 degrees TEU.

**Note**

- If a cable engagement procedure is not utilized with nose gear retracted, lower the nose to the runway while control effectiveness still exists and apply maximum braking.
- The rudder authority switch must be positioned to FULL to obtain full rudder authority if the landing gear handle is not in the DN position.

**On Final:**

17. Fuselage fire pushbutton - Depress.
18. Agent discharge switch - Up.

**After Landing:**

19. Throttles - OFF.
20. Engine fire pushbuttons - Depress.
21. Battery switch - OFF.
22. Exit the aircraft as soon as possible.

## HYDRAULIC MALFUNCTIONS

### PRIMARY OR UTILITY HYDRAULIC FAILURE LANDING. (COMPLETE EVEN IF CONFIGURED FOR LANDING)

Due to the time required for emergency gear and flap extension, adequate time must be allotted during the approach. If a published instrument penetration is to be accomplished, the aircraft may be fully configured prior to the IAF, and penetration made using a minimum of 20 knots above the computed final approach speed. An approach end cable 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 to 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 cable 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. The anti-skid should be turned off since application of brakes at high speed could result in anti-skid cycling and depletion of the hydraulic accumulators, thereby losing the use of normal brakes for directional control and stopping. Do not attempt damper reset unless pressure returns to normal.

1. Fuel - Dump. (As required)
2. Spike control switches - OVERRIDE (Below 0.9 Mach)
3. Wing sweep - Set for landing. (1 degree per second)
4. "Descent Check" and "Before Landing" checklists - Complete.
5. Backup approach airspeed - Compute.
6. Translating cowls - Check open at mach 0.44.
7. Anti-skid switch - OFF.

**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.
- The most efficient braking occurs when the brake pressure is just below the skid threshold for each respective wheel. This pressure is generally exceeded due to difficulty in sensing or detecting the onset of a skid. If the same braking technique is used with anti-skid off as anti-skid on, locked wheels and blown tires may result.

8. Translating cowls emergency override switch - OVRD.
9. Flap/slat emergency extension - Completed.

**WARNING**

Due to the loss of one set of spoilers with hydraulic system failure, attempting abrubt rolling maneuvers or bank angles in excess of 30 degrees with the flight control system in takeoff and land can result in loss of control of the aircraft.

a. Rudder authority switch - FULL.
b. Reduce airspeed to 180 KIAS or 10 degrees angle-of-attack, whichever is higher.
c. Flap and slat switch - EMER.
d. Emergency flap and slat switch - EXTEND and hold. (Emergency extension may require approximately 60 seconds.)

**WARNING**

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.

**CAUTION**

Extreme care must be exercised when operating the flaps/slats with the emergency electric motor. Ensure that SLAT/FLAP RETRACT-EXTEND switch is released immediately when full retract or extend position is reached. Holding the switch in position after the slats and flaps have reached full position can cause damage to the electric motor and the flex drive shafts.

**Note**

- If flaps and slats continue to drive to the up position during or after emergency extension the cause may be a failure of the isolation valve or emergency extension switch. Positioning the flap/slat handle to DOWN will allow the emergency extension to drive the flaps and slats down.

- If the decision is made to land with the flaps/slats in a position that does not agree with the flap/slat handle, a malfunction in the flap/slat control system may cause the flaps/slats to drive to the position commanded by the flap/slat handle.

e. Flap/slat handle - Down, to agree with selected flap/slat position.

10. Landing gear emergency extension - Completed.

a. Landing gear handle - DN.

b. Reduce speed to 160 KIAS or computed approach speed, whichever is higher.

c. Landing gear emergency release handle - Pull.

d. Landing gear indicator lamps - Check.

**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 take up to 10 minutes.

**Note**

After the landing gear emergency release handle is pulled, nosewheel steering will be inoperative, and the nosewheels may be cocked up to 90 degrees. During landing roll, the nosewheels will align and present no directional control problem.

**If Gear Is Not Down and Locked (One Or Both Green Lamps Out) Or The Warning Lamp Is On (Red Warning Lamp On), Refer To "Emergency Gear Extension," This Section.**

11. Consider cable engagement. Refer to "Approach End Cable Engagement," this section.

**If Approach End Cable Engagement Is Not Used or If Cable Is Missed:**

12. Elevator position - Checked. (At 10 degrees angle-of-attack)

a. Wing at 16 degrees - 4 to 10 degrees TEU.

b. Wing at 26 degrees - 2 to 10 degrees TEU.

13. Fly straight-in approach.

14. Utilize aerodynamic braking and maintain directional control with the rudder as long as effective.

15. Hook - Extend. (If required)

16. 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 if accumulator pressure has reduced to below 1100 psi (normal braking no longer available).

17. Stop straight ahead on the runway.

## LANDING WITH ASYMMETRIC STORES

If a large or heavy asymmetry exists, landing can be accomplished using a 10 degree angle-of-attack approach. Landing configuration and approach speed should be established with sufficient altitude remaining to determine specific flying qualtities 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 $\pm 6$ 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.

## LANDING WITH CADC/PITOT-STATIC SYSTEM MALFUNCTION

If a CADC malfunction occurs, the standby airspeed indicator will be reliable. If a pitot-static system malfunction occurs (due to bird-strike, lightning, ice formation, etc), both primary and secondary airspeed indications will be unreliable; however, angle-of-attack indications will be reliable. If both CADC and pitot-static system malfunctions are present, NC groundspeed may be adjusted to approximate approach speed. Crosscheck rpm against NC groundspeed as an additional guide in establishing the correct speed.

**WARNING**

With a CADC malfunction, the stall warning system and gear warning horn will be unreliable.

1. "Descent Check" and "Before Landing" checklists - Complete.
2. IFF mode C mode select/test switch - OUT.
3. Compressor bleed valve control switches - CLOSE.
4. If angle-of-attack and airspeed indications are unreliable compute final approach groundspeed as follows:
   a. Desired final approach KIAS plus 3 knots per 1,000 feet MSL equals desired TAS.
   b. Desired TAS minus runway headwind component reported from ground station equals final approach groundspeed.

**WARNING**

Wind direction/velocity and TAS indications are unreliable with pitot-static system or CADC malfunction present.

5. Fly formation straight-in approach if possible.
6. "Landing Pattern" checklist - Complete.

## LANDING WITH LANDING GEAR GROUND SAFETY (SQUAT) SWITCH FAILURE

### SWITCH FAILED IN GROUND POSITION

1. "Descent Check," "Before Landing." and "Landing Pattern" checklists - Complete, except leave ground roll spoiler switch off.
2. Landing gear handle - Down.
3. Landing gear safety relay circuit breaker - Pulled.
4. The AYC and stall warning devices may be inoperative. Fly a straight-in approach, avoiding abrupt control inputs, and do not exceed 60 degrees bank.
5. Throttles - Maintain above idle until touchdown.

**After Touchdown:**

6. Ground roll spoiler switch - BRAKE.
7. Landing gear safety relay circuit breaker - In.

   Normal anti-skid braking and nose wheel steering will be inoperative until this circuit breaker is pushed in.

## SWITCH FAILED IN AIR POSITION

Failure of the squat switches in the air position is an insidious failure and cannot be detected unless the aircraft is on the ground. If a switch fails to the air position, the first indication to the aircrew will normally be ground roll spoilers failing to extend or nozzles remaining closed when the throttles are retarded to idle during the landing roll. If both switches fail in the air position, the following systems will be inoperative on the ground:

- Ground roll spoiler brakes
- Nosewheel steering
- Wheel brakes will not function below 20 knots with anti-skid switch on. (No caution lamp indications will be present.)
- Ejector air coolers
- Flight control ground test panel.
- Engine bleed opening with throttles below MIL.

The following systems will continue to operate on the ground:

- Adverse yaw compensation (AYC).
- Nozzles remain closed at idle power
- Pitot and alpha/beta probe heaters remain on in the OFF/SEC position
- Artificial stall warning
- Landing gear handle can be moved from down position.
- JSS radiate

**If a Failure of the Squat Switches to the Air Position is Suspected During the Landing Roll, Proceed as Follows:**

1. When rudder effectiveness is lost, maintain directional control using differential braking.
2. Anti-skid switch - OFF before reaching 20 knots.
3. If wheel braking is lost, refer to "Brake Malfunctions," this Section.
4. Shut down engines as soon as practical after landing. R R

**WARNING**

- The absence of ejector air flow will allow fuel and oil vapors to accumulate in the engine bay which may cause explosive conditions to exist. Engine bay subsystems (generators, oil supply, hydraulics) will not be cooled causing associated caution lamps to light and possible decoupling of the generators. R R R R R R R R
- With the landing gear ground safety (squat) switches failed to the air position, the pitot probe and alpha/beta probe heaters will remain on when the aircraft is on the ground.

## TRANSLATING COWL(S) CLOSED LANDING

If a malfunction requires landing with either or both cowls closed, military power can be selected at normal landing approach angle-of-attack, stall free. In the event a go-around or touch-and-go landing is required with the cowls closed, military power can be selected stall-free at or above 100 knots provided the angle-of-attack is restricted to 11 degrees or less. Afterburner power can be selected but may not be completely reliable due to an afterburner mislight possibility.

1. Fuel dump - As required.
2. "Descent Check," "Before Landing," and "Landing Pattern" checklists - Complete.
3. Establish landing pattern as necessary to ensure angle-of-attack does not exceed 11 degrees.

**WARNING**

Inducing a nose high attitude to reduce speed should be avoided to prevent possible excessive sink rate.

## LANDING WITH FLAP AND SLAT MALFUNCTIONS

When a flap/slat malfunction occurs, the conditions should be evaluated to determine if the emergency extension procedures, this section, should be attempted. If conditions require landing with a flap or slat malfunction, factors such as gross weight, approach speed, ground roll distance and runway conditions must be considered. Diversion to a suitable alternate or approach end cable engagement may be necessary. The appropriate course of action will depend upon the amount and combination of slat and flap extension. If landing is to be made, with either a slat or flap malfunction, follow the procedure, this section, which applies to the final flap/slat configuration achieved.

**WARNING**

- If flaps are confirmed or suspected to be extended more than 5 degrees and the slats cannot be verified visually or by gain changer lamps to be approximately 70 percent down, landing should be made using the "Slats Up (Gain Changer Lamps On), Flaps More Than 5 Degrees Landing" procedures, this section.
- If a rolloff condition exists and large lateral control is required, a likely cause is an asymmetric flap condition. An attempt should be made to reduce the asymmetry by returning the flap/slat handle toward the previous position until rolloff stops. Landing should be made using the "Asymmetric Flap Landing" procedures, this section.

**CAUTION**

If the flaps stop at an intermediate position during 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 and slat switch to EMER will relieve hydraulic pressure to the flap motor and could prevent further damage or undesirable flap movement.

**Note**

- The gain changer lamps will provide an indication of slat position. After the landing gear handle is placed down, the gain changer lamps will go out if the slats extend 70 percent or more. The gain changer lamps will remain lighted if the flight control disconnect switch is in OVRD or if there is a malfunction in the adverse yaw compensation circuits. In this case, the lamp will not go out when the control system switch is placed to takeoff and land. All available means should be used in determining slat position, i.e. flap/slat indicator, visually and the gain changer lamps.
- When landing with a missing flap vane, consideration should be given to using 25 degree flaps in order to reduce the split in the horizontal stabilizers.

## SLATS UP (GAIN CHANGER LAMPS ON), FLAPS 5 DEGREES OR LESS LANDING

Approach speeds will vary with configuration (figure 3-6). The important factor to consider, however, is that for wing sweeps between 16 and 45 degrees, the approach should be flown at 11 degrees angle-of-attack. For wing sweeps aft of 45 degrees the approach should be flown at 12 degrees angle-of-attack. Landing should be made from a long, shallow straight-in approach. Excessively shallow approaches make judgement 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. 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. Care should be exercised to avoid tail strikes at touchdown. Landings can be made with up to 14 knots of crosswind but, with wing sweeps greater than 45 degrees, 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 buildup which may be difficult to arrest at approach altitudes.
- Maintain an airspeed compatible with aircraft configuration and gross weight to ensure that 10 degrees angle-of-attack is not exceeded during the maneuvering flight prior to final approach phase.

1. Flaps and slat switch - EMER.

# EMERGENCY LANDING AIRSPEEDS (FLAPS UP)

DATA BASIS: FLIGHT TEST
DATE: SEPTEMBER 1984

CONFIGURATION
- SLATS RETRACTED (EXCEPT AS NOTED)
- FLAPS RETRACTED

CONDITIONS:
- CG – FUEL SYSTEM IN AUTO MODE

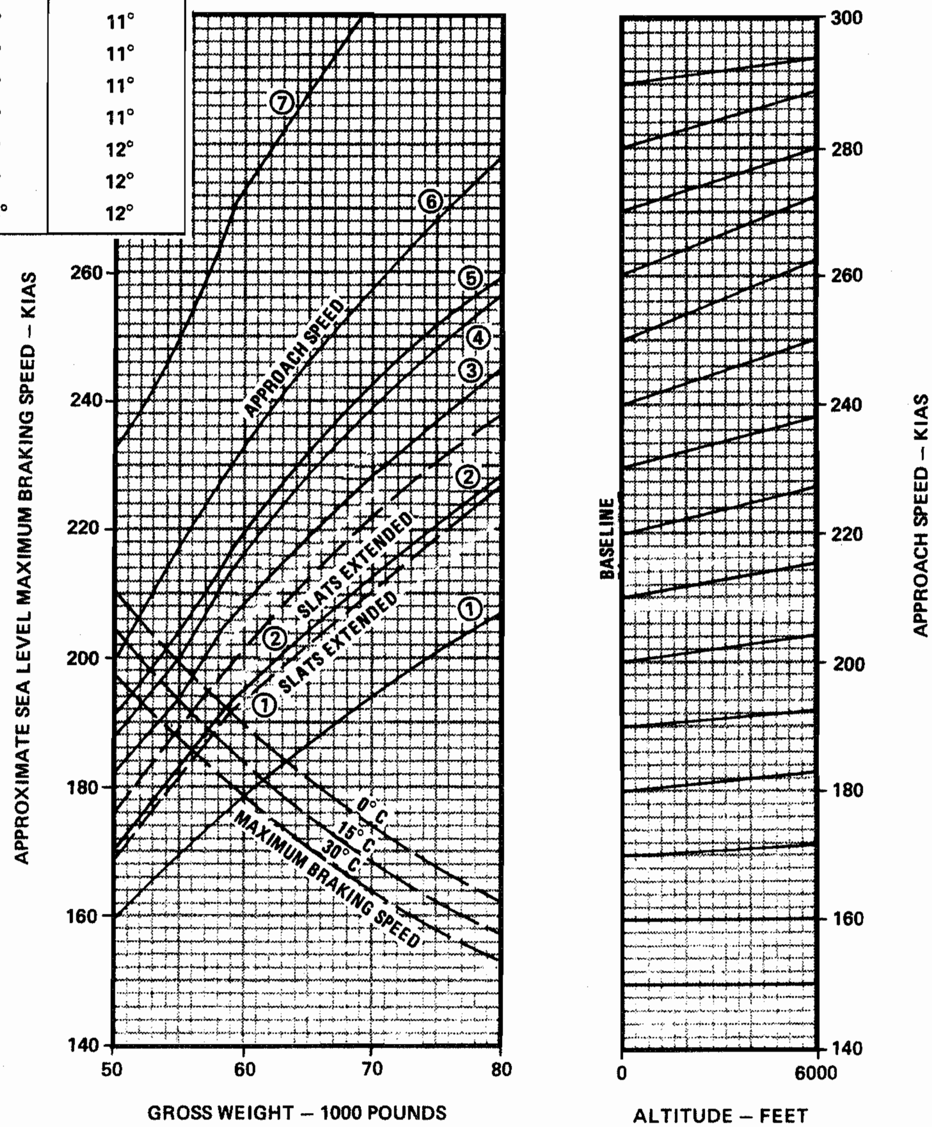
FUEL GRADE: JP-4
ENGINES: TF30-P-3

NOTES:
1. SUBTRACT 10 KIAS FROM APPROACH SPEED TO OBTAIN TOUCHDOWN SPEED
2. DECREASE MAXIMUM BRAKING SPEED 4 KIAS PER 1000 FEET OF ALTITUDE.
3. ADD 0.5 KIAS FOR EACH PYLON STATION WITH STORE PRESENT.

**WARNING**

REFER TO SECTION V FOR BRAKE ENERGY LIMITS AND SECTION VI FOR CG LIMITS

| WING SWEEP | | INDICATED ANGLE OF ATTACK |
|---|---|---|
| 1 | 16° | 11° |
| 2 | 26° | 11° |
| 3 | 35° | 11° |
| 4 | 40° | 11° |
| 5 | 50° | 12° |
| 6 | 60° | 12° |
| 7 | 72.5° | 12° |

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★Figure 3-6.

2. "Descent Check" and "Before Landing" checklists - Complete.

3. Fuel - Dump. (As required)

**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.

4. Fuel quantity and distribution - Check.

5. Wing sweep handle - Set.

   Refer to figure 6-13.

6. Compute emergency landing angle-of-attack, backup airspeed and ground roll distance from figures 3-6 and 3-8.

7. Landing brief - Completed.

8. Control system switch - T.O. & LAND.

9. Landing gear - Down and check.

10. Elevator position - Check at 11° angle-of-attack. (1° TED to 10° TEU for wing sweeps 16° - 26°; 2° to 10° TEU for wing sweeps greater than 26°)

**WARNING**

A crosswind landing will require a crabbed attitude at touchdown. 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.

11. Landing - Fly nose smoothly onto runway.

Power should be reduced to IDLE immediately upon touchdown. The nosewheel should be lowered to the runway as soon as possible if stopping distance is a factor. The aircraft nose will not have the same tendency to fall through, consequently, it will be necessary to "fly" the nose smoothly onto the runway. When brakes are applied, full pedal deflection anti-skid braking in a three-point attitude will provide most effective deceleration for both dry and wet runways.

**Note**

The inboard ground roll spoilers lock down at 45 degrees wing sweep. The outboard ground roll spoilers lock down at 47 degrees wing sweep. If possible, sweep wings forward to obtain ground roll spoiler operation.

12. Hook - As required.

**WARNING**

- The wheel blowout plugs may relieve tire pressure after any high-speed braking effort. Refer to "Brake Limitations," Section V.
- Call the fire department after any emergency landing that 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.

**If Missed Approach Is Accomplished:**

13. Control system switch - As required.

## SLATS UP (GAIN CHANGER LAMPS ON), FLAPS MORE THAN 5 DEGREES LANDING

For a slats up (less than 70 percent down) and flaps more than 5 degrees landing, use the following: fly a shallow, straight-in approach at 7 degrees AOA. AOA should be read from the AOA indicator and not the AOA indexer. The landing can be accomplished at 16 to 26 degrees wing sweep depending on the center-of-gravity considerations.

Speed will vary from the no flap/slat approach speeds. With 6 degrees flap extension, add 15 percent for 16 degrees wing sweep and 20 percent for 26 degrees wing sweep. As flaps are extended the speeds will decrease. At 15 degrees flap extension, use normal no flap/slat speeds with 16 degree wing sweep and add 10 percent with 26 degree wing sweep. Use figures 3-6 or 3-7 to obtain these airspeeds; however, AOA is the primary speed reference for the approach and landing. Figure 3-8 may be used to approximate ground roll distance.

## AIRSPEEDS FOR LANDING @ 7° AOA

| GROSS WEIGHT | 60k | | 65k | | 70k | | 75k | | 80k | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| WING SWEEP | 16° | 26° | 16° | 26° | 16° | 26° | 16° | 26° | 16° | 26° | R |
| FLAPS 6° | 206 | 233 | 215 | 245 | 223 | 254 | 230 | 264 | 238 | 272 | R |
| FLAPS 15° | 179 | 213 | 187 | 224 | 194 | 233 | 200 | 242 | 207 | 250 | R |

★ **Figure 3-7.**

**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 buildup (above 12 degrees) which could result in stall or uncontrolled roll off.
- Maintain an airspeed compatible with aircraft configuration and gross weight to ensure that 7 degrees angle-of-attack is not exceeded during maneuvering flight prior to final approach.

1. Flap and slat switch - EMER.
2. Angle-of-attack - Do not exceed 7 degrees.

R 3. Control system switch - T.O. & LAND.

4. "Descent Check" and "Before Landing" checklists - Complete.

**Note**

Wing sweep may require readjustment between 16 and 26 degrees to keep the elevator position between 0 to 7 degrees TEU.

5. Fuel - Dump. (As required) (At 7 degrees angle-of-attack maintaining 0 to 7 degrees TEU)

Fuel may be dumped as required to reduce landing speeds. Fuel should be dumped at 7 degrees angle-of-attack in stablilized flight. Closely monitor fuel distribution to avoid an aft cg condition which cannot be compensated for by moving the wing aft. Monitor the elevator position to assure that it remains within the 0 to 7 degrees TEU limits. It may be necessary to readjust the wing sweep between 16 and 26 degrees to keep it within limits.

6. Fuel quantity and distribution - Checked.
7. Landing brief - Completed.
8. Landing gear - Down and check.
9. Elevator position - Check at 7 degrees angle-of-attack. (0 to 7 degrees TEU for wing sweeps 16 degrees to 26 degrees)

The wing sweep may be moved forward as long as the elevator remains between 0 to 7 degrees TEU.

**WARNING**

A crosswind landing will require a crabbed attitude at touchdown. 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.

10. Approach angle-of-attack - 7 degrees.

# EMERGENCY LANDING GROUND ROLL DISTANCE (FLAPS UP)

DATA BASIS: FLIGHT TEST
DATE: SEPTEMBER 1984

CONFIGURATION:
- SLATS RETRACTED (EXCEPT AS NOTED)
- FLAPS RETRACTED

CONDITIONS:
- CG – FUEL SYSTEM IN AUTO MODE
- GROUND ROLL SPOILERS EXTENDED (16° TO 40° SWEEP)
- MAXIMUM BACKSTICK (NOSE WHEEL ON RUNWAY)

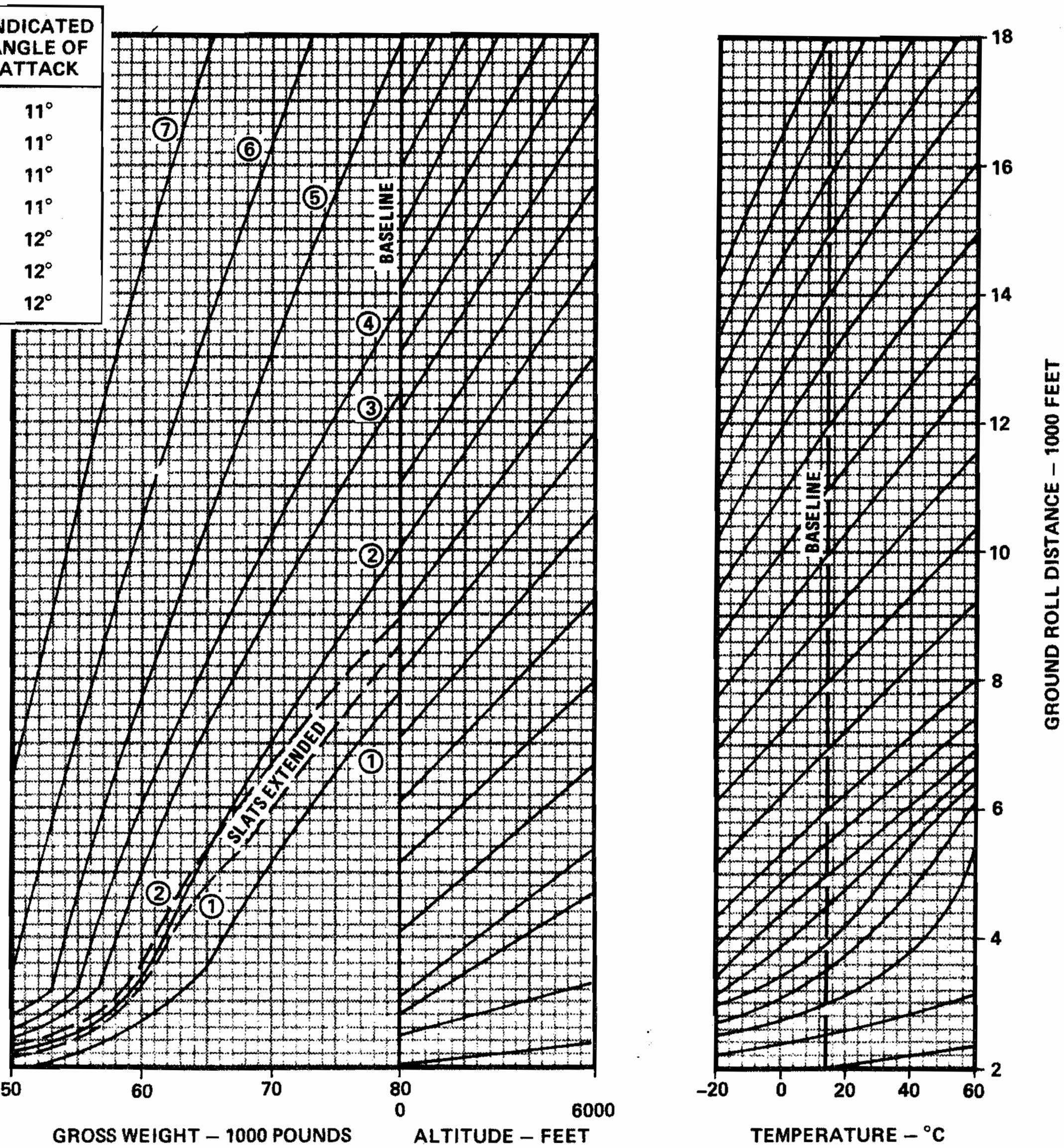
FUEL GRADE: JP-4
ENGINES: TF30-P-3

NOTES:
1. STOPPING DISTANCE CONSIDERS AERODYNAMIC DECELERATION TO MAX BRAKE SPEED (AS DETERMINED FROM "BRAKE ENERGY LIMITS", SECTION V) AND THEN APPLICATION OF MAX EFFORT BRAKING.
2. INCREASE GROUND ROLL 8% IF SPOILERS NOT USED.
3. USE RCR CORRECTIONS IN T.O. 1F-111(E)A-1-1 FOR WET RUNWAY (SPOILERS EXTENDED).
4. INCREASE GROUND ROLL 200 FT FOR EACH KNOT (KIAS) ABOVE RECOMMENDED TOUCHDOWN SPEED.

**WARNING**

REFER TO SECTION V FOR BRAKE ENERGY LIMITS AND SECTION VI FOR CG LIMITS

| WING SWEEP | | INDICATED ANGLE OF ATTACK |
|---|---|---|
| 1 | 16° | 11° |
| 2 | 26° | 11° |
| 3 | 35° | 11° |
| 4 | 40° | 11° |
| 5 | 50° | 12° |
| 6 | 60° | 12° |
| 7 | 72.5° | 12° |

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★ Figure 3-8.

11. Landing - Fly nose smoothly onto runway.

    Power should be reduced to IDLE immediately upon touchdown and nosewheel lowered to the runway as soon as possible. The aircraft nose will not have the same tendency to fall through, consequently, it will be necessary to "fly" the nose smoothly onto the runway. When brakes are applied, full pedal deflection anti-skid braking in a three-point attitude will provide most effective deceleration for both dry and wet runways.

12. Hook - As required.

**WARNING**

- The wheel blowout plugs may relieve tire pressure after any high-speed braking effort. Refer to "Brake Limitations", Section V.
- Call the fire department after any emergency landing that 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.

## SLATS DOWN (GAIN CHANGER LAMPS OUT), FLAPS 15 DEGREES OR LESS LANDING

This flap/slat configuration will provide flight characteristics similar to those described under "Slats Up (Gain Changer Lamps On), Flaps 5 Degrees or Less Landing." The following procedure is aligned with that procedure but steps not applicable to this configuration are deleted.

1. Flap and slat switch - EMER.
2. "Descent Check" and "Before Landing" checklists - Complete.
3. Fuel - Dump. (As required)

**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.

4. Fuel quantity and distribution - Check.
5. Wing sweep handle - Set.

   Refer to figure 6-12.

6. Compute backup airspeed for 11 degrees angle-of-attack using figure 3-6 or 3-9.
7. Landing brief - Completed.
8. Landing gear - Down and check.
9. Elevator position - Check at 11 degrees angle-of-attack. (2 to 10 degrees TEU for 16 to 26 degrees wing sweep).

**WARNING**

A crosswind landing will require a crabbed attitude at touchdown. 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.

10. Landing - Fly nose smoothly onto runway.

    Power should be reduced to IDLE immediately upon touchdown. The nose wheel should be lowered to the runway as soon as possible if stopping distance is a factor. The aircraft nose will not have the same tendency to fall through, consequently, it will be necessary to "fly" the nose smoothly onto the runway. When brakes are applied, full pedal deflection anti-skid braking in a three-point attitude will provide most effective deceleration for both dry and wet runways.

11. Hook - As required.

# Emergency Landing Airspeeds

(SLATS EXTENDED/PARTIAL FLAPS)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONFIGURATION:
- SLATS EXTENDED
- PARTIAL FLAPS
- WING SWEEP 16 DEGREES

CONDITIONS:
- CG — FUEL SYSTEM IN AUTO MODE

NOTES:
1. FOR FLAPS RETRACTED, REFER TO FIGURE 3-6.
2. FOR FLAPS EXTENDED LESS THAN 15 DEGREES, SUBTRACT 10 KIAS FROM APPROACH SPEED TO OBTAIN TOUCHDOWN SPEED.
3. FOR FLAPS EXTENDED MORE THAN 15 DEGREES BUT LESS THAN 34 DEGREES, SUBTRACT 6 KIAS FROM APPROACH SPEED TO OBTAIN TOUCHDOWN SPEED.
4. ADD 0.5 KIAS FOR EACH PYLON STATION WITH STORE PRESENT

**WARNING**

REFER TO SECTION V FOR BRAKE ENERGY LIMITS AND SECTION VI FOR CG LIMITS

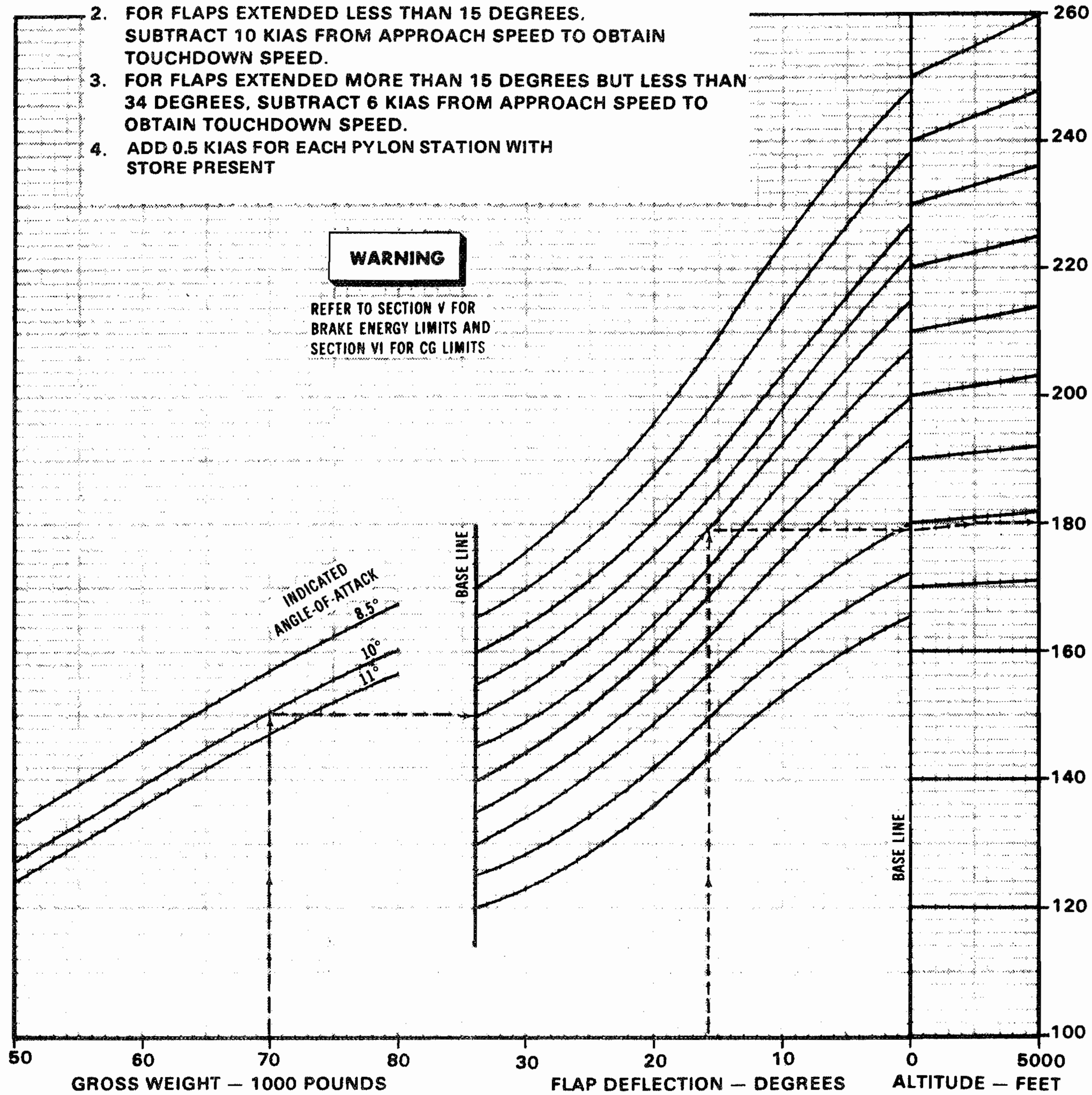


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Figure 3-9. (Sheet 1)

# Emergency Landing Airspeeds

## (SLATS EXTENDED/PARTIAL FLAPS)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONFIGURATION:
- SLATS EXTENDED
- PARTIAL FLAPS
- WING SWEEP 26 DEGREES

CONDITIONS:
- CG — FUEL SYSTEM IN AUTO MODE

NOTES:
1. FOR FLAPS RETRACTED, REFER TO FIGURE 3-6.
2. FOR FLAPS EXTENDED LESS THAN 15 DEGREES, SUBTRACT 10 KIAS FROM APPROACH SPEED TO OBTAIN TOUCHDOWN SPEED.
3. FOR FLAPS EXTENDED MORE THAN 15 DEGREES BUT LESS THAN 34 DEGREES, SUBTRACT 6 KIAS FROM APPROACH SPEED TO OBTAIN TOUCHDOWN SPEED.
4. ADD 0.5 KIAS FOR EACH PYLON STATION WITH STORE PRESENT

WARNING

REFER TO SECTION V FOR BRAKE ENERGY LIMITS AND SECTION VI FOR CG LIMITS

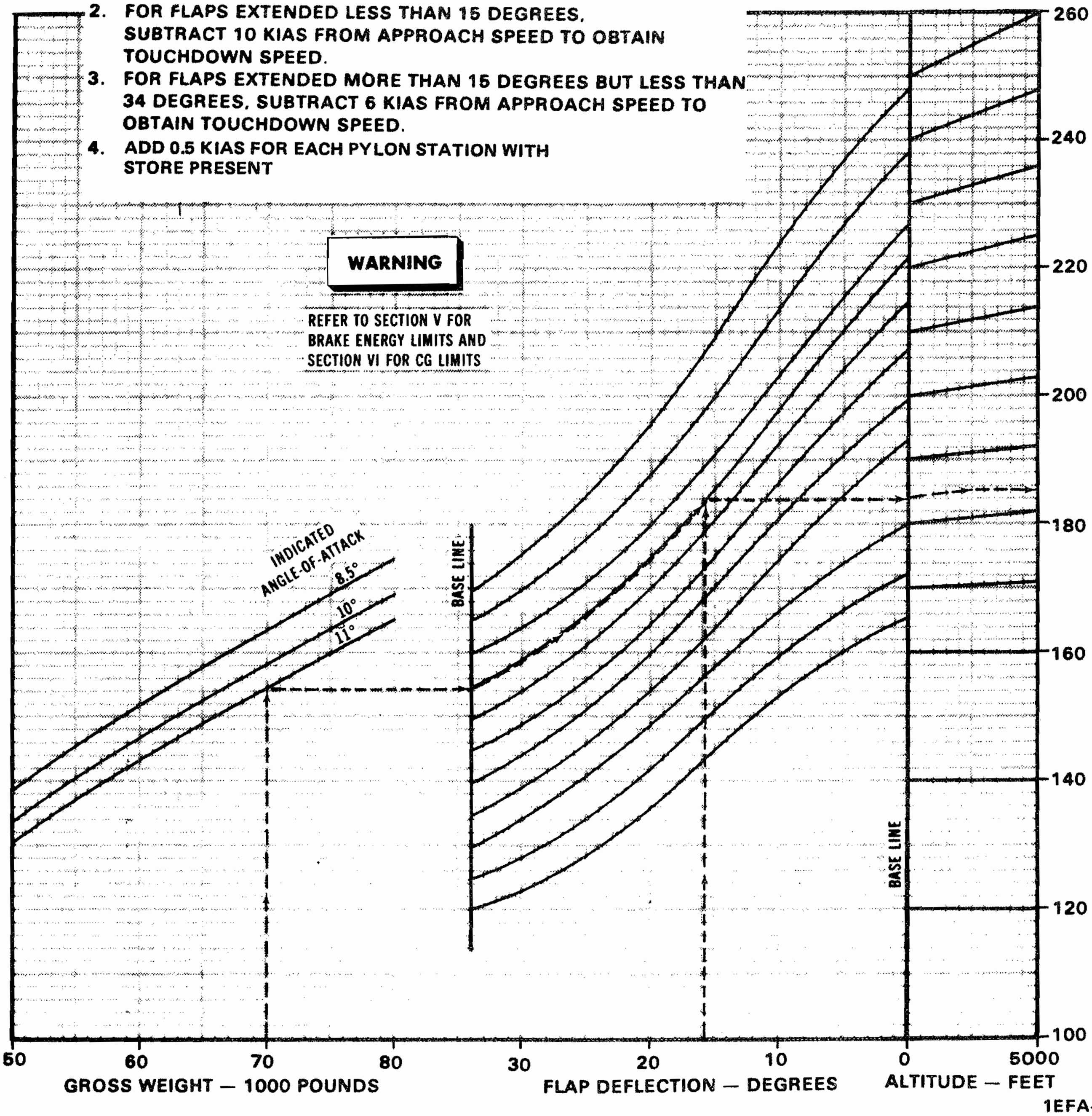



Figure 3-9. (Sheet 2)

**WARNING**

- The wheel blowout plugs may relieve tire pressure after any high-speed braking effort. Refer to "Brake Limitations," Section V.
- Call the fire department after any emergency landing that 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.

## SLATS DOWN (GAIN CHANGER LAMPS OUT), FLAPS MORE THAN 15 DEGREES LANDING

This flap/slat configuration will provide flight characteristics similar to a normal landing.

1. Flap and slat switch - EMER.
2. Fly final approach at 10 degrees angle-of-attack.
3. Backup airspeed - Compute. (Refer to figure 3-9)
4. Refer to normal landing procedures.

## ASYMMETRIC SLATS OR FLAPS DURING EXTENSION

1. Apply lateral stick as necessary followed with rudder for roll control if necessary.
2. Flap/slat handle - Return toward previous position until rolloff stops.
3. Flap and slat switch - EMER.

**CAUTION**

Placing the flap and slat switch to EMER relieves hydraulic pressure to the flap motor and isolates the function of the flap/slat handle, and may prevent further damage or undesirable flap movement.

**If Rolloff Is Reduced to Zero:**

4. Land as soon as practical, using the procedure appropriate to the final flap/slat configuration.

**If Rolloff Is Still Present:**

5. Land as soon as practical, using "Asymmetric Slat Landing" or "Asymmetric Flap Landing" procedures this section.

## ASYMMETRIC SLAT LANDING

1. Control system switch - T.O. & LAND. R
2. Rudder authority switch - FULL.
3. "Descent Check" and "Before Landing" checklists - Complete.
4. Fuel - Dump. (As required)
5. Backup airspeed - Compute for 11 degrees AOA. (Refer to figure 3-6 and use slats extended condition.)
6. Controllability check - Accomplished. (Above 5,000 feet AGL)
7. Fuel quantity and distribution - Check.
8. Wing sweep - Set.
9. Landing brief - Completed.
10. Landing gear - Down and check.
11. Elevator position - Check at 11 degrees angle-of-attack. (2 to 10 degrees TEU at all wing sweeps)
12. Fly long, shallow, straight-in approach at 11 degrees angle-of-attack. (On final approach, maintain ground track with rudder 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. Any of the above can result in excessive sink rate buildup which may be difficult to correct at approach altitudes.

- Maintain an airspeed compatible with aircraft configuration and gross weight to ensure that 10 degrees angle-of-attack is not exceeded during maneuvering flight prior to final approach phase.

13. Landing - Fly nose smoothly onto runway.

    Power should be reduced to IDLE immediately upon touchdown and nosewheel lowered to the runway as soon as possible. The aircraft nose will not have the same tendency to fall through, consequently, it will be necessary to "fly" the nose smoothly onto the runway. When brakes are applied, full pedal deflection anti-skid braking in a three point attitude will provide most effective deceleration for both dry and wet runways.

**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 are extended. Lateral control, augmented with rudder as necessary, should be initiated upon touchdown to maintain desired ground track.

14. Hook - As required.

**WARNING**

- The wheel blowout plugs may relieve tire pressure after any high-speed braking effort. Refer to "Brake Limitations", Section V.

- Call the fire department after any emergency landing that 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.

## ASYMMETRIC FLAP LANDING

With asymmetric flaps, angle-of-attack indication accuracy is affected by Mach number and by the position of the left flap. Angle-of-attack should not be used as the primary speed reference. Airspeed for best handling and final approach can be determined from figure 3-10 (Sheet 1 or Sheet 2). Since the cockpit flap indicator will indicate only the position of the working flap, the position of the failed flap must be estimated. The initial flap handle movement, direction and magnitude of roll tendencies, and the cockpit flap indicator must all be evaluated to determine the relative flap positions and degree of asymmetry. If possible, have a chase aircraft provide an estimate of the left and right flap positions. Compare this information with the cockpit flap position indicator to determine the working (indicated) and failed (estimated) flap positions. If cg conditions permit, an approach wing sweep of 26 degrees will lessen lateral control requirements.

1. Rudder authority switch - FULL.
2. Wing sweep - Set to 26 degrees to reduce lateral control requirements (if cg conditions permit)
3. "Descent Check" and "Before Landing" checklists - Complete.
4. Ground roll spoiler switch - OFF.
5. Fuel - Dump. (As required)
6. Fuel quantity and distribution - Check.
7. Compute airspeeds for best handling and approach. (Use figure 3-10 Sheet 1 or Sheet 2)
8. Landing brief - Completed.
9. Controllability check - Accomplished. (Above 5,000 ft AGL)
10. Landing gear - Down and check.

# EMERGENCY LANDING AIRSPEEDS

## (SLATS EXTENDED/ASYMMETRIC FLAPS—WING SWEEP 16°)

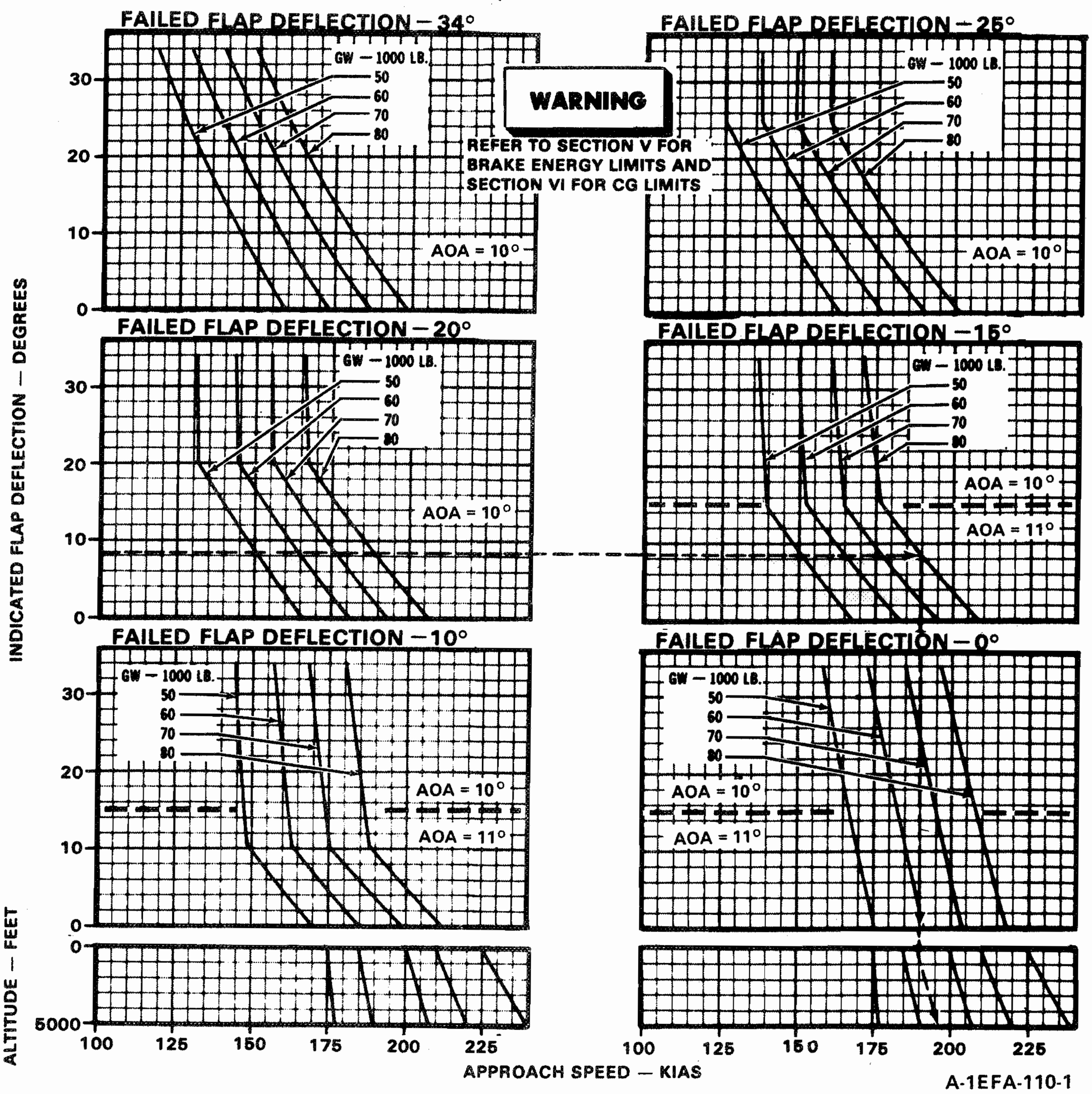
DATA BASIS: ESTIMATED
DATE: 16 SEPTEMBER 1985

**CONFIGURATION:**
- SLATS EXTENDED
- ASYMMETRIC FLAPS
- WING SWEEP—16°

**CONDITIONS:**
- CG—FUEL SYSTEM IN AUTO MODE

FUEL GRADE: JP-4
ENGINES: TF30-P-3

NOTES:
1. ADD 10 KIAS TO APPROACH SPEED FOR BEST HANDLING AIRSPEED.
2. FLAPS 15° OR LESS, SUBTRACT 10 KIAS TO OBTAIN TOUCHDOWN SPEED.
3. FLAPS MORE THAN 15°, SUBTRACT 6 KIAS TO OBTAIN TOUCHDOWN SPEED.




★ Figure 3-10. (Sheet 1)

# EMERGENCY LANDING AIRSPEEDS

## (SLATS EXTENDED/ASYMMETRIC FLAPS—WING SWEEP 26°)

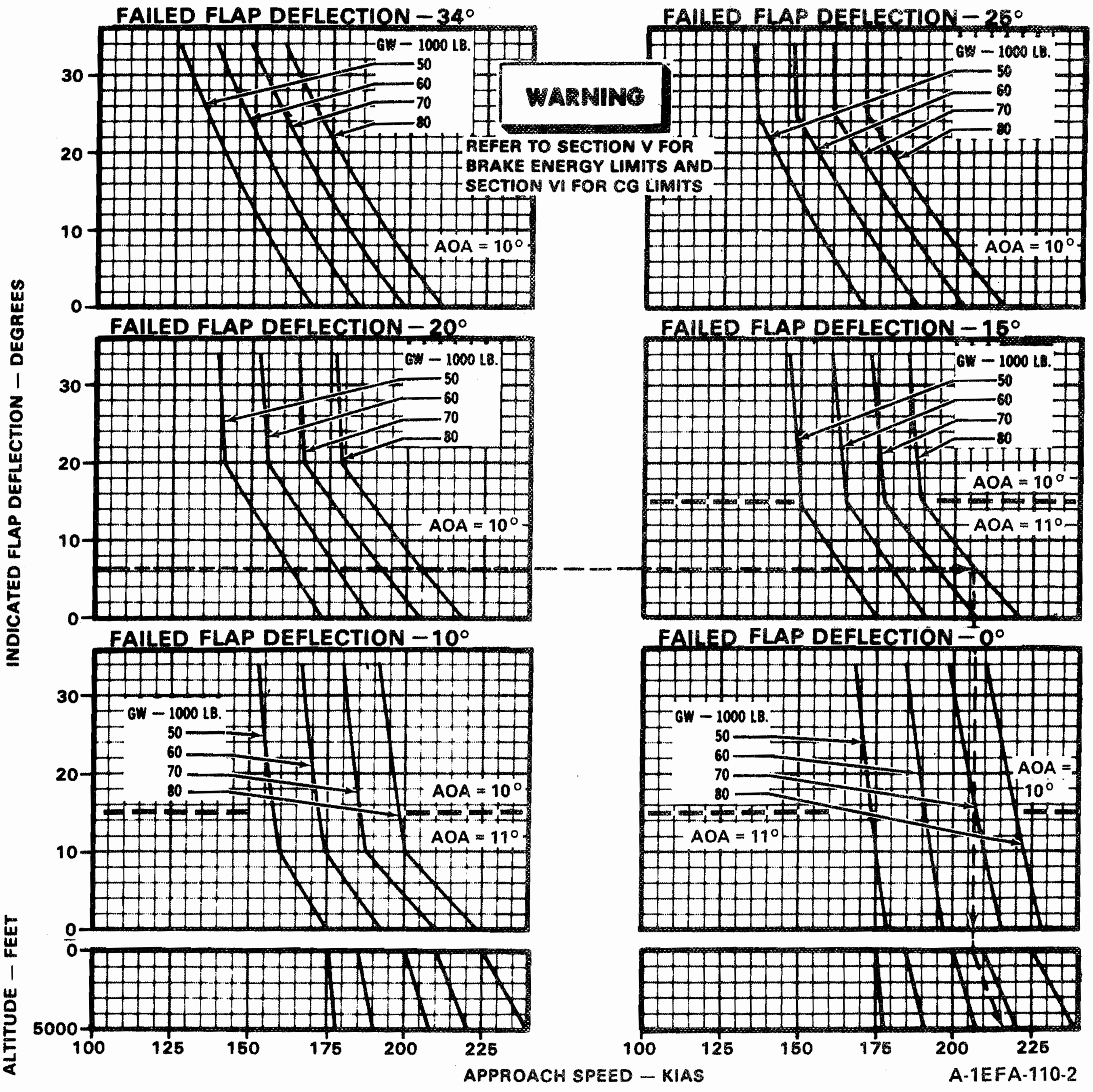
DATA BASIS: ESTIMATED
DATE: 16 SEPTEMBER 1985

**CONFIGURATION:**
- SLATS EXTENDED
- ASYMMETRIC FLAPS
- WING SWEEP—26°

**CONDITIONS:**
- CG—FUEL SYSTEM IN AUTO MODE

FUEL GRADE: JP-4
ENGINES: TF30-P-3

NOTES:
1. ADD 10 KIAS TO APPROACH SPEED FOR BEST HANDLING AIRSPEED.
2. FLAPS 15° OR LESS, SUBTRACT 10 KIAS TO OBTAIN TOUCHDOWN SPEED.
3. FLAPS MORE THAN 15°, SUBTRACT 6 KIAS TO OBTAIN TOUCHDOWN SPEED.




★ Figure 3-10. (Sheet 2)

11. Elevator position - Check at best handling airspeed. (2 to 10 degrees TEU at all wing sweeps)

12. Fly a long, shallow, straight-in approach.

**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 buildup, which may be difficult to correct at approach altitudes.

- Maintain constant ground track by use of rudder and lateral control. A combined wing-low, crabbed approach and touchdown are recommended since the use of rudder to align the aircraft with the runway can cause a rapid rolloff. Large flap asymmetries combined with unfavorable crosswinds can result in touchdown crab angles which exceed 10 degrees.

13. Landing - Fly nose smoothly onto runway.

    Power should be reduced to IDLE immediately upon touchdown and nosewheel lowered to the runway as soon as possible. The aircraft nose will not have the same tendency to fall through; consequently, it will be necessary to "fly" the nose smoothly onto the runway. When brakes are applied, full pedal deflection and anti-skid braking in a three-point attitude will provide most effective deceleration for both dry and wet runways.

14. Ground roll spoiler switch - BRAKE.

**WARNING**

Aircraft will initially tend to veer in the direction away from the most extended flap when the spoiler brakes extend. Lateral control, augmented with rudder as necessary, should be used to maintain desired ground track.

15. Hook - As required.

**WARNING**

- The wheel blowout plugs may relieve the pressure after any high-speed braking effort. Refer to "Brake Limitations," Section V.

- 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.

## SINGLE ENGINE OPERATION

### 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. Aircraft gross weight is especially critical during hot day and high density altitude conditions; however, under any conditions, consideration should be given to reduce gross weight by dumping or R burning fuel, unless the nature of the emergency dictates an immediate landing. The optimum method for reducing gross weight by burning fuel, is to select afterburner on the good engine and extend the speed brake. This results in more drag than with the gear extended. A straight-in approach should be flown with flaps set at 25 degrees. This is the optimum flap setting in case of a go-around. The approach should be planned to minimize the use of afterburner. If conditions permit, do not configure the aircraft until established on final. When descent is established on final, increase angle-of-attack slowly to 8.5 degrees. When landing is assured, slowly increase angle-of-attack to 10 degrees. 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. The hydraulic system failure caution lamps for the failed engine may not light until the aircraft is slowed for landing. If an engine is shut down by depressing the fire pushbutton, the hydraulic caution lamps may not light due to trapped pressure.

**WARNING**

During maneuvering flight, prior to final approach phase, angle-of-attack should not exceed 8.5 degrees.

1. Operating engine - Maintain 85 percent rpm minimum. (90 percent rpm if operating on emergency generator.)

2. Air source selector knob - As required.

3. Fuel - Dump or burn to reduce gross weight (If time and conditions permit)

**WARNING**

Due to possible fuel dump line damage, fuel should not be dumped after a wheel well hot, bleed air duct failure, or fuselage or engine fire unless thrust requirements or landing conditions dictate.

4. Wing sweep - Adjust slowly for landing. (One degree per second) (See figure 6-12).

5. "Descent Check" and "Before Landing" checklists - Complete.

6. Backup airspeed - Compute. Refer to figure 3-9.

7. Translating cowl - Check open at Mach 0.44. (Operating engine)

8. Rudder authority switch - FULL.

9. Flaps/slats - 25 degrees.

**Note**

As a last resort, when stopping distance is critical, a full flap "on-speed" approach should be used. If full flaps are used, they should be selected prior to short final because of the angle-of-attack and airspeed changes which occur during flap extension. With full flaps, go-around capability may be marginal and obstacle clearance should be considered.

10. Final approach:

    a. Landing gear - Down. (Use emergency extension procedures if operating on emergency generator)

**WARNING**

In level flight, thrust required may exceed thrust available at MIL power after lowering gear and flaps. (See Section VI)

    b. Angle-of-attack - 8.5 degrees when descent is established. (When landing is assured, slowly increase to 10 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 buildup which may be difficult to arrest at approach altitudes.

    c. Glide slope - Normal. (Approximately 600 feet per minute)

11. Elevator position - Check at 8.5 degrees angle-of-attack.

    a. Wings at 16 degrees - 4 to 8 degrees TEU.

    b. Wings at 26 degrees - 2 to 8 degrees TEU.

## 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 will be inoperative. Hydraulic system pressures should be monitored closely through 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 Generator," "Single Engine Landing" and "Single Engine Go-Around" procedures, this section.

## 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 15 seconds to increase thrust from IDLE to MIL with full bleed from the accelerating engine.

1. Throttle - MAX AB. (Operating engine)

2. Landing gear - As required.

   Reduction of rudder authority will be felt as the rudder authority switch is placed to AUTO. This may be felt as a kickback if more than 7.5 degrees of rudder deflection is being held.

3. Air source selector knob - OFF or EMER. (As required)

   **Note**

   - Significant thrust is gained with the air source selector in OFF or EMER.
   - With the air source selector in OFF or EMER, no servo air will be available for throttle boost or fuel tank pressurization. Loss of throttle boost will require a significant increase in force required to move the throttle (approximately 40 pounds with maximum throttle friction). Lack of tank pressurization will degrade fuel dump.

4. Climb:

   a. After close-in obstacles are cleared, establish an attitude that will clear terrain and retract the flaps/slats at a rate to maintain 8.5 degrees angle-of-attack. Best single engine climb speed will equate to 8.5 degrees angle-of-attack.

   **WARNING**

   Excessive angle-of-attack may result from retracting flaps too rapidly.

5. Air source selector knob - As required.

6. Rudder authority switch - As required.

This is the last page of Section III.

# SECTION IV

# CREW DUTIES

## TABLE OF CONTENTS



The purpose of this section is to provide a compact collection of material wherein each crewmember 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. Refer to Section II for turn-on procedures and classified supplement T.O. 1F-111(E)A-1-3 for TJS related operations.

## CREW COORDINATION

Coordination of actions within a crew is of prime importance to ensure 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 crewmember's duties will contribute immeasurably toward crew coordination. Each crewmember must be constantly on the alert and should notify the responsible crewmember 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 crewmembers. Prior to flight both crewmembers 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. ECM activities.
6. Normal and emergency communications procedures.
7. Penetration, approach, missed approach, landing patterns, altitudes, and obstructions at both destination and alternate airfields.

Prior to accomplishment of any of the following, coordination between crewmembers 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.
4. Changing interphone control panel setting.

## 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.

## ELECTRONIC WARFARE OFFICER'S DUTIES

The EWO will ensure that sufficient mission preparation is accomplished, in accordance with command directives, to successfully accomplish the briefed mission. During all phases of flight, the EWO will ensure 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 EWO must respond with the required corrective actions and checklists. During critical phases of flight, the EWO will monitor aircraft configuration, flight and engine instruments, and terrain clearance to ensure 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.

## INERTIAL NAV SYSTEM OPERATION

### NORMAL MODE OPERATION

During normal operation, the inertial nav mode selector knob is set to either the GREAT CIRCLE or SHORT RANGE 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 updating or 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.

### 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 variations 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. Inertial nav mode selector knob - ALIGN.

**CAUTION**

To prevent equipment damage allow 5 minutes for gyro spin-down prior to returning the knob to ALIGN.

4. MAN FIX pushbutton - Depress.
5. Present position latitude and longitude counters - Check and set.
6. Inertial 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. Inertial nav mode selector knob - ALIGN.
11. Platform align indicator lamp - On steady within 3 minutes after heat lamp goes out and flashing within an additional 5 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 indicates the platform is aligned.

**Note**

When in the align mode, the magnetic heading synchronization is an indicator of align status. 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. Inertial 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. Inertial 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. Inertial 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 funciton of ambient temperature, local latitude, aircraft attitude and component tolerances. Under optimum conditions the time may be as low as 30 seconds and under worst conditions the time can be as long as 4 minutes.

6. Inertial nav mode selector knob - GREAT CIRCLE or SHORT RANGE.

Place the inertial nav mode selector knob to GREAT CIRCLE or SHORT RANGE (prior to moving the platform alignment control knob from RAPID ALIGN) after the platform align indicator lamp starts flashing and before moving the aircraft.

7. Platform alignment control knob - NORMAL.

   Operation of the Stable Platform has not been degraded if the rapid align control knob has been inadvertently left in RAPID ALIGN.

### 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. Inertial 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. Inertial 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. Inertial nav mode selector knob - GREAT CIRCLE or SHORT RANGE.

   At 100 seconds after step 4 and before moving the aircraft place the inertial 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 inertial 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. Inertial 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. Inertial nav mode selector knob - HEAT.

9. Inertial 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. Inertial nav magnetic variation counter - Set to local value.

2. Inertial nav platform alignment control knob - NORMAL or RAPID ALIGN.

3. Inertial 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. Inertial nav magnetic variation counter - Set to local value.

3. Inertial 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. Inertial 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.

## 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 inertial nav mode selector knob from OFF.
- Do not move the inertial nav mode selector knob from OFF on a system that has been preset for stored heading rapid alignment until ready for alignment.

1. Inertial 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 inertial nav mode selector knob is in ALIGN.

1. Inertial 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 nav 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 nav 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 nav radar ground velocity and ground auto mode cursors will appear on the new destination if it is in range and the inertial nav mode selector knob is in any position other than GREAT CIRCLE, HEAT or OFF.

4. Destination counters - Check and refine as desired.

## ALTITUDE ALIGN

The purpose of altitude align is to null previous altitude calibration correction values to a zero value. This procedure allows uncorrected CADC pressure altitude inputs to the NC sighting circuitry.

1. Inertial nav mode selector knob - ALIGN or above.

2. Altitude/test selector knob - ALIGN.

3. GO lamp - Lighted.

**Note**

The cursor range counter should indicate zero absolute altitude.

4. Altitude/test selector knob - NORM.

## ALTITUDE CALIBRATION

Altitude calibration is necessary prior to position updating, and AILA letdown. It is recommended that, circumstances permitting, calibration be made at pressure altitude, and speed at which up-dating 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 5,000 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.

**Note**

The cursor range counter should indicate absolute altitude equal to that indicated on the radar altimeter.

5. Altitude/test selector knob - NORM.

### High Altitude Calibration (Using Nav Radar)

Altitude calibration should be accomplished in accordance with this procedure when flying over level terrain of known elevation at altitudes normally above 5,000 feet or below 5,000 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. Inertial nav mode selector knob - SHORT RANGE.

4. Fix mode TARGET selector button - Depress.

5. Nav radar function selector knob - XMIT.

6. Nav radar mode selector knob - GND AUTO or GND VEL.

7. Nav radar sector switch - Wide scan.

8. Range select knob - Set minimum range compatible with altitude.

9. Sensitivity time control knob - OFF.

10. Nav radar beta switch - NORM.

11. Altitude/test selector knob - CAL.

12. Antenna tilt - Minus 30°. (−10° TFR on)

13. Using the nav 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.

## AIRCRAFT POSITION UPDATING

The need for aircraft position updating is indicated primarily by a position error observed on the nav radar scope. Nav 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 inertial nav system. There are other reasons why the nav 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 nav 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 nav radar is operating.

1. Altitude calibration - Completed.

2. Fix mode TARGET selector button - Depress.

3. Destination position counters - Set.

4. Fixpoint elevation counter - Set.

5. Inertial nav mode selector knob - SHORT RANGE, or above.

6. Nav radar mode selector knob - GND AUTO or GND VEL.

7. Destination/present position selector switch - PP.

8. Nav radar range selector knob - Use lowest range setting possible.

9. Postively identify target.

10. Nav 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. Nav radar mode selector knob - GND VEL or GND AUTO.

5. Destination/present position selector switch - DEST.

6. Inertial nav mode selector knob - SHORT RANGE.

7. Fix mode TARGET selector button - Depress, after destination position counters have stopped slewing.

8. Nav radar range knob - Use lowest range setting possible.

**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 nav 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 nav radar or TFR group 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 nav 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 nav 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.

## 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.

## 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/attitude 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, and slant range 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 inertial nav generated cursors closely track selected targets on the nav 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 nav 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, ground-speed 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 nav 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 difference 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.

## 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.

- Transmitter selector knob - HF.

- HF monitor knob - On.

- Mode selector knob - Desired mode.

- Desired frequency - Set.

A mute period will indicate the RT unit is setting to the new frequency. The system should not be keyed during this period. If the frequency was already set when the system was turned on, rotate the 1-kilohertz knob one digit off frequency and then back to the desired frequency. This will allow the R-T unit to properly tune to the desired frequency.

- RF sensitvity knob - Adjusted.

Adjust the RF sensitivity knob to receive signals just above the noise level of the receiver, then adjust the interphone monitor knob for a comfortable listening level. Proper balance is indicated when background noise is just audible and a weak signal is raised to comfortable level.

- Microphone switch - TRANS.

After a frequency change, a 1-kilohertz tone will be heard when the mike switch is first placed to TRANS. This indicates that the antenna coupler is tuning to the RT unit frequency set up in step 4. When the tone ceases, the tuning cycle is complete and a side tone will be heard when transmitting.

- Transmitter selector knob - As desired.

## JAM RESISTANT UHF RADIO PROCEDURES

1. Initial TOD (must be on internal aircraft power).

   a. Manual frequency selectors - Set to TSS/RSG broadcast frequency.

   b. A–3–2–T knob - Move to T momentarily and release.

   c. Listen for short tone signal (beep). If no signal is heard within 10 sec proceed to step 2 or 3.

2. TOD update from another user.

   a. Request TOD from another user.

   b. Requester - Move A–3–2–T knob to T and release.

   c. Other User - Momentarily depress tone button to transmit TOD within one minute of step b. (Short tone signal should be heard)

3. Initiating individual TOD. (No TOD available from a TSS/RSG master frequency)

   a. A–3–2–T knob - T and hold.

   b. Tone Button - Depress momentarily.

c. A–3–2–T knob - Release. (No tone signal will be heard)

d. To confirm radio has TOD - Depress and hold tone button. Listen for short TOD tone followed by normal radio tone.

4. WOD Insertion.

a. Mode selector knob - PRESET.

b. Preset channel selector knob - Channel 15.

c. Manual frequency selectors - Set WOD.

d. Channel preset pushbutton - Depress momentarily.

e. Repeat steps b, c, and d for each additional WOD segment, storing successive segments in preset channels 16 thru 20.

f. Transfer WOD to volatile memory - Cycle out of channel 20, then back to channel 20. Successively select each preset channel used for WOD storage. At each storage location user should hear a single tone to indicate successful WOD insertion. At last storage location user should hear a double tone indicating the last segment has been read and stored.

g. Conduct a radio check in active mode on a prebriefed net number.

5. Active mode operation.

a. Mode selector knob - MANUAL.

b. Function selector knob - MAIN or BOTH.

c. Manual frequency selectors - Set assigned net number. (A plus three digits)

d. A steady tone will be heard if the A–3–2–T knob is moved to A without a TOD.

e. Conference capability is disabled with 25 set in the hundredths/thousandths position of the manual frequency selectors.

## 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

- Function selector knob - As required (REC, T/R, A/A REC, or A/A TR).
- Antenna selector switch - AUTO.
- Channel selector knobs - As required.
- Channel mode knob - Set.

  Use X mode for air-to-ground operations as no Y mode ground stations are presently in use. Y mode is recommended for air-to-air operations to preclude possible interference from X mode ground stations; however, all participating aircraft must have Y mode capability.

- Volume control knob - Adjust for desired volume level.
- ISC mode select knob - TACAN.
- Desired HSI course - Set.
- Check that the HSI and BDHI bearing pointers lock onto the ground station and that the pointer bearings agree within 3 degrees.
- Set the HSI course deviation indicator to zero. Check that the course readout and the bearing pointers agree within 3 degrees.
- Check that the TO/FROM indicator indicates the bearing selected in relation to the ground station.
- Check the audio signal for readability.
- If the TACAN signal becomes unusable, check that the OFF flags appear on the course deviation indicator and on the range indicator.
- The range indicator shall have a distance accuracy of ±0.5 mile or 3 percent of the distance from the station, whichever is greater.
- 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.
- Monitor attitude director indicator 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.

### System Press-to-Test

- Function selector knob - T/R (Allow 90 second warmup)
- HSI course - Set to 180 degrees.
- Press-to-test button - Depress and release. Observe HSI and test indicator for following indications:
  - Test indicator flashes momentarily.
  - Course and range warning flags come into view, if not already in view.
  - Bearing pointer slews to 270 degrees for approximately seven seconds.
  - Course and range warning flags go out of view.
  - Distance indicates 0.0 $\pm$0.5 (since negative values are not presented, a value of $-0.5$ will be portrayed as 399.5), bearing pointer slews to 180 $\pm$3 degrees. Course deviation indicator should be within one-half dot deflection from center and to-from indicator should indicate "to". All indications remain for approximately 15 seconds.
  - Course and range warning flags come into view until TACAN returns to normal operation.

### Inflight Operation

- Press-To-Test. If the test indicator lights during flight, perform a press-to-test as described above. If the indicator light remains lighted, repeat the test in the REC mode. If the light goes out in the REC mode, the malfunction is probably in the transmitter and bearing information can be used provided adequate cross-check is available. If the indicator light remains lighted, in both T/R and REC, all TACAN information should be considered unreliable.
- Automatic Self-Test. If the TACAN signal is lost, an automatic self-test is initiated. This is indicated by the bearing pointer turning to 270 degrees for approximately three seconds. The test indicator will normally not light during self-test. If the indicator lights, a system malfunction has occurred and a press-to-test should be accomplished. Changing channel mode or number will not by itself inititate a self-test.

## AUTOMATIC DIRECTION FINDER OPERATION

- UHF function selecıor knob - ADF.
- UHF mode selector switch - MANUAL, PRESET or GUARD

  Select the desired frequency with the preset channel selector knob or with the manual frequency selector knobs. Selection of GUARD will provide ADF information for 243.0 megahertz.
- BDHI mode selector switch - NAV.
- 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 or nonexistent and considerable background noise will be evident. If required, the UHF squelch switch can be placed to OFF to improve audio during ADF operation.
- ADF will not home on guard transmissions unless 243.0 megahertz is a preset channel, dialed in manually, or the mode selector knob is positioned to GUARD.

This is the last page of Section IV.

# SECTION V

# OPERATING LIMITATIONS

## TABLE OF CONTENTS



## INTRODUCTION

This section includes limitations that must be observed for safe and efficient operation of the engines and 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. A summary of limitations is shown in figure 5-2. Approved fuels are contained in figure 5-3.

**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.
- 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 100 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:

| | ONE ENGINE | BOTH ENGINES |
|---|---|---|
| AB Power Setting | See Notes 1, & 3 | See Notes 2, & 3 |
| All Zones 1,2,3 | 6 Minutes | 6 Minutes |
| Zones 4 & 5 (MAX) | 90 Seconds | 30 Seconds |

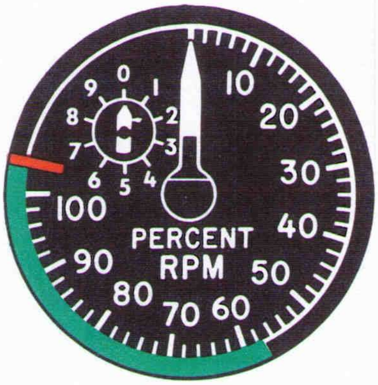
NOTES:

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

# INSTRUMENT MARKINGS

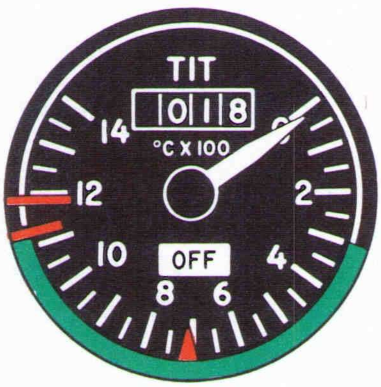
TF30-P3 ENGINES
BASED ON JP-4 FUEL

### TACHOMETER ($N_2$)




| | | |
|---|---|---|
| (Unmarked) | Idle RPM — percent (Ground operation) | 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 | SEE BELOW |
| | Starting: Ground start<br>Airstart (Unmarked) | 710°C<br>870°C | MOMENTARY MAXIMUM |
| | Maximum Military operation | 1130°C | 45 MINUTES |
| | Maximum and Partial Afterburner Limit (Below 660 KIAS) | 1130°C | 45 MINUTES |
| | Maximum and Partial Afterburner Limit (Above 660 KIAS) | 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.
- 35 psi – Minimum during idle.
- 50 psi – Maximum.

NOTE: Oil Pressure Fluctuations of ±5 PSI Within the Normal Range are Permissible.

### HYDRAULIC PRESSURE


- 2950 to 3250 psi – Normal range.
- 3250 psi – Maximum.

A-1EFA-31

**Figure 5-1.**

## SUMMARY TABLE OF LIMITATIONS

**CAUTION**

THOSE ITEMS INDICATED BY ASTERISK (*) HAVE EXPANDED INFORMATION ELSEWHERE IN THIS SECTION. WEIGHTS AND SPEEDS ARE BASED ON 30-PLY RATED MAIN GEAR TIRES.

**Note**

The summary table of limitations lists operating limits for the clean aircraft.

### WEIGHTS - POUNDS

| | |
|---|---|
| MAX TAXI AND TAKEOFF | 100,000 |
| MAX INFLIGHT | 100,000 |
| MAX LANDING (600 FPM RATE OF SINK) | 82,500 |

### LIMITS/SPEEDS - KIAS AND MACH, UNLESS OTHERWISE INDICATED

INFLIGHT:

- Clean — See Figure 5-4

MAX TAXI:

- Straight — 25 KTS Groundspeed (Up to 30,000 ft for GW above 82,500 lbs; up to 40,000 ft for GW 82,500 lb or less)
- Turning — 10 KTS Groundspeed

TIRE SPEEDS:

| | | |
|---|---|---|
| • Max Main Gear Tire | | 196 KTS Groundspeed |
| • Max Nose Gear Tire | (20 Ply) | 196 KTS Groundspeed |
| | (16 Ply) | 179 KTS Groundspeed |
| • Emer Ldg Main Gear Tire | | 217 KTS Groundspeed |
| • Emer Ldg Nose Gear Tire | (20 Ply) | 217 KTS Groundspeed |
| | (16 Ply) | 200 KTS Groundspeed |

*LANDING GEAR:

- 330 or 0.62 whichever is less
- 1.2 g during extension
- No intentional sideslip during gear transition is permitted

*FLAPS EXTEND/RETRACT:

- 0 - 30° 330 or 0.62 whichever is less
- 31° - Full 300 or 0.56 whichever is less

*SLATS ENTEND/RETRACT:

- 330 or 0.62 whichever is less

*SPEED BRAKE:

- 600 or 2.0 whichever is less

FUEL DUMP:

- 350 or 0.75 whichever is less
- Approx 1 g flight

TRANSLATING COWLS MUST BE FULLY CLOSED ABOVE:

- 415 or 0.90 whichever is less

AIR REFUEL RECPT:

- 400 or 1.0 whichever is less

CANOPY HATCH:

- 60 Kts (Relative Wind)

MANEUVER RESTRICTIONS:

- The aircraft shall be limited from initiating rolling maneuvers at normal load factors less than 1 g at all wing sweeps except 45 and 50 degrees. For 45 and 50 degrees wing sweep, the roll limitations shall be defined as follows.

**Figure 5-2. (Sheet 1)**

## SUMMARY TABLE OF LIMITATIONS (cont)

- Do not perform rolling maneuvers at less than 1 g at Mach numbers greater than 0.80 and altitude in excess of 10,000 feet.
- Rolling maneuvers of up to 90 degrees angle of bank, from an upright position, may be performed at load factors as low as zero g. Recovery must be initiated at a bank angle not greater than 45 degrees to avoid substantial bank angle overshoots.

- Do not perform rolling maneuvers in excess of 360 degrees at all wing sweeps.

### SYSTEM LIMITS

STARTER:

- Pneumatic (both) 5 consecutive; then 1 hr cooling
- Cartridge (left) 2 in 15 min period; then 1 hr cooling
- Continuous Operation (Then 15 min cooling):
  - Left 10 min
  - Right 2 min
- Max engagement speed 20% $N_2$
- Discontinue start if minimum of 17% $N_2$ cannot be maintained

*RPM:

- Idle (Ground Operation) 57 - 69%
- Max 105%

TIT:

- Norm Oper Range 300 - 1,130°C
- Starting:
  - Ground 710°C (momentary)
  - Air 870°C (maximum)
- Max Military 1,130°C (45 minutes)
- Max & Partial A/B (<660 KIAS) 1,130°C (45 minutes)
- Max & Partial A/B (>660 KIAS) 1,190°C (45 minutes)
- During Accel 1,160°C (2 minutes)
- Max Continuous 1,000°C (Unlimited)

FUEL FLOW:

- Fluctuations ±300pph
- Zero or negative g 10 seconds max

*ENGINE:

| | Zones 1, 2, 3 | Zones 4, 5 |
|---|---|---|
| • A/B (Ground) | | |
| • One Eng | 6 min | 90 sec |
| • Two Eng | 6 min | 30 sec |
| • Max Continuous Above 1,000°C | | 45 min |
| • A/B Operations in Flight | | 45 min |

OIL PRESSURE:

- Min (Idle) 35 psi
- Norm Operations 40-50 psi
- Maximum 50 psi
- Fluctuation ±5 psi within limits

HYDRAULIC PRESSURE:

- Normal 2,950 - 3,250 psi
- Maximum 3,250 psi

Figure 5-2. (Sheet 2)

## SUMMARY TABLE OF LIMITATIONS (cont)

*CREW MODULE:

- Combined Crew Wt 430 lbs
- Differential Wt 65 lbs

FLYING QUALITIES LIMITS:

- Yaw and/or roll dampers off Mach limit is 1.1.

AVIONICS LIMITS:

- Do not operate JSS transmitters above 40,000 feet.
- Do not operate ILS in flight while the JSS or SPS is transmitting.
- Do not use TFR in flight, in manual or auto TF modes while JSS transmitters are radiating in bands 6 and 8.

CROSSWIND LIMITS:

- Takeoff and landing (flaps/slats extended) on a dry runway is limited to a 90° crosswind component of 28 knots.
- Landing (flaps/slats retracted) on a dry runway is limited to a 90° crosswind component of 14 knots.

EXTERNAL STORES:

Installation of external jettisonable stores is not authorized.

**Figure 5-2. (Sheet 3)**

further AB operation. Continuous AB operation shall be limited to a total time of 6 minutes, in any zone or combination of zones.

### 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.

**WARNING**

An engine stall or afterburner blowout may occur during throttle retards in the afterburning range (including both partial AB transients and retard to non AB power). To avoid simultaneous abrupt thrust loss on both engines, power reductions should be made on one engine at a time (in situations where maintaining thrust is a critical requirement).

### ZERO G AND NEGATIVE G TIME LIMIT

To prevent possible flameout of both engines, do not exceed 10 seconds at 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.
- During zero or negative g conditions it is possible for either or both oil pressure indicators to indicate low or zero oil pressure. This does not constitute an oil system malfunction provided the oil pressure returns to normal when positive g operation is resumed.

# Approved Fuels

| | Recom-mended | Preferred Alternate | Approved Alternate | | | | | Emergency |
|---|---|---|---|---|---|---|---|---|
| Description | JP–4 | JP–8 | JP–5 | Jet A | Jet A–1 | Jet A–1 | Jet B | Aviation Gasoline Blended With Oil (See Note B) |
| NATO No. Specification | F-40 MIL-T-5624 | F-34 MIL-T-83133 | F-44 MIL-T-5624 | — ASTM D1655 | F-34 ASTM D1655 | F-35 ASTM D1655 | F-45 ASTM D1655 | F-22 MIL-G-5572 |
| Icing Inhibitor | Yes | Yes | Yes | No (See Note A) | Yes | No (See Note A) | No (See Note A) | No (See Note A) |
| Freezing Point | −58°C (−72°F) | −47°C (−52.6°F) | −46°C (−51°F) | −38°C (−36°F) | −47°C (−52.6°F | −50°C (−58°F) | −49°C (−56°F) | −60°C (−76°F) |

Note A:

1. 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 read 0 degrees C or less. The additive will prevent ice from accumulating in the fuel controls and strainers.
2. When used without corrosion inhibitor additives, operation is limited to 10 hours.

Note B:

1. Aviation gasoline 115/145 must be blended with 3 percent lubricating oil MIL–L–6082 Grade 1100.
2. This fuel is approved for a one flight emergency situation only. An alternate fuel should be used if available.
3. 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.
4. Throttle movements should be as slow as practical.
5. Altitude should remain as low as practical and must not exceed 35,000 feet.
6. Engine thrust available may be reduced approximately 10 percent.
7. 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.
8. It is permissible to mix this fuel with a preferred or alternate fuel in the aircraft. However the above restrictions are still applicable.

Note C:

T.O. 42B1–1–14 and 42B1–1–15 may be consulted for additional information relating to use of aircraft fuels.

★ **Figure 5-3.**

# AIRSPEED LIMITATIONS

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
FLAPS AND GEAR UP

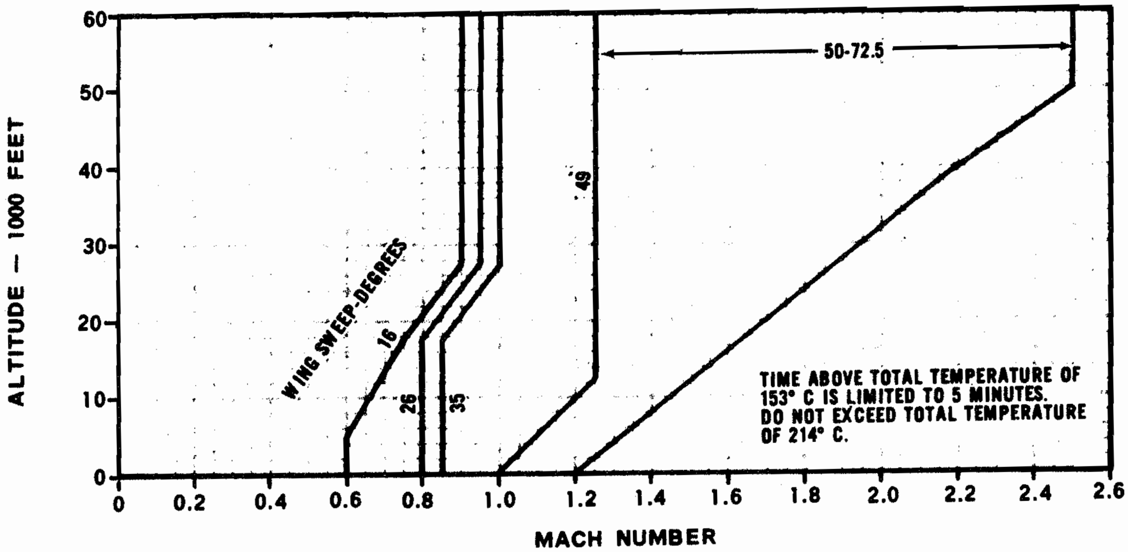



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Figure 5-4.

## AIRSPEED LIMITATIONS

### AIRSPEED AND ALTITUDE OPERATIONAL LIMIT ENVELOPES

Airspeed restrictions are presented in figure 5-4. Limitations shown in figure 5-4 do not reflect operational limitations imposed by cg considerations. Refer to figure 5-10 for additional wing sweep or airspeed limits. The maximum sustained speed is coincident with a total temperature of 153°C (308°F). The maximum dash speed is coincident with a total temperature of 214°C (418°F) or Mach 2.50, whichever is less. Flight at speeds that result in total temperatures greater than 153°C (308°F) is limited to 5 minutes per flight. Refer to figure 5-5 for maximum recommended cable engagement speeds.

### SLATS/FLAPS LIMIT SPEEDS

The flaps and slats are structurally cleared to the following limits during travel, or when fully extended:

| | |
|---|---|
| Slats only | 330 KIAS or 0.62 Mach, whichever is less |
| Flaps 0 to 30 degrees | 330 KIAS or 0.62 Mach, whichever is less |
| Flaps 31 degrees to full | 300 KIAS or 0.56 Mach, whichever is less |

However, to ensure slat/flap extension, and from component life and handling considerations, the following limitations should be observed:

- Flaps

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 |

- Slats limit speed is 295 KIAS or 0.62 Mach, whichever is less

# Maximum Recommended Cable Engagement Speeds

DATA BASIS: ARRESTING CABLE LIMITS.

NOTE: THE MAXIMUM CABLE THE ARRESTING HOOK WILL ACCEPT IS 1-3/8 INCHES IN DIAMETER.

| Aircraft Weight–(Pounds) | Bak-9* | Std Bak-12* | 1200' Runout Bak-12 | Dual Bak-12 | Bak-13 |
|---|---|---|---|---|---|
| 55,000 | 135 | 145 | 175 | 180 | 160 |
| 60,000 | 130 | 140 | 175 | 180 | 160 |
| 65,000 | 120 | 130 | 170 | 180 | 160 |
| 70,000 | 115 | 125 | 165 | 180 | 160 |
| 75,000 | 105 | 115 | 160 | 180 | 155 |
| 80,000 | 100 | 110 | 155 | 180 | 150 |
| 85,000 | 95 | 100 | 150 | 180 | 145 |
| 90,000 | 90 | 95 | 145 | 180 | 140 |
| 100,000 | — | — | — | 180 | — |

*Use only when it is impractical to engage one of the other three systems.

**Figure 5-5.**

## LANDING GEAR OPERATION LIMIT

Do not exceed 1.20 g during landing gear extension. The landing gear is structurally cleared to 330 KIAS or Mach 0.62, whichever is less, during extension, retraction, or flight with the gear extended. However, the normal operational limit is a maximum of 295 KIAS or Mach 0.62, whichever is less.

**Note**

No intentional sideslip during gear transition is permitted.

## 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 crew comfort when operating in a ram air mode (RAM or EMER), airspeed should be kept between 260 KIAS and 460 KIAS.

**CAUTION**

During ram air operation, all 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.

## MANEUVERABILITY LIMITATIONS

### LIMIT MANEUVER LOAD FACTORS

Limit maneuver load factors are presented in figure 5-6.

### ROLL LIMITATIONS

1. The aircraft shall be limited from initiating rolling maneuvers at normal load factors less than 1 g at all wing sweeps except 45 and 50 degrees. For 45 and 50 degrees wing sweep, the roll limitations shall be defined as follows:

   - Do not perform rolling maneuvers at less than 1 g at Mach numbers greater than 0.80 and altitude in excess of 10,000 feet.

   - Rolling maneuvers of up to 90 degrees angle of bank, from an upright position, may be performed at load factors as low as zero g. Recovery must be initiated at a bank angle not greater than 45 degrees to avoid substantial bank angle overshoots.

2. Do not perform rolling maneuvers in excess of 360 degrees at all wing sweeps.

### ANGLE-OF-ATTACK AND RUDDER DEFLECTION (SIDESLIP) LIMITATIONS

The angle-of-attack and rudder deflection limitations presented in figure 5-7 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 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.

**WARNING**

Angle-of-attack limiting is not absolute. Large abrupt aft stick inputs or repetitive push-pull inputs at high angle-of-attack can result in overshoots of angle-of-attack limits and produce susceptibility to loss of control. Also, large abrupt cross controlling and reversals at maximum angle-of-attack may force the aircraft into departure from controlled flight.

Stall warning will activate above 14 degrees wing angle-of-attack for all wing sweeps when the combination of pitch rate in degrees per second and wing angle-of-attack in degrees total 18 ($\pm$1). Sideslip limitaitons 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**

Intentional overpowering of the yaw damper through large, prolonged rudder inputs can result in loss of roll control and eventual loss of control of the aircraft.

### FLIGHT WITH DAMPERS OFF

Figure 5-8 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 5-8 and the affected damper turned off. Continuing flight should be accomplished with a wing sweep of 26 degrees observing the airspeed limitation for this sweep presented in figure 5-8, 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 shown in 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. Limit lateral maneuvers to 60 degrees of bank angle and do not exceed angle-of-attack limits.

**CAUTION**

With loss of yaw and/or roll dampers the aircraft is presently restricted to Mach 1.1 for wing sweeps of 46 degrees to 72.5 degrees at all altitudes.

### CROSSWIND LANDINGS

The maximum allowable crosswinds during landings with flaps extended and/or retracted are presented in figure 5-9. For a complete discussion refer to "Crosswind Landing", Section II.

# Limit Maneuver Load Factors

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:

FLAPS AND GEAR UP
PIVOTING PYLONS (W-W/O MAU-12C/A BOMB RACKS) ON OR OFF.
(A) SYMMETRIC MANEUVER AT ANY WING-SWEEP
(B) ASYMMETRIC (ROLLING PULLOUT) MANEUVER
(C) SYMMETRIC MANEUVER DURING WING SWEEP
(C-1) SYMMETRIC MANEUVER DURING WING SWEEP. 16°-50°
(C-2) SYMMETRIC MANEUVER DURING WING SWEEP. 51°-72.5°

CONFIGURATION:

GEAR UP OR DOWN
SLATS ONLY EXTENDED OR FLAPS EXTENDED
PIVOTING PYLONS (W-W/O MAU-12C/A BOMB RACKS) ON OR OFF.
(D) SYMMETRIC MANEUVER } 16-26 DEGREES
(E) ASYMMETRIC MANEUVER } WING SWEEP
(E-1) IN ANY POSITION OTHER THAN FULLY EXTENDED OR RETRACTED:
AVOID ABRUPT ROLL ENTRY AND TERMINATION
LIMIT BANK ANGLES TO 60 DEGREES

NOTE: DO NOT SWEEP WINGS DURING ASYMMETRIC (ROLLING PULLOUT) MANEUVER.

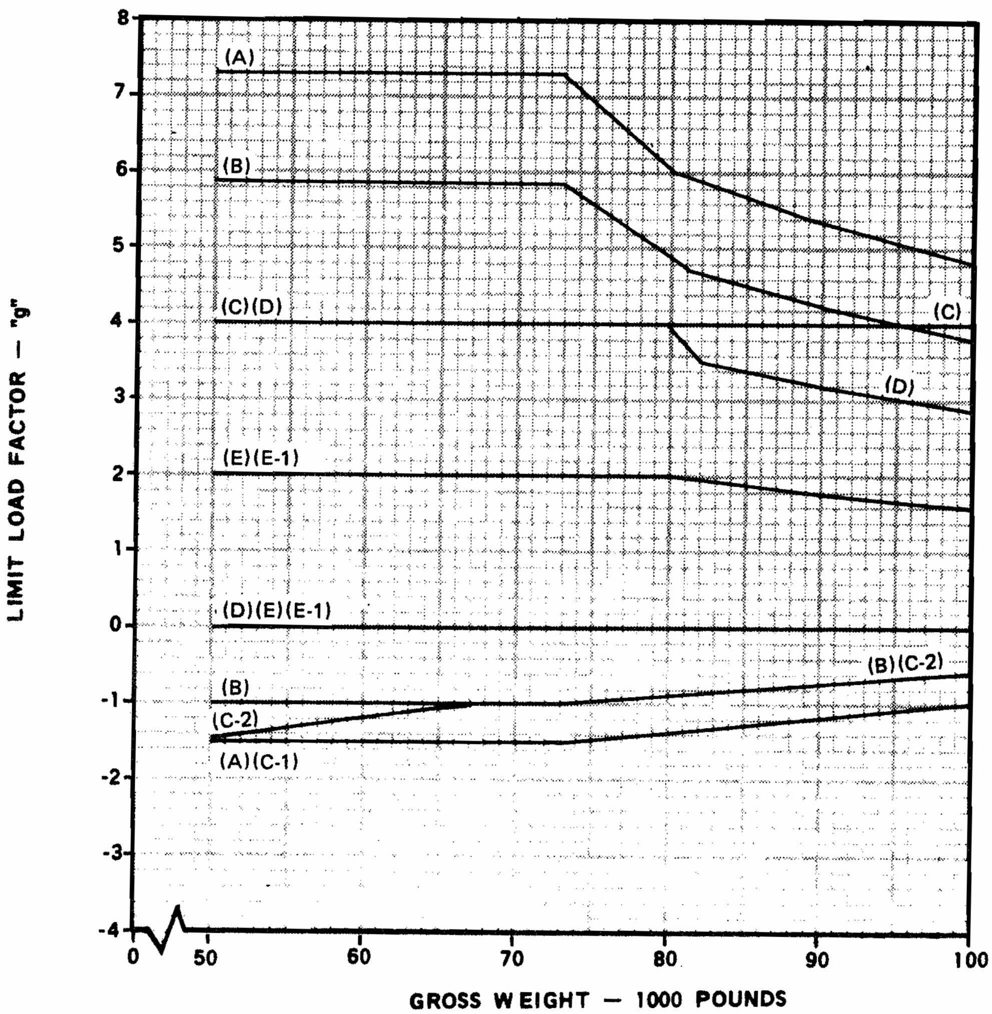


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Figure 5-6.

## Angle-of-Attack and Rudder Deflection (Sideslip) Limitations

| Wing Sweep (Degrees) | Configuration | Angle-of-Attack | Rudder Deflection Limit Applicable for all angles-of-attack |
|---|---|---|---|
| 16-26 | Gear and slats/flaps down. | Below 0.40 Mach:<br><br>14 degrees or stall warning system activation, whichever occurs first. | Yaw Damper On.<br>15 degrees. |
| | | Above 0.40 Mach:<br><br>1. 10 degress for flaps greater than 15 degrees.<br><br>2. 12 degrees for flaps at 15 degrees or less.<br><br>3. Up to 14 degrees with slats only. | Yaw Damper Off.<br>12 degrees, do not make abrupt rudder inputs. |
| 16-49 | Gear and slats/flaps up. | 18 degrees or stall warning whichever occurs first. | Yaw Damper On.<br>6 degrees below Mach 0.80.<br>3 degrees above Mach 0.80. |
| | 14 degrees with pitch damper off. | Yaw Damper Off. | No intentional sideslip. |
| 50-72 | Gear and slats/flaps up. | 18 degrees or stall warning whichever occurs first. | Yaw Damper On.<br>6 degrees below Mach 0.80.<br>3 degrees above Mach 0.80.<br><br>Yaw Damper Off.<br>No intentional sideslip. |

**Figure 5-7.**

## 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 5-10, as a function of maximum allowable down horizontal stabilizer position and minimum fuel remaining.

For wing sweeps greater than 35 degrees at airspeeds below Mach 2.0, there is no aft center-of-gravity limit within the normal loading capability of the 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 5-10 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

## Dampers Off Operating Limits

| DO NOT EXCEED THE FOLLOWING AIRSPEEDS/ALTITUDES: | | |
|---|---|---|
| **PITCH DAMPER OFF** | | |
| **Wing Sweep** | **Altitude** | **Airspeed** |
| 16-25 Degrees | No Restriction | 265 KIAS or Mach 0.50 whichever is less |
| 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 1.1 |

**Figure 5-8.**

horizontal stabilizer positions and minimum fuel values presented are for stabilized flight, with speed brake retracted and 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 5-10 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 only and can be used to determine if the aircraft is within the aft center-of-gravity limit regardless of fuel distribution, aircraft ballast, or airspeed. For example, with 26-degree wing sweep, the maximum allowable down horizontal stabilizer position presented in figure 5-10 (1.5 degree down) does not change as the 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, push takeoff trim button and check for 3.8 degrees trailing edge up.)

**WARNING**

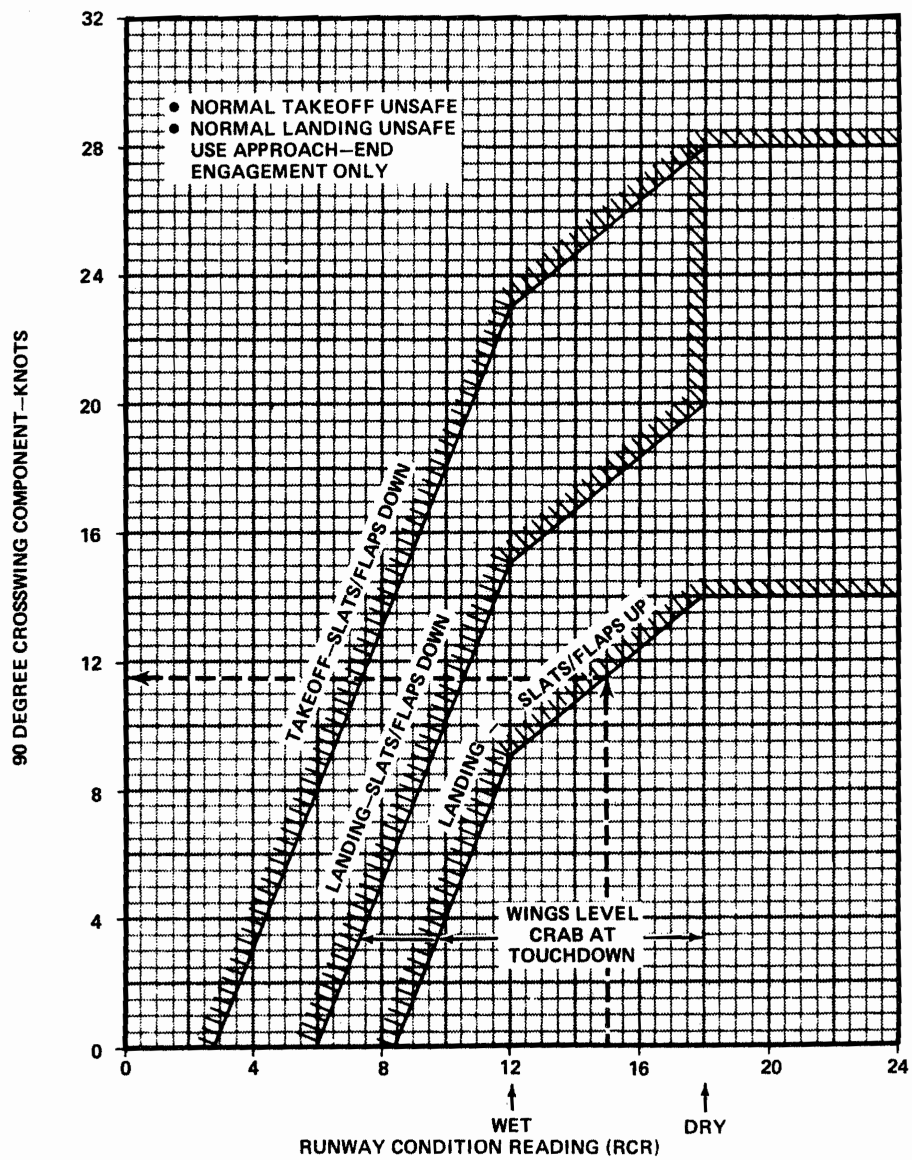
- When the maximum allowable down horizontal stabilizer position is reached, the aircraft is at the aft center-of-gravity limit for that specific wing sweep/airspeed, even though the minimum fuel values may have been exceeded. The aft center-of-gravity limit will be exceeded if flight is continued and the wing sweep and airspeed are not changed to maintain the center-of-gravity within the limits specified in figure 5-10.

- 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 5-10 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.

# CROSSWIND TAKEOFF AND LANDING LIMITS

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981




NOTES:
- ON WET OR SLIPPERY RUNWAYS, WHEEL CORNERING CAPABILITY IS REDUCED BY BRAKING ACTION WHICH MAY INCREASE SUSCEPTIBILITY TO SKIDDING. IF SKIDDING IS ENCOUNTERED DURING BRAKING ACTION, RELEASE BRAKING PRESSURE UNTIL DIRECTIONAL CONTROL IS ASSURED.
- LANDING AT RCR'S LESS THAN SIX IS NOT RECOMMENDED.
- USE MAXIMUM GUST VELOCITY IN COMPUTING CROSSWIND.

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Figure 5-9.

# AFT CENTER-OF-GRAVITY LIMITS (BASED ON STABILIZER POSITION/MINIMUM FUEL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

FUEL GRADE: JP-4
ENGINES: TF30-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 POSITION EXCEEDS THIS VALUE IN A CLOCKWISE DIRECTION WHILE IN 1.0 "g" FLIGHT, THE AIRCRAFT CENTER OF GRAVITY HAS EXCEEDED THE LIMIT FOR THAT WING SWEEP. THE HORIZONTAL STABILIZER POSITION IS A FUNCTION OF WING SWEEP ONLY.

CONSIDERATIONS:
MINIMUM FUEL REMAINING IS BASED ON:

A. AN 8,200 POUND FUEL DIFFERENTIAL HAS BEEN MAINTAINED, AND/OR ALL THE REMAINING FUEL IS IN THE FORWARD TANK.
B. ZERO FUEL 26° WING SWEEP CENTER OF GRAVITY IS MAINTAINED AT OR FORWARD OF 47.5 PERCENT MAC.
C. TEN TRANSMITTERS AND 5 EXCITERS INSTALLED

| | GEAR DOWN | | | | | |
|---|---|---|---|---|---|---|
| WING SWEEP | 16 DEGREES | | | 26 DEGREES | | |
| | FLAPS/SLATS | | | FLAPS/SLATS | | |
| | UP | SLATS ONLY | FLAPS 1° TO FULL | UP | SLATS ONLY | FLAPS 1° TO FULL |
| CONFIGURATION | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER |
| | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS |
| CLEAN | 1° DOWN | 1° UP | 4° UP | 1° DOWN | 1° UP | 2° UP |
| | 4,400 | 5,700 | 7,000 | 0 | 0 | 0 |
| CARGO PODS ON STATIONS 3 AND 6 | 0.5° DOWN | 1.5° UP | 4.5° UP | 0.5° DOWN | 1.5° UP | 2.5° UP |
| | 5,500 | 6,800 | 7,500 | 0 | 0 | 0 |
| CARGO PODS ON STATIONS 3, 4, 5 AND 6 | 0.5° DOWN | 1.5° UP | 4.5° UP | 0.5° DOWN | 1.5° UP | 2.5° UP |
| | 5,100 | 6,500 | 7,200 | 0 | 0 | 0 |

| | GEAR AND FLAPS UP | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| WING SWEEP | 16 DEGREES | | 26 DEGREES | | 35 DEGREES | AFT WING SWEEP | | |
| AIRSPEED | 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 | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MAXIMUM STABILIZER | MINIMUM FUEL-POUNDS | MINIMUM FUEL-POUNDS | MINIMUM FUEL-POUNDS |
| | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | MINIMUM FUEL POUNDS | | | |
| CLEAN | 1° DOWN | 1° DOWN | 1.5° DOWN | 1.5° DOWN | 1° DOWN | 0 | 0 | 5,000 |
| | 7,600 | (1) | 0 | 2,700 | 0(2) | | | |
| CARGO PODS ON STATIONS 3 AND 6 | 0.5° DOWN | 0.5° DOWN | 1° DOWN | 1° DOWN | 0.5° DOWN | NOT AUTHORIZED | | |
| | 13,000 | (1) | 1,600 | 4,100 | 0 | | | |
| CARGO PODS ON STATIONS 3,4,5 AND 6 | 0.5° DOWN | 0.5° DOWN | 1° DOWN | 1° DOWN | 0.5° DOWN | NOT AUTHORIZED | | |
| | 10,200 | (1) | 900 | 3,800 | 0 | | | |

NOTES

THESE NOTES ARE FOR PLANNING PURPOSES ONLY. SURFACE POSITION INDICATIONS SHOULD BE USED FOR INFLIGHT DETERMINATION OF ALLOWABLE CG.

(1) USING THE MINIMUM FUEL CONSIDERATIONS ABOVE, THERE IS NO FUEL LOADING AT WHICH CENTER-OF-GRAVITY IS WITHIN AFT LIMIT.
(2) FOR WING SWEEPS GREATER THAN 35 DEGREES AT AIRSPEEDS BELOW MACH 2.2, THERE IS NO FUEL LOADING AT WHICH THE CENTER-OF-GRAVITY EXCEEDS THE AFT LIMIT.

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**Figure 5-10.**

## FORWARD CENTER-OF-GRAVITY

For center-of-gravity locations forward of 25 percent MAC, a correction to takeoff speed is necessary to assure proper rotation beginning 15 KIAS below takeoff speed. Takeoff speed should be increased 2 KIAS for each 1 percent center-of-gravity forward of 25 percent MAC. The resulting correction should be compared to any required Delta $V_{SE}$ and the larger of the two values used as Delta $V_{SE}$. (See T.O. 1F-111(E)A-1-1.) 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 large amounts of fuel remaining, 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.

## BRAKE AND TIRE LIMITATIONS

The brakes are limited in the amount of energy they can absorb and dissipate in the form of heat. A measure of the amount of heat absorbed by the brakes is the kinetic energy expended, measured in millions of foot-pounds. The amount of heat added to the brakes for each braking effort during a landing rollout or taxiing is cumulative and is a function of the speed of the aircraft and its gross weight at the time the brakes are applied. The heat generated in the brakes is transferred to the wheel and tire, and depending on the severity of the stop, can cause the tire pressure to rise to dangerous levels. Thermal release plugs within the wheel are designed to prevent wheel explosion by relieving pressure from the tire when the wheel attains a particular temperature. However, the thermal release plugs will not protect against tire explosion resulting from tire sidewall heat build-up due to excessive taxiing. In those stops where tire explosion or thermal release plug activation occurs, a hydraulic fluid fire is a possibility due to the deterioration of the rubber seals within the brake assembly. Such a fire can occur before tire explosion or thermal release plug activation.

## BRAKE LIMITATIONS

Brake energy limits for slats-flaps-spoilers extended are provided in figure 5-11. The example lines explain how to determine the amount of energy absorbed by the brakes during a stop. The brake energy limit charts should be used whenever a takeoff is aborted, for all flaps-up landings, for planned lengthy ground maneuvers, and when the pilot suspects that the combination of gross weight, runway altitude, temperature, IAS, and the number of stops and decelerations will result in brake energies in the caution or danger zones. Operating procedures and restrictions are provided with the brake energy limits charts to preclude the possiblity of injury to personnel and damage to equipment from overheated brakes and tires.

**WARNING**

Takeoff, after an abort or full stop taxi back landing, requires consideration of the cumulative heating effect on the brakes should the second takeoff result in an abort. Additionally, a 30,000 foot taxi distance restriction for aircraft above 82,500 pounds gross weight, (40,000 feet at gross weights of 82,500 pounds or less) might prevent an immediate takeoff attempt. See the restrictions in figure 5-11 under the Normal Zone heading.

## TIRE TAXI LIMITATIONS

Continued operation of the aircraft is restricted should a total taxi distance of 30,000 feet be reached. If aircraft

# BRAKE ENERGY LIMITS

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONDITIONS:
FLAPS/SLATS/SPOILERS–EXTENDED
WING SWEEP–26 DEGREES OR LESS
ONE CONTINUOUS BRAKE APPLICATION

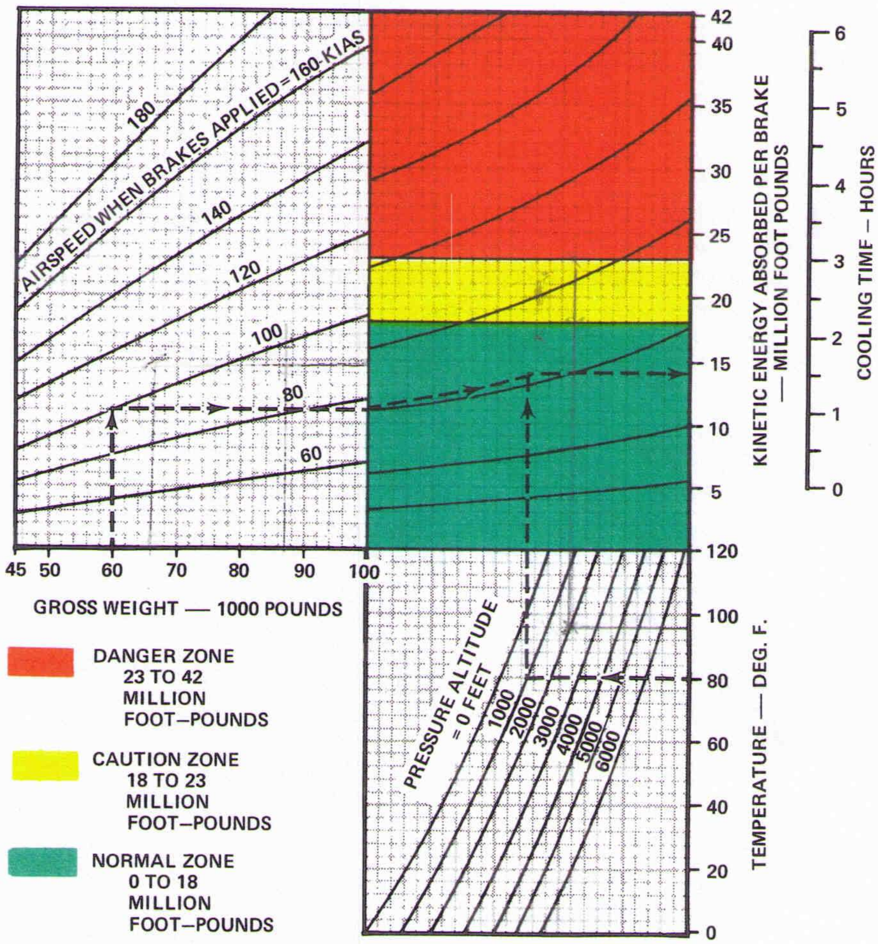
NOTES:

1. SUBTRACT 50 PERCENT OF THE HEADWIND COMPONENT, MEASURED BY THE TOWER, FROM THE INDICATED AIRSPEED. THE FULL TAILWIND COMPONENT MUST BE ADDED TO THE INDICATED AIRSPEED.
2. BRAKE ENERGY USE IS 15 PERCENT HIGHER FOR LANDINGS WITH FLAPS/SLATS RETRACTED.

THE FOLLOWING INFORMATION EXPLAINS ACTION TO BE TAKEN WHEN A STOP IN THE DANGER, CAUTION, OR NORMAL ZONE IS PERFORMED.

**WARNING**

When a landing results in Danger or Caution zone operation, the area extending at least 300 feet in a 45 degree cone around the axle on both sides of the wheel should be regarded as unsafe during at least the first one hour and 15 minutes after the stop unless the thermal release plugs have allowed the tires to deflate.




## 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. Request fire fighting equipment.

**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 aircraft and personnel without stopping and as quickly as possible. Do not set parking brakes.

3. Hydralic 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 firefighting 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. Request fire fighting eqipment. Do not set parking brakes or shut down engines until fire fighting equipment arrives. Hydraulic fluid fire is possible.

2. 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 the kinetic energy absorbed is within the NORMAL ZONE a subsequent takeoff may be performed provided the following conditions are met:

a. The sum of the brake energy absorbed in the completed stop and the brake energy absorbed in a possible rejected takeoff at decision speed (refusal speed) will not exceed 40 million foot pounds.

b. The distance already taxied, including the landing and/or takeoff rolls, plus the distance to travel and clear the runway should the next takeoff attempt be rejected at decision speed (refusal speed) will not exceed taxi distance limitations.

3. Should an abort occur after an initial abort or full stop landing, and the elapsed time between the two events is less than one hour, the brake energies for both stops shall be considered to be cumulative and shall be added together. The resulting brake energy shall be used with the chart for determining whether the brakes are in the CAUTION or DANGER ZONES.

1EFA-32

Figure 5-11.

gross weight is 82,500 pounds or less at initial landing, the taxi distance limitation is 40,000 feet. Taxi distance includes takeoff and landing rolls. If the distance limitation is reached or exceeded, the aircraft must be parked to allow the tires to cool. The restrictions and safety procedures listed in the Caution Zone of figure 5-11 are applicable for this case.

### MAXIMUM BRAKING SPEED

The maximum braking speed is that speed from which a stop will develop the maximum allowable brake energy (42 million foot-pounds). The brakes alone cannot be depended on to stop the aircraft from a speed faster than the maximum braking speed. The maximum braking speed is dependent on gross weight, pressure altitude, and ambient temperature. See figure 5-11.

## MISCELLANEOUS OPERATIONAL LIMITATIONS

### SPEED BRAKE LIMITS.

1. Speed brake operation is limited to 600 KIAS or Mach 2.0, whichever is less. However, to ensure integrity of the Lower Nose structure, normal Speed Brake operation should be restricted to 500 KIAS and 0 to +2 "g"s.

### MXU–648 CARGO POD

The following limitations apply to the carriage of the MXU–648 cargo pod:

1. The preferred pylon stations for pod carriage are 3 and 6.

2. If pylon stations 4 and 5 are used, wing sweep will be limited to 54 degrees. The wing sweep handle lockout control must be utilized.

3. Maximum cargo weight is 300 pounds per pod. Refer to T.O. 16W41-2-1 for proper cargo loading procedures.

4. Airspeed is restricted to Mach 0.90.

5. Acceleration g:

    a. Symmetrical - 0.0 to +3.0

    b. Rolling - +0.5 to +2.4

6. TFR operation is not permitted.

7. Operation of the JSS transmitters is authorized. Band 1 and band 2 transmitter output performance will be degraded and may be ineffective with pod carriage.

For asymmetric loading the following restrictions apply in addition to the above: For asymmetric pod use, limit total aircraft asymmetric load to 2,000 pounds.

### TERRAIN FOLLOWING RADAR OPERATION

Terrain following radar operation is limited to the following:

1. With 500 set clearance selected, limit maximum Mach to 0.85.

2. For all set clearances except 500, limit maximum Mach to 1.2.

3. For turning flight during TF operations the following limits apply:

    - Bank angles of 30 degrees or less are recommended for normal TF turning flight operations.

    - If bank angles greater than 30 degrees are required, a maximum bank angle of 40 degrees may be used with a minimum airspeed of 460 KTAS.

4. Strict observance of minimum TF operation airspeed (Appendix I).

5. Wing sweep angles of 26 to 72 degrees.

6. Blind letdown to an initial clearance of 1,000 feet.

7. A minimum altitude of 500 feet when in manual mode using the E scope only.

8. Set clearances for each route segment that will be at least 200 feet higher than any obstacle in the flight path that may not provide a reliable radar return.

9. No TFR operation with external stores permitted.

10. Do not fly auto TF with 200 foot set clearance plane and hard ride selected.

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.

11. Any known pitch trim malfunction or any pitch axis caution lamp that will not reset.
12. The yaw channel caution lamp will not reset.
13. The TF fly-up off caution lamp on.
14. When auto TF is selected and the reference not engaged, caution lamp is on.

## WINDSHIELD RAIN REMOVAL

Observe the following limits when using rain removal:

1. Ground Operation.

   a. Wet or dry windshield - Operate for short periods (less than 5 seconds) as required.

2. Inflight Operation.

   a. Dry windshield - Do not operate.

   b. Wet windshield - Operate as required.

**Note**

In order to minimize ECS water boiling, the use of the anti-ice and rain-removal systems should be kept to a minimum. Operation of both systems simultaneously should be limited to a maximum total of 15 minutes for any one mission.

This is the last page of Section V.

# SECTION VI

# FLIGHT CHARACTERISTICS

## TABLE OF CONTENTS



## INTRODUCTION

The flight characteristics information presented in this section is based on quantitative and qualitative data obtained to date. The EF-111A configuration modifications and utilization of the variable sweep concept do not introduce any unusual flight characteristics. The main features of the flight control system (self-adaptive gain changing and command augmentation) significantly minimizes variations in stability and control characteristics over the large Mach-altitude operating spectrum of the aircraft. The low friction and breakout forces associated with the flight control system enhance ease of handling and maneuverability. Angle-of-attack limiting is provided by the stall inhibitor system (SIS) which greatly reduces the possibility of inadvertently exceeding angle-of-attack limits. The system increases pilot awareness of the approach to high angles-of-attack through increasing stick force per g and allows more precise longitudinal maneuvering. The SIS also provides a beta reducer that reduces sideslip during rolling maneuvers and improves rolling performance. Wing sweep transition will not be reflected to the pilot in the form of a trim change due to the command augmentation feature of the flight control 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.

**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.

## 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.

# LONGITUDINAL REFERENCE ANGLES

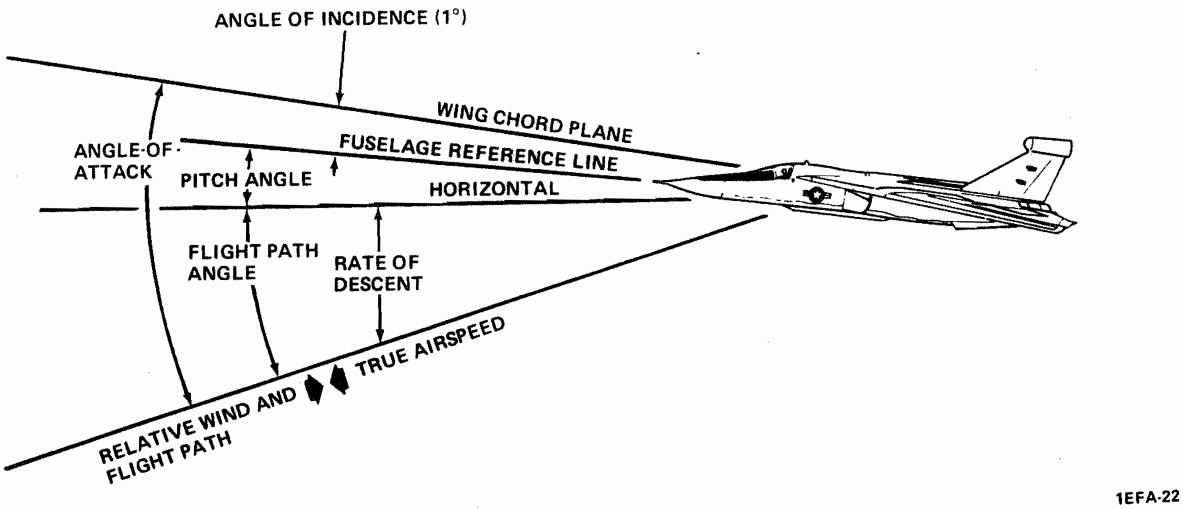



**Figure 6-1.**

## ANGLE-OF-ATTACK

The angle-of-attack 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 degrees angle-of-attack regardless of gross weight. The angle-of-attack indexer is programmed so that the onspeed symbol is lighted in the range of 10 degrees ($\pm$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 are considered excellent based on flight and estimated data. 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. Roll coordination is provided by the beta reducer at wing angles-of-attack above 7 degrees. 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. Inadvertent stall and post-stall gyrations may result from exceeding these limits.

Increasing back stick force is required during a slowdown, which gives the pilot a constant reminder of the slowdown, and wing angle-of-attack will be limited to about 20 degrees. However, sink rate will be quite high. Refer to "Minimum Airspeeds", this section.

### 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. 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. 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, response and damping characteristics are considered 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," Section V.

## MANEUVERING FLIGHT CHARACTERISTICS

### LONGITUDINAL FLIGHT

Wing sweep angles for maneuvering flight are compatible with those previously described for level flight characteristics. During maneuvering flight with the slats and flaps extended or retracted, angle-of-attack should not be allowed to exceed the angle-of-attack limits in Section V. Stall warning will activate above 14 degrees angle-of-attack when the sum of angle-of-attack in degrees and pitch rate in degrees per second equals 18 for all aircraft configurations. Increased stick forces above 11 degrees angle-of-attack improve awareness of approach to high angle-of-attack as well as provide more precise control of angle-of-attack and improve lateral maneuvering. However, angle-of-attack should be maintained within limitations. The artificial stall warning system provides a safety margin between warning signals and stall. Figure 6-2 depicts the safety margins at various wing sweeps.

**WARNING**

> Angle-of-attack limiting is not absolute. Large abrupt aft stick inputs or repetitive push-pull inputs at high angles-of-attack can result in overshoots of angle-of-attack limits and produce susceptibility to loss of control. Also, large abrupt cross controlling and reversals at maximum angle-of-attack may force the aircraft into departure from controlled flight.

Stick force per g and stick deflection per g during maneuvers are relatively independent of wing sweep and altitude up to 11 degrees wing angle-of-attack. A mild variation with Mach number, however, does exist. Above 11

# STALL WARNING SAFETY MARGIN

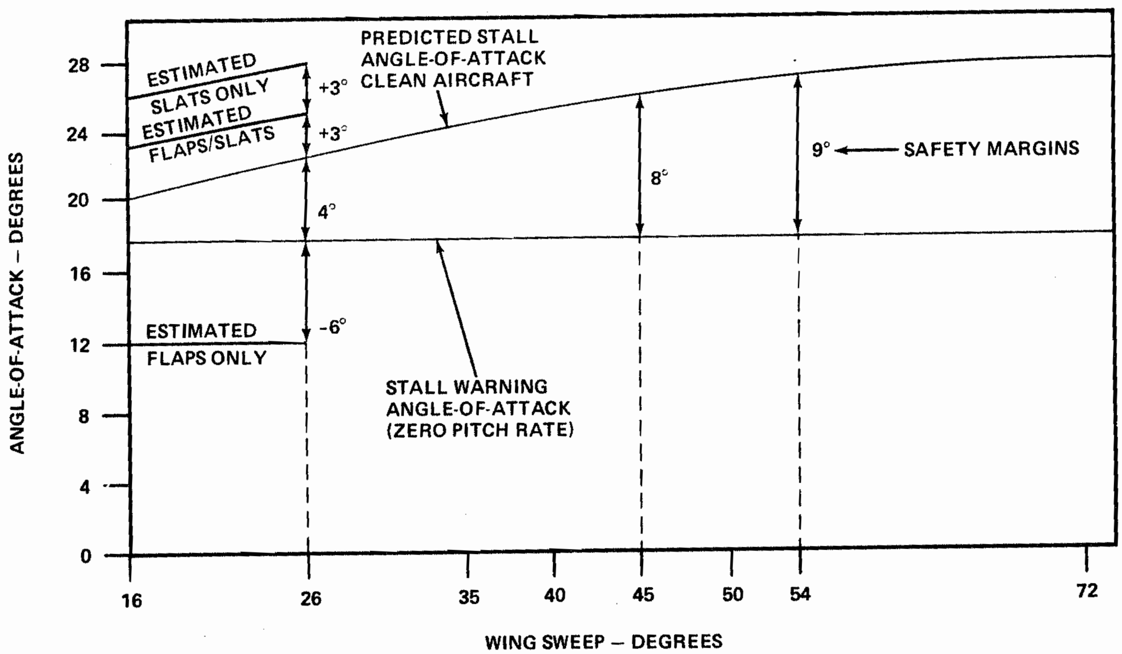



**Figure 6-2.**

degrees wing angle-of-attack, however, stick force per g and stick deflection per g increase such that, regardless of flight conditions or gross weight, wing angle-of-attack is limited to about 20 degrees with full back stick. The increasing stick force per g with increasing angle-of-attack provides the pilot with a direct cue as to angle-of-attack condition and allows more precise maneuvering with less monitoring of the angle-of-attack indicator. During supersonic flight at altitudes above 30,000 feet with aft wing sweeps, full back longitudinal control maneuvers can result in some stick talk-back being detected. This characteristic is a result of the pitch damper and mechanical input commanding full nose-up 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 relieving the back pressure being held. This same characteristic is exhibited at negative load factors for the aft sweep throughout the 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 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 6-3 presents the angle-of-attack for buffet onset. These boundaries are based on ±0.05 g buffet intensity.

# ANGLE-OF-ATTACK FOR BUFFET ONSET

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR AND FLAPS UP

FUEL GRADE: JP-4
ENGINES: TF30-P-3

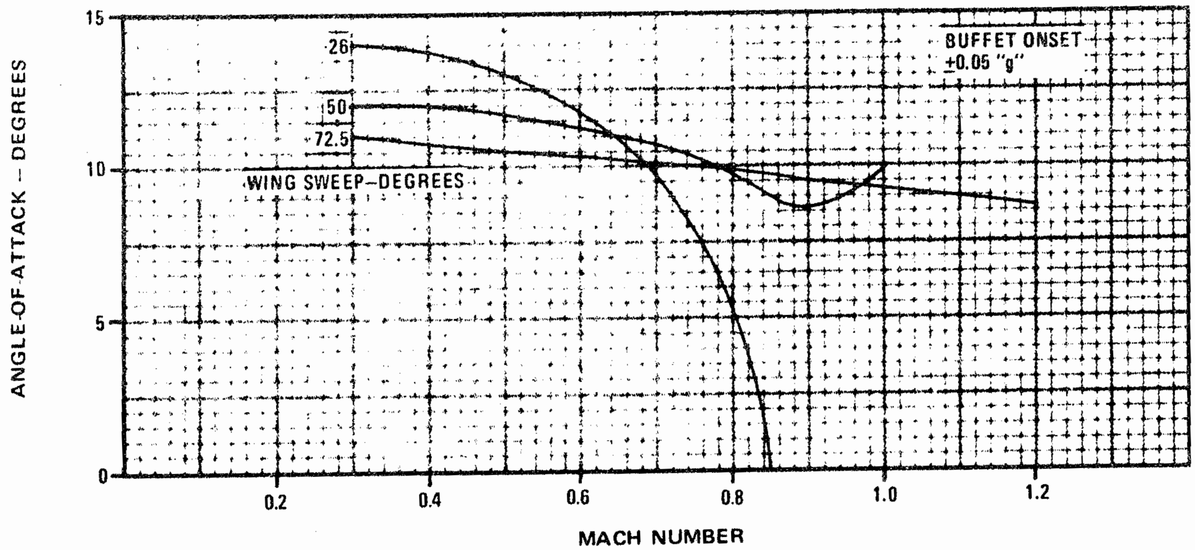



**Figure 6-3.**

## WING SWEEP/MANEUVERABILITY EFFECTS

Instantaneous longitudinal maneuver capability for wing sweeps of 26, 45, and 72.5 degrees is presented in figure 6-4 (sheets 1 through 4). For sustained maneuver load factors, refer to Appendix I. Two typical gross weights are shown: 60,000 pounds and 80,000 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. For reference, an estimated maximum afterburner ceiling for 1 g flight is presented. Refer to "Airspeed Limitations," Section V, for airspeed limitations.

**WARNING**

Flight into heavy buffet is prohibited.

**CAUTION**

- The wing seals can be damaged when maneuvering at supersonic speeds in excess of 3 g's with a full aft wing sweep. If higher g loads are planned or anticipated, use the most forward allowable wing sweep.
- With full aft wing sweep, at supersonic speeds, negative g's may cause minor flap damage.

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 6-5 (sheets 1 through 4) presents data for angle-of-attack versus wing

# MANEUVER CAPABILITY

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
WING SWEEP - 26 DEGREES
GEAR AND FLAPS UP

FUEL GRADE: JP-4
ENGINES: TF30-P-3

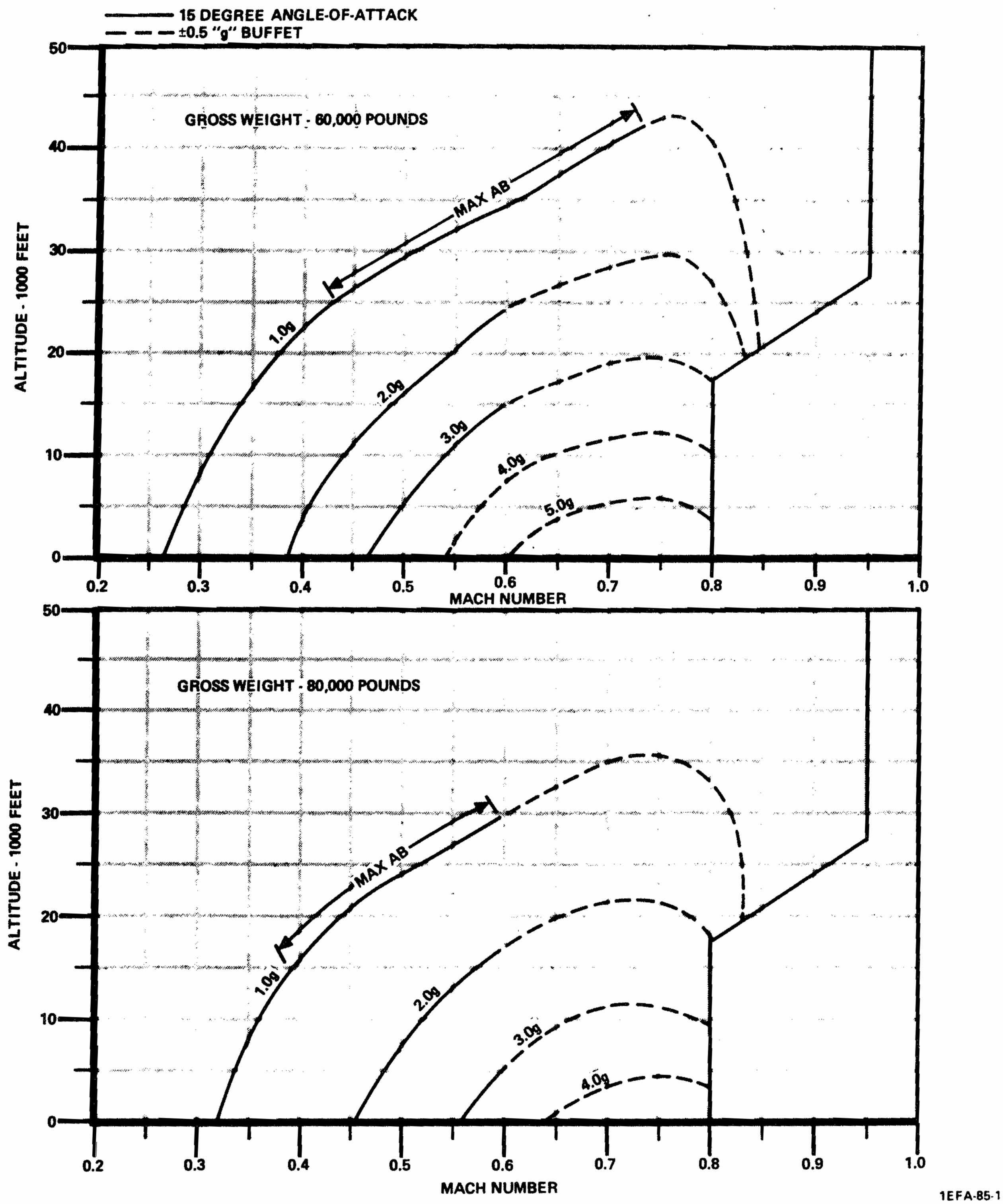


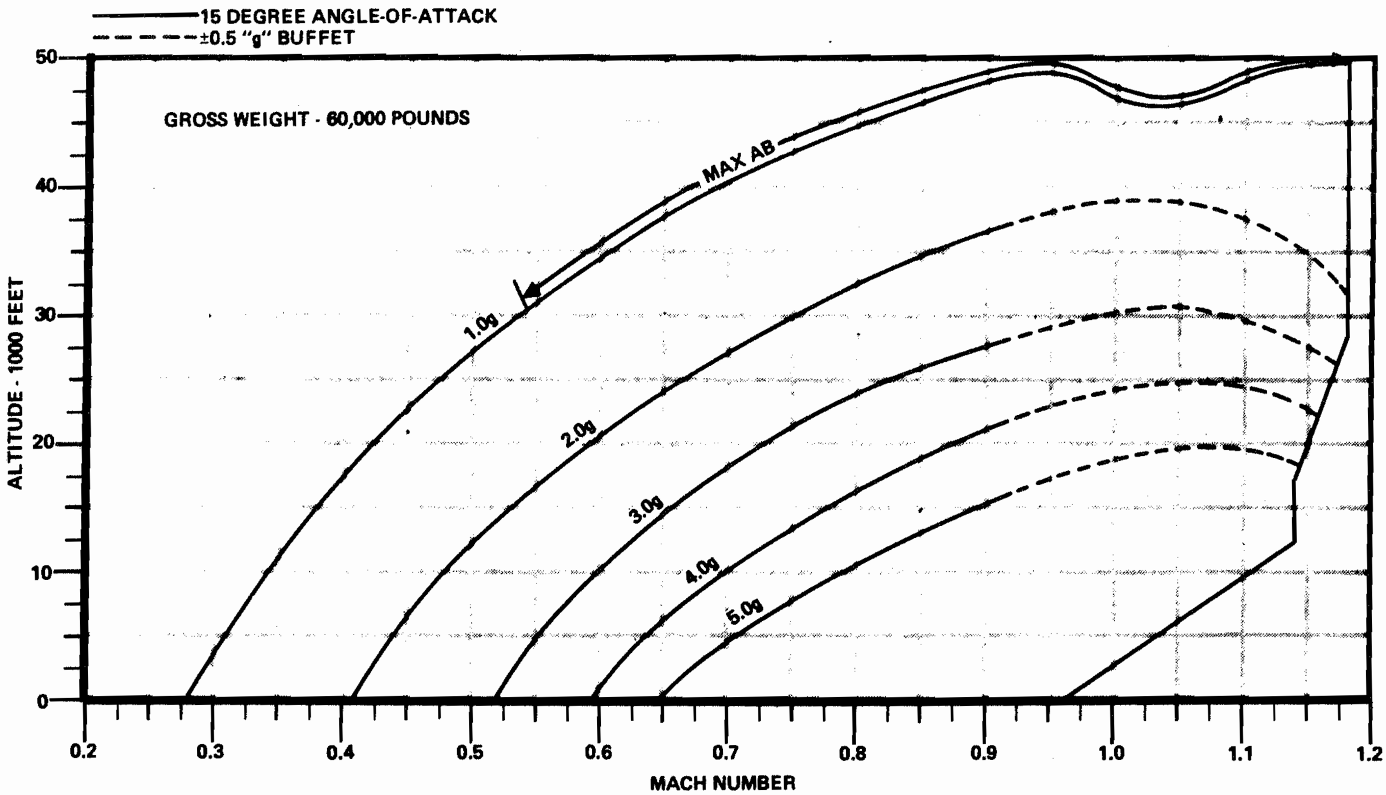

Figure 6-4. (Sheet 1)

# MANEUVER CAPABILITY

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
WING SWEEP - 45 DEGREES
GEAR AND FLAPS UP

FUEL GRADE: JP-4
ENGINES: TF30-P-4



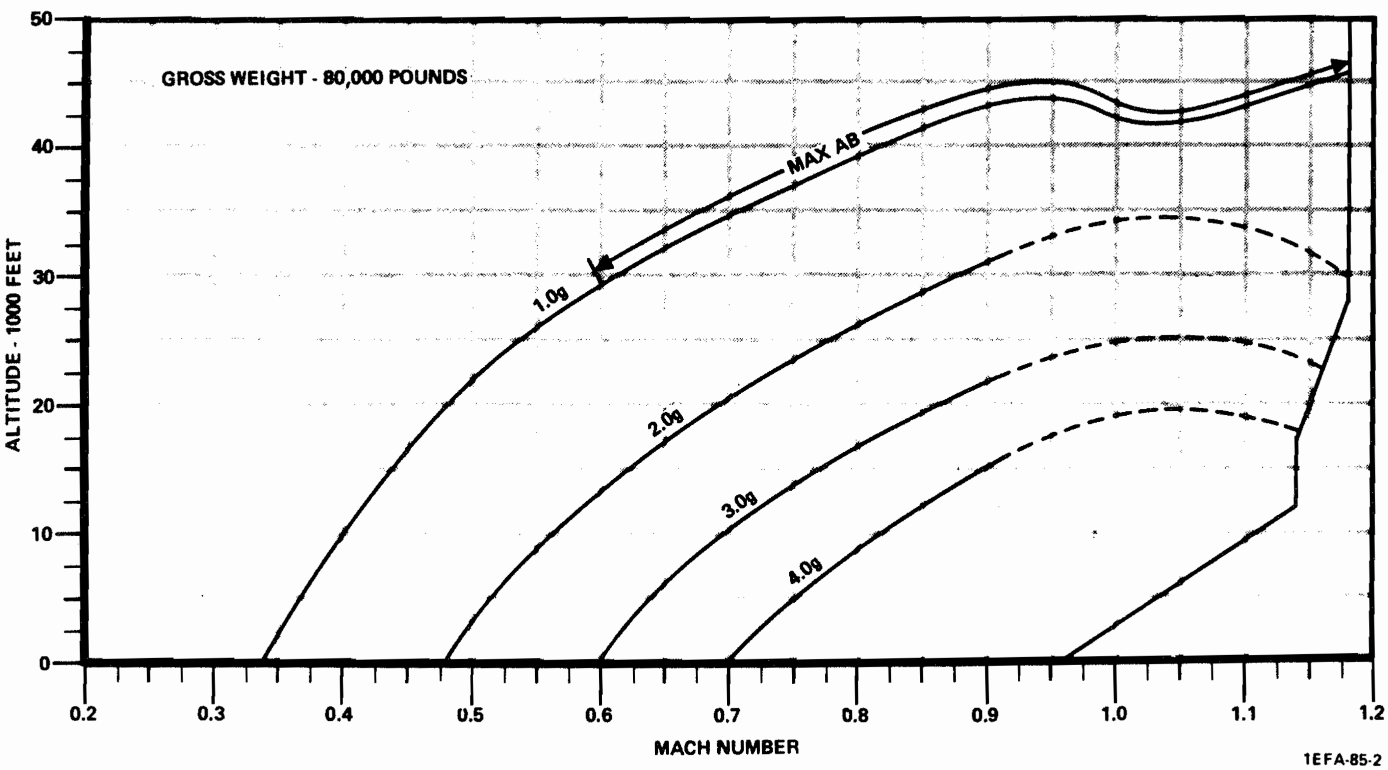




Figure 6-4. (Sheet 2)

# MANEUVER CAPABILITY

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONFIGURATION:
WING SWEEP - 72.5 DEGREES
GEAR AND FLAPS UP

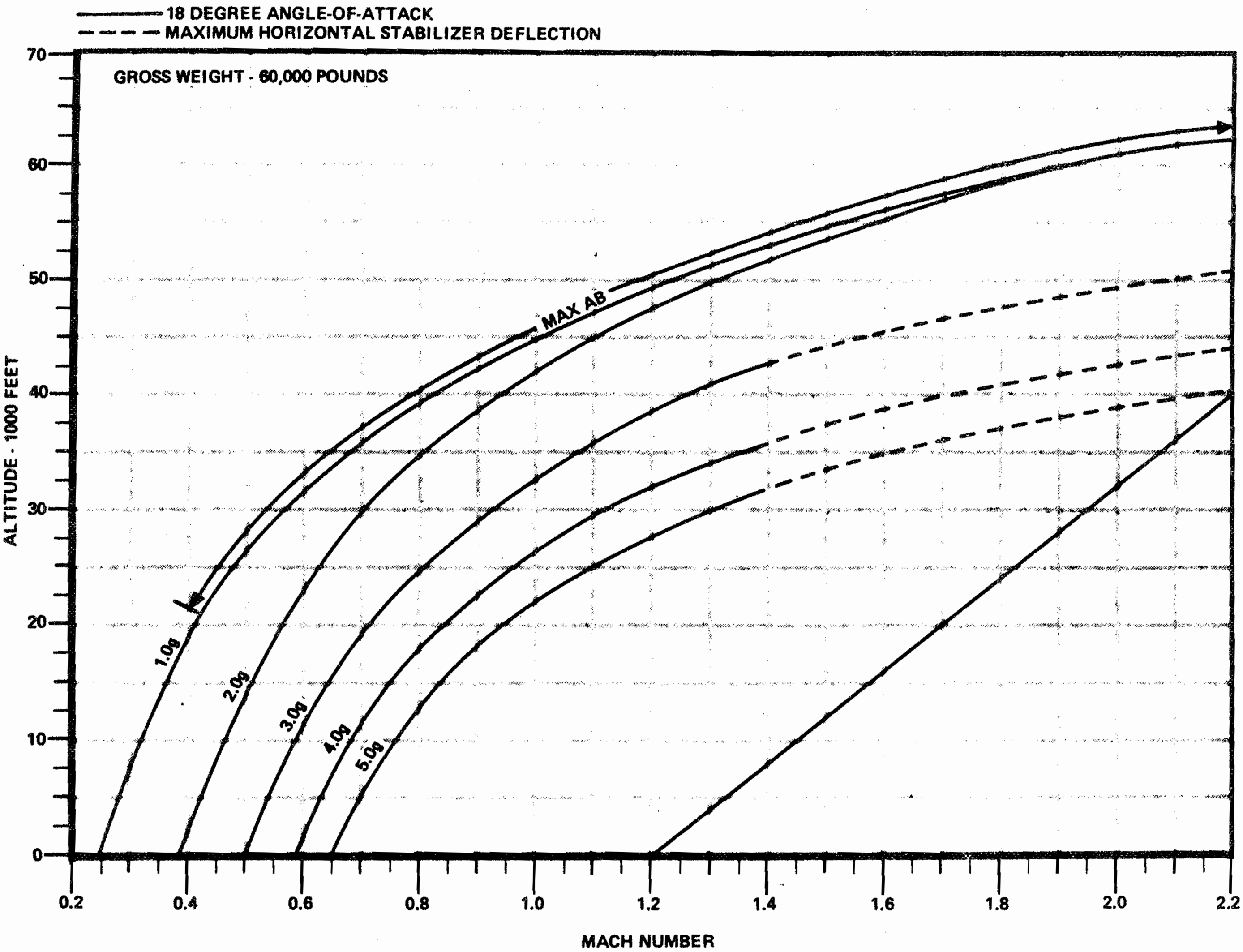



Figure 6-4. (Sheet 3)

# MANEUVER CAPABILITY

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
WING SWEEP - 72.5 DEGREES
GEAR AND FLAPS UP

FUEL GRADE: JP-4
ENGINES: TF30-P-3

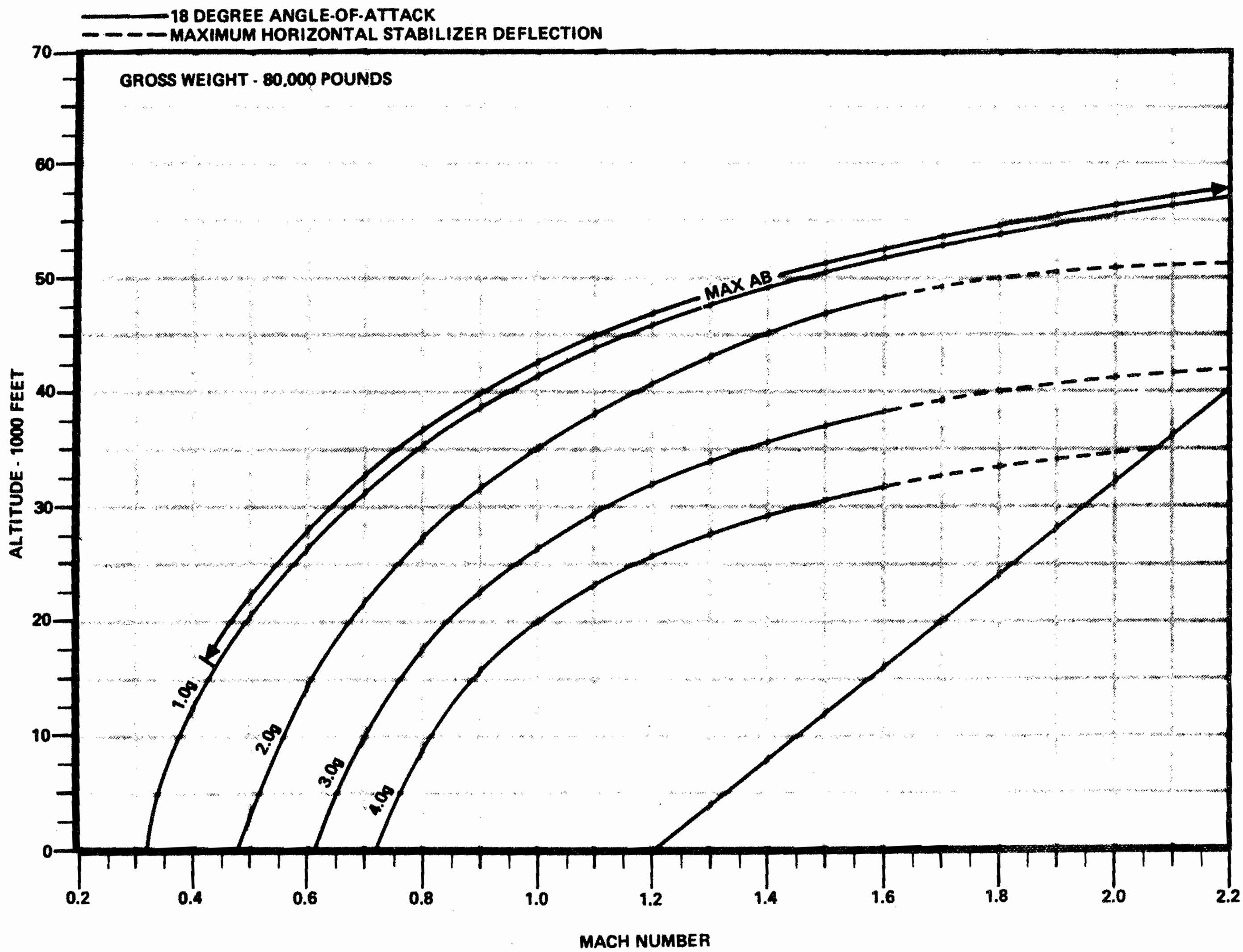



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Figure 6-4. (Sheet 4)

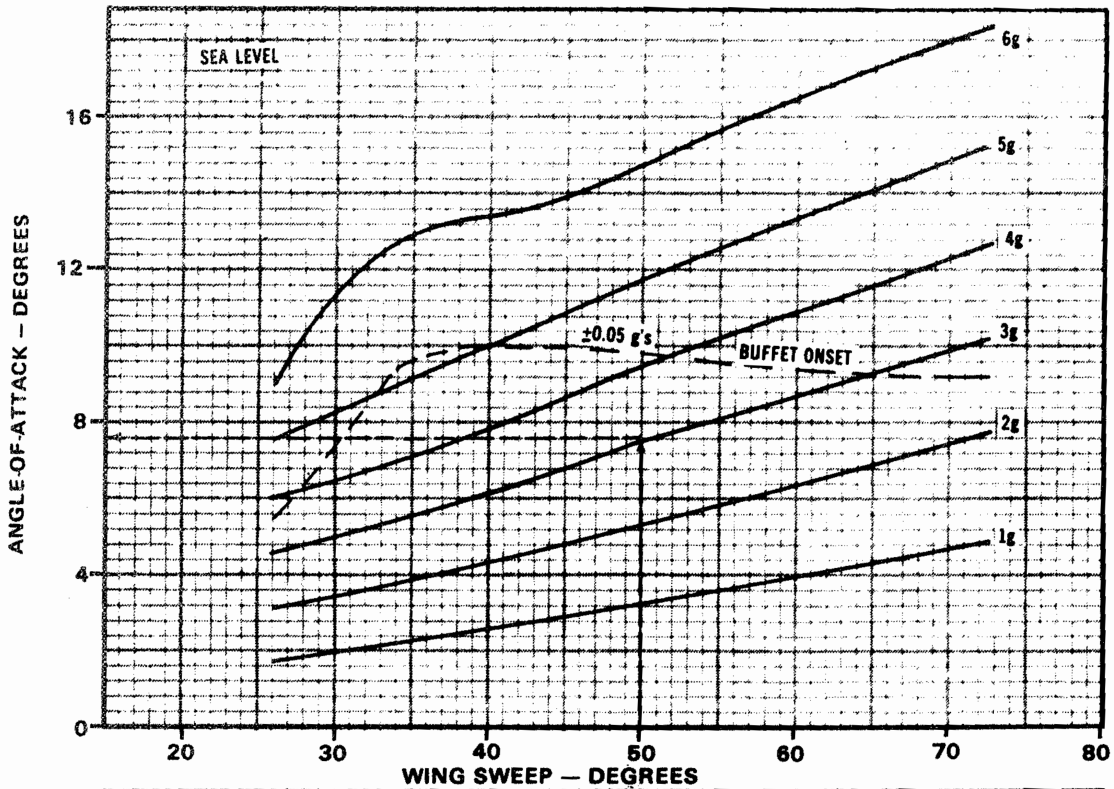
# WING SWEEP MANEUVERABILITY EFFECTS

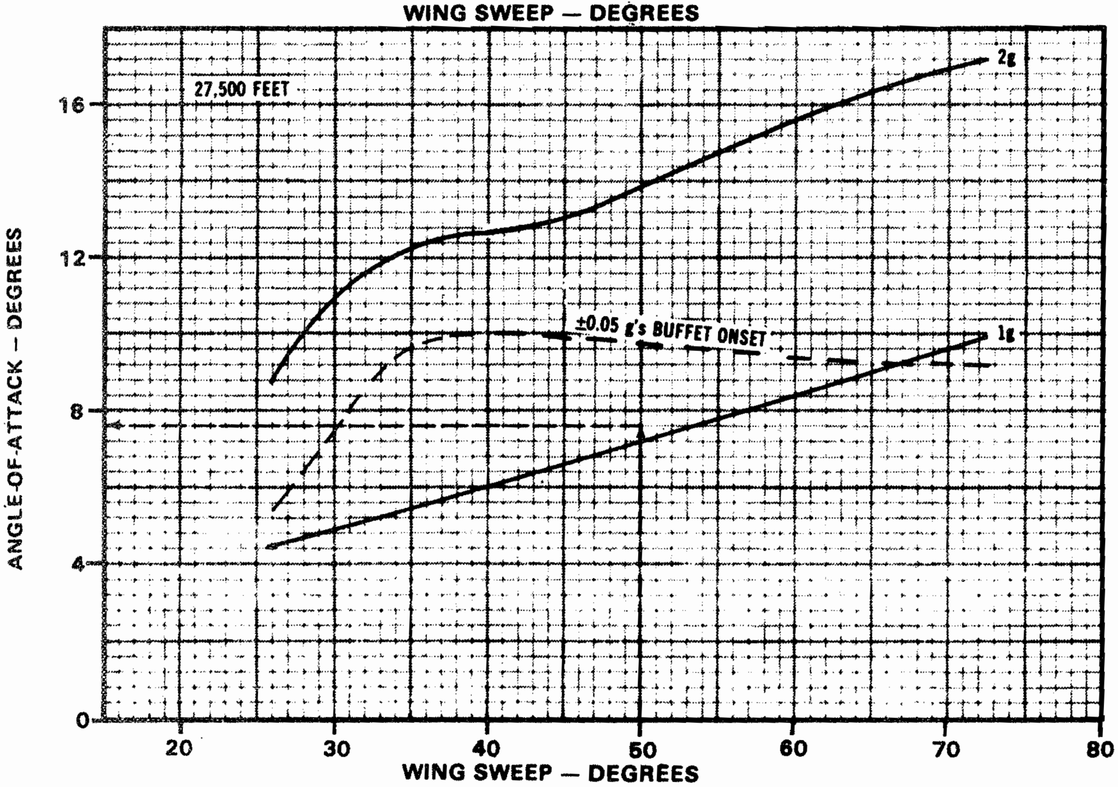
DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION
GEAR AND FLAPS UP
SLATS RETRACTED

CONDITIONS:
GROSS WEIGHT = 70,000 POUNDS
MACH = 0.80

FUEL GRADE: JP-4
ENGINES: TF30-P-3




1EFA-86-1

Figure 6-5. (Sheet 1)

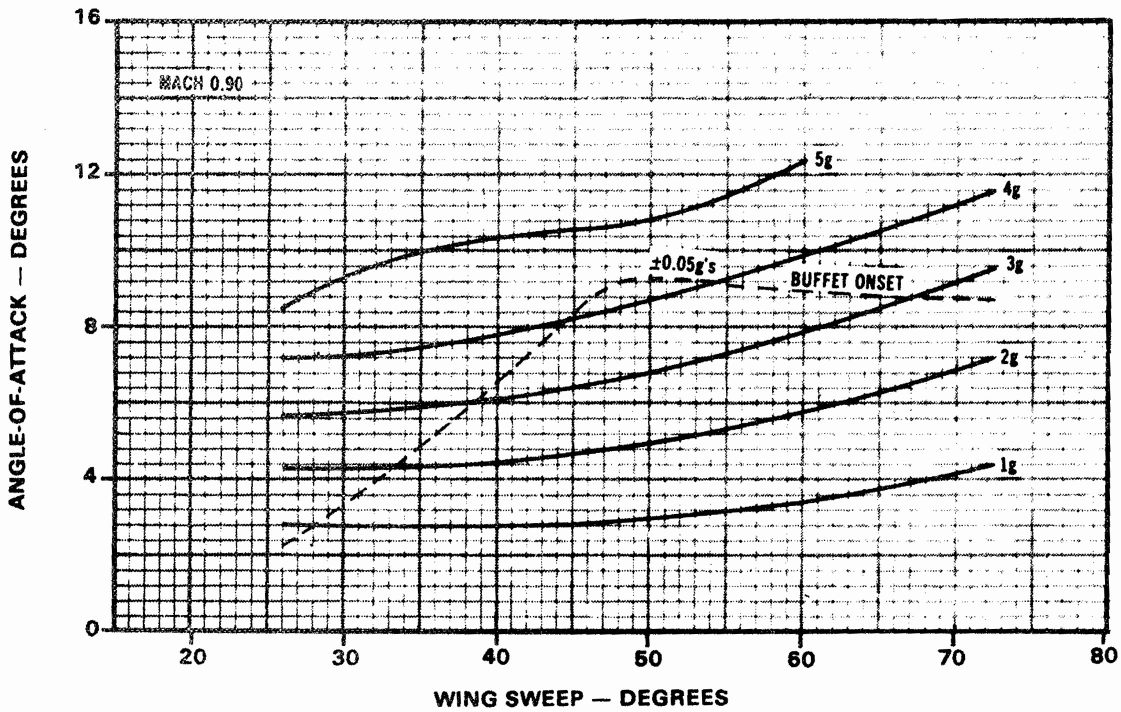
# WING SWEEP MANEUVERABILITY EFFECTS

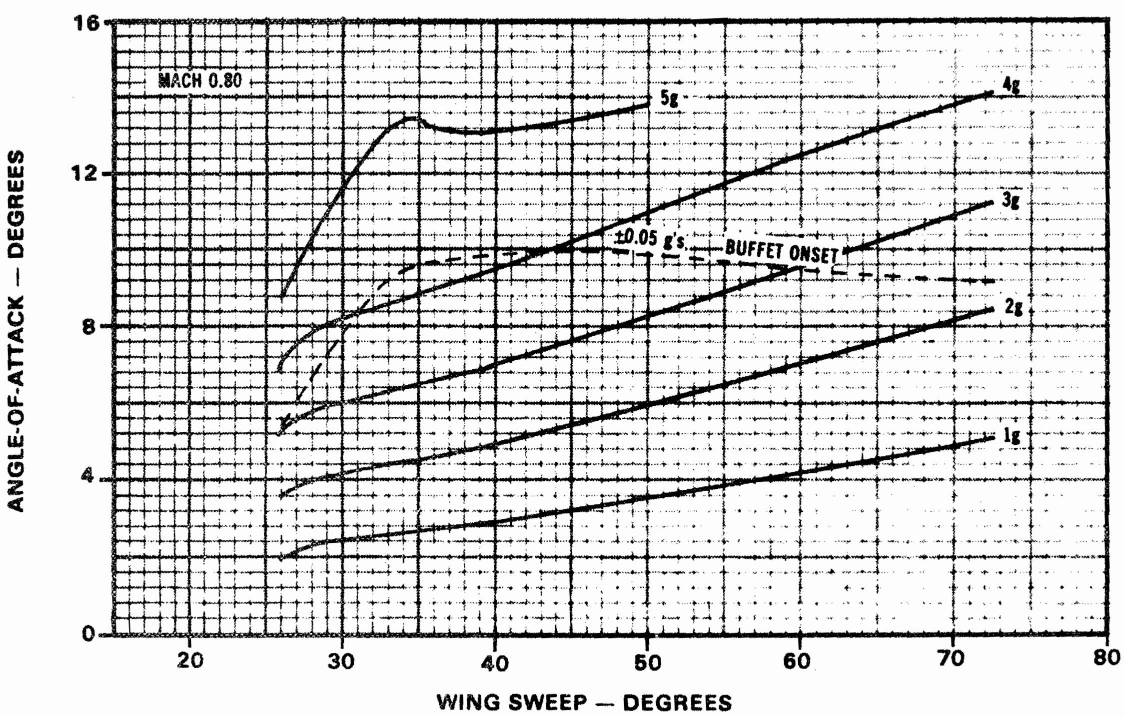
DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR AND FLAPS UP
SLATS RETRACTED

CONDITIONS:
GROSS WEIGHT = 70,000 POUNDS
ALTITUDE = 5,000 FEET

FUEL GRADE: JP-4
ENGINES: TF30-P-3




1EFA-86-2

Figure 6-5. (Sheet 2)

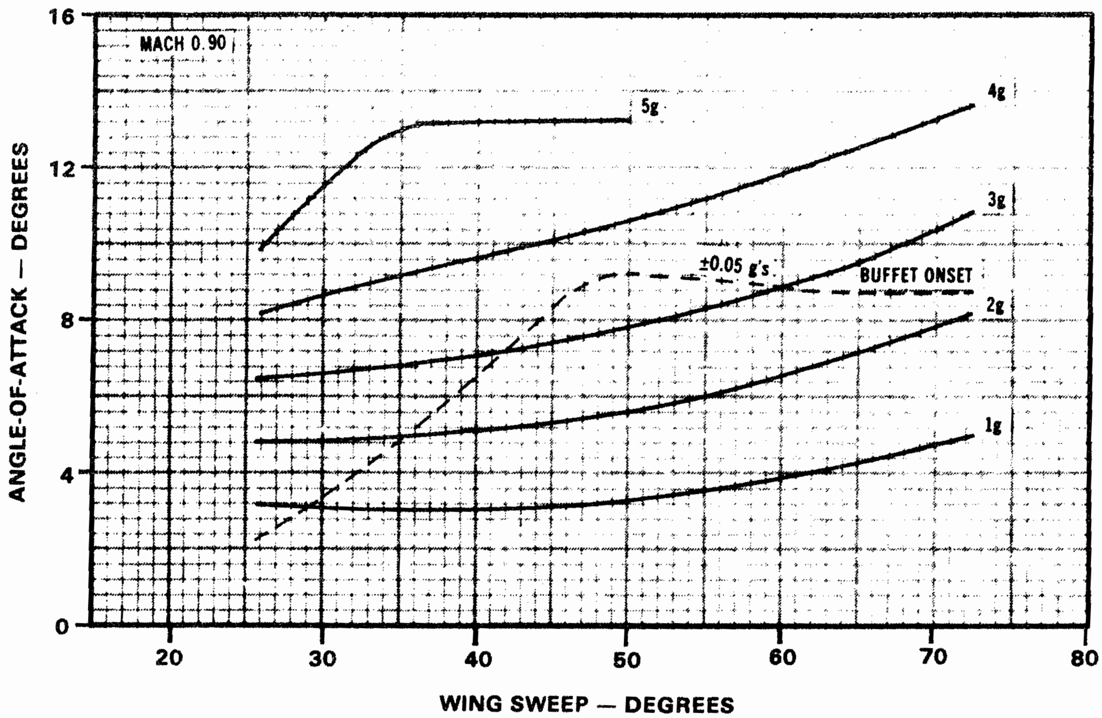
# WING SWEEP MANEUVERABILITY EFFECTS

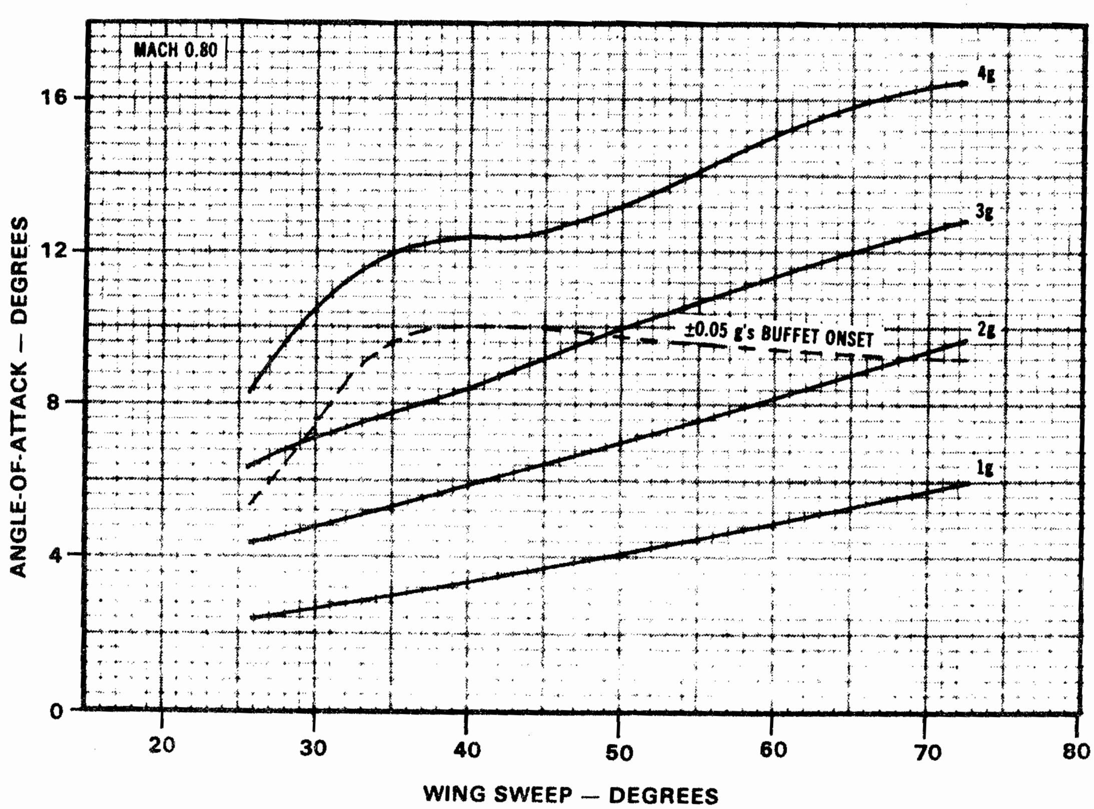
DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR AND FLAPS UP
SLATS RETRACTED

CONDITIONS:
GROSS WEIGHT = 70,000 POUNDS
ALTITUDE = 10,000 FEET

FUEL GRADE: JP-4
ENGINES: TF30-P-3




1EFA-86-3

Figure 6-5. (Sheet 3)

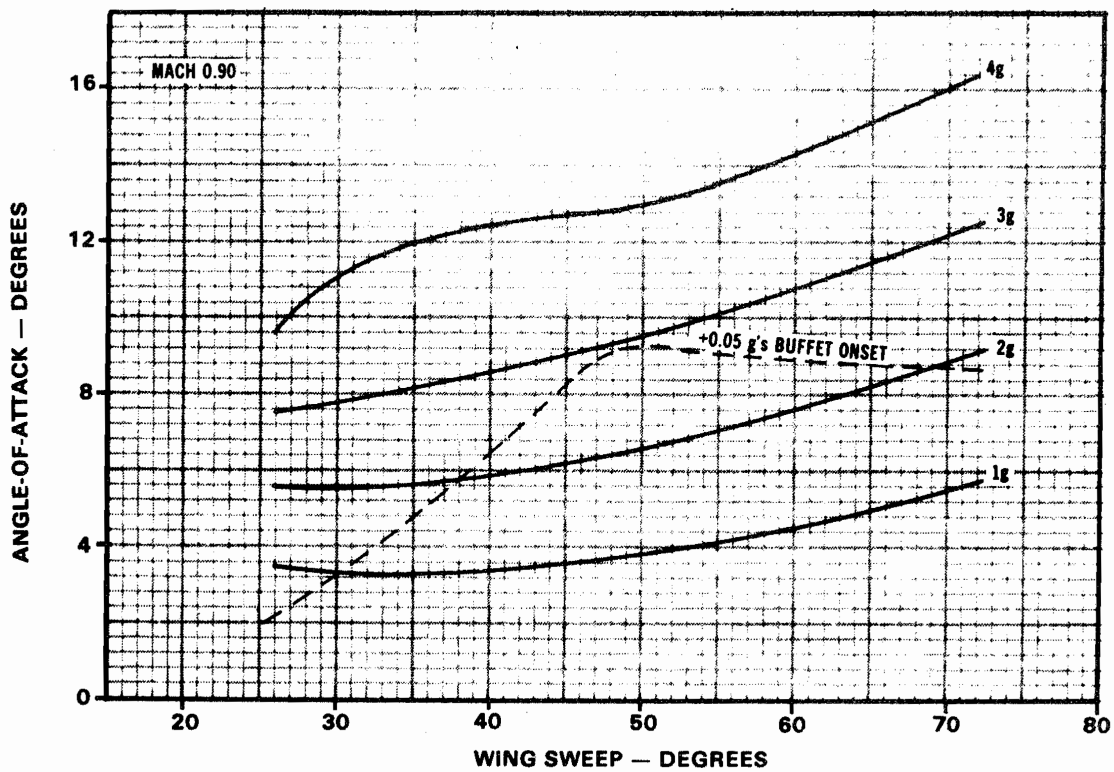
# WING SWEEP MANEUVERABILITY EFFECTS

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR AND FLAPS UP
SLATS RETRACTED

CONDITIONS:
GROSS WEIGHT = 70,000 POUNDS
ALTITUDE = 15,000 FEET

FUEL GRADE: JP-4
ENGINES: TF30-P-3



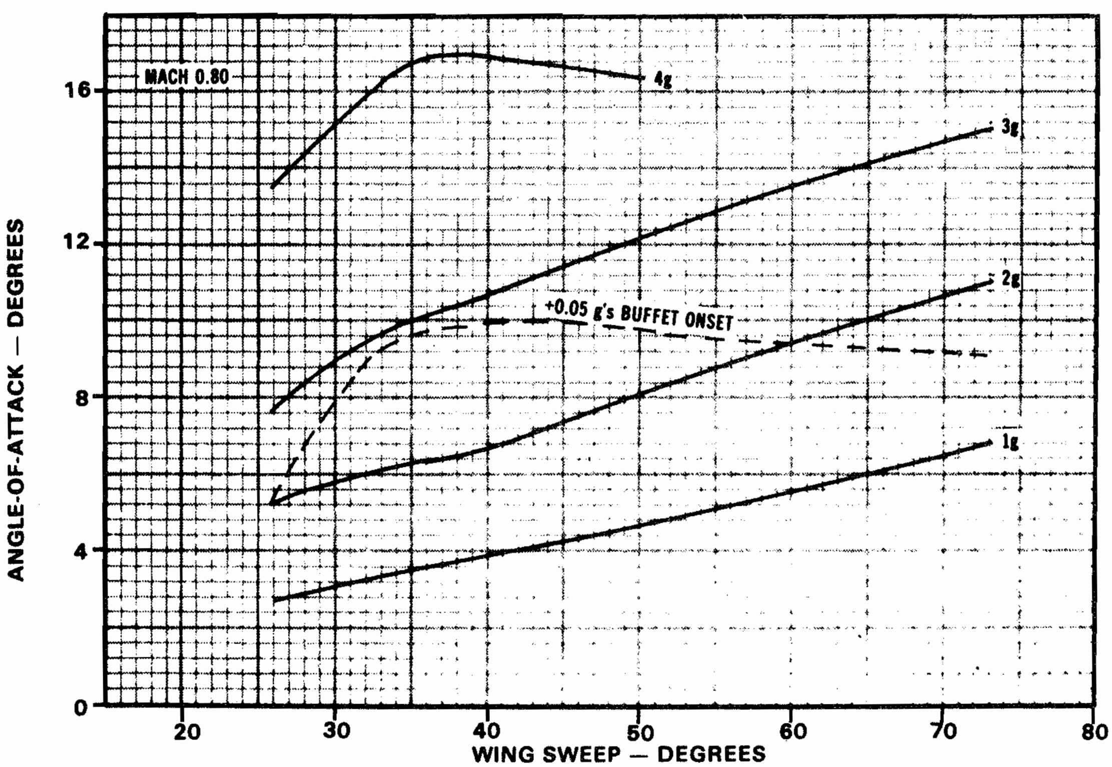




Figure 6-5. (Sheet 4)

sweep with a parameter of constant load factors for typical high subsonic speeds (Mach 0.80 and 0.90). Data are presented for 70,000 pounds gross weight and altitudes of sea level, 5,000, 10,000, 15,000, and 27,500 feet. 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 wings are swept aft, the buffet onset margin improves until the wings are 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 15 pounds. Normal rolling performance below Mach 0.80 decreases significantly when the wings are swept aft of 45 degrees because the spoilers become inoperative. Adverse yaw (aircraft nose moving in a direction opposite to the roll) is minimized by action of the beta reducer, which begins to function at 7 degrees wing angle-of-attack and reaches its maximum effectiveness at 16 degrees angle-of-attack. The beta reducer increases the average roll rate so that roll performance is improved and more precise maneuvering is possible.

**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.
- For rolling maneuver limitations, refer to Section V.

## 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 improve the probability of 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 VMC 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. Limit lateral maneuvers to 60 degrees of bank angle, and do not exceed angle-of-attack limits.

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 and loss of angle-of-attack limiting. 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.

### SLATS AND FLAPS RETRACTED

Loss of the pitch damper will result in degrading damping characteristics, loss of angle-of-attack limiting above 11 degrees angle-of-attack and 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 aircraft 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 supersonic speeds above Mach 1.1. 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 yaw damper will also result in loss of beta reducer operation. 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. Aircraft roll damping can be improved by sweeping wings forward to 50 degrees at supersonic speeds. Roll trim can be accomplished by using rudder trim.

**CAUTION**

With loss of yaw and/or roll dampers the aircraft is presently restricted to Mach 1.1 for wing sweeps of 46 to 72.5 degrees at all altitudes.

## MINIMUM AIRSPEEDS

**WARNING**

- Under no circumstances should the angle-of-attack limits or stall warning activation be exceeded. Inadvertent stall and post-stall gyrations may result from exceeding these limits.

- Minimum airspeeds shown in figure 6-6 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 18 degrees. Figure 6-6 presents these speeds for 1g 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 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 wing angle-of-attack of 18 degrees for all wing sweeps. Figure 6-6 also presents these speeds for 1 g flight and 2 g flight (60-degree bank). These speeds are based on center-of-gravity positions representative of normal fuel sequencing.

## THRUST REQUIREMENTS

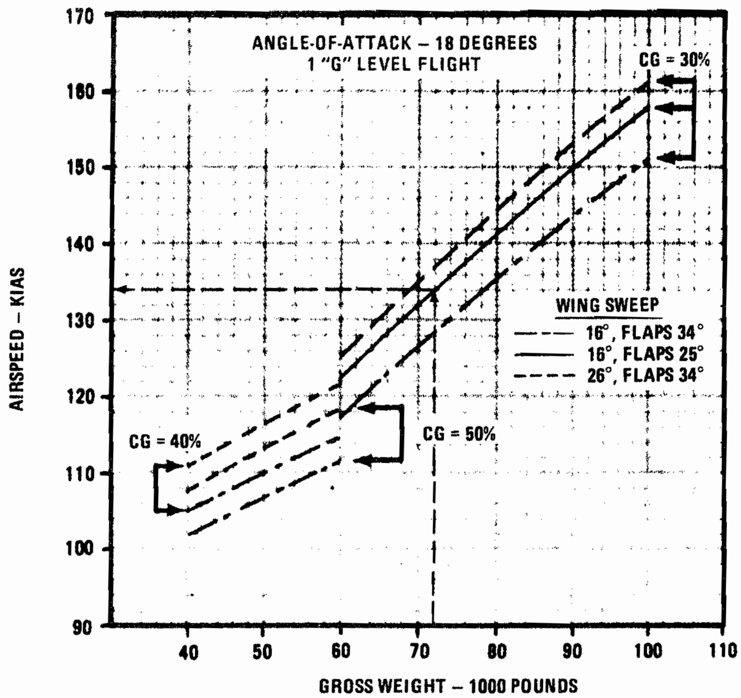
Particular attention to thrust requirements versus airspeed is essential in this aircraft because of its variable sweep wing, sharp drag increase at high angles-of-attack (typical of aircraft with high wing loading), and relatively slow thrust buildup during engine acceleration. Figure 6-7 shows how 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 increase or increase in thrust requirement that exists at higher angles-of-attack. This drag increase (the steep slope in the left side of the thrust required curve) occurs over a very small range of airspeeds. Angle-of-attack limiting will preclude entry into a stall, but high sink rates will result. 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 aircraft, especially at high angles-of-attack. Each of these changes will be discussed separately.

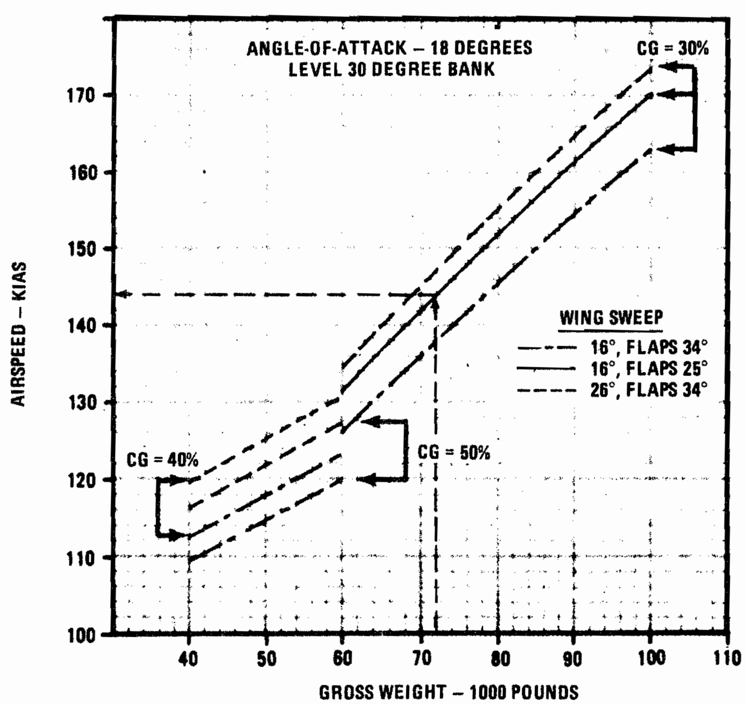
# MINIMUM AIRSPEED

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR AND FLAPS DOWN
SLATS EXTENDED
CLEAN AIRCRAFT

FUEL GRADE: JP-4
ENGINES: TF30-P-3




1EFA-87-1

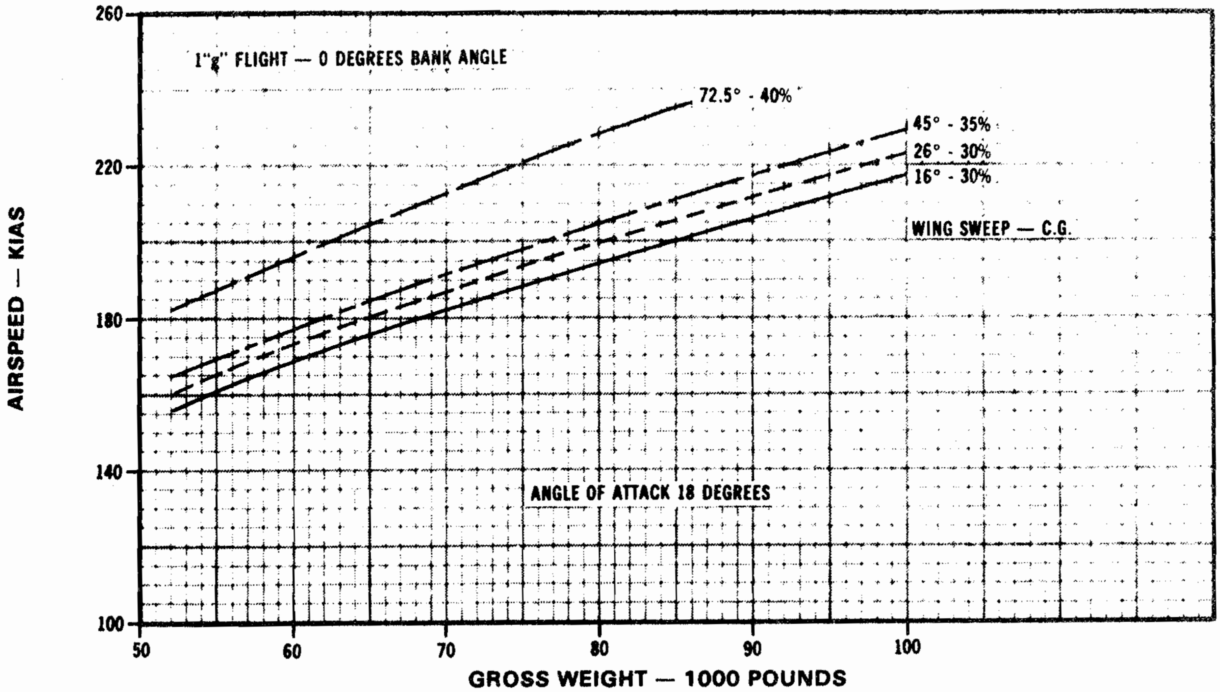
Figure 6-6. (Sheet 1)

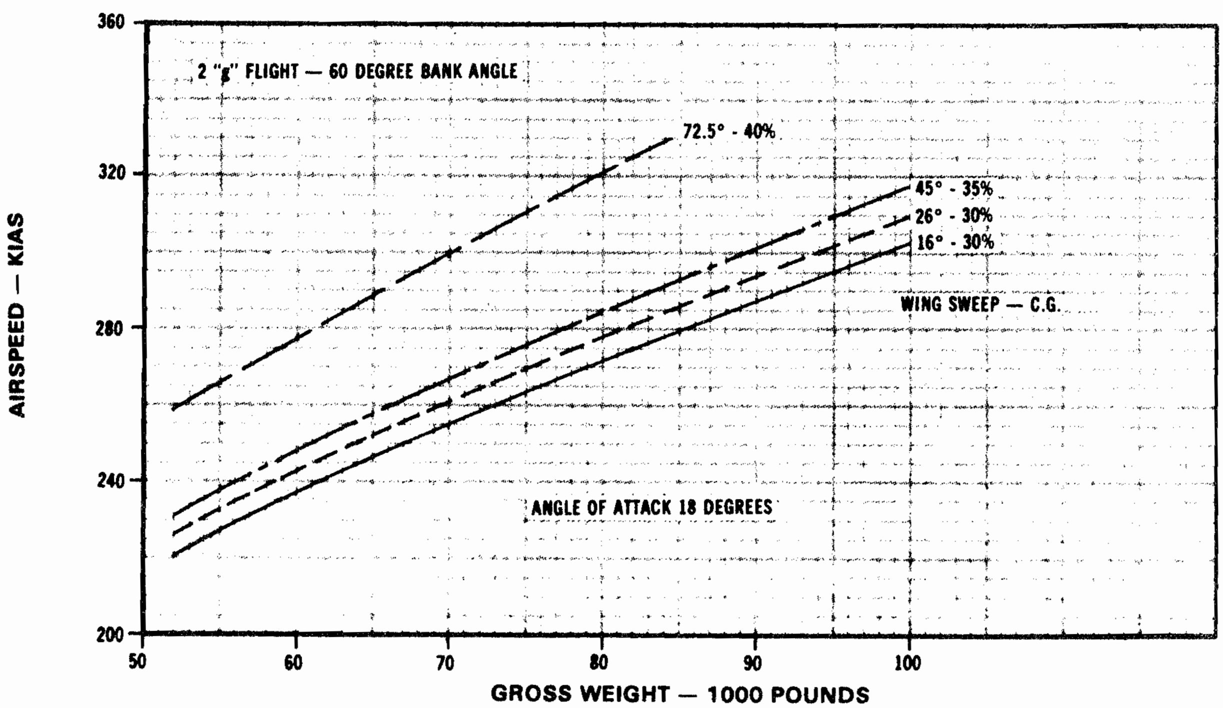
# MINIMUM AIRSPEED

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

CONFIGURATION:
GEAR UP AND FLAPS UP
SLATS RETRACTED
CLEAN AIRCRAFT

FUEL GRADE: JP-4
ENGINES: TF30-P-3




1EFA-87-2

Figure 6-6. (Sheet 2)

# THRUST VERSUS AIRSPEED

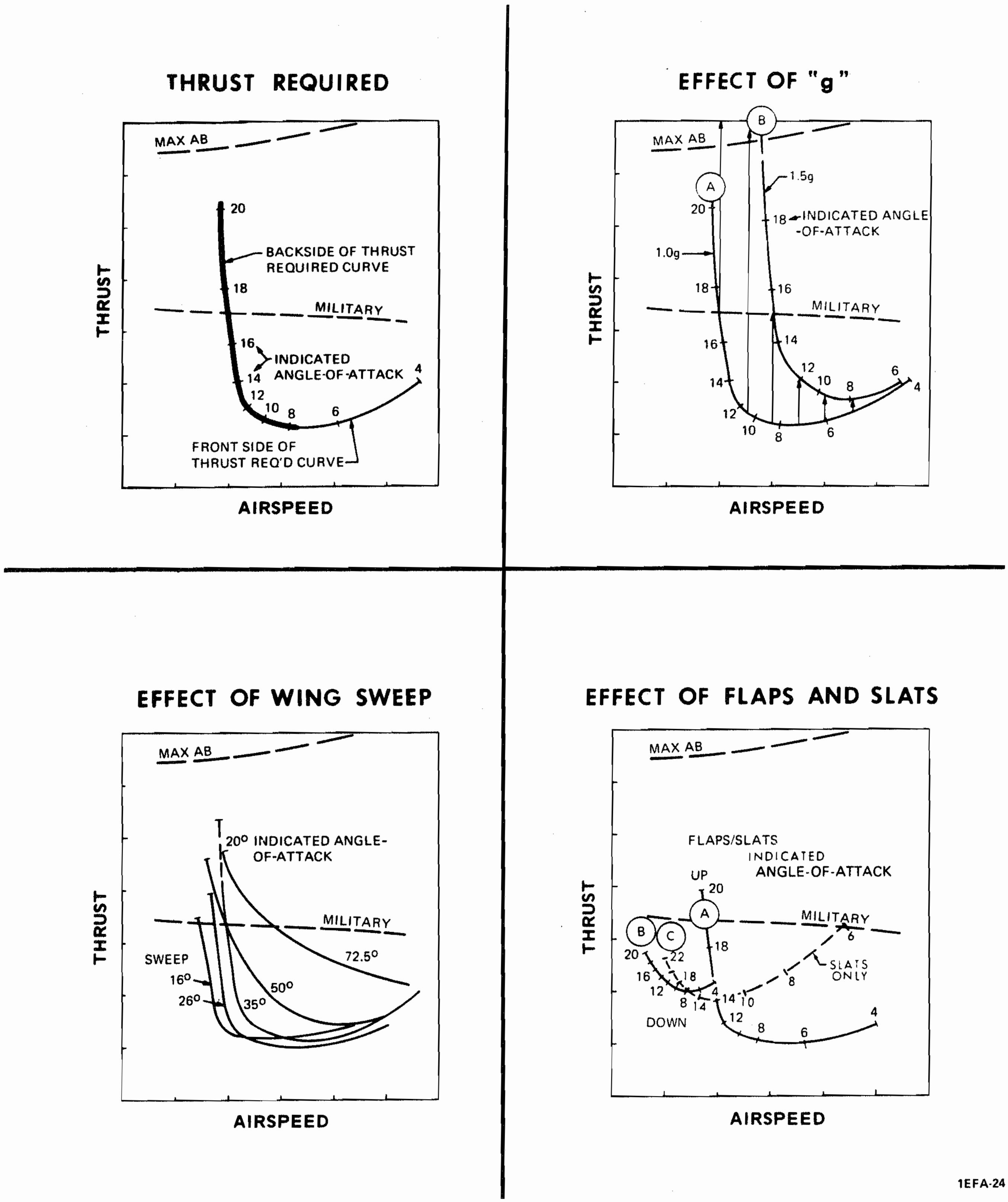



Figure 6-7

## 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 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 6-7. 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. Angle-of-attack will be limited to approximately 20 degrees but large sink rates will result.

## EFFECT OF G

An understanding of the effect of g on thrust required may be obtained by referring to figure 6-7 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 6-7 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 6-7 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**

Delay in selection of flaps and slats can be critical during decelerating flight.

## EFFECT OF WING SWEEP

Figure 6-7 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, increasing back stick force or nose-up trim is required when angle-of-attack exceeds 11 degrees. Although the angle-of-attack limits will not prevent flight on the backside of the power required curve, it will provide the pilot with a direct indication of the decelerating condition without reference to the angle-of-attack indicator.

## CORRECTIONS

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.

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.

## 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 indicated 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. Angle-of-attack will be limited to 20 degrees, but large sink rates can develop before maximum afterburner thrust can be attained. 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.

**WARNING**

Sink rate control is critical during no flap approaches due to operation in this high angle-of-attack region.

## SINGLE ENGINE THRUST REQUIREMENTS

Flight in the landing configuration with one engine shut down requires close attention to the effects of gross weight, altitude, and ambient temperature on thrust required/available. Level flight in the landing configuration may require thrust in excess of MIL power on the operating engine. Figure 6-8 shows four plots that illustrate various aspects of single engine thrust requirements. The Effect of Gross Weight plot shows that thrust required increases as gross weight increases. For comparison, MIL and MAX AB thrust available at both sea level and 5,000 feet MSL are illustrated. This plot should be used for comparison with the thrust required/available relationships on the other three plots to understand the effects of increased ambient temperature, retracting the landing gear, or descent rate. As ambient temperature increases, thrust available decreases; retracting the landing gear or descending reduces thrust required.

# Single Engine Thrust Required

CONFIGURATION:

- WING SWEEP 16 DEG
- FLAPS 25 DEG
- ONE ENGINE WINDMILLING

FUEL GRADE: JP-4
ENGINES: TF30-P-3

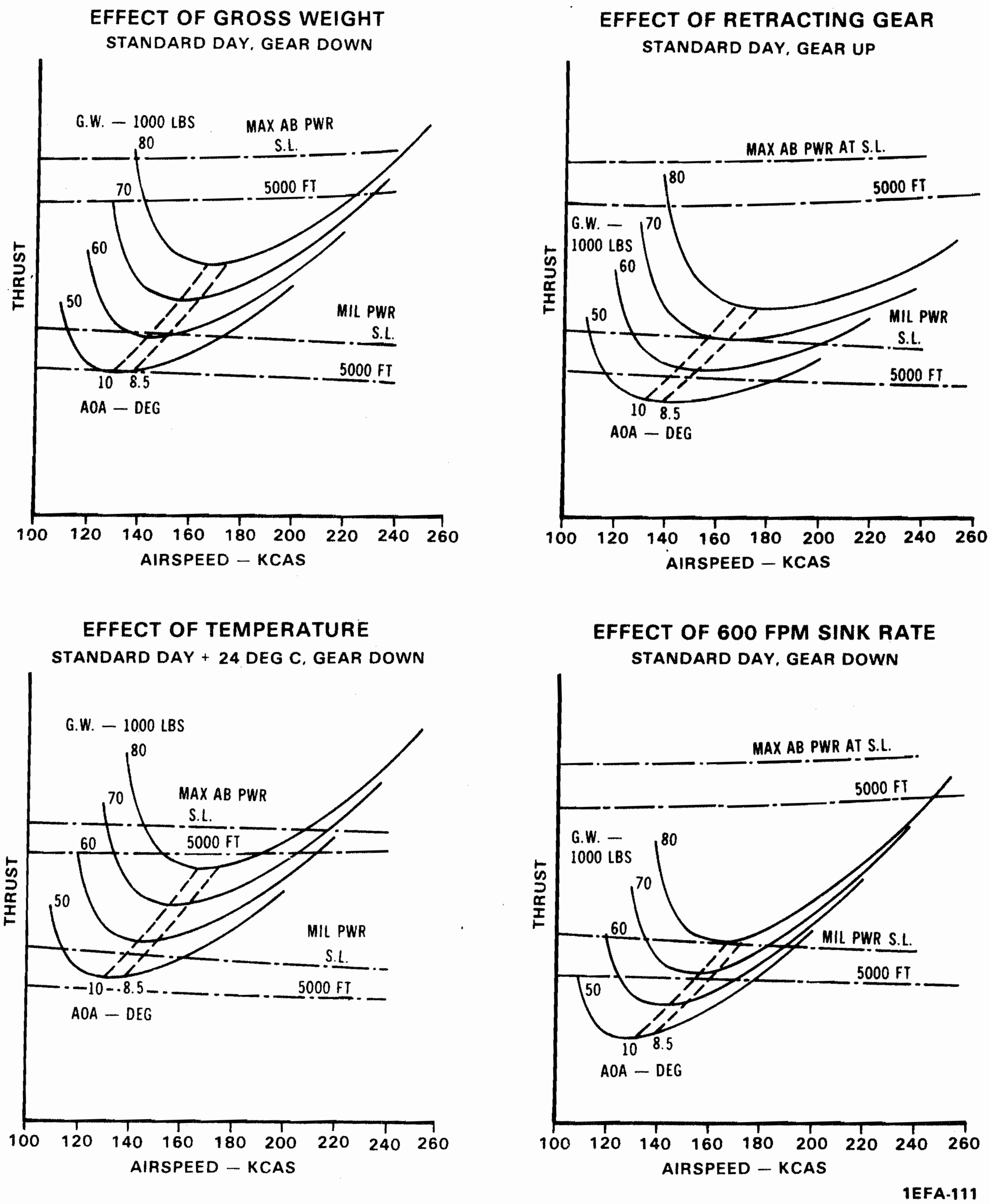



**Figure 6-8.**

## LIFT AVAILABLE

The relationship of angle-of-attack and lift coefficient (lift available at a given airspeed) is depicted in figure 6-9. This figure demonstrates that lift available at a given airspeed increases with angle-of-attack through stall (directional stability zero). It also shows the effect of configuration changes. Landing gear position does not affect the lift available.

## 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. Refer to Figure 6-9.

**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 high sink rates 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.

## STALL WARNING CHARACTERISTICS

The aircraft will not exhibit sudden, abrupt motions 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. The first two characteristics rely on correct operation of the stall warning and angle-of-attack systems. Malfunctions of these systems could give false indications of a stall or delay recognition of an actual stall.

1. Artificial Stall Warning System. This system provides simultaneous actuation of the pedal shaker, a flashing red lamp and a steady tone. System operation is described in Section I. The stall warning safety margin is depicted in figure 6-2.

2. 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 airspeds, g loadings, gross weights, wing sweeps, attitudes, thrust settings, configuration 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.

3. A High Sink Rate Most Evident During 1 g Stalls. During 1 g flight stall approaches at MIL thrust, sink rates of 3,000 to 6,000 feet per minute can develop. Pre-stall sink rates are greater at aft wing sweeps. Sink rates will be noticed through increasing back stick force or nose-up trim requirements.

4. 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.

5. 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.

6. 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.

# AOA RELATIONSHIPS

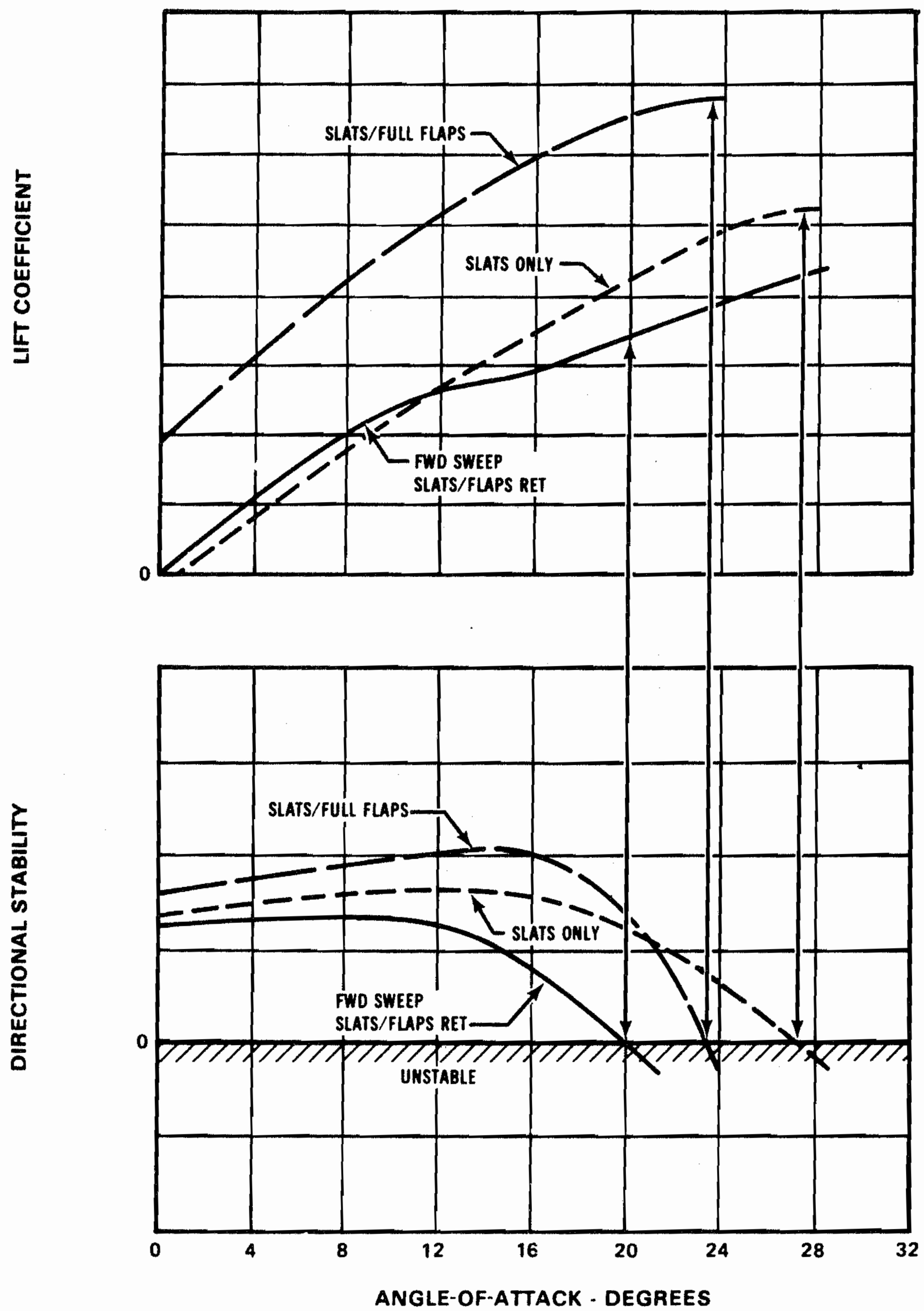



**Figure 6-9.**

**WARNING**

- During maneuvering flight at high angles-of-attack within limits, large, abrupt aft stick inputs can result in overshoots of angle-of-attack limits and produce susceptibility to loss of control. Large, abrupt cross-controlling and reversals may force the aircraft into departure from controlled flight.
- 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.

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**

The immediate action that must be taken when an aircrew realizes that the aircraft has departed controlled flight (above 20 degrees angle-of-attack) is to reduce angle-of-attack. The forward stick and neutral rudder recovery inputs must be given time to be effective. Maintain these controls until a spin or recovery is identified.

## 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 F-111A 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 "Stall Prevention/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 22 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 "Stall Prevention/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 use less than full forward stick. 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 in excess of 200 degrees per second may occur and be sustained. To recover, neutralize controls.

**WARNING**

Abrupt forward stick movement during rolling maneuvers below 20 degrees angle-of-attack may result in uncommanded roll rate increases in excess of 200 degrees per second.

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 (rapidly increasing angle-of-attack), 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 14 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 14 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 14 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. 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 over-rotation and lead to another out-of-control condition. The dive pullout should be conducted at no more than 14 degrees angle-of-attack.

## 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.

A dive recovery chart is presented in figure 6-10 and may be used as follows:

Given:

| | | |
|---|---|---|
| Wing Sweep | - | 26 degrees |
| Dive Angle | - | 30 degrees |
| Airspeed | - | 500 KIAS |
| Ambient Temperature | - | 0°C |
| Start Recovery | - | 6,000 feet |
| Desired Load Factor | - | 3.0 g's |
| Gross Weight | - | 60,000 lbs. |

Find:

Altitude required to recover

Solution:

Enter figure 6-10 at 500 KIAS A, proceed horizontally to the right to 6,000 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 6-10 on the left airspeed scale 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.

## FLIGHT WITH SPEED BRAKE EXTENDED

With dampers off, extension of the speed brake will cause pitch attitude to increase. Extension of the speed brake will also result in aircraft buffet. Refer to Section V for operating limitations of speed brake extension.

## FLIGHT WITH ABNORMAL CENTER OF GRAVITY CONDITION

The relationship between aircraft cg position and the aft allowable cg position can be determined by the use of T.O. 1-1B-40 and figure 6-11 of this manual or by the elevator position check in Section II. An approximate determination of this relationship may be obtained from figures 6-11 and 6-12. If the cg is forward of normal, the aircraft will exhibit a higher degree of positive static longitudinal stability with some corresponding reduction in pitch maneuverability. With an aft abnormal cg the static longitudinal stability of the aircraft will be reduced, and the pitch response of the aircraft will become increasingly sensitive. If the aft allowable cg limit is exceeded, the aircraft may exhibit neutral or negative static longitudinal stability. With neutral stability, and dampers off, the aircraft will be difficult to trim, and if momentary angle-of-attack disturbances are encountered, the aircraft will tend to stabilize at some new angle-of-attack. If negative longitudinal stability exists with dampers off, any disturbances of the angle-of-attack will continue to increase in magnitude until corrected by the pilot. With the dampers on, the command augmentation and stability augmentation features of the flight control system will conceal the effects of degraded stability until damper travel limits are reached. With an extreme aft center of gravity condition, elevator authority will be inadequate to control the resulting negative longitudinal stability.

## DETERMINATION OF AFT ALLOWABLE CENTER-OF-GRAVITY POSITION

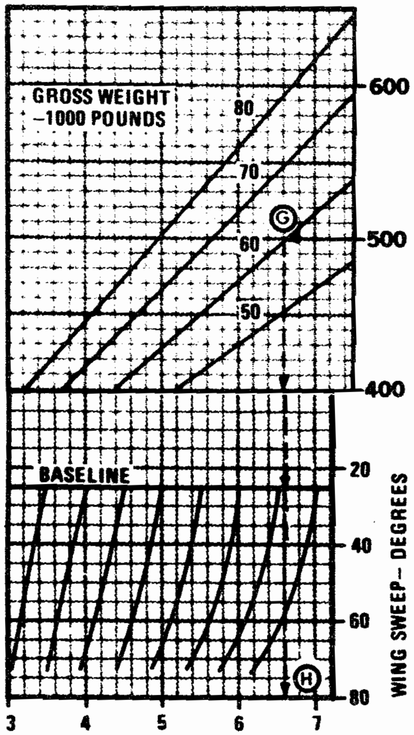
The aft allowable center-of-gravity positions presented in this section are given to allow calculation of, and verification of the minimum fuel values presented in "Center-of-Gravity Limitations," Section V. In order to calculate the minimum fuel loadings presented in Section V, the user must not only 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 6-11 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 0.3, 0.55 and 0.80 Mach or greater. For Mach numbers between

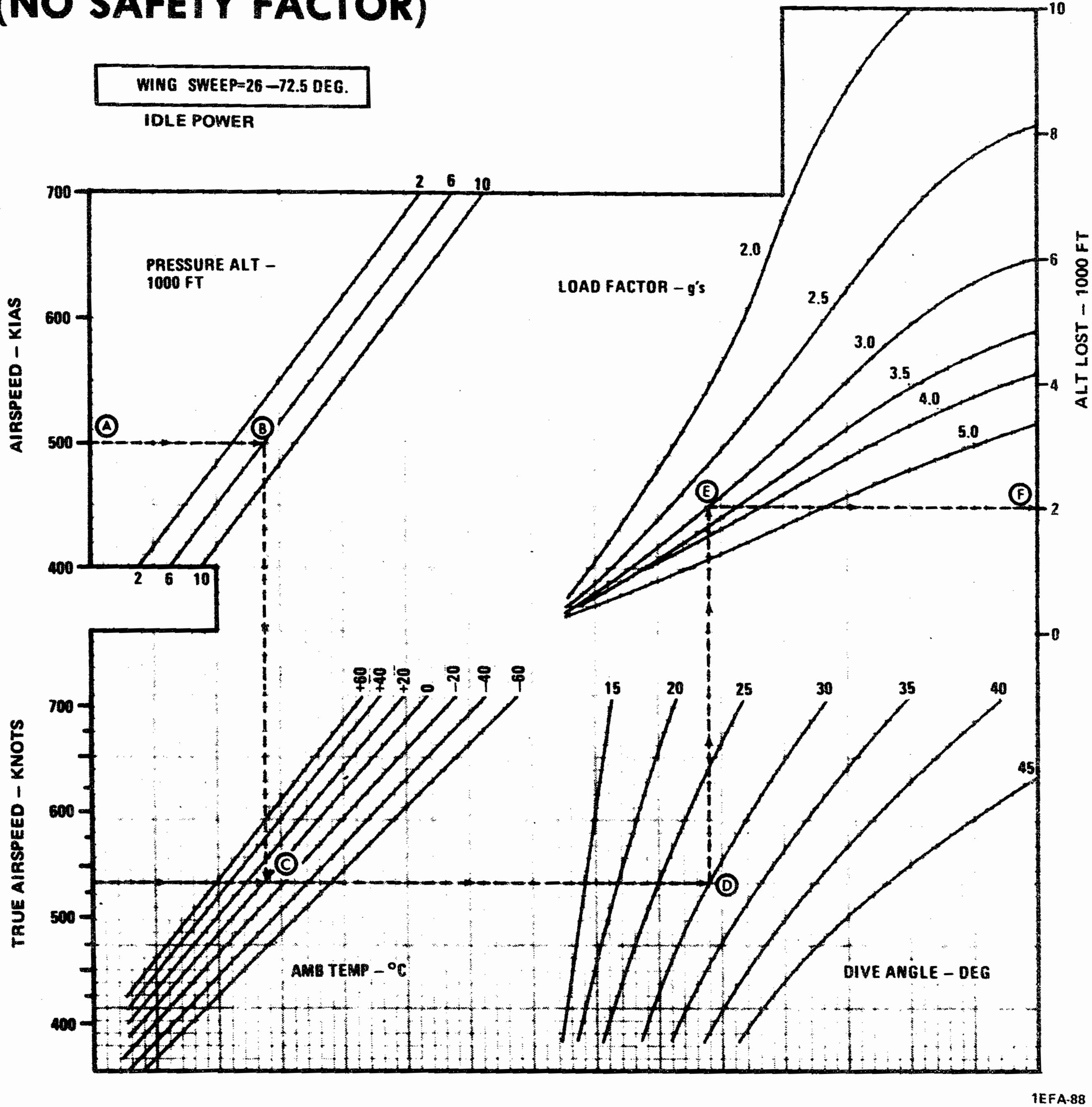
# DIVE RECOVERY (NO SAFETY FACTOR)

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

WING SWEEP=26 –72.5 DEG.

IDLE POWER




NOTES:

* (1) LOAD FACTOR BASED ON

| ANGLE-OF-ATTACK | WING SWEEP |
|---|---|
| 15° ............ | 26°– 45° |
| 18° ............ | 46°– 72.5° |

(2) RECOVERY BASED ON INITIATION OF:

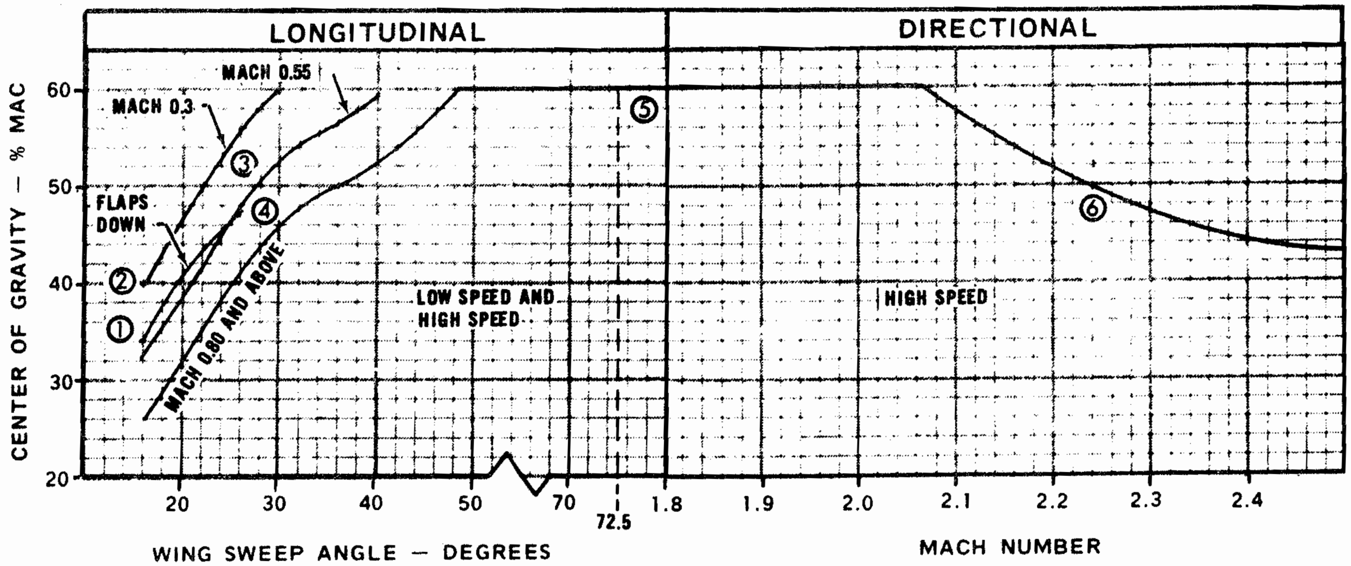
2 g's IN 1 SEC
3 g's IN 1.5 SEC
4 g's IN 2 SEC
5 g's IN 2 SEC

Figure 6-10.

# AFT ALLOWABLE CENTER OF GRAVITY POSITIONS

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
CLEAN AIRCRAFT




## LONGITUDINAL

Flaps Down
- 25 degrees flaps.................. – 1.0%
- (2) or (4) MXU-648 cargo pods...... – 1.0%

Flaps Up
- Slats only ...................... – 2.0%
- (2) or (4) MXU-648 cargo pods...... – 2.0%

## STABILITY INCREMENTS

Note

- Add the sum of all applicable limits at left to the limits in the above chart.
- Linear interpolation required between Mach 0.3, 0.55 and 0.80.

A-1EA-89

**Figure 6-11.**

0.80 and 0.3, 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 and all applicable increments. The longitudinal aft allowable positions for wing sweep angles of 16 through 50 degrees are based on maintaining a 1 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 wing sweep and airspeed for specific phases of flight.

Given:

a. Takeoff: Low speed, flaps and gear down, 16 degree wing sweep, 25 degrees flap.

b. Low Speed: Flaps and gear up, 16 degree wing sweep, slats extended, Mach 0.3.

c. Low Speed: Flaps and gear up, 26 degree wing sweep, clean aircraft, Mach 0.3.

d. High speed: 26 degree wing sweep, Mach 0.70.

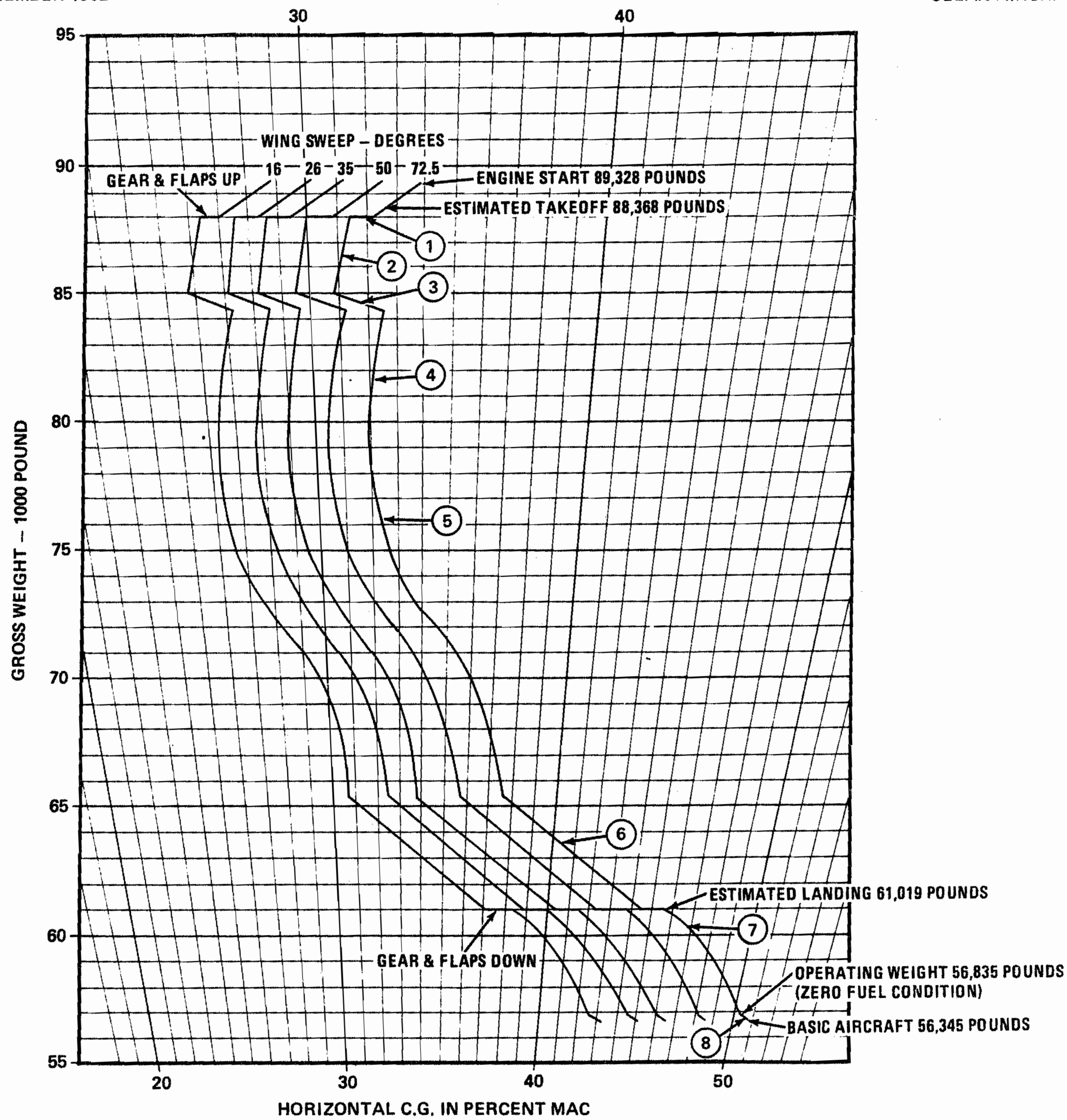
e. High speed: 72.5 degree wing sweep, Mach 2.20.

f. Landing: Low speed, flaps and gear down, 26 degree wing sweep.

# CENTER OF GRAVITY ENVELOPE (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATON:
CLEAN AIRCRAFT




| FULL SEQUENCE (8200 POUNDS DIFFERENTIAL) | | |
|---|---|---|
| | TANK | POUNDS FUEL USED |
| 1 | WING | 960 (TAXI AND TAKEOFF) |
| 2 | WING | 4100 (WINGS EMPTY) |
| 3 | FORWARD | 767 (8300 FWD AND AFT TANKS) |
| 4 | FORWARD AND AFT | 5158 (A2 EMPTY) |
| 5 | FORWARD AND AFT | 13,124 (A1 EMPTY) |
| 6 | FORWARD | 5688 (F1 AND F2 EMPTY) |
| 7 | FORWARD (RESERVOIR) AND LINES | 2696 (RESERVOIR [2512] EMPTY), (LINES [184] EMPTY) |
| | TOTAL FULL USED | 32,493 POUNDS |
| 8 | CREW, OIL AND OPERATING WEIGHT MISCELLANEOUS – 490 POUNDS | |

**CAUTION**

T.O. 1-1B-40 MUST BE USED TO DETERMINE THE ACTUAL WEIGHT AND BALANCE CONDITIONS OF A SPECIFIC AIRCRAFT.

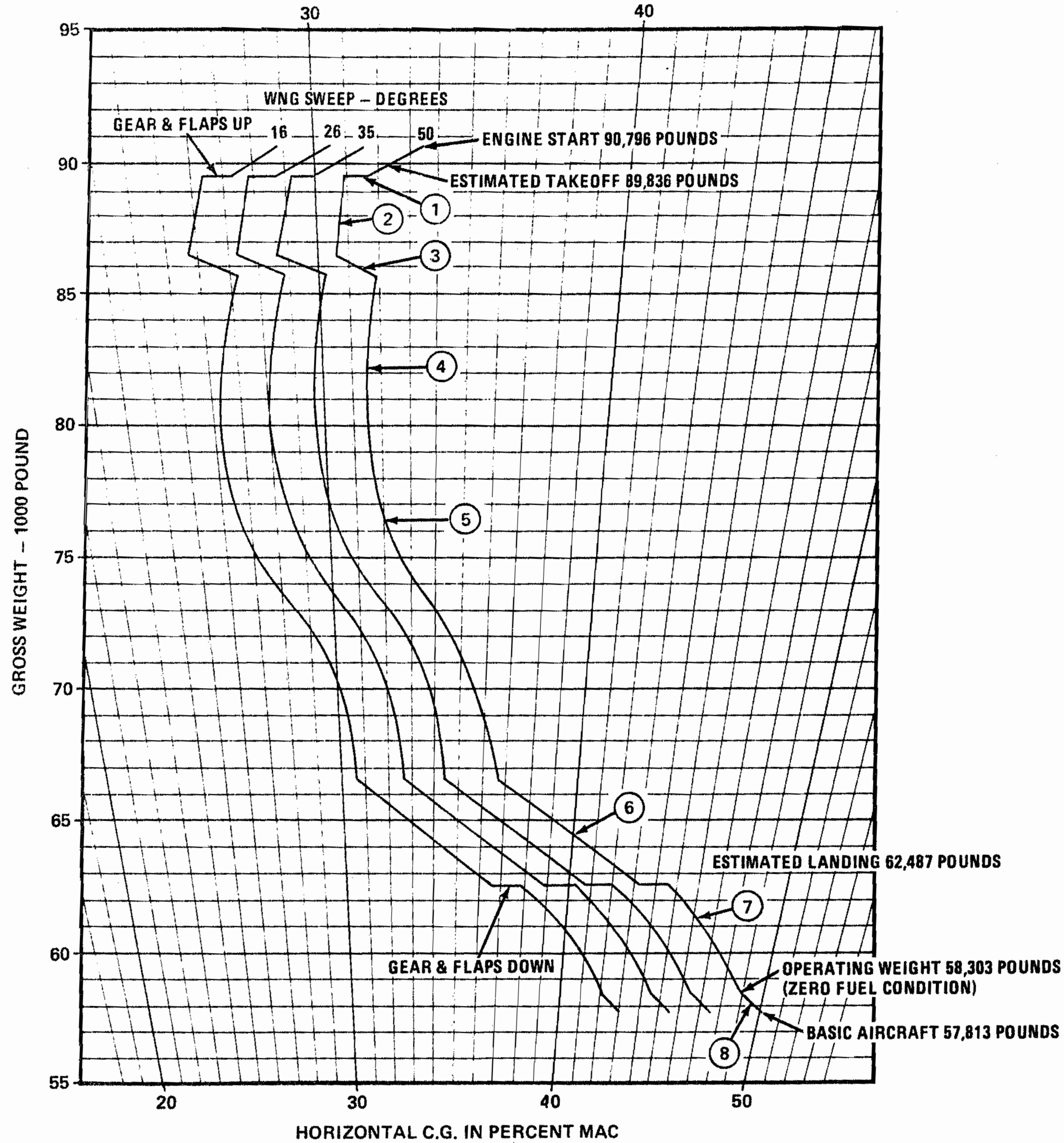
A-1EFA-90-1

**Figure 6-12. (Sheet 1)**

# CENTER OF GRAVITY ENVELOPE (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
(2) MXU-648 CARGO PODS




| FULL SEQUENCE (8200 POUNDS DIFFERENTIAL) | |
|---|---|
| TANK | POUNDS FUEL USED |
| 1 WING | 960 (TAXI AND TAKEOFF) |
| 2 WING | 4100 (WINGS EMPTY) |
| 3 FORWARD | 767 (8300 FWD AND AFT TANKS) |
| 4 FORWARD AND AFT | 5158 (A2 EMPTY) |
| 5 FORWARD AND AFT | 13,124 (A1 EMPTY) |
| 6 FORWARD | 5688 (F1 AND F2 EMPTY) |
| 7 FORWARD (RESERVOIR) AND LINES | 2696 (RESERVOIR [2512] EMPTY), (LINES [184] EMPTY) |
| TOTAL FULL USED | 32,493 POUNDS |
| 8 CREW, OIL AND OPERATING WEIGHT MISCELLANEOUS – 490 POUNDS | |

**CAUTION**

T.O. 1-1B-40 MUST BE USED TO DETERMINE THE ACTUAL WEIGHT AND BALANCE; CONDITIONS OF A SPECIFIC AIRCRAFT.

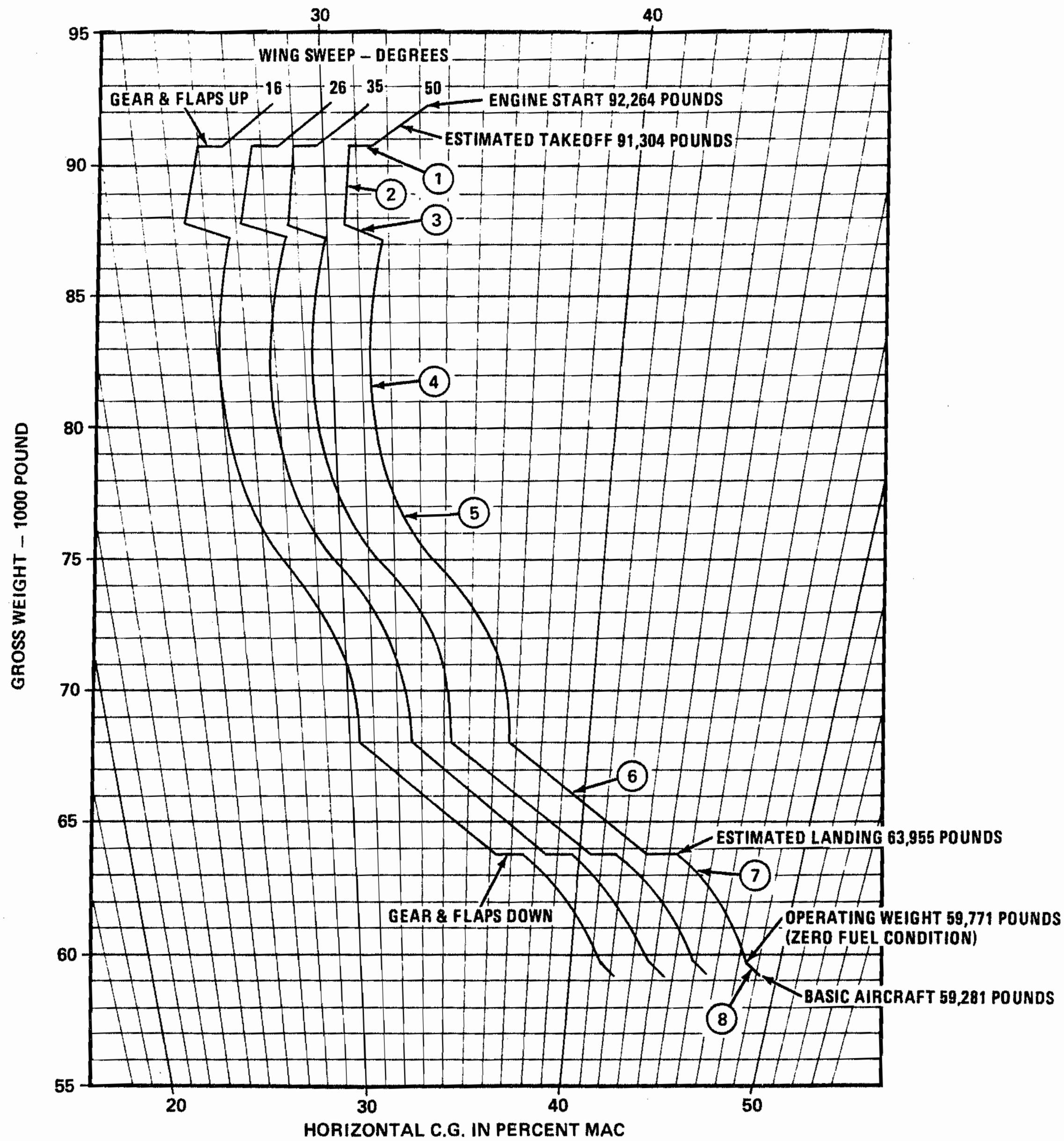
1EFA-90-2

Figure 6-12. (Sheet 2)

# CENTER OF GRAVITY ENVELOPE (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
(4) MXU-648 CARGO PODS




| FULL SEQUENCE (8200 POUNDS DIFFERENTIAL) | | |
|---|---|---|
| | TANK | POUNDS FUEL USED |
| 1 | WING | 960 (TAXI AND TAKEOFF) |
| 2 | WING | 4100 (WINGS EMPTY) |
| 3 | FORWARD | 767 (8300 FWD AND AFT TANKS) |
| 4 | FORWARD AND AFT | 5158 (A2 EMPTY) |
| 5 | FORWARD AND AFT | 13,124 (A1 EMPTY) |
| 6 | FORWARD | 5688 (F1 AND F2 EMPTY) |
| 7 | FORWARD (RESERVOIR) AND LINES | 2696 (RESERVOIR [2512] EMPTY), (LINES [184] EMPTY) |
| | TOTAL FULL USED | 32,493 POUNDS |
| 8 | CREW, OIL AND OPERATING WEIGHT MISCELLANEOUS – 490 POUNDS | |

**CAUTION**

T.O. 1-1B-40 MUST BE USED TO DETERMINE THE ACTUAL WEIGHT AND BALANCE CONDITIONS OF A SPECIFIC AIRCRAFT

1EFA-90-3

**Figure 6-12. (Sheet 3)**

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 0.3

c. Low speed, flaps and gear up, 26 degree wing sweep, clean aircraft, Mach 0.3

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.

Use figure 6-11 in the appropriate speed regime for the particular configuration specified above to determine the following:

Aft Allowable Position

a. Takeoff ........................................... 33.0%

| | |
|---|---|
| Basic | 34.0% |
| 25° Flap | −1.0% |
| | 33.0% |

b. Low speed, flaps and gear up, 16 degree wing sweep, slats extended, Mach 0.3 .............................. 38.0%

| | |
|---|---|
| Basic | 40.0% |
| Slats Extended | −2.0% |
| | 38.0% |

c. Low speed, flaps and gear up, 26 degrees wing sweep, Mach 0.3 ........ 56.0%

| | |
|---|---|
| Basic | 56.0% |

d. High speed, Mach 0.70, 26 degree wing sweep .......................................... 46.4%

| | |
|---|---|
| *Basic | 46.4% |

*Determine basic by linear interpolation between 0.55 and 0.80 Mach.

e. High speed, Mach 2.20, 72.5 degree wing sweep .......................................... 51.0%

(1) Longitudinal

| | |
|---|---|
| Basic | 60.0% |

(2) Directional

| | |
|---|---|
| Basic | 51.0% |

**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 ...........47.5%

| | |
|---|---|
| Basic | 47.5% |

After determining the aft allowable center-of-gravity position, either longitudinal or directional, refer to figure 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 that result in 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 6-12 present a typical aircraft center-of-gravity envelope. Fuel sequencing (pounds used) for an 8,200-pound differential between the forward and aft tanks is maintained if auto engine feed is selected.

## WING SWEEP POSITION FOR LANDING

Figure 6-13 may be used to determine the wing sweep setting required to remain within center-of-gravity limits for landing. A forward cg line is shown for a flap setting of 34° (full) flaps and is applicable only in the clean configuration as there is no attainable forward cg limit with 25° or less flap setting. This chart assumes normal fuel sequence (auto engine feed) and will not be used in conjunction with abnormal fuel distributions.

# WING SWEEP FOR LANDING (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
CLEAN

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONSIDERATIONS:

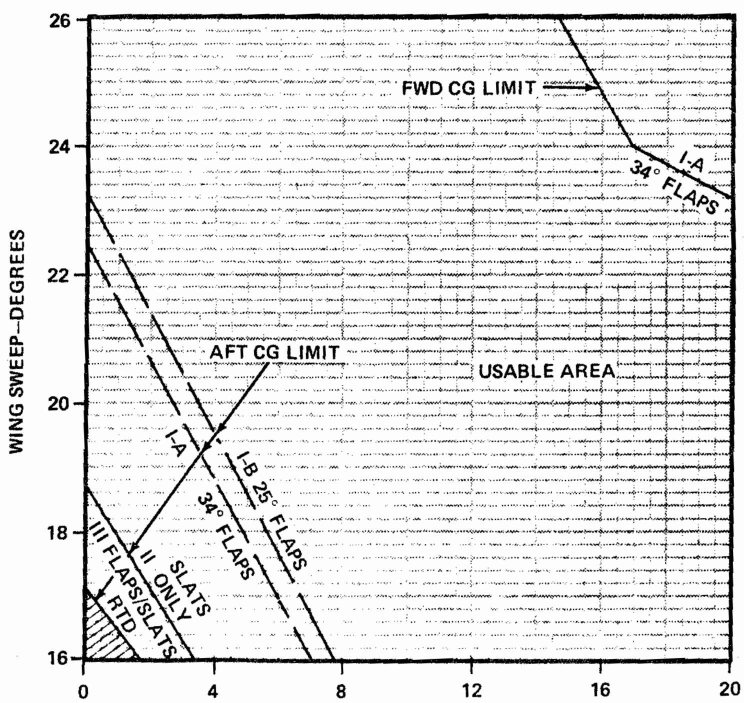
- GEAR EXTENDED
- AUTO FUEL SEQUENCE
- OPERATING WEIGHT CG 47.5% MAC AT 26° WING SWEEP

| | CONFIG I | CONFIG II | CONFIG III |
|---|---|---|---|
| • AFT LIMITS | | | |
| 26° SWEEP | 2° EPI–TEU | 2° EPI–TEU | 1° EPI–TED |
| 16° SWEEP | 4° EPI–TEU | 2° EPI–TEU | 1° EPI–TED |
| • FORWARD LIMITS | | | |
| 26° SWEEP | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| 16° SWEEP | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| • CHECK EPI AT ANGLE-OF-ATTACK | 10° | 11° | 11° |

- WING CONFIGURATION
  - I – SLATS DOWN, FLAPS GREATER THAN 15° (1-A FLAPS AT 34°; 1-B FLAPS AT 25°)
  - II – SLATS DOWN, FLAPS 15° OR LESS (SLATS ONLY)
  - III – SLATS UP, FLAPS 5° OR LESS (RETRACTED)

NOTES:

1. EXCEPT FOR WING CONFIGURATION 1-A, FORWARD CG LIMIT IS NOT APPLICABLE FOR COMBINATIONS OF FUEL REMAINING/WING SWEEP SHOWN.
2. CHOOSE AFT CG LIMIT CORRESPONDING TO ACTUAL WING CONFIGURATION. FOR WING CONFIGURATION 1-A, USABLE AREA IS BETWEEN FORWARD AND AFT CG LIMIT LINES. FOR ALL OTHER WING CONFIGURATIONS, USABLE AREA IS ANYWHERE ON CHART TO RIGHT OF APPLICABLE AFT CG LIMIT.




A-1EFA-91-1

Figure 6-13. (Sheet 1)

# WING SWEEP FOR LANDING (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
(2) MXU-648 CARGO PODS

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONSIDERATIONS:

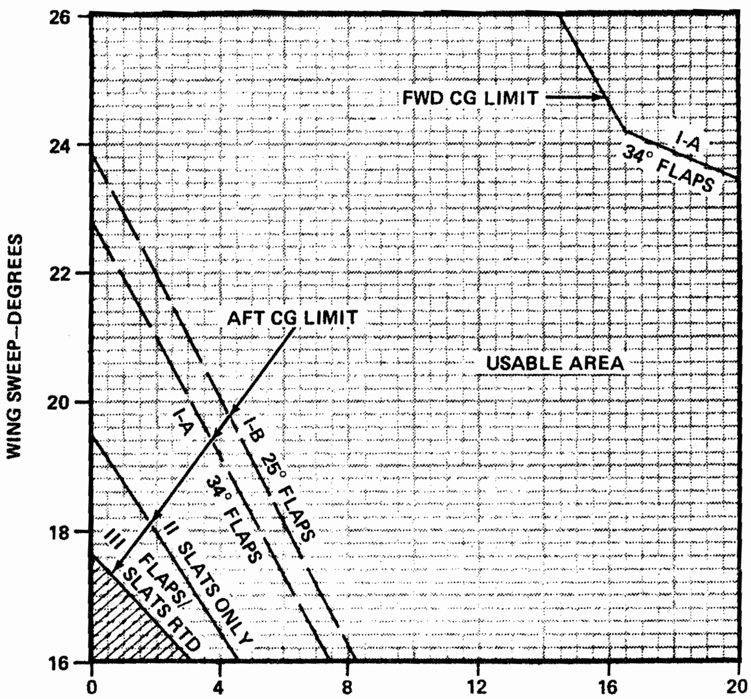
- GEAR EXTENDED
- AUTO FUEL SEQUENCE
- OPERATING WEIGHT CG 47.5% MAC AT 26° WING SWEEP

| | CONFIG I | CONFIG II | CONFIG III |
|---|---|---|---|
| • AFT LIMITS | | | |
| 26° SWEEP .......... | 3° EPI–TEU | 3° EPI–TEU | 0° EPI |
| 16° SWEEP .......... | 5° EPI–TEU | 5° EPI–TEU | 0° EPI |
| • FORWARD LIMITS | | | |
| 26° SWEEP........... | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| 16° SWEEP........... | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| • CHECK EPI AT ANGLE-OF-ATTACK... | 10° | 11° | 11° |

- WING CONFIGURATION
  - I – SLATS DOWN, FLAPS GREATER THAN 15° (1-A FLAPS AT 34°; 1-B FLAPS AT 25°)
  - II – SLATS DOWN, FLAPS 15° OR LESS (SLATS ONLY)
  - III – SLATS UP, FLAPS 5° OR LESS (RETRACTED)

NOTES:

1. EXCEPT FOR WING CONFIGURATION 1-A, FORWARD CG LIMIT IS NOT APPLICABLE FOR COMBINATIONS OF FUEL REMAINING/WING SWEEP SHOWN.
2. CHOOSE AFT CG LIMIT CORRESPONDING TO ACTUAL WING CONFIGURATION. FOR WING CONFIGURATION 1-A, USABLE AREA IS BETWEEN FORWARD AND AFT CG LIMIT LINES. FOR ALL OTHER WING CONFIGURATIONS, USABLE AREA IS ANYWHERE ON CHART TO RIGHT OF APPLICABLE AFT CG LIMIT.




1EFA-91-2

Figure 6-13. (Sheet 2)

# WING SWEEP FOR LANDING (TYPICAL)

DATA BASIS: ESTIMATED
DATE: 10 DECEMBER 1982

CONFIGURATION:
(4) MXU-648 CARGO PODS

FUEL GRADE: JP-4
ENGINES: TF30-P-3

CONSIDERATIONS:

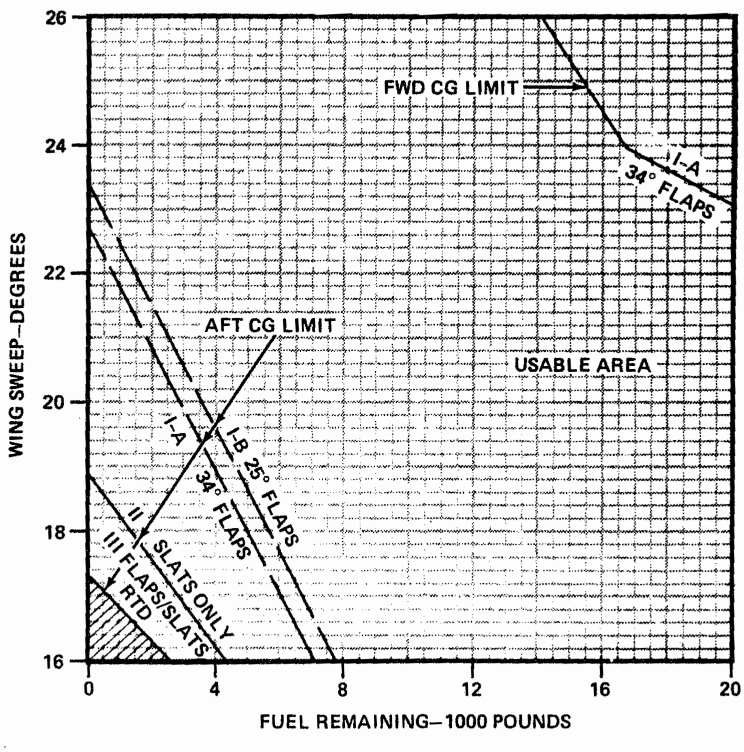
- GEAR EXTENDED
- AUTO FUEL SEQUENCE
- OPERATING WEIGHT CG 47.5% MAC AT 26° WING SWEEP

| | CONFIG I | CONFIG II | CONFIG III |
|---|---|---|---|
| • AFT LIMITS | | | |
| 26° SWEEP.......... | 3° EPI–TEU | 3° EPI–TEU | 0° EPI |
| 16° SWEEP.......... | 5° EPI–TEU | 5° EPI–TEU | 0° EPI |
| • FORWARD LIMITS | | | |
| 26° SWEEP........... | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| 16° SWEEP........... | 10° EPI–TEU | 10° EPI–TEU | 10° EPI–TEU |
| • CHECK EPI AT ANGLE-OF-ATTACK.. | 10° | 11° | 11° |

- WING CONFIGURATION
  - I – SLATS DOWN, FLAPS GREATER THAN 15° (1-A FLAPS AT 34°; 1-B FLAPS AT 25°)
  - II – SLATS DOWN, FLAPS 15° OR LESS (SLATS ONLY)
  - III – SLATS UP, FLAPS 5° OR LESS (RETRACTED)

NOTES:

1. EXCEPT FOR WING CONFIGURATION 1-A, FORWARD CG LIMIT IS NOT APPLICABLE FOR COMBINATIONS OF FUEL REMAINING/WING SWEEP SHOWN.
2. CHOOSE AFT CG LIMIT CORRESPONDING TO ACTUAL WING CONFIGURATION. FOR WING CONFIGURATION 1-A, USABLE AREA IS BETWEEN FORWARD AND AFT CG LIMIT LINES. FOR ALL OTHER WING CONFIGURATIONS, USABLE AREA IS ANYWHERE ON CHART TO RIGHT OF APPLICABLE AFT CG LIMIT.




1EFA-91-3

**Figure 6-13. (Sheet 3)**

**Note**

The elevator position indicator (EPI) should be used to assure that the aircraft is within CG limits for landing.

All computations are based upon aircraft angle-of-attack of 10 degrees for normal flap/slat landing configurations and on 11 degrees angle-of-attack for a malfunctioning flap/slat landing configuration. Enter the chart with the predicted landing fuel and project vertically to the aft center-of-gravity line for the appropriate aircraft configuration, then horizontally to read wing sweep. This is the most forward allowable wing sweep for landing. Continue the vertical projection to the forward center-of-gravity line corresponding to the desired flap setting, then horizontally to wing sweep. This is the most aft allowable wing sweep for landing. Selecting a wing sweep between these two values will provide an elevator trailing edge deflection within center-of-gravity limits for landing.

### ELEVATOR POSITION (EPI) DURING LANDINGS WITH FORWARD CENTER-OF-GRAVITY

The increase of EPI during landing at usual wingsweeps with forward center-of-gravity locations is normal and expected. Corrective action is to sweep the wings forward to bring the EPI within the proper range.

## ENGINE STALL CHARACTERISTICS

Stalls are caused by an aerodynamic disruption of the airflow through 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 stalls have been experienced as a result of flying through another aircraft's exhaust gases/jet wash with little nose-to-tail clearance. The stall may not be noticed due to airframe buffet created by the turbulent airflow.

## ENGINE OPERATING ENVELOPE

In order to minimize engine stalls, Mach-altitude engine operating envelopes are defined in figure 6-14 (Sheet 1) for nonjamming and jamming operations. An angle-of-attack envelope for fixed throttle settings is defined in figure 6-14 (sheet 2). 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 for nonjamming operations and 30,000 feet for jamming operations, the throttle should not be positioned lower than 80 percent RPM for nonjamming operations and 85 percent RPM for jamming operations (approximately halfway from MIL to IDLE). If inlet duct rumble is encountered when selecting 80/85 percent RPM, do not retard the throttle further, but advance as necessary to eliminate the inlet duct rumble. Below 1.5 Mach and 35,000 feet for nonjamming and below 1.5 Mach and 30,000 feet for jamming, throttle position is unrestricted.

In order to minimize engine turbine inlet (TIT) overtemperature, a jamming Mach-altitude avoidance envelope has been included in figure 6-14 (Sheet 1). An increased probability of engine overtemperature results from the high rotor power extraction associated with TJS jamming.

**CAUTION**

Due to the possibility of engine overtemperature, avoid jamming operations in the identified area depicted in figure 6-14. (Sheet 1).

## 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. 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.

# ENGINE OPERATING ENVELOPE

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

FUEL GRADE: JP-4
ENGINES: TF30-P-3

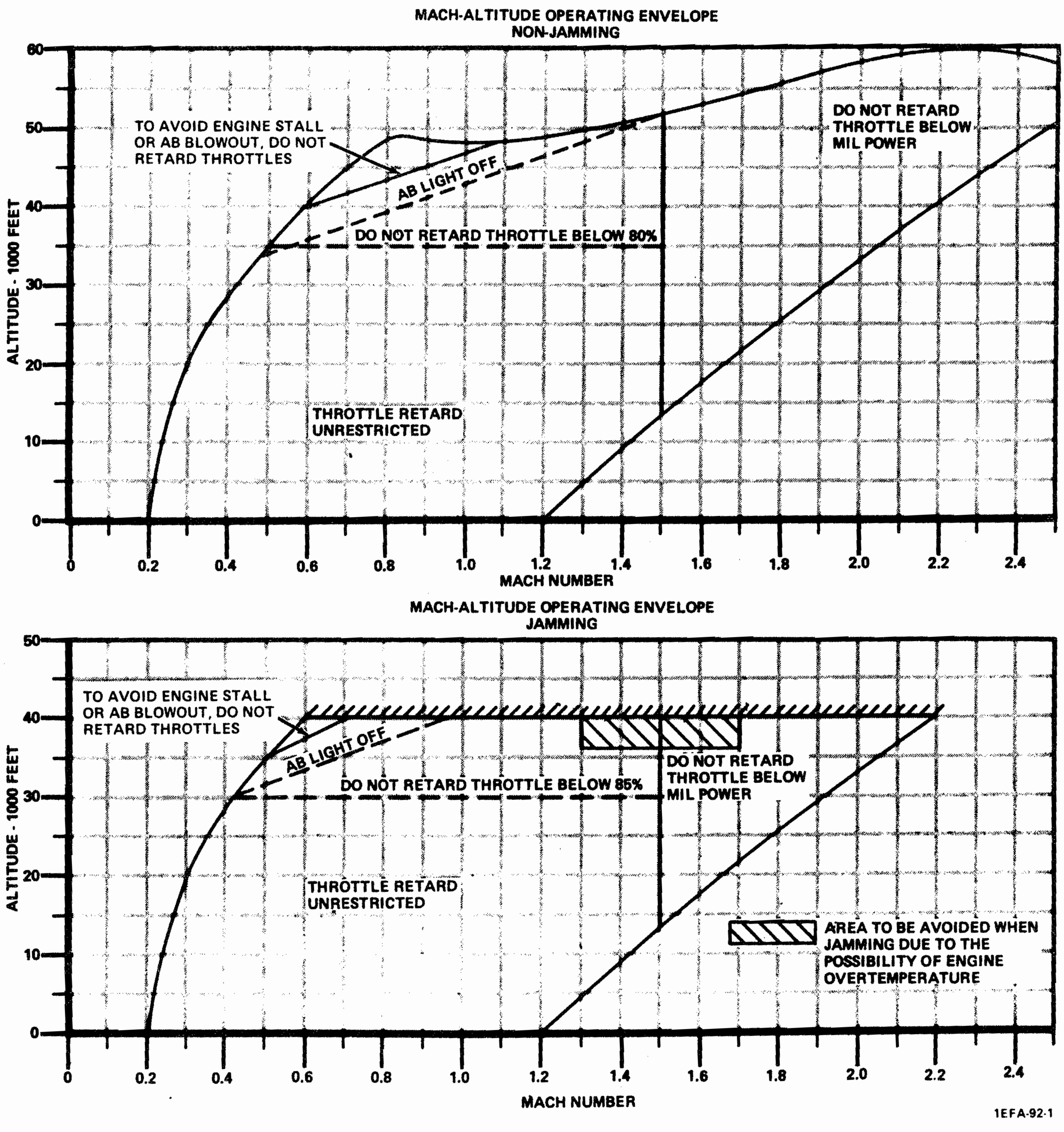



Figure 6-14. (Sheet 1)

# ENGINE OPERATING ENVELOPE

DATA BASIS: ESTIMATED
DATE: 12 JANUARY 1981

FUEL GRADE: JP-4
ENGINES: TF30-P-3

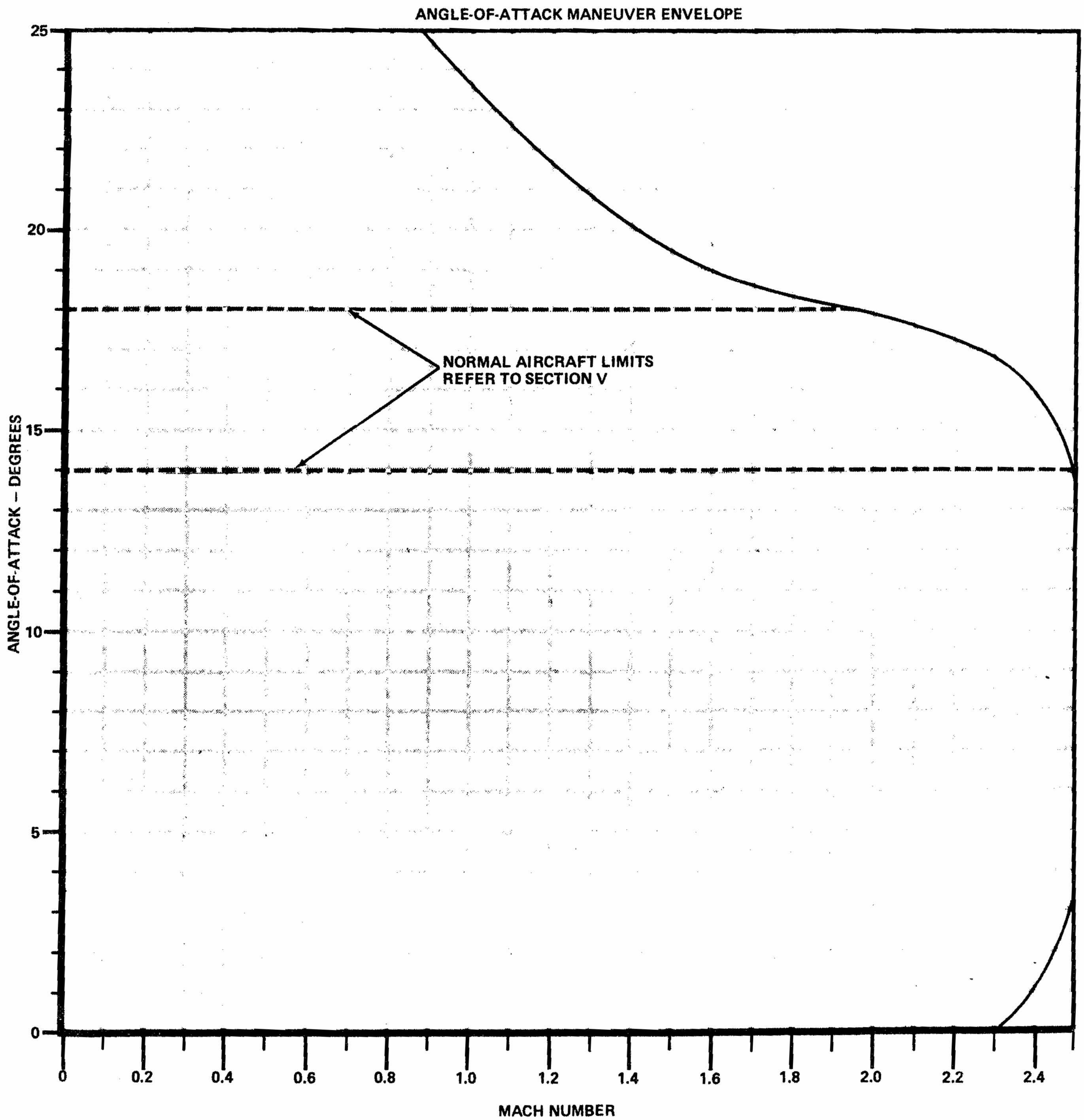



**Figure 6-14. (Sheet 2)**

- 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 indications of a stalled engine are a drop in EPR and a significantly different TIT 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.

**Note**

- Engine stalls may be expected during supersonic flight under uncoordinated flight conditions.

- Stalls have been experienced above 30,000 feet during deceleration from high Mach numbers with the speed brake extended. These stalls do not have the characteristic bang and the pilot may not be aware of the stalled condition. The stall will become apparent when power is advanced and the engine does not respond.

## STALL RECOVERY

For engine stall recovery, refer to "Engine Stall", Section III.

This is the last page of Section VI.

# SECTION VII

# ADVERSE WEATHER OPERATION

TABLE OF CONTENTS



**Note**

In general, this section consists of procedures and information that 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.

### 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.

## 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, or 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

This aircraft is considered a category D aircraft for instrument approaches.

## 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 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. Reduce airspeed to a minimum of computed approach speed plus 20 knots (160 KIAS minimum) until aircraft is rolled out on the final approach course. Prior to final approach fix, establish desired angle-of-attack/airspeed.

## 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 (160 KIAS minimum). Maintain this airspeed until completing of turn onto final approach. After rolling out on final, establish desired angle-of-attack/airspeed. 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

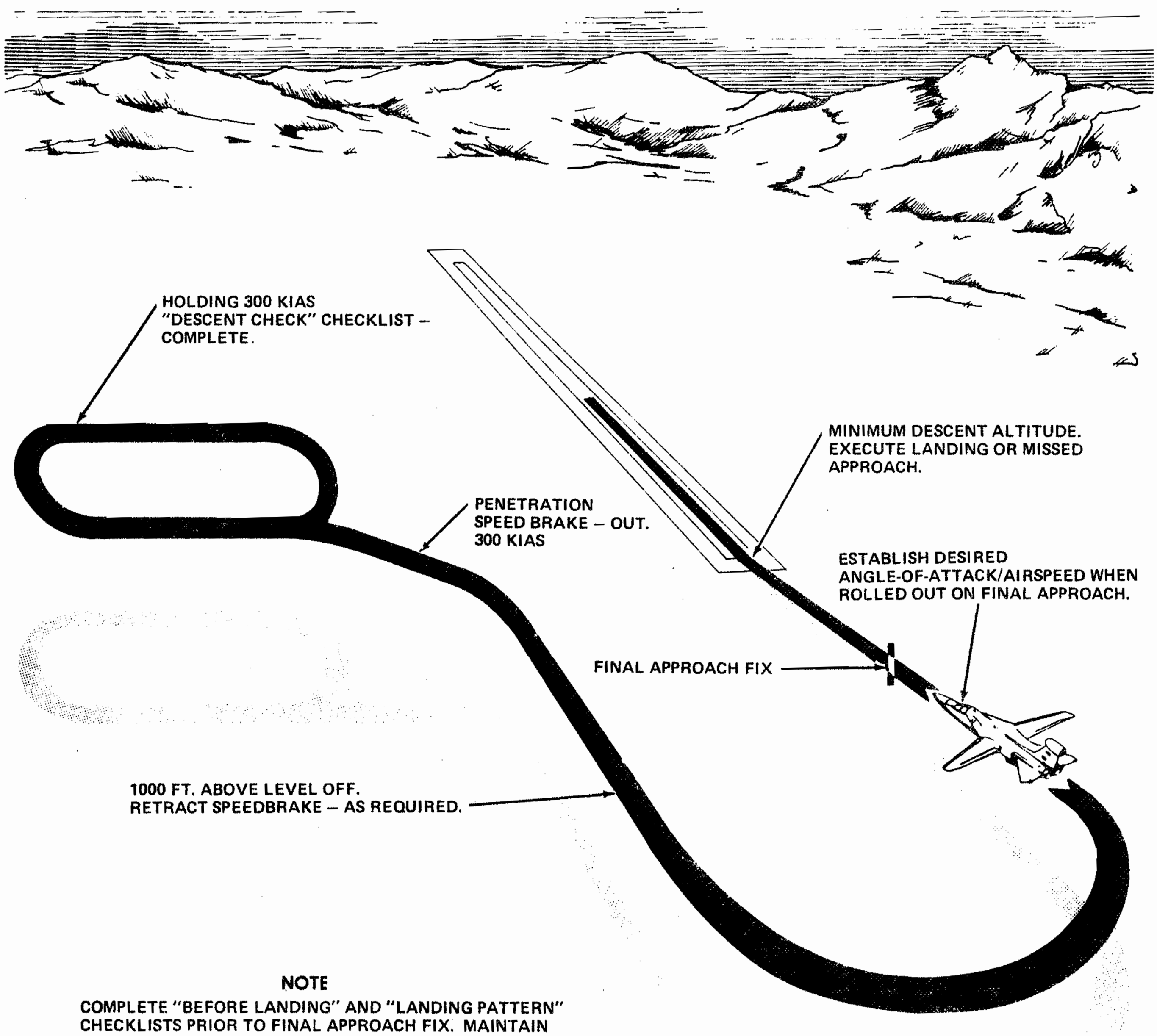
Refer to landing charts in Appendix I for appropriate airspeeds. Figure 7-2 illustrates a typical radar/ILS/AILA approach.

- Magnetic variation - Set.
- ILS power switch - POWER.
- ILS frequency selector knob - Set.

  Set the ILS frequency selector knob to the frequency of localizer to be used for the approach and adjust volume control for identification.

- Instrument system coupler mode selector knob ILS.

# TACAN PENETRATION/APPROACH (TYPICAL)



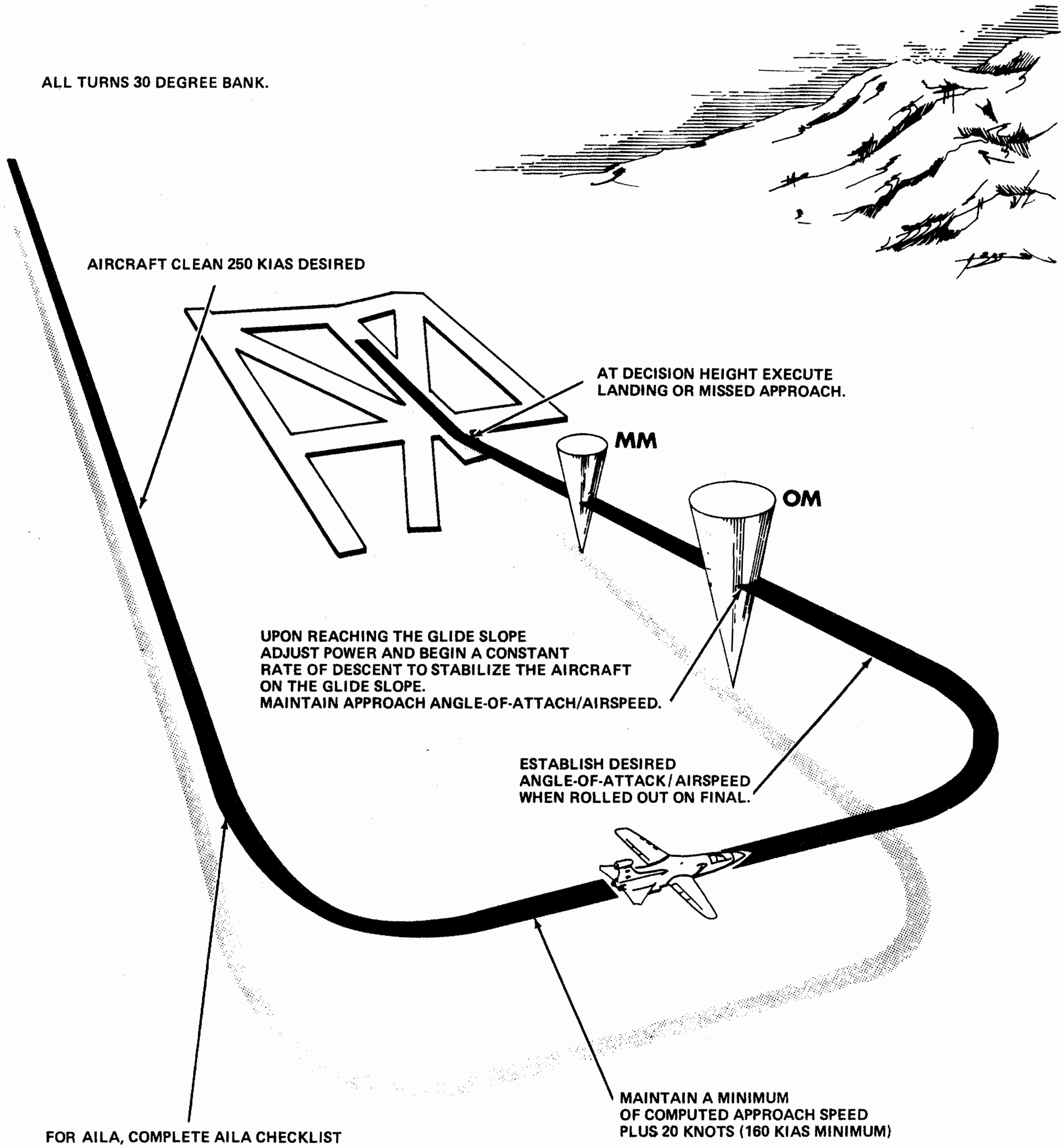

**NOTE**

COMPLETE "BEFORE LANDING" AND "LANDING PATTERN" CHECKLISTS PRIOR TO FINAL APPROACH FIX. MAINTAIN A MINIMUM OF COMPUTED APPROACH SPEED PLUS 20 KNOTS (160 KIAS MINIMUM) UNTIL ROLLED OUT ON FINAL.

1EFA-26

Figure 7-1.

# RADAR/ILS/AILA APPROACH (TYPICAL)




**NOTE**

COMPLETE "BEFORE LANDING" AND "LANDING PATTERN" CHECKLISTS PRIOR TO FINAL APPROACH FIX.

1EFA-27

Figure 7-2.

**WARNING**

Inadvertent selection of AILA will result in erroneous course and glide slope information.

- Horizontal situation indicator - Set.

  Set the inbound localizer course in the course selector window.

- Radar altimeter - Set to minimum altitude AGL.
- Landing Pattern checklist - Complete.
- Localizer course - Intercepted.

  Intercept the localizer, by centering the bank steering bar approximately 10 miles from touchdown.

- Glide slope - Intercepted.

**Note**

When the aircraft penetrates the altitude set on the radar altimeter index pointer, 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 inertial 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 course deviation is displayed on the HSI. Glide path information is also displayed on the ADI on the glideslope deviation indicator. The course deviation displayed on the HSI is a synthetic localizer deviation scaled for $\pm 2.5$ degrees 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 inertial nav computers. Glide path information is provided on the glideslope deviation indicator when the glideslope deviation is less than $\pm 0.9$ 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 inertial nav computer. When the inertial nav system is furnishing simulated localizer, the pitch steering bar will remain in view even when the aircraft deviates more than $\pm 2$ dots from the glideslope or localizer. 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. Minimum altitude must be manually set on the radar altimeter indicator. Figure 7-2 illustrates a typical airborne instrument landing and approach.

1. Inertial mode selector knob - SHORT RANGE.
2. Destination position counters - Set.

   Set destination position counters to touchdown coordinates.

3. TARGET fix mode selector button - Depressed.

**WARNING**

OFFSET and AILA are incompatible switch positions. Glide slope information will be computed to the crosshairs.

4. Altitude calibration - 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. Nav 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. Nav radar range selector - Minimum setting.

8. Glide/Dive angle counter - Set.

   Set the glide/dive angle counter to the approach glide slope.

9. Magnetic variation - Set.

10. "Landing Pattern" checklist - Complete.

11. Instrument system coupler mode selector knob - AILA.

**WARNING**

Inadvertent selection of ILS will result in erroneous course and glide slope information.

12. Horizontal situation indicator - Set.

    Set course selector window to runway heading.

13. Nav radar 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.

14. Runway heading - Intercepted.

    Intercept runway heading by maintaining ADI bank steering bar centered.

15. Glide slope-Intercepted.

    The ADI pitch steering bar will come into view at glide slope interception.

16. ADI pitch steering bar - Centered.

    Establish descent as required to maintain the pitch steering bar centered.

# ICE AND RAIN

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 flight characteristics.

## OPERATION IN RAIN OR ICING CONDITIONS

Icing is possible under the following conditions:

**Ground Operation:**

- In visible moisture - Temperature between +5°C and −15°C (+41°F to +5°F).
- In clear air - Relative humidity above 70 percent and dew point temperature −4°C to +2°C (+25°F to +36°F).

**Inflight Operation:**

- In visible moisture - Total temperature of +5°C (+41°F) or below.
- In clear air - No limitations related to icing.

Aircraft with an operational anti-icing system are cleared for flight in limited icing conditions as follows:

**Note**

- Maintain a minimum of 80 percent rpm until clear of icing conditions.
- In order to minimize ECS water boiling, the use of the anti-ice and rain-removal systems should be kept to a minimum. Operation of both systems simultaneously should be limited to a maximum total of 15 minutes for any one mission.
- For definitions of light, moderate, and heavy icing conditions, refer to Air Weather Service Manual 105–39.

a. 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 necesssary. Takeoffs, descents, penetrations, low approaches and landings, when icing conditions exist, should be planned to minimize the occurrence of ice ingestion and engine stalls.

b. 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 tolerated for perhaps 5 minutes at airspeeds less than 250 KIAS and for longer periods at higher airspeeds as long as no ice is observed accumulating on the aircraft.

c. The following icing conditions should be avoided:

- Heavy icing conditions.

- Icing conditions associated with thunderstorms at airspeeds which produce total temperatures from +5°C to −15°C (+41°F to +5°F).

## GROUND OPERATION

Operate the aircraft and systems as indicated in "Cold Weather Procedures," this section. Rain removal should be used when needed to improve visibility.

**WARNING**

During ground operation in icing conditions, select MAN on the engine/inlet anti-icing switch until after takeoff and clear of icing conditions, then return to AUTO. Have the engine cowls checked for ice build-up prior to takeoff.

**Note**

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 addtition, 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. Apply brakes and advance throttles for takeoff. If aircraft starts to slide on ice or snow before full power is reached, release brakes and begin takeoff run. Continue advancing power during takeoff run and check engine instruments for proper operation. 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 or slush ingestion if takeoff is attempted with accumulated water or slush on the runway.

**CAUTION**

Takeoffs with accumulated slush on the runway should be avoided as the nose gear tires may deflect slush into the engine intakes resulting in the formation of ice and ensuing engine damage.

## 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. Do not operate in rain, sleet, or hail longer than absolutely necessary. If it becomes necessary to fly in these conditions, constantly check the aircraft leading edges, including radome, for indications of peeling or other structural deterioration of the aircraft surfaces. In the event deterioration of surfaces is observed, maintain airspeed as low as practicable and land at the nearest suitable airfield as soon as possible. If heavy precipitation conditions of the above type are encountered at any speed or light to moderate conditions exist at high airspeeds, an entry must be made in Form 781.

**CAUTION**

To minimize impact damage from rain or sleet, do not exceed 450 KTAS.

## LANDING

- At 10 degrees angle-of-attack, indicated airspeed may be higher than computed final approach airspeed due to structural icing.

- Loss of Visibility-Reduced visibility may accompany a power decrease. This is due to a loss of air pressure to the windshield rain removal system, which is directly related to the engine rpm. Therefore, by selecting (L) left or (R) right rain removal, visibility will be improved.

CAUTION

- Landings with accumulated slush on the runway should be avoided as the nose gear tires may deflect slush into the engine intakes resulting in the formation of ice and ensuing engine damage.

- Hydroplaning may cause the failure detect feature of the anti-skid system to turn the anti-skid system off, thus lighting the anti-skid caution lamp. When this occurs, the system reverts to manual braking and tires may be inadvertently blown even though only slight brake pressure is applied.

Note

If the tail hook is extended during operations in icy, snow or slush conditions, the HOOK DOWN lamp may not light due to a frozen limit switch.

# TURBULENCE, THUNDERSTORMS, AND WIND SHEAR

## 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 nav 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. Windshield 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 engine stall and/or airstart procedures contained in Section III.

- When using terrain following radar, the back scatter from drizzle or rain and other forms or 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 angle-of-attack with resultant marginal engine performance.

- Turbulence, gusty wind or wind shear conditions may induce variations in angle-of-attack or airspeed and may 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 touchdown characteristics and susceptibility to a pilot induced oscillation, this additional airspeed should be dissipated so that an on-speed indication exists prior to touchdown.

## LIGHTNING

Lightning strikes are more prevalent within 1000 feet of the freezing level and when flying through precipitation, or in the vicinity of thunderstorms. Most strikes occur in the area of the radome and pitot boom. They are characterized by a loud bang, high intensity light causing temporary loss of night vision, and a sensation of electrical discharge within the body. These factors, coupled with sporadic cockpit caution/warning lamps and possible activation of the stall warning system may cause extreme confusion and disorientation. Flight instruments, particularly heading indicators, may be unreliable. Electronic components, including the radios, nav radar, and inertial nav system may be degraded or inoperative. Sporadic, uncommanded, possibly violent flight control movements may be experienced immediately or after an indeterminate delay. Amplitude and frequency of movements may vary.

**WARNING**

Aircraft receiving lightning strikes may experience erroneous electrical inputs affecting electrical supply, CADC, flight control computer system, and aircraft instruments and displays. In addition arcing, fusing of circuit breakers, fire and explosion may occur. Refer to "Lightning Strike/Static Discharge," Section III.

## WIND SHEAR

Wind shear phenomena can affect an aircraft in all phases of flight, however, failure to recognize and cope with wind shear during the approach phase may result in an unexpected long or short landing. Wind shear is best described as a change in wind direction and/or speed over a short distance along the flight path. These changes can occur at any altitude, but are most hazardous during takeoff, initial climbout, approach, and landing. Low level wind shear is frequently associated with thunderstorms, fronts, high winds and effects of temperature changes. (For a more complete discussion of the weather phenomena which cause wind shear, refer to AFM 51-12, Volume I.) Preflight weather briefings should include a discussion of thunderstorm and frontal activity near the departure and destination airports to determine if the potential for wind shear exists. Pilot reports are also a valuable aid in making this determination.

## DEFINITIONS

### Minimum Groundspeed (VMGS)

VMGS is the expected groundspeed on final approach. It is computed by adding the surface tailwind component to, or subtracting the surface headwind component from, approach true airspeed.

### Gain Shear

A rapid increase in runway headwind component, or a rapid decrease in tailwind component, resulting in a rapid gain in indicated airspeed, a significant gain in lift and vertical velocity, and an increase in altitude.

### Loss Shear

A rapid decrease in runway headwind component, or a rapid increase in the runway tailwind component, resulting in a rapid loss in indicated airspeed, a significant loss of lift, and vertical velocity and a decrease in altitude.

## TAKEOFF PROCEDURES

### Loss Shear

When experiencing a loss shear on takeoff, anticipate a loss in indicated airspeed and altitude. This situation is potentially dangerous if the aircraft is close to the ground, during reduced thrust from simulated/actual single engine or heavy weight, mil power climbouts. An excessive loss could cause angle-of-attck build up and/or stall. Obstacle clearance may also be critical.

## APPROACH AND LANDING PROCEDURES

The INS should be used in determining the actual groundspeed of the aircraft. When the aircraft approaches the initial approach fix or is established on a portion of the approach below 3000 feet AGL, a comparison of the actual runway wind with the INS wind will indicate whether or not a wind shear is possibly present. A second wind comparison, at the final approach fix, can more accurately determine whether a shear can be expected and the extent of the shear. VMGS should be compared to INS groundspeed. A difference of 15 knots or more indicates the presence of significant wind shear and wind shear procedures should be used. A large difference in wind direction, and/or wind speed most likely indicates the presence of a shear.

**WARNING**

> If a wind shear is encountered on final approach, do not delay a go-around if the proper approach profile cannot be promptly reestablished. Delaying go-around may result in unexpected short or long landing.

If a wind shear is anticipated during approach and landing, use the following procedures.

### Gain Shear

Anticipate an increase in indicated airspeed and altitude. Airspeed flown on final approach should be normal approach speed. When experiencing the shear, the indicated airspeed will increase and the aircraft will balloon above the glidepath. Reduce power and lower the nose to regain the normal glidepath. Power setting to maintain a normal approach glidepath will be higher than was used prior to experiencing the shear. There is a tendency for the aircraft to descend below the normal glidepath while the shear correction is being made, which could result in early landing or possible ground impact, depending on shear level altitude above ground, if power corrections or pitch changes are not made promptly and correctly.

### Loss Shear

With a loss shear, anticipate a loss in indicated airspeed and altitude. The aircraft will drop below the normal glidepath, and airspeed will decrease below normal approach speed, if on speed when the shear occurs. To recover, power will have to be increased and a climb back to normal glidepath accomplished. Once the normal glidepath and airspeed are regained, power will have to be lower than the original setting prior to the shear level. When regaining the normal glidepath, there will be a tendency for the aircraft to balloon above the glidepath and possibly land long.

**Note**

> Wind shear conditions may induce variations in angle-of-attack or airspeed, and may 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. This additional airspeed should be dissipated so that an on-speed indication exists prior to touchdown.

## UNANTICIPATED WIND SHEAR

Normally, wind shear can be predicted and procedures followed which will ensure a safe takeoff, climb out, approach and landing. However, the possibility of experiencing unanticipated severe low level wind shear does exist. Pilots must be able to recognize the characteristics of gain/loss shear and apply prompt recovery procedures. Unlike recovery procedures at high altitudes, lowering the nose and trading altitude for airspeed may result in ground/obstacle impact. Low level shear recognition must be made as soon as possible since recovery may require the full aerodynamic capabilities of the aircraft. Indicators of a condition that may require immediate application of recovery procedures include:

- A rapid decrease in airspeed below approach or climbout speeds.
- A rapid decrease in climb rate during takeoff or go-around.
- A rapid increase in sink rate during approach.
- An increase in angle-of-attack indication approaching stall warning.
- Monitor VASI (visual approach slope indicator), ILS glideslope indications, and runway visual aimpoint for pilot awareness of deviations. Know pitch attitude and thrust levels required for approaches.

# COLD WEATHER PROCEDURES

Most cold weather operating difficulties are encountered on the ground due to cold soaking and ice accumulations.

The following instructions are to be used in conjunction with Normal Procedures, Section II, when cold weather operation is necessary.

## BEFORE ENTERING AIRCRAFT

Remove all protective covers and duct plugs; check to see that all surfaces, ducts, struts, drains and vents are free of snow, ice and heavy 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 shrinking of packings. Inspect areas behind the aircraft to ensure that water or snow will not be blown onto personnel and equipment during engine start. After cold soaking at temperatures below 0°F, the cockpit, the main wheel well and the forward wheel well should be preheated by portable heaters for 15 minutes with 80° to 120°F warm air just prior to engine start. This will aid actuators and switches to function freely.

**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-111(E)A-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 aircraft stall speeds, and cause engine foreign object damage.
- 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. R

**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-111(E)A-2-1.

**Note**

- Moisture may accumulate in the low point of the engine intakes even with the intake covers installed.
- Extreme cold temperature may require cockpit preheating to ease operation of rotary type switches.

## STARTING ENGINES

Use normal procedures for starting engines.

WARNING

During ground operations in icing conditions, select MAN on the engine/inlet anti-icing switch until after takeoff and clear of icing conditions, then return to AUTO. Have the engine cowls checked to ensure no ice build-up prior to takeoff.

CAUTION

To prevent the aircraft from sliding on icy surfaces or blowing ice and snow during engine start, it may be necessary to utilize an external power source for starting both engines.

Note

At ambient temperatures below 0°F, unless fast warmup of the hydraulic systems is required, do not attempt hydraulic actuator movement until three minutes after engine start or until the hydraulic fluid, monitored at hydraulic reservoirs, has warmed up to 0°F, whichever is later. Flight control checks should be delayed until 10 minutes after engine start.

Cartridge starts using JP–5 or JP–8 fuel are unreliable at temperatures below −10°C, and using JP–4 below −20°C. Pneumatic starts at these colder temperatures may require a longer time.

## 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. Prior to taxiing, depress and release brake pedals to insure freedom of operation in both directions.

## 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. Ensure that all instruments have warmed sufficiently to insure operation. Check for sluggish instruments during taxiing.
- After cold soaking at temperatures below 0°F, be alert for flat main landing gear struts when taxiing and during engine run-up. Ensure that struts are fully serviced prior to takeoff.

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

Ensure that takeoff data accounts for reduced braking capability due to ice and snow on runways in event of an abort. Apply brakes and advance throttles for takeoff. If aircraft starts to slide on ice or snow before full power is reached, release brakes and begin takeoff run. Continue advancing power during takeoff run and check engine instruments for proper operation. Care should be exercised to avoid exceeding climb schedule speeds due to additional thrust available at low temperatures.

CAUTION

If the ICING caution lamp lights during engine runup, do not take off until the lamp goes out.

Note

With the landing gear handle up and the landing gear handle warning lamp out, a nose gear position indicator lamp that remains on may be caused by frozen moisture in the gear position sensing switch. The mission may be continued if a visual check confirms the nose gear is retracted.

## LANDING

After the aircraft is in the landing configuration, depress and release the brake pedals to insure freedom of operation in both directions.

## 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 GROUND 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, hydrualic, pitot-static, etc.), which are susceptible to contamination, malfunction, or damage from sand and dust. 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. 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. Leave the canopy open if the aircraft is to be exposed to direct sunlight at ambient temperatures above 90 degrees F. This will prevent excessive heat buildup within the cockpit.

**CAUTION**

- Do not attempt takeoff or engine operation in a sand storm or dust storm if avoidable.
- To minimize canopy and windshield damage, consideration should be given to parking the aircraft tail into the wind.

This is the last page of Section VII.

**APPENDIX I**

## PERFORMANCE DATA

Performance Data is contained in Appendix I of Supplement T.O. 1F-111(E)A-1-1.

# ALPHABETICAL INDEX



***Denotes Illustration**

# ALPHABETICAL INDEX (cont)



## B



***Denotes Illustration**

# ALPHABETICAL INDEX (cont)



***Denotes Illustration**

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## ALPHABETICAL INDEX (cont)



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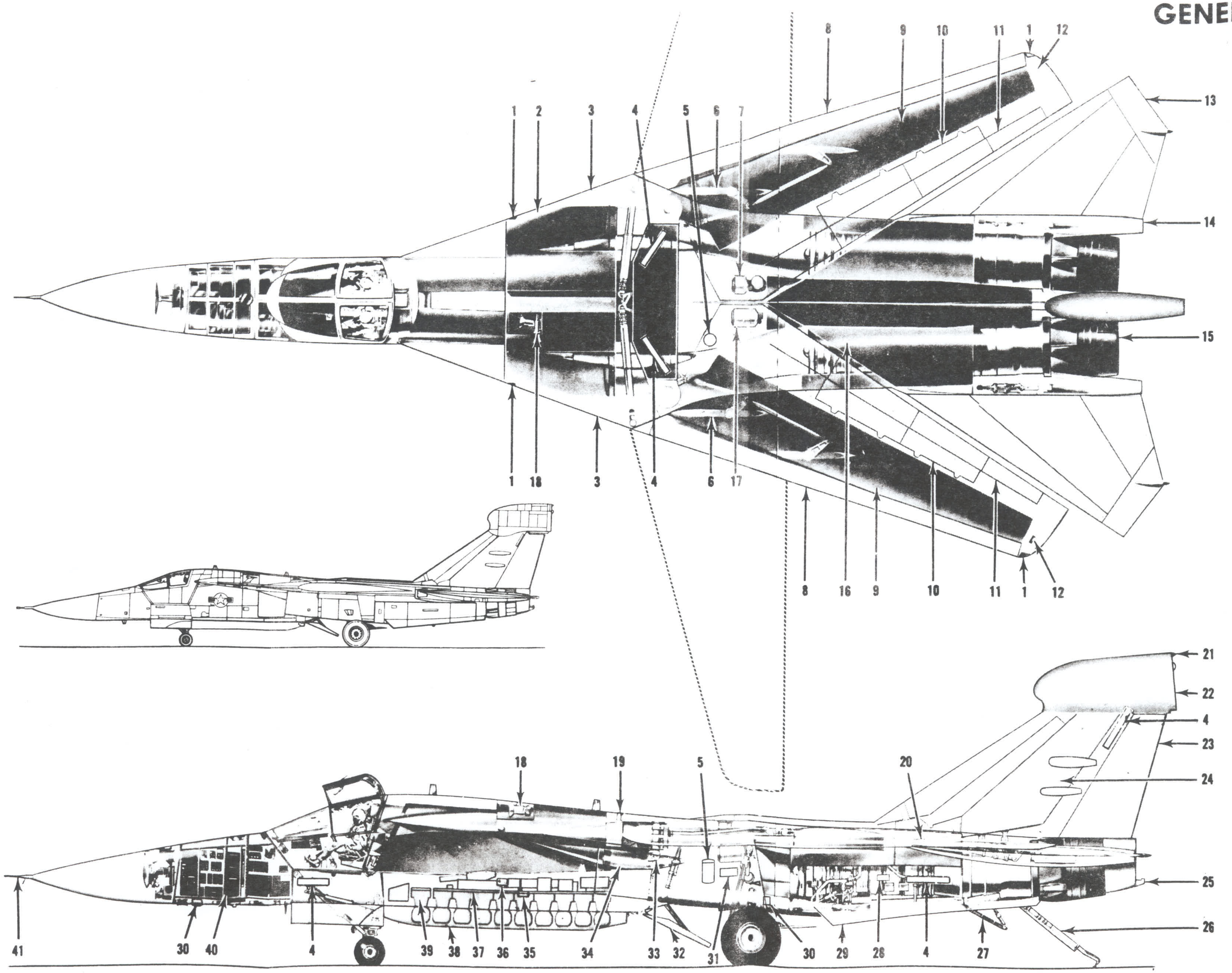
# FOLDOUTS

## TABLE OF CONTENTS

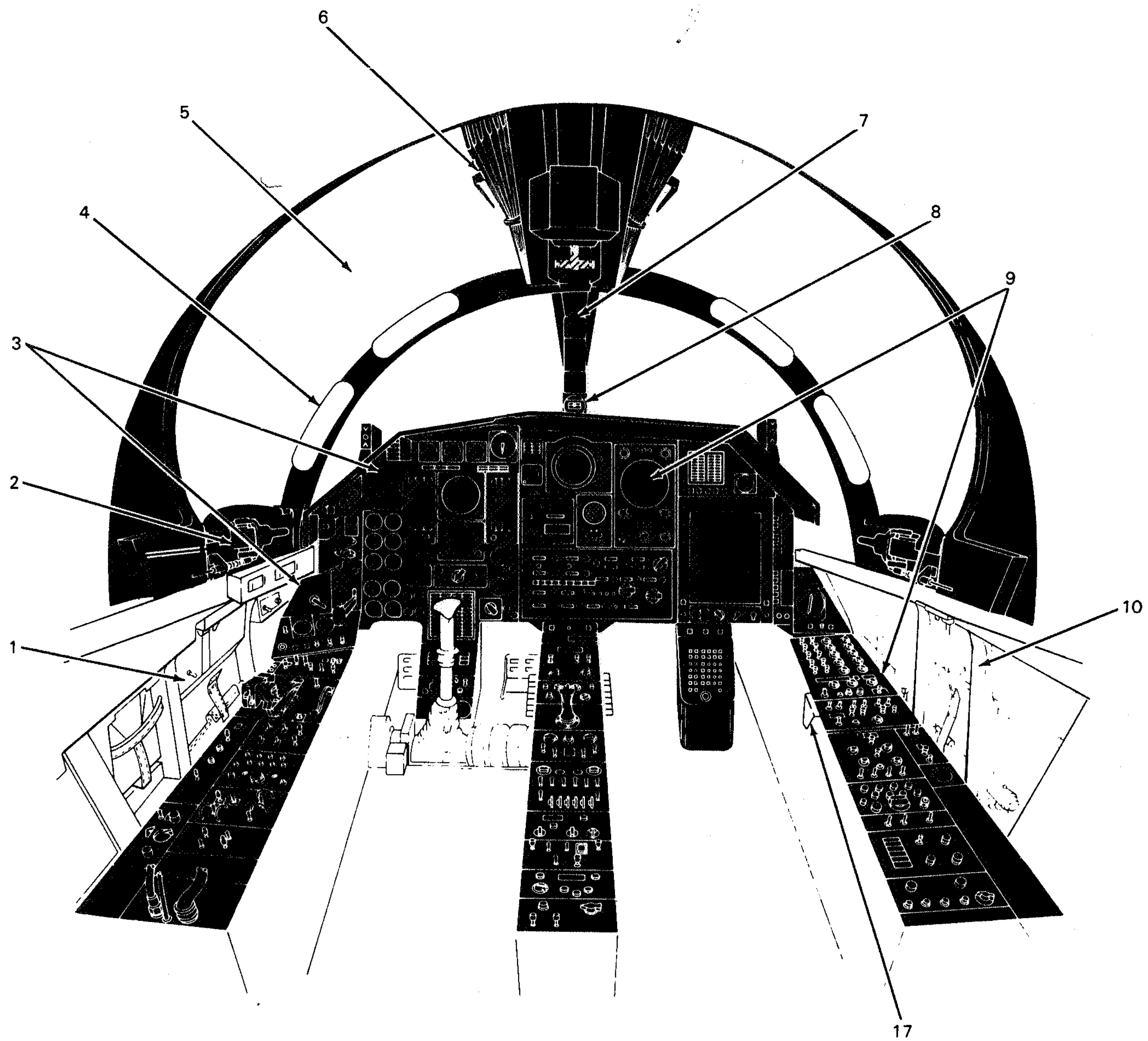
# GENERAL ARRANGEMENT DIAGRAM

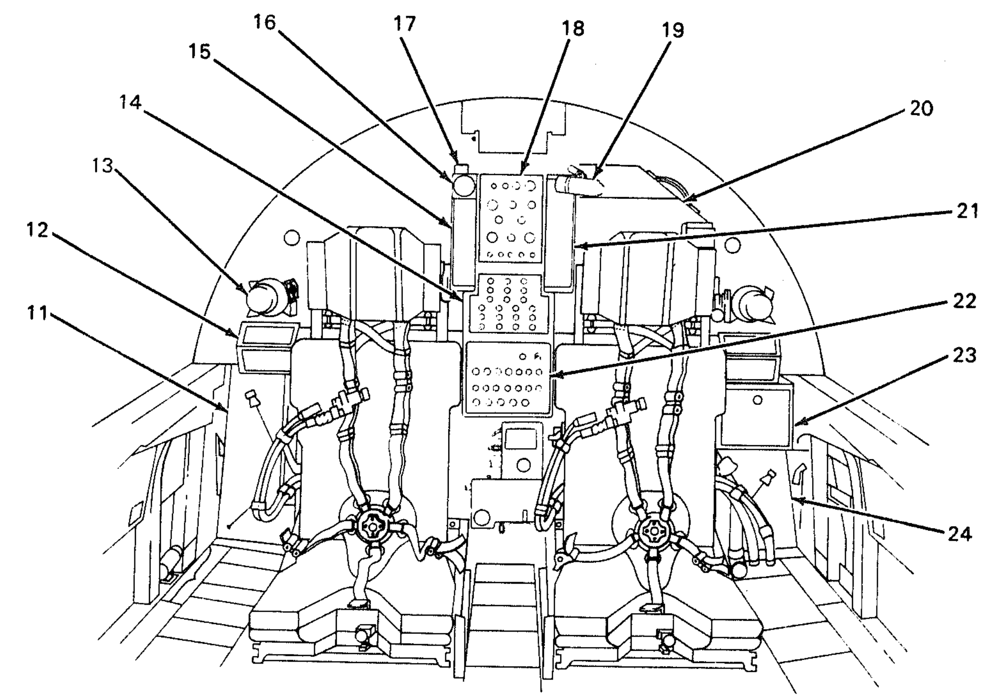



1. WING POSITION LIGHTS
2. FORWARD FUEL TANK
3. ROTATING GLOVE
4. STRIP LIGHTS
5. SPS LIQUID COOLANT SYSTEM RESERVOIR
6. PIVOTING PYLONS
7. PRIMARY HYDRAULIC SYSTEM RESERVOIR
8. SLATS
9. WING FUEL TANKS
10. SPOILERS
11. WING FLAPS
12. WING FORMATION LIGHTS (UPR & LWR)
13. HORIZONTAL STABILIZER
14. SPEED BUMPS
15. ENGINES
16. AFT FUEL TANK
17. UTILITY HYDRAULIC SYSTEM RESERVOIR
18. AIR REFUELING RECEPTACLE
19. ANTI-COLLISION LIGHT (UPPER)
20. RAM AIR HEAT EXCHANGER (2) (BOTH SIDES)
21. TAIL POSITION LIGHT
22. FIN RADOME
23. RUDDER
24. FUEL VENT TANK
25. FUEL DUMP/VENT OUTLET
26. ARRESTING HOOK
27. TAIL BUMPER
28. FUSELAGE FORMATION LIGHTS (4)
29. STRAKE (2)
30. ANTI-COLLISION LIGHT (2 LOWER)
31. JSS COOLANT ACCUMULATOR
32. FORWARD LANDING GEAR DOOR/ SPEED BRAKE
33. AIR CONDITIONING SYSTEM COOLING AIR INTAKE
34. SPIKE
35. FUEL SYSTEM PRECHECK SELECTOR PANEL
36. SINGLE POINT REFUELING ADAPTER RECEPTACLE
37. AFT ELECTRONIC EQUIPMENT BAY
38. WEAPONS BAY RADOME
39. JSS TRANSMITTERS
40. FORWARD ELECTRONIC EQUIPMENT BAY
41. PITOT STATIC PROBE

Figure FO–1.

# CREW STATION GENERAL ARRANGEMENT




1. LEFT SIDEWALL (SEE FIG. 1-11)
2. INTERNAL CANOPY LATCH HANDLES (2)
3. CREW STATION (PILOT) (SEE FIG. FO-3)
4. MIRRORS (4)
5. CANOPY
6. THERMAL CURTAIN (2)
7. CANOPY CENTER BEAM ASSEMBLY
8. MAGNETIC COMPASS
9. CREW STATION (EWO) (SEE FIG. FO-4)
10. RIGHT SIDEWALL (SEE FIG. 1-23)
11. HOOD STOWAGE COMPARTMENT
12. AIR DIFFUSERS (2)
13. LIQUID CONTAINERS (2)
14. CIRCUIT BREAKER PANEL (SEE FIG. 1-7)
15. LETDOWN CHART STOWAGE COMPARTMENT
16. TOTAL TEMPERATURE INDICATOR
17. INTERNAL LIGHTING CIRCUIT BREAKER PANELS (SEE FIG 1-22)
18. LIGHTING CONTROL PANEL (SEE FIG. 1-21)
19. UTILITY LIGHT
20. QUICK RESCUE KIT
21. APPROACH CHART HOLDER STOWAGE COMBINATION
22. GROUND CHECK PANEL (SEE FIG. 1-17)
23. FOOD STOWAGE COMPARTMENT
24. RELIEF CONTAINER STOWAGE COMPARTMENT (2)
25. OXYGEN SUIT CONTROL PANEL AND DILUTER (SEE FIG. 1-28)

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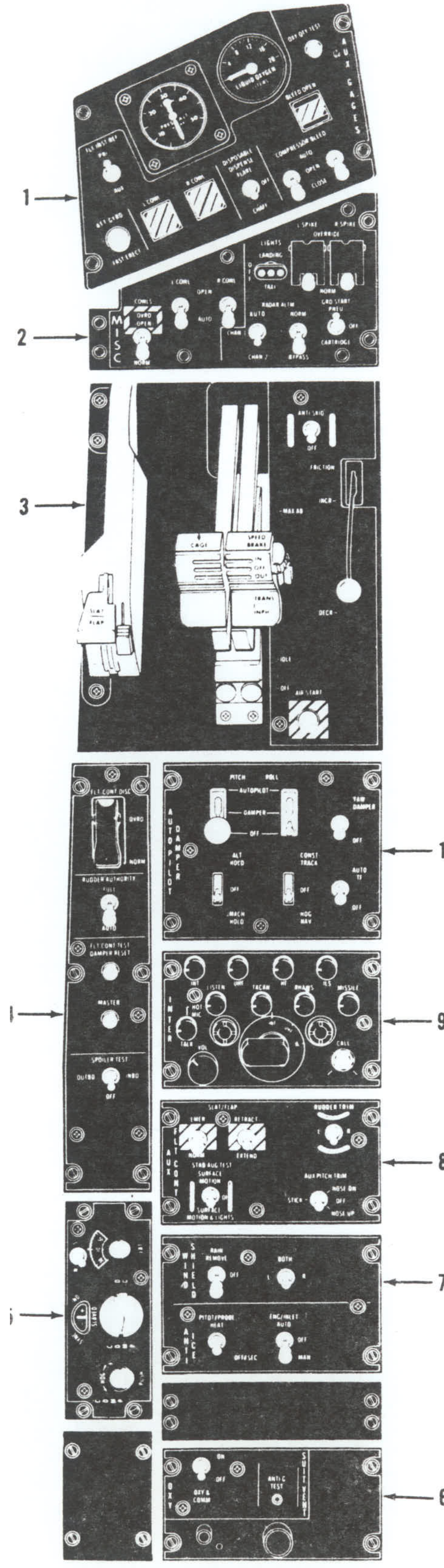
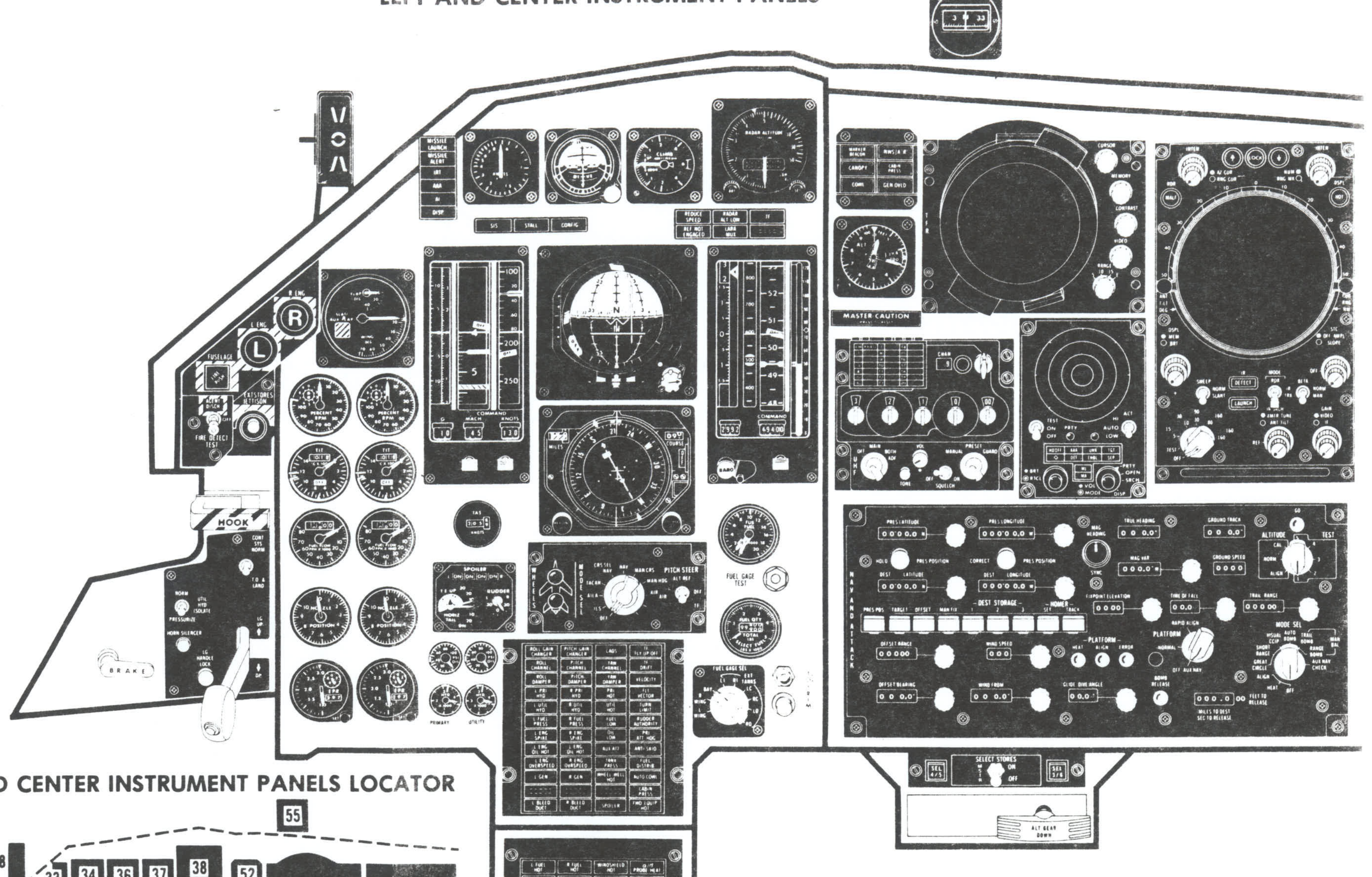
Figure FO–2.

# CREW STATION (PILOT)

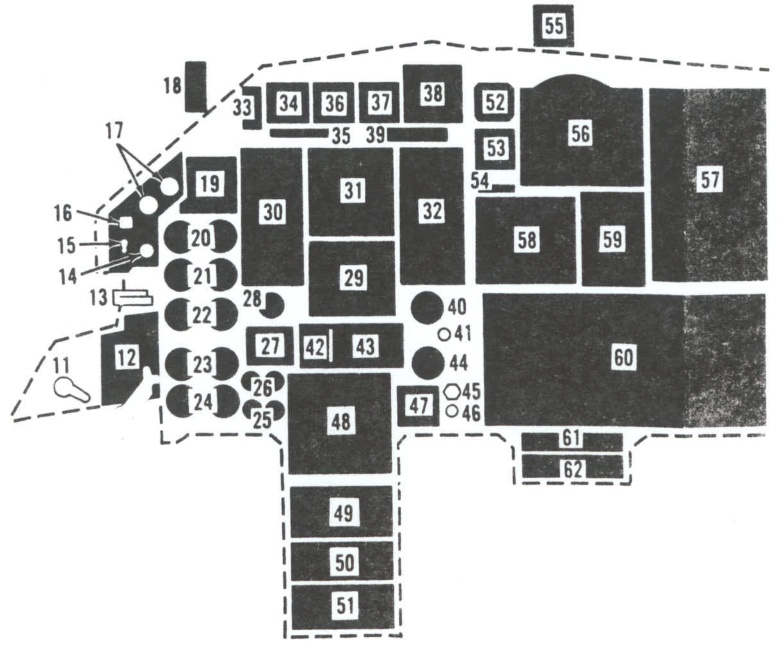
LEFT CONSOLE



LEFT AND CENTER INSTRUMENT PANELS



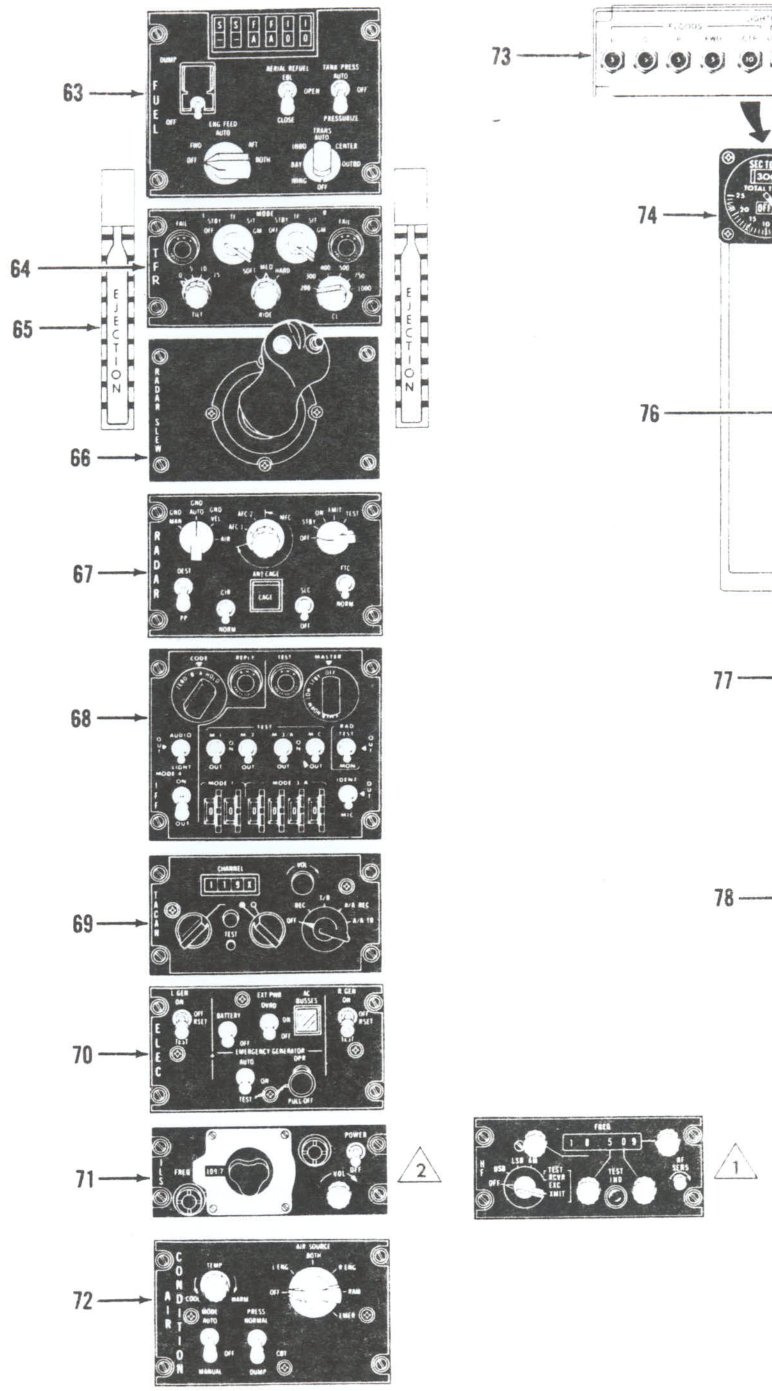
LEFT AND CENTER INSTRUMENT PANELS LOCATOR



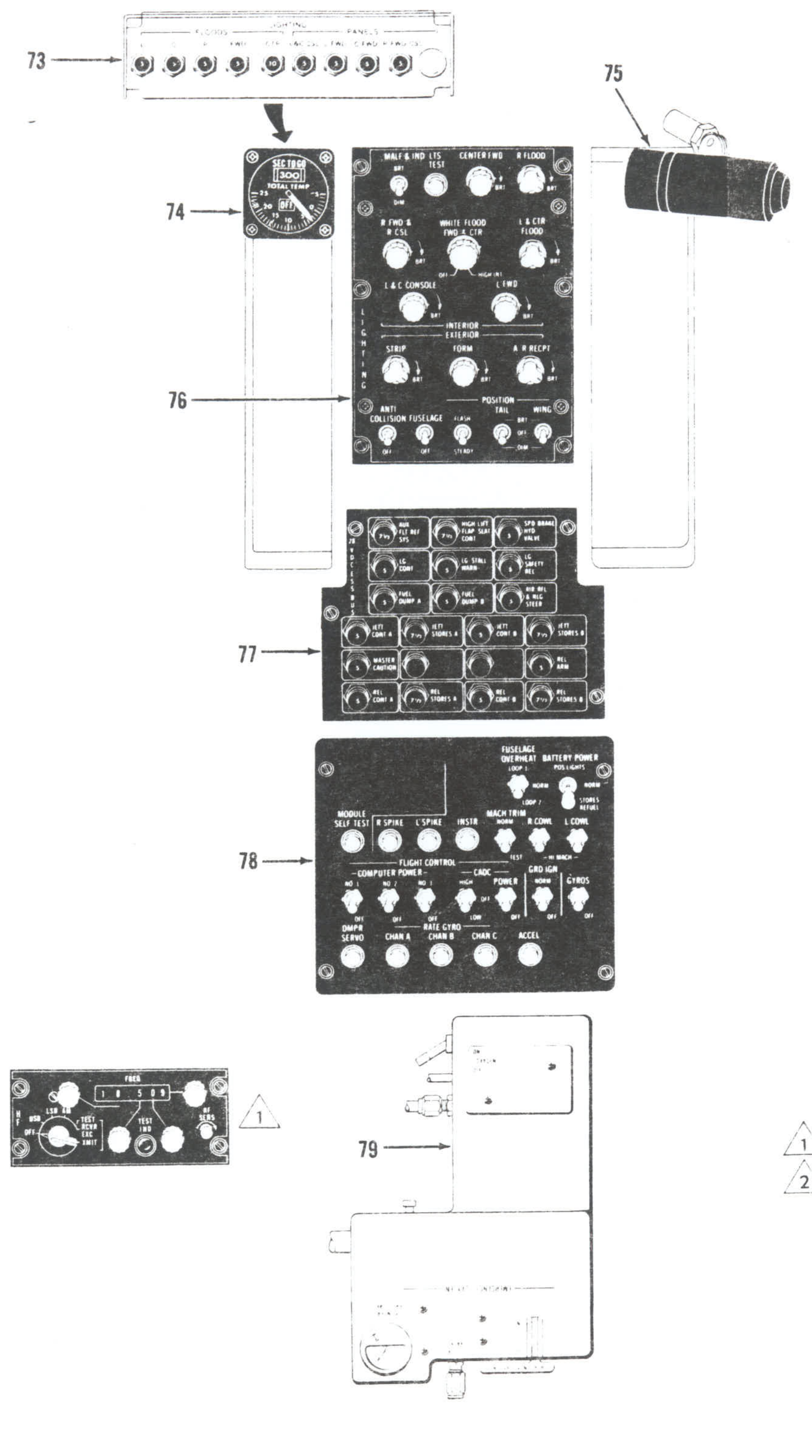
1 PRIOR TO T.O. 1F-111(E)A-501

2 AFTER T.O. 1F-111(E)A-501

CENTER CONSOLE



AFT BULKHEAD



LEFT CONSOLE

1. AUXILIARY GAGE PANEL
2. MISCELLANEOUS SWITCH PANEL
3. THROTTLE PANEL
4. FLIGHT CONTROL TEST PANEL
5. COMPASS CONTROL PANEL
6. OXYGEN – SUIT CONTROL PANEL
7. ANTI-ICING/WINDSHIELD RAIN REMOVAL CONTROL PANEL
8. AUXILIARY FLIGHT CONTROL PANEL
9. INTERPHONE CONTROL PANEL
10. AUTOPILOT DAMPER PANEL

LEFT AND CENTER INSTRUMENT PANELS

11. AUXILIARY BRAKE HANDLE
12. LANDING GEAR CONTROL PANEL
13. ARRESTING HOOK HANDLE
14. EXTERNAL STORES JETTISON BUTTON
15. AGENT DISCHARGE/FIRE DETECT TEST SWITCH
16. FUSELAGE FIRE PUSHBUTTON WARNING LAMP
17. ENGINE FIRE PUSHBUTTON WARNING LAMPS
18. ANGLE-OF-ATTACK INDEXER
19. WING SWEEP FLAP/SLAT POSITION INDICATOR
20. ENGINE TACHOMETERS
21. ENGINE TURBINE INLET TEMPERATURE INDICATORS
22. ENGINE FUEL FLOW INDICATORS
23. ENGINE NOZZLE POSITION INDICATORS
24. ENGINE PRESSURE RATIO INDICATORS
25. HYDRAULIC PRESSURE INDICATORS
26. ENGINE OIL PRESSURE INDICATORS
27. CONTROL SURFACE POSITION INDICATOR
28. TRUE AIRSPEED INDICATOR
29. HORIZONTAL SITUATION INDICATOR
30. AIRSPEED-MACH INDICATOR
31. ATTITUDE DIRECTOR INDICATOR
32. ALTITUDE-VERTICAL VELOCITY INDICATOR
33. THREAT INDICATOR
34. STANDBY AIRSPEED INDICATOR
35. LEFT STATUS INDICATOR
36. SELF CONTAINED ATTITUDE INDICATOR
37. STANDBY VERTICAL VELOCITY INDICATOR
38. RADAR ALTIMETER INDICATOR
39. RIGHT STATUS INDICATOR
40. FUSELAGE FUEL QUANTITY INDICATOR
41. FUEL QUANTITY INDICATOR TEST BUTTON
42. LANDING GEAR POSITION INDICATOR LAMPS
43. INSTRUMENT SYSTEM COUPLER CONTROL PANEL
44. TOTAL/SELECT FUEL QUANTITY INDICATOR
45. TAKEOFF TRIM INDICATOR LAMP
46. TAKEOFF TRIM BUTTON
47. FUEL QUANTITY INDICATOR SELECT KNOB
48. MAIN CAUTION LAMP PANEL
49. LOWER CAUTION LAMP PANEL
50. 1 ILS CONTROL PANEL
    2 CMRS CONTROL PANEL
51. RAM DOORS/OIL QUANTITY PANEL
52. ANNUNCIATOR INDICATOR
53. STANDBY ALTIMETER
54. MASTER CAUTION LAMP
55. MAGNETIC COMPASS
56. TFR SCOPE PANEL
57. NAV RADAR SCOPE PANEL
58. UHF RADIO CONTROL PANEL
59. TTWS CONTROL INDICATOR PANEL
60. INERTIAL NAV CONTROL PANEL
61. STORES JETTISON SELECT PANEL
62. LANDING GEAR EMERGENCY RELEASE HANDLE

CENTER CONSOLE

63. FUEL CONTROL PANEL
64. TFR CONTROL PANEL
65. EJECTION HANDLES (2)
66. NAV RADAR SLEW PANEL
67. NAV RADAR CONTROL PANEL
68. IFF CONTROL PANEL
69. TACAN CONTROL PANEL
70. ELECTRICAL CONTROL PANEL
71. 1 HF CONTROL PANEL
    2 ILS CONTROL PANEL
72. AIR CONDITIONING CONTROL PANEL

AFT BULKHEAD

73. INTERNAL LIGHTING CIRCUIT BREAKER PANEL
74. TOTAL TEMPERATURE INDICATOR
75. UTILITY LIGHT
76. LIGHTING CONTROL PANEL
77. DC CIRCUIT BREAKER PANEL
78. GROUND CHECK PANEL
79. OXYGEN-SUIT CONTROL PANEL (EWO)

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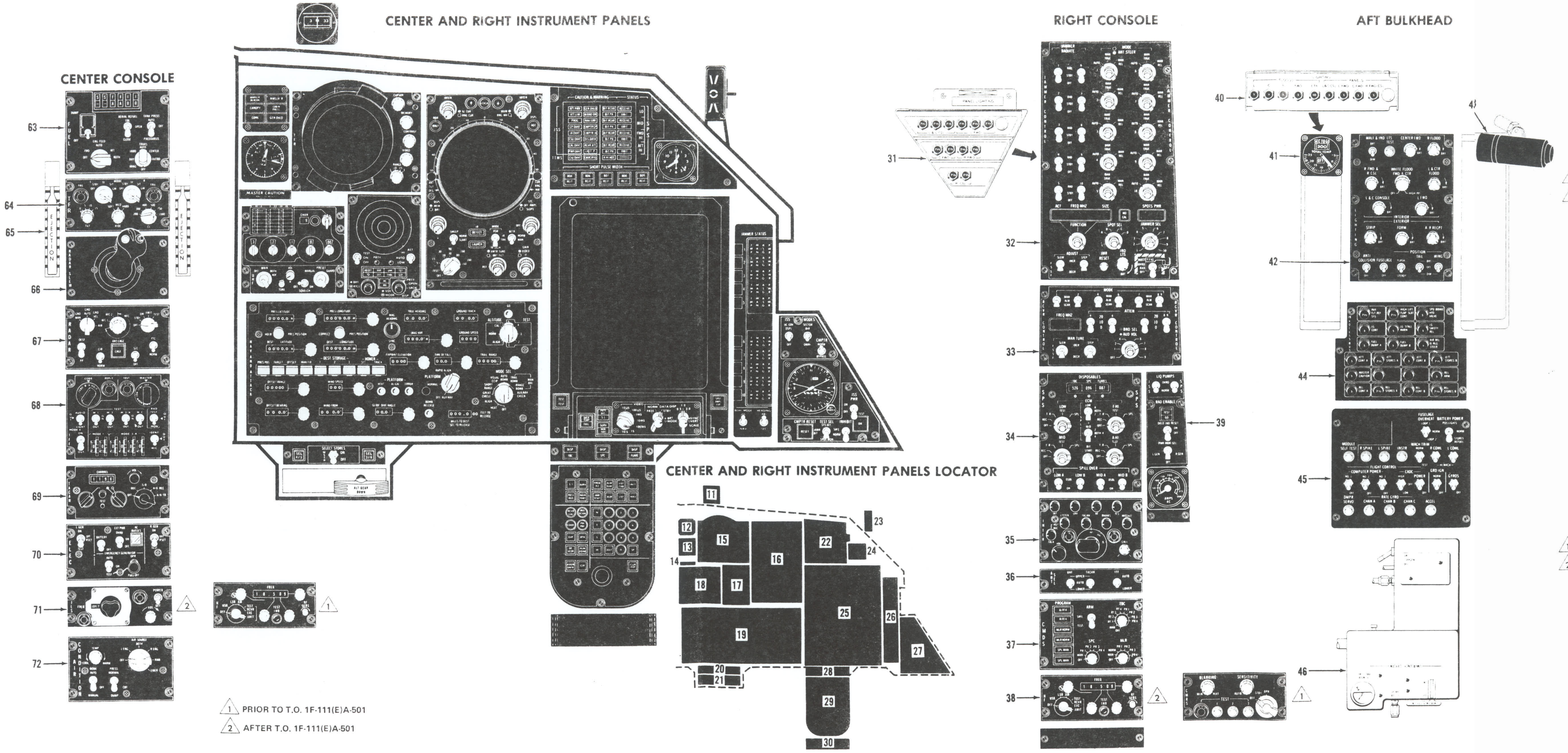
Figure FO-3.

# CREW STATION (EWO)




CENTER CONSOLE

1. FUEL CONTROL PANEL
2. TFR CONTROL PANEL
3. EJECTION HANDLES (2)
4. NAV RADAR SLEW PANEL
5. NAV RADAR CONTROL PANEL
6. IFF CONTROL PANEL
7. TACAN CONTROL PANEL
8. ELECTRICAL CONTROL PANEL
9. 1 HF CONTROL PANEL
   2 ILS CONTROL PANEL
10. AIR CONDITIONING CONTROL PANEL

CENTER AND RIGHT INSTRUMENT PANELS

11. MAGNETIC COMPASS
12. ANNUNCIATOR INDICATOR
13. STANDBY ALTIMETER
14. MASTER CAUTION LAMP
15. TFR SCOPE PANEL
16. NAV RADAR SCOPE PANEL
17. TTWS CONTROL INDICATOR PANEL
18. UHF RADIO CONTROL PANEL
19. INERTIAL NAV CONTROL PANEL
20. STORES JETTISON SELECT PANEL
21. LANDING GEAR EMERGENCY RELEASE HANDLE
22. JSS/SPS WARNING AND CAUTION PANEL
23. ANGLE-OF-ATTACK INDEXER
24. CLOCK
25. DIGITAL DISPLAY INDICATOR
26. JAMMER STATUS PANEL
27. JSS MODES SELECT PANEL
28. DISPOSABLES CONTROL PANEL
29. DIGITAL DISPLAY INDICATOR CONTROL PANEL
30. STOWAGE BAG

RIGHT CONSOLE

31. INTERNAL LIGHTING CIRCUIT BREAKER PANEL
32. JAMMER CONTROL PANEL
33. RECEIVER CONTROL PANEL
34. SPS CONTROL PANEL
35. INTERPHONE CONTROL PANEL
36. ANTENNA SELECT PANEL
37. CMDS CONTROL PANEL
38. 1 CMRS CONTROL PANEL
    2 HF CONTROL PANEL
39. COOLING AND POWER MONITOR PANEL

AFT BULKHEAD

40. INTERNAL LIGHTING CIRCUIT BREAKER PANEL
41. TOTAL TEMPERATURE INDICATOR
42. LIGHTING CONTROL PANEL
43. UTILITY LIGHT
44. DC CIRCUIT BREAKER PANEL
45. GROUND CHECK PANEL
46. OXYGEN-SUIT CONTROL PANEL (EWO)

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Figure FO–4.

# WARNING, CAUTION AND INDICATOR LAMPS (TYPICAL)

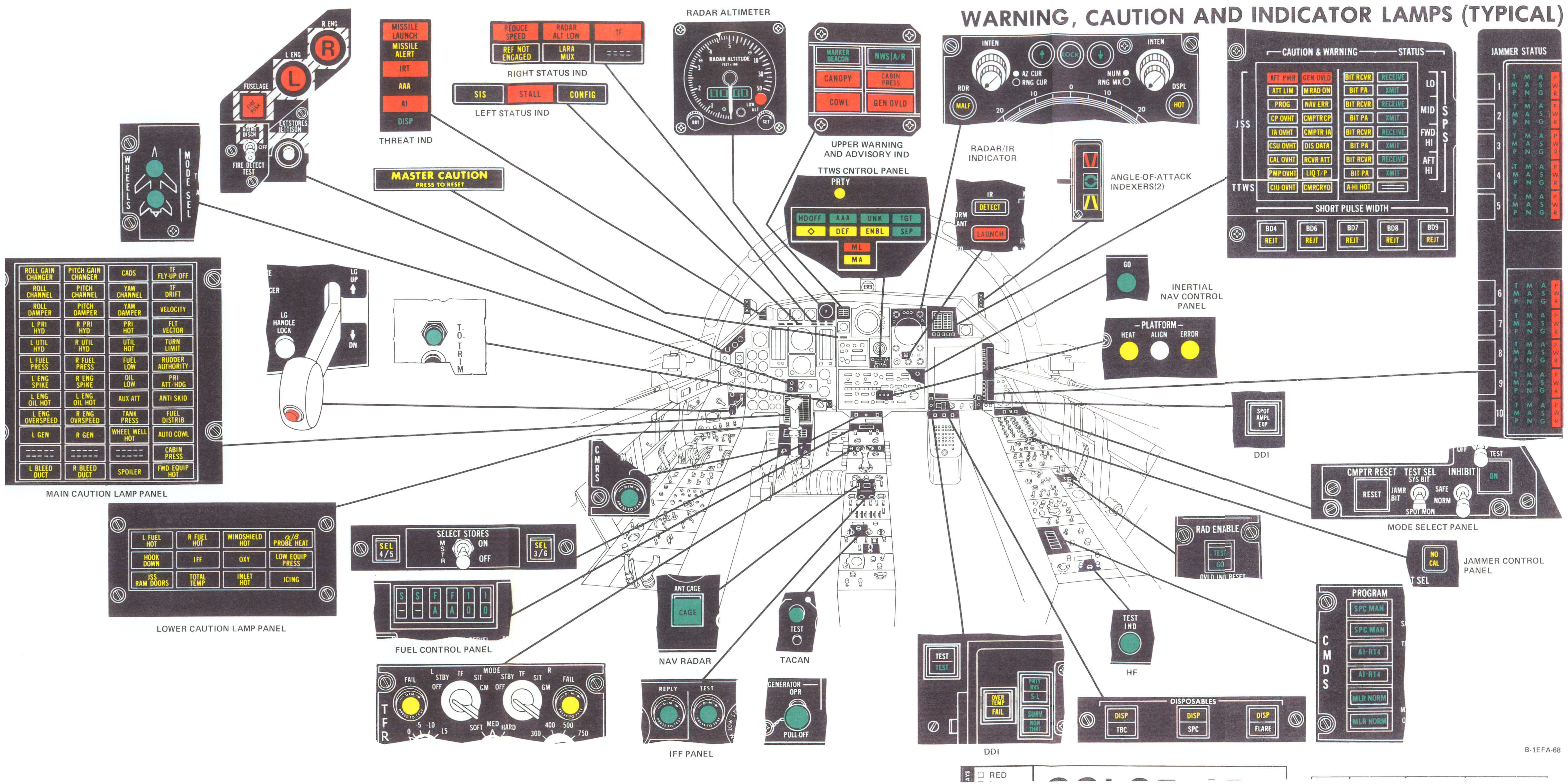
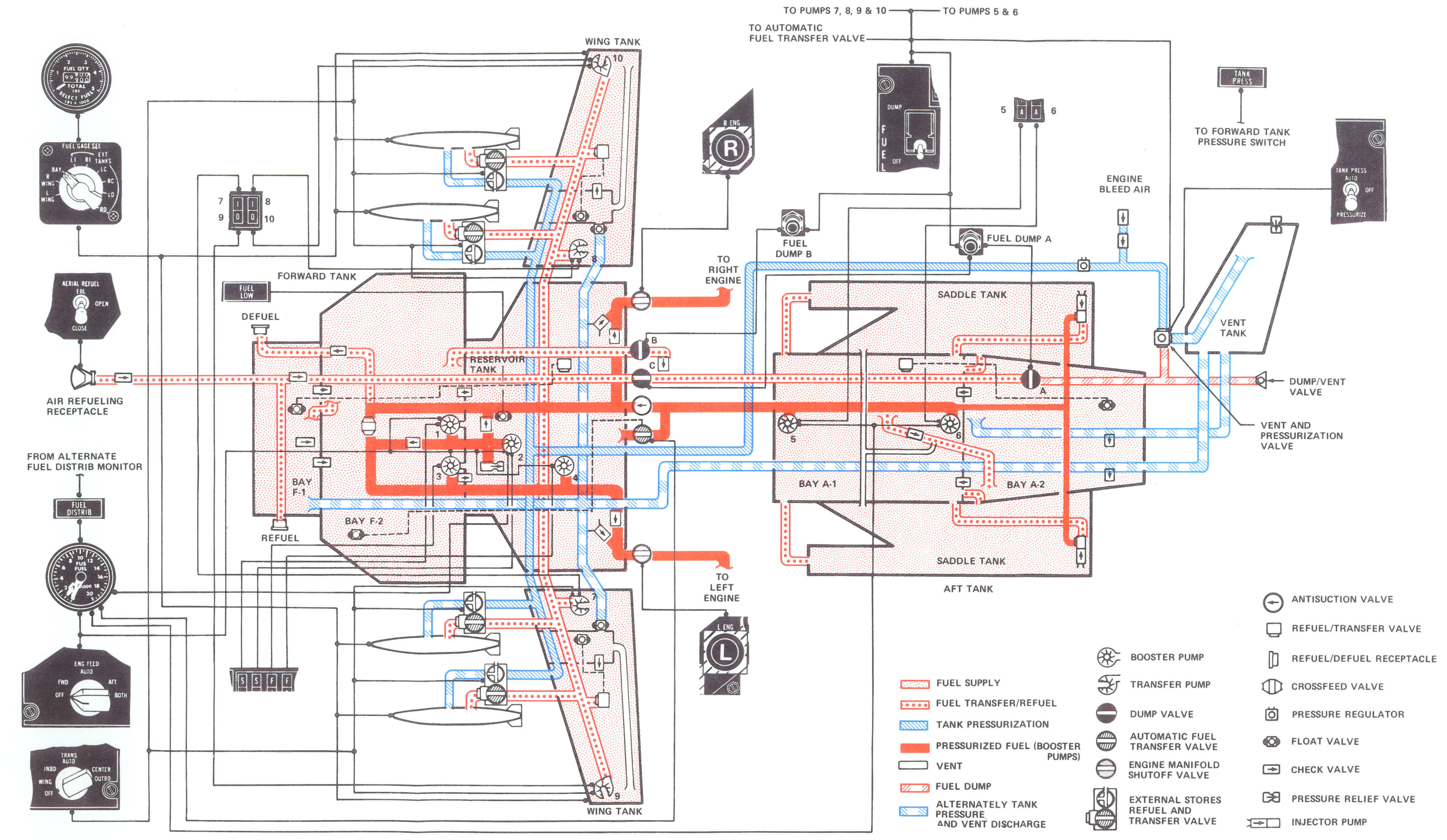



Figure FO-5.

# ENGINE FUEL CONTROL SYSTEM



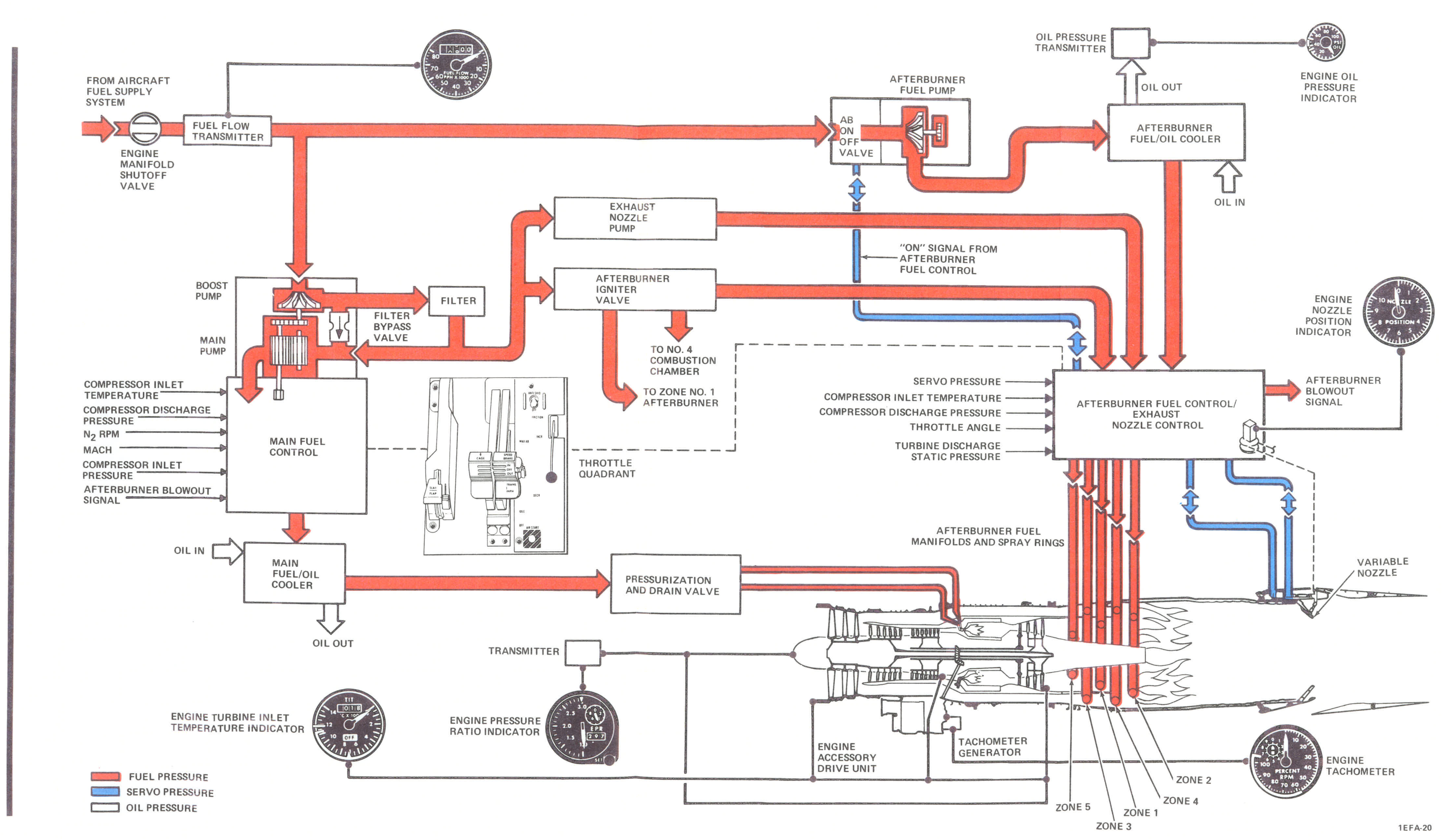
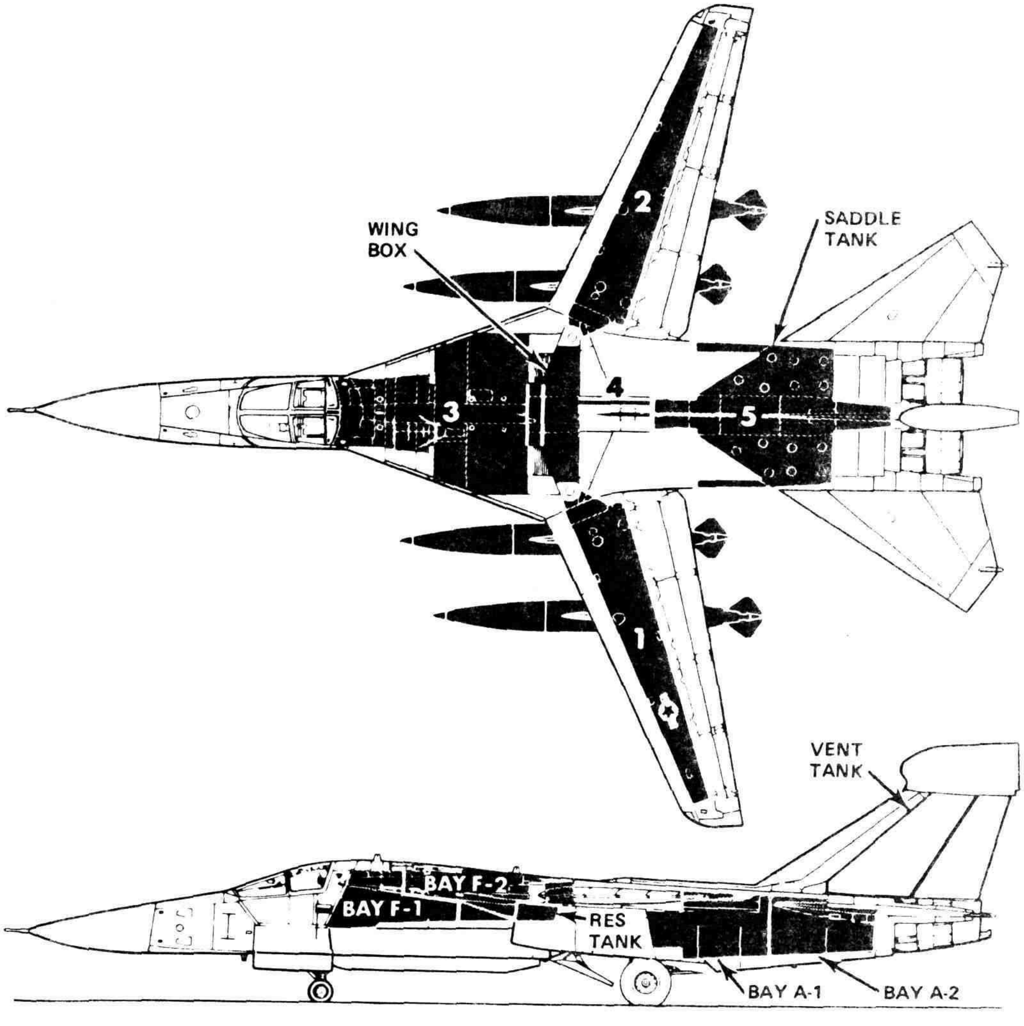




Figure FO-6.

# FUEL SUPPLY SYSTEM
## FUEL QUANTITY AND TANK ARRANGEMENT




| | 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 | 2,822.9 | 18,349 |
| 4 | FUEL LINES | 28.3 | 184 | 53.4 | 347 |
| 5 | AFT FUSELAGE TANK | 1,383.8 | 8,995 | 1,385.9 | 9,008 |
| | TOTAL | 4,998.8 | 32,493 | 5,043.6 | 32,784 |

NOTES:
1. THESE ARE AVERAGE FIGURES BASED ON SINGLE POINT REFUELING AT NORMAL RAMP ATTITUDE. WEIGHTS BASED ON JP-4 FUEL AT 6.5 POUNDS PER GALLON. (STD. DAY ONLY). FOR JP-5/JP-8 USE 6.8 POUNDS PER GALLON.
2. EACH EXTERNAL TANK, WHEN CARRIED, WILL HAVE THE FOLLOWING CAPACITIES:

| USABLE FUEL | | FULLY SERVICED | |
|---|---|---|---|
| GALLONS | POUNDS | GALLONS | POUNDS |
| 601.2 | 3,908 | 603.4 | 3,922 |

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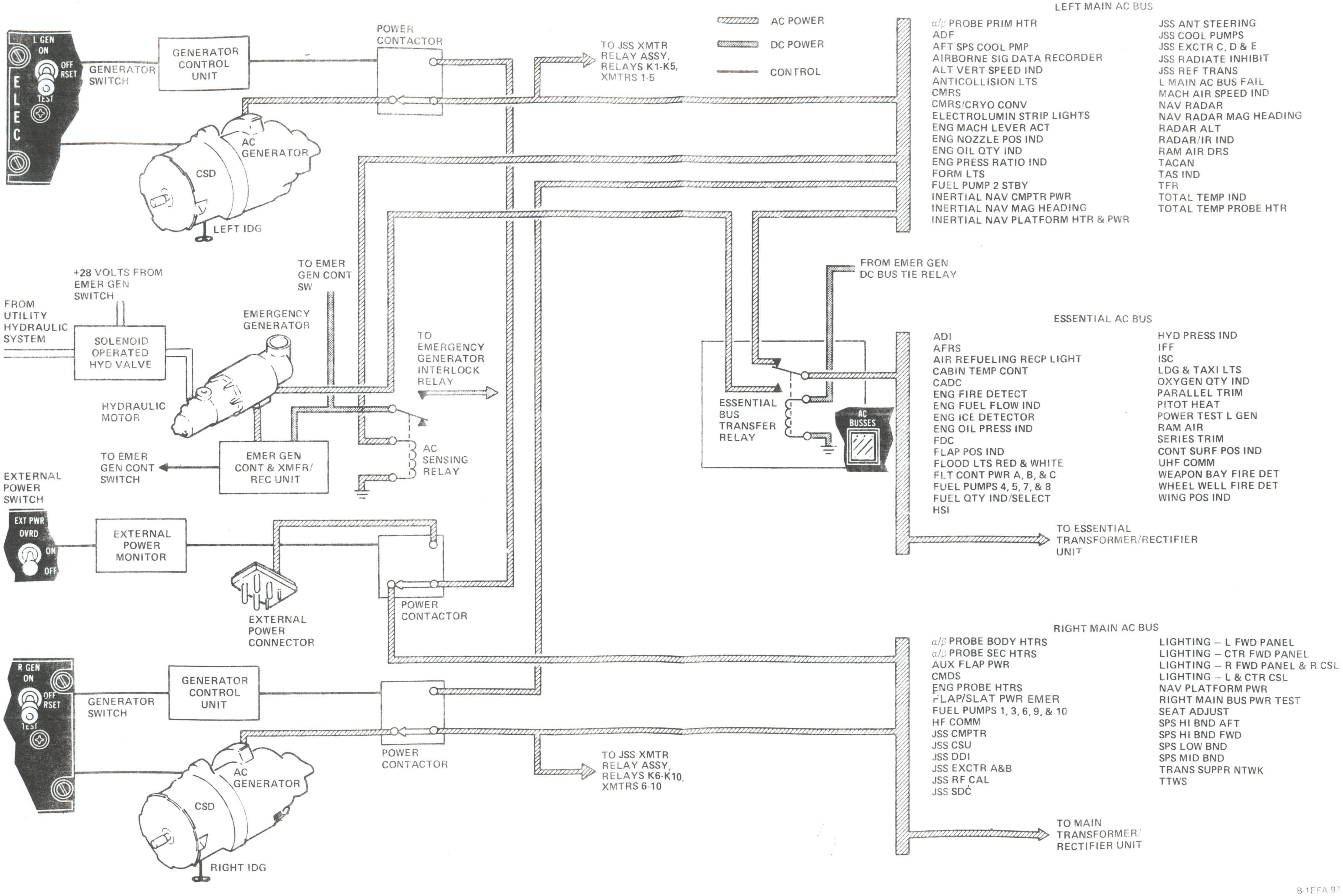
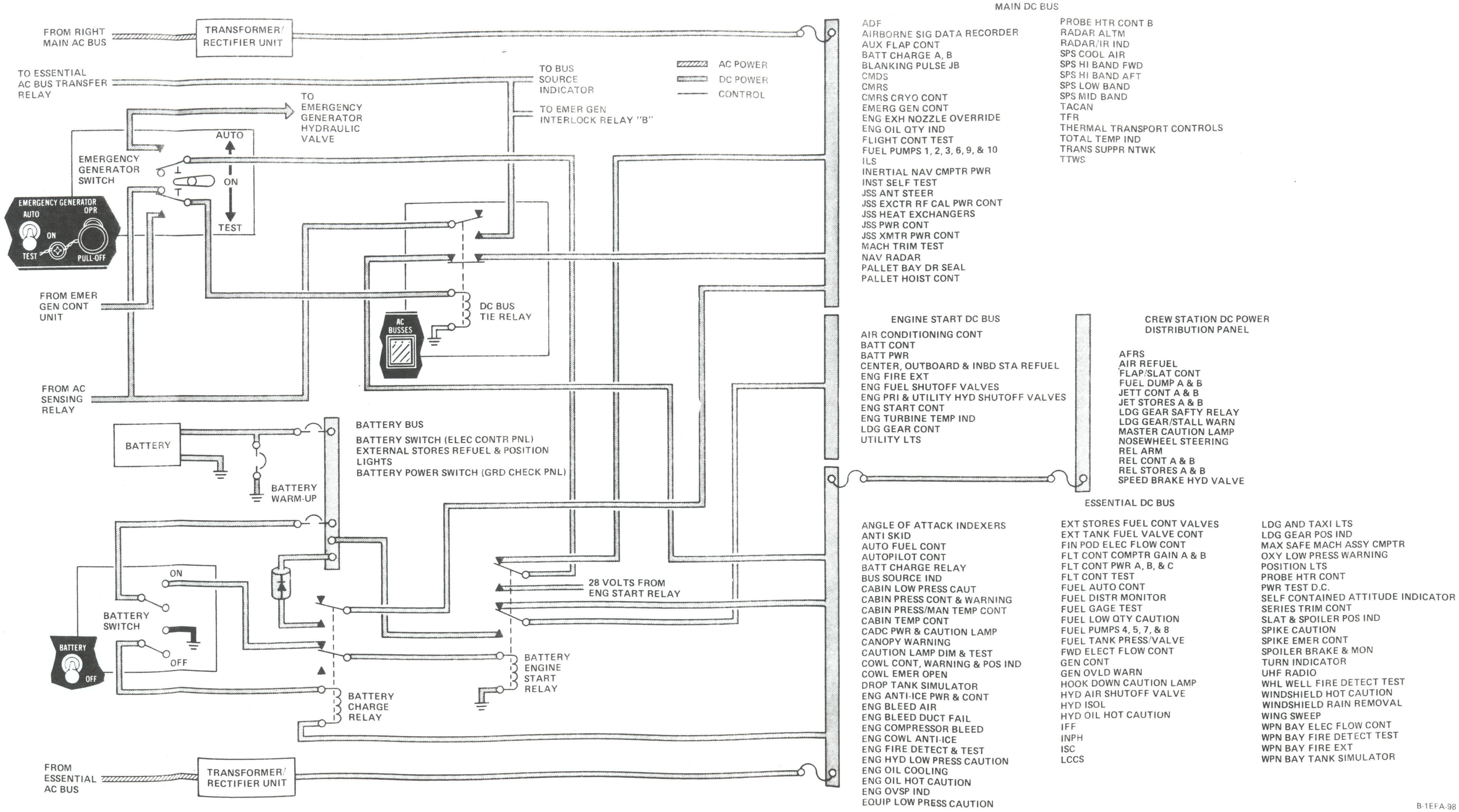
Figure FO–7.

# AC ELECTRICAL POWER SUPPLY SYSTEM



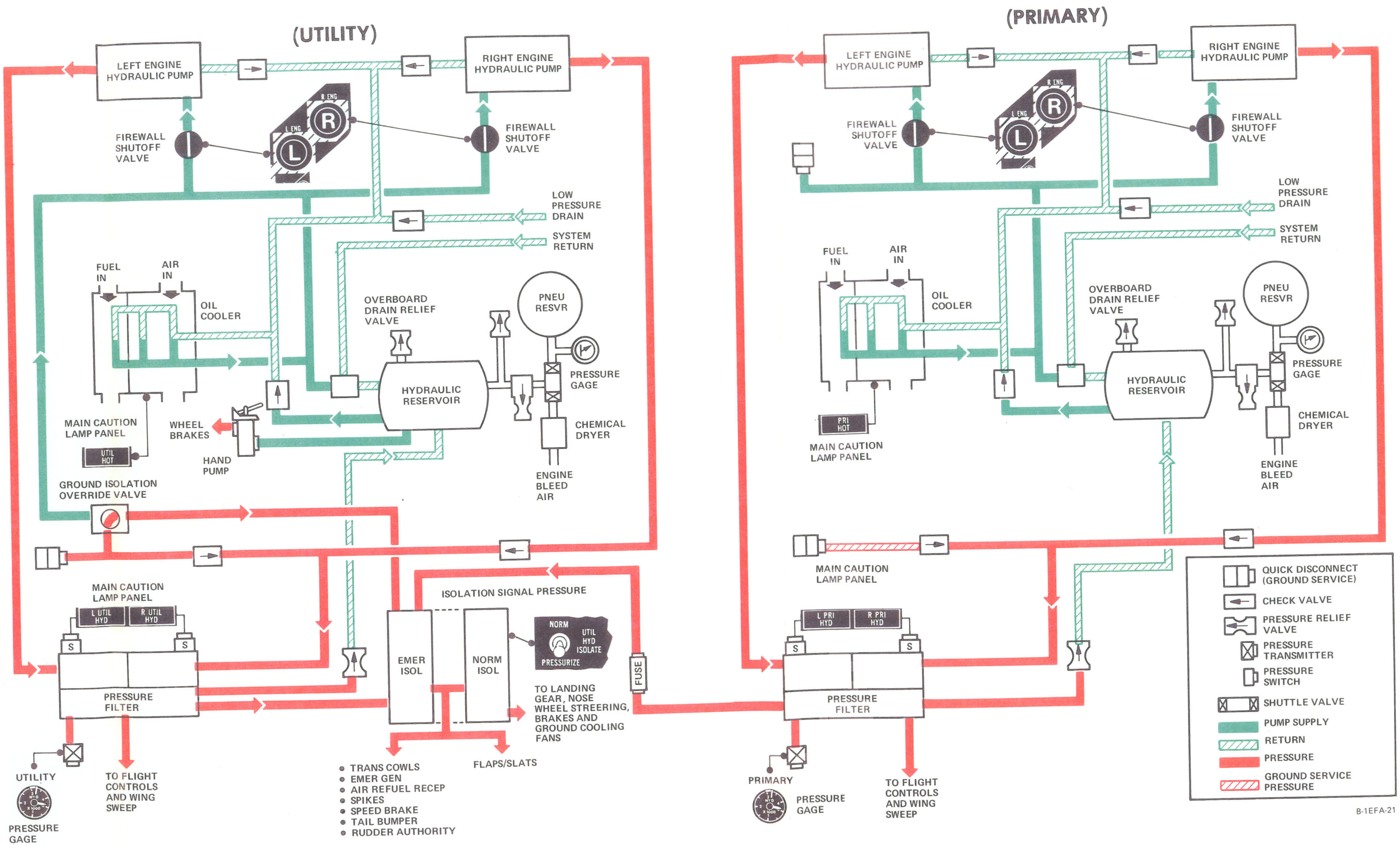

★Figure FO-8.

# DC ELECTRICAL POWER SUPPLY SYSTEM




★Figure FO–9.

# HYDRAULIC POWER SUPPLY SYSTEM




★Figure FO-10.

# PITCH AND ROLL MECHANICAL SCHEMATIC

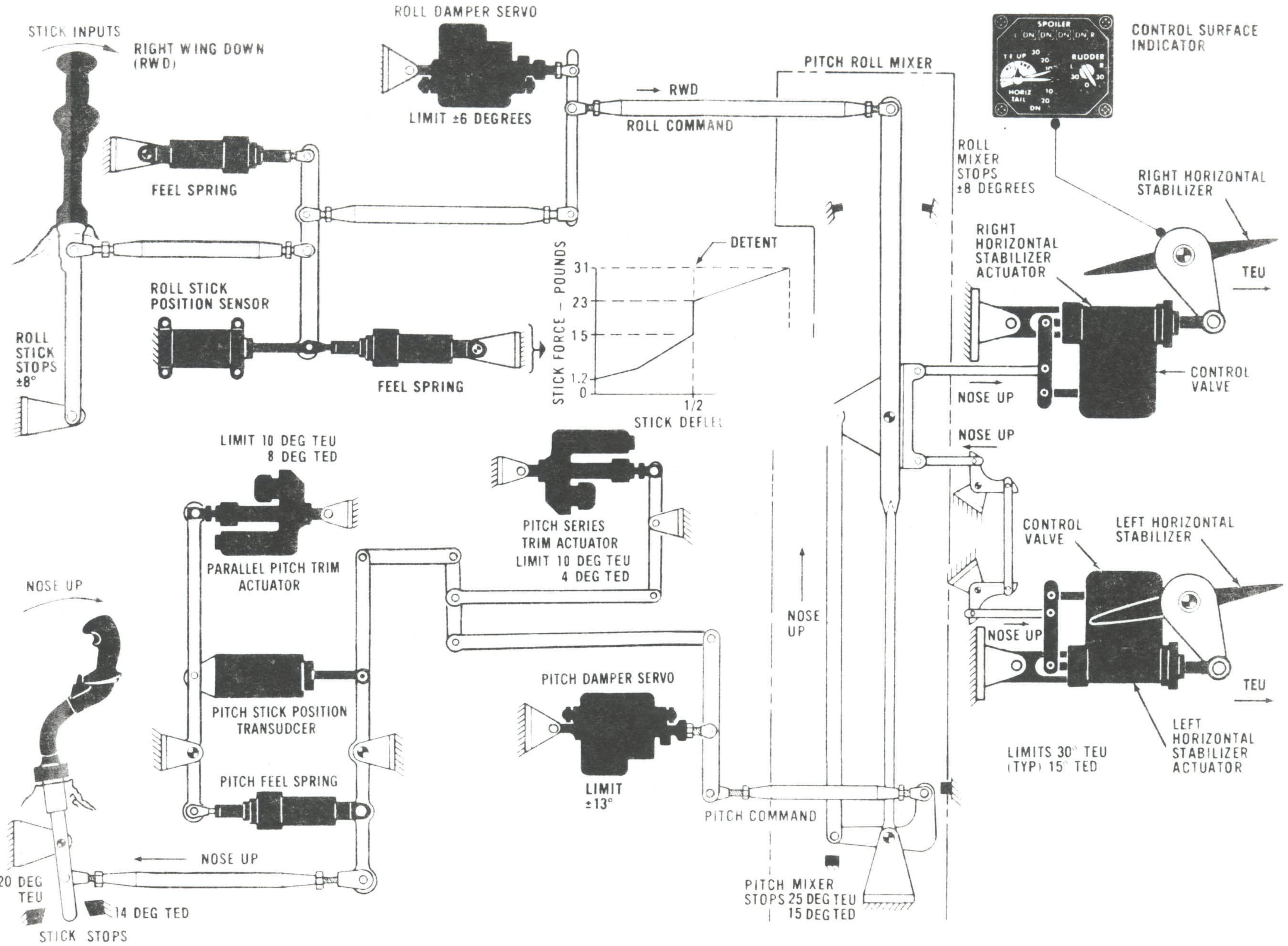



Figure FO–11.

# PITCH CHANNEL ELECTRICAL SCHEMATIC

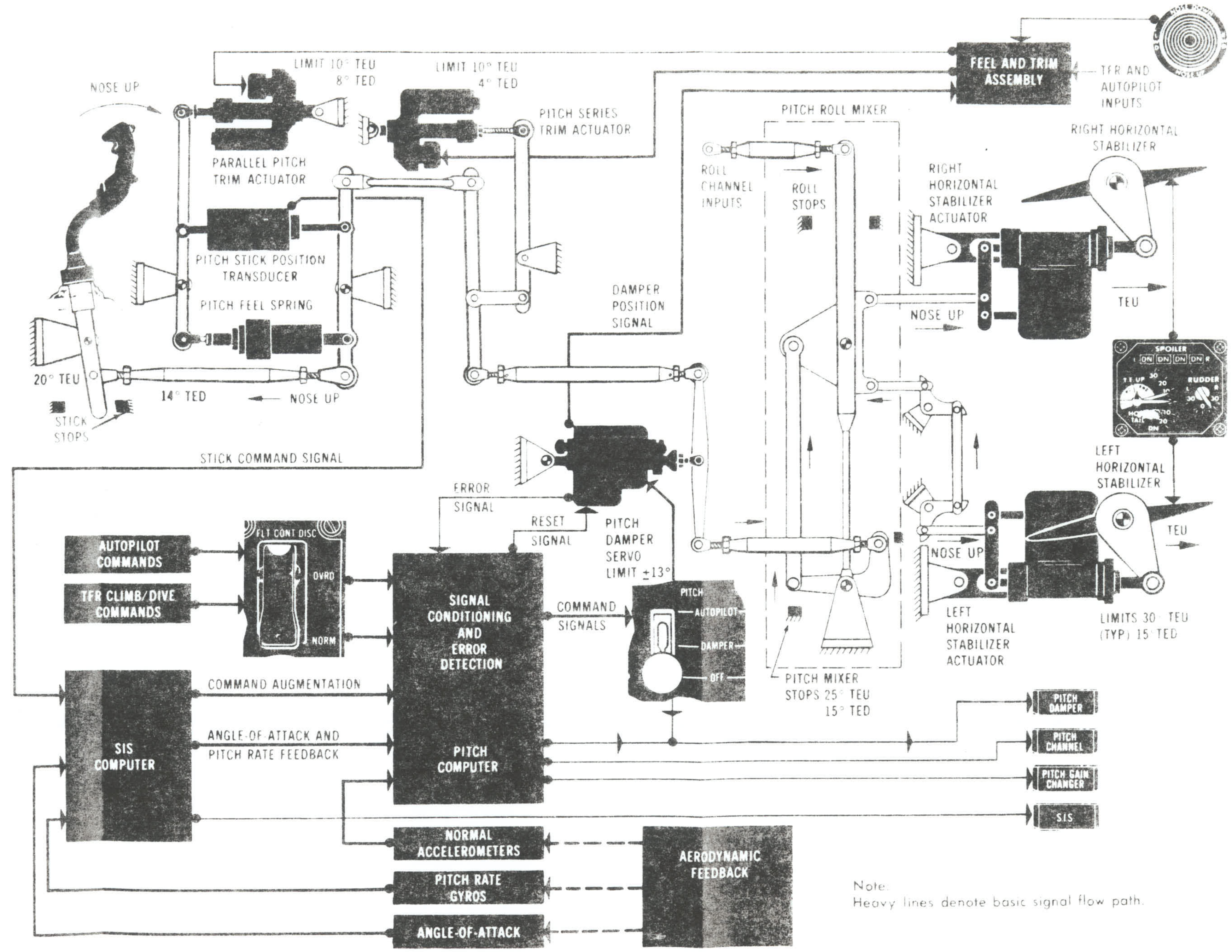



Figure FO-12.

# ROLL CHANNEL ELECTRICAL SCHEMATIC

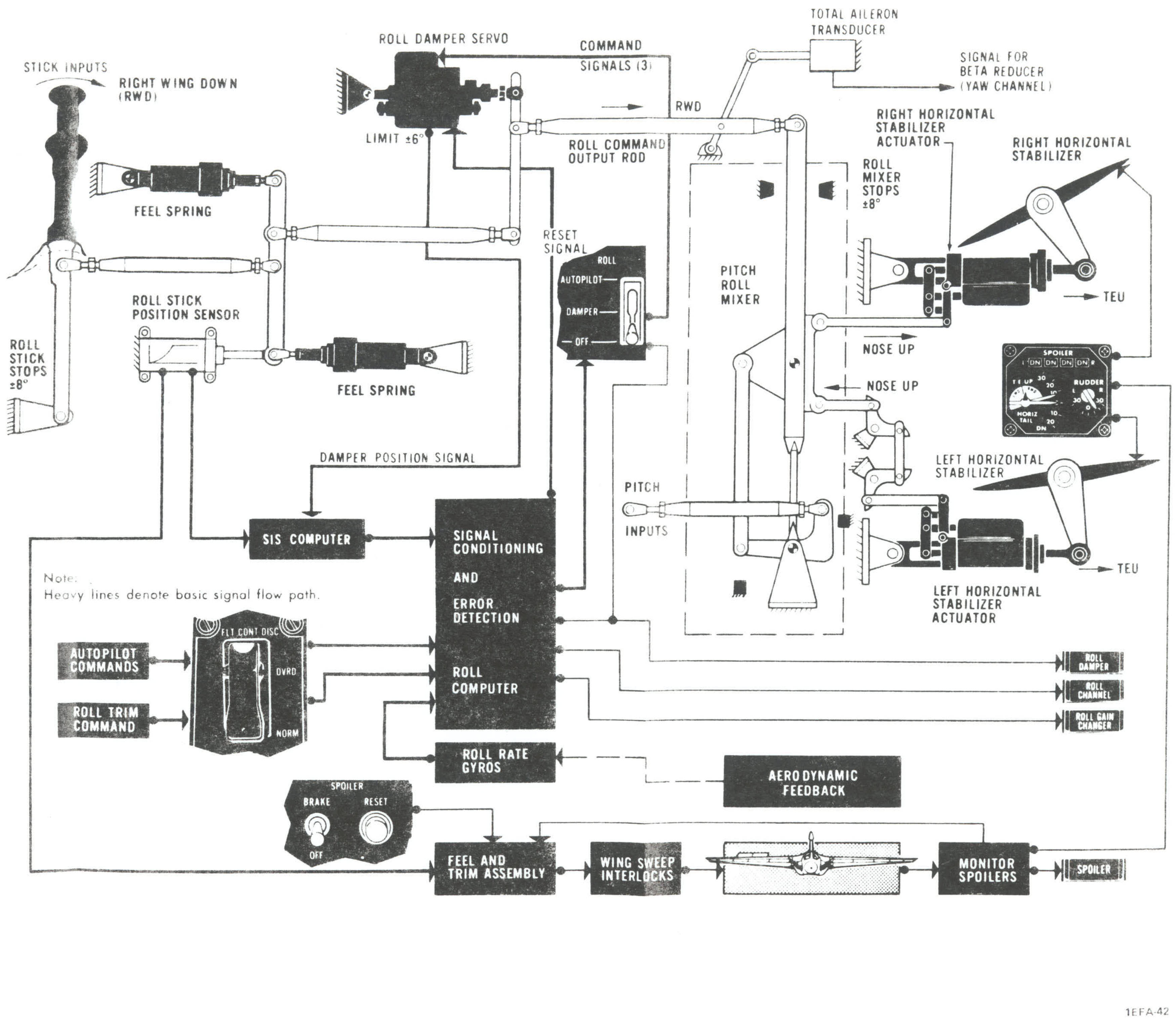



Figure FO-13.

# YAW CHANNEL SCHEMATIC

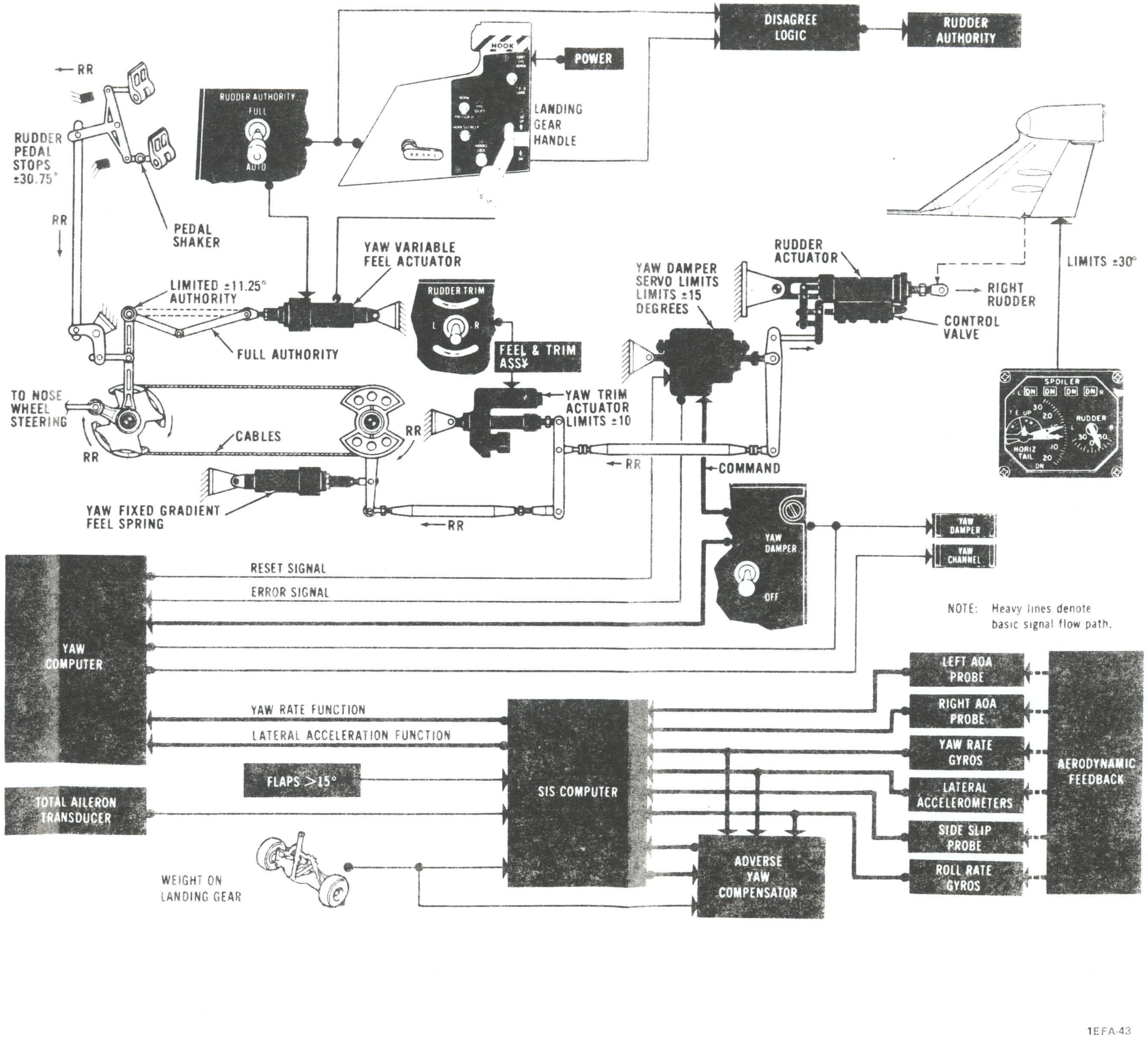



Figure FO-14.

# INSTRUMENT WARNING FLAG ANALYSIS/HSI AND ADI STEERING

## WARNING FLAGS

| 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 SCAI |
| | 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. |

## WARNING FLAGS (CONT)

| | | | | |
|---|---|---|---|---|
| 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 |
| SELF CONTAINED ATTITUDE INDICATOR | POWER (OFF) WARNING FLAG | IN VIEW | PITCH AND ROLL INFORMATION GOOD TO WITHIN 6 DEGREES OF VERTICAL FOR A MINIMUM OF 9 MINUTES. DISPLAYS NOT RELIABLE THEREAFTER. | USE ADI |

## 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 | 8 TO 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 |

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Figure FO-16.

# AIR CONDITIONING AND PRESSURIZATION SYSTEM

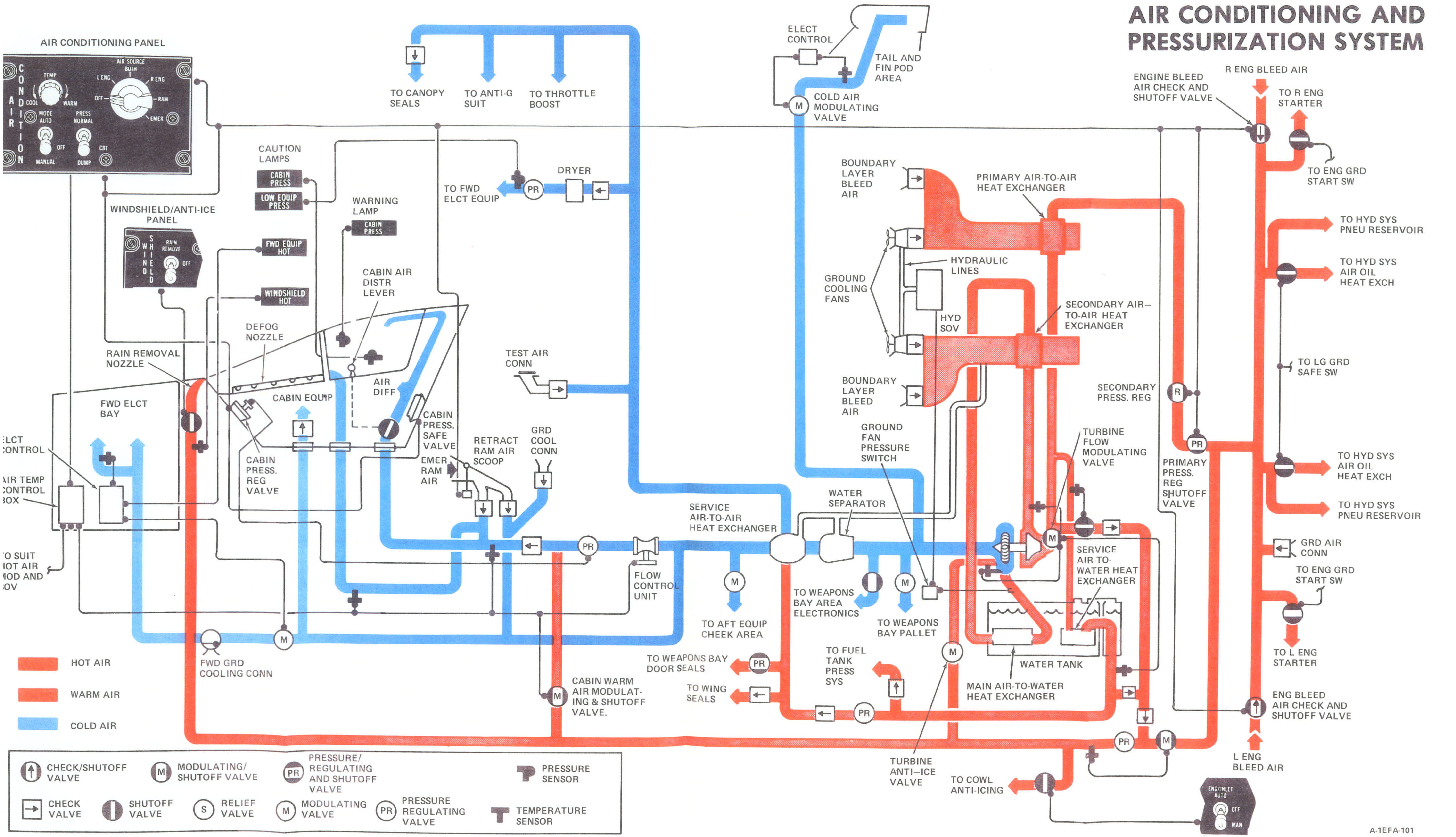


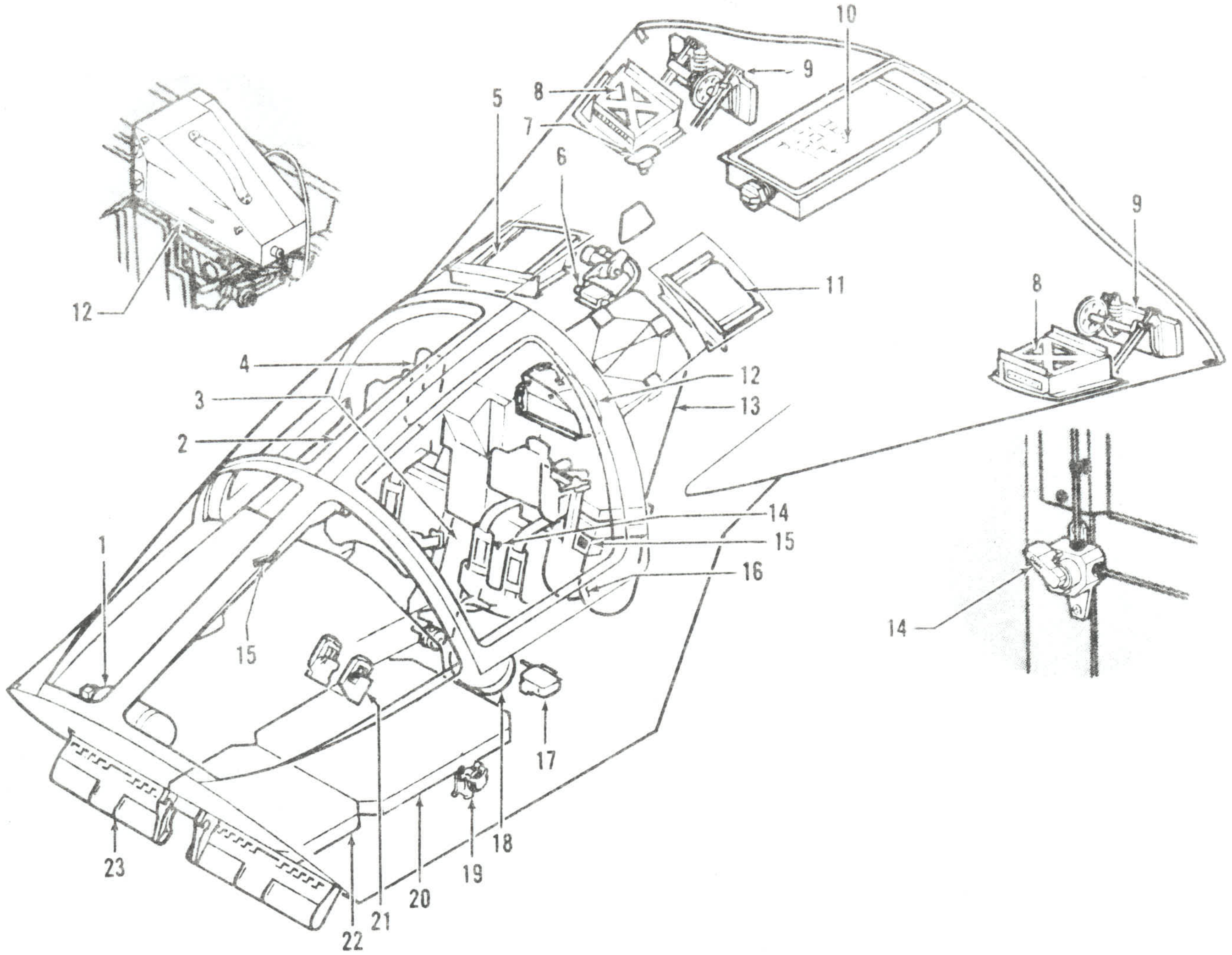

Figure FO–17.

# THERMAL TRANSPORT SYSTEM

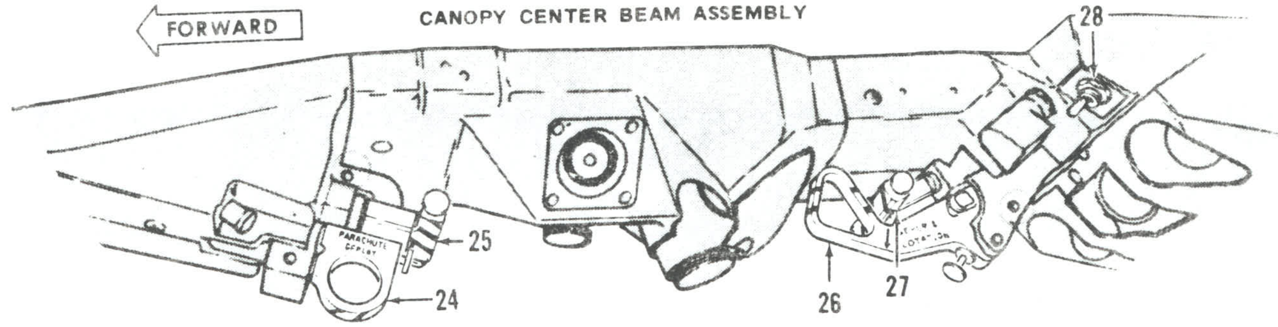

Figure FO-18.

# CREW MODULE GENERAL ARRANGEMENT

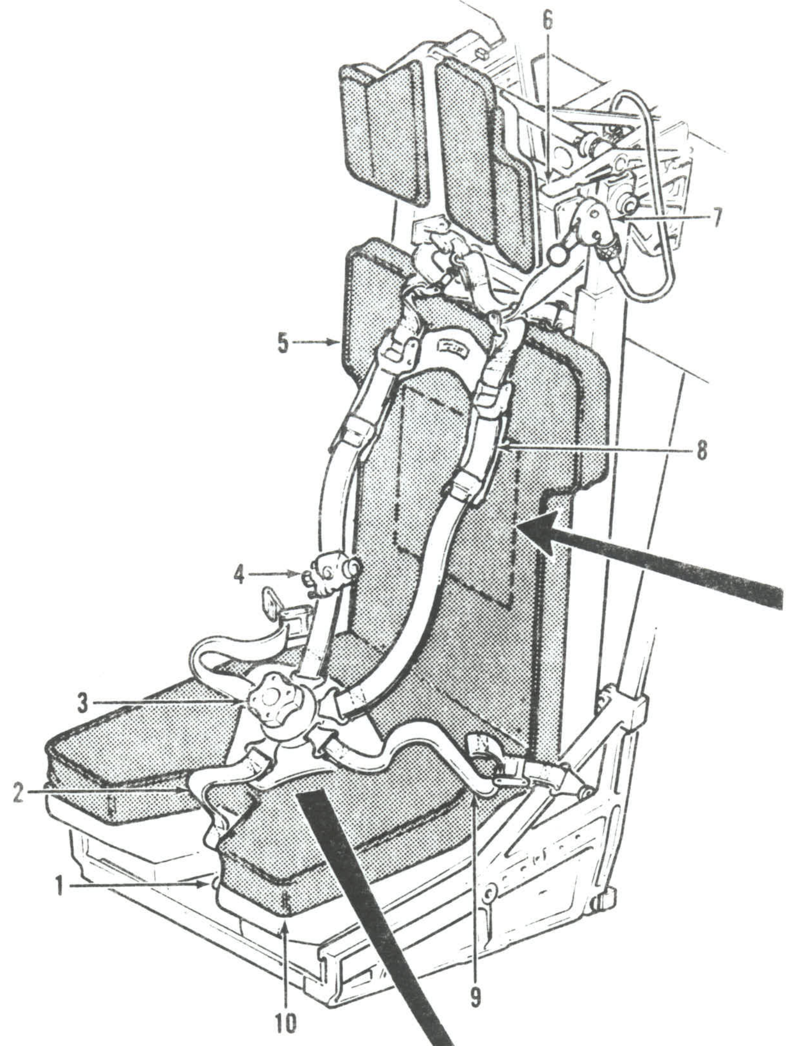
## CREW MODULE SEATS

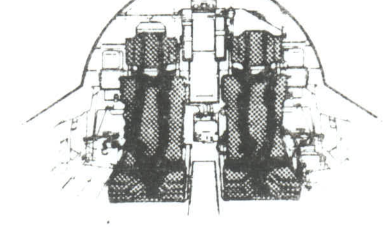



CANOPY CENTER BEAM ASSEMBLY




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

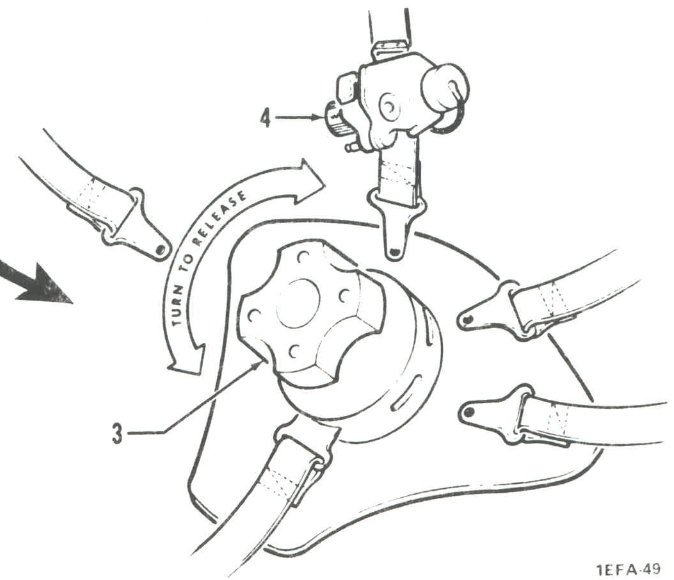


PEEL CLOTH FROM TAPE TO REMOVE INSTRUCTIONS

SURVIVAL GEAR ACCESS

1. DEPRESS EITHER HEADREST ADJUSTMENT LEVER FOR RH SEAT; SLIDE HEADREST FULLY FORWARD.
2. PULL UP ON RH SEAT PAN FORE AND AFT ADJUSTMENT HANDLE; SLIDE SEAT FULLY FORWARD.
3. LH CREWMEMBER REACHES BEHIND RH SEAT AND DISCONNECTS SEAT ADJUSTMENT ACTUATOR ELECTRICAL CONNECTOR.
4. LH CREWMEMBER REACHES BEHIND RH SEAT TO PULL BALL LOCK ATTACH PIN FROM SEAT ACTUATOR WHILE RH CREWMEMBER RELIEVES HIS WEIGHT ON SEAT TO AID PIN REMOVAL. PULL PIN OUT COMPLETELY. CAUTION: SEAT WILL SUDDENLY DROP DOWN WHEN PIN IS REMOVED. MAKE SURE THAT CREWMEMBER'S FEET ARE NOT UNDER SEAT PAN AND HANDS ARE ABOVE AND CLEAR OF SEAT SIDES.
5. WITH CREWMEMBER IN RH STATION FACING AFT, POSITION BOTH HEADREST STOPS LOCATED ON HEADREST TRACKS IN THE UP POSITION.
6. DEPRESS ONE OR BOTH HEADREST ADJUSTMENT LEVERS AND PULL HEADREST TOWARD CREWMEMBER (FORWARD) UNTIL HEADREST DISENGAGES FROM ITS SUPPORT TRACKS.
7. RAISE ENTIRE HEADREST ASSEMBLY UPWARD UNTIL SEAT BACK UPPER SUPPORTS CLEAR THE SEAT BACK TRACKS. PLACE HEADREST ASSEMBLY UPON HEADREST SUPPORT.
8. ROTATE SEAT BACK ASSEMBLY AND SEAT ACTUATOR TOWARD CREWMEMBER (FORWARD).
9. TURN TWO (2) QUARTER-TURN WING STUDS TO REMOVE UPPER AND LOWER SURVIVAL GEAR COMPARTMENT COVERS.
10. REMOVE SURVIVAL GEAR.




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

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Figure FO-19.