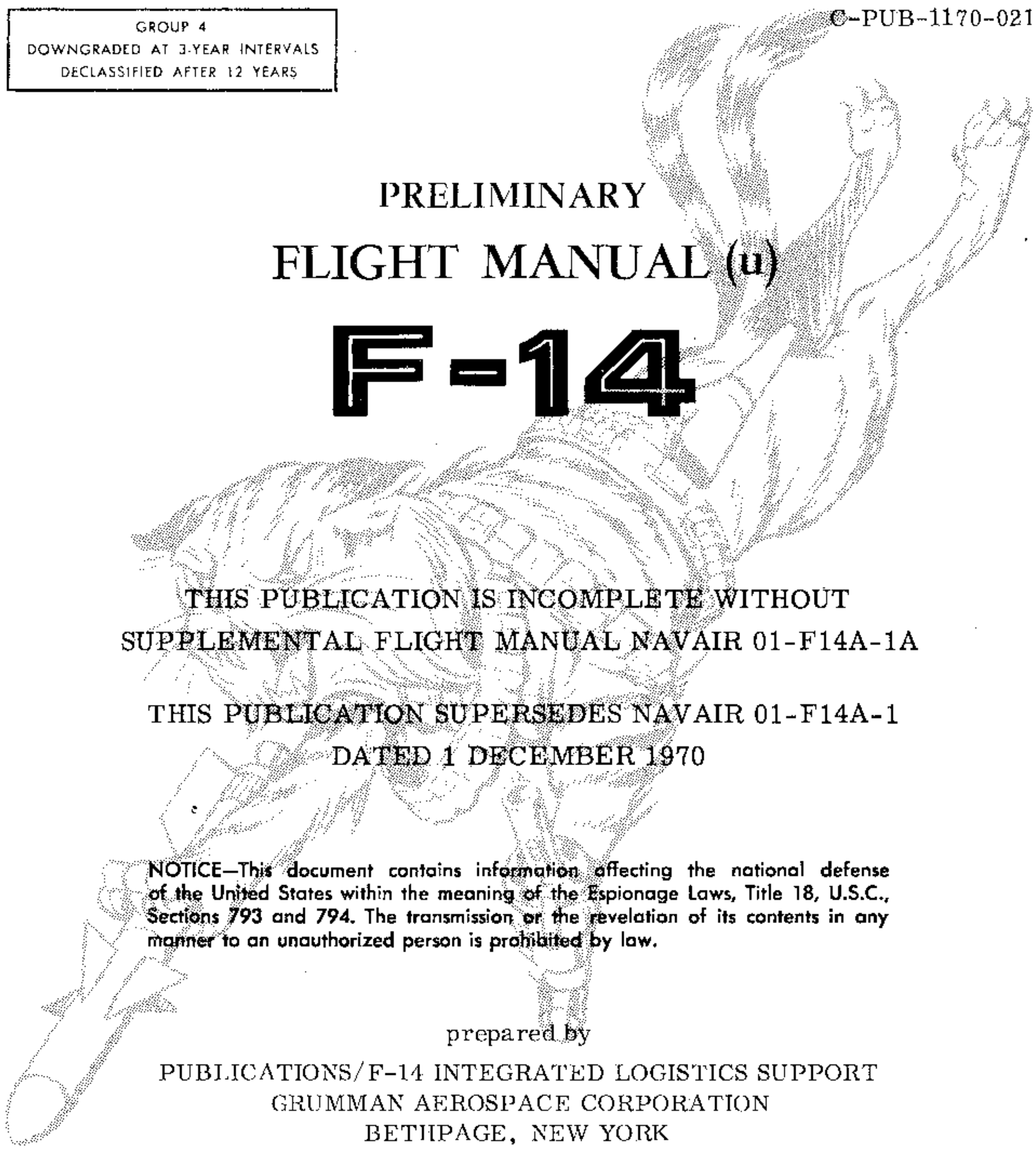
60-1-52

NAVAIR 01-F14A-1
C-PUB-1170-021

GROUP 4
DOWNGRADED AT 3-YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS



PRELIMINARY

FLIGHT MANUAL (u)

F-14

THIS PUBLICATION IS INCOMPLETE WITHOUT
SUPPLEMENTAL FLIGHT MANUAL NAVAIR 01-F14A-1A

THIS PUBLICATION SUPERSEDES NAVAIR 01-F14A-1
DATED 1 DECEMBER 1970

NOTICE—This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws, Title 18, U.S.C., Sections 793 and 794. The transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

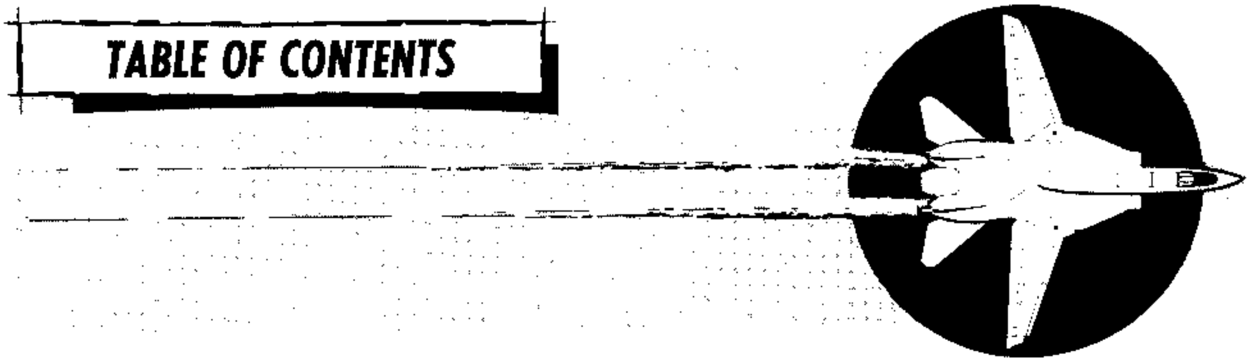
prepared by

PUBLICATIONS/F-14 INTEGRATED LOGISTICS SUPPORT
GRUMMAN AEROSPACE CORPORATION
BETHPAGE, NEW YORK

1 JUNE 1972

THIS PAGE IS UNCLASSIFIED

TABLE OF CONTENTS



NOTE

List of illustrations – Titles included in alphabetical index.

*INDICATES DATA THAT WILL BE ADDED AT A LATER DATE

0-F050-004
114-0

FOREWORD

SCOPE

(U) This is a provisional issue of the Preliminary F-14A Flight Manual. It is developmental in nature and contains information necessary for safe and efficient operation of the F-14A Tomcat. This manual is prepared for use by Navy/Grumman flight test aircrew members and ground personnel in direct support of the F-14 research, development, test, and evaluation program. A Preliminary NATOPS Flight Manual will be issued for the Board of Inspection and Survey (BIS) trails.

CHANGE RECOMMENDATIONS

(U) In order to provide a more comprehensive and accurate Flight Manual, the individual effort and contribution of all users is requested and encouraged. Comments, corrections, and suggestions for improvement to this manual should be brought to the attention of:

Publications, F-14 Integrated Logistics Support,
Plant 15

UPDATING THE MANUAL

(U) This manual will be changed or revised to keep it up to date as additional information becomes available.

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to "WARNINGS", "CAUTIONS", and "NOTES" found through the manual.

WARNING

An operating procedure, practice, or condition, etc., which may result in injury or death, if not carefully observed or followed.

CAUTION

An operating procedure, practice, or condition, etc., which may result in damage to equipment, if not carefully observed or followed.

Note

An operating procedure, practice, or condition, etc., which is essential to emphasize.

WORDING

The concept of word usage and intended meaning which has been adhered to in preparing this Manual is as follows:

"Shall" has been used only when application of a procedure is mandatory.

"Should" has been used only when application of a procedure is recommended.

"May" and "need not" have been used only when application of a procedure is optional.

"Will" has been used only to indicate futurity, never to indicate any degree of requirement for application of a procedure.

GLOSSARY

A

A/B	Afterburner
ac	Alternating Current
A/D	Analog-to-Digital
ACL	Automatic Carrier Landing
ADC	Air Data Computer
ADD	Airstream Direction Detector
ADF	Automatic Direction Finder
ADI	Attitude Director Indicator
AFCS	Automatic Flight Control System
AHRS	Attitude Heading Reference System
AICS	Air Inlet Control System
AIM-54A	Phoenix Missile
AOA	Angle-Of-Attack
APC	Approach Power Compensator
ATDC	Airborne Tactical Data Control
AWCL	All Weather Carrier Landing
AWCLS	All Weather Carrier Landing System
AWCS	Airborne Weapons Control System

B

BDHI	Bearing Distance Heading Indicator
BINGO	Return Fuel State
BIT	Built-In-Test
Bolter	Hook down, unintentional touch and go

C

CAP	Combat Air Patrol
CARQUAL	Carrier Qualifications
CAS	Calibrated Air Speed
CAT	Catapult
CG	Center of Gravity
Charlie Time	Expected Time Over Ramp
CSD	Constant Speed Drive
CV	Aircraft Carrier
CVA	Aircraft Carrier (Attack)

D

dBA	Decibel
dc	Direct Current
DDI	Digital Data Indicator
DECM	Defensive Electronic Counter - Measure
DG	Directional Gyro
DLC	Direct Lift Control
D/L	Data Link

E

EAS	Equivalent Airspeed
ECM	Electronic Countermeasures
ECMD	Electronic Countermeasures Display
ECS	Environmental Control System
EGT	Exhaust Gas Temperature
EPR	Engine Pressure Ratio

F

FAM	Familiarization
FF	Fuel Flow
FL	Flight Level
FMLP	Field Mirror Landing Practice
FOD	Foreign Object Damage

G

G	Guard Channel
g	Gravity
GCU	Generator Control Unit

H

Hang Start	A Start That Results in a Stagnated rpm and Temperature
Hot Start	A Start That Exceeds Normal Starting Temperatures
HSD	Horizontal Situation Display
HSI	Horizontal Situation Indicator
HUD	Heads-Up Display

I

IAS	Indicated Airspeed
ICAO	International Civil Aviation Organization
ICS	Intercommunications
IDG	Integrated Drive Generator
IFF	Identification Friend or Foe
IFR	Instrument Flight Rules
ILS	Instrument Landing System
IMN	Indicated Mach Number
INS	Inertial Navigation System
IR	Infrared

K

KCAS	Knots Calibrated Airspeed
KH2	Kilohertz
KIAS	Knots indicated airspeed
KTS	Knots

L

L_E	Leading Edge
LOX	Liquid Oxygen
LSO	Landing Signal Officer (Paddles)

M

M	Mach
MAC	Mean Aerodynamic Chord
MAX	Maximum
Meatball	Glide Slope Image of Mirror Landing System
MHz	Megahertz
MIL	Military
MLP	Mirror Landing Practice
MSL	Mean Sea Level
MTDS	Marine Tactical Data System

N

NATOPS	Naval Air Training and Operating Procedures
NFO	Naval Flight Officer
NOTAM	Notices to Airmen
NOZ	Nozzle
NTDS	Naval Tactical Data System
NWP	Naval Warfare Publications
N_1	Low Pressure Compressor Rotor speed

O

OAT	Outside Air Temperature
OBC	On-Board Check

P

Paddles	Landing Signal Officer
PCD	Precision Course Direction
PH	Phoenix Missile
PPH	Pounds Per Hour
P_s	Static Pressure
P_t	Total Pressure
PT	Engine Power Trim
psi	Pounds per Square Inch
P_{T7}	Turbine Exhaust Pressure

Q

Q	Dynamic Pressure

R

RPM	High Pressure Compressor Rotor Speed (N_2)

S

SAS	Stability Augmentation System
SIF	Selective Identification Feature
SW	Sidewinder Missile

T

TACAN	Tactical Air Navigation
TAS	True Airspeed
TID	Tactical Information Display
TIT	Turbine Inlet Temperature
T_{T2}	Compressor Inlet Temperature
T_{T4}	Compressor Discharge Temperature
T_S	Static Temperature

U

UHF	Ultra-Height Frequency

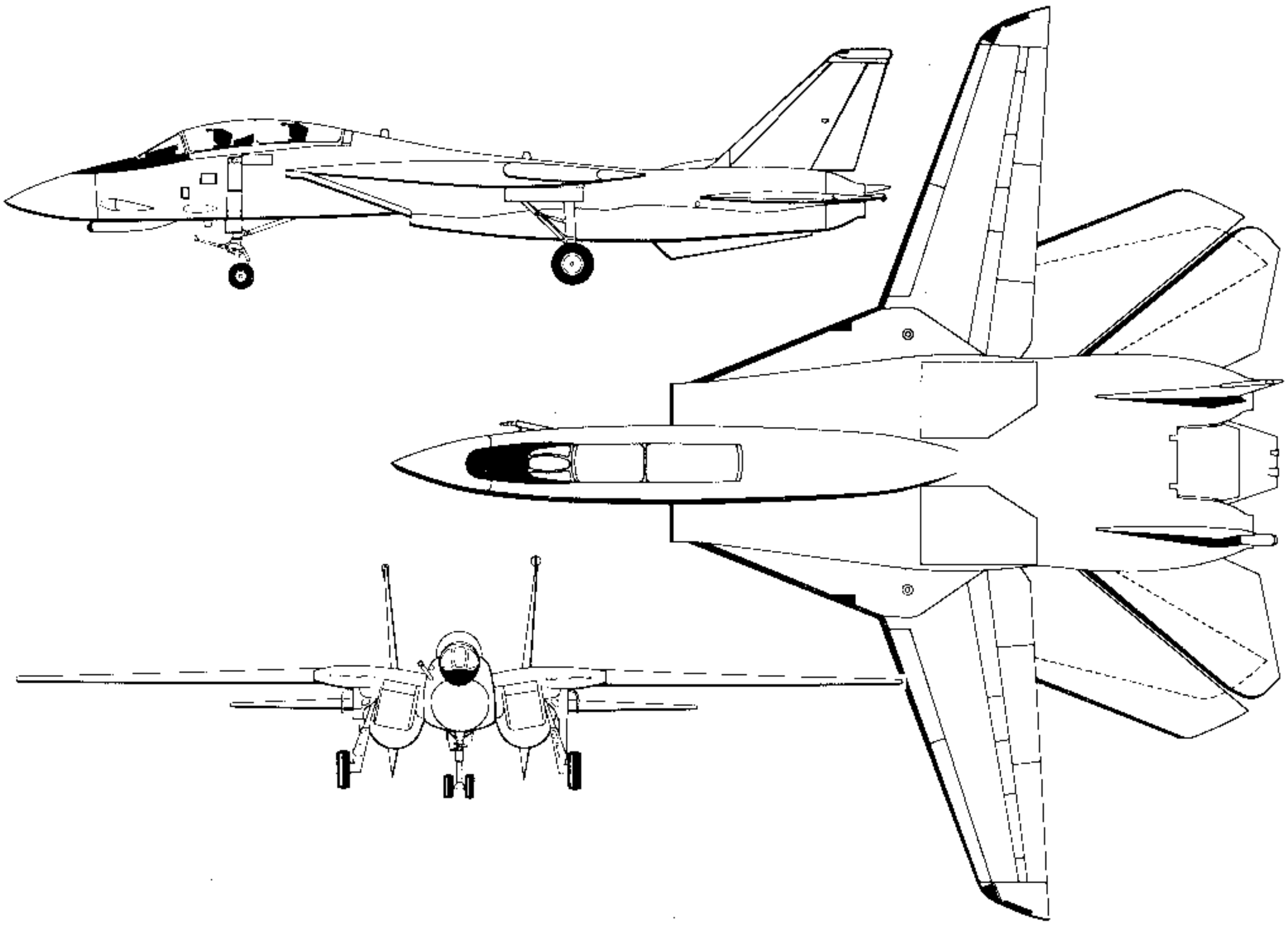
V

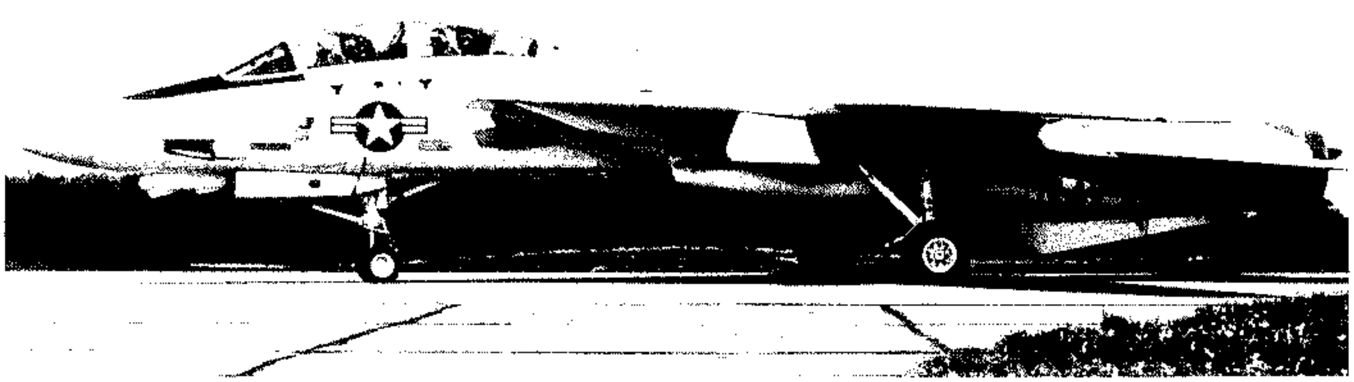
VDIG	Vertical Display Indicator Group
VDI	Vertical Display Indicator
V_R	Rotation Speed
V_1	Critical Engine Failure Speed

W

WST	Weapons System Trainer

F-14A tomcat





1-F050-004
078-0

section I
AIRCRAFT

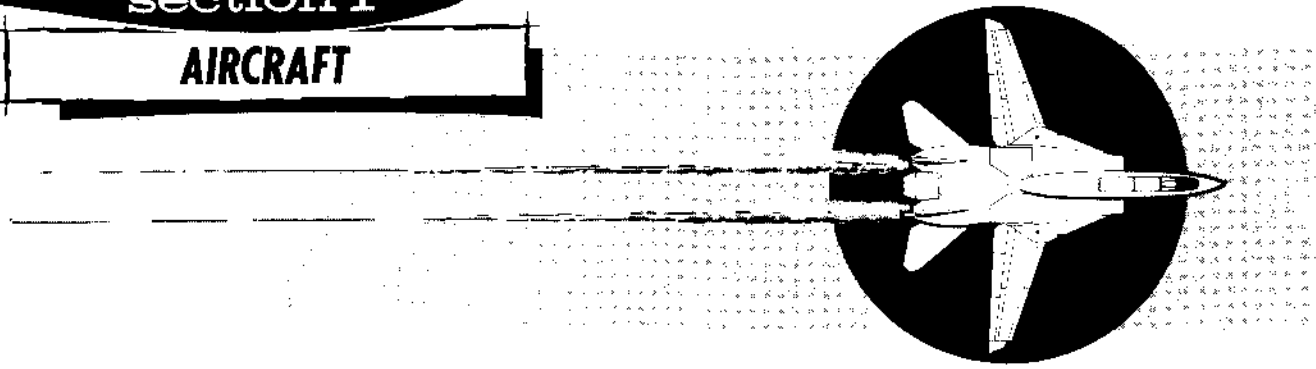


TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

part 1 General Description

AIRCRAFT

(U) The F-14A aircraft is a supersonic, two place, twin engine, swing-wing air superiority fighter designed and manufactured by Grumman Aerospace Corporation. In addition to its primary fighter role carrying missiles (Sparrow and/or Sidewinder) and an internal 20-millimeter gun, the aircraft is designed to perform fleet air defense (Phoenix missiles) and ground attack (conventional ordnance) missions without compromising the basic design. Armament and peculiar auxiliaries used only during secondary missions are installed in low drag, external configurations. Mission versatility and tactical flexibility are enhanced through independent operational capability or integration under existing tactical data systems.

(U) The forward fuselage, containing the crew and electronic equipment, projects forward from the mid fuselage and wing glove. Outboard pivots, contained in the highly swept wing glove, support the movable wing panels, which incorporate integral fuel cells and full-span leading edge slats and trailing edge flaps for supplemental lift control. In flight, the wings may be varied in sweep, area, camber and aspect ratio by selection of any leading edge sweep angle between 20 and 68 degrees. Wing sweep can be automatically or manually controlled to optimize performance and thereby enhance aircraft versatility. Separate variable geometry air inlets offset from the fuselage in the glove, direct primary airflow to two TF30-P-412, dual axial compressor, turbo-fan engines equipped with afterburners for thrust augmentation. The displaced engine nacelles extend rearward to the tail section supporting the twin vertical tails, horizontal tails and ventral fins. The mid and aft fuselage which contains the main fuel cells, tapers off in depth to the rear where it accommodates the speed brake surfaces and arresting hook. Retractable vanes in the glove leading edges extend to supplement lift and compensate for changes in the aircraft aerodynamic center. All control surfaces are positioned by irreversible hydraulic actuators to provide desired control effectiveness throughout the flight envelope. Stability augmentation features are incorporated in the flight control system to enhance flight characteristics and thereby provide a more stable and maneuverable weapons delivery platform. The tricycle-type, forward retracting landing gear is designed for nose tow catapult launch and carrier landings. Missiles and external stores are carried from eight hardpoint stations on the center fuselage between the nacelles and under the nacelles and wing glove; no stores are carried on the movable portion of the wing. The fuel system incorporates both inflight and single point ground refueling capabilities. Aircraft general dimensions are provided in figure 1-1, and figure 1-2 (Sheets 1 and 2) represents the general placement of the components within the aircraft.

AIRCRAFT WEIGHT

(C) The zero fuel/stores gross weight of the aircraft is approximately 37,781 pounds. Internal fuel capacity is approximately 16,000 pounds of JP-5 type fuel and the aircraft has a maximum take-off gross weight of 71,000 pounds. Consult the applicable Handbook of Weight and Balance for the exact weight of any series aircraft.

COCKPITS

(U) The aircraft accommodates a two man crew consisting of the pilot and naval flight officer (NFO) in a tandem seating arrangement. In order to maximize external field of view, the tandem cockpits are predominently located atop the forward fuselage and enclosed by a single clamshell canopy. Integral boarding provisions to the cockpits and aircraft top deck are provided on the left side of the fuselage. Each cockpit incorporates a rocket ejection seat that is vertically adjustable for crew accommodation. A single environmental control system provides conditioned air to the cockpits and electronics bays for pressurization and air conditioning. Oxygen for breathing is supplied to the crew under pressure from liquid storage bottles. The cockpit arrangement provides for a minimum duplication of control capability which of necessity requires two crewmen for flight.

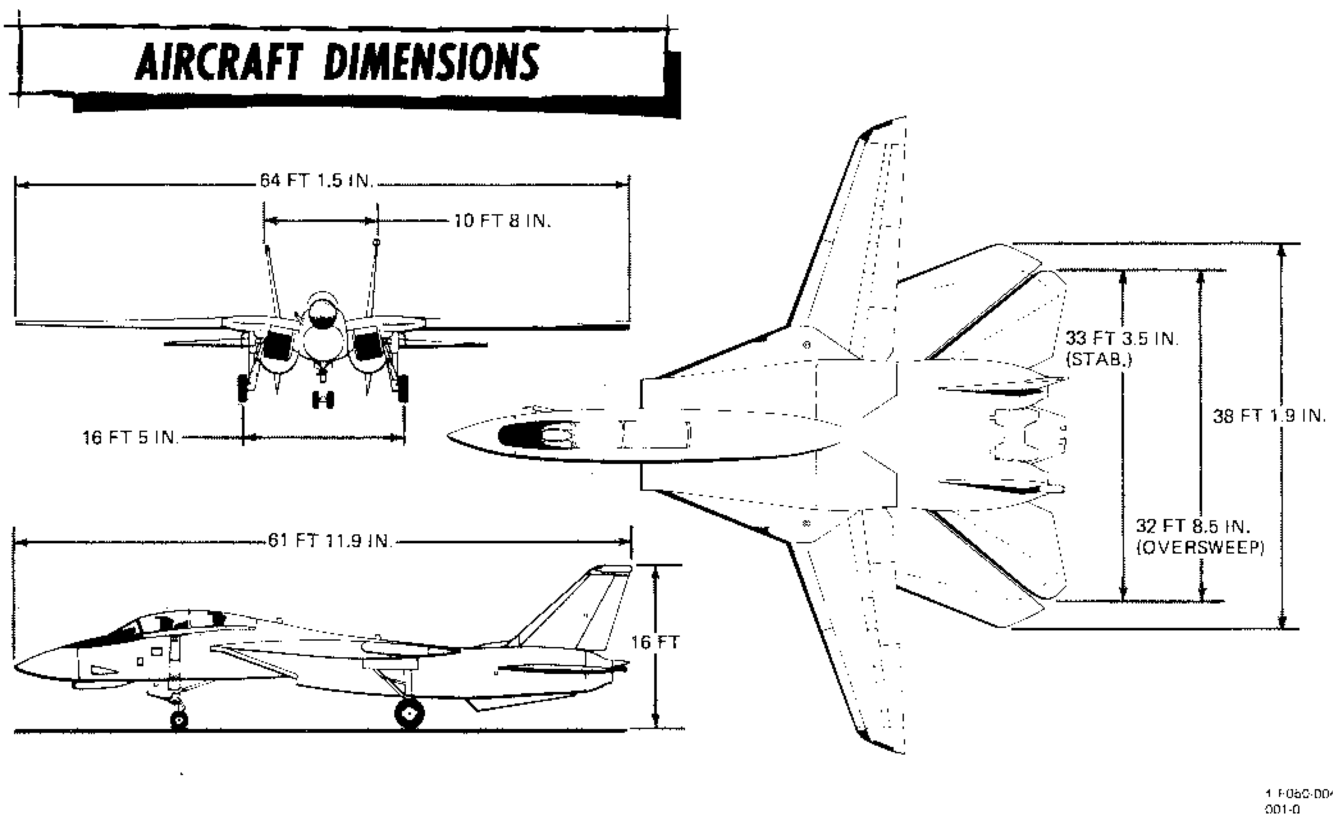
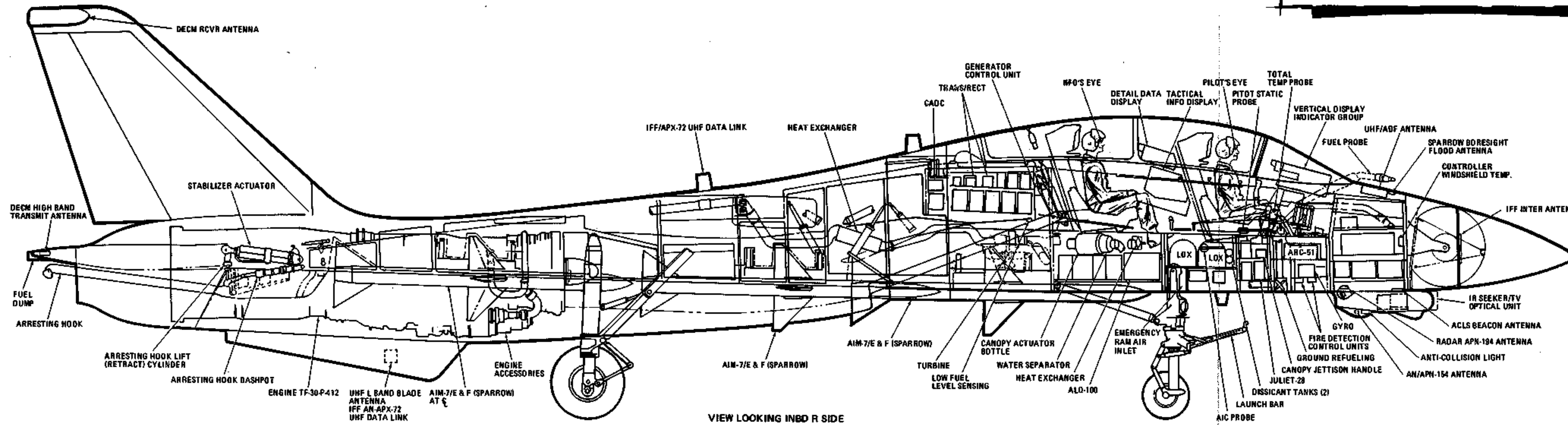
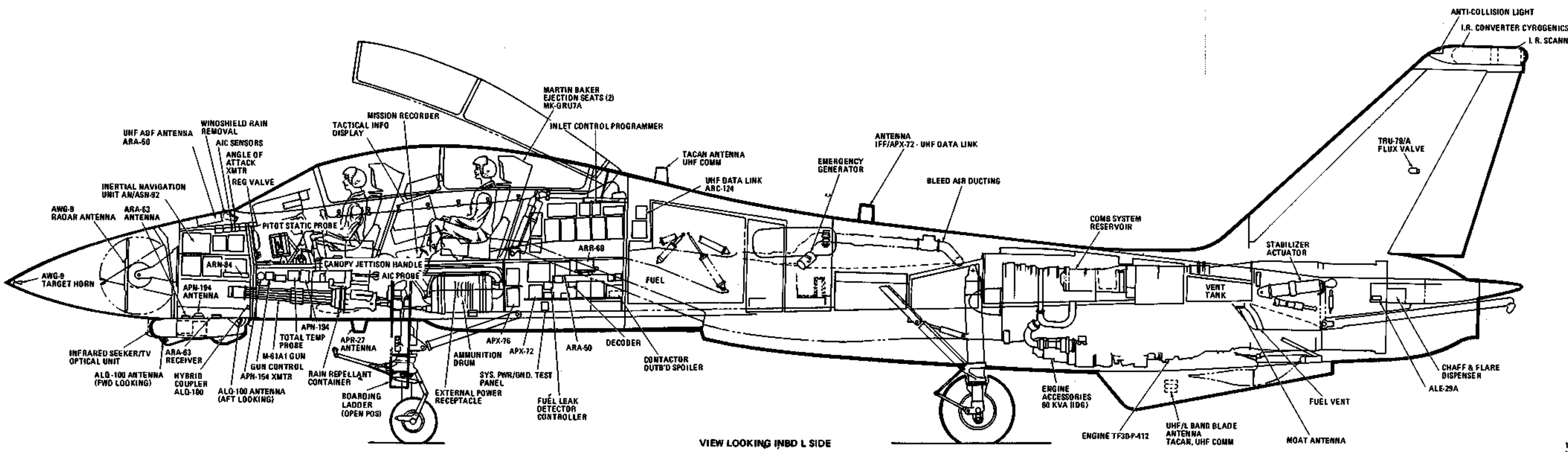


Figure 1-1. (U)

GENERAL ARRANGEMENT

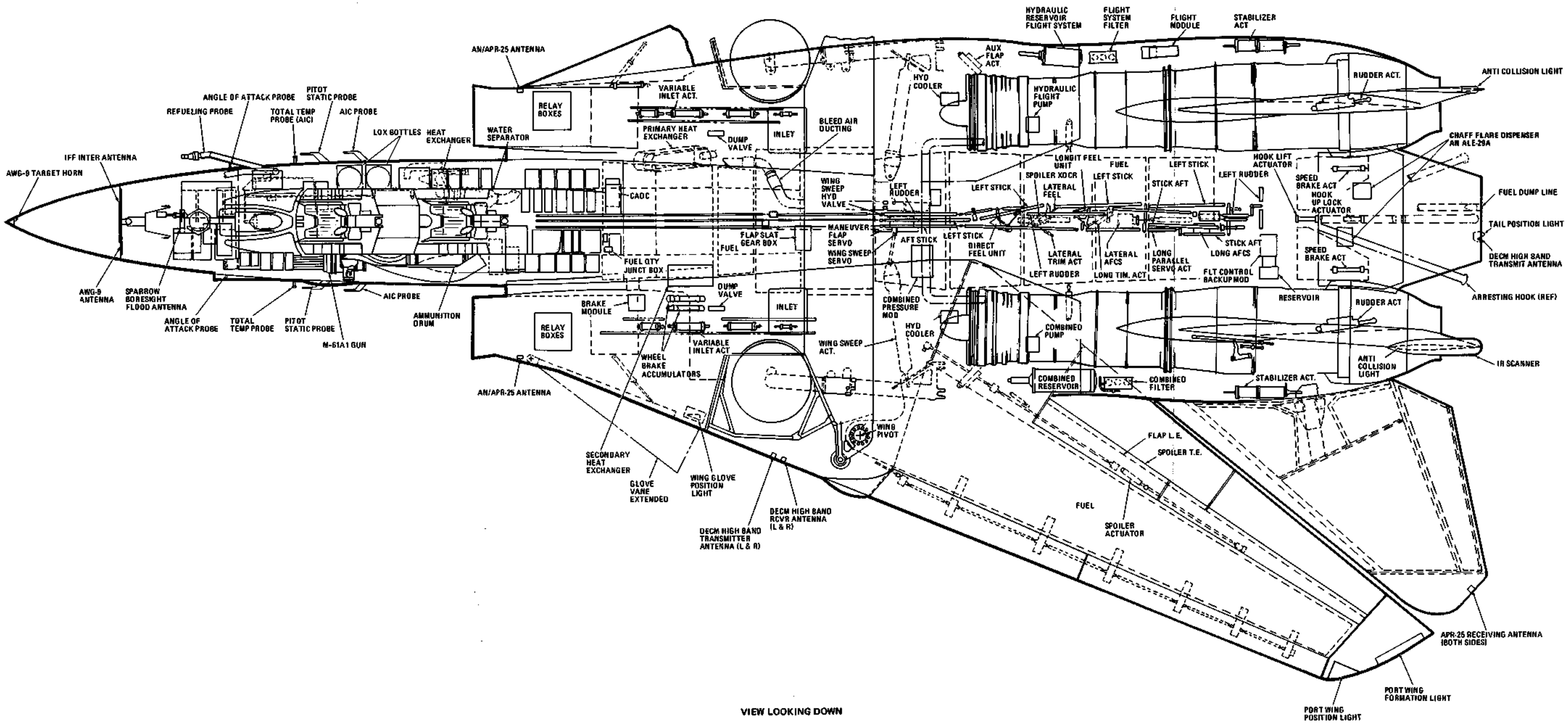




1-F050-004
019-1

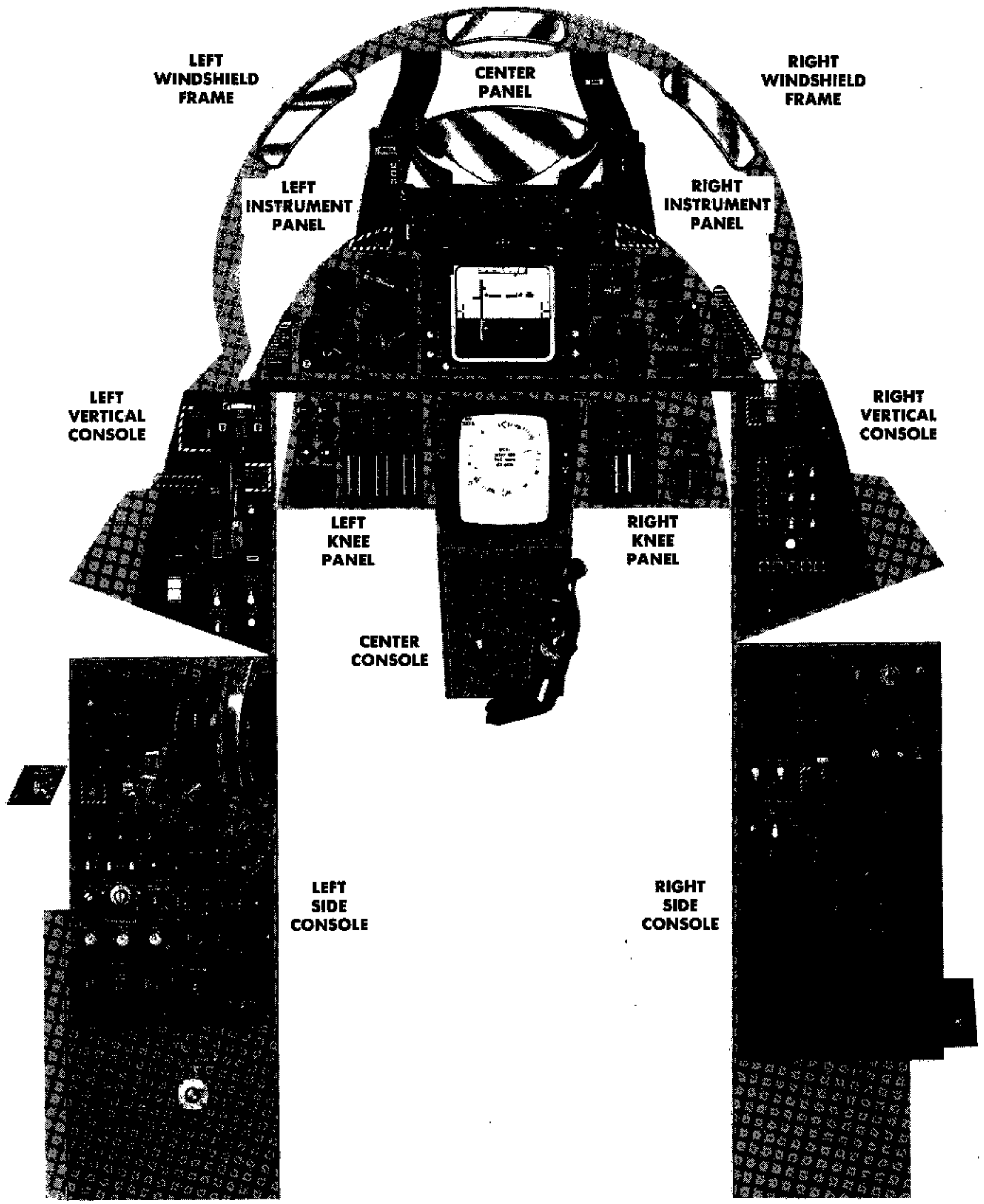
Figure 1-2. (Sheet 1)

GENERAL ARRANGEMENT

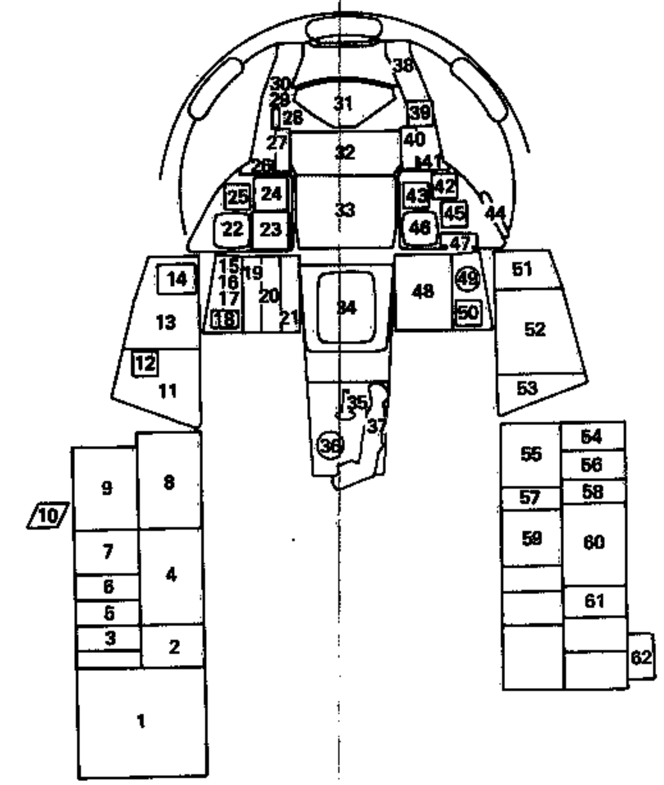


1-F050-004
019-2

Figure 1-2. (Sheet 2)



PILOT'S INSTRUMENT PANEL AND CONSOLES



LEFT SIDE CONSOLE
1. G VALVE PUSHBUTTON
2. OXYGEN-VENT AIRFLOW CONTROL PANEL
3. COMM/NAV COMMAND CONTROL PANEL
4. INTEGRATED CONTROL PANEL
5. TONE VOLUME CONTROL PANEL
6. ICS CONTROL PANEL
7. AFCS CONTROL PANEL
8. THROTTLE QUADRANT
9. EXTERNAL ENVIRONMENT/THROTTLE CONTROL PANEL
10. TARGET DESIGNATE SWITCH

LEFT VERTICAL CONSOLE
11. FUEL MANAGEMENT PANEL
12. CONTROL SURFACE POSITION INDICATOR
13. LANDING GEAR CONTROL PANEL
14. WHEELS-FLAPS POSITION INDICATOR

LEFT KNEE PANEL
15. ENGINE PRESSURE RATIO INDICATOR
16. EXHAUST NOZZLE POSITION INDICATOR
17. OIL PRESSURE INDICATOR
18. HYDRAULIC PRESSURE INDICATOR
19. ELECTRICAL TACHOMETER INDICATOR (RPM)
20. THERMOCOUPLE TEMPERATURE INDICATOR (TIT)
21. RATE OF FLOW INDICATOR (FF)

LEFT INSTRUMENT PANEL
22. SERVOPNEUMATIC ALTIMETER
23. RADAR ALTIMETER
24. AIRSPEED MACH INDICATOR
25. VERTICAL VELOCITY INDICATOR
26. LEFT ENGINE FUEL SHUTOFF HANDLE
27. ANGLE-OF-ATTACK INDICATOR

LEFT FRONT WINDSHIELD FRAME
28. APPROACH INDEXER
29. WHEELS WARNING LIGHT
30. ACLS/AP WARNING LIGHT

CENTER PANEL
31. HEADS UP DISPLAY
32. AIR COMBAT MANEUVER PANEL
33. VERTICAL DISPLAY INDICATOR (VDI)
34. HORIZONTAL SITUATION DISPLAY INDICATOR (HSI)
35. PEDAL ADJUST HANDLE
36. BRAKE PRESSURE INDICATOR
37. CONTROL STICK

RIGHT FRONT WINDSHIELD FRAME
38. SAM WARNING LIGHT
39. STANDBY COMPASS

RIGHT INSTRUMENT PANEL
40. WING SWEEP INDICATOR
41. RIGHT ENGINE FUEL SHUTOFF HANDLE
42. ACCELEROMETER
43. STANDBY ATTITUDE INDICATOR
44. CANOPY JETTISON HANDLE
45. CLOCK
46. BEARING DISTANCE HEADING INDICATOR (BDHI)
47. UHF REMOTE INDICATOR

RIGHT KNEE PANEL
48. FUEL QUANTITY INDICATOR
49. LIQUID OXYGEN QUANTITY INDICATOR
50. CABIN PRESSURE ALTIMETER

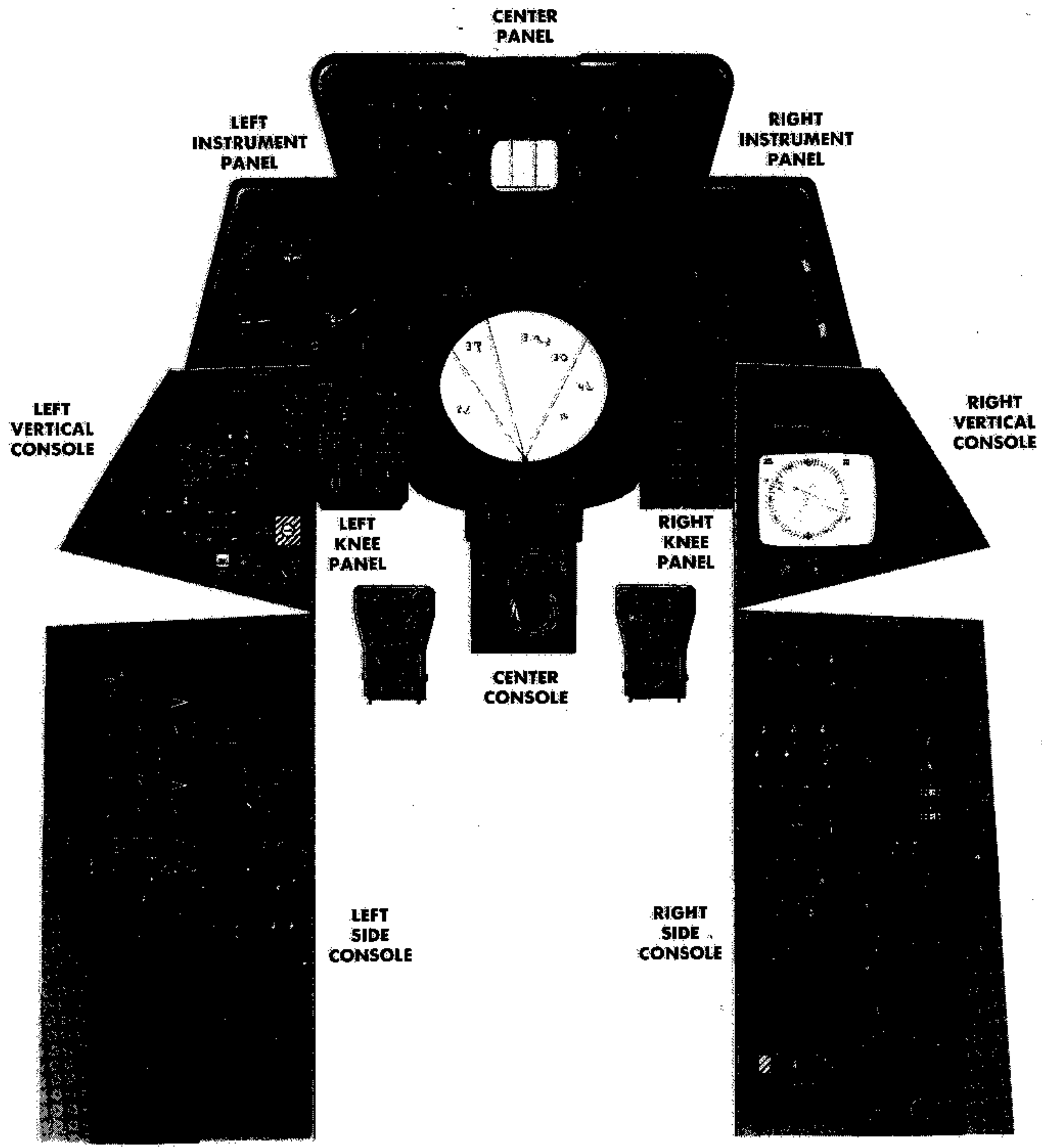
RIGHT VERTICAL CONSOLE
51. ARRESTING HOOK PANEL
52. DISPLAYS CONTROL PANEL
53. ELEVATION LEAD PANEL

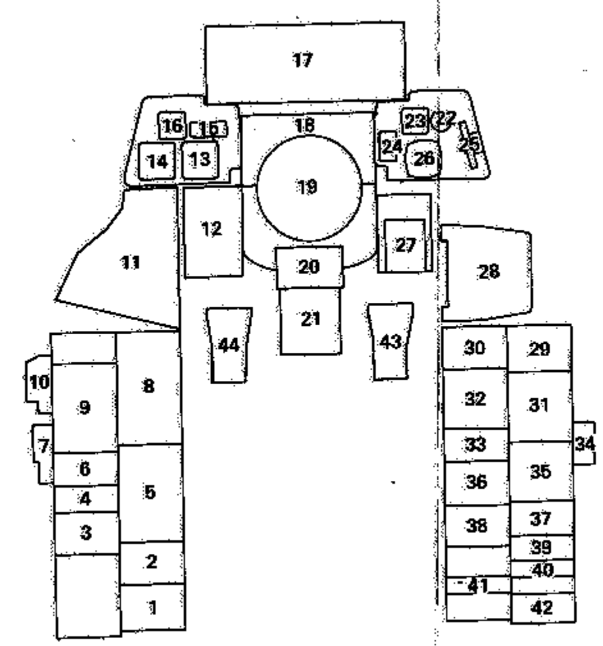
RIGHT SIDE CONSOLE
54. COMPASS CONTROL PANEL
55. CAUTION—ADVISORY INDICATOR
56. TACAN CONTROL PANEL
57. MASTER GENERATOR CONTROL PANEL
58. ARA-63 CONTROL PANEL
59. AIR CONDITIONING CONTROL PANEL
60. MASTER LIGHT CONTROL PANEL
61. MASTER TEST PANEL
62. DEFOG CONTROL LEVER

1 F050 004
020 0

Figure 1-3.

NFO INSTRUMENT PANEL AND CONSOLES





LEFT SIDE CONSOLE
1. G VALVE PUSHBUTTON
2. OXYGEN-VENT AIRFLOW CONTROL PANEL
3. COMM/NAV COMMAND PANEL
4. ICS CONTROL PANEL
5. INTEGRATED CONTROL PANEL
6. TACAN CONTROL PANEL
7. LIQUID COOLING CONTROL PANEL
8. COMPUTER ADDRESS PANEL
9. RADAR IR/TV CONTROL PANEL
10. EJECT COMMAND PANEL

LEFT VERTICAL CONSOLE
11. ARMAMENT PANEL

LEFT KNEE PANEL
12. SYSTEM TEST—SYSTEM POWER PANEL

LEFT INSTRUMENT PANEL
13. SERVOPNEUMATIC ALTIMETER
14. AIRSPEED MACH INDICATOR
15. UHF REMOTE INDICATOR
16. STANDBY ATTITUDE INDICATOR

CENTER PANEL
17. DETAIL DATA DISPLAY PANEL (DDD)

CENTER CONSOLE
18. NAVIGATION CONTROL AND DATA READOUT
19. TACTICAL INFORMATION DISPLAY (TID)
20. TACTICAL INFORMATION CONTROL PANEL
21. HAND CONTROL UNIT

RIGHT INSTRUMENT PANEL
22. FUEL QUANTITY TOTALIZER
23. CLOCK
24. THREAT ADVISORY LIGHTS
25. CANOPY JETTISON HANDLE
26. BEARING DISTANCE HEADING INDICATOR (BDHI)

RIGHT KNEE PANEL
27. CAUTION-ADVISORY PANEL

RIGHT VERTICAL CONSOLE
28. MULTIPLE DISPLAY INDICATOR

RIGHT SIDE CONSOLE
29. DIGITAL DATA INDICATOR (DDI)
30. ECM DISPLAY CONTROL PANEL
31. DATA LINK REPLY AND INTERIOR LIGHT CONTROL PANEL
32. ECM CONTROL PANEL
33. DECM CONTROL PANEL
34. DEFOG CONTROL LEVER
35. IFF TRANSPONDER CONTROL PANEL
36. CHAFF/FLARE DISPENSE PANEL
37. AA1 CONTROL PANEL
38. AN/ALE-29A PROGRAMMER
39. IFF ANTENNA ANT AND TEST PANEL
40. RADAR BEACON CONTROL PANEL
41. KY-28 CONTROL PANEL
42. ELECTRICAL POWER SYSTEM TEST PANEL

LEFT AND RIGHT FOOT WELLS
43. MIC FOOT BUTTON
44. ICS FOOT BUTTON

1-F050-004
021-0

Figure 1-4.

(U) The forward cockpit is arranged and equipped for the pilot. In addition to three multipurpose electronic displays for viewing tactical, flight, navigational, and ECM data, the pilot's instrument panel also contains armament controls, flight and engine instruments. Engine controls, fuel management, auxiliary devices, autopilot and communications control panels are located on the left console. Display, power, lighting, and environmental controls are on the right console.

(U) The aft cockpit of the tandem arrangement is equipped for the NFO. This instrument panel contains controls and displays for the AN/AWG-9 Airborne Weapon Control Systems, navigational and flight instruments. Armament controls, sensor controls, keyboard panels, and communications panels are located on the left console. The right console contains an ECM/navigational display, ECM controls, data link controls and lighting, and the IFF panel. Refer to figures 1-3 and 1-4 for illustrations of cockpit arrangements.

part 2 Systems

AIR INLET CONTROL SYSTEM

INTRODUCTION

(U) The air inlet control system (AICS) consists of two primary air inlets, one on each side of the fuselage at the intersection of the wing glove and fuselage. The rectangular cross-sectional inlet sidewalls are spaced away from the fuselage to minimize forebode boundary layer air ingestion and are highly raked to optimize operation for high angle of attack conditions. Variable inlet geometry is provided by automatically positioned ramps on the upper side of the inlets to control the two-dimensional external airflow compression field, airflow contraction at the throat, boundary layer air bypass and diffuser duct geometry. Downstream of the inlet throat the subsonic diffuser section directs air straight aft to the engine compressor face. A bleed door on top of the glove allows the inlet to be self-compensating to some degree across the aircraft speed regime. Inlet ramps and bleed doors are positioned by electro-hydraulic actuators which respond to fixed schedules in the AICS programmers. The purpose of the AICS is to decelerate free stream air in flight at an optimum rate to provide high pressure recovery at the engine compressor face for maximizing installed thrust and minimizing aircraft drag while providing high quality, matched airflow for compatible propulsion system operation throughout the aircraft flight envelope. Separate probes, sensors, AICS programmers, actuators and hydraulic power systems provide for completely independent operation of the left and right air inlet control systems. Extensive electronic monitoring is provided within the AICS to detect failures that would degrade system operation and performance. As a result, no pilot control is required during normal modes of operation. AICS malfunction lights, test controls and overriding pilot controls for failure modes of operation are illustrated in figure 1-5. Electrical power supply for AICS operation is derived from essential ac/dc busses and hydraulic power is supplied individually from the two main hydraulic systems.

NORMAL MODE OF OPERATION

(U) To obtain high propulsive efficiency and airflow compatibility for the turbofan engine it is necessary that the pressure and flow across the face of the compressor be as uniform as possible under all attainable operating conditions. Variable geometry provides for scheduling the inlet shape as a function of aircraft Mach number so that the inlet duct operates with maximum attainable efficiency at all flight conditions. Sectional side views of representative variable geometry inlet configurations with descriptive nomenclature

AICS CONTROLS AND INDICATORS

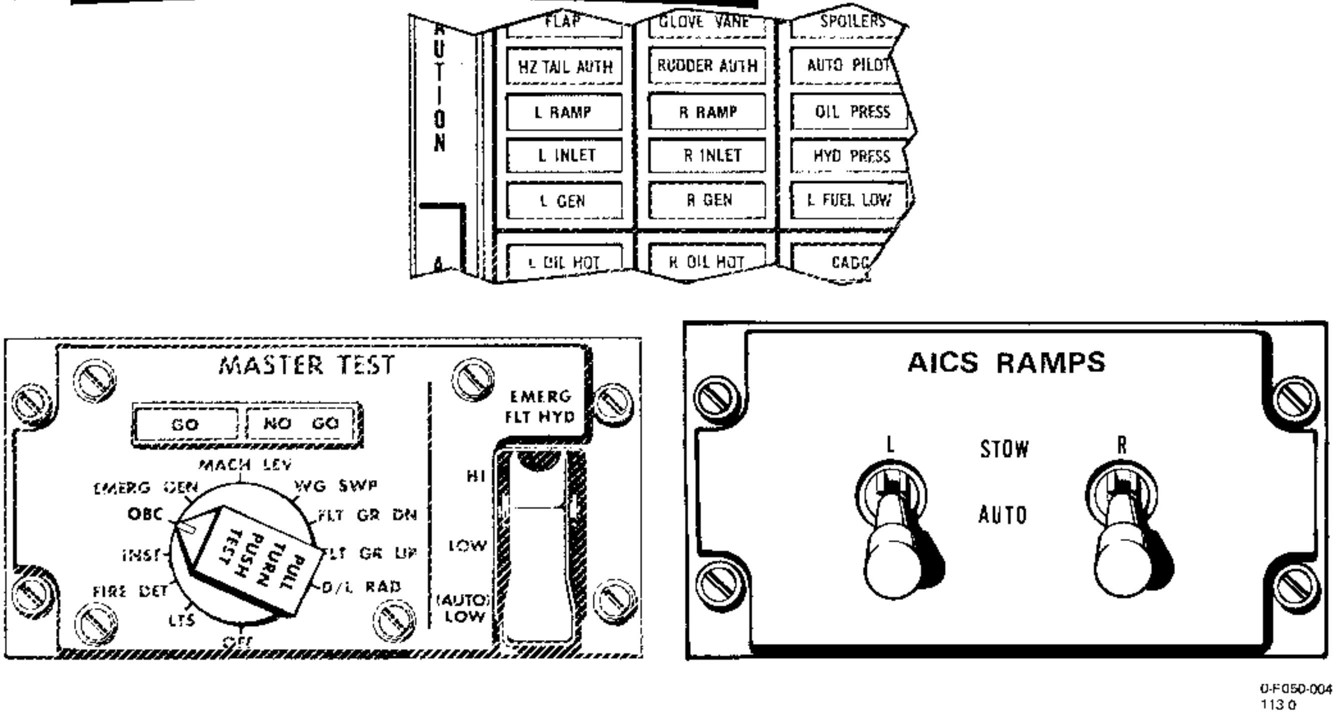


Figure 1-5.

are presented in figure 1-6. Inlet geometry is varied by automatically controlled hinged ramps on the upper side of the inlets which are independently positioned to decelerate the air efficiently, i.e., control the formation of shock waves in the external compression field and/or to regulate capture area at the throat. The first (#1) movable ramp is laterally hinged on its leading edge to the fixed ramp structure. The hinge line for the second (#2) movable ramp provides a direct juncture for the trailing edge of the first ramp to the leading edge of the second ramp; therefore, the second ramp position is directly related to the first ramp position. The forward facing third (#3 or diffuser) movable ramp is hinged on the trailing edge to the duct structure and forms the subsonic diffuser as well as defining throat height in conjunction with the second ramp. A forward hinged, two-positioned bleed door with side plates provides for a wide range of airflow self regulation in conjunction with the throat slot between the trailing edge of the second ramp and the leading edge of the third ramp.

(U) During ground static and low speed (Mach < 0.2) conditions the inlet ramps are mechanically restrained in the stowed (retracted) position and the bleed door is closed (100 sq. in. exit area) to provide a fixed geometry configuration with the ramps independent of the hydraulic pressure source. The predominant airflow activity is concentrated about the station defined by the inlet lower lip and supplemented by reverse airflow through the bleed door around the forward lip of the third ramp. As flight speed is increased to 0.35 Mach hydraulic power is ported to the ramp actuators, but the ramps are not scheduled out of the

VARIABLE GEOMETRY INLET CONFIGURATION

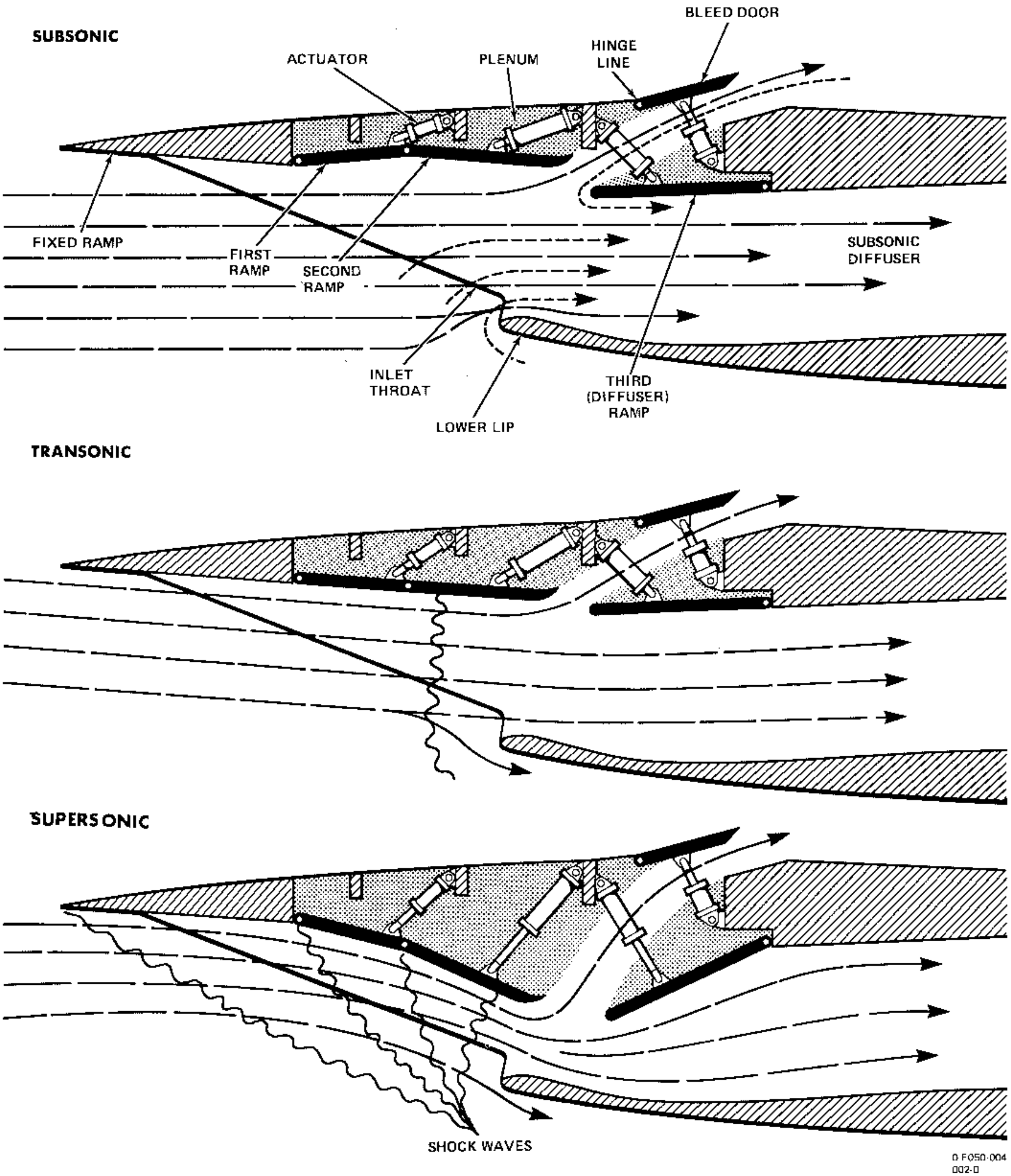


Figure 1-6.

stowed position until 0.50 Mach in response to the fixed Mach schedule presented in figure 1-7. The throat slot now serves to bleed off low energy boundary layer air from the fixed and two movable ramps upstream and discharge the bypass airflow over the top deck. The airstream flow converges at the primary inlet in subsonic flight with no spillage at the lower lip. At transonic speeds a normal shock wave attaches to the second movable ramp which marks the formation of the external compression field characteristic of supersonic flight. The third ramp deflects in conjunction with the first two movable ramps to maintain a nominal throat slot height for transonic and low supersonic flight speeds. At supersonic flight speeds the throat slot assumes an additional function as the bypass of air induces a turning action to the airstream downstream of the throat to permit a wider range of diffuser ramp deflections without inducing airflow separation on the back side. At high supersonic speeds the inlet propagates three predominant oblique shock waves in the external compression field converging with the normal shock wave ahead of the inlet lower lip. The first oblique shock wave emanates from the fixed ramp leading edge while the second and third oblique shocks are generated by the first and second ramp hinge line discontinuities, respectively. Although the four shock waves do not intercept at the inlet lower lip for maximum thermodynamic efficiency, the trade-off for slightly increased spillage drag with a more simple control schedule and inherent degree of self regulation was adopted. The degree of control exercised over inlet throat area ranges from________ sq. in. at minimum Mach (Mach < 0.35) to_________ sq in. at 2.5 Mach is equivalent to an area regulation factor of______: 1. The basic shcedule for bleed door operation is also directly related to aircraft Mach number but is additionally biased by angle of attack (discrete and rate) and engine

INLET RAMP SCHEDULE

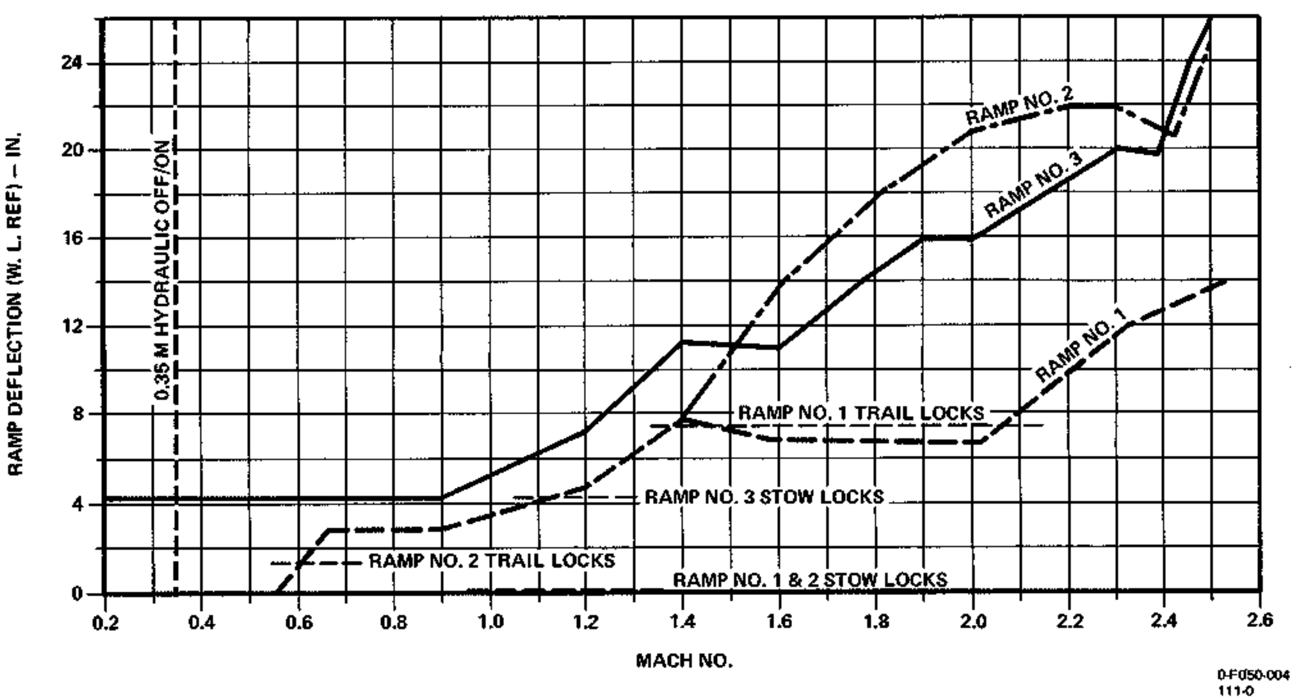


Figure 1-7.

airflow variables. Because of the modifying functions to the Mach governed bleed door schedule, a significantly higher actuator slew rate is provided to meet response rate requirements. Identification of boundaries for the two-positioned bleed door is provided in figure 1-8. A summary of the AICS normal mode of operation is presented in Table I.

AICS CONTROL

(U) Control of the AICS is completely automatic for normal and initial failure modes of operation. Each inlet has a self-controlled system which is completely independent from the probes, sensors, AICS programmers, and movable surface actuators to the hydraulic power systems. Because of differences in controlling functions, control of the AICS ramps and bleed doors will be discussed separately. A functional schematic of the AICS is presented in figure 1-9.

Ramp Control

(U) The ramps which are positioned by electro-hydraulic actuators are controlled solely as a function of aircraft Mach number according to a fixed schedule in the AICS programmer. Pitot and static pressures tapped by freestream probes located on each side of the forward fuselage are converted by sensors to electrical signals which are transmitted to the AICS programmer on the respective side. Although the ratio of these two values can be equated to aircraft Mach number, angle of attack is also fed to the AICS programmer to provide for probe pressure defect correction in the Mach number calculation. Local angle of attack is derived from differential pressure sensed between two angular spaced total sources in freestream alpha probes located on either side of the forward fuselage. The differential pressure is converted to proportional voltage equivalent using a differential transformer. Knowing the basic impact pressure (P_t) applied to the probe, the angle of attack equation may be solved in the AICS programmer. Within the AICS programmer the electrical input signals are fed to a multiplexer which serves as a commutating section that passes only one input in a given time frame to the computer via an analog-to-digital (A/D) converter. The A/D converter is a device for converting dc voltage into a digital word that is an acceptable format for the special purpose digital computer. Based on the Mach number calculation in the computer, specific ramp actuator position commands in digital word form are directed to the digital-to-analog (D/A) converter which transmits the commands in analog form to sum joints (Σ) where they are compared to actuator feedback position commands. As an integral part of each actuator a feedback transformer provides a position reference to the sum joint in the AICS programmer. If the commanded position is different than the actual position, the respective servo amplifier (∇) will detect this error and command an actuator position change accordingly to null the error. Each actuator contains a servo valve which directs hydraulic pressure to extend or retract the actuator in response to servo amplifier electrical commands and additionally implements fail-safe operation. In addition to variable ramp actuator position commands, the AICS programmer output signals control the hydraulic shut-off valve. At Mach numbers less than 0.35 M the hydraulic valve, which controls the supply of hydraulic motive power to the three ramp actuators on the respective side, is closed and the ramps are mechanically restrained by internal finger latches in the actuators at the stowed (fully retracted) position. At speeds greater than 0.35 M the programmer provides a signal to open the hydraulic shut-off valve which ports pressurized fluid to the actuators to unlock the finger latches and enable motive response to position commands from the programmer. Hydraulic power supplied to the left and right inlet actuators is derived from the combined and flight hydraulic systems, respectively.

INLET BLEED DOOR SCHEDULE

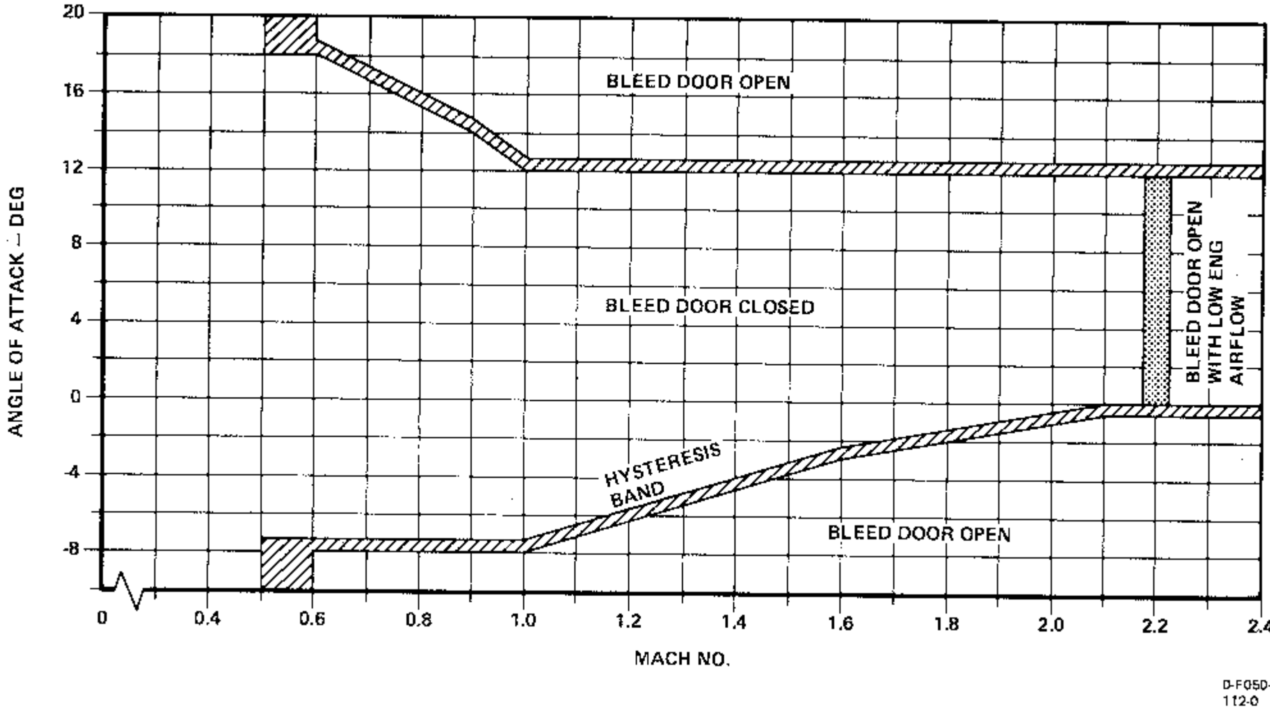


Figure 1-8.

Bleed Door Control

(U) Although bleed door actuator position is controlled by more functions than the ramp actuators, bleed door commands provide for either a closed (100 sq. in. exit area) or open (150 sq in. exit area) discrete position. Thus, there is no need for an actuator feedback transformer and a single microswitch on the door is used to provide the feedback discrete. In addition to Mach number the bleed door is scheduled as a function of angle of attack, rate of angle of attack change, and engine corrected airflow (resultant of total temperature and engine low pressure rotor speed). Corrections to indicated values of angle of attack and total temperature are applied within the AICS programmer to compensate for freestream probe defects. Rate of change of angle of attack logic is used to modify the discrete angle of attack switching boundaries to provide an anticipation factor during high pitch rate maneuvering and compensate for inherent hardware and computation delays. Coincident with bleed door discrete open-close commands the AICS programmer also provides discrete open-close commands for control of the engine mid-compression bypass (MCB) bleed valves. Open commands to the bleed door and MCB are provided at extreme angle of attack and airflow conditions to maintain compatibility, (i.e., engine stall margin) at flight conditions where the quality of inlet airflow is degraded. Normally the bleed door is closed and is commanded to the open position only above 0.5 Mach as a vernier function of the modifying parameters. The position of the AICS hydraulic shut-off valve does not affect bleed door

Table I. AICS Normal Mode of Operation

FLIGHT CONDITION	HYDRAULIC POWER		ACTUATOR POSITION				AICS→MCB COMMAND
	RAMP ACTUATORS	BLEED DOOR ACTUATOR	RAMP #1	RAMP #2	RAMP #3	BLEED DOOR	
$M < 0.35$	OFF	ON	Mechanically restrained by finger locks in stowed position; electrical stow commands output from AICS programmer.			CLOSED	CLOSED
$.35 < M < 0.5$	ON	ON	Electrical stow commands output from AICS programmer.			CLOSED	CLOSED
$.5 < M < 1.2$	ON	ON	Variable position scheduled by AICS programmer as a function of Mach number.			CLOSED, or OPEN with exceedance of angle of attack boundary.	
$M > 2.2$	ON	ON	Variable position scheduled by AICS programmer as a function of Mach number.			CLOSED, or OPEN with exceedance of angle of attack boundary and/or minimum engine airflow condition.	

AICS CONTROL SYSTEM

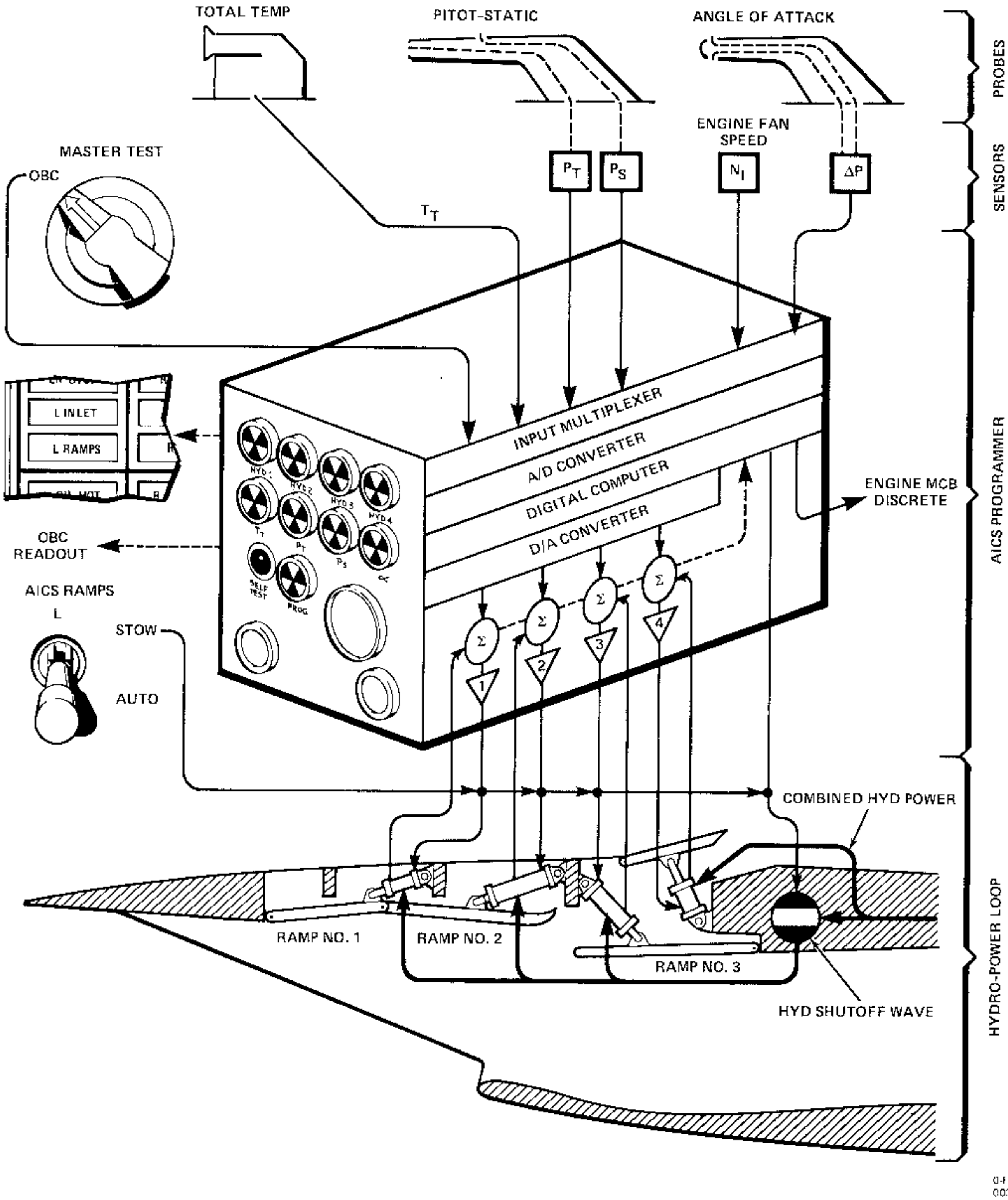


Figure 1-9.

operation as its source of hydraulic power is derived from the upstream side of the valve. Subsequent to engine post-flight shutdown the bleed doors open with the loss of electrical power in the presence of residual hydraulic pressure.

AICS TEST

(U) Two types of AICS tests are provided to check the general "health" of the air inlet control system and pinpoint detected failures to the appropriate system component.

Built-in-Test

(U) Built-in-test (BIT) is included in the AICS computer program and is automatically initiated within the programmer every fifth frame in the program cycle to check the following components when the programmer is energized. If a failure is detected, BIT derates its attention to that failure repeatedly until the failure clears or exceeds its time limit (trips failure mode of operation).

- (U) Sensor input values (total pressure, static pressure, angle of attack, total temperature and engine fan speed) to the AICS programmer are checked for exceedance of preset high and low limits. There must be five consecutive failure indications in program sequence before a failure output is developed. In each case a detected failure will be registered by tripping the appropriate flag (latching) on the face of the AICS programmer and by inflight performance monitoring displays on the HSD/TID. The operational status of the AICS will differ depending on the detected sensor(s) failure. Since Mach number is the fundamental value in the AICS programmer computations, a detected failure in total or static pressure input values will cause the AICS to automatically transfer to the fail-safe mode of operation. Conversely, a detected failure in angle of attack, total temperature or engine fan speed input signals will cause a nominal value to be substituted for the failed value in the AICS programmer. However, the normal operational mode of the AICS prevails except for the trimming action of the bleed door.

- (U) Programmer electronics are checked for value exceedance and an output to input comparison. Detection of a failure in either test for a consecutive period of 0.5 seconds will generate an AICS programmer failure and cause the AICS to automatically revert to the fail-safe mode of operation.

- (U) Hydro-power loop status from the output of the D/A converter to the actuator feedback position signal is compared for agreement in the three variable ramp control loops as well as a discrete check of bleed door actuator position. A consecutive detected disagreement in ramp actuator position above an established threshold for a period of 0.5 seconds will generate a failure and cause the AICS to automatically revert to the fail-safe mode of operation. An apparent bleed door actuator control loop discrete failure will trip a bleed door failure which allows the AICS to operate in a fail operational status.

On-Board Checkout

(U) On-board checkout (OBC) is a manually initiated self test program that can be performed on the deck with electric and hydraulic power energized. The pilot initiates the self test as part of the pre-taxi check list to obtain a quick operational check of the air inlet control system prior to flight. Additionally, on the face of the AICS programmer the self test can be initiated for the convenience of maintenance personnel without the need to satisfy remote test interlocks. The following specific conditions must be satisfied before a remote self test can be initiated at the pilot's station:

- Throttle(s) - IDLE or OFF
- Weight-on-wheels
- Mach < 0.15
- Engine fan rotor speed < 6000 RPM

(U) Pilot control is accomplished by rotating the Master Test Knob in the raised position to the OBC position and depressing the knob to initiate the OBC sequence. Concurrent with self test of the left and right AICS, a similar form of self test is conducted for the automatic flight control system (AFCS), auto throttle and central air data computer (CADC). The regular AICS BIT program plus the following additional checks are performed in the OBC program:

- Sensor input value checks for total pressure equal to static pressure and no angle of attack sensor differential pressure is made since velocity and velocity angle will be zero. A detected failure in pitot-static pressures will cause the AICS to revert to the fail-safe mode of operation whereas an angle of attack failure will provide a fail operational status.

- Programmer electronics is checked by pseudo total and static pressure signals inserted in place of regular sensor values. The program varies the pseudo pressures upwards to a value equivalent to 2.5 Mach (maximum ramp extension stroke) at 30,000 ft. altitude and back to static sea level conditions in an approximate 65 second time frame. A detected failure will cause the AICS to revert to the fail safe mode.

- Hydro-power loop status from the hydraulic shutoff valve to the ramp and bleed door actuators are exercised through the complete schedule as the AICS programmer pseudo pressures are varied to simulate a flight sequence to limit speed conditions and back to static sea level conditions. Cycling of the ramp and bleed door actuators through their full authority exercises the complete air inlet control system for preflight failure detection and permits the plane captain to check for external evidence of hydraulic leaks in the inlet area prior to taxiing. If a hydro-power loop occurs during the OBC exercise the AICS will revert to the fail-safe mode.

FAILURE MODES OF OPERATION

Detected Failures

(U) Failures in the air inlet control system are normally detected by the AICS programmer which automatically initiates appropriate initial corrective action. AICS detected failure modes of operation may either be fail operational or fail safe as delineated in Table II. The fail operational mode results in no significant degradation in AICS operating characteristics and the mission can be continued without pilot corrective action. The fail safe mode results in degraded AICS characteristics which restrict mission completion to speeds less than 1.2 Mach. In addition to providing for illumination of the appropriate INLET caution light and OBC legends in the event of a detected failure, the AICS programmer also outputs ramp actuator solenoid actuated fail valve commands, an AICS hydraulic shutoff valve close signal, and bleed door/MCB open discrete signals. The servo valve blocks the ramp actuator pressure and return ports upon sensing a loss of hydraulic pressure and responds to an electrically operated solenoid, which is energized with Mach < 1.2, to port both sides of the actuator to return so that the actuator is free-floating in response to air loads on the ramps. Electronic failures in the AICS programmer may be circumvented by pilot activation of the appropriate AICS RAMPS control switch to the STOW position which provides an electrical stow command directly to the ramp actuators while commanding the appropriate hydraulic shutoff valve open; such action should not be initiated at speeds greater than 1.2 Mach. In event of a hydraulic system pressure failure at speeds where one or more ramps actuators are extended out of the stowed position the stow position finger latches in such actuators will not restrain them from further extension. However, the first and second ramp actuators have an additional set of finger (trail) latches at the positions indicated in figure 1-9 which prevent further ramp actuator extension subsequent to a failure within the trail locks. The actuator stow and trail locks restrain actuator movement in tension only and may be engaged by aerodynamic blow-up of the ramps; hydraulic pressure (> 500 psi) is required to unlock the finger latches. Flight test results indicate that a safe ramp configuration (first ramp actuator trail or stowed, second ramp floating in or out of trail locks, and third ramp stowed) results due to ramp aerodynamic blow-up characteristics. In such case the engine may be operated unrestricted at speeds less than 0.95 Mach. A detected AICS failure will result in a slight degradation of cruise/excess thrust performance due to the off optimum configuration and MCB open at speeds greater than 0.35 Mach. A hydraulic system power failure allows the bleed door to float unrestricted between the open and closed positions independent of electrical commands.

Undetected Failures

(U) Failures in the air inlet control system which go undetected by the AICS programmer (because of error magnitude or multiple failures) will probably induce an engine stall at speeds greater than 1.2 Mach or, depending on the failure, even at speeds as low as 0.35 Mach. The occurrence of an engine stall is predicated on the incompatibility of the inlet ramp configuration, engine airflow demands and flight conditions. Additional caution lights (L/R RAMPS) are provided outside the AICS to warn the pilot of an otherwise undetected or detected failure at critical flight conditions (take-off and landing) and thus act as a double check of AICS performance monitoring. The lights also provide an inlet configuration status indication subsequent to the AICS programmer reverting to the fail safe mode of operation. Table III defines the logic which will cause individual illumination of the

Table II. AICS Detected Failure Modes of Operation

MODE	DETECTED FAILURE	FAILURE INDICATION: INLET CAUTION LIGHT	FAILURE INDICATION: OBC HSD/TID (1)	FAILURE INDICATION: BIT FLAG	CAUSE	RESULTANT
FAIL OPERATIONAL	ANGLE OF ATTACK	NO	AIC S	α	Sensor signal exceedance of established max/min limits.	Programmer substitutes +2° α constant. Bleed door and MCB open M > 0.35, closed M < 0.35.
	ENGINE FAN RPM		None, Pilot unable to initiate AICS OBC.			Programmer substitutes 6300 RPM constant.
	TOTAL TEMPERATURE		AIC S	T_T		Programmer substitutes 60°F constant.
	BLEED DOOR		AIC A4	HYD 4	(Bleed door not checked for failure 0.45M) Sustained error in actuator command-feedback position signal comparison	M < 0.35 Bleed door and MCB closed. M > 0.35 Bleed door and MCB open.
FAIL SAFE	HYDRO LOOP: RAMP #1	YES	AIC A1	HYD 1		• Hyd shutoff valve-CLOSED • Bleed door OPEN M > 0.35, CLOSED M < 0.35 • M < 0.5: Fail mode valve remains locked. Ramp actuators remain mechanically restrained within stow locks. (3) .5 < M < 1.2: Fail mode valve dumps hyd pressure to return and ramps float in response to aero loads. (3) (4) M > 1.2: Fail mode valve locks and ramps remain in last commanded position. Engine stall may occur. (3) (4)
	HYDRO LOOP: RAMP #2		AIC A2	HYD 2		
	HYDRO LOOP: RAMP #3		AIC A3	HYD 3		
	AICS PROGRAMMER		AIC P	PROG	Failure in end-to-end electronic BIT	
	TOTAL PRESSURE		AIC S	P_T	Sensor signal exceedance of established max/min limits. (2)	
	STATIC PRESSURE		AIC S	P_S		
	HYDRAULIC PRESSURE	M > 0.5 YES	Any combination of AIC A1, A2, A3 or A4.	Any combination of HYD 1, 2, 3 or 4.	Error signal generated by failure of ramp actuator(s) to respond to commands or ramp float.	Bleed door floats
		M < 0.5 NO	Fail-safe mode not generated until exceedance of 0.5 Mach. Bleed door may float open for fail operational mode.			

Notes:
(1) AIC symbol has L or R appended (AICL, AICR) to identify on which side failure was detected.
(2) Rate of change of total and static pressure signal is also checked. Exceedance of established max limit will cause ramps to hold last commanded valve (passive Mach failure) until signal rate of change diminishes below limit rate. If signal rate exceedance persists, max/min limit will be encountered - prevents hardover failure.
(3) At M < 1.2 pilot should actuate AICS RAMP switch to STOW position which opens the hyd valve and provides an overriding command to stow the ramp actuators.
(4) Subsequent operation of the aircraft is speed restricted.

L/R RAMPS caution lights. Pilot corrective action consists of actuating the applicable AICS RAMPS control switch to the STOW position which overrides the AICS programmer output signals and provides an electrical stow command directly to the ramp actuators as well as a hydraulic shutoff valve open signal. As a pre-flight check of RAMP light operational status the RAMP caution lights should be illuminated during the entire AICS OBC with the AICS RAMP control switch in the AUTO position.

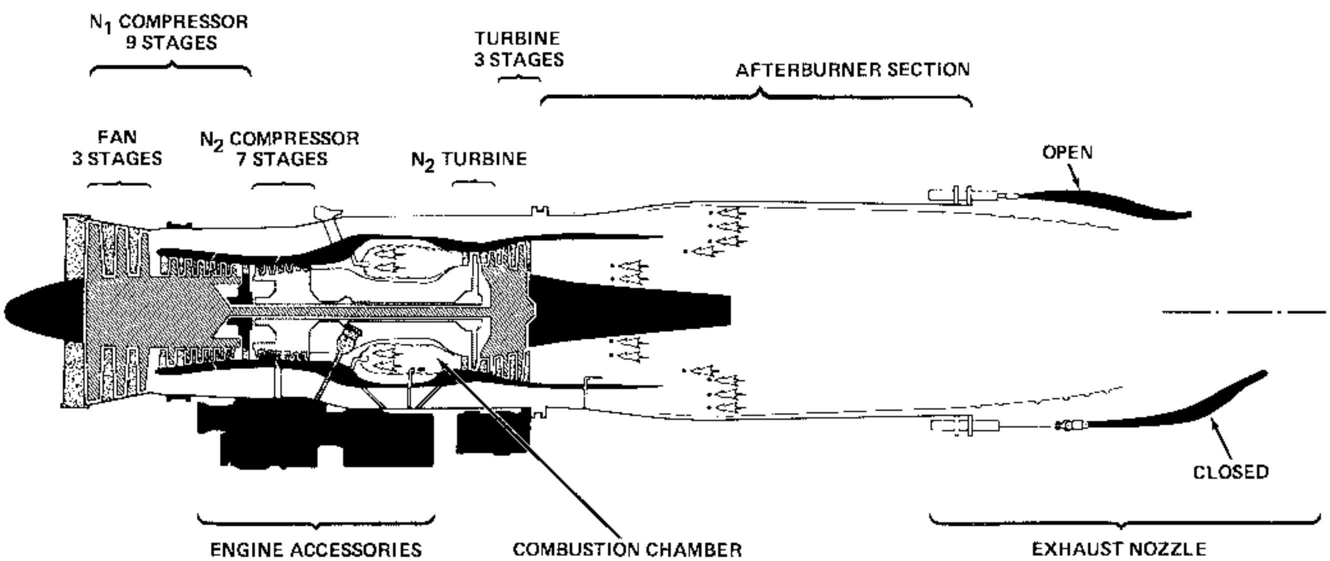
Table III. Ramps Caution Light Logic

To be supplied when available.

ENGINES

(C) The aircraft is powered by two Pratt & Whitney TF30-P-412, dual axial flow compressor, turbofan engines (figure 1-10) equipped with an afterburner for thrust augmentation. At sea level, static conditions, each engine develops 10,500 pounds installed thrust at military and 1700 pounds thrust at maximum afterburner. A low thrust specific fuel consumption is characteristic of basic engine turbofan operation, however significantly higher levels of thrust requiring afterburner operation are less efficient than contemporary turbojet engines. Primary airflow is directed straight aft through the inlet duct, engine and exhaust to maximize propulsive efficiency. Each engine is slung in a nacelle with the thrust axis laterally offset approximately 4 1/2 feet from the aircraft centerline. The engine mates to the inlet duct at the main landing gear bulkhead which forms the forward firewall. Side-hinged, split engine access doors form the lower nacelle structure and provide access to the lower hemisphere of the engine from the compressors aft to the afterburner section.

ENGINE TF30-P-412



1-F050-004
024-0

Figure 1-10. (U)

The lower aft most portion of the nacelle need only be lowered for engine removal and encompasses the drop-out frame for the canted bulkhead containing the horizontal tail pivot. The build-up of engine components is identical regardless of which side the engine is installed. Engine suspension in the nacelles is provided by three mounts. The thrust mount is located forward on the outboard side of the nacelle and the two aft engine mounts which carry only vertical and horizontal loads attach just forward of the engine nozzle shroud. The ring clamp mating the engine to the inlet can take engine thrust loads in an emergency.

(U) The engine forward fan section is composed of three fan stages secured to the front of the low pressure compressor. The fan and the low pressure compressor rotor are driven as a single rotating unit by the last three turbine stages. Fan discharge air divides into two separate streams with approximately 54% of the airflow directed into the low pressure compressor and the remaining air flows through the full-annular fan duct, by-passing the basic engine, to mix with basic engine airflow in the afterburner duct downstream of the turbine discharge. The forward fan develops approximately 45% of the total basic engine thrust. The core of fan discharge air is further compressed by six additional stages of axial compression making the front or low pressure compressor consist of nine stages. Air discharged from the low pressure compressor is directed aft through seven stages of high pressure compression, which unit is driven through concentric shafting by the first stage turbine. Characteristic of twin spool engines, the low and high pressure compressors are completely free to rotate with no mechanical connection with each other. The high pressure compressor rotor is "speed-governed" by the engine main fuel control and the low pressure compressor is rotated by the three turbine stages at whatever speed will ensure optimum flow through the compressor. Flow matching is achieved between the compressors and turbines as the front and rear rotors work in harmony with a minimum of interstage bleed to prevent a compressor stall or surge. The power take-off for engine accessory drive is geared to the high pressure compressor rotor. Sixteenth stage compressed air discharged from the high pressure compressor is used to support combustion and diverted for air conditioning and ancillary purposes. In the combustion section of the basic engine, compressed air and fuel are mixed and burned within eight can-annular type combustion cans each of which contains four fuel ejector nozzles. Engine start ignition is provided by a dual ignition system with a spark ignitor located in each of the two bottom combustion chambers. Cross-ignition tubes provide for flame propagation between combustion cans. The burning gases are directed rearward from the combustion section through the split, four stage turbine where some 60% of the available energy is converted into torque to drive the engine compressors and accessories. The fan discharge air and basic engine turbine exhaust gases are mixed in the forward section of the afterburner duct. Afterburner fuel is injected through five circular manifolds with zones No. 1 and No. 4 located in the turbine exhaust core and zones No. 2, No. 3 and No. 5 positioned in the fan discharge stream. Afterburner ignition is initiated by a dual hot streak of overrich fuel mixture torching through the turbine stages to the Zone No. 1 manifold and flame holder. With the annular fan duct serving as an insulation shroud for the basic engine, fan discharge airflow serves to cool the liner in the afterburner duct and exhaust nozzle flaps. Turbine expansion pressure ratio and exhaust gas flow for both basic engine and afterburner operation are controlled by a variable area, convergent-divergent, exhaust nozzle. The nozzle is operated by four fuel pressure operated actuators which drive a unison ring to position the nozzle flaps. For basic engine operation, the nozzle is closed to the minimum area (except for ground idle operation to reduce residual thrust), and in afterburning, the nozzle area is infinitely variable to a full open position which represents a 110% increase in exit area.

THROTTLE CONTROL

(U) Two throttle levers for regulating engine thrust are located on the left console of the forward cockpit. Unrestricted engine operation under independent control is afforded, however, normal symmetric thrust control is provided by collective movement of the throttle levers with the grips matched. Numerous engine control and subsidiary functions are performed by movement of the throttle levers within the full range of travel as illustrated in figure 1-11. The forward and aft throw of each throttle lever in the quadrant is restricted by hard detents at the OFF, IDLE, MIL and MAX positions which provide a hand feeling sense of throttle limits of travel within the cut-off, basic engine and afterburner augmentation sectors of operation. At the OFF and IDLE detents the throttles are spring loaded to the inboard position to inhibit inadvertent operation, whereas at the MIL detent a gear shift mechanism allows the throttles to be shifted outboard to the afterburner augmentation sector or inboard to the basic engine sector of operation by merely overcoming a lateral breakout force. Lateral shifting of the throttles at the MIL detent does not affect engine control so that placement of the throttles outboard at the MIL position provides a natural catapult detent to prevent inadvertent retardation of the throttles during the launch. This however, does not inhibit the selection of afterburner thrust augmentation in emergency situations. A friction control lever is mounted on the outboard side of the quadrant to

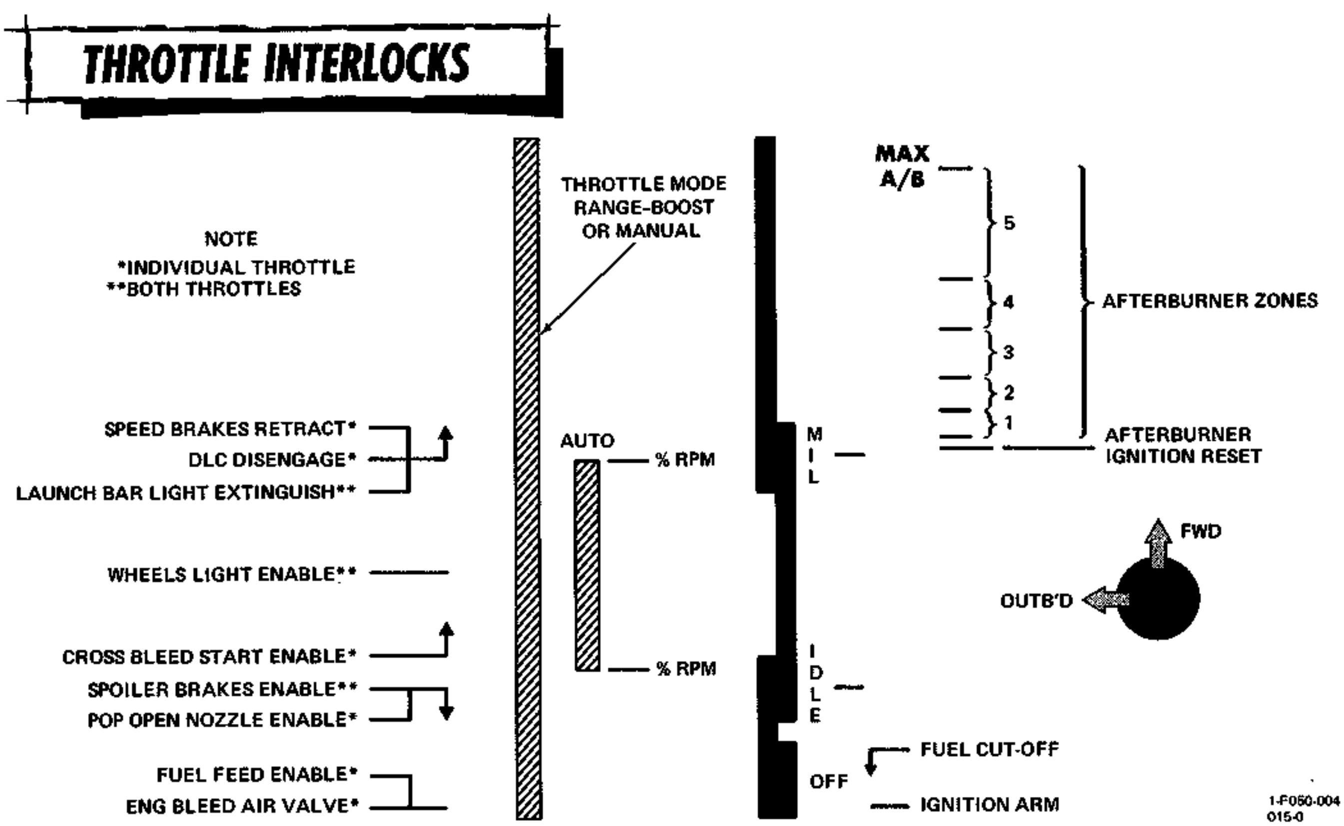


Figure 1-11.

permit adjustment of throttle friction to suit individual requirements. With the friction lever in the full aft position, no throttle friction is applied at the quadrant; increased throttle friction is obtained by forward movement of the lever.

THROTTLE CONTROL MODES

(U) Manual, Boost and Auto are the three modes of throttle control over engine operation selectable by the throttle mode switch (MAN, BOOST, AUTO positions) located outboard of the quadrant on the pilot's left console. A functional schematic of throttle control modes, including system major components, is illustrated in figure 1-12. Except for the auto throttle computer and mode control switch, the throttle control system for each engine is completely redundant. Independent engine operation is possible in the manual or boost mode of throttle control, however full system operation is necessary in the auto mode since operation under single engine control can be impracticable due to asymmetric thrust considerations.

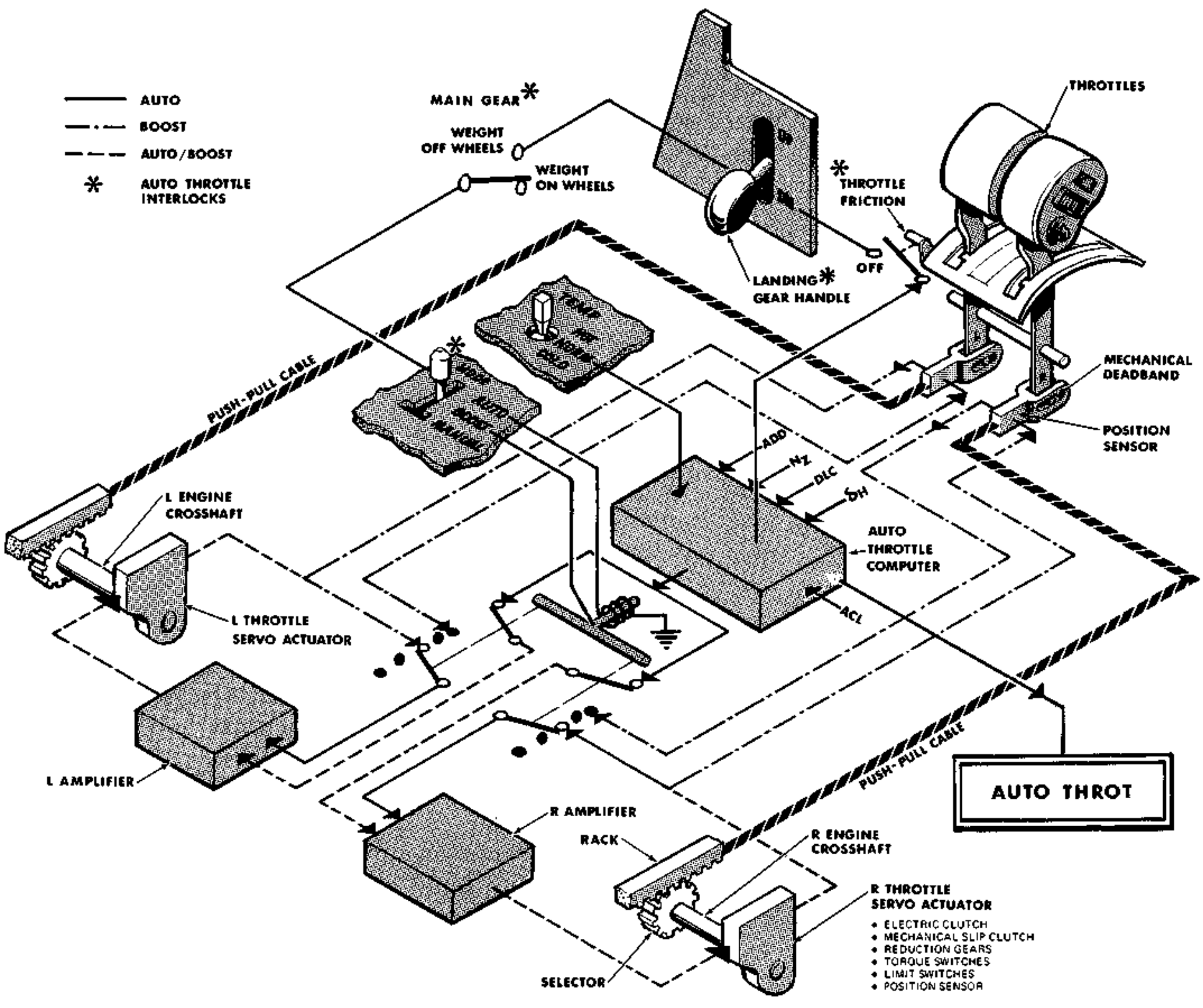
MANUAL THROTTLE

(U) In the manual mode of operation movement of each throttle is mechanically transmitted to the respective engine cross-shaft by a push-pull cable. A rack and sector mechanism converts cable translation into rotary movement of the cross-shaft. As the primary control interface device for the engine, the cross-shaft is mechanically linked to the engine main and afterburner fuel controls and the exhaust nozzle control unit. The electric clutch in the throttle servo actuator which is splined to the cross-shaft is disengaged in the manual mode to reduce operating forces by that amount necessary to back-drive the actuator. With the throttle friction lever full aft in the OFF position, approximately 6 pounds of force per throttle must be applied at the grip to operate the throttles in the IDLE to MAX range; frictional forces increase to 9 pounds in the IDLE to OFF range.

THROTTLE BOOST

(U) The boost mode of throttle control is provided for normal operations to decrease the required effort necessary to overcome operating forces of the manual control system. In this mode a force of 1 to 2 pounds at the grip is required to move each throttle through all sectors with the friction lever in the OFF position. Response rate of the boost system is compatible with abrupt throttle manipulations without undue increases in throttle frictional forces, yet it provides satisfactory positional accuracy for effecting small adjustments in engine thrust settings. Essentially the boost mode provides electric throttle operation with the push-pull cable serving as a back-up control path. A mechanical deadband at the throttle allows the throttle position transducer to detect a commanded change and follow-up before the deadband limits are exceeded. Throttle movement produces an error signal between the throttle position transducer and its counterpart in the electro-mechanical servo actuator splined to the engine cross-shaft. The error signal is resolved in the amplifier to provide positional follow-up commands to the actuator to null the error. Movement of the actuator rotates the engine cross-shaft which back drives the push-pull cable to recenter the mechanical deadband. In the event of a boost system malfunction, exceeding a torque load at the actuator equivalent to 14 pounds at the throttle will automatically revert throttle control to the manual mode by disengagement of the actuator electric clutch. Additionally, a malfunction of the actuator which imposes a cross-shaft torque equivalent to 26 pounds at the throttle will cause a mechanical clutch in the actuator to slip so that pilot control

THROTTLE CONTROL



0-F050-004
014-0

Figure 1-12.

is overriding. There is no visible warning of these occurrences aside from the noticeable increase in forces necessary for manipulation of the affected throttle lever. Subsequently, the throttle mode switch must be cycled to MAN then BOOST to regain normal system operation (assuming the malfunction is not persistent).

AUTO THROTTLE

(U) The auto mode of throttle control is a closed-loop system which automatically regulates basic engine thrust to maintain the aircraft at an optimum approach angle-of-attack for landing. In this regard the auto throttle is compatible with the all weather-carrier landing system (AWCLS) and direct lift control (DLC) modes of flight path control besides pilot manual flight control. All components of the throttle control system except the throttle position transducer, (figure 1-12), are used in the auto mode of control. The angle-of-attack signal from the airstream direction detector (ADD) probe mounted on the left side of the forward fuselage is the controlling parameter within the auto throttle computer. However, additional parameters are integrated within the computer for providing anticipatory lead, compensation, damping, washout and gain changes to produce responsive but stable system operation under abrupt control, turbulent, modified control, and varied atmospheric temperature conditions. All changes to the computer inputs are automatically effected except for the air temperature switch. This switch is located on the pilot's left console outboard of the throttle quadrant and effects a computer gain change to compensate for engine thrust response characteristics as a function of air temperature. Outside air temperature ranges corresponding to the three switch positions are: COLD, less than 40°F; NORM, 40° to 80°F; and HOT, greater than 80°F. Electronic signals emanating from the computer are limited to a pre-set authority in the AUTO mode. Auto throttle controls are illustrated in figure 1-13.

(U) Engagement of the auto throttle cannot be accomplished with the throttles outside of the auto mode limits or until all controlling interlocks are satisfied. With all conditions met, subsequent movement of the throttle mode switch to the AUTO position will result in the switch remaining in the selected position by action of a holding solenoid. Throttle mismatch upon engagement of the auto mode will be automatically nulled. Engagement of the auto throttle permits computer commands to be transmitted to the two control amplifiers where the error relative to actuator position feed-back signals on the respective side is resolved. The resultant error in the amplifier comparator network produces follow-up commands to the actuator to null the error. Actuator authority is limited by internal switches and rotation of the cross-shaft back-drives the throttle to provide a tactile sense of auto throttle commands. The auto throttle reference angle-of-attack is automatically changed within the angle-of-attack indicator upon engagement or disengagement of DLC. Without DLC the approach angle-of-attack reference is 15 units and with DLC engaged the reference is 18 units to provide optimum landing approach speeds. In addition to changing the reference for the auto throttle, the change is also visually displayed by the angle-of-attack indicator, approach indexer, and external approach lights.

(U) The throttle control mode automatically reverts from the auto to the boost mode upon interruption of any interlock in the system or by manually overriding the throttles. A force of approximately 8 pounds per throttle in either direction results in auto throttle disengagement by action of the throttle torque switch(es). Secondarily, the actuator electric clutch disengages at a force of 14 pounds per throttle and the actuator mechanical clutch slips at 26 pounds overriding force at the throttle. Disengagement of the auto throttle by any means results in the throttle mode switch automatically returning to the BOOST position and illumination of the AUTO THROT caution light on the pilot's DDI panel. A time delay relay holds the AUTO THROT light illuminated for a 10 second duration following disengagement whereafter it is extinguished.

Auto Throttle Test

(U) An automatic check of the auto throttle control system while on-deck can be initiated by activating the master test switch at the OBC (On-Board Checkout) position. The test requires approximately 1 1/2 minutes to run its course during which time the auto throttle computer, AICS, ADC and AFCS are being concurrently self-tested. Signals to the servo actuator are inhibited during the auto throttle test so that a complete end-to-end check of the auto throttle control system is not achieved. This is necessary to enable the test to be performed with the engines at idle thrust on the deck. Because of this constraint, an end-to-end check of the auto throttle control system by manual control may be performed by rotating and depressing the master test knob in the FLT GR DN position which by-passes the auto-throttle weight-on-wheels interlock.

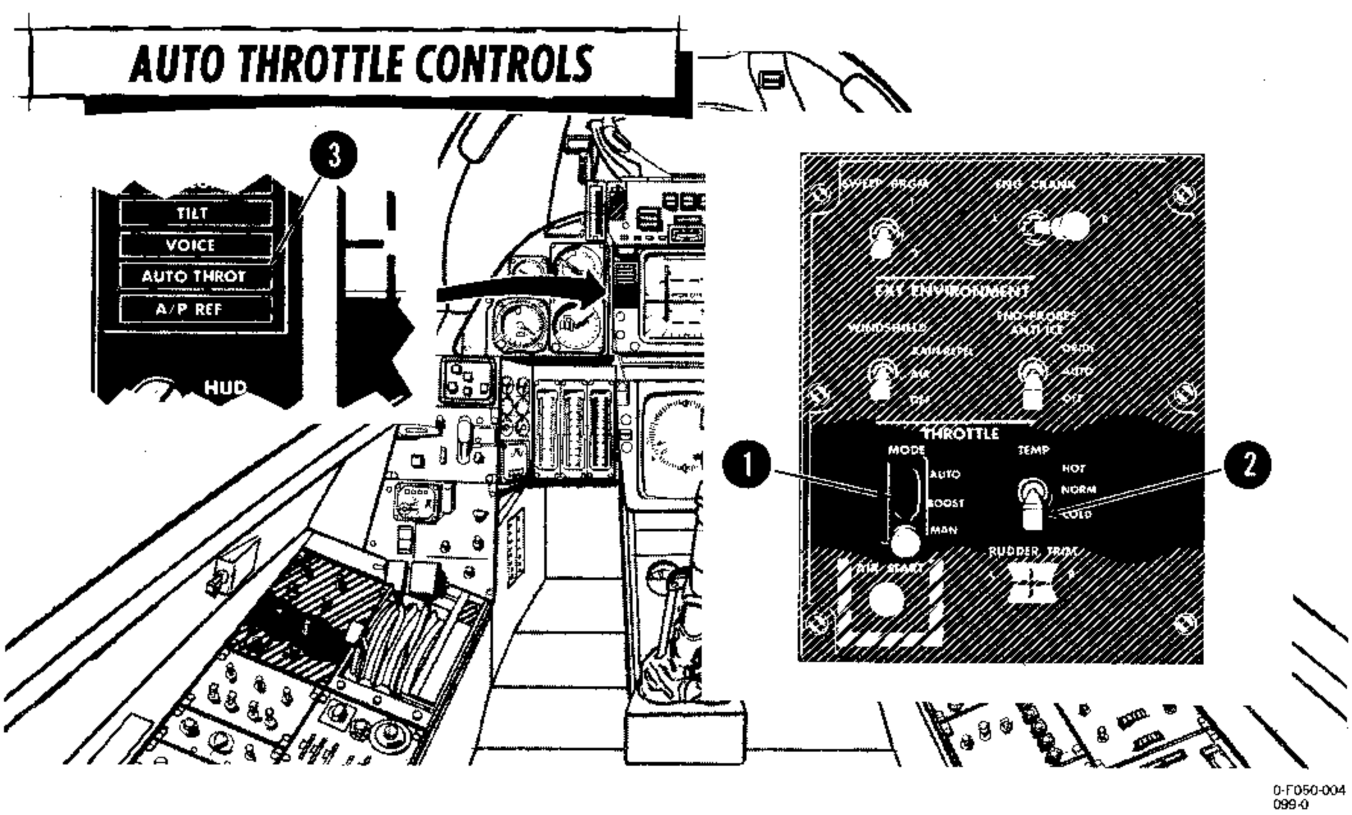


Figure 1-13.

AUTO THROTTLE CONTROLS

NOMENCLATURE	FUNCTION
(1) THROTTLE MODE SWITCH	AUTO - Engine thrust is automatically regulated to maintain optimum angle of attack for landing by the throttle control computer BOOST - Normal operating mode. Reduces effort required to move throttles manually with friction control aft. MAN - Movement of each throttle is mechanically transmitted to the respective engine cross-shaft by a push-pull cable.
(2) THROTTLE TEMP SWITCH	Used with the AUTO throttle mode to effect throttle computer gain changes to compensate for air temperature. HOT - Greater than 80°F NORM - Between 40° and 80°F COLD - Less than 40°F
(3) AUTO THROT LIGHT	Auto throttle mode is disengaged. Remains lit for 10 seconds, then is extinguished.

ENGINE FUEL SYSTEM

(U) The engine fuel system, which is identical for each engine, provides motive flow fuel to effect fuel transfer and metered fuel for combustion as a function of throttle commands and numerous engine operating parameters. A schematic of the engine control system is provided in figure 1-14. A split fuel feed system is provided whereby fuel discharged from the left and right engine fuel feed pumps is separately routed to the engine on the respective side. The fuel feed lines, which pass through the aft fuselage tanks but do not offer self-sealing protection, incorporate an isolation valve before penetrating the nacelle firewall. Activation of either FUEL SHUTOFF handle in emergency situtations will manually isolate the respective engine nacelle from the supply of additional flammable fluids. Resetting of an actuated FUEL SHUTOFF control, opens the fuel and hydraulic isolation valves without the need for maintenance action. Although the fuel feed lines in the engine compartment will withstand negative pressures incurred by closing the fuel isolation valve at maximum thrust conditions without collapsing the line, the FUEL SHUTOFF control should not normally be activated with the respective engine running.

Motive Flow Fuel Pump

(U) A centrifugal pump is mounted on the engine accessory gear box to return high pressure fuel to the fuselage and wing tanks to effect normal fuel transfer. The engine feed line is tapped to recirculate motive flow fuel at a rate consistent with the pumping capacity of the motive flow pump. The pump recirculating fuel flow varies from 10,000 pounds per hour (250 psi) at IDLE to 27,000 pounds per hour (400 psi) at MIL. This motive flow fuel is the medium used to power the turbine connected to the engine feed pump in the respective fuel feed tank and continue through control valves to ejector pumps in the respective fuselage and wing fuel tanks. There is no cockpit control provided for the motive flow fuel pump. Failure of the pump will cause illumination of either R or L FUEL PRESS advisory light and also a reduction in the rate of fuel transfer. Failure of a motive flow fuel pump will not restrict full afterburner operation below 15,000 feet MSL altitude or inhibit the transfer of fuel from any tank.

Main Fuel Pump and Filter

(U) A two stage engine driven pump is mounted on the accessory gear box to provide 1st stage boosted fuel to the A/B hydraulic pump and 2nd stage, high pressure fuel to the engine fuel control. All interstage fuel is subject to filtration in the main fuel filter (10 micron, paper type) which incorporates a mechanical pin (red tip) that is tripped to a protruding position by excessive (5 psi) pressure differential across the filter. The clogged filter indication is latched in the protruded position and cannot be manually reset except by removal of the filter. No cockpit indication is provided for a clogged main fuel filter and

ENGINE CONTROL SYSTEM

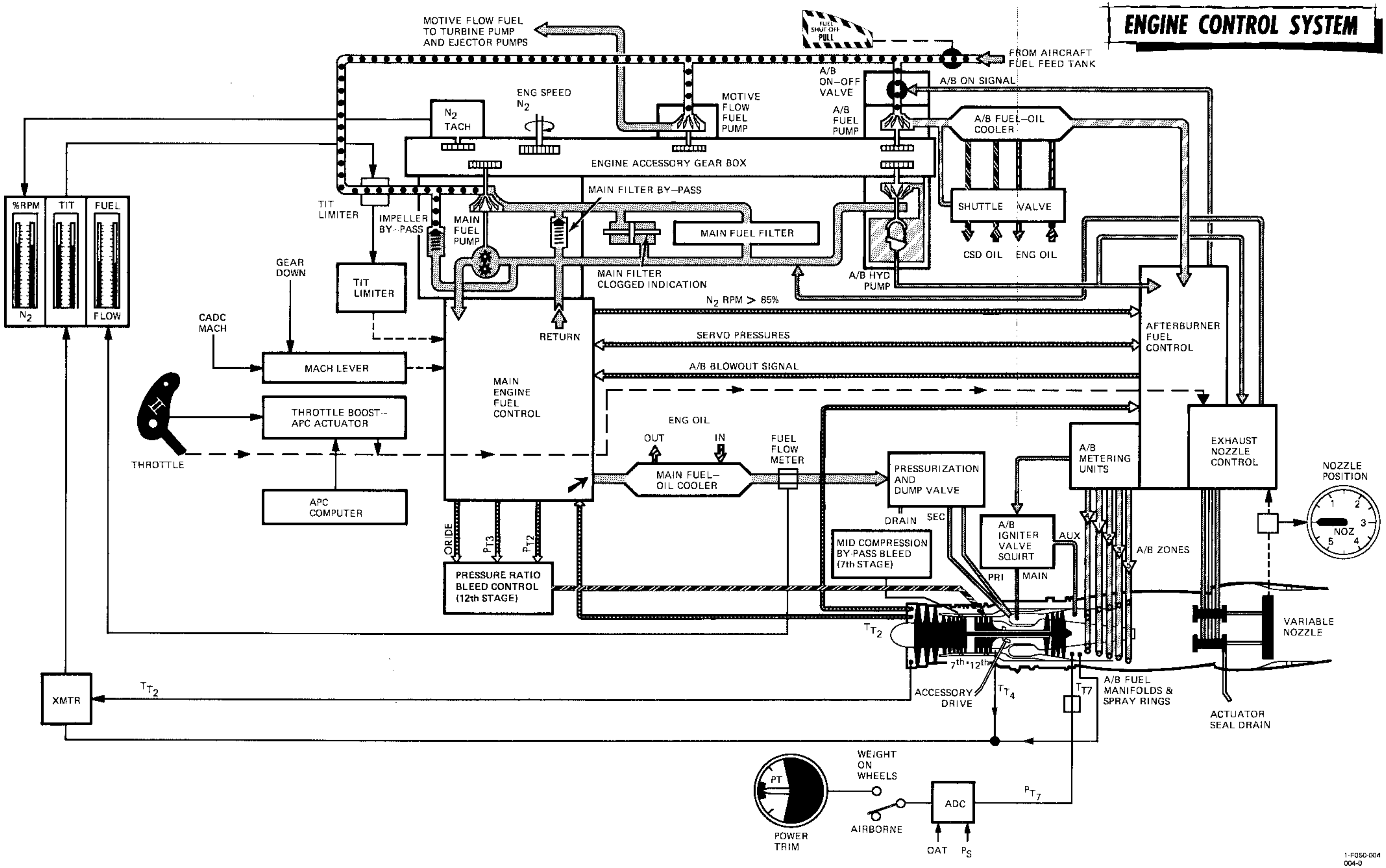


Figure 1-14.

engine operation will not be restricted because of a spring loaded filter by-pass valve (opens at 10 psi differential). Proper maintenance of the filter is important since JP-5 fuel is very susceptible to entrained moisture in the fuel which when frozen forms ice crystals. Maximum pumping capacity is approximately 17,000 pounds per hour with the pump pressure relief valve set at 1100 psi differential. Throughout the engine start and operating speed range, the pumping capacity of the main fuel pump is several times the metered requirements of the main engine fuel control so that the excess is returned to the pump interstage level downstream of the main fuel filter. No degradation in engine operation will occur with a failure of the centrifugal impeller of the pump's first stage as the engine feed pump will supply a positive head of pressure to the 2nd stage gear pump via an impeller by-pass valve; no cockpit indication is provided for such a failure. Conversely, failure of the 2nd stage gear pump will result in a flameout due to insufficient pressure for operation of the hydro-mechanical main engine fuel control. Although no cockpit indication is provided for such a failure, it will be evident when attempting an air-start that no fuel flow registers with the throttle at IDLE and an engine speed greater than 15% N_2 speed. In such a situation, the throttle of the afflicted engine should be placed to OFF.

MAIN ENGINE FUEL CONTROL

(U) The main engine fuel control is a hydro-mechanical device mounted in tandem to the main fuel pump which serves as an intermediary over pilot control of the engine. Based on pilot commands through the throttles, the main fuel control on each engine automatically and independently establish engine thrust by taking full cognizance of the numerous operating variables to which the engine is sensitive. Throughout the aircraft flight envelope the control system automatically attempts to provide stable (stall and flutter free) engine operation under steady state and transient conditions, protection against exceeding engine limit conditions, and compatibility with the air indication system. A fuel metering valve is the sole regulating control of fuel flow to the main engine. Pilot control over the engine has been established in such a manner as to provide a near linear net thrust-to-throttle relationship for sea level static conditions. Rotational movement of the engine cross shaft is mechanically transmitted to the main engine fuel control. Absolute cut-off of fuel to the engine is provided with the throttle less than one-half the travel between OFF and IDLE and flow modulation is provided beyond that point to the IDLE detent.

(U) At throttle settings of MIL or greater, engine output is limited by turbine inlet temperature, corrected airflow, burner case pressure and creep of rotating components for which the main engine fuel control provides automatic compensation. The engine acceleration schedule protects the engine from compressor stalls and exceeding turbine inlet temperature limits during transients. A cold start feature enriches the acceleration schedule in the starting speed range as a function of fuel temperature. During deceleration transients a signal is provided to the pressure ratio bleed control to open intercompressor bleeds (12th stage) to protect against deceleration stall or flame out. High pressure compressor rotor (N_2) governing limits in the fuel control are also biased by compressor inlet pressure and temperature signals. Satisfactory engine operation is dependent upon proper setting (trimming) of the engine fuel controls to ensure development of rated thrust, prevent limit exceedance and ensure airflow compatability with the inlet duct to maximize aircraft performance. The control unit incorporates ground adjustments (specific gravity and minimum fuel flow) for use of JP-4 as an alternate for JP-5 type fuel. Most controls are spring-loaded to the increased position in the control unit for failure conditions.

Mach Lever

(U) The Mach lever provides a mechanical input to the main fuel control as a function of aircraft Mach number to control engine airflow at high aircraft speeds by providing a maximum and minimum N_2 limitation. The Mach lever on each engine receives signals from the air data computer (CADC) which varies its position linearly between 0.25M and 2.5M. Above 1.5M on non-standard days, engine airflow starts to effect inlet stability to a point where limits must be imposed. Airflow is kept within a topping limit which prevents excessive distortion at maximum thrust and a bottoming limit which prevents engine buzz at idle thrust settings. Mach lever positioning below 1.5M does not impose any limits on engine airflow. With the lever failed in the maximum Mach position thrust on the affected engine cannot be reduced below 4,300 pounds. If the Mach lever fails in the 1.6 to 1.8M range, a maximum reduction of 22% in the level of MIL thrust will result. In order to minimize the consequences of a failed Mach lever, the absence of a signal or zero volts commands the Mach lever motor to the minimum Mach condition; a landing gear down discrete signal also performs such action as an overriding function. Additionally, a retract spring is incorporated on the Mach lever to ensure the return of the lever with a motor failure.

Mach Lever Test

(U) An automatic check of the Mach lever control unit while on-deck can be initiated by selecting the MACH LEV position and depressing the master test switch on the pilot's MASTER TEST panel. This test is normally conducted during the PRE-TAXI checklist with engines at idle thrust. The test simulates an input signal from the air data computer to the Mach lever control unit, which compares input and feedback signals to each engine actuator. If the comparison indicates proper operation the GO light on the MASTER TEST panel illuminates; otherwise, the NO GO light illuminates indicating a malfunction. Landing gear handle interlock circuits and weight on wheels switch interlocks prevent initiating the test while airborne.

TIT Limiter

(U) The turbine inlet temperature (TIT) limiter is a proportional and integral control device that limits TIT at high steady state thrust levels to the preset maximum allowable value (1175°C). The limiting device consists of a solid state electronic unit with an electric motor inputing to the governor schedule in the main engine fuel control. The electronic package is protected from the high temperature environment of the engine nacelle by passing fuel supplied to the inlet of the main engine fuel pump through its case. Turbine inlet temperature signals to the limiter are the same as displayed on the pilot's TIT indicator. Proportional control is accomplished by varying the duty cycle of the motor as a percentage of the 6 second full duty cycle of the motor. Turbine inlet temperature errors less than ±3.5°C above the reference value result in no limiter action. On the other extreme, errors greater than ±10°C result in continuous corrective action until the limit of authority is reached. When the TIT is below the reference value (a condition existing at virtually all thrust settings below MIL) the limiter attempts, within its authority, to raise the TIT level to eliminate the sensed error. Limiter authority approximates 2% N_2 speed with the upper limit set at the engine ground trimmed MIL governor schedule; the lower limit is set to prevent excessive reduction of engine speed by the limiter without hindering its TIT limiting capability. Therefore, the limiter only serves to downtrim the engine at high steady state thrust levels where TIT allowable limits are exceeded. The limiter is not capable of causing the engine to exceed the ground trimmed condition.

Fuel Oil Heat Exchanger

(U) Metered fuel from the engine fuel control passes through the coolant tubes of the main fuel oil heat exchanger enroute to the pressurization and dump valve. The fuel serves as the coolant for engine lubricating oil which flows around the heat exchanger tubes.

Fuel Pressurization and Dump Valve

(U) The fuel pressurization and dump valve divides the fuel flowing to the engine into two separate streams to optimize the spray pattern for fuel atomization in the combustion cans. Low pressure starting fuel is directed only through the primary stream to the fuel nozzles until fuel pressure at the pressurization and dump valve is sufficient to overcome a spring loaded pressure valve to unport the secondary stream. Upon engine shutdown, the stoppage of metered fuel flow allows the dump valve to open and drain fuels in the primary manifold overboard through the lower nacelle door. Forward and aft drains controlled by reed valves are also located in the lower portion of the combustion chamber to route residual fuel overboard. These drains normally deposit minimal residual of fuel on the deck upon shutdown and their proximity to the wheel brakes does not impose a hazard.

Fuel Nozzles

(U) Four flow-divider type fuel nozzles in each combustion can deliver metered fuel, in the proper state of atomization for maximum burning, into the compressor discharge air entering the combustion chambers. The nozzles produce a uniformly distributed, cone shaped spray pattern. Nozzle air baffles, fuel spray pattern and combustion liner design serve to reduce the tendency for carbon build-up and incomplete combustion so as to minimize aircraft visual detection by the exhaust gas trail.

IGNITION SYSTEM

(U) The basic engine ignition system provides a redundant means of electrically igniting the atomized fuel-air mixture in the combustion cans during the engine start cycle and automatic and manual airstart ignition in the event of an engine stall or flameout. Each engine incorporates a dual main ignition system comprised of an ignition alternator, ignition terminal transformer, automatic restart switch, ignition exciters and spark ignitor plugs as illustrated in figure 1-15.

(U) Electrical power for the engine ignition system is provided by an engine driven, low voltage (2 to 5 Volts ac) alternator which incorporates two independent, current-generating circuits. Ignition alternator voltage output is sufficient above 8% N_2 speed to effect a start. In the de-energized mode, the ignition system is grounded in the ignition terminal transformer to provide a positive short of the alternator output. When energized, the short circuit is opened to permit alternator current from redundant generating circuits to independently flow to two ignition exciter boxes. By means of a power transformer and capacitor circuits each 4 joule exciter box steps up the low voltage output of the alternator windings to 25,000 volts. Shielded ignition leads separately route the high tension voltage output from the two exciter boxes to independent spark ignitors in the lower two combustion cans. The atomized fuel-air mixture in the remaining six combustion cans is ignited by propogation of the flame through crossover tubes.

(U) Selection of either engine for start with the ENG CRANK switch arms the redundant ignition circuitry of the selected engine so that high energy ignition is auromatically provided when the throttle is advanced from OFF to IDLE with the engine turning over. During the start cycle electrical ignition is de-energized when the ENG CRANK switch spring returns to OFF at about 45% engine speed when the starter centrifugal switch opens the CRANK switch holding solenoid. Ignition is also terminated during an aborted start if the throttle is retarded to OFF. Limits for the start ignition duty cycle are 2 minutes on, 3 minutes off, 2 minutes on and 23 minutes off.

(U) An automatic restart switch is incorporated on each engine to automatically energize the individual engine ignition system for an approximate 30 second duration upon detection of a rapid decay in engine burner pressure. Additionally, an AIR START ignition button on the pilot's left console outboard of the throttles provides a manual means of energizing the ignition system. Depression of the AIR START button provides 50 seconds of continuous ignition for both engines if engine rotor speed is in excess of 8% N_2 speed.

IGNITION SYSTEM

BASIC ENGINE - TF30-P-412

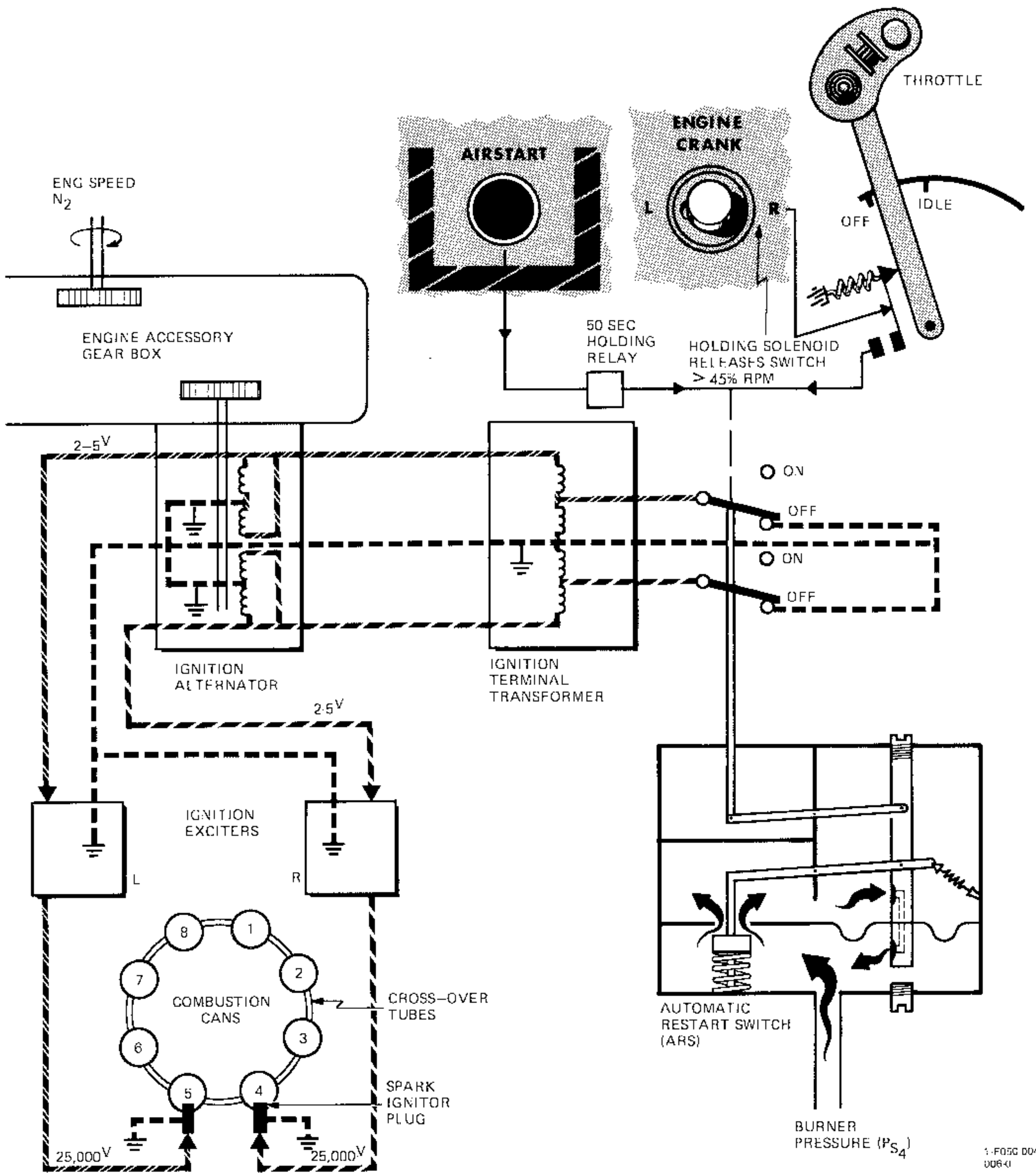


Figure 1-15.

AFTERBURNER FUEL SYSTEM

(U) The afterburner augments main engine thrust by injecting fuel through one to five manifold zones into the turbine and fan discharge airstream in the forward section of the afterburner duct. The afterburner increases the local temperature and thereby increases the exhaust nozzle velocity at the nozzle. Initial afterburner ignition is provided by a "hot streak" ignition system. Thrust augmentation can be infinitely modulated between 11, 700 pounds and 17, 000 pounds installed net thrust at static sea level conditions; the minimum thrust augmentation represents a 1, 200 pound thrust increase over that developed by the main engine without augmentation. The efficiency of afterburner thrust augmentation in the lower zones is on a par with that of turbojet engines of comparable thrust ratings. The efficiency of operation is significantly less with maximum thrust augmentation; thrust specific fuel consumption (pounds of fuel per pound of thrust) between MIL and MAX ratings differs by a factor of five.

Afterburner Hydraulic Pump

(U) The afterburner (A/B) hydraulic pump receives fuel from the main engine fuel pump interstage (downstream of the main fuel filter) and boosts the pressure to provide a hydraulic medium for actuating the exhaust nozzle and supply the initial flow of metered fuel for afterburner operation. The A/B hydraulic pump consists of a centrifugal 1st stage and a variable displacement 2nd stage with a rated output of 11, 000 pounds per hour at 2500 psig discharge pressure. A pulsation damper is incorporated downstream of the pump to reduce the amplitude of flow pulsations. Approximately 2, 000 pounds per hour of the discharge circulates through the afterburner fuel control and exhaust nozzle control back to the main engine fuel pump interstage for fuel cooling purposes. Aside from using pump discharge flow for operating the variable area exhaust nozzle, the pump supplies the first 3, 500 pounds per hour of afterburner fuel flow metered in the afterburner fuel control. Failure of the A/B hydraulic pump will result in the exhaust nozzle failing to an open position due to the bursting pressure differential and blowout of the afterburner. Subsequent attempts to relight the afterburner will be unsuccessful.

Afterburner Fuel Pump

(U) During non-afterburning operations or less than 3, 500 pounds per hour afterburner fuel flow, the afterburner fuel pump impeller runs dry with the A/B pump turn-on valve closed. The bearings of the single stage, centrifugal pump are lubricated by the engine oil system. With afterburner fuel flow demands less than 3, 500 pounds per hour, excessive fuel temperatures would result if the A/B fuel pump were not deactivated. Upon development of an afterburner fuel flow demand exceeding 3, 500 pounds per hour, the afterburner fuel control ports A/B hydraulic pump pressure to actuate the A/B pump turn-on valve to the open position. This action introduces airframe boosted fuel feed to the centrifugal impeller of the A/B fuel pump. Rated pumping capacity of the engine driven afterburner fuel pump is 70,000 pounds per hour. Initial flow from the pump closes the pump drain and vent valve, sends a signal to the afterburner fuel control transfer valve to provide a cut-off for A/B hydraulic pump contribution of fuel to the afterburner metering system, and sends a signal to the A/B fuel oil heat exchanger shuttle valve to permit engine and CSD oil to flow through the cooler. Afterburner fuel pump operation continues to deliver fuel to the afterburner fuel control until afterburner metering flow demands reduce to less than 3,500 pounds per hour. Thereafter, the A/B pump turn-on valve closes, the A/B fuel pump discharge ceases, engine and CSD oil ceases to flow through the A/B fuel oil heat exchanger, and the pump overboard drain and vent valve unseats. Failure of the A/B fuel pump will result in an A/B blowout with greater than 3,500 pounds per hour afterburner flow demand; however, minimum zone afterburner operation with demands less than 3,500 pounds per hour will not be impaired.

Afterburner Fuel Oil Heat Exchanger

(U) Fuel discharged from the A/B fuel pump flows through a fuel oil heat exchanger enroute to the A/B fuel control unit. In those conditions where the A/B fuel pump is running dry with no discharge, a spring loaded shuttle valve bypasses engine and CSD oil circulation around the A/B fuel oil heat exchanger. The A/B fuel oil heat exchanger only serves to cool engine and CSD oil with a fuel discharge from the A/B fuel pump. A pilot line from the A/B fuel pump discharge line serves to control shuttle valve position to permit engine and CSD oil to flow around the heat exchanger tubes for cooling, with a pump discharge pressurized.

Afterburner Fuel Control

(U) The afterburner fuel control is a hydro-mechanical unit which contains five A/B fuel metering units, one for each zone of operation, that regulate the flow of fuel to the afterburner spray manifolds. Except for step thrust increases at the turn-on/off point for each zone, pilot control over the afterburner provides a near linear net thrust-to-throttle relationship at steady state conditions between the minimum and maximum ranges of afterburning. In addition to the throttle input to the A/B fuel control via the engine crosshaft, metering unit positions are biased by burner can pressure and zones (2, 3, and 5) discharging into the fan stream are also biased by compressor inlet temperature. The A/B meters schedule a fixed fuel air ratio for each zone of thrust augmentation. Initiation of afterburner operation requires that the high pressure compressor rotor speed be greater than 85% N_2 speed, control ratio be near maximum and the throttle sufficiently advanced in the afterburner range. An exhaust nozzle position feedback is used to schedule the rate of change of A/B fuel flow during transient operation. At altitude the basic engine match must be suppressed during A/B operation to insure compatibility with inlet design requirements. Suppression is a relative reduction in airflow through the engine and is accomplished by controlling the exhaust nozzle area and temperature during afterburning. Operation in zones 1 through 4 results in a constant exhaust nozzle temperature with variable area whereas in zone 5, the area is constant and the temperature is increased. A sudden decrease in turbine discharge pressure initiates an afterburner blowout signal which shuts off all A/B fuel, momentarily deriches the main engine fuel control and opens 12th stage intercompressor bleeds until the exhaust nozzle closes. A ground adjustment is also provided on the A/B fuel control to account for specific gravity differences in JP-5 and JP-4 type fuel.

Afterburner Ignitor Valve

(U) The A/B ignitor valve controls the phasing of the "hot streak" ignition and afterburner zone/manifold fuel flow to effect an afterburner light in response to A/B fuel control signals and turbine discharge pressure. As the throttle is advanced approximately one-half inch forward of the MIL detent (outboard), fuel from the A/B metering unit is injected into the zone 1 manifold. Initial afterburner ignition is provided by a "hot streak" ignition system whereby a main squirt of raw fuel is injected into the lower (No. 4) combustion can. This causes an overrich fuel mixture which imparts a burst of flame through the turbine section into the forward portion of the afterburner. Since at high altitudes it is necessary to supplement the main squirt to ensure that the flame travels rearward far enough to ignite fuel being sprayed from the zone 1 manifold, a second (auxiliary) squirt of raw fuel is injected just aft of the fourth stage turbine disk. Both the main and auxiliary squirts last only momentarily, so that subsequent to an unsuccessful A/B light or an A/B blowout, the A/B igniter valve must be reset before a successful relight may be attempted. To reset the A/B igniter valve, the throttle must be retarded aft of the reset position standard procedure would be to retard the throttle to the MIL detent (outboard) and hesitate momentarily seconds before advancing for a relight attempt. The hesitation allows the A/B igniter valve to close and recharge the main squirt plenum. Failure of the A/B igniter valve will most probably inhibit afterburner lights.

Afterburner Manifold Zones

(U) The engine employs five metered zones of afterburning to provide smooth thrust augmentation with throttle movement in the afterburner range. Fuel from the A/B metering unit is directed into five concentric manifolds located in the afterburner diffuser just forward of the flameholders. Initial light-off of the afterburner is achieved with fuel sprayed from the zone 1 manifold in the turbine exhaust stream. As the throttle is advanced more fuel is metered for burning; zones 2, 3 and 5, dispense fuel into the fan discharge airstream and likewise zone 4 in the turbine exhaust stream. At MAX thrust all five zone manifolds are supplying fuel to the afterburner. Rapid throttle movement through the full afterburner range results in automatic staging through the five zones in consecutive order at the maximum rate consistent with maintaining afterburner stability. Nominal minimum acceleration and deceleration time through the full afterburner range is seconds.

VARIABLE AREA EXHAUST NOZZLE

(U) The convergent-divergent geometry of the variable area exhaust nozzle (figure 1-15) provides the optimum exhaust throat area and controls the expansion of the exhaust gases for maximizing efficiency and performance under varying engine mass flow and thermal conditions. The iris action of the variable area nozzle is achieved by close tolerance overlapping seals which bridge adjacent leaf segments to provide a relatively smooth surface contour. Engine exhaust gases at the higher thrust settings are discharged through the nozzle throat at sonic velocity and are accelerated to supersonic velocity by the controlled expansion of the gases.

VARIABLE AREA EXHAUST NOZZLE

TF30-P-412 ENGINE

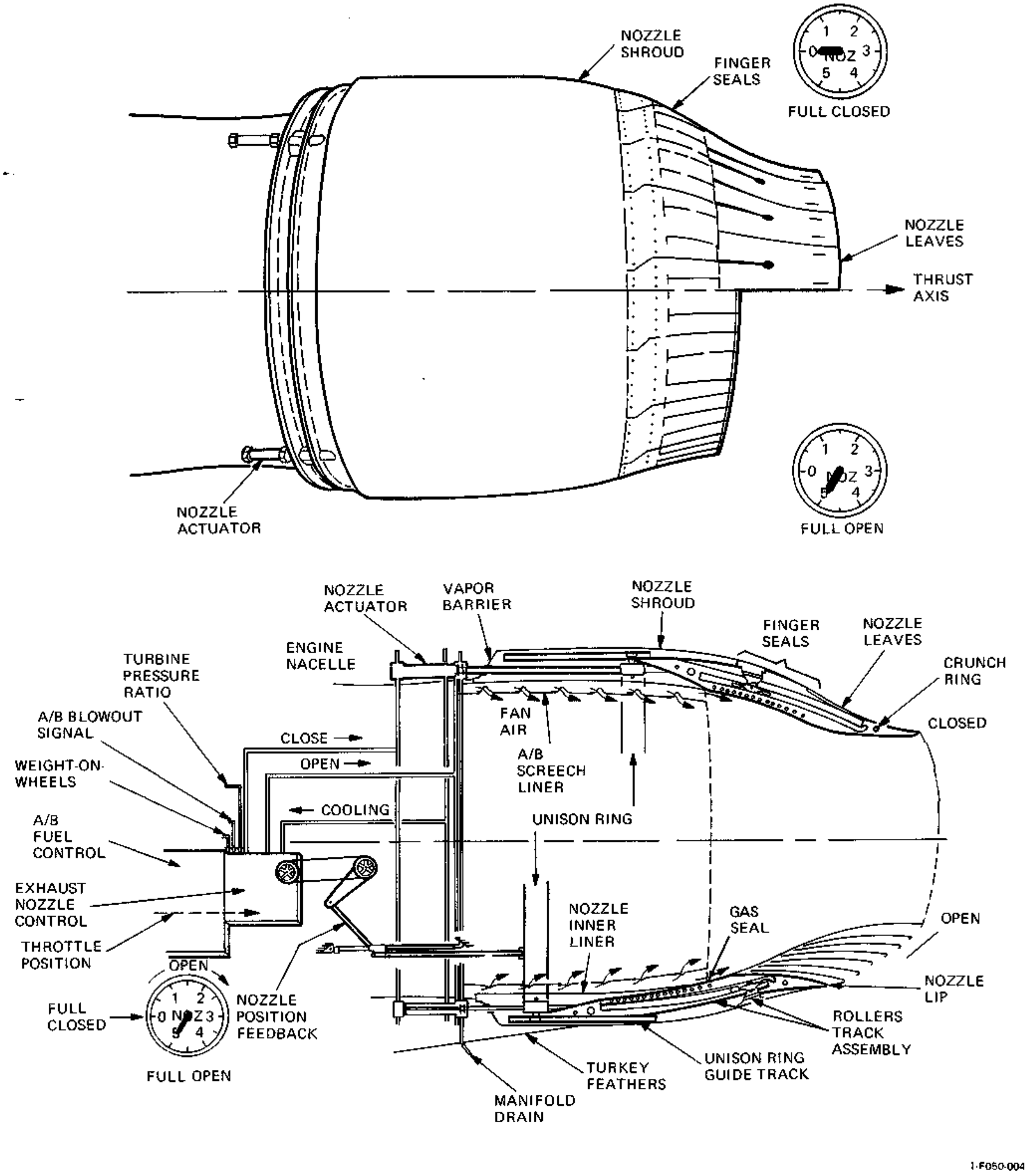


Figure 1-16.

(U) Eighteen peripheral mounted sheet-metal leaves form the internal and external contours of the nozzle. Alignment of the leaves is provided by translating tracks which ride on roller assemblies attached to the nozzle shroud. The roller assemblies maintain nozzle peripheral alignment and exhaust area under both bursting and collapsing radial loads. Nozzle area is infinitely variable between the closed (3.53 sq.ft.) and full open (7.5 sq.ft.) positions. Positioning of the nozzle is accomplished automatically by four synchronized hydraulic actuators which translate a circumferential unison ring interconnecting the translating leaf assemblies. High pressure (2,500 psi) fuel discharged from the A/B hydraulic pump is used as the hydraulic medium for positioning the actuators. The engine exhaust nozzle control unit is a hydromechanical computing device that regulates fuel to the actuator ports for setting the exhaust nozzle position in response to controlling parameters. A nozzle position feedback system, consisting of slider rod, link, crank and pulley-cable assembly, mechanically transmits unison ring position to the exhaust nozzle control unit. The nozzle position transmitter for the cockpit indication is mounted at the exhaust nozzle control feedback.

(U) The four nozzle actuators are interconnected by tubing routed around the circumference of the afterburner shroud which transmits pressure from the exhaust nozzle control unit for extending and retracting the actuators and provides for fuel circulation to prevent overheating by a controlled internal leakage in the actuators. In addition, the cylinder rod ends incorporate double seals around the actuating shaft with the interspace drain port manifolded between the four actuators. Manifold fuel drainage is provided by a tube protruding sufficiently through the nacelle shroud on the underside of the nacelle to carry the fuel droplets free of the nozzle. The four nozzle actuators are exposed to the engine nacelle environment and vapor seals are provided around the piston at the point of nozzle shroud penetration. A static air environment is provided between the nozzle shroud and the inner liner; however, gas seals between the afterburner shroud and nozzle casing prevents the back flow of exhaust gases. Fan discharge air ducted between the afterburner screech liner and nozzle inner shroud exits at the aft end of the afterburner section through a corregated spacer to form a thermal boundary layer over the inner nozzle surface.

(U) Close aerodynamic tolerances are maintained between the engine and airframe by location of the aft engine mounts just forward of the nozzle shroud and the use of turkey feather seals contained by a cable on the inner surface to provide a smooth interface. Premature separation of the boundary layer airflow ahead of the nozzle lip in the closed nozzle condition can induce a significant increase in aircraft boattail drag since the nozzle shroud cross-sectional area represents a significant proportion of the maximum cross-sectional area of the airframe. In the closed nozzle position the nozzle actuators overdrive to maintain a preload on the leaf assemblies. A "crunch ring" consisting of a series of circumferential tubes containing rigging bolts is provided internally near the nozzle lips so that the critical nozzle closed position is accurately maintained by a preload on the abutted bolts which exceeds the radial burst pressure. Close tolerances under elevated temperatures are necessarily maintained to prevent a performance degradation due to nozzle efficiency and aircraft drag.

(U) For on-deck operations the nozzle is scheduled full open with the throttle within 1/4 inch of the IDLE position to reduce engine residual thrust (450 pounds per engine). Otherwise, during ground or inflight engine operation in the non-afterburning region, the nozzle is maintained in the full closed position. Afterburner ignition and light-off is accomplished with the nozzle closed subject to immediate opening whereafter opening of the nozzle is scheduled by throttle angle as biased by turbine pressure ratio. Although nozzle position is a linear function of throttle position in the afterburner range, variations (+60% to -25%) in nozzle position will occur in an attempt to maintain an optimum turbine pressure ratio under airflow and fuelflow conditions that differ from the nominal. An afterburner blowout signal, detected by a sudden and substantial decay in turbine exhaust pressure, automatically closes the nozzle to prevent a thrust loss below the military thrust level as long as the throttle remains in the afterburner range without initiating a relight. Additionally, nozzle closure upon afterburner blowout is necessary to prevent a low pressure compressor (N_1) overspeed condition due to the reduced back pressure on the fan stages.

(U) Failure of the afterburner hydraulic pump or a pressure line will cause the exhaust nozzle to float or oscillate. Failure of the nozzle position feedback cable will automatically shut down the afterburner and close the nozzle. A cocked unison ring or an actuator failure may cause up to 1/2 degrees misalignment of the thrust axis. Binding in the nozzle mechanism or a failed actuator can best be detected by reduced slew rates during full authority step commands such as the pop-open nozzle action during on-deck operations. Normal nozzle position slew rates through full authority are 1.5 seconds to close and 2.0 seconds to open.

ENGINE BLEED AIR

(U) Bleed air is extracted from various stages of the engine low and high pressure compressors to increase compressor operating stall margins, perform engine associated services, and supply high pressure/temperature bleed air for operation of auxiliary equipment in the aircraft. Figure 1-17, illustrates the four stages of compression where bleed air is extracted and summarizes the purposes of each.

Compressor Stall Margin

(U) In the twin spool axial compressor, the low and high pressure units are mechanically independent yet are aerodynamically balanced to efficiently provide a high compression ratio (18.7:1). Control of axial airflow velocity through the compressior is the key to the control of performance and stall margin. For any particular compressor rotational speed an increase in axial airflow velocity through the compressor decreases the effective angle-of-attack of the compressor blades. Compressor stall margin is provided by select bleeding of air from the interstages of the low and high pressure compressors in addition to governing schedules in the engine main fuel control. Since compressor bleeds reduce engine thrust levels, the select use of such bleeds is scheduled to minimize the impact on engine operating efficiency and performance. Opening of interstage bleeds increases the axial airflow velocity upstream and the 7th and 12th stages of compression provide the desired effect on the low and high pressure compressors with minimum thrust loss. Although no direct control from the cockpit is exercised over these bleeds, it is important to understand the significant aspects which directly affect the operation of the propulsion system.

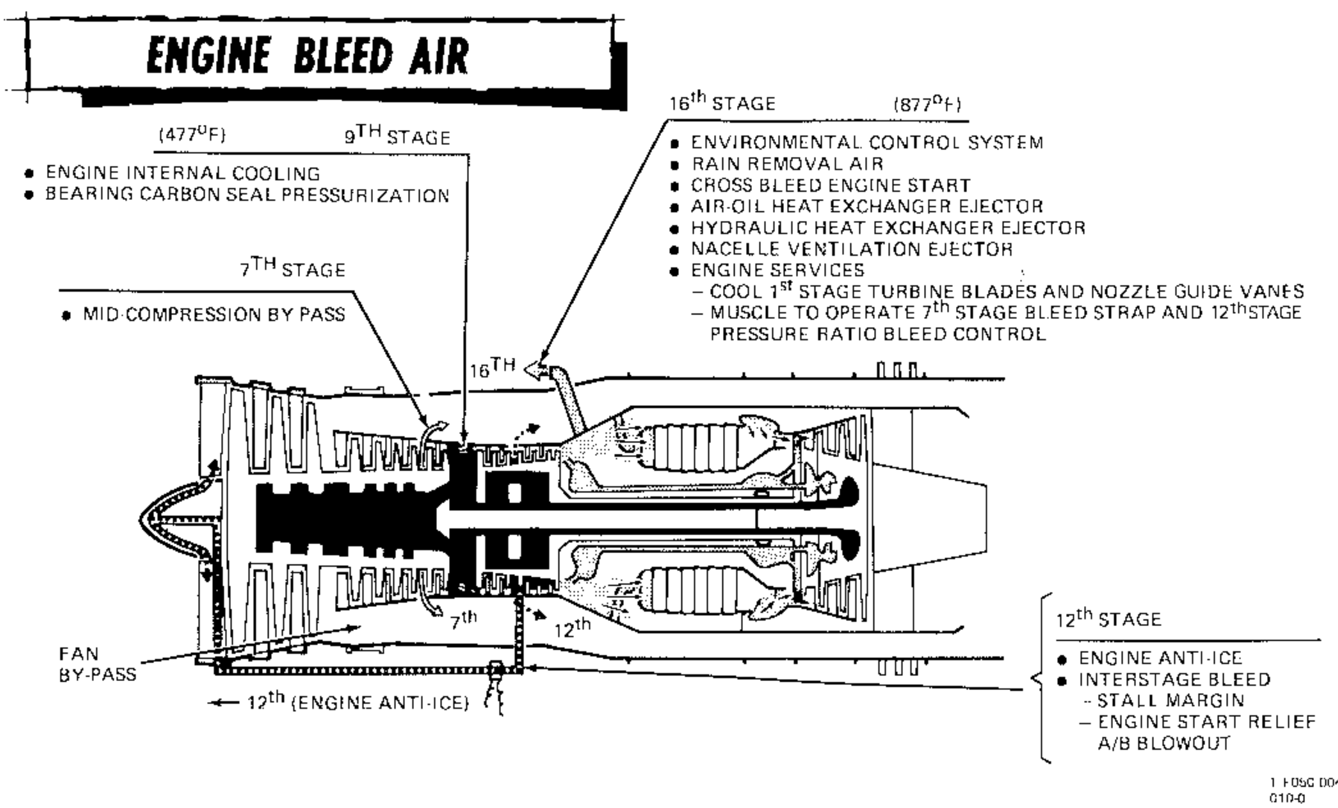


Figure 1-17.

(U) Pressure distortion at the engine compressor face results in a maldistribution and reduction in axial airflow velocity downstream through the compressors. Extremes in distortion may be produced by high aircraft angle-of-attack and/or yaw conditions, firing of the internal gun, or air turbulence. In order to minimize such effects, the mid compression by-pass system is provided to control the porting of 7th stage bleed air from the low pressure compressor into the fan by-pass duct. Such action is primarily directed towards decreasing the blade angle-of-attack in the initial stages of the low pressure compressor, however it does improve the quality of airflow downstream. Controlling parameters for operating the electric (essential dc No. 2 bus) solenoid are illustrated in figure 1-18. The solenoid uses 16th stage bleed air to close the bleed valves and in the absence of electrical power the device fails to the closed position.

(U) Engine speed transients impose an additional stall margin problem over that of steady state operating conditions. The high pressure compressor, with its smaller mass and accessory drive horsepower extraction, characteristically accelerates and decelerates faster than the low pressure compressor, with its larger mass and diameter. As a result of this transient operating imbalance, airflow and rotor speed in some parts of the compressors are no longer compatible and without some corrective measures a stall could occur. For the acceleration condition, the additional fuel produces a relatively high burner pressure downstream which slows down the airflow through the high pressure compressor. Whereas for a deceleration, the more rapid deceleration of the high pressure compressor in effect creates a partial blockage to reduce the axial airflow velocity through the low pressure compressor. Although the main fuel control schedules provide some protection during engine speed transients, it is mainly concerned with governing fuel to air ratio and N_2 speed and cannot effectively react to prevent an aerodynamic imbalance between the two compressors.

MID COMPRESSION BYPASS

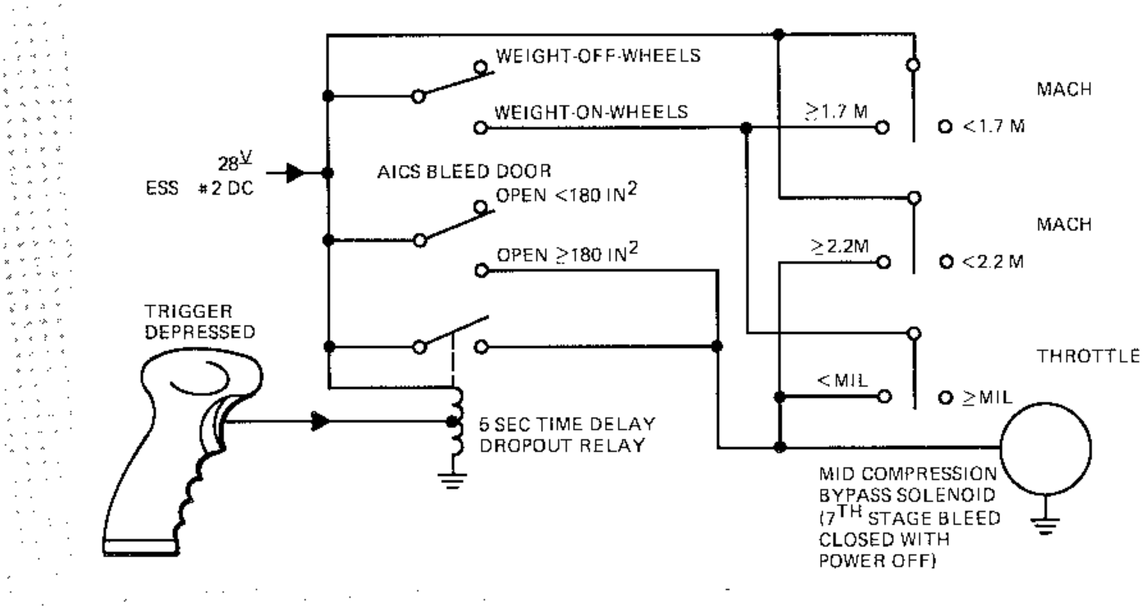


Figure 1-18.

(U) In addition to the transient speed characteristics, at the lower engine speeds the low pressure compressor operates at a higher angle of attack with a high pressure ratio, both of which tend to reduce the available stall margin. As a corrective measure to alleviate stalls under the above conditions, interstage bleed is extracted from the high pressure compressor at the 12th stage and ported into the fan by-pass duct. This action effects the aerodynamic balance in both compressors.

(U) The pressure ratio bleed control is the controlling unit for 12th stage bleed (exclusive of the engine anti-ice function) to reduce the stall possibility during rapid engine decelerations and at low engine operating speeds. Under steady state or gradual engine transient speed conditions the controlling parameter in the pressure ratio bleed control is the low pressure compressor pressure ratio; however, under rapid decelerations the normal schedule is overriden by a signal from the main fuel control causing the 12th stage bleed valves to open at a higher speed. During gradual changes in engine speed, the 12th stage bleeds close at 78% N_2 speed on acceleration from IDLE and open at 76% N_2 speed upon deceleration from higher thrust settings. This range of operation is of particular concern because it can effect the preciseness of engine thrust modulation during the landing approach if operating at the lower thrust levels. Additionally, an afterburner blowout signal will momentarily open the 12th stage bleed valves to prevent a stall much the same as the snap deceleration condition. The pressure ratio bleed control uses 16th stage bleed air as a medium for operating the 12th stage bleed valves.

Engine Anti-Ice

(U) The compressor inlet guide vanes and nose dome are susceptible to icing under a wider range of conditions, particularly at static or low speed conditions with high engine RPM, than that which causes ice to form on external surfaces of the airframe. Ice formation at the compressor face can restrict engine maximum airflow which results in a thrust loss, decreased stall margin, and dislodgement of ice can damage the compressor. The engine anti-icing system is designed to anti-ice (prevent the formulation of ice) rather than de-ice the inlet guide vanes and nose dome. Hot bleed air (12th stage) is passed through the hollow compressor inlet guide vanes to the nose dome and discharged into the engine at the rotor hub. No inlet duct anti-icing is provided. Cockpit control of the engine anti-icing system is affected through the ENG-PROBES ANTI ICE switch with three positions (OFF, AUTO, and ORIDE). An ice detector is located in the left inlet duct wall slightly forward of the engine compressor face. Accumulation of ice on the detector changes its natural frequency which in turn causes illumination of the INLET ICE caution light when the ENG/PROBES ANTI-ICE switch is in either the OFF or AUTO position. With the control switch in the AUTO position, the ice detector signal also activates the anti-icing systems on both engines. The ORIDE position activates anti-ice air independent of the ice detector signal and illuminates the INLET ICE caution light independent of icing detected conditions. The engine anti-icing system should only be used during flight and ground operations whenever icing conditions exist or are anticipated because of its adverse effects on engine stall margin and thrust. The thrust loss associated with the use of the engine anti-icing system should be especially accounted for as its degraded effect on take-off and wave-off performance. The engine anti-icing control valve on the engine is electrically energized from the essential dc No. 2 bus.

Miscellaneous Bleed Air Functions

(U) A continuous bleed of 9th stage air is used exclusively for internal cooling throughout the engine and for pressurization across the rotor bearing carbon seals. Similarly, 16th stage bleed air is used for internal cooling in the combustion and turbine sections; and is discharged through the 1st stage turbine nozzle guide vanes and blades for cooling. Aside from a cross-bleed engine start function, other applications for 16th stage bleed air in the engine compartment include operation of 7th and 12th stage bleed valves and on-deck (interlocked with weight-on-wheels) operation of the air-oil/IDG heat exchanger ejector, and hydraulic oil heat exchanger ejector. The main supply of 16th stage of bleed air is ducted out of the engine compartments and joined to supply hot bleed air to the environmental control system.

ENGINE STARTING SYSTEM

(U) Each engine is provided with an air turbine starter which may be pressurized from an external ground starting cart or by cross-bleeding high pressure bleed air from the other engine (if operating). A low pressure (50 psi) air source and 115 volt, 400 H_Z ac power are required for the initial engine start on the deck. Figure 1-19 illustrates the components associated with the engine starting system.

(U) The air hose from the external starting cart is connected to the aircraft fitting located in the left sponson area behind the main gear strut. Ground start air is ducted into a central bleed air (16th stage) manifold which interconnects the air turbine starters on both engines. The air supply to each air turbine starter is pressure regulated (45 psi) and controlled by a regulator and shutoff valve at the turbine. Each pneumatic starter is composed of a turbine, geartrain, over-running clutch with a speed-sensing device, and an overspeed disengagement mechanism with a shear pin. The turbine drive is clutched to the accessory gear box to back-drive the high pressure compressor for starting. Several shutoff valves are incorporated in the bleed air manifold to selectively isolate engine starters, subsidiary bleed lines and the environmental control system air supply for effecting a start. Torque requirements for cranking enable the start cycle for an engine to be completed in approximately 35 seconds. Starting load is reduced by automatically maintaining the 12th stage engine bleed valves open during the start cycle. Maximum engine motoring speed with the pneumatic starter is pproximately 24% N_2 speed. During the crank cycle the turbine driven starter discharges air into the nacelle.

(U) A complete crank/abort control of the engine start cycle is provided with the ENG CRANK switch and throttles. Placement of the ENG CRANK switch to the L or R position opens the corresponding starter pressure shutoff valve to allow pressurized air to drive the turbine. Additionally, the ENG CRANK switch in conjunction with throttle position energizes the ignition system and appropriate shutoff valves to condition the bleed manifold for starting. During the initial phase of the starting cycle, the ENG CRANK switch is maintained in the L or R position by a holding coil. The supply of pressure to the air turbine starter is interrupted at approximately 45% N_2 speed by activation of a speed-sensing centrifugal switch. Actuation of the cut-out switch closes the turbine shutoff valves and open the holding coil circuit to enable the ENG CRANK switch to spring return to the center (OFF) position. This action in turn reconditions the bleed air manifold valves to permit 16th stage bleed air to flow to the environmental cooling system and ejectors in the engine compartment. Engine cranking procedures during a cross-bleed start are the same as with a ground start cart except that the throttle on the operating engine must be advanced to 85% N_2 speed in order to supply sufficient 16th stage bleed air for motoring the air turbine starter on the opposite engine. Preparatory to a start, the air conditioning source selector button must be on BOTH and applicable circuit breakers must be in.

ENGINE CRANK SYSTEM

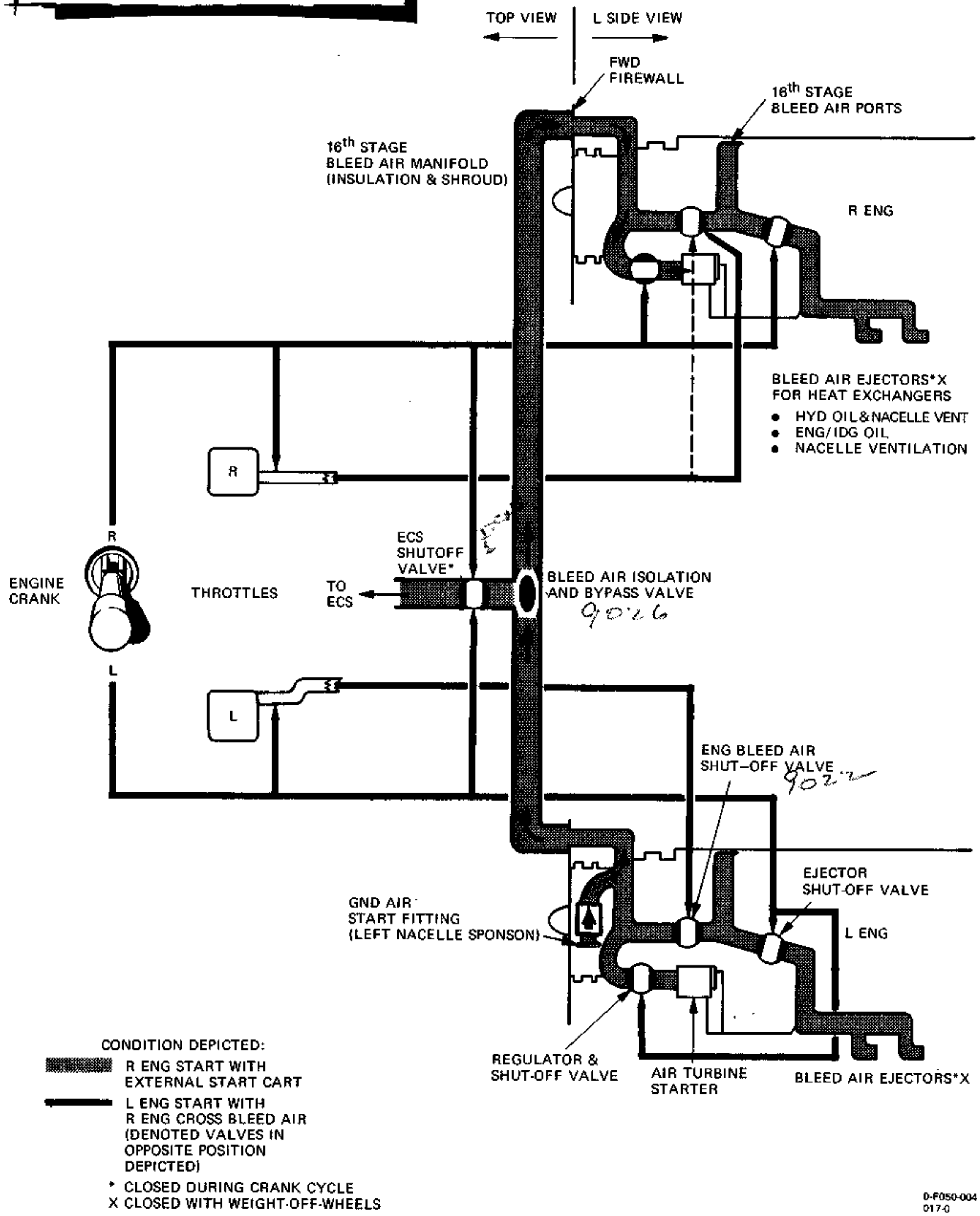


Figure 1-19.

(U) Minimum airspeeds to sustain sufficient windmill speed ($N_2 \geq 15\%$) for an airstart are provided in Section XI, Part 10. The increased speed necessary to effect left engine windmill airstarts is based on emergency generator hydraulic demands. Depression of the AIRSTART button on the pilot's left console provides 50 seconds continuous ignition to both engines for a windmill airstart. With one engine running, a cross-bleed airstart may be effected in the same manner as a ground start except that circumstances permitting, the starter should not be engaged with engine RPM greater than 10% and the throttle of the operating engine should not exceed 85% RPM.

ENGINE COMPARTMENT COOLING

(U) Each engine compartment is completely isolated from the primary air inlet, and the efficiency and cooling of the variable area exhaust nozzle are not dependent on nacelle airflow. Therefore, within the bounds of the forward firewall (landing gear bulkhead) and the nozzle shroud, the cooling system for each engine compartment is a separate identity. Cooling requirements for the turbofan engine are minimized by virtue of the annular fan bypass duct. Figure 1-20 illustrates cooling airflow patterns through the engine compartment during ground and flight operations. Two air cooled heat exchangers are also illustrated, however only the hydraulic heat exchanger cooling airflow is associated with engine nacelle cooling.

(U) On deck, cooling airflow through the engine compartment is induced by the hydraulic heat exchanger ejector in the forward end of the compartment. Air enters through the nacelle ram air scoop on the left side, passes through the hydraulic heat exchanger and is discharged into the engine compartment. The air flows through the full length of the nacelle to discharge overboard through a louvered port atop the nacelle on the outboard side of the vertical tail.

(U) The ejector action in the engine oil/IDG heat exchanger draws air from the ventral scoops and discharges it overboard without any mixing with nacelle air. Since ejector bleed airflow is shut off on the operating engine during a cross-bleed engine start, the ENG CRANK switch should not be prematurely selected to the opposite engine to avoid an excessive nacelle temperature rise.

(U) Inflight cooling of the engine compartment is accomplished by nacelle ram air scoops circulating boundary layer air through the entire length of the compartment and expelling the air overboard through louvered exits at the rear just forward of the engine nozzle shroud. Bleed airflow to the heat exchanger and ventilation ejectors is shut off inflight by a weight-on-wheels interlock. The ram scoops maintain minimum pressure differential across the nacelle doors and provide adequate ventilation airflow through the nacelle. Nacelle mixing doors remain closed and ram airflow from the ventral strakes is sufficient for engine oil/IDG heat exchanger operation.

(U) Fire access doors are located on the outboard side of the nacelles at the forward end to permit insertion of fire suppressing agents by ground personnel in event of an engine compartment fire. The aft fuselage fuel cells which abut to the nacelle compartment are protected by a titanium fire shroud, as illustrated in figure 1-20, which serves as a flash insulation blanket. A fire warning system is provided in each engine compartment to warn the crew of a nacelle fire/overheat condition.

ENGINE COMPARTMENT COOLING

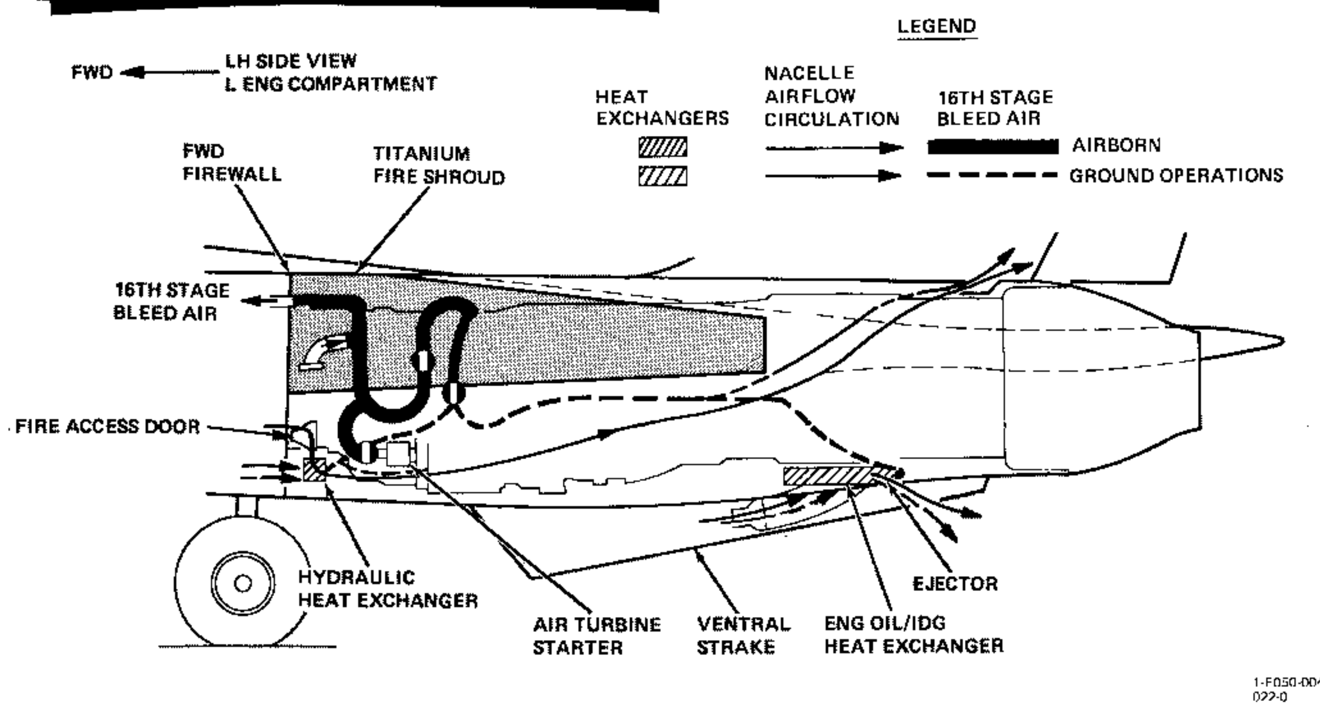


Figure 1-20.

FIRE DETECTION SYSTEM

(U) The fire detection system provides a cockpit indication of fire or overheat conditions in either engine compartment. Separate systems are provided for each engine compartment, each consisting of a thermistor type sensing loop monitored by a transistorized control unit. False-alarm-free operation is provided by the grounded circuit type system employing short discrimination. Figure 1-21 is a functional schematic of the system which is powered by 28 volts from the essential dc No. 2 bus.

(U) The sensing loop for each engine compartment consists of a 45 foot continuous tubular element routed throughout the entire length of the engine compartment on both sides above the nacelle door hinge line. The tube sheath, which is clamped in grommets to the engine compartment structure, contains a ceramic-like thermistor material in which are embedded two electrical conductors; one of the conductors is grounded at both ends of the loop. Electrical resistance between the two conductors varies inversely as a function of temperature and their length so that heating of less than the full length will require a higher temperature for the resistance to decrease to the alarm point. The L or R FIRE warning lights in the cockpit illuminate when the respective entire sensing loop is heated to approximately 600°F or when any six inch section is heated to approximately 1,000°F. Since the system averages the varying temperatures throughout the sensing loop, the alarm temperature is set close to the maximum nacelle ambient temperature, giving greater sensitivity to a general temperature rise without being subject to false alarms from local, non-hazardous hot spots. The fire alarm output relay to the light is a latching type which remains in the last energized position independent of power interruptions until the fault clears. False alarms resulting from short circuits in the sensing element loop are prevented by a discriminator circuit in the control unit. This circuit discriminates between a resistance lowered at a finite rate due to fire or overheat and resistance lowered instantaneously due to a short circuit. The system is automatically deactivated upon the detection of an instantaneously low resistance without alarming the crew and remains inoperative until the fault clears. Manual FUEL SHUTOFF handles are provided in the forward cockpit to enable the pilot to isolate the supply of fuel to either engine compartment in the event of fire.

(U) False alarms triggered by moisture entering the sensing element and connectors or by damage resulting in short circuits or grounds in the sensing element are eliminated by the system design. For contaminants (moisture, fluid or any electrolyte) to cause a false alarm, they must appear as a low resistance across the sensing element circuits; however, this system uses a low dc potential which causes moisture, even salt water, to appear as a high resistance. Thus the system insensitivity to the presence of moisture in the sensing element connectors, or moisture abosrbed by broken elements, means that not only will it not false-alarm but that its effectiveness in detecting a fire is unimpaired by the presence of moisture. Additionally, there is no loss or impairment of fire detector capability from a single break in the sensing element as long as there is no electrical short. The control unit continues to monitor the sensing element from each end to each side of the break. With two breaks in the sensing element, that section between the breaks becomes inactive although the remaining end segments remain active.

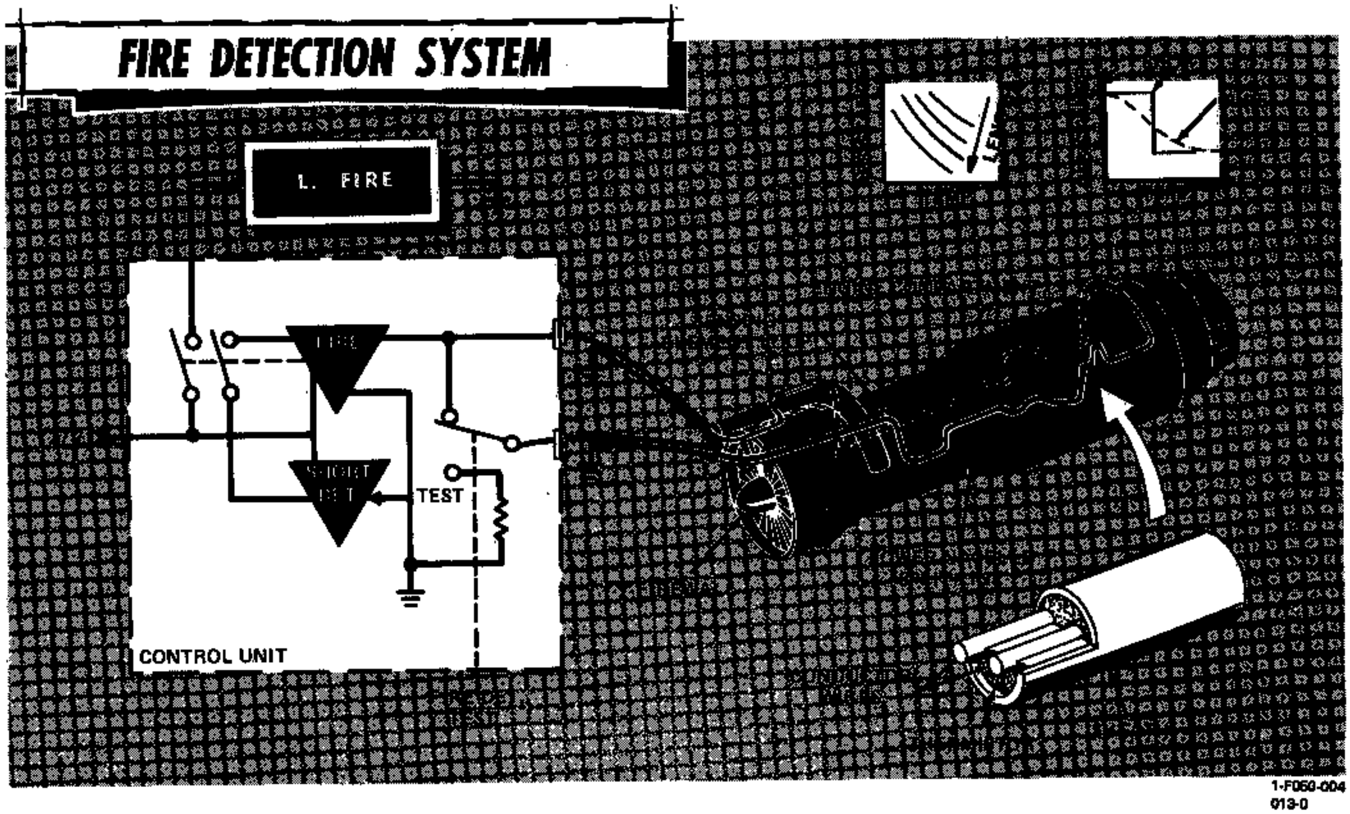


Figure 1-21.

Fire Detection Test

(U) An integrity test of the fire detection system can be performed during both flight and on deck operations by selection of the FIRE DET position on the pilot's master test rotary switch on the outboard right console. The integrity test simultaneously checks the sensing element loops of both engine compartments for continuity and freedom from short circuits, and the fire alarm circuits and FIRE warning lights for proper functioning. For the integrity test, one end of the sensing loop is electrically disconnected from the control unit and connected through a test resistor to ground. The test resistor prevents the short discriminator circuit from activating and allows the individual FIRE warning lights to illuminate contingent on satisfactory system continuity. Presence of a short circuit or control unit malfunction will cause the warning light to remain extinguished. As part of the preflight, maintenance personnel can verify proper functional operation of the short discriminator circuits using the FIRE SHORT simulation switch on the SYS TEST/SYS PWR ground check panel on the NFO left knee panel in conjunction with the FIRE DET test.

ENGINE OIL SYSTEM

(U) Each engine incorporates a self-contained oil system which provides pressure lubrication to the engine bearings and accessory drives (figure 1-22) using MIL-L-23699 grade oil. Total capacity of the system is 5.0 gallons with 4.0 gallons usable; normal oil consumption rate is 0.3 gallons per hour. Other than normal continuous operation under positive g flight conditions, the oil system permits engine operation under negative g flight for 60 seconds and under zero g flights for 30 seconds. During such flight conditions, oil pressure may drop, but as soon as positive g flight is resumed the pressure will return to normal.

(U) The forward section of the engine accessory gear box serves as the oil stowage tank. Filters and dipsticks are provided on both sides of the forward gearbox for checking the oil quantity and gravity servicing, however only the outboard dip sticks should be used for checking the oil service level. Engine oil quantity should be checked within 15 minutes after engine shutdown to ensure proper servicing.

(U) Oil from the main pump, which is pressure regulated to 45 ± 5 psi, is pumped in series through the air-oil cooler, main fuel-oil cooler and afterburner fuel-oil cooler (if A/B fuel pump is operating). If oil flow through any of the coolers becomes restricted, pressure buildup will open the applicable bypass valve to permit unrestricted oil flow. Additionally, oil flow will bypass the main fuel-oil cooler if fuel temperature out of the cooler becomes hot enough (235° F) to actuate the bypass valve. Downstream of the coolers the oil flows to a temperature control valve which opens below a preset value (185° F) to bypass the coolers until after engine starting the oil temperature rises above the preset valve. The oil is then directed to the engine bearings and to the accessory gear box. Scavenge pumps and drains located in the bearing compartments and the aft accessory gear box return the oil to the forward accessory gear box for re-use. Oil breather tubes are provided to vent the engine bearing compartments to the aft gear box. Air in the accessory gear box is de-oiled (oil particles removed from the breather air) with the excess internal pressure being vented overboard through a port in the engine lower nacelle door.

(U) Transducers for operating the engine oil pressure indicator and OIL HOT (>245° F) caution light are located near the terminal end of the oil pressure line routing through the engine just prior to the No. 1 bearing discharge jet. The oil pressure and temperature indicating systems operate on 26 -volt ac power that is supplied from the essential ac No. 2 bus through a transformer. The OIL PRESS caution light illuminates whenever either engine oil pressure drops below 25 ± 3 psi. No cockpit indications or maintenance

ENGINE OIL SYSTEM

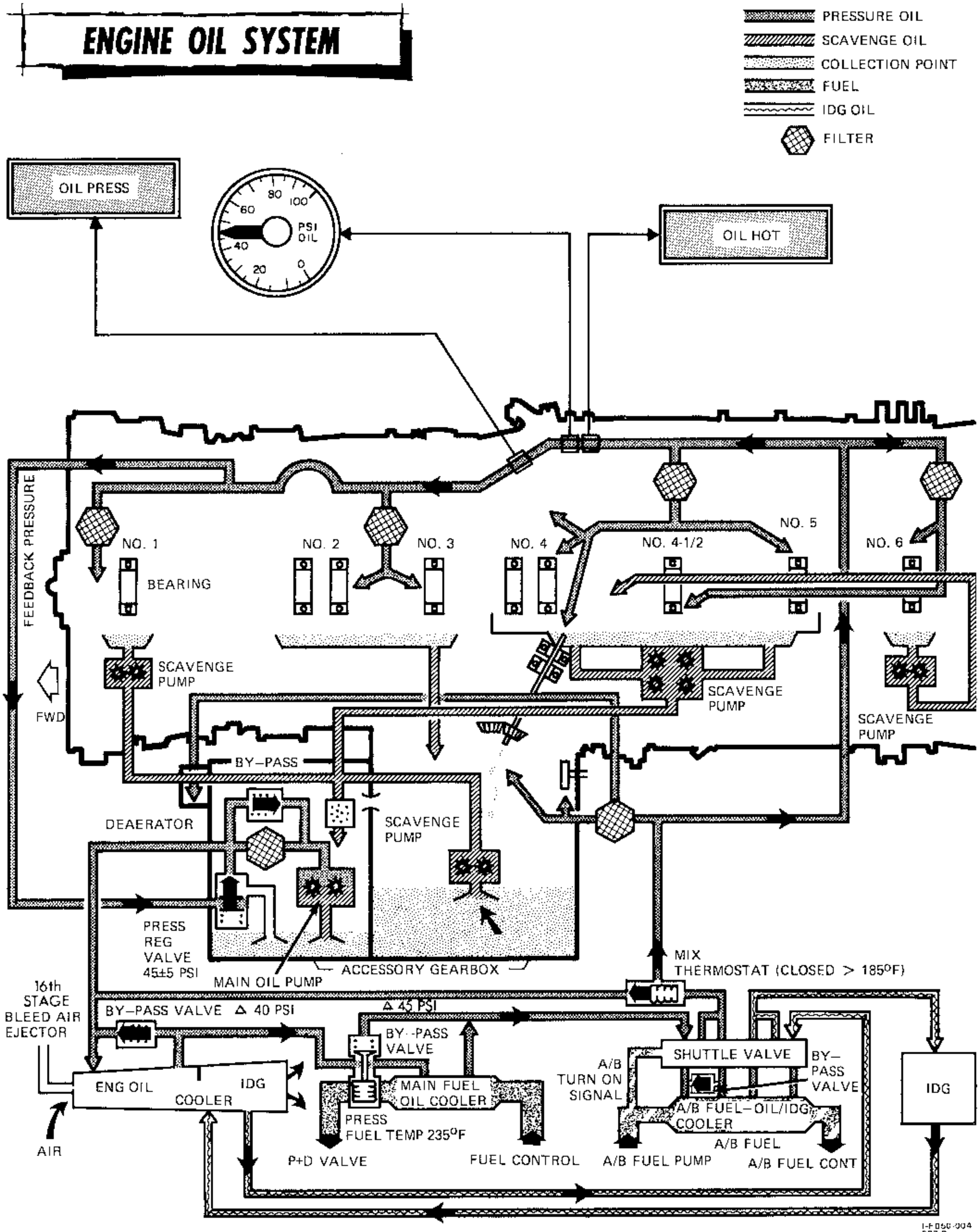


Figure 1-22

indicators are provided to detect an oil by-pass or clogged filter conditions aside from visible oil pressure fluctuations and routine maintenance inspections.

(U) The efficiency of the air-oil heat exchangers can be compromised during engine ground operations by a tailwind condition. A tailwind will cause recirculation of hot ejector induced discharge air into the ram inlet scoop on the ventral. Although 85% RPM is the best engine speed for overall efficiency of the main fuel-oil heat exchanger, even that relief can be insufficient if high engine power operation is allowed to persist with a tailwind. In flight illumination of an OIL HOT caution light will be normal following a high speed dash in MAX A/B, whereafter an IDLE thrust deceleration is performed. If feasible, an increase in engine fuel flow is the best corrective action for reducing oil temperature to an acceptable level. This will extinguish the OIL HOT caution light because of increased fuel-oil cooler efficiency.

ENGINE INSTRUMENTS

(U) Instruments for monitoring engine operation are located on the lower left side of the pilot's stepped instrument panel as illustrated in figure 1-23. Vertical scale instruments are used for the integral dual (left and right) display of high pressure rotor speed (N_2), turbine inlet temperature (TIT) and basic engine fuel flow. Adjacent to these instruments are located dual (left and right) but separate circular instruments for power trim, engine oil pressure and exhaust nozzle position. Take-off checks at MIL thrust should display evenly matched tapes on corresponding vertical scale instruments and all pointers on the circular instruments should be oriented at the 9 o'clock position. Electrical power for the operation of all engine instruments is obtained from the essential No. 2 electrical busses. Engine operating limits are illustrated in figure 1-24.

Engine RPM Indicator

(U) A tachometer generator mounted on the engine accessory gearbox provides a signal to the pilot's RPM indicator for displaying high pressure compressor rotor speed (N_2). Since the RPM indicator registers the fuel control governed speed, it enables the pilot to directly monitor main engine fuel control performance. The readout is in terms of percent with 100% equivalent to an N_2 rotor speed of __. The speed bias characteristics of the engine result in a speed variance with ambient temperature at MIL rated thrust. No display is provided for the low pressure compressor rotor speed (N_1) since the two compressors are aerodynamically balanced; however, failure of the exhaust nozzle to close upon afterburner blowout or during high power, non-afterburning conditions can induce an N1 rotor overspeed condition due to low back-pressure on the fan. This will illuminate the corresponding N_1 OVSP caution light.

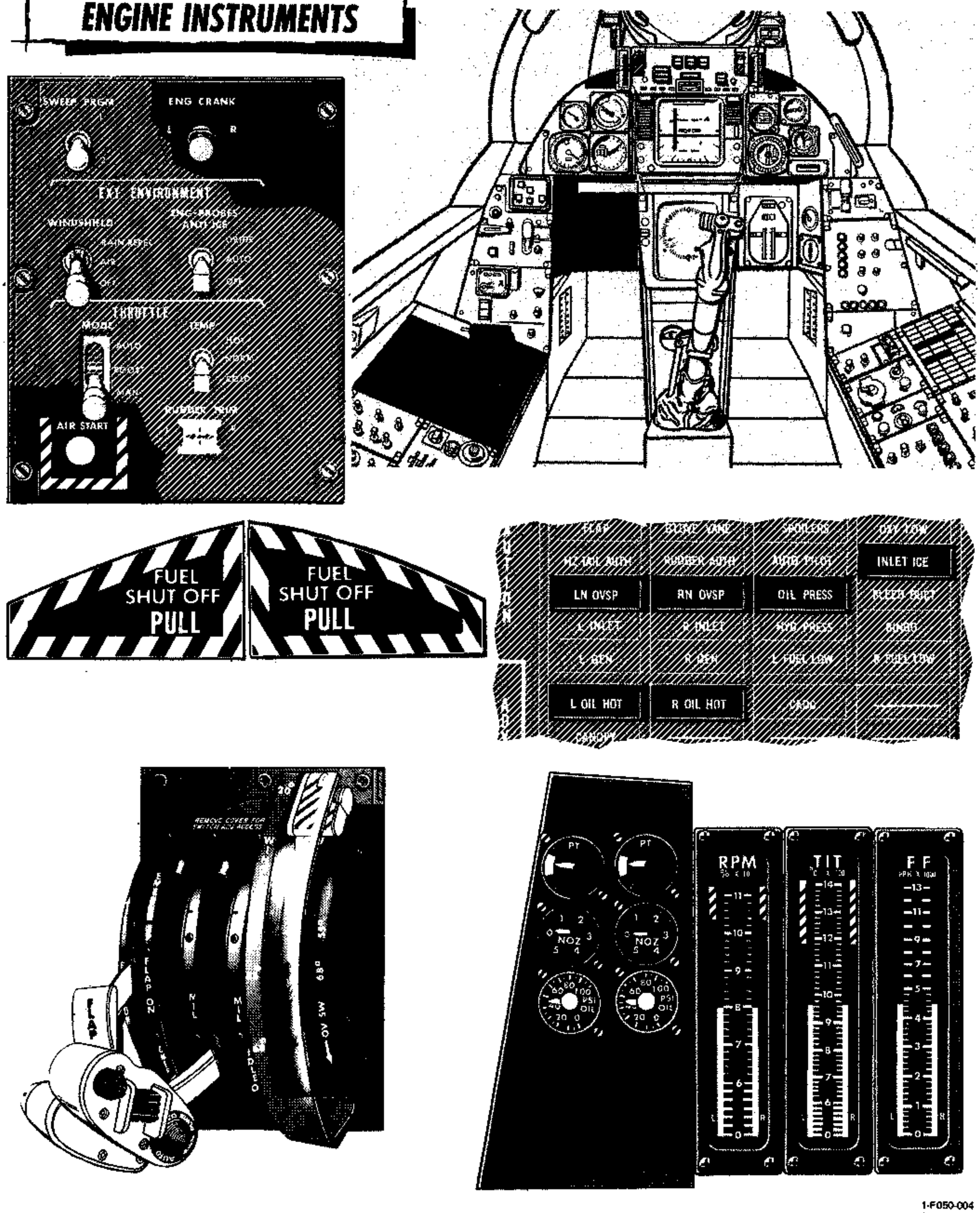
Engine TIT Indicator

(U) The turbine inlet temperature (TIT) signal is not the result of a direct measurement on the engine but is indirectly computed from measurements of compressor inlet temperature (T_{T2}), compressor discharge temperature (T_{T4}) and turbine exhaust temperature (T_{T7}). An overall indication of temperature conditions within the engine is provided by the computed TIT readout. The TIT signal to the cockpit is the same valve which the TIT limiter compares with a present maximum allowable valve (1175° C) to limit turbine inlet temperature at high steady state thrust levels.

Engine Fuel Flow Indicator

(U) The transmitter supplying the signal for the display of fuel flow in the cockpit is located in the engine fuel line downstream of the main fuel control so that it accurately reflects the fuel being supplied to the basic engine for combustion; no indication of fuel flow to the afterburner is provided.

ENGINE INSTRUMENTS



1-F050-004
032-0

Figure 1-23

ENGINE OPERATING LIMITS

TF30-P-412 ENGINE

OIL: MIL-L-7808D OR MIL-L-23699
FUEL: MIL-J-5624 JP-5 (JP-4 ALT)

OPERATING CONDITIONS		OPERATING LIMITS	
THRUST SETTING	TIME LIMIT (MINUTES)	MAXIMUM MEASURED TURBINE-INLET TEMP (C°)	OIL PRESSURE (PSIG) NORMAL
MAXIMUM (AFTERBURNING)	45	1175°	45 ± 5
PARTIAL AUGMENTATION	45	1175°	45 ± 5
MILITARY	45	1175°	45 ± 5
NORMAL RATED	CONTINUOUS	1015°	45 ± 5
IDLE	CONTINUOUS	--	30 MINIMUM
STARTING	MOMENTARY	705°	---
TRANSIENT	2	1195°	45 ± 5

NOTES

1. The maximum measured turbine inlet temperature limit for the transient operating condition for the TF30-P-412 engine is 1195°C. This temperature is permissible, in excess of the maximum steady-state limit, up to two minutes.

2. Momentary excessive oil temperature following thrust reduction: Should the oil hot caution light illuminate following a rapid reduction in the aircraft throttle, it is recommended that the throttle be advanced to a higher thrust setting, provided this can be accomplished without exceeding the maximum thrust allowed for the prevailing conditions. This will increase the fuel flow to the cooler, and will increase the cooling capacity of the oil cooling unit until the heat rejection from the engine can be accommodated by a lower fuel flow through the fuel-oil cooler. Retarding, rather than advancing, the throttle will not normally reduce the oil-in temperature. If the oil hot caution light does not extinguish within 10 seconds, or if it is impossible to use a higher throttle setting or to otherwise maintain oil-in temperature within limits, either the engine should be shut down or a landing should be made as soon as possible.

RPM LIMITS	
ANY OVERSPEED LIMIT EXCEEDED SHOULD BE REPORTED AS A DISCREPANCY AND MAXIMUM RPM NOTED.	
OPERATING CONDITIONS	OPERATING LIMITS
IDLE	64-73 PERCENT RPM
MAXIMUM N_2 SPEED, ENGINE INSPECTION REQUIRED IF EXCEEDED.	104.2 PERCENT RPM
N_2 OVERSPEED, ENGINE CHANGE REQUIRED.	TO BE SUPPLIED WHEN AVAILABLE

1-F050-004
013-0

Figure 1-24

Engine Power Trim Indicator

(U) Power trim in a sense is a measure of what the engine is accomplishing in an effort to produce thrust. Basically, it is a non-dimensional display of engine pressure ratio (EPR) corrected for ambient temperature conditions. Turbine exhaust pressure (P_{T7}) signals to the air data computer are used in conjunction with manifolded static pressure (P_S) in the ADC to digitally calculate engine pressure ratio. The sensed EPR is then biased with an incremental EPR which is a function of the difference between the sensed outside air temperature (T_{AMB}) and standard day temperature. This incremental EPR is determined from the MIL power trim relationship stored in the ADC and is not applicable for MAX thrust settings. The power trim indicator has a solid marked band with high and low indices at the band-width extremities. The low band index corresponds to the minimum standard day (59° F) EPR, and the bandwidth is 0.066 EPR's. During a MIL thrust check prior to take-off, the pointer should be within the solid band to indicate that the engine is developing rated thrust for the ambient temperature conditions. At MAX thrust settings the power trim indication will be high since the trim band is only representative of MIL thrust conditions. The power trim indicator is deactivated with weight-off-wheels where the pointer is driven behind a mask.

Exhaust Nozzle Position Indicator

(U) The transmitter for the nozzle position indication senses the mechanical feedback position signals from the nozzle to the exhaust nozzle control unit. The indicator displays equal graduations between 0 to 5 to indicate relative position from full closed to full open, respectively; however, these divisions are not numerically equal to the five zones of afterburning. Staging of the afterburner through the various zones is apparent on the indicator so that the relative degree of afterburning can be discerned. During afterburner operation, a mismatch in nozzle position is indicative of thrust asymmetry condition between the two engines which should be equalized for maximum efficiency. The pop-open (position 5) nozzle feature is provided with the throttle at IDLE and weight-on-wheels to reduce residual thrust for ground operation.

Engine Oil Pressure Indicator

(U) The engine oil pressure sensor is mounted in the pressure supply line to the forward bearings on the engine. The system is pressure regulated to 45 ± 5 psi and the oil pressure at IDLE thrust settings should not be less that 30 psi. During engine start the oil pressure rises correspondingly with N_2 speed so that at lower rotor speeds, oil pressure may be used as a cross-reference for the N_2 tachometer. In extremely cold weather, the rise of oil pressure during initial engine start will be sluggish. An OIL PRESS caution light located on the pilot's caution advisory panel will be illuminated whenever either engine oil pressure is less than 25 ± 3 psi.

Engine Instrument Test

(U) A test of engine instruments can be performed during both airborne and on deck operations. Selecting the INST position and depressing the master test switch on the Pilot's MASTER TEST panel will drive the tapes on the vertical scale instruments to a mid-point range (figure 1-19) for easy comparison. The RPM indicator will read 80% , the TIT indicator 967° C, and the FF indicator 4340 PPH.

AIRCRAFT FUEL SYSTEM

(C) The aircraft fuel system provides an internal usable fuel capacity of approximately 16,000 pounds (2,365 gallons) and two wet stations for external carriage of an additional 3,600 pounds (530 gallons) based on the use of JP-5 fuel (6.8 pounds/gallon). The location of the fuel tanks and associated plumbing provides for low vulnerability and failure isolation management. Fuel tankage in the fuselage is concentrated in the center and aft sections between the two engine nacelles. Integral wing fuel tanks and external fuel drop tanks complete the fuel loading capability. The aircraft fuel system normally operates as a split feed system with the left and aft tanks feeding to the left engine and the right and forward tanks feeding the right engine. Except for the external tanks, the tankage system is non-pressurized and uses motive flow fuel to effect fuel transfer. The supply of high pressure fuel from engine driven motive flow fuel pumps operates fuel ejector pumps to effect fuel transfer without the need of moving parts. The hydro-mechanical system is not dependent on electrical power to effect normal fuel transfer and feed. Total internal and external fuel quantity gaging is provided with a selective readout capability for individual tanks. Fuel system management requirements are minimal under normal operation in the feed, transfer, dump, and refuel modes of operation. Sufficient cockpit control is provided to manage the system under failure conditions, however, not at the compromise of flight safety. The aircraft fuel system design dictates the depletion of all usable fuel under twin or single engine operating conditions before an engine flameout occurs due to fuel starvation. Refer to figure 1-25 for fuel system controls.

FUEL CONTROLS AND CAUTION LIGHTS

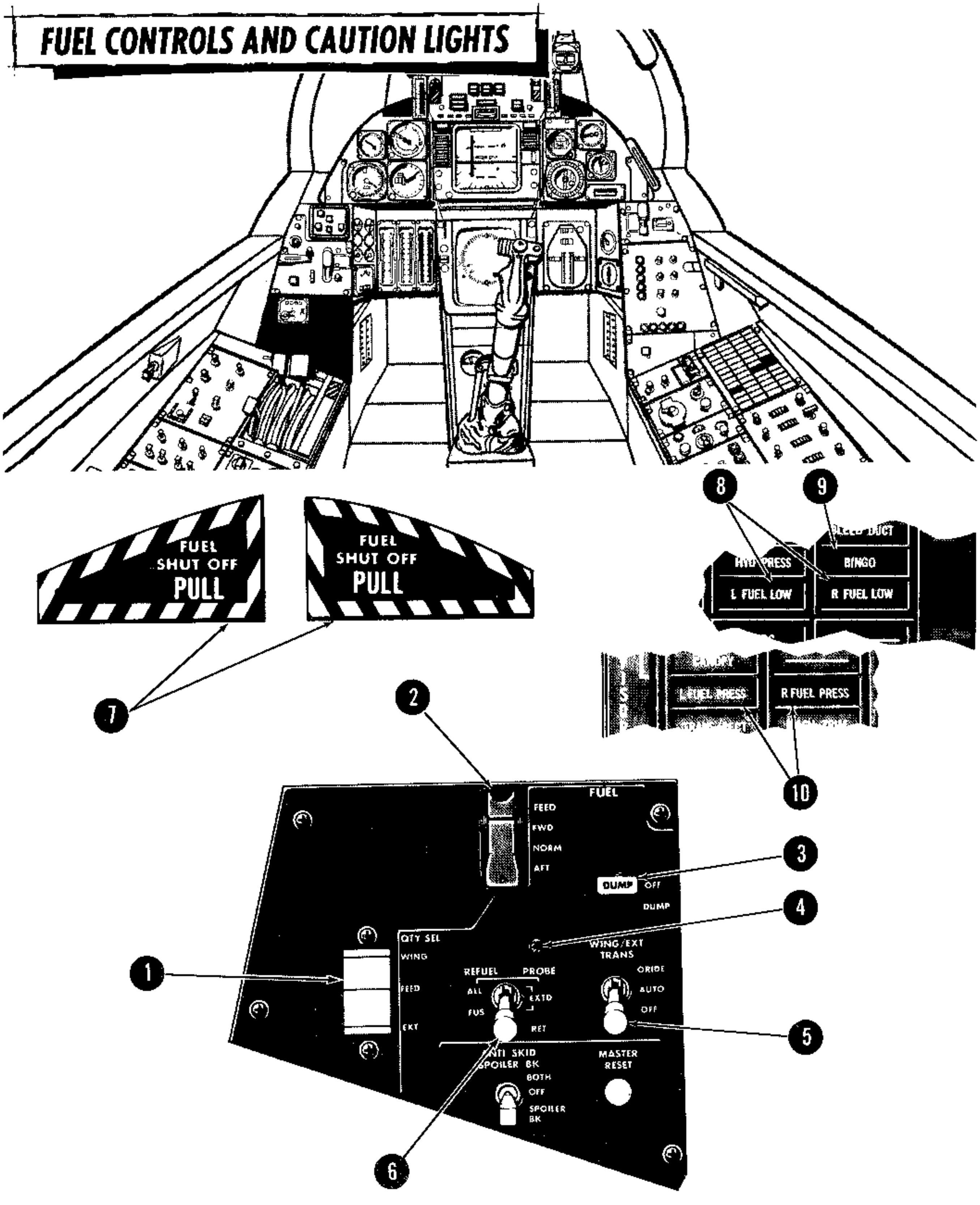


Figure 1-25

NOMENCLATURE		FUNCTION
1 QTY SEL SWITCH	WING -	Internal wing fuel quantity displayed on L and R counter pilot's fuel quantity indicator.
	FEED -	Rocker switch spring returns to FEED when not held in WING or EXT. Feed tank fuel quantity displayed on the L and R counter of pilot's fuel quantity indicator.
	EXT -	Fuel quantity in each external tank displayed on L and R counter of pilot's fuel quantity indicator.
2 FEED SWITCH	FWD -	Opens the feed tank interconnect valve with motive flow pressure and shuts off motive flow fuel to the aft tank ejector pumps.
	NORM -	Guarded position. Motive fuel flow supplied to ejector pumps in forward and aft tanks. Feed tank interconnect valve opens automatically when either thermistor device is uncovered in fuel cells 2 or 5.
	AFT -	Opens the feed tank interconnect valve with motive flow pressure and shuts off motive flow fuel to the forward tank ejector pumps.
3 DUMP SWITCH	OFF -	Dump valve closed.
	DUMP -	Opens a solenoid operated pilot valve, which ports motive flow fuel pressure to open the dump valve and allows gravity fuel overboard from cells 2 and 5. Dump electrically inhibited with weight on wheels or speed brakes extended.
4 INFLIGHT REFUELING PROBE TRANSITION LIGHT		Illuminates red whenever the probe cavity forward door is open during retraction or extension of the probe.

NOMENCLATURE	FUNCTION
(5) WING/EXT TRANS SWITCH	ORIDE - Allows pressurization and transfer of external tanks with landing gear down.
	AUTO - Normal position. Wing fuel is automatically transferred. Transfer of external fuel is automatic with landing gear retracted. Automatic shut-off of wing fuel when empty.
	OFF - Solenoid held in OFF position. Spring returns to AUTO with L/R FUEL LOW caution light illuminated. Closes solenoid operated valve to shut-off motive flow fuel to the wing and also inhibits external tank transfer.
(6) REFUEL PROB SWITCH	ALL/EXTD - Extends refueling probe. Shuts off wing and external drop tank transfer to permit refueling of all tanks.
	FUS/EXTD - Extends refueling probe. Normal transfer and feed. Used for practice plug-ins, fuselage only refueling, or flight with a damaged refueling probe.
	RET - Retracts refueling probe.
(7) L/R FUEL SHUTOFF HANDLES	Pulling the respective handle manually shuts off fuel to that engine. Push forward resets shut-off valve open.
(8) L/R FUEL LOW LIGHTS (Also single light on NFO caution panel.)	Fuel thermistors uncovered in aft/left or fwd/right feed group. Illuminates with approximately 1,000 pounds remaining in individual feed group.
(9) BINGO CAUTION LIGHT	Illuminates when the total fuel quantity is lower than the BINGO counter valve.
(10) L/R FUEL PRESS ADVISORY LIGHTS	Indicates insufficient discharge pressure from the respective sump tank turbine driven fuel boost pump.

FUEL TANKAGE AND HARDWARE

(U) Figure 1-26 illustrates the general fuel tankage arrangement in the aircraft. Fuel is compartmentized into eight separate fuselage cells, two wing box cells, two integral wing cells, and an optional loading of two external fuel drop tanks.

Feed Tanks

(C) The engine feed tanks, which span across the fuselage and glove slightly forward of the mid center of gravity position of the aircraft, consist of two self-sealing (except on the top surface) sump tanks on the under side inter-connected to two integral tanks which constitute the wing box directly above the sump tanks. The left and right portions of these tanks are partitioned so that the supply of fuel to the left and right engines is normally through the respective sump tank. The usable fuel capacity of each feed tank is approximately 1600 pounds (235 gallons) of which approximately 350 pounds (50 gallons) is in the sump tank. A negative "g" type check valve serves to trap fuel in the sump tank during inverted flight conditions. Vent lines also interconnect the upper side of the sump tank with the wing box above to prevent the formation of air pockets during refueling. The feed tanks contain turbine driven boost pumps and are not connected to the fuel dump manifold.

(C) The upper part of the feed tanks consist of the partitioned wing box which, because of its welded titanium structure, provides cells for protected fuel. The usable fuel capacity of each half of the wing box is approximately 1,250 pounds (184 gallons). Fuel in each side of the wing box gravity flows to the sump tank on the respective side. Vertical baffles in the wing box contain open ports at the top for vent and overflow and ports at the base for gravity transfer.

Forward Tank

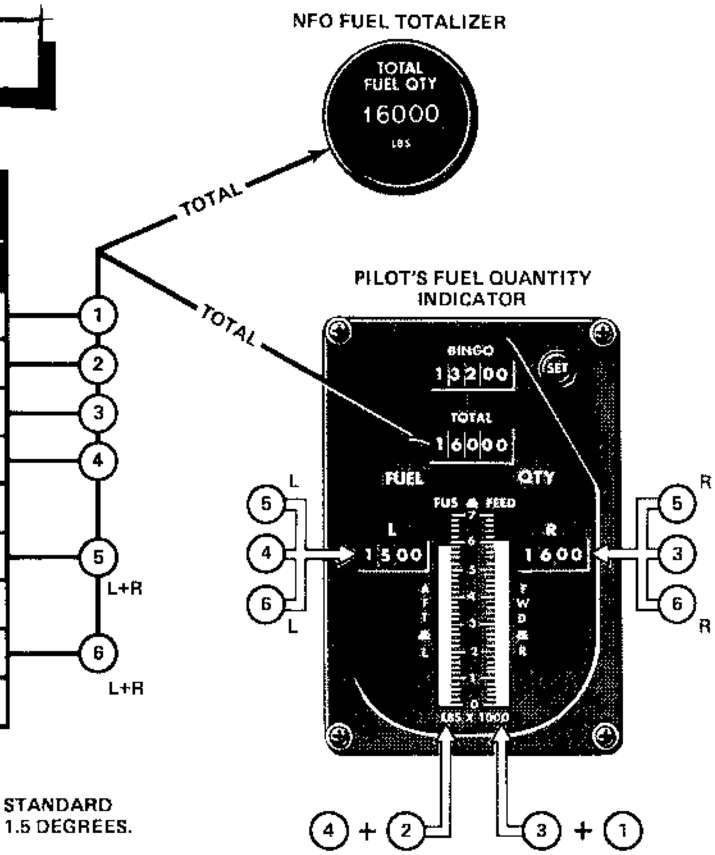
(C) Except for local cut-outs, the forward fuselage fuel tank comprises the center fuselage between the inlet ducts immediately ahead of the feed tanks. The forward tank, which is partitioned into two bladder cells, has a usable fuel capacity of approximately 4,500 pounds (662 gallons). The two cells (#1 and #2) are interconnected by open ports at the top for vent and overflow purposes. Flapper valves at the base provide for forward to aft fuel gravity transfer and serve as inertia check valves for arrestment and catapulting to minimize fuel surge loads.

Aft Tank

(C) The "U" shaped aft fuselage fuel tank group, which is partitioned into four bladder cells (#5, 6, 7 and 8) and a vent tank, has a usable fuel capacity of approximately 4,400 pounds (648 gallons). The most forward cell in the aft tank group lays laterally across the center fuselage. Extending aft are two "coffin" shaped tanks which contain two cells (#6 and #8) on the right side and one cell (#7) plus an integral fuel vent tank on the left side. The "coffin" tanks straddle the center trough area which contains the control runs, Sparrow missile launchers, electrical and fluid power lines. The right and forward cells in the aft tank are interconnected by upper ports for vent and overflow purposes. All fuel cells in the aft tank group are interconnected by one-way flapper valves at the base for aft to forward fuel gravity transfer which also serve as inertia check valves for catapulting and arrestment.

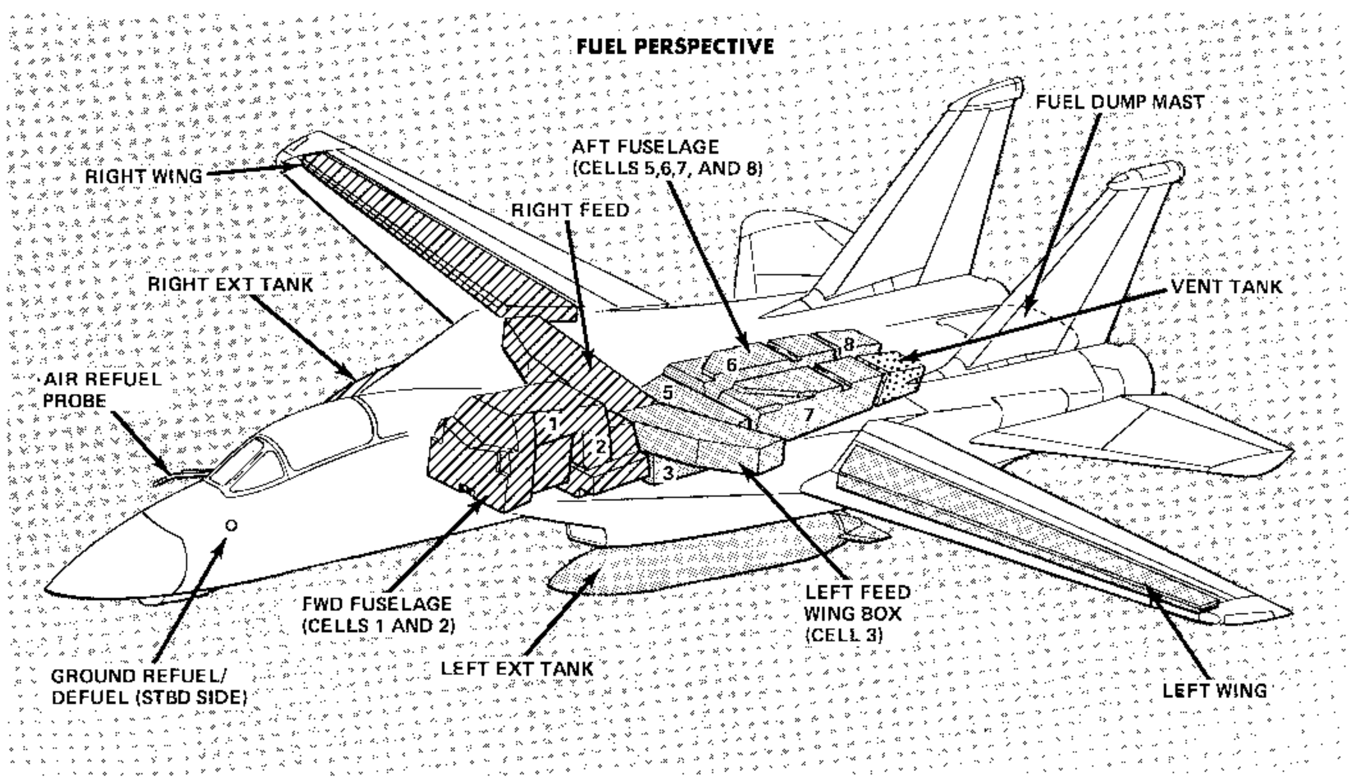
FUEL QUANTITY DATA TABLE

TANKS	APPROXIMATE USABLE FUEL	
	GALLONS	POUNDS
FWD FUSELAGE	662	4,500
AFT FUSELAGE	648	4,400
RIGHT FEED	235	1,600
LEFT FEED	221	1,500
TOTAL FUSELAGE	1,766	12,000
INTERNAL WING	600	4,000
TOTAL INTERNAL	2,365	16,000
EXTERNAL TANKS	530	3,600
TOTAL CAPACITY	2,895	19,600



NOTE

WEIGHTS ARE BASED ON 6.8 POUNDS PER GALLON OF JP-5 FUEL, STANDARD DAY CONDITIONS. USABLE FUEL BASED ON GROUND ATTITUDE 1.5 DEGREES.



1-F050-004
008-0

Figure 1-26.

Wing Tanks

(C) Integral fuel cells are provided in the movable wing panels between the front and aft wing spars. Each wing tank has a usable fuel capacity of approximately 2,000 pounds (300 gallons). Because of the wing sweep pivot location and the extensive span (20 feet) of the wing tanks, wing fuel loading provides a variable aft center of gravity contribution to the aircraft's longitudinal balance as a function of wing sweep angle (figure 1-27). Each wing panel comprises one integral fuel cell which is designed to withstand loads due to fuel slosh incident to catapulting, arresting and extreme rolling maneuvers with partial or full wing fuel. Fuel system plumbing (transfer/refuel, motive flow, vent and precheck lines) to the wing tanks incorporate telescoping sealed joints at the pivot area to provide for normal operation independent of wing sweep position.

External Drop Tanks

(C) Fuel, air, electrical, and fuel precheck line connectors are provided under the engine inlet nacelles for the external carriage of two fuel drop tanks. Check valves in the connectors provide an automatic seal with the tank removed. Although the location is designated as armament stations 2 and 7, no other store is designed to be suspended there so that the carriage of external fuel tanks does not limit the weapon loading capability of the aircraft. The submerged rack design for the tank is provided to minimize drag and provide adequate ground clearance. Suspension of the drop tanks and their fuel content provides an

CENTER OF GRAVITY VARIATION WITH FUEL LOADING

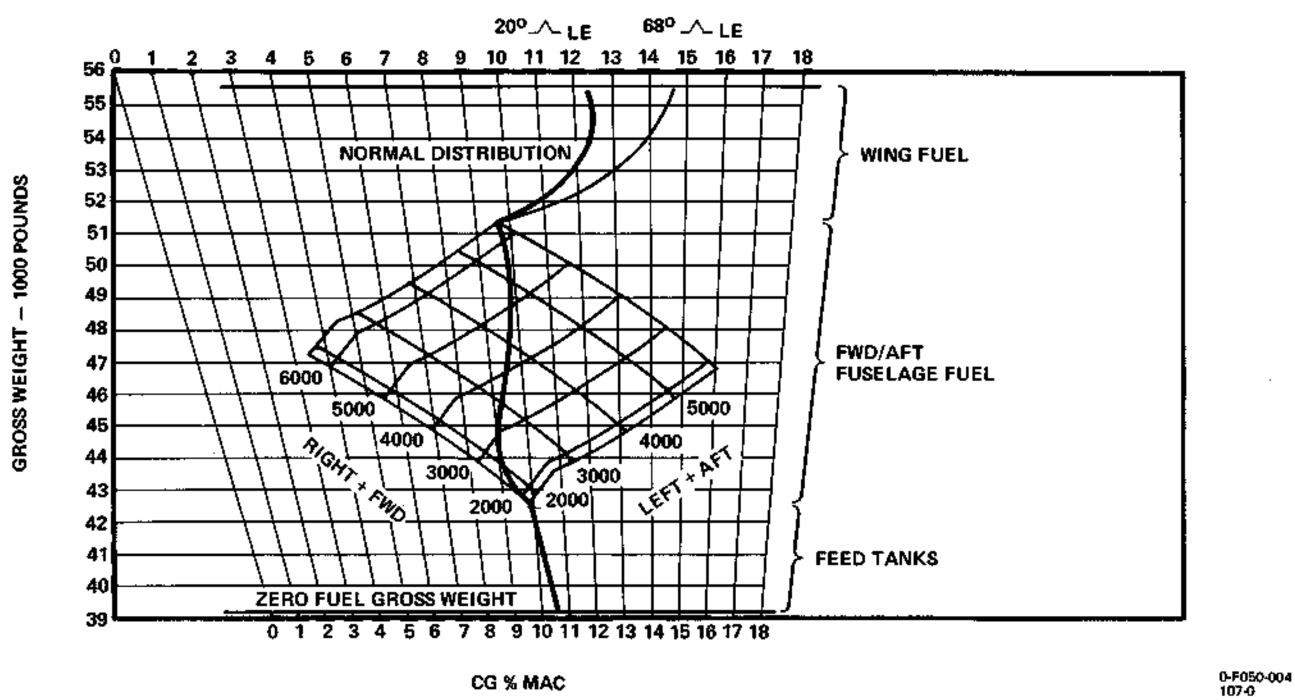


Figure 1-27.

insignificant effect on the aircraft's longitudinal center of gravity, and under the most adverse fuel asymmetry loading condition the resultant moment can be compensated by lateral trimming. The drop tank is cleared for carriage up 1.8M or_____KIAS and imposes no restriction on inflight maneuvering regardless of fuel loading. Tank separation (ejection mode) under level flight conditions is cleared for speeds less than_____. The usable fuel capacity of each external tank is approximately 1800 pounds (265 gallons). Catapult operations with partial fuel or arrestment with full or partial fuel in the drop tanks is not permitted due to surge load considerations.

Tank Pressurization and Vent

(U) The internal fuel system is a non-pressurized (except by ram air), open vent type system which incurs an insignificant fuel loss due to fuel vaporization or boil-off, even at high speed conditions. Engine bleed air (16^{th} stage) is regulated (25 psi) to pressurize the external fuel drop tanks during non-refueling operations with the landing gear retracted or during transfer override conditions with the landing gear extended. Except for the external fuel drop tanks, fuel transfer and dump operations are not dependent on pressurization for operation. A schematic of the fuel tank pressurization and vent system is presented in figure 1-28.

(U) Internal fuel tanks are continuously vented through an open vent system to interconnecting lines which terminate in a fuel vent tank. Venting of the wing box tanks is controlled by solenoid operated valves to provide for suction transfer through the inverted standpipes in cells #2 and #5. External drop tanks are provided with individual external vent outlets that are opened during gear down or refueling operations. The vent system prevents the build-up of excessive fuel cell-to-cavity pressure differential during climb or dive maneuvers and normal transfer, dump, refueling or defueling operations. Additionally, it prevents the formation of air pockets in the fuel tanks during refueling operations. A vent line pressure indicator is incorporated on the ground refuel panel for monitoring system pressure during ground refueling. The vent tank is an integral cell appended to the aft end of the left "coffin" cell (#7) in the aft tank group. The vent tank, which has a 50 gallon capacity, accepts fuel tankage overflow and spillage due to aircraft attitude effects or expansion characteristics while causing no spillage overboard until it overflows or is inverted. With the left engine running and the FUEL FEED switch in the NORM or AFT position, fuel in the vent tank is continuously purged by a scavenge ejector pump which transfers trapped fuel into the adjacent forward cell (#7) of the aft tank group. The vent tank cannot be filled under refueling or transfer operations unless a high-level pilot valve fails to shut off fuel to a fuel tank.

(U) A line extends from the high point in the vent tank to the forward part of the tailhook fairing to vent the fuel system overboard. The mouth of the overboard vent line in the tailhook vent plenum incorporates a flame arrester (polyurethane, foam netted) which inhibits propagation of a flame in the overboard vent upstream into the vent tank. A dual chamber vent is provided in the leading edge of the tailhook fairing to allow fuel drainage overboard and control vent back pressures under all flight conditions. In order to control the vent back pressure, only the forward chamber with its ram air and discharge port is used in low speed flight conditions. As the speed increases, spring loaded flaps open, exposing the second chamber and its larger discharge port, to prevent excessive back pressures in the vent system.

FUEL SYSTEM

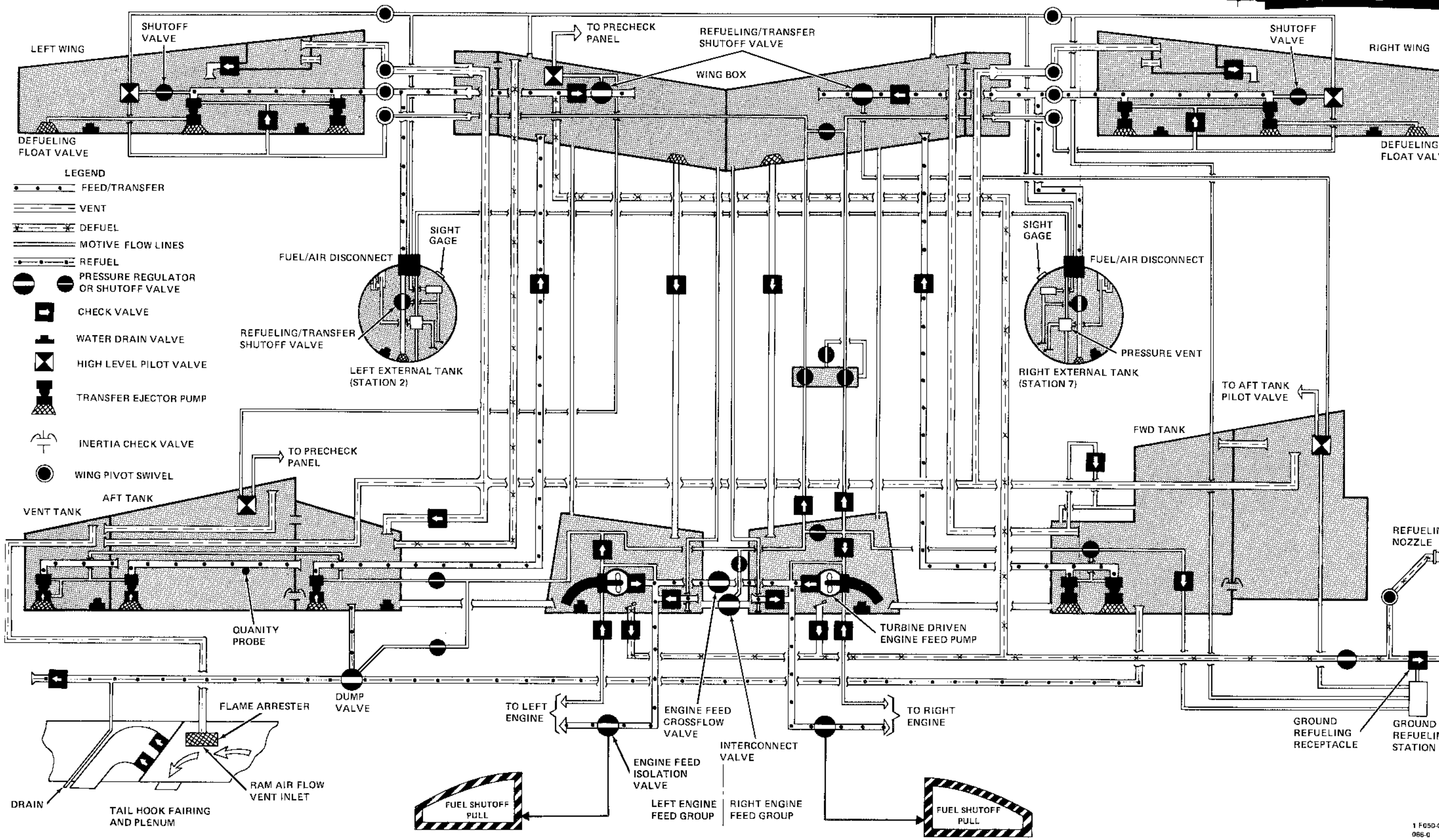


Figure 1-28.

Fuel Ejector Pumps

(U) The fuel ejector pumps use motive flow fuel to induce fuel transfer or scavenge by venturi action without the need for moving parts. As shown in figure 1-28, high pressure fuel is discharged through a nozzle into the transfer line. Fuel surrounding the ejector pump is induced to flow into the transfer line so that pump operation is solely dependent on motive flow fuel and an immersed inlet. The inlet contains a fuel strainer to prevent fouling of the pump. The pump transfer rate is predicted on the head pressure, inlet pressure and motive flow fuel supply, all of which are affected by aircraft attitude. In the absence of surrounding fuel, none of the motive flow fuel is recirculated back into the tank.

High-Level Shutoff Valves

(U) Discharge of wing and external fuel transfer into the feed tanks (wing boxes) is controlled hydro-mechanically by high-level pilot valves. A single high-level pilot valve located in the forward tank (cell #1) controls the shutoff valve in the right feed tank (wing box) and thereby prevents overfilling of the forward and right tank group. Two high-level pilot valves, one in the left feed tank (wing box) and the other in the aft tank (cell #7), control in series the shutoff valve in the left feed tank (wing box). Although normally the left and right wing and external tanks transfer fuel to the feed tank on the respective side, a restrictor valve is provided in the interconnecting line to enable fuel crossflow for replenishment of fuselage fuel. These same high-level shutoff valves are used to prevent fuselage overfilling during refueling. Additionally, hydro-mechanical high-level shutoff valves are provided in the wing and external tanks to prevent overfilling during refueling.

Tank Drains

(U) As illustrated in figure 1-28, spring loaded, water drain valves are provided on the underside of the fuselage for forward, sump, and aft cavity drain to prevent accumulation of fuel in the event of a cell bladder leak. Overboard drain holes are also provided for the air refueling probe cavity with the discharge ports located on the underside of the right forward fuselage.

FUEL QUANTITY SYSTEM

(U) The fuel quantity measurement and indication system provides the flight crew with a continuous indication of total internal and external fuel remaining, a selective readout for all fuel tanks, independent low fuel detection, and automatic fuel system control features to prevent mismanagement by the pilot. Figure 1-26 tabulates fuel quantity data, correlates these with the cockpit quantity displays and provides a perspective view of the tankage arrangement. The quantity measurement system uses dual element, capacitance type fuel probes and intermediate compensator devices to provide the crew with a continuous display of fuel quantity remaining. Fuel thermistor devices and caution light displays are used as a back-up FUEL LOW level indicating system independent of the capacitance type gaging system. Additionally, the pilot is provided with a BINGO set capability on the fuel quantity indicator to preset the total quantity level for activation of a BINGO caution light.

Fuel Capacitance Probes

(U) Dual element fuel probes are positioned vertically in the fuel cells and, either by locating them through the centroid of the cell or through the use of multiple probes, fuel quantity gaging errors are minimized as a function of aircraft attitude. Usable fuel that may be momentarily contained in the vent tank is not gaged. An electrical bridge circuitry measures changes in probe capacitance as a function of the probe immersed length. Through intermediate devices the system is balanced to provide an accurate display of fuel quantity (calibrated in pounds) aboard. The dual element probes provide the crew with a functionally independent total quantity indication separate from the individual tank display source. The pilot can check the accuracy of the capacitance measurement and indication system by comparison of the total fuel quantity reading with the summation of individual tank readings. Pilot and NFO totalizer readings are measured from the same set of fuel capacitance probes. The system accuracy is a function of the aircraft attitude and total fuel aboard. It should be recognized that the redundant measurement system can be degraded by the influences of fuel contaminants such as water, bacterial growth, fuel additives, tank cleaner, etc.

Fuel Quantity Indicators

(U) The pilot's and NFO fuel quantity indicators are illustrated in figure 1-26 as well as a definitive breakdown of tape and counter readings. The white vertical tapes on the pilot's indicator provide a peripheral viewing instrument for fuselage fuel quantity directly available for engine consumption and, by the height differential, a relative indication of longitudinal fuel balance for center of gravity considerations. The L and R labeled counters display either feed (wing box and sump) tank, wing tank or external drop tank fuel quantity, on the respective side, as selected using the QTY SEL rocker switch on the fuel management panel. The rocker switch spring returns to the FEED position when not held at either extreme position. The pilot's TOTAL quantity display and the NFO's display indicate total internal and external fuel aboard. The totalizer reading is valid only on the premise that all fuel is transferable and will be available for engine feed. As an adjunct to the pilot's fuel quantity indication, a BINGO set knob and counter is provided to trigger the illumination of a BINGO caution light on the pilot's annunciator panel whenever the pilot's total fuel quantity reading is lower than the preset BINGO counter valve. The BINGO level may be reset any number of times during a flight.

Fuel Thermistor Devices

(U) Four thermistors are provided in the aircraft fuel system (forward, left wing box, right wing box and aft tank) as shown in figure 1-28. They are independent devices attached to the fuel quantity probes that pass a discrete electrical signal when uncovered but cause an open circuit when submerged in fuel. Operation of the thermistors is less subject to the influence of fuel contaminants than the capacitance type probes. The discrete levels of the thermistors in these cells are set to provide a separate series signal from the split pairs (aft and left wing box tank, forward and right wing box tank) for a FUEL LOW indication at 1,000 pounds remaining in either the left or right engine feed group. The use of dual thermistors in the left and right engine feed group is necessary because the individual feed (wing box and sump) tank level varies with the rate of engine fuel consumption and fuel remaining. In addition to the low fuel signal function, signals from the thermistors in the forward and aft tanks (cells #2 and #5) automatically and independently open the feed tank interconnect valve, open the fuselage motive crossflow shutoff valve, and release (for 5 seconds) the solenoid which holds the WING/EXT TRANS switch in the OFF position (when selected). By these automatic actions the feed tank levels will be balanced at low fuel conditions and an engine flameout due to fuel starvation during twin or single engine operation (non-afterburning) will not accur with fuel remaining in the forward or aft fuselage tanks, wing, or external drop tanks (with landing gear handle up).

FUEL LOW Caution Lights

(U) Uncovered thermistor devices in the aft and left wing box tanks or the forward and right wing box tanks provide the pilot with a discrete L FUEL LOW or R FUEL LOW caution light indication, respectively. Illumination of either light should occur with 1,000 ± 200 pounds fuel remaining displayed on the applicable vertical tape and feed tank quantity indication. The NFO is provided with a single FUEL LOW caution light that illuminates when either the left or right engine feed group is at 1,000 pounds fuel remaining.

Quantity Indication Test

(U) Actuation of the Master Test switch in the INST position causes the pilot's and NFO's fuel quantity indicators to drive to 2,000 pounds and illuminates the FUEL LOW caution lights on the pilot's and NFO's annunciator panels. The test is performed to check operation of all display drive units, check the circuitry downstream of the intermediate devices, and, upon completion of the test, check that measurement indications are consistent by observing the return to valves displayed prior to test activation. Such a test can be performed on the ground or inflight. The test does not check the fuel probes or the thermistor devices and their circuits. A test of the BINGO set device can be concurrently obtained with the INST test if the BINGO level is set at greater than 2,000 pounds; in such a case the BINGO caution light will illuminate when the fuel totalizer reading decreases to a value less than the BINGO setting.

CENTER OF GRAVITY EFFECTS

(U) Tankage, automatic distribution and sequencing of the aircraft's fuel system is designed to minimize excursions in center of gravity during transfer, dump, and refueling operations. Figure 1-27 shows the longitudinal center of gravity excursions that occur during the fuel burn sequence with normal transfer sequence and distribution. The extremes (±6% MAC) in center of gravity variation with an unbalance in forward and aft fuselage fuel is also indicated on the grid. The significance of these characteristics, in conjunction with store loading and aerodynamic contributions, must be taken into account from an aircraft maneuverability, controlability, and stability viewpoint.

FUEL SYSTEM OPERATION

Engine Feed System

(U) A turbine driven engine feed pump is submerged in each sump tank to provide boosted fuel pressure (10 to 50 psi) to each engine. The feed pump discharge capacity is sufficient to supply boosted fuel pressure to the engine main fuel pump, afterburner fuel pump, and motive flow pump under all operating conditions. The engine driven motive flow pumps draw fuel from each engine feed line and produce a supply of high pressure (maximum 400 psi) fuel which is used to operate the boost pumps and transfer system. This fuel drives the turbine in the boost pump which is shafted to the feed pump impeller. The motive flow continues through the turbine and controlling valves to the fuselage and wing ejector pumps. The feed pump is provided with a flexible inlet which seeks the fuel content of the sump tank under various aircraft "g" conditions. One-way check valves are provided on the feed pump discharge line and the feed pump bypass inlet to permit unrestricted fuel flow to the engine with either one or both feed pumps inoperative. Although normally operating independently as a split feed system, pump failures automatically cause a valve to open in the crossfeed line interconnecting the discharge ports of both feed pumps with both engines operating. This causes one feed pump to automatically supplement the other or supply fuel boost pressure to both engines without imposing any main engine operating restriction. However, at altitudes above 15,000 feet, a boost pump failure limits afterburner operation to zone 1 throttle settings. Operation of the fuel crossfeed is automatically initiated by either feed tank boost pump discharge pressure switch which also illuminates the respective L or R FUEL PRESS caution light on the pilot's caution/advisory panel.

(U) Individual engine fuel feed shutoff valves are positioned in the left and right feed lines at the point of nacelle penetration. These valves are connected by control calbes to the FUEL SHUTOFF handles on the pilot's instrument panel. During normal operation, the handles should remain pushed in so that fuel flow to the engine fuel feed system is unrestricted. If a fire is detected in the engine nacelle, the pilot should pull the FUEL SHUTOFF handle on the affected side to stop the supply of fuel to that engine. No other action is required to reset the shutoff valves to the normal open position than pushing in the respective FUEL SHUTOFF handle.

Fuel Feed

(U) Figure 1-28 illustrates aircraft fuel system components associated with fuel feed operations. Engine fuel feed is normally (except under crossfeed operation) supplied from the feed (sump) tank on the respective side. Under normal (split feed) operation, the turbine driven engine feed pump in each of these tanks supplies fuel to its respective engine and the cross-feed valve is closed. Each feed tank is continuously replenished by motive flow transfer provisions and remains full under non-afterburner fuel flow demands until depletion of transferable fuel. At high fuel feed conditions characteristic of afterburner demands the feed tank quantity may decrease prematurely and the motive flow transfer must be augmented by gravity transfer. Failure of a feed pump to supply discharge pressure is normal with either engine shutdown and can also occur due to a malfunction (sump tank fuel depletion, motive flow system failure, feed pump turbine or impeller failure). If one of the above failures does occur, the crossfeed valve will be open so that the feed pump in the other sump tank supplies boosted fuel pressure to both engines. If twin engine fuel flow demands exceed the flow capacity of the single feed pump, the pump by-pass valves in either sump tank will allow suction feed. Thus, under failure conditions, engine feed continues uninterrupted, but the total engine feed will now come from either the forward or aft tank leading to a fuselage fuel unbalance. To correct this unbalance a fuel feed control system is provided.

(U) Cockpit control of fuel feed is provided by the FUEL FEED switch on the fuel management panel. Basically, this switch controls the sequence of fuel consumption between the forward and aft tanks and thereby the distribution which effects the aircraft's longitudinal center-of-gravity; it does not alter the conditions for each engine feeding from its respective sump tank. In the guarded NORM position, the feed tank interconnect valve is normally closed subject to automatic opening by the uncovering of either thermistor device in the forward (cell #2) or aft (cell #5) fuel tanks. Selection of either FWD or AFT positions on the FUEL FEED switch opens the feed tank interconnect valve and shuts off motive flow fuel to and stops transfer from the ejector pumps in the aft and forward tanks, respectively. These selections are not normally required and are provided to enable control over the fuel distribution between the forward and aft tanks for abnormal conditions which dictate corrective actions. Since the ejector pumps in the tank (forward or aft) other than that selected are deactivated, ejector pumps in the tank selected attempt to maintain both feed (wing box and sump) tanks full. This dictates that all fuel consumed comes from the selected tank. Regardless, the feed tanks are still provided with a gravity fuel transfer system from their respective fuselage (forward or aft) tanks. The amount of engine fuel feed available under twin or single engine operation is not altered because of FUEL FEED selection except by attitude effects on gravity transfer. The FUEL FEED circuit is energized in the FWD and AFT positions by the essential dc No. 2 bus. In the NORM position and with power off, the motive flow shut-off valves are deenergized to the open position and the feed tank interconnect valve to the closed position.

Fuel Transfer

(U) Figure 1-28 illustrates aircraft fuel system components pertinent to fuel transfer operations. Fuel is transferred from the left and right wing and external drop tanks to the wing box tank on the respective side; if the feed tank is full, the fuel overflows to the aft and forward tanks respectively. Wing and fuselage tank transfer is effected by fuel ejector pumps; whereas the drop tanks use pressure regulated engine bleed air from the environmental control system to force fuel transfer. Figure 1-29 provides a matrix of transfer conditions selectable on the fuel management panel. The wing and fuselage ejector pumps are powered by the same motive flow fuel used to run the turbine driven engine feed boost pumps.

Forward Fuselage Fuel Transfer

(U) Fuel transfer from the forward tank (cell #2) into the right feed (wing box) tank is accomplished by two ejector fuel pumps which collectively have a maximum pumping capacity of 10,000 pounds per hour. In addition, a gravity transfer line interconnects the forward tank (cell #2) with the right feed (sump) tank to supplement the ejector transfer pumping capacity. An electrically actuated motive flow shut-off valve in the forward tank controls operation of the ejector transfer pumps. Selecting the AFT position on the FUEL FEED switch supplies electrical power to shut off motive flow fuel to the forward tank ejector pumps; however, when NORM or FWD is selected, the valve is open and the power deactivated.

FUEL TRANSFER/DUMP MANAGEMENT

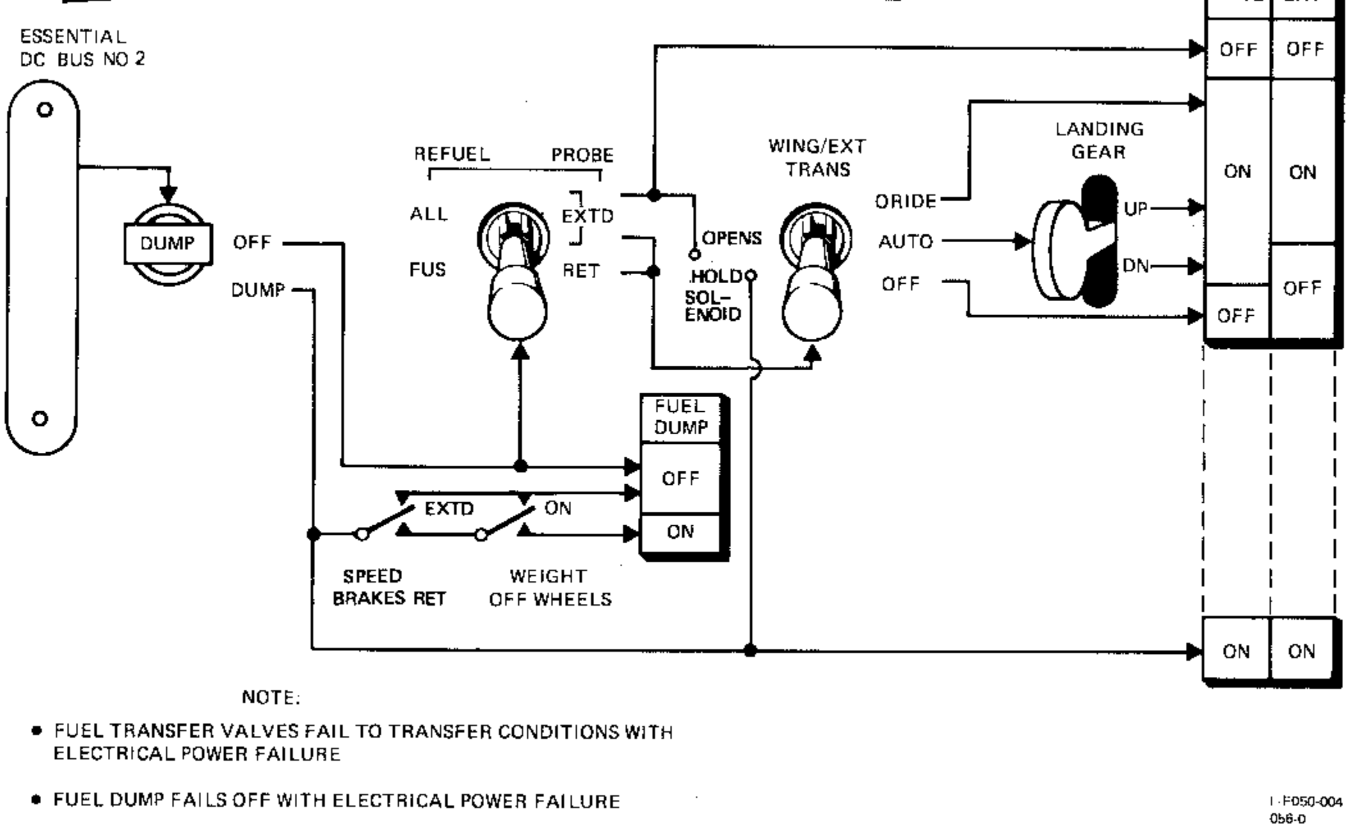


Figure 1-29.

Aft Fuselage Fuel Transfer

(U) Fuel transfer from the aft tank (cell #5) into the left feed (wing box) tank is accomplished by two ejector transfer fuel pumps which collectively have a maximum pumping capacity of 10,000 pounds per hour. In addition, a gravity transfer line interconnects the aft tank (cell #5) with the left feed (sump) tank to supplement the ejector transfer pumping capacity. Inter-cell motive transfer of fuel in the aft tank, which is effected by two ejector scavenge and two ejector transfer pumps, serves to empty the most rearward cells in sequence by transferring fuel to the next forward cells. Two low capacity ejector pumps in the aft most cells (#8 and vent) scavenge fuel forward into adjacent cells (#6 and #7, respectively). Two ejector pumps in the intermediate cells (#6 and #7) in turn transfer fuel forward to the common adjacent cell (#5). An electrically actuated motive flow shut-off valve in the left sump tank controls operation of the aft tank and vent tank ejector pumps. Selection of the FWD position on the FUEL FEED switch supplies electrical power to shut off motive flow fuel to the aft and vent tank ejector pumps; however, when NORM or AFT is selected, the valve is open and the power deactivated.

Wing Fuel Transfer

(U) Fuel transfer from the left and right wing tanks to the feed (wing box) tank on the respective side is accomplished by two ejector transfer fuel pumps and one scavenge pump (at the tip) in each wing which provide a nominal transfer capacity of 300 pounds per minute per side. An electrically actuated wing motive flow shut-off valve in the forward tank controls operation of all wing ejector transfer pumps. With the WING/EXT TRANS switch in the AUTO (normal) position the wing motive flow shut-off valve is open with power deactivated until automatically energized closed upon depletion of fuel in both wing tanks to reduce unnecessary recirculation of motive flow fuel and attendant foam introduction to the feed tanks. This automatic valve shut-off feature is energized by the uncovering of fuel thermistors at the wing tips and can be overridden by pilot selection of ORIDE on the WING/EXT TRANS switch. Selection of OFF on the WING/EXT TRANS switch directs electrical power to close the valve and shut off motive flow fuel to the wings independent of wing fuel quantity. Activation of fuselage fuel dump automatically initiates wing fuel transfer in sequence after external drop tank transfer by automatically moving the WING/EXT TRANS switch to the AUTO position if selected OFF. Positioning the REFUEL PROBE switch to the ALL/EXTD position also releases the solenoid holding the WING/EXT TRANS switch in the OFF position. Replenishment and transfer of wing tank fuel is accomplished through a common line.

External Fuel Transfer

(U) Fuel transfer from the external drop tanks is accomplished by engine bleed air extracted downstream of the primary heat exchanger which is pressure regulated to approximately 25 psi. Air pressure to left and right drop tanks forces fuel to transfer out an inverted standpipe through the drop tank refueling/transfer shut-off valves to the feed tanks (wing boxes). The maximum fuel transfer rate from the drop tanks is approximately 750 pounds per minute per side. Fuel transfer is automatically sequenced with the landing gear retracted so that the external tanks transfer before the wing tanks. A low level pilot valve at the bottom of each drop tank hydraulically shuts off the transfer control valve with fuel depletion to prevent air circulation through the transfer lines. Activation of fuselage fuel dump automatically initiates drop tank fuel transfer with the landing gear retracted in sequence prior to wing tank transfer.

(U) Fuel transfer sequence under normal gear up conditions causes the external tanks to complete transfer before the wing tanks transfer. With the landing gear down, under normal conditions the drop tank transfer is inhibited; whereas, the wing tank transfer is uninhibited. Independent of manual selections the uncovering of the fuel thermistor device in the aft (cell #5) or forward (cell #2) tank automatically releases (for 5 seconds) the holding solenoid for maintaining the WING/EXT TRANS switch in the OFF position; the switch spring returns to the AUTO position. This feature minimizes the probability of causing engine fuel starvation with usable fuel aboard due to mismanagement of the fuel switches. The 5 second time delay enables reselection of the OFF transfer condition in the event of a wing motive flow line failure or battle damage in the wings.

(U) External drop tank transfer can be checked on the deck by placing the WING/EXT TRANS switch to the ORIDE position and noting depletion of external fuel quantity. Although this operation is a valid test of drop tank transfer integrity on both sides, it is time consuming with a full fuselage fuel load and aggravates fuel slosh loads in the drop tank during catapulting. In lieu thereof, a normal pre-takeoff check of drop tank transfer is provided by depressing the Master Test switch in the FLT GR UP position and observing the GO/NO GO lights on the Master Test panel. A green GO light indicates that the common external fuel drop tank line is pressurized; the pressure switch providing GO/NO GO intelligence is internal to the aircraft, not in the tanks. Since the FLT GR UP position serves to bypass the landing gear down interlock in the drop tank transfer circuit, the WING/EXT TRANS switch may remain in the AUTO (normal) position for this check.

Fuel Dump

(U) Figure 1-28 illustrates aircraft fuel system components associated with fuel dump operations. Fuel dump standpipes located in the forward (cell #2) and aft (cell #5) fuselage tanks are connected to the fuel dump manifold at the dump shut-off valve. The manifold extends aft to the fuselage boattail where it incorporates a flame arrestor on the dump mast. Actuation of the FUEL DUMP switch to the DUMP position supplies power (essential dc No. 2) to open the solenoid operated pilot valve, which ports motive flow fuel pressure to hydraulically open the dump shut-off valve with weight off the main landing gear and the speed brakes retracted. The FUEL DUMP switch circuit is deactivated on the deck or with speed brakes extended. Fuel dump with the speed brakes extended is inhibited due to the resulting flow field disturbance which would result in fuel impingement on the fuselage boattail and nozzles. Dump operations with either engine in afterburner should be avoided since the fuel dump mast discharge can be torched. A flame arrestor is provided at the mast discharge so as not to endanger the aircraft in such an occurrance. Dump mast fuel drain is provided through the mast at positive pitch attitudes with a supplemental residual fuel drain line at the base of the tail hook attach fairing. Fuel in the wings and drop tanks is dumped by transferring to the fuselage. When the fuselage fuel dump circuit is activated, wing and external drop tank transfer to the feed tanks (wing boxes) is automatically initiated. Fuel dump is accomplished by gravity flow with a nominal discharge rate of 1500 pounds per minute. The dump rate is affected by aircraft pitch attitude and total fuselage fuel quantity with discharge flow inhibited at nose down conditions. The standpipes in the fuel cells control the minimum fuel dump level in the tanks, which under normal operations (feed tank full) is approximately 4,000 pounds.

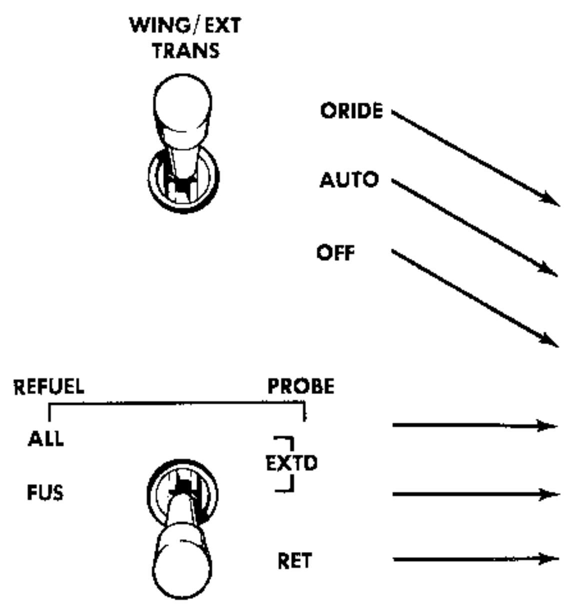
Ground Refueling

(U) The aircraft is equipped with a single-point refueling system which enables pressure filling of all aircraft fuel tanks from a single receptacle. The receptacle is located at the recessed ground refuel/defuel station behind a quick access door on the lower right side of the forward fuselage. The maximum refueling rate is 3,875 pounds per minute (425 gpm) at a pressure of 57 psi. A schematic of the aircraft fuel system associated with ground refueling is shown in figure 1-28.

(U) The ground refueling receptacle connects to the refueling manifold that divides downstream to provide split refueling of the fuel tanks, which comprise the left and right engine feed group, L and R wing, and L and R external tanks. Since ground and air refueling connections use a common manifold, the refueling sequence is the same except for the mode of control. Ground refueling control is provided by two precheck selector valves and a vent pressure gage adjacent to the refueling receptacle on the ground refuel/defuel panel. The precheck valves functionally test high-level pilot valve operation incident to ground pressure refueling; the valves separately check the pilot valves in the fuselage tanks and the wing external tanks. In addition to this precheck function, the precheck valves can be used for ground selective refueling of only the fuselage or all tanks. Since the precheck valves, which are manually set by the ground crew, port pressurized servo fuel to the high-level pilot valves and subsequently to the shut-off valves, no electrical power is necessary on the aircraft to perform ground refueling operations. Additionally, ground refueling control without engines running is completely independent of switch positioning on the fuel management panel. The direct reading vent pressure indicator is provided as a monitor of pressure in the vent lines. The gage consists of a pointer on a scale having two bands, one green and one red. The green band indicates a safe pressure range (0 to 4 psi) and the red band indicates an unsafe range (4 to 8 psi). Refueling should be discontinued if the indicator is in the red band.

(U) Hot refueling (ground pressure refueling with the left engine running) may be accomplished by observing standard safety precautions; for such operations refer to Section III, Normal Operating Procedures. Control of fuel distribution to the tanks during hot refueling may be exercised by the pilot or the ground crewman; however, only the ground crewman is able to monitor the vent line pressure and exercise precheck control over the fuselage high-level pilot valves. Cockpit fuel management switch positions for hot refueling control are shown in figure 1-30.

HOT REFUELING SWITCH POSITIONS



	HOT REFUELING		
	PILOT CONTROL		GND CREW CONTROL
	FUS ONLY	ALL TANKS	
ORIDE	✓		
AUTO			
OFF		✓	✓
ALL			
FUS / EXTD			
RET	✓	✓	✓

1-F050-004
051-0

Figure 1-30.

(U) Fuselage refueling and wing/external fuel transfer/defuel lines terminate in each side of the wing box. The common port in the left and right wing box tanks for these manifolded lines contains a fuselage refueling shut-off valve which is hydraulically operated and independently closed by servo fuel pressure from the high-level pilot valves in the left/aft and forward fuel tanks. Standpipes are provided to refuel the aft and forward fuselage tanks by overflow from the left and right wing box tanks. Replenishment of fuel to the forward fuselage tank is accomplished by an overflow line from the right wing box incident to refueling or wing/external tank transfer. A high-level pilot valve is positioned at the high point of the forward tank to hydraulically shut off the fuselage refueling valve in the right wing box when the forward tank group is full. Replenishment of fuel to the aft fuselage tank is accomplished by an overflow from the left wing box incident to refueling or wing/external tank transfer. The fuel flows to the most forward aft cell whereafter it overflows to the right side, then the left side. High-level pilot valves are positioned at the high point of the left wing box tank and aft tank (cell #7) to hydraulically shut off the fuselage refueling valve in the left wing box when the aft tank group is full. Individual wing and external tank filling is accomplished by flow through a shut-off valve located in each tank.

(U) Gravity refueling of the aircraft fuel system should be accomplished under emergency situations only. Care should be exercised in performing such an operation to avoid introducing contaminants into the fuel tanks or damage to the fuel quantity probes and wiring.

Air Refueling

(U) The air refueling system permits partial or complete (1000 pounds below shut-off) refueling of the aircraft fuel tanks while in flight. A retractable refueling probe is provided with an MA-2 nozzle which is compatible with any drogue type refueling system. A split refueling system is provided with fuel routed into the left and right feed tanks (wing boxes) for initial replenishment of feed tank fuel. Selectable fuel management controls dictate the extent of further fuel distribution to the wing tanks, external drop tanks and/or fuselage tanks. The maximum refueling flow rate is approximately 2,600 pounds per minute at a pressure of 50 psi. A schematic of the aircraft fuel system associated with an air/refueling is shown in figure 1-28.

(U) The retractable air refueling probe is housed in a cavity located on the right side of the forward fuselage section immediately forward of the pilot's right vertical console panel. The probe cavity is a sealed compartment from the surrounding forward avionics bay. It incorporates low point drains which vent overboard at the right main door of the nose landing gear to prevent the accumulation of fuel in the event of a nozzle leak. The probe mast has a swivel fuel fitting at its aft end to form a pivot. A hydraulic actuator powered by the combined hydraulic system (operates from the hydraulic hand pump in the pilot's cockpit as a back-up power source) is contained within the probe cavity to extend and retract the probe. Shaping of the probe mast provides the proper alignment geometry in the extended position. A forward recycling door and a fairing attached to the aft portion of the probe mast totally enclose the probe in the retracted position within the fuselage mold line. The forward door, which is hinged along the upper side, is mechanically cammed to recycle closed with the probe in both the retracted and extended positions. A red light, located immediately above the REFUEL PROBE switch on the fuel management panel, illuminates whenever the probe cavity forward door is not in the closed position. Since the closed door position is indicative of both the probe retracted and extended position, the light serves as a probe transition indicator as well as a terminal status indicator. Extension of the probe cams the external red probe light, which is contained within the cavity, into a properly oriented position at the base of the probe mast. Illumination of the probe external light is automatic upon probe extension with the EXT LTS master switch ON.

(U) The fuel line from the air refueling probe swivel fitting incorporates a one-way check valve prior to its juncture with the refueling manifold adjacent to the ground refueling station. Regardless of fuel management panel switch positioning, at low fuel states the initial resupply of fuel is discharged into the left and right feed tanks (wing boxes). Thereafter, distribution of the fuel to the forward, aft, wing, and external drop tanks is controlled by the fuel management switch position matrix shown in figure 1-25. The split refueling system to the left and right engine feed group of tanks provides for a relatively balanced center of gravity condition during refueling. Selective refueling of the FUS (fuselage) or ALL tanks is provided on the REFUEL PROBE switch with the probe extended. In the FUS/EXTD position, normal fuel transfer and feed is unaltered. This position is used for practice plug-ins, fuselage only refueling, or return flight with a damaged air refueling probe. The ALL/EXTED position shuts off wing and external drop tank transfer to permit the refueling of all tanks. Two electrical extend solenoids are provided as a redundant means of controlling operation of the air refueling probe; one is activated in each of the probe EXTD positions. The REFUEL PROBE switch circuit uses essential dc No. 2 power to control operation of the probe actuator through redundant extend solenoids and a single retract solenoid. Overfilling the fuselage tanks is automatically prevented by high-level pilot valves in the left feed/aft and forward fuselage tanks, which individually shut off the fuselage refueling line discharge ports in the left and right wing box tanks with full fuel quantity in the tank groups. Likewise, the wing and drop tanks each incorporate high-level pilot valves and refuel line shut-off valves to stop refueling of the individual tanks when full.

Ground Defueling

(U) The aircraft fuel system provides the ground crew with the capability for defueling all tanks. The defueling hose attachment fitting, which is the same receptacle used for connection of the ground refueling hose, is located on the ground refuel/defuel panel on the lower right side of the forward fuselage. A schematic of the aircraft fuel system associated with the ground defueling is shown in figure 1-28.

(U) Fuel suction without electrical power is all that is needed to defuel the aircraft. A defuel enable selector valve is provided at the refuel/defuel panel to port suction pressure from an external source into a pilot line to open the defueling check valve. Defueling lines to the manifold originate in each sump and wing tank and incorporate float valves over each inlet to permit suction defueling without sucking air. Forward and aft tank fuel flows into the right and left sump tanks, through gravity transfer ports interconnecting the cells. Defueling of the fuselage is best accomplished with the nose strut in the extended position to provide minimum residual fuel. Drop tank transfer to the feed tanks for defueling may be accomplished by connecting a high pressure air source to an adaptor in the external drop tank pressurization line or the drop tanks may also be defueled individually by fuel suction through an external access on the tank.

ELECTRICAL POWER SUPPLY SYSTEM

(U) Internally generated power to the aircraft electrical system is normally supplied by redundant engine-driven generators which provide ac (alternating current) power. The conversion of ac to dc (direct current) power is accomplished by redundant transformer rectifier units. A single or collective failure in ac and dc power sources does not restrict operations which impose a full electrical system load. Additionally, a hydraulic-driven emergency generator provides an independent back-up supply of both ac and dc power for electrical operation of flight essential components. Ground operation of all electrically powered components is also provided through the supply of external ac power to the aircraft. Electrical power is distributed to the aircraft systems by buses which are tied to the various power supply sources. All electrical circuits are protected by circuit breakers which are accessible inflight within the pilot's and NFO's cockpits. Switching between power supply systems is automatically accomplished without pilot action; however, sufficient control is provided for the flight crew to selectively isolate power sources and distribution in emergency situations. Refer to figure 1-31 for a functional description of the control switches.

POWER SUPPLY

Main Generators

(U) Two engine-driven, oil-cooled integrated drive generators (IDG's) provide the normal source of electrical power (115/200V ac, 400 Hz, 3-phase) developed in the aircraft. The normal rated output of each generator is 75 kVA which is sufficient to individually assume the complete electrical load of the aircraft. One IDG, consisting of a generator and a constant speed drive (CSD), is mounted on the accessory gear box of each engine. The CSD is a hydro-mechanical device which uses a self-contained oil system as the hydraulic medium to regulate the generator at a constant speed for maintaining the 400 Hz frequency. The oil is also used as a cooling and lubricating medium for both the CSD and the generator. The output of the CSD is capable of turning the generator at a sufficient speed to develop normal prescribed electrical power at engine speeds greater than approximately 48% N_2 speed. Fail safe provisions within the CSD cause the speed governor to fail to the underspeed condition. Cooling of the IDG oil is provided by a combined fuel-engine oil-IDG oil heat exchanger. Should an excessive amount of heat be developed in an IDG, a thermal (400°F) actuated device will automatically decouple the drive clutch which immobilizes the IDG. There are no provisions for recoupling the IDG unit in flight. The servicing level of the IDG unit is visible through a sight gage on the unit housing and servicing is accomplished through a pressure or gravity filler. A magnetic plug in the CSD oil sump is used for detecting metallic particles in the circulating oil, and filters are provided to prevent recirculation of solid contaminates.

GENERATOR PANEL

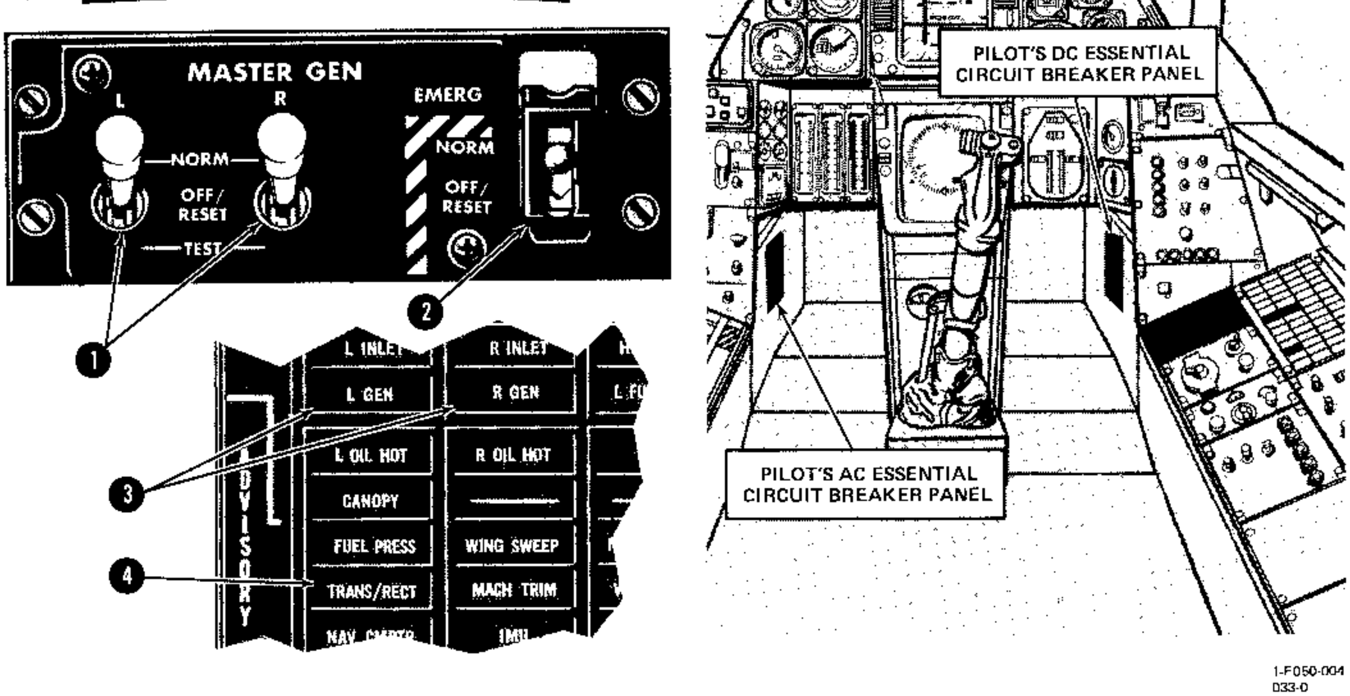


Figure 1-31.

	NOMENCLATURE	FUNCTION	
1	GENERATOR SWITCH	NORM -	connects the generator to the main buses through the line contractors.
		OFF/RESET -	disconnects generators from the buses. Resets the generator if tripped by an overvoltage, under-voltage, or fault condition.
		TEST -	the generators are energized but are not connected to the buses. Provides a means to analyze a system malfunction indicated by a generator caution light when an attempt to reset a generator is unsuccessful.
2	EMERGENCY GENERATOR SWITCH	NORM -	safety guard down. Emergency generator is automatically connected to the essential buses if the primary or secondary power sources fail.

NOMENCLATURE	FUNCTION
	OFF/RESET - safety guard must be lifted. Disconnects the emergency generator from the essential buses. Resets the generator if tripped by an under voltage or under frequency condition.
(3) L GEN/R GEN LIGHTS	Generator caution lights are located on the pilot's caution/advisory light panel. Each light is tied to its respective main ac contactor and is powered by the essential bus No. 2. Failure in the ac primary power source will cause a light to come on.
(4) TRANS/RECT LIGHT	A transformer/rectifier green advisory light is located on the lower half of the pilot's caution/advisory indicator panel. The light will come on if either transformer/rectifier malfunctions.

(U) Generator output voltage and frequency are individually monitored by generator control units (GCU) which prevent application of internally generated power to the aircraft bus system until the generator output is within prescribed operating limits. Indication of a main power supply malfunction is provided by a GEN caution light for each side (left and right). Each main ac generator is controlled by a separate switch on the pilot's master generator control panel. With the switch in the NORM position, the applicable generator is self-excited so that during the engine start cycle it automatically comes on the line at approximately 48% N_2 speed under normal load conditions. Likewise during engine shutdown, the GCU automatically trips the generator off the line as the power output decreases below prescribed limits at approximately the same engine speed. During normal operations the generator control switches remain in the NORM position continuously. However, subsequent to an engine shutdown/stall/flame-out in flight where the GCU has tripped the generator off the line, the reattainment of normal engine operation will not automatically reset the generator on the line unless the engine speed decreases below about 30% N_2 speed (GCU maximum auto reset speed). A transient malfunction or condition which causes the generator to trip must be manually reset by cycling the applicable generator control switch to the OFF/RESET position then back to NORM. In the OFF/RESET position, the generator is not excited, the power contactor to the aircraft buses is open and the GCU is reset. When normal reset cannot be accomplished, the TEST position on the generator control switch, which allows the generator to be excited but not connected to the aircraft buses, is used for fault isolation between the power supply (CSD, generator or GCU failure causes the GEN light to remain illuminated) and the power distribution circuits (GEN light to go out).

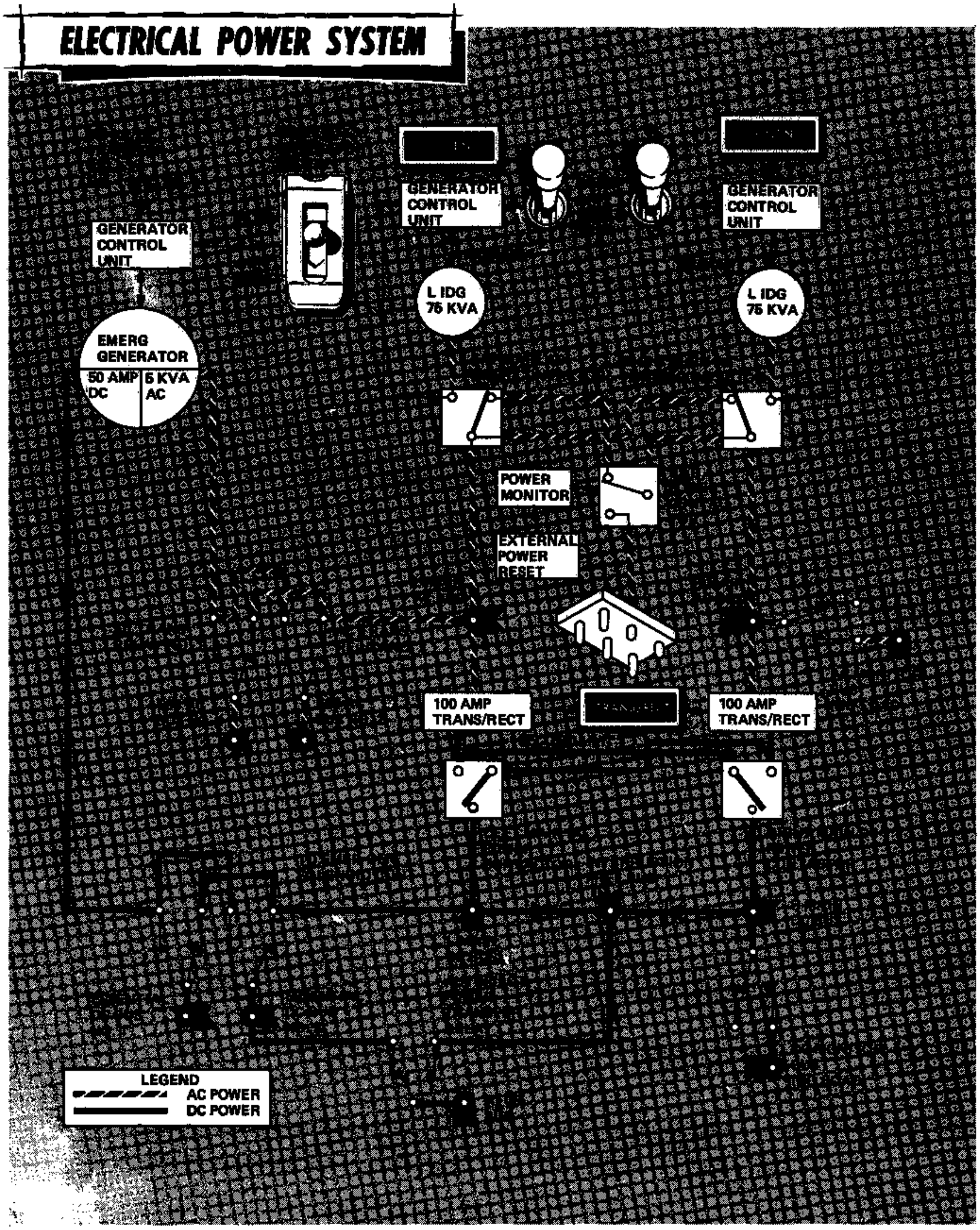
Transformer Rectifiers

(U) The operation of the two transformer rectifier units, which convert 115/200 V ac to 28 V dc power, is dependent on the availability of an ac power supply (internal) or external). A single TRANS/RECT advisory light on the pilot's annunciator panel provides an output failure indication for either transformer rectifier unit. No aircrew control is exercised over transformer rectifier operation aside from controlling the ac power supply or circuit breakers for the power converters. The transformer rectifier units are static converters which individually have a rated output of 100 amperes. Each unit is capable of assuming the complete dc electrical load of the aircraft. Forced air cooling is provided with engines running to dissipate the heat generated by the power converters. No transformer rectifier operating time restriction is imposed under ground external power/engines off conditions without a forced air conditioning source as the compartment environment and volume adequately prevents overheating the units.

Emergency Generator

(U) A hydraulic-driven emergency generator provides a limited but independent back-up source of ac (5 kVA, 115/200 volts) and dc (50 amperes, 28 volts) power for flight essential components throughout the aircraft flight envelope. Fluid from the combined system is the hydraulic media used for driving the generator motor. Approximately 1300 psi hydraulic pressure is required to drive the motor-generator unit to deliver its maximum rated output. With combined system pressure below ______ psi, the emergency generator output is automatically reduced to 1 kVA ac by an under frequency or voltage condition which occurs when a full load from the essential buses is required. There is no automatic cut-off because of essential has No. 1 power supply deviations. Operation of the hydraulic motor which drives the generator is controlled by a solenoid operated hydraulic shut-off valve. The valve automatically returns to the open position with loss of dc electrical power which holds the solenoid operated valve closed. This action occurs automatically with the loss of both the main ac and/or dc power sources which requires a double power supply failure before the emergency generator is activated. Approximately one second time elapses from the point of automatic initiation before the generator delivers rated power to the flight essential ac and dc buses. No time limit is imposed on emergency generator operation as the power unit, which is located forward of the wing box on the left side of the fuselage, is oil cooled to prevent overheating. Operation of the emergency generator unit produces an audible whine, which can be heard in the cockpits during engine off conditions (as during engine shutdown) but the sound is not so discernible with engines running.

(U) Pilot control of the emergency generator is through the guarded EMERG switch on the master generator control panel. With the switch in the NORM position, activation of the emergency generator is automatic with the loss of main ac or dc power. The OFF/RESET position of the switch provides the pilot with the capability of isolating emergency electrical power from the aircraft buses (as in the case of an electrical fire) or resetting the generator.



1-F050-004
012-0

Figure 1-32.

Emergency Generator Test

(U) An inflight or on the deck operations check of the emergency generator can be accomplished by selecting the EMERG GEN position on the MASTER TEST panel, and depressing the switch. This provides 28 volts dc to activate the emergency motor-generator and checks the tie contactors by connecting electrical power to the essential ac and dc busses. Illumination of the GO light on the MASTER TEST PANEL indicates a satisfactory check. Conversely, a malfunction in the emergency generator operation is indicated by illumination of the NO GO light.

External Power

(U) Ground operation of all electrically operated components with engines shutdown is provided through the supply of external ac power (115/200 volts, 400 H_Z, 3-phase) to the aircraft. External power is supplied through a standard six-pin receptacle on the underside of the forward fuselage. An external power monitor prevents the application of external power to the aircraft buses if power is not within prescribed limits. The monitor will disconnect external power from the buses if undervoltage, overvoltage, underfrequency, overfrequency or phase reversal occurs. Power can be re-applied to the aircraft by pressing the reset button adjacent to the receptacle provided the voltage, frequency and power phasing is within prescribed limits. Holding the button depressed will not result in forced coupling if external power is not within limits. Application of a poor quality of external power to the aircraft electrical system must be averted to prevent damage to avionics units. No cockpit control is exercised over the application of external power to the aircraft. The generator control unit allows automatic switchover to occur between external power and aircraft generator power during engine start with the generator switches in the NORM position. Although there is no direct indication in the cockpit as to the stature of external power to the aircraft after starting one engine, the hydraulic transfer pump will not operate with external power connected to the aircraft without activating a switch on the ground check panel. The shorter two pins on the external power receptacle are not connected to an aircraft bus but control the application of external power so that the design inherently prevents electrical arcing between the power plug and the four long pins during hook-up and disconnect operations with a hot plug.

POWER DISTRIBUTION

Buses

(U) Leads from the internal and external power supply systems distribute electrical power to the various aricraft systems through a series of buses. The buses are interconnected by tie contactors which automatically control the split of power distribution from the supply sources to the buses under normal and power supply failure conditions. A general schematic of the electrical power supply and bus tie arrangement is presented in figure 1-32.

(U) Under normal operation, the main ac generator power distribution is split between the left and right main ac buses. Failure of either main ac generator trips a tie contactor to connect both buses to the operative generator. No separate cockpit indication of bus tie status is provided since each GEN caution light provides a direct indication of main generator bus tie contactor position. The left and right main ac buses in turn supply ac power directly to the respective transformer rectifiers, and the left main ac bus also supplies power to both essential ac buses under normal operation. Like the main ac generators, dc power distribution from the two transformer rectifiers under normal operations is split between the left and right main dc buses. Failure of either transformer rectifier (as indicated by an illuminated TRANS/RECT advisory light) trips the respective tie contactor to connect both main dc buses to the operative transformer rectifier. The TRANS/RECT light provides a direct indication of dc bus tie status. An interruption-free dc bus interconnects (through one-way diodes) the left and right main dc buses to provide a continuous source of dc power with failure of either main ac generator and/or transformer rectifier. The left main dc bus additionally supplies power to both essential dc buses under normal operations. Power to the AFCS bus is normally supplied from the interruption-free dc bus; however, with an output failure from both transformer rectifiers, the AFCS bus load is automatically transferred to the essential dc No. 2 bus. Loss of main dc power automatically activates the emergency generator which in turn trips power transfer relays to change essential ac and dc bus loading from the left main ac and dc buses to the emergency generator, regardless of main generator output status. Dual (No. 1 and No. 2) essential ac and dc buses are provided to accommodate the low (No. 1 buses only) and high (No. 1 and No. 2 buses) power supply ratings of the emergency. External power is distributed through the aircraft electrical system in the same manner as main generator power.

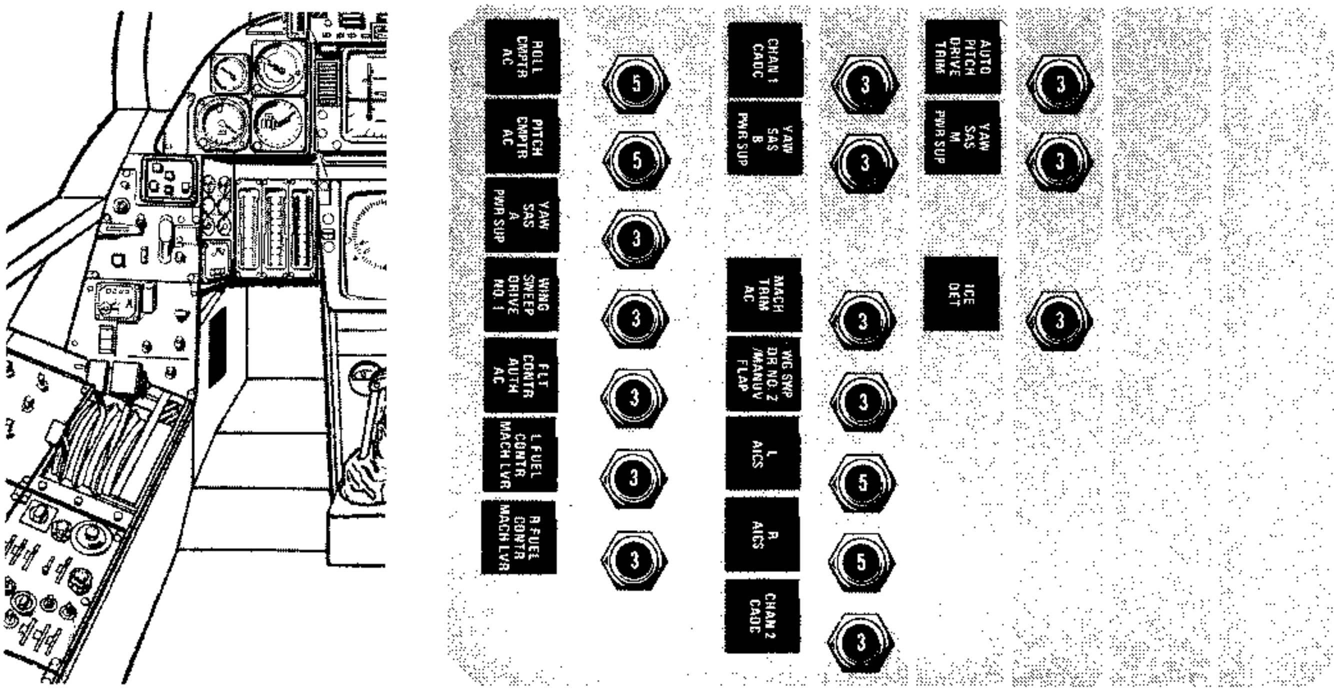
Circuit Breakers

(U) Individual circuit protection from an overload condition is provided by thermal type, trip-free circuit breakers which are all located in the cockpits for accessibility in flight. The appropriate circuit breaker will pop out and isolate a circuit that draws too much current, thus preventing equipment damage and a possible fire. Do not repeatedly reset or forcibly hold circuit breakers depressed as such a condition is indicative of an overload condition or malfunction which may produce far more serious consequences than the mere loss of certain components. Figure 1-33 (Sheets 1 through 5) illustrates the circuit breaker panels in the cockpits. Circuit breakers in the pilots cockpit comprise the majority of those required for essential aircraft systems. The circuit breakers are arranged in rows and oriented so that the white banded shaft of popped breakers is readily visible for aircrew surveillance. Panels, rows and columns of breakers are identified to facilitate breaker location and designation. Placards adjacent to the breakers serve to identify individual circuit breakers by amperage rating and affected components.

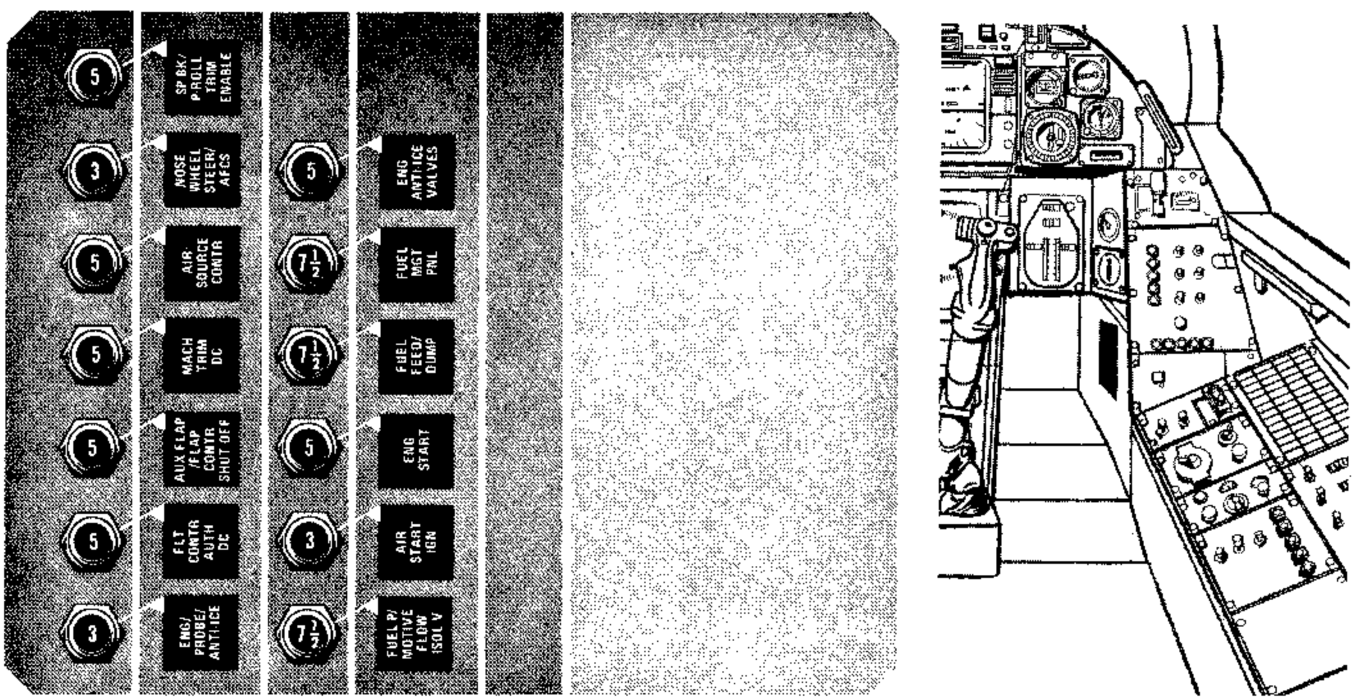
Wiring

(U) Wire runs throughout the airplane are joined, wherever practical, into braided bundles to permit the use of small diameter wire without compromising strength for handling. Wiring connectors do not require soldering or potting for making pin contact or sealing.

PILOT'S CIRCUIT BREAKER PANELS



PILOT'S LEFT KNEE CIRCUIT BREAKER PANEL

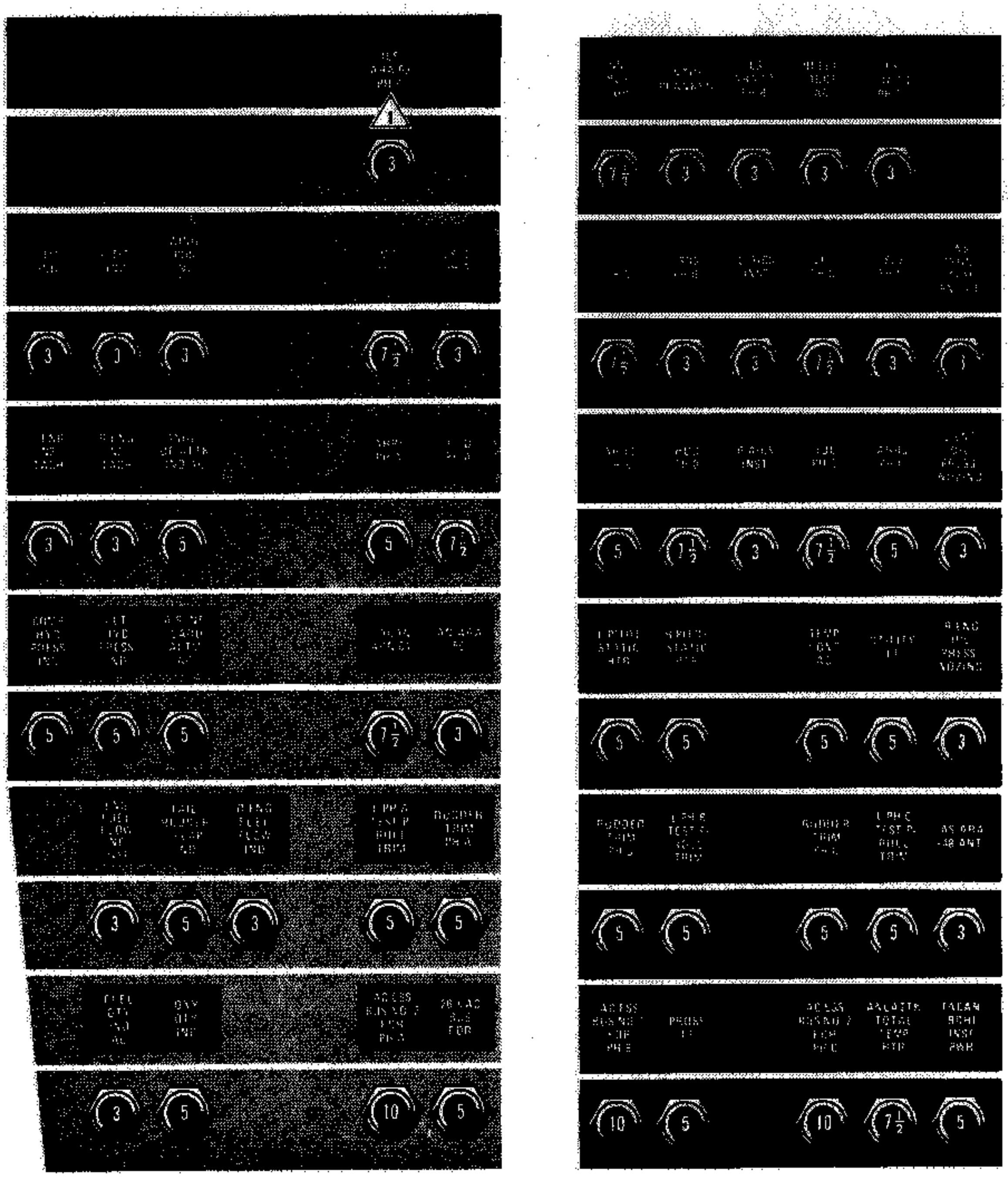


PILOT'S RIGHT KNEE CIRCUIT BREAKER PANEL

0-F050-004
071-0

Figure 1-33. (Sheet 1 of 5)

NFO LEFT SIDE CIRCUIT BREAKER PANELS



AC ESSENTIAL NO. 2 PHASE A
CIRCUIT BREAKER PANEL

AC ESSENTIAL NO. 2 PHASE C
CIRCUIT BREAKER PANEL

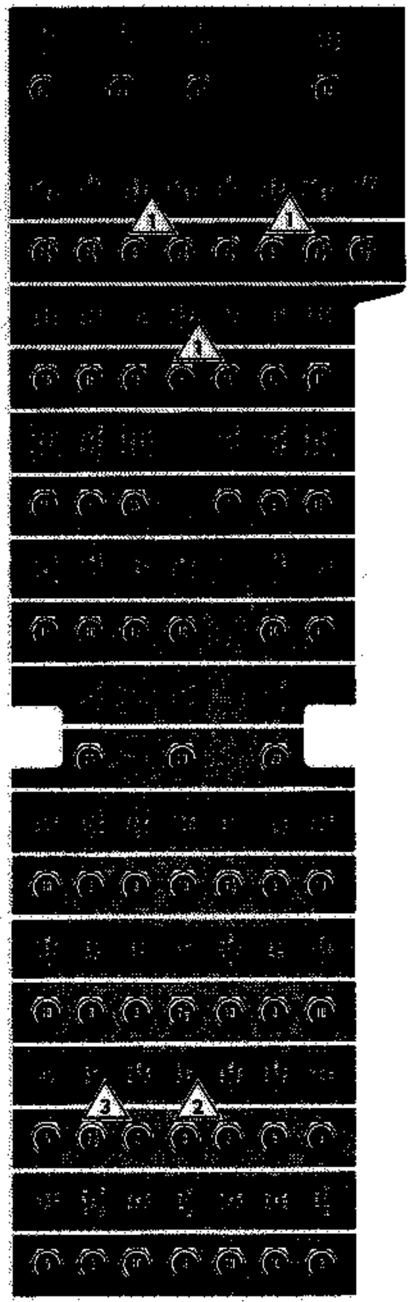
EFFECTIVITY

△1 Aircraft serial no. 157988, 157990, 158612 and subsequent

0-F050-004
095-1

Figure 1-33. (Sheet 2 of 5)

NFO LEFT AFT CIRCUIT BREAKER PANELS



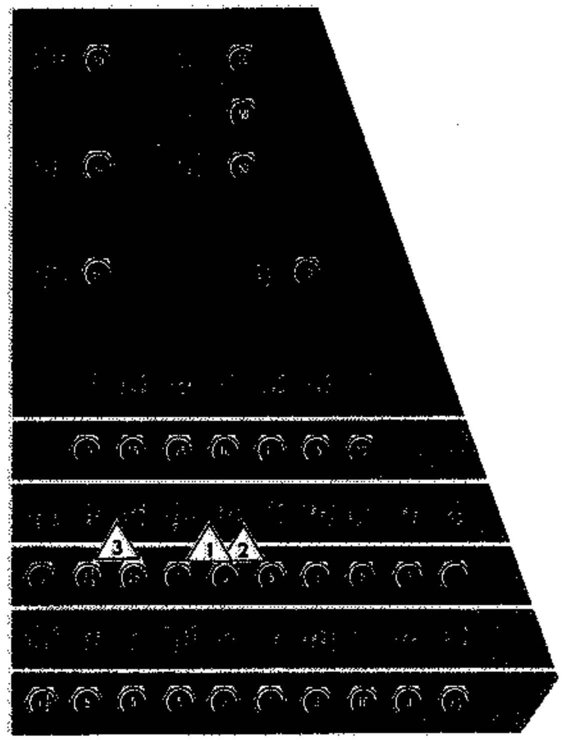
AC LEFT MAIN CIRCUIT BREAKER PANEL

EFFECTIVITY

3 Aircraft serial no. 157983 through 157985, 157988, and 157990

2 Aircraft serial no. 157984, 157985, 157990, 158612 and subsequent

1 Aircraft serial no. 157984, 157985, 157988, 157990, 158612 and subsequent



AC RIGHT MAIN CIRCUIT BREAKER PANEL

EFFECTIVITY

3 Aircraft serial no. 157983 and 157988

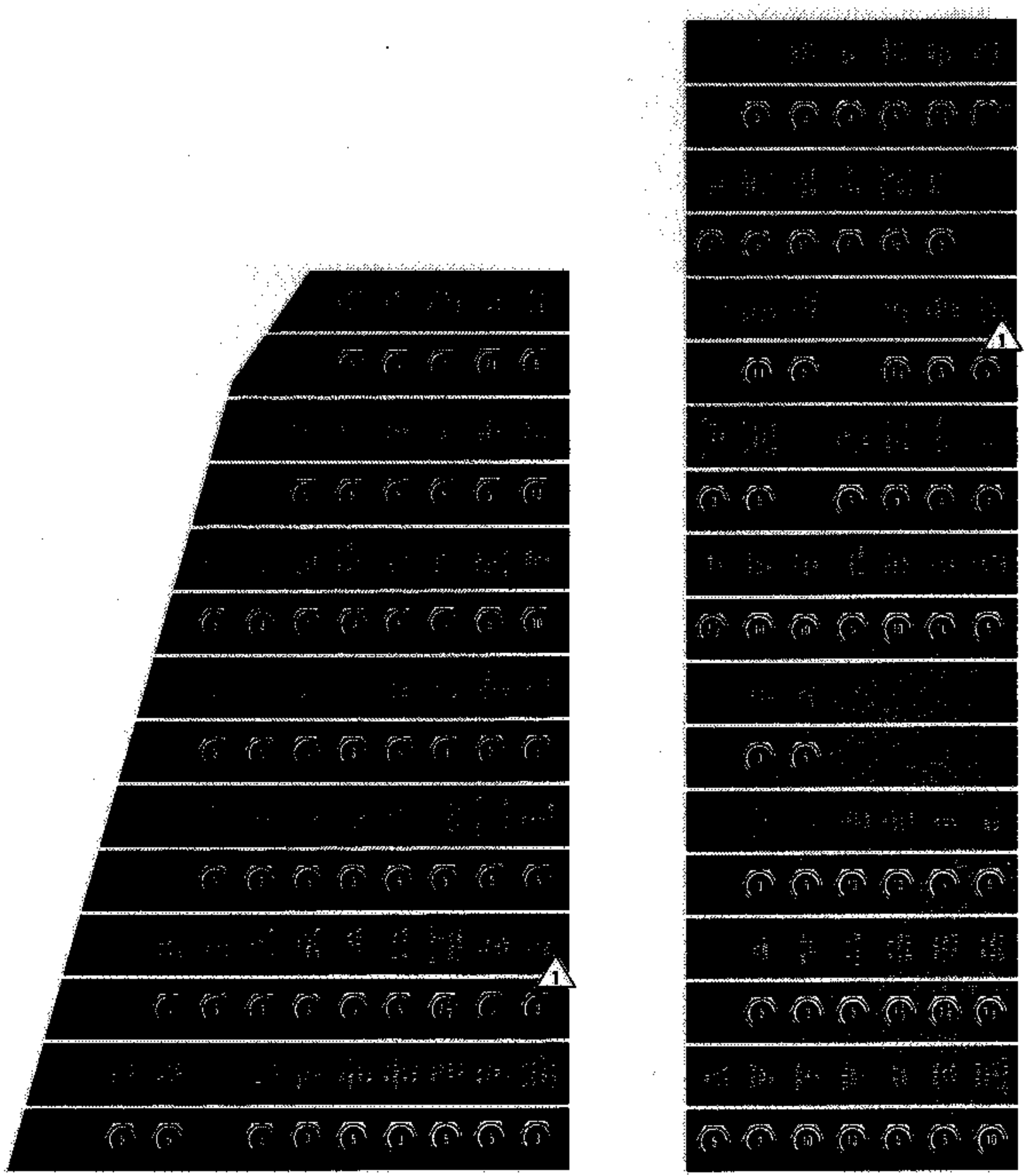
2 25 ampere circuit breaker, titled AWG-9 INST PH C, on aircraft serial no. 157983 and 157988

1 Aircraft serial no. 157990, 158612, and subsequent

0-F050-004
096-2

Figure 1-33. (Sheet 3 of 5)

NFO RIGHT AFT CIRCUIT BREAKER PANELS



DC ESSENTIAL NO. 2 CIRCUIT BREAKER PANEL

DC MAIN CIRCUIT BREAKER PANEL

EFFECTIVITY

1 Circuit breaker title FUEL PRESS ADVSY on aircraft serial no. 158612 and subsequent

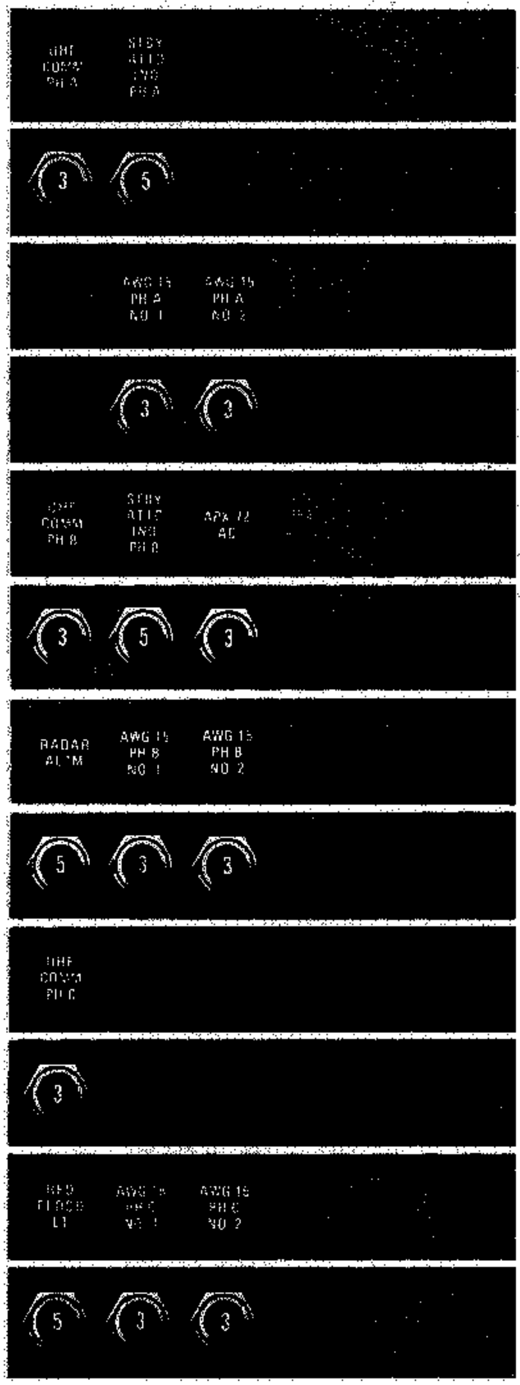
EFFECTIVITY

1 Aircraft serial no. 157983 through 157985, 157988, and 157990

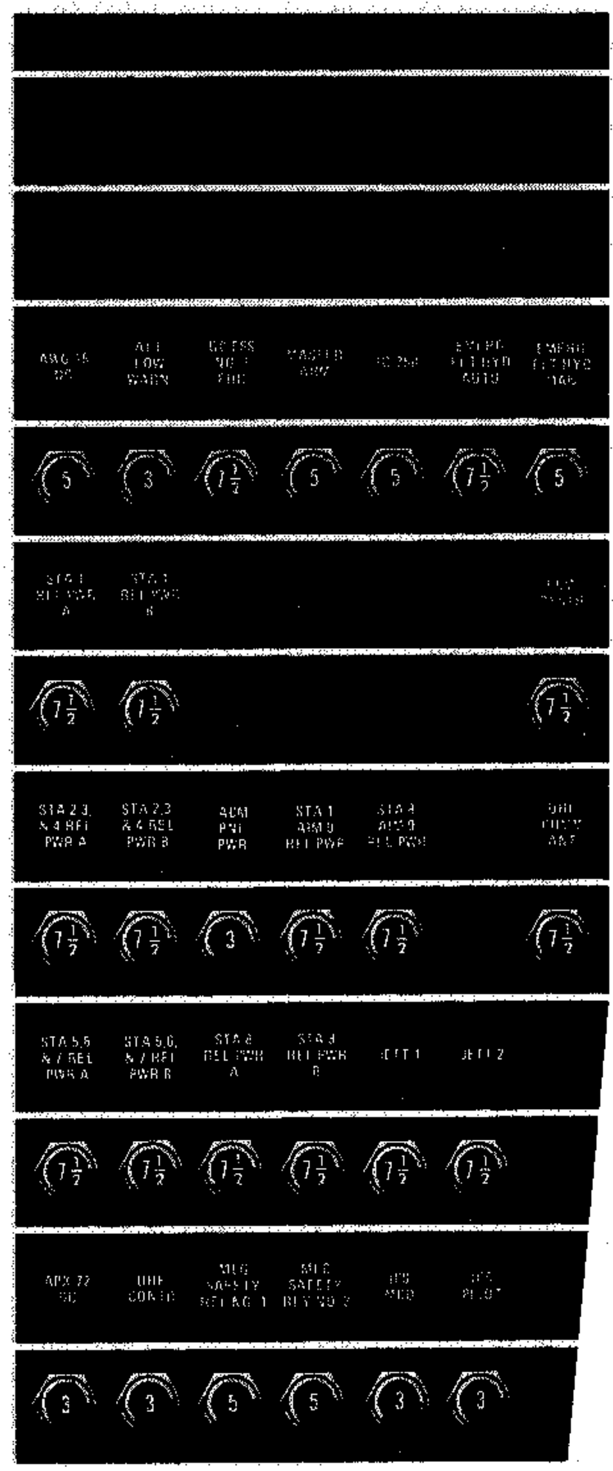
0-F050-004
095-3

Figure 1-33. (Sheet 4 of 5)

NFO RIGHT SIDE CIRCUIT BREAKER PANELS



AC ESSENTIAL NO. 1 CIRCUIT BREAKER PANEL



DC ESSENTIAL NO. 1 CIRCUIT BREAKER PANEL

0-F050-004
095-4

Figure 1-33. (Sheet 5 of 5)

SYSTEMS POWER/GROUND TEST PANEL

(U) A ground check panel (figure 1-34) is located on the NFO left knee panel (accessible from the boarding ladder with the canopy open) for controlling the activation of electrical circuits using ground external power. The panel cover is designed so that when it is closed the switches inside are in the proper position for flight. With the cover closed none of the switches are active when operating on internal power. The panel serves a maintenance and pre-flight purpose and is not intended for use by the flight crew.

HYDRAULIC POWER SUPPLY SYSTEMS

(U) Hydraulic power is supplied to components of the aircraft that require application of heavy forces and/or high response. The aircraft employs two main independent engine powered hydraulic systems which are supplemented by two electro-hydraulic power modules, a bi-directional transfer unit, and a cockpit hand pump. The systems which are pressurized to 3,000 psi use MIL-H-5606 hydraulic fluid circulated through titanium lines with brazed unions. The division of component servicing and hydraulic line routing reduces the possibility of combat damage causing loss of multiple fluid power systems. The versatility of the electro-hydraulic power modules enables on-deck, engines-off operation of certain

SYSTEMS POWER/GROUND TEST PANEL

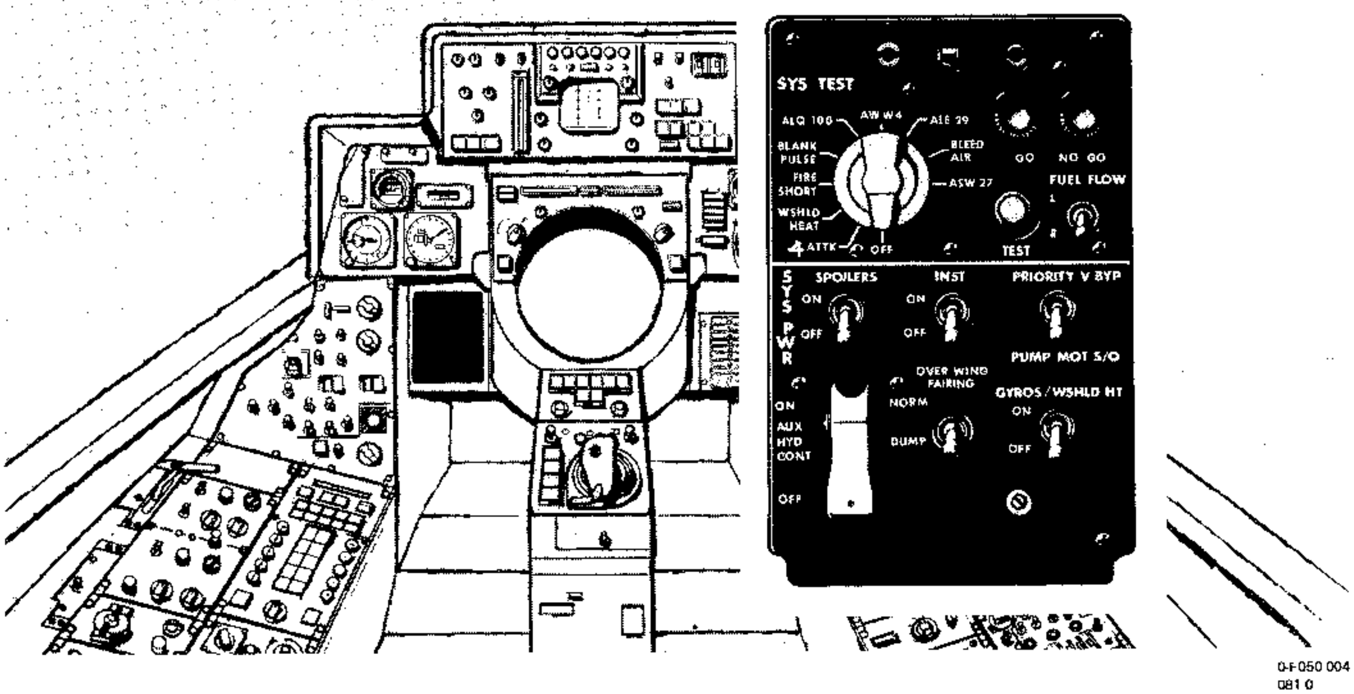


Figure 1-34.

hydraulically powered components at a reduced rate using external ac electrical power in lieu of an external hydraulic power cart. Hydraulic power systems controls and indications are presented in figure 1-35. Figure 1-36 delineates the components serviced by each fluid power system.

FLIGHT AND COMBINED SYSTEMS

Engine-Driven Pumps

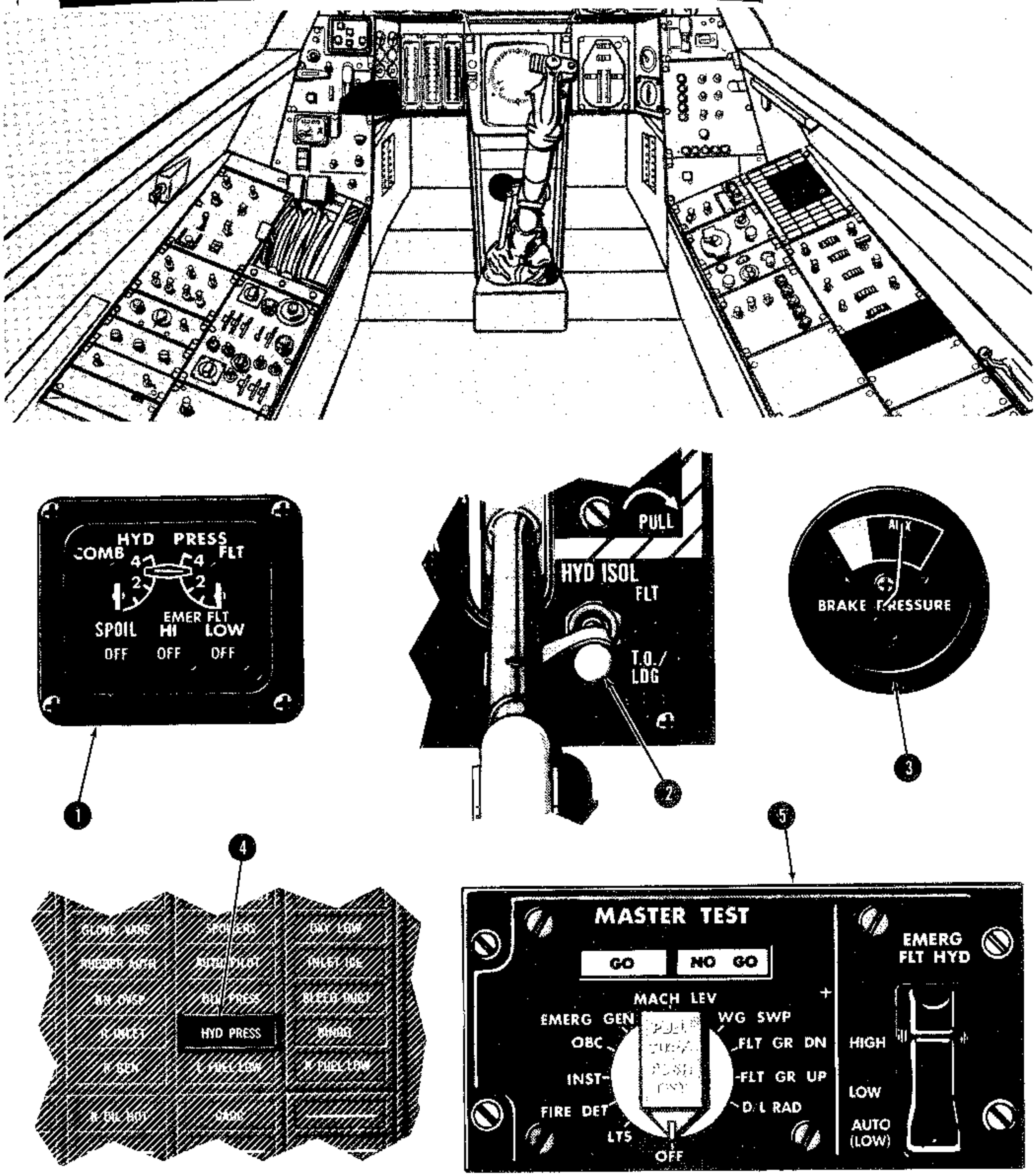
(U) The flight and combined systems, which are the prime independent sources of hydraulic power, are each pressurized by a single pump on the engine accessory gearbox. The flight system pump is driven by the right engine and the combined system pump by the left engine. The single stage, variable delivery, reciprocating piston-type pumps, each have a rated capacity of 84 gpm at 2,900 psi discharge pressure. Each of the main systems is normally pressurized to 3,000 ± 100 psi any time the respective engine is operating. Pressure relief valves protect the systems from pressure surges and dump pressure in excess of 3,450 psi to return lines. The complete separation of power sources decreases the number of common vulnerability points in the hydraulic power systems. A HYD PRESS caution light illuminates when the discharge pressure from either engine-driven hydraulic pump falls below 2,100 psi; thereafter, the light will extinguish with the pressure in both systems exceeding 2,400 psi. If the HYD PRESS caution light has been illuminated by a low pressure condition in one main system, subsequent pressure failure in the other system will not cause the MASTER CAUTION light to flash. The HYD PRESS caution light serves to direct the pilot's attention to the hydraulic pressure indicator. The COMB and FLT gages on the hydraulic pressure indicator reflect pump discharge pressure (measured downstream of the check valve) on the left and right engines, respectively, or pressure derived through the bi-directional transfer unit. With both systems normally pressurized to 3,000 psi, the gage needles form a horizontal straight line.

Bi-Directional Transfer Unit

(U) In order to assure the continuance of main systems hydraulic pressure with an engine or engine-driven pump inoperative, a second source of pressure is provided by a bi-directional transfer unit. This unit consists of two motor/pump units, one in each of the main hydraulic systems, which are interconnected by a common mechanical shaft. Thus, a pressure deficiency in one system is automatically augmented using pressure in the other system as the motive power. The unit results in bi-directional transfer of energy without an interchange of system fluid. The efficiency of the unit is such that a 3,000 psi system on one side will pressurize the other system to approximately 2,400 psi at a rated capacity of 20 gallons per minute without any contribution from the engine driven pump on the deficient side.

(U) The absence of pressure (less than 500 + 50 − 0 psi) on either side of the unit is sensed by one of the transfer unit pressure switches and, after an elapsed time of 10 seconds with insufficient pressure, an electrical signal causes the solenoid controlled hydraulic valve on the opposite side of the motor-pump unit to close. Such action is provided to prevent damage to the bi-directional transfer unit with the loss of system fluid on one side. With ground electrical power connected to the aircraft, the transfer pump is deactivated and can

HYDRAULIC SYSTEM CONTROLS AND INDICATORS



1-F050-004
047-0

Figure 1-35.

NOMENCLATURE		FUNCTION
(1) HYD PRESS INDICATOR	COMB/FLT	- Indicate pump discharge pressure on each engine. Normal pressure is 3,000 psi.
	SPOIL	- When the outboard spoiler hydraulic module is pressurized (1950 to 2050 psi) the ON flag appears. Pressure below 1900 to 1800 psi the OFF flag appears.
	EMERG FLT HI LOW	- When the back-up flight control hydraulic module is pressurized (500 ± 50 psi) the ON flag appears. Pressure below 350 ± 50 psi the OFF flag appears.
(2) HYD ISOL SWITCH	FLT	- Combined system hydraulic pressure is shut-off to landing gear, auxiliary flaps, nose wheel steering, wheel brakes, and tail hook retraction.
	T.O./LDG	- Hydraulic pressure is available to all combined system components barring automatic isolation.
(3) BRAKE PRESSURE GAGE	AUX	- Green segment indicates hydraulic pressure (1,500 to 3,000 psi) in the brake accumulator; auxiliary braking may be applied by rudder-toe pedals.
	EMER	- Red segment indicates 1,000 to 1,500 psi. The parking/emergency brake handle must be pulled to apply auxiliary braking.
(4) HYD PRESS LIGHT		Illuminates when hydraulic pressure from either engine-driven pump is below 2,100 psi. It will go out with pressure in both systems above 2,400 psi.

NOMENCLATURE	FUNCTION	
(5) EMERG FLT HYD SWITCH	HI	- Guard must be lifted. Activates the power module (high speed mode) bypassing flight and combined interlock switches.
	AUTO (LOW)	- Safety guard down. The back-up flight control system is activated when pressure in both the flight or combined system is less than 2,600 psi.
	LOW	- Guard must be lifted. Activates the backup power module (low speed mode) bypassing flight and combined interlocks.

only be energized by a switch on the ground check panel. When starting the right engine first, the combined hydraulic pressure will remain at zero psi until ground electrical power is disconnected. At that moment (approximately 3 seconds time delay) the transfer pump will come on the line and pressure the combined system to nominally 2,400 psi. The converse is true if the left engine is started first. Normally with both engines running or electrical power deactivated the transfer pump is conditioned for the flight hydraulic system to supplement combined hydraulic pressure since the combined system services are more extensive and thus subject to higher pressure demands. However, with less than 2,100 psi flight hydraulic pump discharge pressure and the combined system pressurized, the direction of energy transfer shifts so that the combined system augments flight system pressure. Therefore, the pilot has no direct control over the direction of transfer pump energy transfer as the system automatically shifts in the direction that supplemental power is needed. Because of the location of the flight and combined systems pressure switches, the pressurization contribution of the bi-directional transfer unit is reflected on the hydraulic pressure indicator, but not by the illumination status of the HYD PRESS caution light.

Cockpit Hand Pump

(U) A manually operated pump is provided in the nose wheel well as a supplementary source of power for ground operations with engines shut down and as a back-up for the loss of combined system fluid and/or pressure. Linked thereto, is an extendable handle in the pilot's cockpit between the left console and ejection seat. Forward and aft stroking of the hand pump actuates the double-acting wobble pump. The pump, which draws fluid from the combined system return line, recharges wheel brake accumulator pressure when the landing gear handle is down. With the landing gear handle in the up position, it serves as a back-up means of extending the inflight refuel probe; a reservoir for the hand pump, which is continually serviced by the combined system, traps sufficient fluid to provide for one extension of the probe. Also, the hand pump is the sole means of pressurizating the radome fold actuator which is an operation that must be manually selected and radome unlocked on deck from the nose wheel well. The charging or component operation rate using the hand pump power source, is a function of the number of components selected.

Hydraulic Power Distribution

(U) The distribution of hydraulic power in the flight and combined systems is shown in figure 1-36. Except for the left empennage control surfaces, the flight system services only those components on the right side of the aircraft and does not penetrate into the wings. The combined system distribution is more extensive throughout the aircraft, yet its services are predominently concentrated to the left side and extend to the inboard sections of the movable wing panels and to the landing gear. Although the flight and combined systems are completely independent of each other, in certain components both pressure sources are used without an interchange of fluid. Both systems operate in parallel to supply power for operation of the primary flight control surfaces, (except spoilers), and stability augmentation actuators. If one system fails, the other can continue to supply pressure for operation (with reduced power capability of such components.) If either or both main hydraulic systems should fail, back-up power sources provide the capability for safe return flight and landing. Components serviced by only one of the main hydraulic systems will fail to operate upon system pressure failure unless serviced by a back-up power source. For components to which no back-up power source is provided, alternate means for accomplishing control is provided or the inoperative condition is tolerated for degraded capabilities and/or characteristics.

(U) Figure 1-37, illustrates major components in the main hydraulic power supply systems. Each system has a piston-type reservoir, located in the nacelle sponsor aft on the main landing gear strut on the respective side (combined-left; flight-right), for hydraulic fluid storage that also acts as a surge damper for return line pressures. The reservoirs are pressurized by a bootstrap principal whereby pump discharge pressure ported to the inside of the piston shaft, preloads the piston to provide sufficient pressure head at the pump inlet for all operating conditions. No reservoir precharge servicing is required.

(U) Because of the reservoir vertical location relative to the engine driven pumps, gravity provides a sufficient head of pressure at the pump inlet during the initial engine crank cycle. Reservoir/system fluid capacities for the combined and flight systems are 2.7/21.6 gallons and 1.6/11.8 gallons, respectively. The service level of each reservoir is displayed by a mechanical sensor which is visible from the ground through a sight gage on the forward end of the reservoir through a quick access door in the sponson. The fluid level sensor consists of a preloaded metal tape internal to the reservoir that registers piston position in inches. Actuation of a bleeder/relief valve on a line from the reservoir head dissipates air trapped therein so that the sight gage will not provide an erroneous indication of reservoir fluid quantity. A pressure servicing fitting for replenishment of system fluid is provided in the respective sponson area. Only the flight system can be pressurized directly from an external hydraulic cart with the fittings provided in the right nacelle sponson forward of the reservoir. Flight system pressure contributed by an external fluid power source is reflected on the cockpit hydraulic pressure indicator. The combined system may be indirectly pressurized on deck without the left engine running through the bi-directional transfer unit with the flight system pressurized or alternately by use of the back-up flight control module.

(U) The supply of fluid from each reservoir passes through a heat exchanger enroute to the engine driven pump. Cooling of the hydraulic fluid is accomplished by an air-hydraulic heat exchanger located at the forward end of the engine compartment. The air intake to the hydraulic cooler is located on the left side of the nacelle and the air is discharged into the engine compartment where it ventilates the nacelle before being exhausted overboard at the aft end of the engine compartment. During on-deck operations, bleed air ejector nozzles induce circulating airflow through the heat exchangers. Hydraulic fluid temperature is limited to 275°F and temperature gages on the filter modules in the sponson areas are provided for maintenance monitoring. The autogenous ignition point of MIL-H-5606P hydraulic fluid is 475°F. A flow control valve at the heat exchanger bypasses fluid flow exceeding the heat exchanger capacity (10 gallons per minute) and by-passes all fluid when fluid temperatures are less than 160°F.

(U) Fluid discharged from either engine-driven pump passes through filter before entering the pressure supply lines. The pressure supply lines incorporate priority valves and check valves to control fluid power distribution under system partial pressure conditions. Fluid flows from the pressure lines into the various actuating cylinders and motors, and the

COMBINED AND FLIGHT HYDRAULIC SYSTEMS

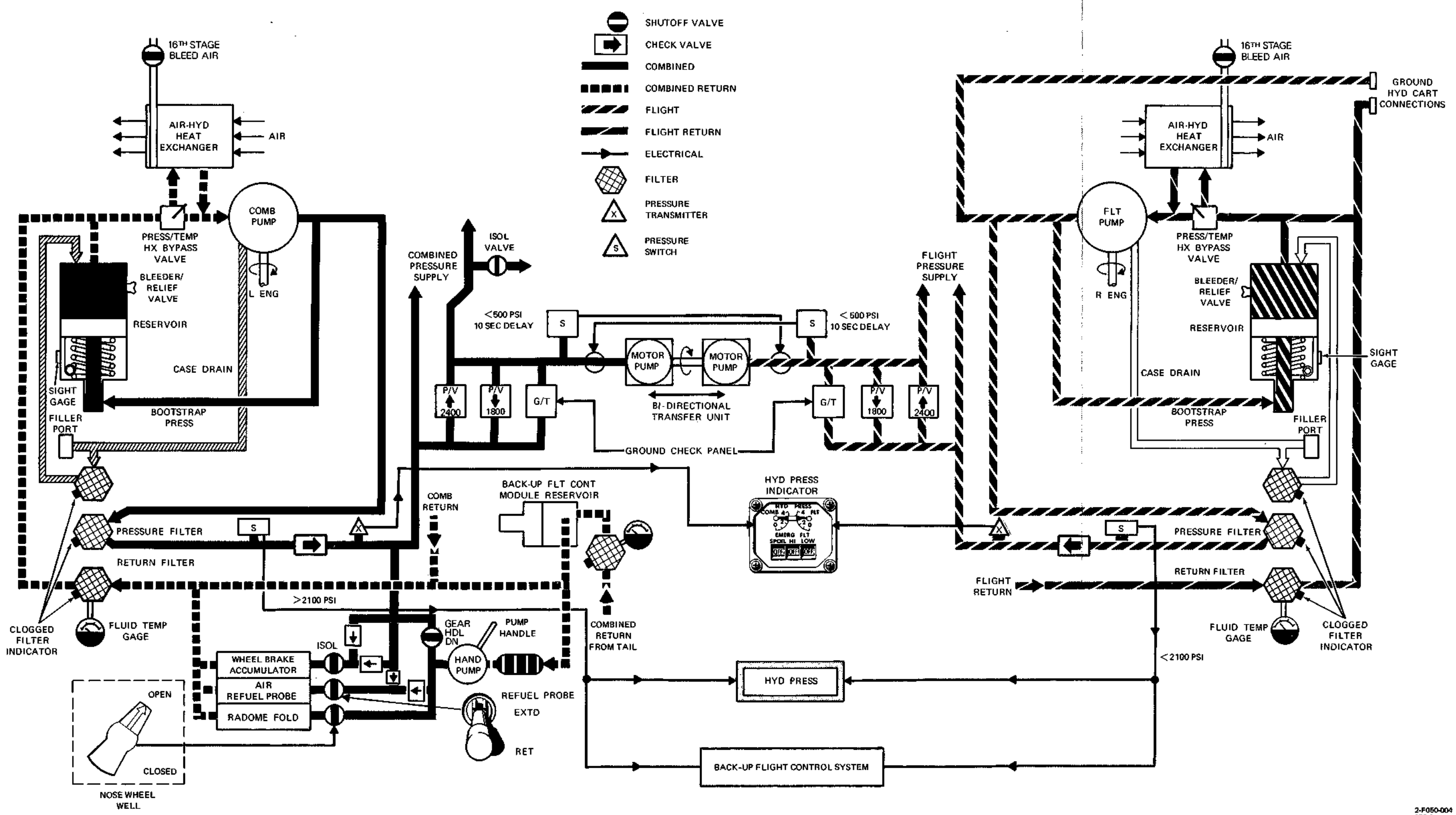


Figure 1-37.

expended output therefrom ports into the applicable system return lines where it is directed back to the reservoirs via the filter modules. Each filter module contains three filters: pressure, return and case drain. The filter modules are located aft of the reservoirs in the nacelle sponson. Protrusion of mechanical pins on each filter module indicates a clogged flow condition in any one of the three filters in each module.

Hydraulic Priority Valves

The combined and flight hydraulic systems each incorporate two priority valves (1800 psi and 2400 psi, figure 1-37) in the pressure modules which perform a subtile but significant function. Hydraulic fluid will not pass through the one-way priority valves unless the input pressure exceeds the cracking threshold of the valve. Basically on each side the 2400 psi priority valves serve to give priority of the individual engine driven pump discharge pressure to the primary flight controls (horizontal tails, rudders, inboard spoilers) and stability augmentation actuators. Conversely, the 1800 psi priority valves on each side give priority to the remaining systems on the other side of the pressure modules (inlet ramps, wing sweep, glove vanes, etc.) with energy derived through the transfer pump unit. Under such circumstances the pilot should be aware of the motive energy available and demands of the various system components. Large and abrupt control commands can rapidly consume total system power with the engine(s) at IDLE speed; the priority valves will govern the ultimate sequence of power distribution. In certain cases the nose wheel steering and wheel brake (normal and anti-skid modes) motive power can be temporarily lost.

Normal Isolation

(U) Because of its extensive services, the combined system incorporates isolation circuits to limit distribution to flight essential components. Normal isolation of the combined system is selectable by the pilot after landing gear retraction has been completed. Normal isolation electrically shuts-off hydraulic pressure to components (landing gear, nose wheel steering, wheel brakes and tail hook retraction) associated with landing. It is activated by placement of the HYD ISOL switch on the landing gear panel to the FLT position. Placement of the gear handle to the DN position mechanically cams the HYD ISOL switch to the T.O./LDG position or the pilot can manually select it prior to lowering the landing gear. Such action returns all combined system components to normal operation barring the existence of automatic isolation.

Automatic Isolation

(U) The occurrence of a flight system pressure failure (less than 2,100 to 2,400 psi detected at the flight system pump discharge port) electrically shuts off hydraulic pressure to all components on the normal isolation circuit if not previously selected; such action is coincident with the illumination of the HYD PRESS caution light by the flight system pressure sensor. Automatic isolation circuits are de-energized with power removed from the essentail electrical bus, with Flight pump discharge pressure greater than 2100 psi, or with weight-on-wheels. The automatic isolation circuit is deactivated with weight-on-wheels to permit on-deck use of the wheel brakes, nose wheel steering (only if the landing gear was not extended by the emergency means) and tail hook retraction components.

OUTBOARD SPOILER SYSTEM

(U) A separate independent electro-hydraulic power module supplies hydraulic pressure for normal operation of the outboard sets of wing spoilers and serves as a back-up power supply for operation of the main flaps and slats. The outboard spoiler system is a separate closed-loop system which is independent of the main hydraulic systems (see figure 1-38). Inflight the module is electrically interlocked with wing sweep position such that it is activated at wing sweep angles less than 63 degrees (outboard spoiler operation is inhibited aft of 57 degrees wing sweep angle). Ground (weight-on-wheels) operation of the outboard spoiler module on aircraft power requires that an additional interlock (flap handle not in the UP position) be satisfied. The purpose of this additional interlock is to reduce the on deck operating time of the module since no heat exchanger is contained within this independent system. Inflight cooling of the system fluid does not present a thermal problem because of heat rejection through the pressure and return lines. Fluid from the reservoir passes through the motor case enroute to the pump to assist in dissipating system heat. Electrical power for the outboard spoiler system motor (single speed) is tapped from the right main ac bus and the module can be activated using external ac electrical power. With the module pressurized (greater than 2,000 psi discharge pressure) the ON flag appears in the SPOIL window at the bottom of the hydraulic pressure indicator; otherwise, an OFF indication is displayed in the window. A thermal (275°F fluid temperature) cutout circuit prevents overheating of the module system. Since the efficiency of the wing flaps is dependent on the spoilers being in the drooped configuration, the thermal cutout circuit is inhibited with weight-off-wheels and the gear handle down. The thermal cutout circuit automatically resets the outboard spoiler system to normal operation when the fluid temperature falls below the prescribed level (245°F).

(U) Rated capacity of the system is 6.5 gallons per minute at 1,800 psi decreasing to no flow at 3,100 psi. The module, which is an integrated package containing a motor, pump, reservoir and filter unit, is located in the left nacelle sponson forward of the main wheel well. A spring in the reservoir provides a starting head of fluid to the pump inlet under all attitude conditions whereafter pump discharge pressure directed through the piston shaft maintains the operating head pressure by the bootstrap principal. Reservoir servicing level is accurately reflected by an indicator rod protruding from the integral power package after trapped air has been bled from the reservoir. Servicing is accomplished through a pressure fitting in the power module compartment. A fluid temperature gage, with actual and retained maximum needles for registering current and peak system temperatures, is provided on the power module. Protrusion of a red-tipped pin on the integrated package is a positive indication of a clogged filter condition.

(U) Rated capacity of the system is 6.5 gallons per minute at 1,800 psi decreasing to no flow at 3,100 psi. The module, which is an integrated package containing a motor, pump, reservoir and filter unit, is located in the left nacelle sponson forward of the main wheel well. A spring in the reservoir provides a starting head of fluid to the pump inlet under all attitude conditions whereafter pump discharge pressure directed through the piston shaft maintains the operating head pressure by the bootstrap principal. Reservoir servicing level is accurately reflected by an indicator rod protruding from the integral power package after

OUTBOARD SPOILER SYSTEM

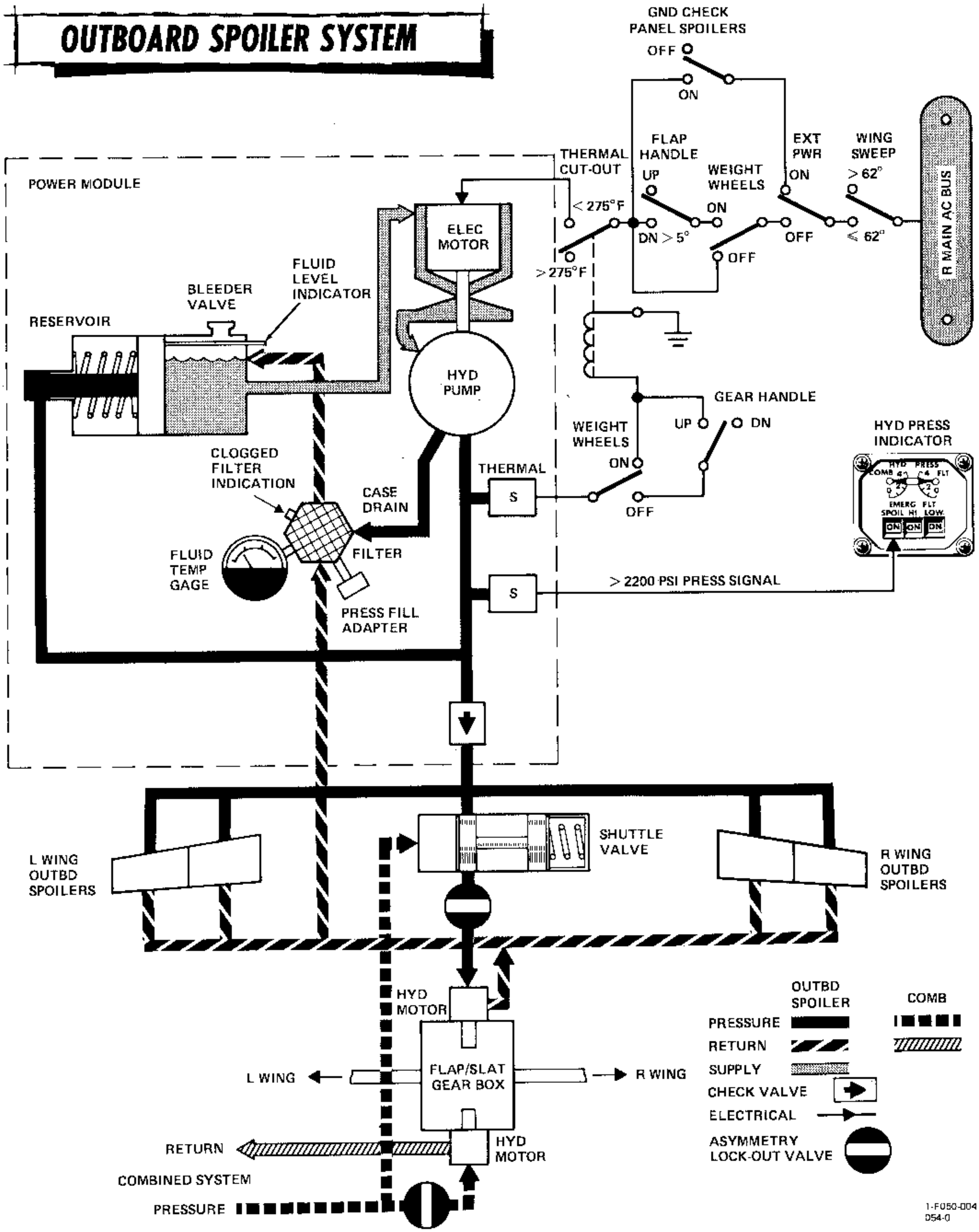


Figure 1-38.

trapped air has been bled from the reservoir. Servicing is accomplished through a pressure fitting in the power module compartment. A fluid temperature gage, with actual and retained-maximum needles for registering current and peak system temperatures, is provided on the power module. Protrusion of a red-tipped pin on the integrated package is a positive indication of a clogged filter condition.

(U) Routing of system pressure and return fluid is through flexible lines in the wing pivot area, along the wing forward spar, across to the two outboard sets of wing spoilers. Although normal operation of the main flap and slat segments is powered by a combined system motor on the flap/slat gear box, an auxiliary motor powered by the outboard spoiler system is geared to the same shaft to provide for emergency operation (retraction and extension) of the main flaps and slats at a reduced rate. A spring-loaded valve shuttled by combined system pressure by-passes outboard spoiler system fluid in a closed loop to minimize the drag load as the auxiliary motor acts as a pump under normal operations. failure of combined system pressure automatically causes the valve to shuttle whereby outboard spoiler system pressure powers the auxiliary motor to drive the flap/slat gear box when selected by the normal flap handle or maneuvering thumbwheel.

BACK-UP FLIGHT CONTROL SYSTEM

(U) The back-up flight control system, or emergency flight system, consists of a two-speed electro-hydraulic power module tied into the combined hydraulic system lines as a closed loop system for providing motive fluid energy to operate the empennage control surfaces at a reduced rate (see figure 1-39). Emergency powering of the empennage control surface (horizontal tails and rudders) actuators provides sufficient pitch, roll and yaw control for return flight and landing with both main hydraulic power supply systems inoperative. Control of the power module is automatic or pilot selected when operating in aircraft electrical power as dictated by the EMER FLT switch on the pilot's master test panel. In the AUTO (LOW) position loss (less than 2100 psi) of both the flight and combined hydraulic systems results in automatic activation of the back-up flight control module in the low speed mode. With the module pressurized (greater than 500 psi discharge pressure) the ON flag appears in the EMERG FLT LOW window at the bottom of the hydraulic pressure indicator; an OFF indication will be displayed in the HI window. Positioning the switch to the LOW position accomplishes the same action on ground or aircraft power without the need to satisfy the hydraulic pressure switch interlocks. The HI position of the switch is the only means of activating the module to the high speed mode which should be done immediately incident to transitioning into the landing configuration; in this case the ON flag will appear in the EMER FLT HI window. Right main ac electrical power is the source of energy for powering the module. Loss of both engine-driven electrical generators obviates inflight use of the module, however it can be activated on the deck using external ac electrical power. A thermal (275°F fluid temperature) cutout circuit prevents overheating of the module system when activated from the switch on the ground check panel but not by the pilot's EMER FLT switch. The thermal cutout circuit automatically resets the back-up module to normal operation when the fluid temperature subsides below the established level.

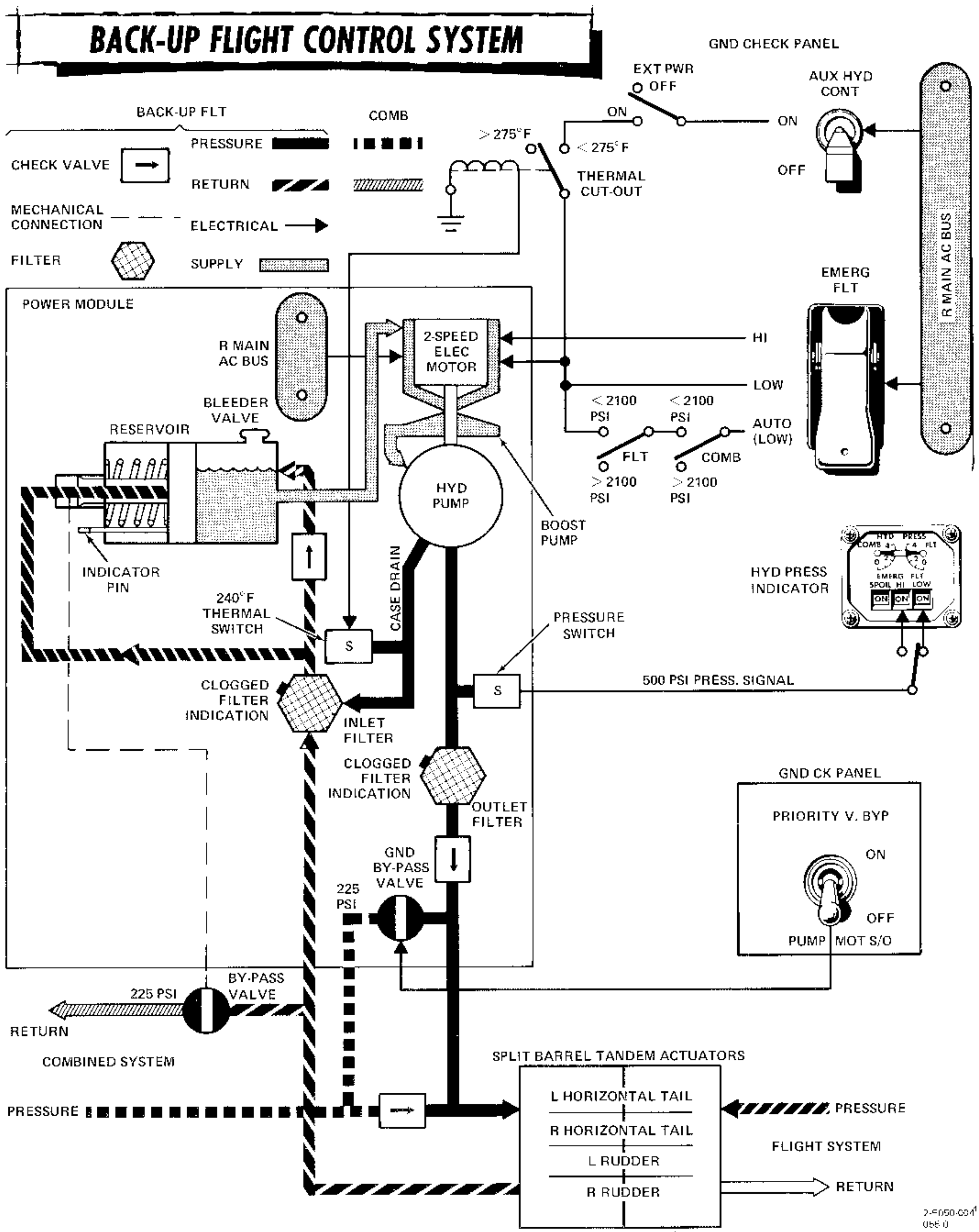

BACK-UP FLIGHT CONTROL SYSTEM
BACK-UP FLT
COMB
CHECK VALVE
PRESSURE
RETURN
MECHANICAL CONNECTION
ELECTRICAL
FILTER
SUPPLY
GND CHECK PANEL
EXT PWR
OFF
ON
AUX HYD CONT
ON
OFF
> 275°F
< 275°F
THERMAL CUT-OUT
POWER MODULE
R MAIN AC BUS
2-SPEED ELEC MOTOR
HI
LOW
EMERG FLT
R MAIN AC BUS
BLEEDER VALVE
RESERVOIR
< 2100 PSI
< 2100 PSI
FLT
COMB
AUTO (LOW)
> 2100 PSI
> 2100 PSI
HYD PUMP
BOOST PUMP
INDICATOR PIN
CASE DRAIN
HYD PRESS INDICATOR
240°F THERMAL SWITCH
PRESSURE SWITCH
S
S
CLOGGED FILTER INDICATION
INLET FILTER
500 PSI PRESS. SIGNAL
CLOGGED FILTER INDICATION
OUTLET FILTER
GND CK PANEL
PRIORITY V. BYP
ON
OFF
PUMP
MOT S/O
GND BY-PASS VALVE
225 PSI
225 PSI
BY-PASS VALVE
RETURN
COMBINED SYSTEM
PRESSURE
SPLIT BARREL TANDEM ACTUATORS
L HORIZONTAL TAIL
R HORIZONTAL TAIL
L RUDDER
R RUDDER
PRESSURE
FLIGHT SYSTEM
RETURN

Figure 1-39.

(U) In the high speed mode the rated capacity of the back-up flight control system varies from 14.5 gpm at 500 psi discharge pressure to 2.4 gallons per minute at 3,000 psi. System flow rates in the low speed mode are approximately one-half those of the high speed mode at corresponding operating pressures. Rapid control inputs exceeding the power module pumping capacity will lag in response against opposing loads. Inflight operation of the power module is time limited to prevent over-heating and subsequent damage to system components. Operating time limits vary as a function of the flight condition.

(U) In the low speed mode, the system can operate indefinitely at altitudes above 27,000 feet and up to 17 minutes at sea level. For maximum range and endurance using the high speed mode, an altitude as close to 44,000 feet and Mach .8 is recommended. For ground inspection purposes protrusion of a red-tipped button on both the inlet or outlet filter cases is a positive indication of a clogged filter. Both such indications may be observed through an access door on the underside of the aft fuselage.

(U) The back-up flight control system reservoir is continually charged by the combined system return flow which obviates the requirement for ground servicing. A spring in the reservoir provides a continuous head of fluid pressure at the module pump inlet under all attitude conditions. When activated, the power module pressurizes the combined system pressure lines to the horizontal tail and rudder actuators. Combined system return lines from these actuators close the loop by porting the return fluid to the power module reservoir via the filter unit. Operation of the back-up flight control system can be checked on the ground or inflight by lifting the guard over the BACK UP HYD switch and selecting the LOW or HIGH position. This action activates the power module to the respective speed mode independent of the flight and combined system pressure interlock switches. On-deck activation of a hydraulic bypass switch on the systems power/ground test panel enables the module to pressurize the entire combined system; if selected, a solenoid operated valve on the combined system side of the bi-directional transfer unit closes to isolate the flight from the combined system pressure to conserve available energy.

PNEUMATIC POWER SUPPLY SYSTEMS

(U) The pneumatic power supply systems consist of three independent, stored pneumatic pressure sources which provide for normal and auxiliary operation of the canopy and emergency extension of the landing gear. The high pressure bottles are ground charged, through a common filler connection in the nose wheel well, to 3,000 psi at 70°F ambient temperature. Individual bottle pressure is registered on separate gages on the right side of the nose wheel well. Charges may be compressed air; however, pressurized dry nitrogen is preferred because of its low moisture content and non-combustible properties.

NORMAL CANOPY CONTROL

(U) The bottle which supplies a pressurized charge for normal operation of the canopy is located on the right side of the forward fuselage inboard of the air refuel probe cavity.

Expenditure of bottle pressure for normal operation of the canopy is controlled by the three (pilot, NFO and ground) canopy control handles. A fully charged bottle provides for approximately 8 complete cycles (open and close) of the canopy before reaching the minimum operating pressure of 300 psi.

AUXILIARY CANOPY OPEN CONTROL

(U) The bottle which supplies a pressurized charge to translate the canopy aft, to a position when the counterpoise action of the canopy actuator will facilitate opening, is located on the turtleback behind the canopy hinge line. Activation of the auxiliary mode can be effected from either of the three (pilot, NFO or ground) canopy control handles. Subsequent to activation of the auxiliary open mode the control system will not return to the normal mode of operation (canopy will lower but will not translate forward) until the auxiliary selector valve on the aft canopy deck is manually reset (lever in vertical position). Servicing of the auxiliary canopy open air bottle is performed in the small access panel immediately behind the canopy on the turtleback. A fully charged bottle provides for greater than 20 operations in the auxiliary open mode.

EMERGENCY GEAR EXTENSION

(U) The bottle which supplies a pneumatic charge for a single emergency extension of the landing gear is located on the right side of the nose wheel well. Expenditure of bottle pressure is controlled by a twist-pull operation of the landing gear handle. Minimum bottle pressure for accomplishing emergency extension of the gear to the down-and-locked condition is 1800 psi.

WING SWEEP SYSTEM

(U) The variable geometry of the wing sweep system provides the pilot with considerable latitude for controlling wing lift and drag characteristics to optimize aircraft performance over a broad flight spectrum. Complementing this control capability, automatic limiting of wing sweep authority is provided under normal inflight control modes to prevent mispositioning of the wings at conditions that could result in the penetration of unsafe structural or stability and control boundaries. Additionally, the automatic wing sweep position limiter provides a pilot relief mode of control for positioning the wings. All computational functions are performed in the dually redundant air data computer (ADC). The wing sweep actuating system embodies the same principles of redundant control as applied to the primary flight control system. Normal modes of control are effected through a "fly-by-wire" system whereas a mechanical back-up control system is provided for emergency and oversweep operations. Details of the wing sweep system are illustrated in figure 1-40.

(U) The wing carry-through box, which lays across the mid fuselage and extends into the wing glove, is the structural backbone used to carry the direct loads of the movable wing panels. In addition, the welded titanium wing box contains integral fuel feed cells and the member also serves to distribute direct loads imposed by the glove pylons, main landing gear drag brace and hoisting sling. Tangs on the extremities of the wing box contain the permanently lubricated spherical bearings in which the outboard wing panels pivot. The outboard location of the wing pivot (9 feet from centerline) serves to reduce the change in longitudinal stability as a function of wing sweep angle. Two independently powered, hydromechanical screw jack actuators, mechanically interconnected for synchronization, position the wings in response to pilot or ADC commands. Inflight the wings can be positioned between 20 degrees and 68 degrees wing leading edge sweep angle and on the deck the range is extended aft to 75 degrees (oversweep position) to reduce the span for spotting. Such authority results in a variation of wing span from approximately 64 feet to 33 feet.

(U) Aerodynamic sealing between the wing leading edge and the glove is achieved by a radius on the movable wing panel which has a constant cross-sectional area. Cavities above the engine nacelles and mid fuselage accommodate the inboard portions of the wing panels as they sweep aft. Sealing of the underside is accomplished by a wiper seal and air bag. An overwing fairing encloses the wing cavity and provides a contoured seal along the upper surface of the wing for the normal range of inflight sweep angles. The forward edge of the overwing fairing on each side is hinged along the aft side of the wing box and is sprung in place by two hydraulic actuators to maintain a tight seal. The left and right overwing fairing actuators are pressurized by the combined and flight hydraulic systems, respectively, and each uses an accumulator as a back-up source of pressure. Hydraulic

WING SWEEP

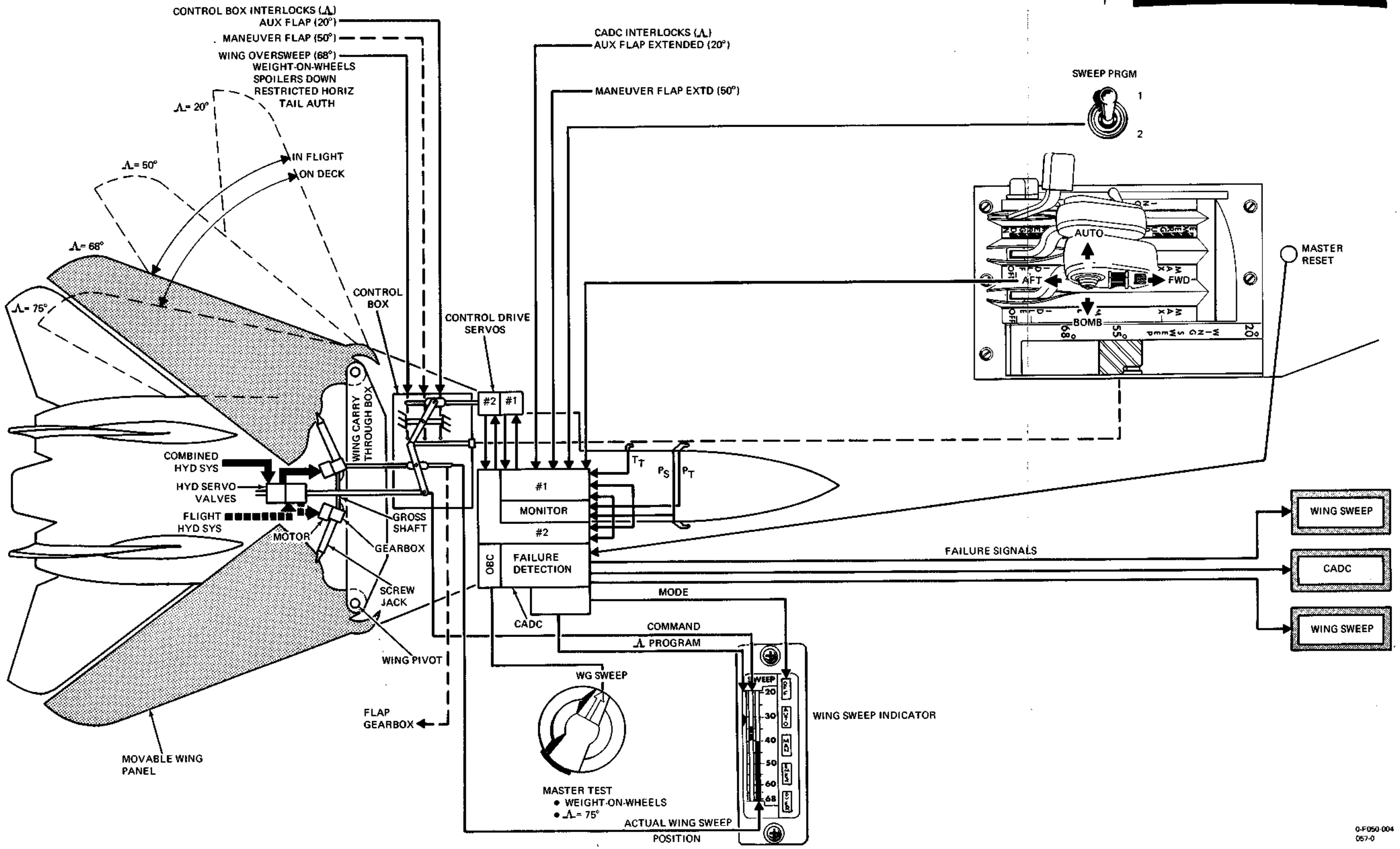


Figure 1-40.

pressure must be relieved in the overwing fairing system before the wing can be swept aft of 68 degrees on the deck. Proper operation of this system must be maintained to prevent a performance degradation and/or structural failure.

WING SWEEP PERFORMANCE

(U) The performance of the wing sweep system is characterized by the wing sweep rate capability under load conditions. Major loads on the wing sweep actuators are imposed by aerocordwise forces, pivot bearing friction and wing panel weight. Wing sweep rate is primarily a function of normal load factor (g's) and the exceedance of the performance envelope will lead to actuator saturation and a system stall. However, sufficient capability has been provided in the system consistent with sustained performance capabilities of the aircraft. Critical performance conditions occur at low linear deceleration/acceleration of the aircraft under high g's (split-S or loop maneuvers) and at high linear deceleration/acceleration under low g's (steep climbs with low thrust or steep dives with maximum thrust). The rate of operation during unsweeping motion exceeds that during sweeping motion at elevated positive load factor condition. Because of the attachment geometry of the wing sweep actuators, the rate of sweep operation progressively increases with the wings aft of 50 degrees. Failure of either the combined or flight hydraulic system permits the wings to move at a reduced rate (nominal 3 degrees per second under 1 g conditions).

WING SWEEP CONTROLS

(U) Normal control of wing sweep position in AUTO, MANUAL and BOMB modes is exercised using the four-way electrical button on the inboard side of the right throttle grip, see figure 1-41. The location permits efficient transition between modes and does not tax the pilot with a physical separation of energy control devices (throttles, speed brakes and wing sweep). As a back-up mode of control, changes in wing sweep position can be manually selected with the emergency handle located on the inboard side of the throttle quadrant. The handle is directly connected to the wing sweep control box by a controllex cable and moves under the transparent guard during normal wing sweep position command changes. The guard must be rotated out of the way before operation of the handle can be effected. Vertical extension of the emergency handle provides for better accessibility and leverage. An initial force of approximately 18 pounds is required to release the emergency control from the spider detent whereafter reduced forces are necessary for operation (unless passing through the detented position again). A light force detent is provided at the 55 degrees commanded wing sweep position for the emergency handle. Further vertical extension of the handle is necessary to command wing oversweep on the deck. Once in the 75 degrees wing sweep commanded position, the handle may be stowed and the guard rotated back in place. A button on the handle must be depressed before the handle may be pushed down to the stowed position in the 20 degrees to 68 degrees command range.

WING SWEEP INDICATOR

(U) The wing sweep indicator (figure 1-41) is located on the upper right side of the pilot's instrument panel at the transition line between the heads-up and heads-down scan pattern. A transducer on the left wing sweep actuator feedback shaft to the control box provides signals of actual wing sweep position for display on the indicator vertical scale tape.

WING SWEEP CONTROLS

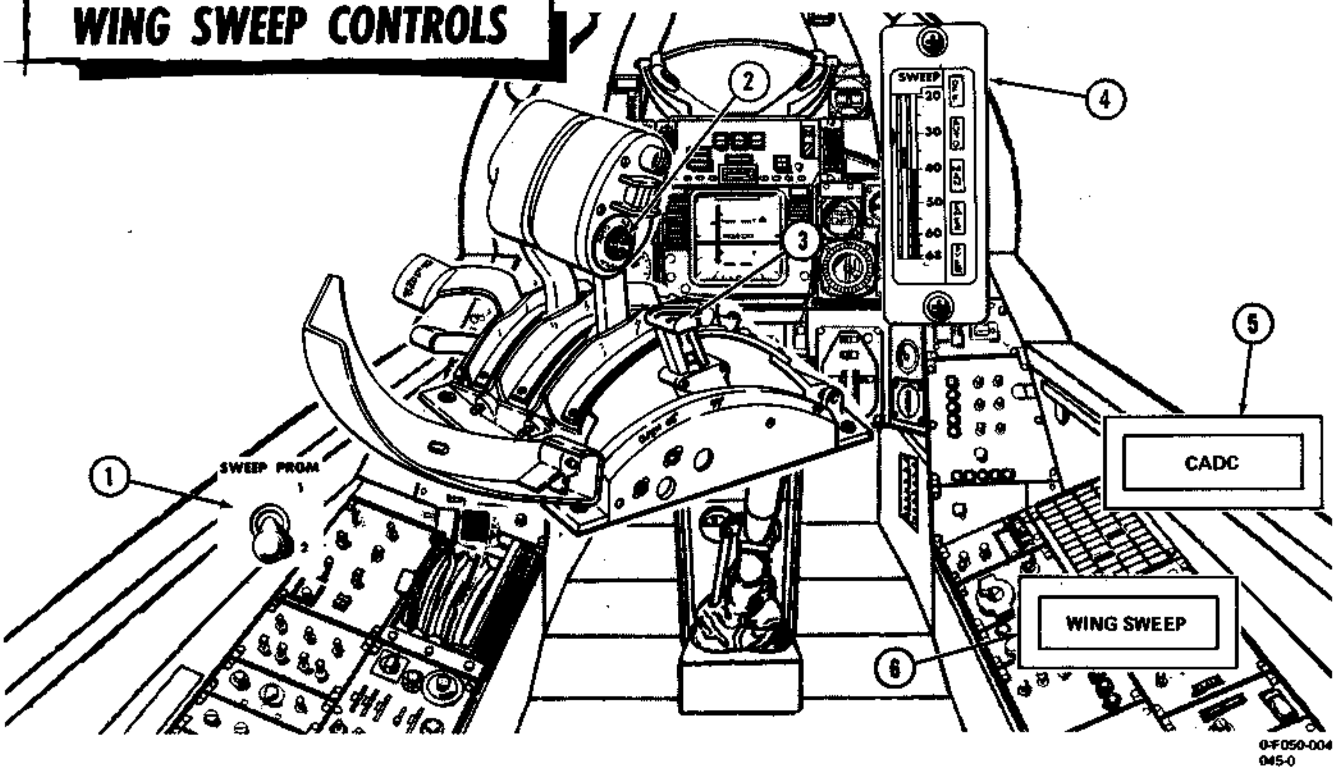


Figure 1-41.

NOMENCLATURE	FUNCTION	
(1) SWEEP PRGM SWITCH	1	- Program 1 schedules wing position automatically when AUTO is selected. Normal program selected for optimum wing sweep angle.
	2	- Is automatically selected if 1 fails. Program 2 positions wings at optimum sweep angle for maximum usable lift for maneuvering.
(2) WING SWEEP CONTROL BUTTON	AUTO	- Wing sweep angles are determined by ADC according to program selected.
	BOMB	- Wings are positioned at 55 degrees or further aft if permitted by the ADC programmer.

NOMENCLATURE	FUNCTION
	AFT/FWD - The pilot can select AFT or FWD wing positions within limits imposed by the automatic programmer. Switch is spring-loaded to the center position. When the forward limit is intercepted the mode is transferred to AUTO.
3 EMERGENCY WING SWEEP HANDLE	Provides a mechanical back-up means of wing sweep control which overrides the ADC programmer commands. Wing sweep angles between 20 and 68 degrees are unrestricted except for flap interlocks. Oversweep to 75 degrees is provided with weight-on-wheels only.
4 WING SWEEP INDICATOR	Displays actual wing sweep position, command position, program position which is the maximum forward angle at present airspeed. Indicator windows show the operating mode.
5 CADC CAUTION LIGHT	Computations in the air data computer are unreliable. Pilot should manually select wing sweep position by means of the emergency wing sweep handle.
6 WING SWEEP ADVISORY LIGHT	Illumination of the advisory light indicates failure of a single channel failure in the ADC programmer. Failure of the second (remaining) channel will illuminate the WING SWEEP warning light on the pilot's DDI panel. Manual wing sweep positioning is required using the emergency wing sweep handle.

(U) Commanded wing sweep position from the control box (independent of operational mode) is displayed by the bars alongside the wing position tape. Optimum wing position as determined by the programmer in the ADC is displayed by the triangular index which also indicates the most forward sweep angle that may be commanded under normal control modes. The five indicator windows on the right side of the wing sweep indicator display the operational control mode in effect.

WING SWEEP INTERLOCKS

(U) Electrical interlocks in the ADC must be satisfied before commanded changes to the control drive servos may be effected. Wing sweep interlocks within the ADC are depicted in figure 1-42. The main and auxiliary flap interlocks in the ADC are redundant to such interlocks in the wing sweep control box. Both auxiliary flaps must be retracted in order to remove that interlock. The normal acceleration (N_Z) input to the ADC inhibits wing sweep electrical command changes at less than -0.7g conditions. Under high negative g conditions it is necessary to inhibit sweeping of the wings to avert structural damage due to wing-fuselage contact.

CONTROL DRIVE SERVOS

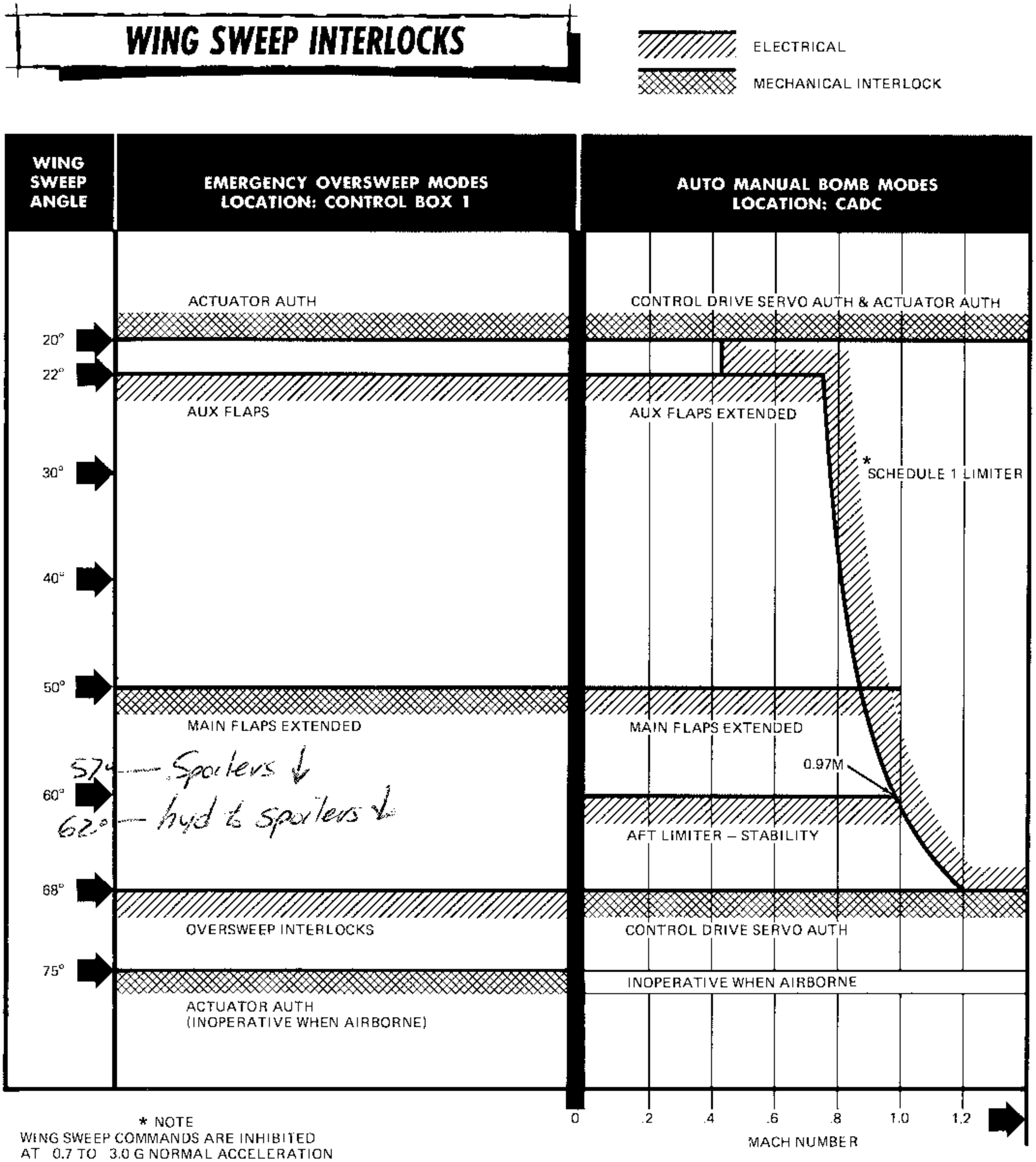
Electrical output signals from the ADC are converted into a common mechanical (rotary) input to the wing sweep control box by redundant control drive servos. This is the command path for wing position commands in the AUTO, MANUAL and BOMB modes.

Wing Sweep Control Box

(U) The wing sweep control box sums mechanical inputs and provides a mechanical output that is the difference between the input and the feedback. Electrical (auxiliary flaps, oversweep enable) and mechanical (main flap) interlocks input to the control box limit aft wing sweep commanda at 20 degrees, 68 degrees and 50 degrees respectively. The main flap interlock is mechanically inserted directly from the flap/slat control box. A surface retracted signal must be received from both auxiliary flaps before the 20 degrees electrical interlock may be withdrawn to enable aft wing sweep commands. Interlocks in the control box are illustrated in figure 1-42. These interlocks, which serve as a back-up to the electronic interlocks in the ADC, are imposed on both the normal and emergency control inputs to the control box and assure noninterference of movable surfaces.

WING SWEEP ACTUATING SYSTEM

(U) The mechanical output of the control box positions tandem servo valves which control the porting of combined and flight hydraulic system pressure to the left and right wing sweep actuating systems, respectively. The actuating system on each side consists of two hydraulic motors, one for sweeping and the other for unsweeping, driving an integral gear box and irreversible screw-jack actuators. During wing sweeping, the larger motor drives the gearbox while the unsweep motor acts as a pump; the reverse action holds true during wing unsweeping. This principal controls the wing sweep rate during aiding and opposing aerodynamic loads to prevent wing chatter during sweep operations. A cross shaft between left and right actuator gear boxes provides mechanical synchronization of the wings. In the event of a pressure failure in either of the main hydraulic systems, the remaining system



0-F050-004
060-0

Figure 1-42.

is capable of driving both actuators at a reduced rate via the cross shaft. Wing sweep position feedback to the control box is mechanically provided by a rotary shaft from the left wing sweep gearbox. The no-back characteristics of the wing sweep actuators are achieved by irreversible self-energizing brakes that prevent positional creep of the wings and provide sufficient rigidity to permit barricade engagements at the forward wing sweep condition (actuator fully extended).

WING SWEEP MODES

(U) The command source for positioning the wings is dependent upon the mode selected by the pilot or in certain cases is automatically selected. Electrical and mechanical wing sweep command paths are presented in figure 1-43, and described in the following subparagraphs. Pilot control through electrical command paths is accomplished using the wing sweep control switch on the inboard face of the right throttle, and mechanical commands are effected through the guarded handle on the inboard side of the throttle quadrant. Wing sweep interlocks for both the electrical and mechanical command paths are illustrated in figure 1-42. The electrical command path does not enable the pilot to penetrate structural or stability boundaries although subsonically in the MANUAL or BOMB mode, the wing may be set at too aft a sweep angle from a performance (lift/drag) standpoint.

Auto Mode

(U) In the auto mode of operation, the wings are automatically positioned according to a program in the ADC which is primarily a function of Mach number but includes pressure altitude biasing. Discounting transient positional lag or a system failure, the wing position will automatically track the programmer to relieve the pilot of this control task. Pilot selection of either of the two wing sweep programs in the ADC is effected by a SWEEP PRGM switch located on the left console outboard of the throttle quadrant. Program 1 schedules wing position to optimize specific excess energy whereas program 2 positions the wings at the optimum sweep angle for developing maximum usable lift for maneuvering. The two program schedules are illustrated in figure 1-43. In addition to serving an automatic wing positioning function, the program also defines the forward sweep command limit which cannot be penetrated using any of the other electrical command (manual or bomb) modes. The forward sweep limiter serves to prevent mispositioning of the wings from a wing structural standpoint.

(U) Selection of the auto mode is accomplished by placing the four-way, wing sweep control switch in the upper detented (hold) position. Such a selection permits the ADC wing sweep program to automatically position the wings based on air data inputs. Electrical position signals from the ADC are translated into a mechanical (rotary) input to the wing sweep control box by the No. 1 control drive servo. The No. 2 control drive servo is a redundant control path which becomes active with detection of a failure in the No. 1 command path of the ADC or control drive servo. The force exerted by the control drive servo input to the control box is insufficient to break out the "spider detent" so that the detent serves as a pivot point. The arm connected to the emergency wing sweep control handle moves with the control drive servo input such that the emergency handle in the cockpit serves as a secondary indication of commanded wing sweep angle. The primary sense of commanded wing sweep angle is electrically transmitted to the wing sweep indicator from the emergency

WING SWEEP MODES

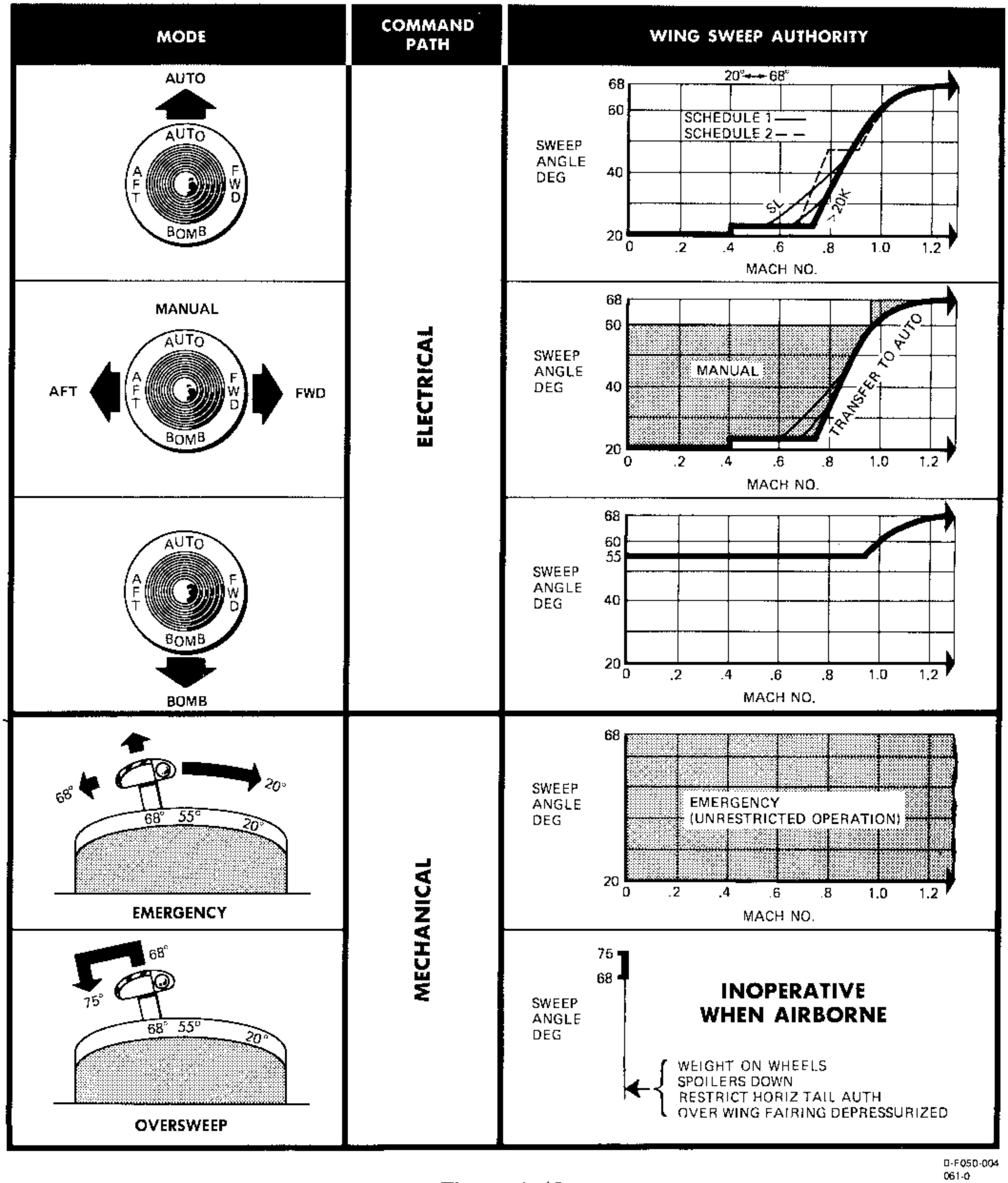


Figure 1-43.

handle input at the control box. Pilot selection of the auto mode or automatic transfer from the manual mode causes the AUTO flag to appear in the wing sweep indicator. Aside from a dual failure, transfer from the auto mode to some other command mode can result only from pilot selection of either remaining electrical or a mechanical command path. Once in the auto mode the four-way, wing sweep control switch can be in the center position without changing the command mode.

Manual Mode

(U) The manual mode of wing sweep control enables selection of wing sweep position within the forward sweep limit imposed by the automatic program and the aft sweep stability (60 degrees subsonic) or inflight authority (68 degrees) limits. Operational application of this mode includes pilot anticipatory action incident to performing specific maneuvers and use in fast cruise conditions. Pilot control in the manual mode is accomplished by forward and aft movement of the wing sweep control button which in turn slews the wing forward and aft following the same directional sense. The button is spring-loaded to return to the center position from the forward and aft momentary detents. Movement of the wing sweep control button to either extreme position provides a 15 degrees per minute wing sweep slew command (through the ADC and control drive servo) which remains in effect as long as the button is held displaced until a limit is intercepted. Transfer to the auto mode is automatic upon interception of the forward sweep limit imposed by the wing sweep programmer. Additionally, with the wing set at a discrete position using the manual mode, transfer to the auto mode will occur by accelerating the aircraft to a flight condition where the forward sweep limit of the program intercepts the manually set wing position. Therefore, with the wing sweep control button in the center position, either the auto or manual command mode may exist. Indication of the existing mode is provided by the AUTO and MAN flags in the wing sweep indicator.

Bomb Mode

(U) In the bomb mode of wing sweep control, the wings are positioned at 55 degrees or further aft by limiting action of the programmer as shown in figure 1-43. At these aft sweep positions, the wing bending moments are reduced such that a nominal 1g increase in normal acceleration is permitted over that of the forward, subsonic wing sweep angles at corresponding gross weight conditions. Selection of the bomb mode is effected by displacing the wing sweep control button to the downward position where it will remain engaged until deselected by the pilot. No separate discrete indication is presented in the wing sweep indication to denote the bomb mode. The MAN flag is displayed for both the manual and bomb modes with the wings at 55 degrees sweep angle. As the aircraft accelerates to a flight condition which is coincident with the 55 degrees wing sweep forward limit of the programmer, the AUTO flag appears in place of the MAN flag and the wing sweep continues aft following the program. Upon deceleration from supersonic conditions, the wing sweeps forward tracking the program until reaching 55 degrees where it remains until the bomb mode is deselected, the aircraft is again accelerated, or the emergency mode is activated.

Emergency Mode

(U) The emergency mode is a back-up means of wing sweep control which bypasses the electrical command path through the ADC and control drive servos by mechanically inputing directly to the control box. Any input (greater than 2 degrees sweep command) through the emergency handle overrides inputs from the fly-by-wire system. Therefore, except for wing flap (main and auxiliary) and oversweep interlocks in the control box, the emergency command mode does not prevent pilot mispositioning of the wings from a structural or stability standpoint as shown on the interlock illustration in figure 1-42. Because of the absence of normal (forward and aft) limiter cues in the electrical command path (unless the wing sweep program in the ADC is still operational), the pilot should follow the following schedule:

Mach Number	Wing Sweep
0 to .4	20 degrees
.4 to .7	25 degrees
.7 to .8	40 degrees
.8 to .9	50 degrees
.9 to 1.0	60 degrees
1.0 and greater	68 degrees

(U) The handle for emergency wing sweep control travels under the transparent guard located on the inboard side of the throttle quadrant. A series of actions is required to engage this mode. First, the transparent guard must be rotated aft out of the way to expose the emergency sweep handle. Thereafter, the handle may be raised (2 inches) to provide better access and leverage. Neither of these actions interrupt the existing command mode. A force of approximately 18 pounds in either direction (unless against an interlock stop in the control box) is necessary to disengage the "spider detent" which allows direct movement of the handle to command changes in wing sweep angle. Displacement of the emergency sweep handle out of the "spider detent" causes the EMER flag to appear in the wing sweep indicator and causes the WING SWEEP warning light to illuminate independent of the operational status of the electrical command paths. The sweep command bar on the wing sweep indicator is electrically transmitted from a pot at the other end of the mechanical control path in the control box.

(U) Additionally, the position of the emergency handle on the quadrant provides a cursory indication of the commanded sweep angle. Except for the force breakout discontinuity at the "spider detent" position, deflection of the handle in the raised position is unrestricted between the 20 degrees and 68 degrees command positions except for flap interlocks in the control box.

Oversweep Mode

(U) The wing oversweep mode enables on-deck sweeping of the wings aft from 68 degrees to 75 degrees thereby reducing aircraft plan size for spotting. At the 75 degrees position, the wings are swept over the horizontal tails so that the wing and tail span are coincident.

All interlocks in the wing sweep system must be satisfied before the wing position command in the control box can exceed the 68 degrees setting. Failure of a wing oversweep interlock (figure 1-42) can result in structural damage to the aircraft. Command inputs to the horizontal tail actuators restrict surface deflection to 18 degrees trailing edge up and 10 degrees trailing edge down. The restricted authority of the horizontal tail surfaces in the oversweep mode prevents wing-tail contact in the hydraulic power on condition. Except for horizontal tail authority and spoiler checks, a complete checkout of the primary flight control system can be performed with the wing in the oversweep position. The mass balancing of the horizontal tail about its pivot causes the trailing edge to droop downward with depletion of hydraulic pressure whereupon wing-tail contact occurs at rub strips.

(U) Initiation of the oversweep mode follows the same procedures as initiating the emergency mode except that the emergency wing sweep handle must be extended upward an additional 1/2 inch at the 68 degrees command position. An oversweep enable switch is activated by the extended vertical travel of the handle with weight-on-wheels. Approximately six seconds are required to depressurize the overwing fairings and drive the electrical oversweep stops into place in the horizontal tail control linkage. A HZ TAIL AUTH caution light will illuminate during transition of the authority stops to the commanded position. Thereafter, the emergency sweep handle may be moved aft to the 75 degrees oversweep position (approximately 18 pounds force must be applied to break out the "spider detent") and the handle and guard may be returned to the stowed position. The WING SWEEP warning light remains illuminated with the wing in the oversweep position and the OVER flag appears in the wing sweep indicator.

(U) With the "spider detent" disengaged in the emergency and oversweep modes, the electrical command path cannot induce changes to the hydraulic servo valves. Coming out of the oversweep mode is performed by the reverse actions except that no hesitation is required at the 68 degrees oversweep enable position. (If self-test of the wing sweep system has been performed while in the oversweep position, spider detent engagement should occur at the 68 degrees position emergency sweep handle.) Otherwise, forward movement of the emergency sweep handle may be necessary to engage the "spider detent".

WING SWEEP PREFLIGHT TEST

(U) A preflight test capability (enabled in the wing oversweep position only) of the wing sweep system assures proper operation of both electrical command channels without physically moving the wings. Activation of the test is initiated by rotating the master test switch to the WG SWP position and depressing the switch. Prior to initiation of test, depress the MASTER RESET button to extinguish the WING SWEEP advisory light, then initiate test within 5 seconds. During the test which takes approximately 10 seconds, the control drive servo of the channel I is commanded to the 44 degrees position and subsequently disabled. Thereafter, the channel II control drive servo takes over and moves the control input to the 68 degree position where it is disabled. At the same time, the wing sweep program index on the wing sweep indicator moves to 44 degrees, then 68 degrees, and returns back to the 20 degrees position. Any channel failure is indicated by illumination of the WING SWEEP advisory light during the test. The wings do not move during the self-test since the hydraulic servo valves are locked in the oversweep position by the emergency sweep handle being in the oversweep detent. Hydraulic pressure is not required for the execution of this test.

WING SWEEP FAILURES

(U) The electrical command path of the wing sweep control is continually monitored for failure detection purposes whereas mechanical and sensor failures provide more subtle failure indications. No single failure in the electrical command path can drive the wing to an unsafe position. Transient failures in the electrical command path may be reset to regain normal operation by depression of the MASTER RESET button located ahead of the throttle quadrant. This button recycles the failure detection process and 5 seconds must elapse before the final operational status may be ascertained.

(U) A failure detection system in the ADC governs the switching between channel I and II or the shutting off of the control drive servos and wing sweep related elements of the ADC. The two channel redundancy combined with on-line monitoring provides a fail operational system with immediate (built in 5 sec lag) detection of a failure condition in the electrical command path. A single channel failure in the wing sweep electrical command path is indicated by illumination of the WING SWEEP advisory light whereafter normal operation continues on the remaining channel. Failure of the remaining channel is indicated by illumination of the WING SWEEP warning light whereafter wing sweep control must be exercised through the emergency sweep handle. Transient failures in the ADC may be reset to normal operation by depression of the MASTER RESET button.

HIGH LIFT SYSTEM

MAIN FLAPS AND SLATS

(U) The main flaps (figure 1-44) consists of two simply hinged, single slotted flap panels that work in conjunction with two leading edge slat panels on each wing. During take-off and landing, flaps are extended 35 degrees and slats 17 degrees, with mechanical sequencing requiring 10 degrees main flap rotation before slat and auxiliary flap extension begins. In the trans-sonic range, main flaps may be extended to a maximum of 10 degrees for maneuvering. Flap cove doors which are mechanically slaved to the main flaps, fit flush to the flaps when they are retracted and provide optimum airflow over the flap surfaces when the flaps are extended.

(U) Each main flap panel is driven by two irreversible mechanical actuators; each slat panel is driven by a rack and pinion at the track supports. An asymmetry sensor attached to the flap drive, stops hydraulic power to the drive motors when asymmetry between wing flaps exceeds 3 degrees. There is no asymmetry provisions for the slat drive, but an overtorque release mechanism allows the flaps to continue to operate if the slats jam.

(U) The main flap and slat drive (figure 1-45) consists of a center gear box powered by two hydraulic motors on a common shaft. One motor, driven by the combined hydraulic system, is for normal operation. The second motor, driven by the outboard spoiler hydraulic module is for emergency back-up operation.

(U) Flaps may be operated manually or automatically. Manual repositioning the flap handle is transmitted to the hydraulic control valves by a controlled cable with a mechanical feedback from the center gear box.

MANEUVER FLAPS

(U) With the wing sweep angle less than 50 degrees, the main flap segments can be extended from 0 to 10 degrees as maneuver flaps, with proportional slats extending from 0 to 8.5 degrees. This maneuvering flap/slat feature is provided to augment the aircraft maneuver capability in the subsonic, mid-wing at higher load factors, over a wide range of altitudes. When wing lift is increased, maneuvering flap actuation is coordinated with wing sweep for maximum delay of buffet onset.

(U) Pilot induced maneuver flap commands are induced through the DLC thumbwheel on the control stick. The thumbwheel which is spring loaded to the center position, is also used for direct lift control and to manually extend the wing glove vanes. Commands from the

WING CONTROL SURFACES

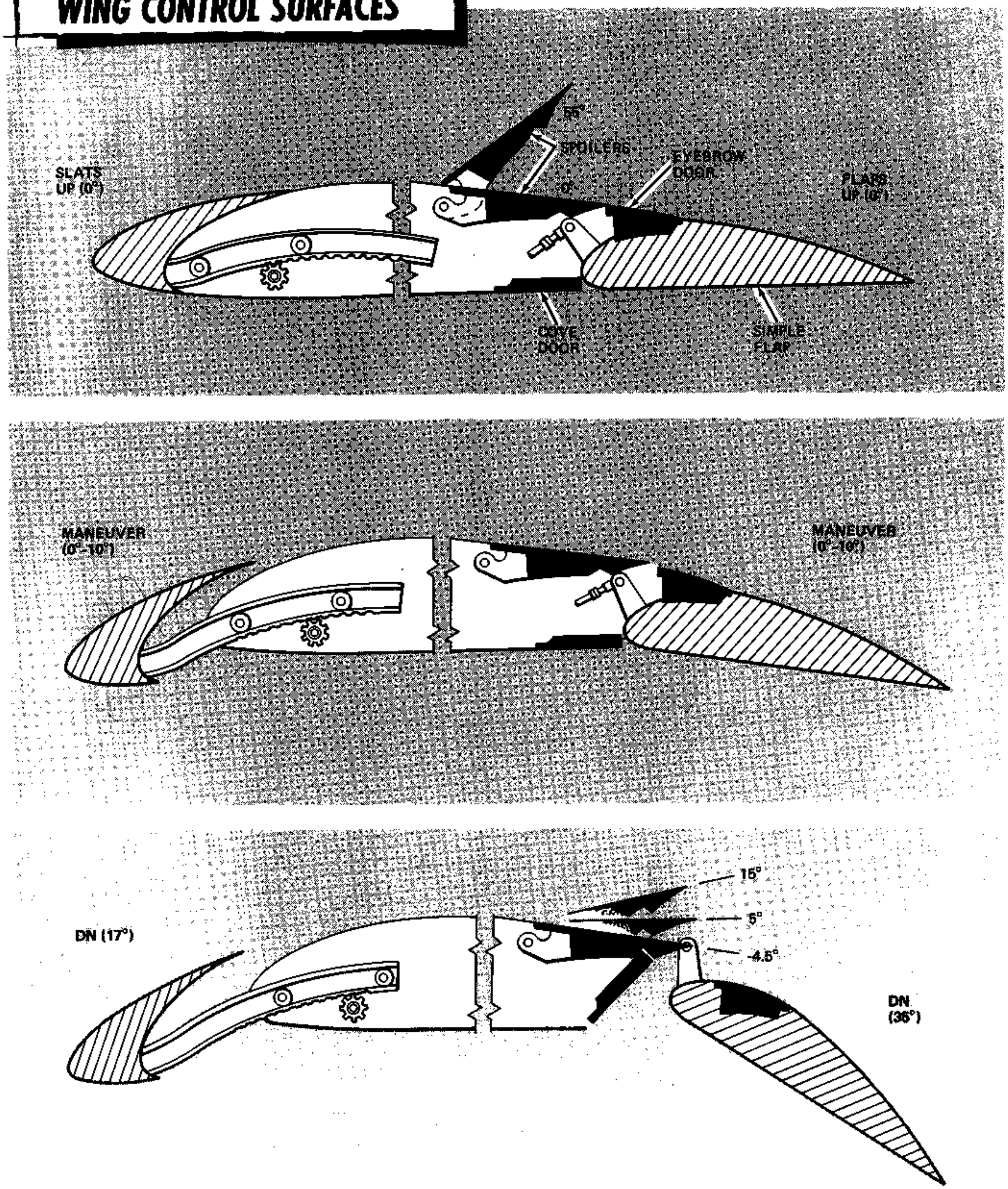


Figure 1-44.

HIGH-LIFT SYSTEM

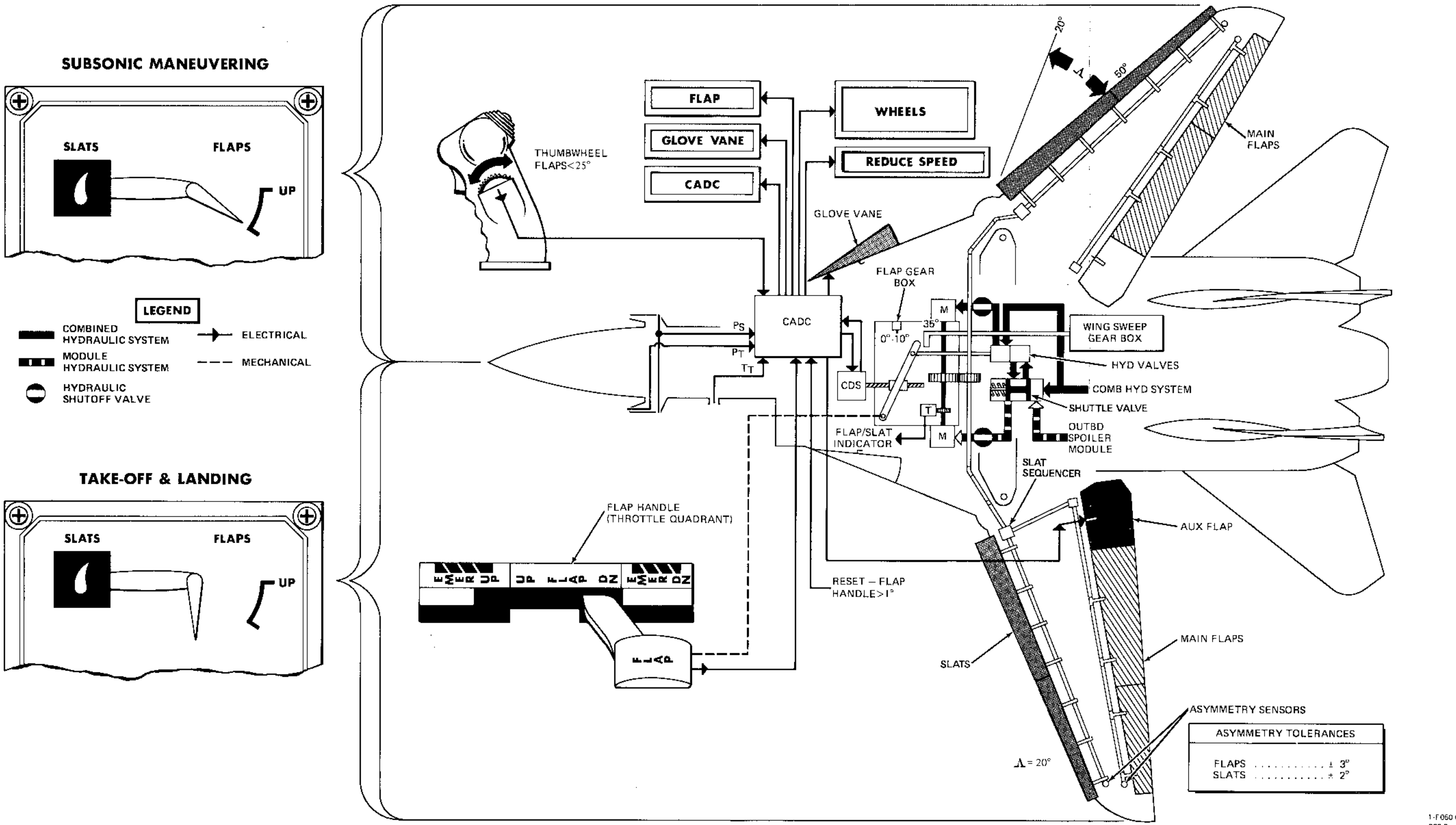


Figure 1-45.

thumbwheel are processed through the central air data computer where a limiter-inverter allows only safe value computed commands to activate flap/slat actuation. Maneuver flap actuation is inhibited unless:

- landing gear is up
- wing sweep angle is less than 50 degrees
- flap lever is up
- flap deflection limit (figure 1-46)

(U) Flap deflection limit computations are determined by the differential between the flap dynamic pressure and Mach number. As the maneuver flap limit deflection schedule (figure 1-46) indicates, permissible flap deflection for a given Mach number is increased until Mach 0.9. Because maneuver flaps are ineffective above Mach 1.0, the flap deflection limit drops sharply above Mach 0.95, assuring full maneuver flap retraction in the supersonic envelope.

(U) Asymmetry between wing flaps is maintained within 2 1/2 degrees by asymmetry switches. Comparison between flaps/slats and glove vane, and failure sensing are monitored by the ADC. A failure of the ADC, disagreement between flap handle and flap actuators, or flap and glove vanes positions' illuminates the FLAP caution light and the REDUCE SPEED caution light. The auxiliary flaps are not affected by asymmetry lockout switches.

MANEUVER FLAP-LIMIT DEFLECTION SCHEDULE

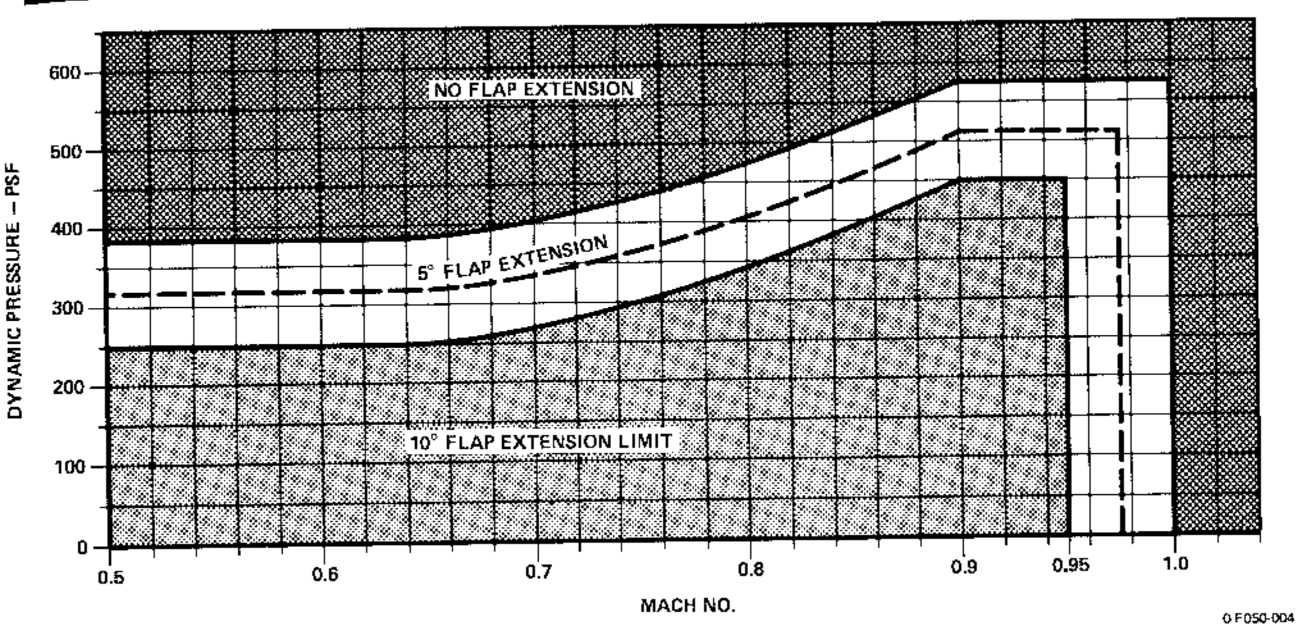


Figure 1-46.

AUXILIARY FLAPS

(U) The auxiliary flaps, (Fowler type) provide additional lift and drag during takeoff and landing. Each auxiliary flap is independently driven by a separate hydraulic actuator powered by the combined hydraulic system. The auxiliary flap actuators consist of double acting cylinders for lowing and raising the flaps commanded by the flap handle. Auxiliary flap actuation is restricted to wing sweep of 20 degrees (electrical interlock). Full down deflection is 35 degrees. Excessive air loads will cause the auxiliary flaps to retract to the up position, relieving pressure on the flaps down cylinder.

GLOVE VANE

(U) Each wing glove vane (figure 1-47) includes a stowed vane surface which, when extended at higher airspeeds increases maneuverability and decreases trim drag. The glove vane consists of a triangular panel, pivoted at the forward apex of the wing glove and supported by a track mechanism. They are controlled manually by the DLC thumbwheel (spring-loaded to neutral) on the pilot's control stick or automatically as a function of Mach number. With the landing gear down the DLC thumbwheel on the pilot's control stick commands direct left control (DLC). When the landing gear is up the DLC thumbwheel commands both maneuver flaps and glove vane. A single hydraulic system failure will not disable both vanes, and a safe carrier landing may be effected with one vane extended. Each vane extends 15 degrees at approximately 11 degrees per second (no load).

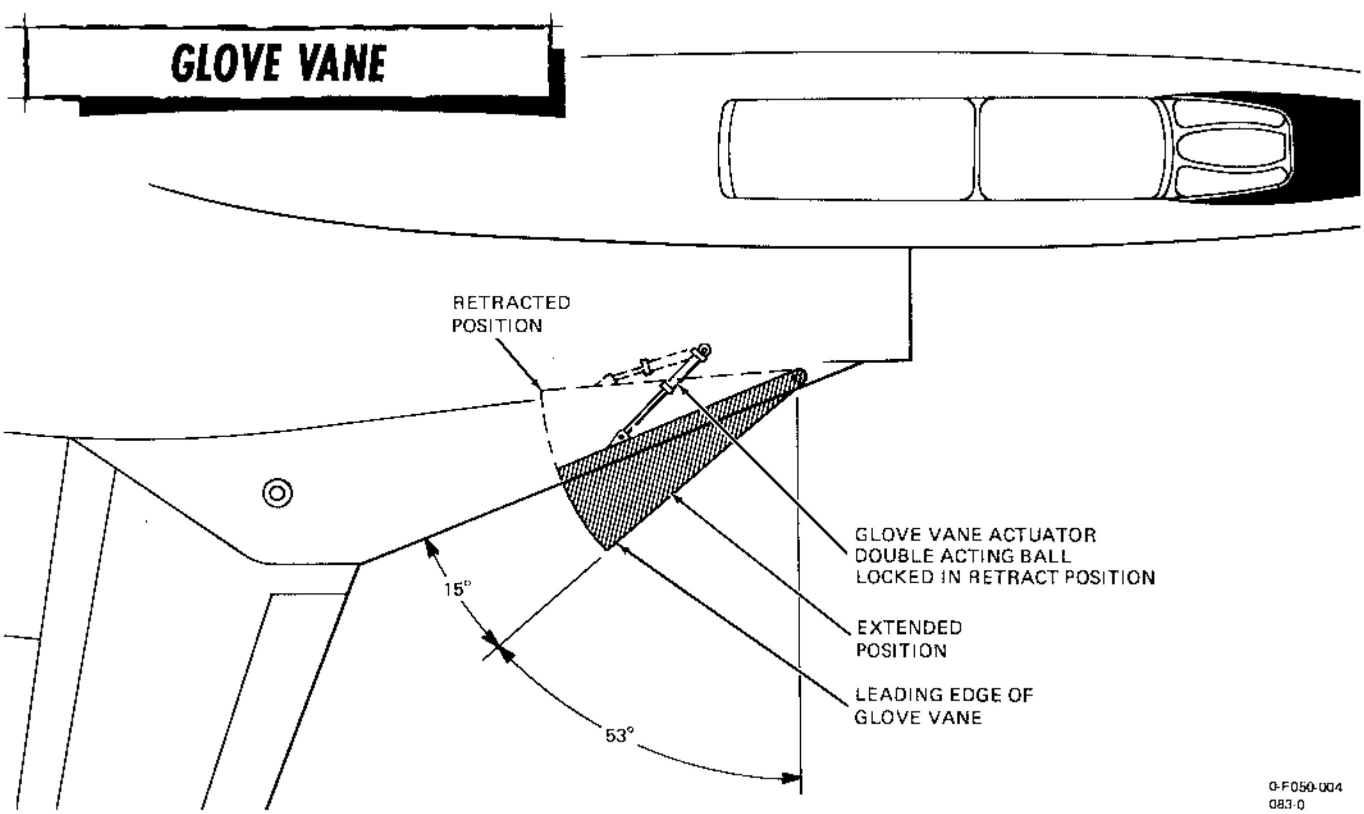


Figure 1-47.

(U) The vanes are extended through the leading edge of the wing glove structure by individual electro-hydraulic actuators. During flight the glove vane actuators are continuously pressurized. The left glove vane actuator is powered by the combined hydraulic system, while the right glove vane is supplied by the flight hydraulic system. Command signals from the ADC to the electro-hydraulic servo valves control the actuators that extend and retract the vanes. Independent of pilot command an optimum glove vane position schedule is computed by the ADC. The supersonic automatic program overrides any manual commands above Mach 1.5. Subsonic commands are limited by the wing sweep angle. Supersonic automatic glove vane extension commands are limited by altitude and Mach number. Pilot command signals are always compared to this Mach schedule in the ADC, and above Mach 1.4 the vanes are always fully extended. Between Mach 1.1 and 1.4, the supersonic glove vane program can be manually over-ridden by pilot commands.

(U) Failure detection provisions for ground test and inflight monitoring are contained in the ADC. Failsafe is provided by the glove vanes retracting to the closed position. Disparity between position sensor and commanded position signals will result in retraction of the glove vanes. Any failure of the glove vane system illuminates the GLOVE VANE caution light. Other than the caution light and the DLC thumbwheel there are no other cockpit controls or indicators for the wing glove vanes.

SPEED BRAKE SYSTEM

(U) The speed brakes consist of three individual surfaces, one upper and two lower panels, on the aft fuselage between the engine nacelles (figure 1-48). As a drag control device, the speed brake surface deflections may be infinitely modulated on the extension cycle whereas the retraction cycle is accomplished in a single step. Operating time for full deflection is approximately 2 seconds, and a blowback feature is incorporated to prevent exceeding structural limits. The upper speed brake is operated by two electrohydraulic actuators and the resulting motion is mechanically transmitted via interconnecting links to deflect the two lower surfaces. Hydraulic power is supplied by the combined hydraulic system (non-isolation circuit) and electrical power is through the essential No. 2 dc bus with circuit overload protection on the pilot's right circuit breaker panel.

(U) Speed brake position is displayed by a flip-flop indicator on the pilot's flap/gear position indicator, and pilot control of the speed brakes is effected directly by use of the three-position speed brake switch on the inboard side of the right throttle grip (figure 1-49). Aside from this control, automatic retraction of the speed brakes occurs with placement of

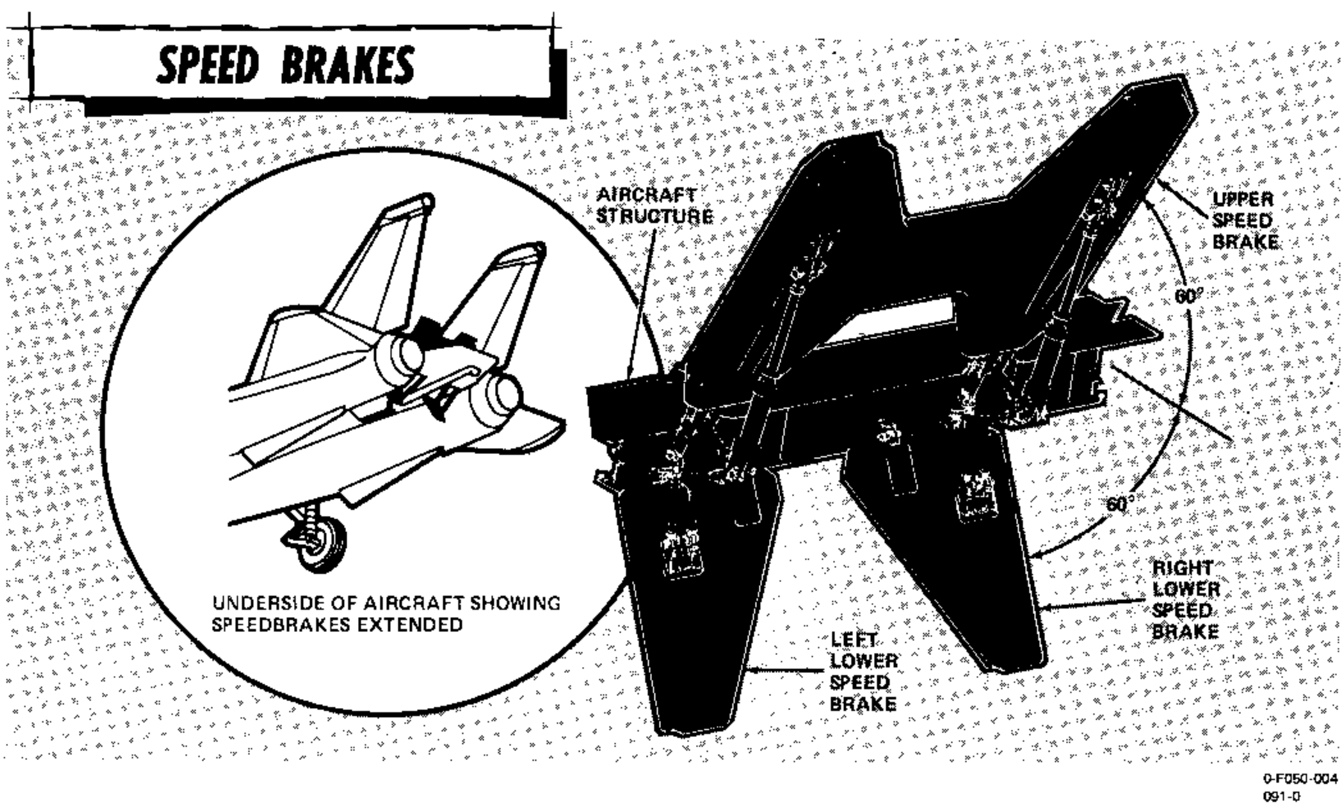
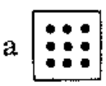


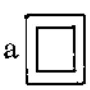
Figure 1-48.

either or both throttles at MIL or inflight loss of combined hydraulic pressure or electrical power. Automatic speed brake retraction with the throttle(s) at MIL or greater settings is provided as a safety and pilot relief function to facilitate aircraft clean-up during a wave-off/touch-and-go landing and to prevent incompatible thrust-drag conditions if the pilot forgets that the speed brakes are extended. The speed brake switch affects the energized state of the control valve which controls hydraulic power to operate the speed brakes. In the blowback mode the control valve relieves when pressure on the actuator side of the valve exceeds combined hydraulic system supply pressure and resets automatically. Subsequent to encountering the blowback mode, a reduction in aircraft speed will not cause the speed brakes to automatically extend to the original command position. Fuel dump operations are inhibited with the speed brakes not retracted to avoid fuel impingement on the fuselage boattail and nozzles. In this regard the pilot should refrain from extending the speed brakes in flight within nominally one minute after terminating fuel dump operations to allow residual fuel in the dump mast to drain.

SPEED BRAKE CONTROLS AND INDICATIONS

(U) Speed brake position is selected by a three-position thumb switch on the inboard side of the right throttle grip. The center position (temporary hold) is the normal position of the switch. Positioning the switch forward (RET) retracts the speed brakes; holding it aft (EXT) extends them.

(U) The EXT position is momentary and allows partial extension. When the switch is released it returns to the center position. Partial extension is indicated by a  flag on the wheel-flap position indicator. Full extension (60 degrees) of the speed brakes is shown by a  flag. The center position deenergizes the selector valve and blocks hydraulic fluid from both the extend and retract sides of the speed brake cylinders, hydraulic holding the speed brakes to any desired position.

(U) Automatic retraction of the speed brakes occurs by placing either or both throttles at MIL or inflight loss of combined hydraulic pressure or electrical power. When the speed brakes are in a fully retracted position a  flag appears on the wheels-flap position indicator.

FLIGHT CONTROL SYSTEMS

(U) Flight control is achieved through an irreversible, hydraulic power system operated by conventional stick and rudder controls. The stabilizer and rudders are powered by the flight and combined hydraulic systems and controlled by a mechanical linkage series of pushrods and bellcranks. A third independent flight control power source is provided by the backup flight control system. Spoiler control is effected by an electro-hydraulic fly-by-wire system and powered by the combined hydraulic system. The stabilizer and rudder control surfaces are mechanically driven by dual tandem power servo actuators. The spoilers are actuated by single system power servo actuators.

(U) Aircraft pitch is controlled by symmetrical deflection of the horizontal stabilizers. Roll control is effected by differential stabilizer deflections and augmented by spoilers at wing sweep positions less than 57 degrees. Directional control is provided by dual rudders. A mechanical lateral stick rudder interconnect (LSRI) provides adverse yaw compensation and assists the pilot to maintain coordinated flight in the landing approach configuration.

(U) The automatic flight control system (AFCS) includes a stability augmentation system (SAS) to augment the natural damping characteristics; an autopilot with numerous pilot relief modes; and provides auxiliary control functions for spoiler control, rudder authority control, lateral stick authority control and mach trim compensation.

LONGITUDINAL CONTROL

(U) Longitudinal control (figure 1-49) is provided by symmetric deflection of independently actuated horizontal stabilizer slabs. Control stick motion is transmitted to the stabilizer power actuators by pushrods and bellcranks routed under the right cockpit console and aft on the top center trough of the turtleback. Combat survivability was a major consideration for control system routing and component location. The hydraulic power actuator is a tandem design of two double acting power cylinders with split body construction and one common actuating shaft. The split body construction prevents the propagation of combat damage over the entire actuator force and possible loss of stabilizer control. Independent power is supplied to the dual tandem actuator by the flight and combined hydraulic systems. The actuator piston rod end connects to the stabilizer drive horn to rotate the stabilizer about the cantilevered stub shaft. The dual tandem power actuators control the stabilizers symmetrically for longitudinal control and differentially for lateral control. This is accomplished by mechanically summing pitch and roll stick commands algebraically at the pitch-roll mixer assembly. The composite pitch/roll command is transmitted mechanically from the pitch-roll mixer assembly to the individual stabilizer actuators. Pitch response to a control stick input is tailored by the stick-to-stabilizer gearing assembly. The non-linear stick-to-stabilizer gearing provides appropriate stick sensitivity for responsive and smooth control.

LONGITUDINAL CONTROL SYSTEM

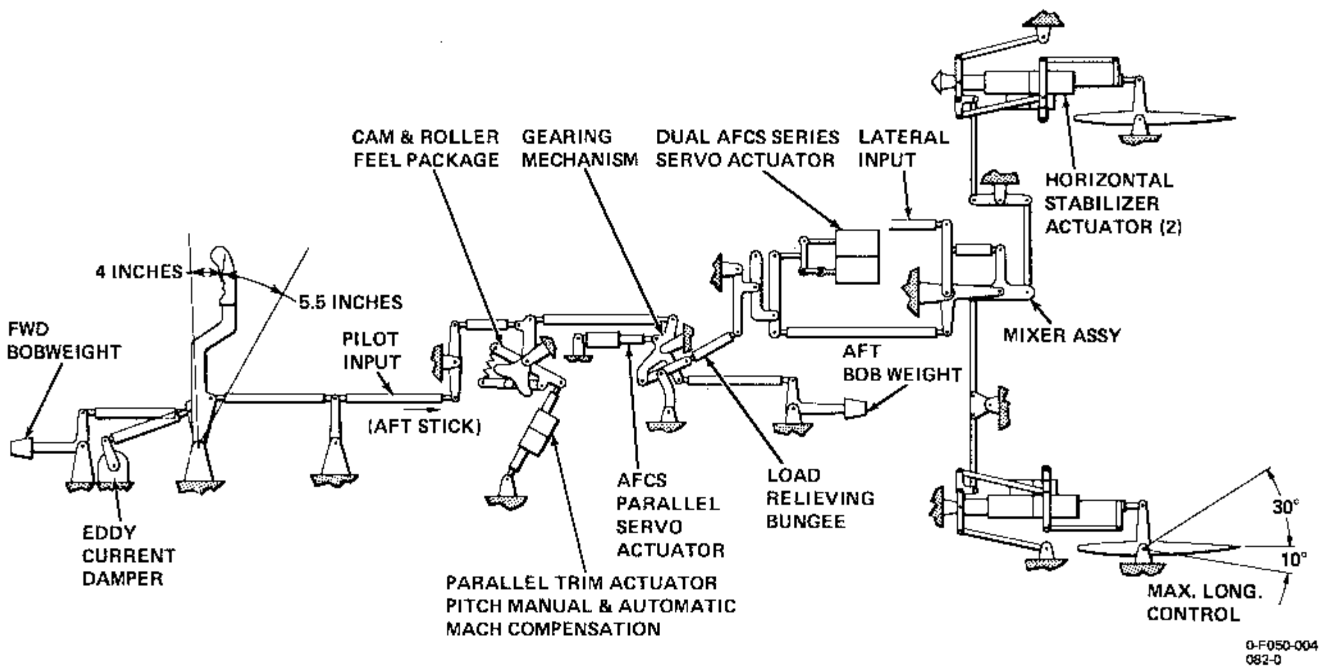


Figure 1-49

Longitudinal Feel

(U) Artificial fuel devices have been included in the control system to provide the pilot with force cues and feedback. A spring loaded cam and roller assembly produces increasing stick forces proportional to control stick displacement. The cam detent requires a 2.5 pounds breakout force to displace the stick from neutral. The stick displacement force gradient is 5 pounds per inch for the first inch and 3 pounds per inch over the remaining stick travel. This provides the pilot with a distinct force reference about the stick neutral position and a reduced gradient to minimize fatigue from large stick deflections during aggressive maneuvering. Control stick forces proportional to normal acceleration (g forces) and pitch acceleration are produced by fore and aft bobweights. Normal accelerations generate a 3 pounds per g stick force to provide the pilot with a dynamic maneuvering reference. Aircraft overstress from abrupt stick inputs is prevented by the eddy current damper. The eddy current damper generates opposing stick forces proportional to stick velocity and resists large rapid control deflections.

LONGITUDINAL SYSTEM AUTHORITY

COCKPIT CONTROL			STAB. SURFACE		PARALLEL TRIM	
ACTUATION	MODE	MOTION	AUTH	RATE	AUTH	AVG RATE
Control Stick	Manual	4" Fwd 5.5" Aft	10° Leu 30° Led	36°/Sec	6.0° Leu 16.1° Led	1°/Sec
AFCS	Series	None	± 3°	20°/Sec	-	-
	Parallel (AWCL Only)	4" Fwd 5.5" Aft	10° Leu 30° Led	36°/Sec	6.0° Leu 16.1° Led	0.1°/Sec
DLC Thumbwheel	Series	± 45° Dlc Thmbwl	8.4° Leu Max Δ is	36°/Sec	-	-

Longitudinal Trim

(U) Longitudinal trim is provided by varying the neutral position of the cam and roller feel assembly with an electro-mechanical screwjack actuator. The trim actuator incorporates two electric motor driven screwjack assemblies mounted back-to-back in a common housing with the jack screw operating rods in line and extending from opposite ends. One assembly of the actuator provides manual trim or AFCS controlled automatic trim. This section provides for two-speed trim operation with a high rate for manual trim and a lower speed for AFCS trim. The manual trim control switch on the stick is a five position switch spring-loaded to the center OFF position. The fore and aft switch positions produce corresponding nose down and nose up trim respectively. The manual trim switch is deactivated when AFCS pilot relief or ACLS modes are engaged.

(U) The other section of the trim actuator provides automatic mach trim compensation during transonic and supersonic flight. Mach trim control is provided by the AFCS and is continuously engaged. A failure of mach trim compensation is indicated by the MACH TRIM advisory light. Transient failures can be reset by depressing the MASTER RESET button.

(U) The manual/AFCS and Mach trim actuator is installed in parallel with the flight control system. Trim actuation produces a corresponding stick and control surface movement.

(U) The flight control system surface positions are indicated on the composite surface position indicator located on the left quarter panel, forward of the throttle. The trailing edge positions of the left and right stabilizer are independently displayed in degrees. Position information is transmitted from individual sensors located on the stabilizer drive horns.

LATERAL CONTROL SYSTEM

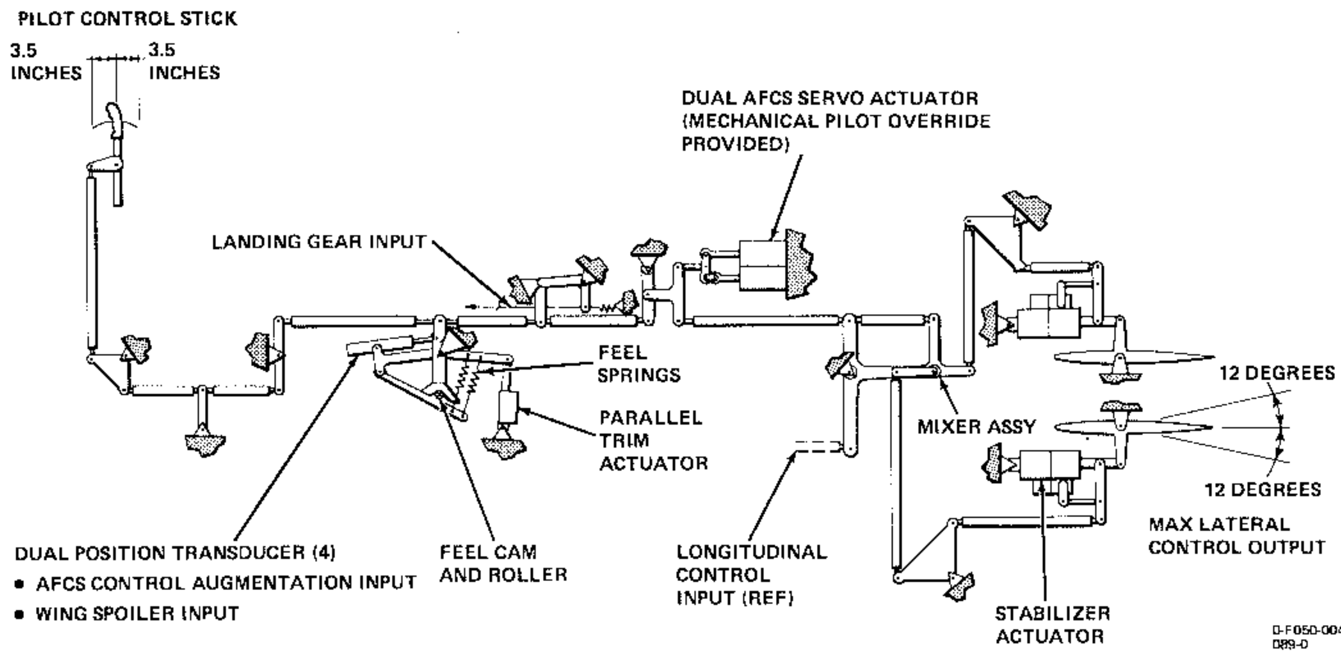


Figure 1-50.

LATERAL CONTROL

(U) Lateral control (figure 1-50) is effected by differential displacement of the stabilizers and augmented by wing spoilers at wing sweep positions of less than 57 degrees. The spoilers are interlocked in the flush down position at wing sweeps greater than 57 degrees and roll control is provided entirely by differential stabilizers. Lateral stick command is transmitted by pushrods and bellcranks routed under the right cockpit console and aft in the top turtleback trough to the pitch/roll mixer assembly. Pitch and roll commands are algebraically summed at the mixer assembly and a composite pitch/roll command is mechanically transmitted to the independent stabilizer power actuators. The power actuator piston operating rod is connected to the stabilizer drive horn to produce stabilizer rotation.

Lateral Feel

(U) An artificial feel system is incorporated to provide the pilot with force cues and feedback. The lateral feel mechanism is a spring roller cam assembly with a neutral stick position detent and a constant stick deflection force gradient. The neutral stick breakout force is 1.5 pounds and the stick deflection force gradient is 2.5 pounds per inch. This provides the pilot with a distinct neutral stick reference while maintaining a comfortable lateral force level for quick and sensitive roll control.

Lateral Trim

(U) Lateral trim is achieved by differential deflection of the horizontal stabilizers. The wing spoilers are not actuated for lateral trim control. Trim is provided by adjusting the neutral position of the spring roller cam feel assembly with an electro mechanical screw-jack. Left or right deflection of the trim switch on the stick grip produces corresponding stick movement and left or right wing down trim respectively. The "coolie hat" momentary trim switch is spring loaded to the center OFF position. The normal stick grip trim switch is deactivated when AFCS or ACL modes are engaged.

Lateral Control Stops

(U) A structural weight and wing loading penalty would be imposed if maximum differential stabilizer deflections were permitted throughout the flight envelope. The nominal increase in usable performance would not be commensurate with the increased weight and wing loading penalty. To limit the torsional fuselage loads variable lateral control authority stops are installed. The lateral stick stops vary according to dynamic pressure airloads from full stick authority at low q, to one half stick throw limits at high q conditions. Failure of the lateral stick stops is indicated by illumination of the HZ TAIL AUTH caution light. Transient failures can be reset with the MASTER RESET button. The position of the stops following a failure can be estimated by the pilot from the available stick deflection limits. Failure of the stops in the one half stick position does limit low q rolling performance, however, ample roll control is available for all landing conditions and configurations. Failure in the open position requires the pilot to manually limit stick deflection, or, under dynamic maneuvering conditions when following the schedule in not practicable, to limit stick deflections to one half.

LATERAL SYSTEM AUTHORITY

CONTROL SURF.	COCKPIT CONTROL			SURFACE		PAR. TRIM	
	ACTUATION	MODE	MOTION	AUTH	RATE	AUTH	RATE
Stab (DIFF)	Control Stick	MAN.	3.5" LT 3.5" RT	±7°	36°/SEC	±3°	3/8° SEC
	AFCS	Series	None	±5°	33°/SEC	-	-
INBD & OUTBD Spoilers	Control Stick	MAN. 57°	4.5" LT 4.5" RT	+55°	250°/SEC	None	None
	AFCS(ACL)	Series	None	15°MAX			
	DLC	MAN	DLC THMBWL ±45°	+9°, -7 1/2° About +3° NEUT. POS	125°/SEC		
	GRD Roll BRK Arm SW&WOW	Series	None	55°UP	250°/SEC		
LAT Stops	CONT Stick RES	MAN	1.75"LT 1.75"RT	±3 1/2° A 28° SP	36°/SEC 250/SEC		

DIRECTIONAL CONTROL

(U) Directional control (figure 1-51) is effected by twin rudders, one on each vertical stabilizer. The twin stabilizer design provides excellent directional stability and control for smooth air-to-air tracking, responsive air combat maneuvering and steady carrier landing control. A single tail to achieve equal control would require the added complexity and weight of a power fold mechanism to provide hanger deck clearance. Full rudder throws of ±30 degrees correspond to ±3 inches of rudder pedal travel. The rudder pedals adjust through a 10 inch range in one inch increments with the adjust control located on the lower center pedestal forward of the stick.

(U) The rudder positions are indicated on the control surface position indicator. Individual rudder pointers marked R and L display the trailing edge position of the rudders in degrees.

(U) Directional commands are transmitted mechanically from the rudder pedals to the rudder power actuators by pushrods and bellcranks. Control linkage routing is aft under the right cockpit consoles and aft in the upper turtleback trough to provide maximum structural protection from combat damage. The power actuators are a tandem arrangement of two double acting power cylinders around a common piston operating rod. The actuator rod end is connected to the rudder control horn of the rudder base. The tandem cylinders are of a split body construction to minimize combat damage effects. The two power cylinders are powered independently by the flight and combined hydraulic systems. Full directional control is available with a single system hydraulic failure and sufficient control for a safe return and field landing is provided by the back-up flight control module should both the flight and combined systems fail.

Directional Feel

(U) Artificial feel is provided with a spring roller-cam mechanism similar to the longitudinal and lateral feel systems. A rudder pedal neutral detent in the cam requires 8 pounds breakout force for initial rudder deflection. Rudder force with pedal deflection is non-linear with a relatively steep gradient about neutral and gradually decreasing with increased pedal travel. The artificial feel forces provide the pilot with a distinct neutral reference and comfortable rudder forces for sustained maneuvering and nose wheel steering operation.

Directional Trim

(U) Directional trim is effected by varying the neutral position of the feel assembly with an electro-mechanical screwjack actuator. Trim control is actuated by a three position switch on the left console outboard of the throttle quadrant. Left and right lateral switch movement commands left and right rudder trim respectively. The switch is spring loaded to the center OFF position. Trim actuation produces an associated movement of the rudder pedals and rudder trim indicator.

DIRECTIONAL CONTROL SYSTEM

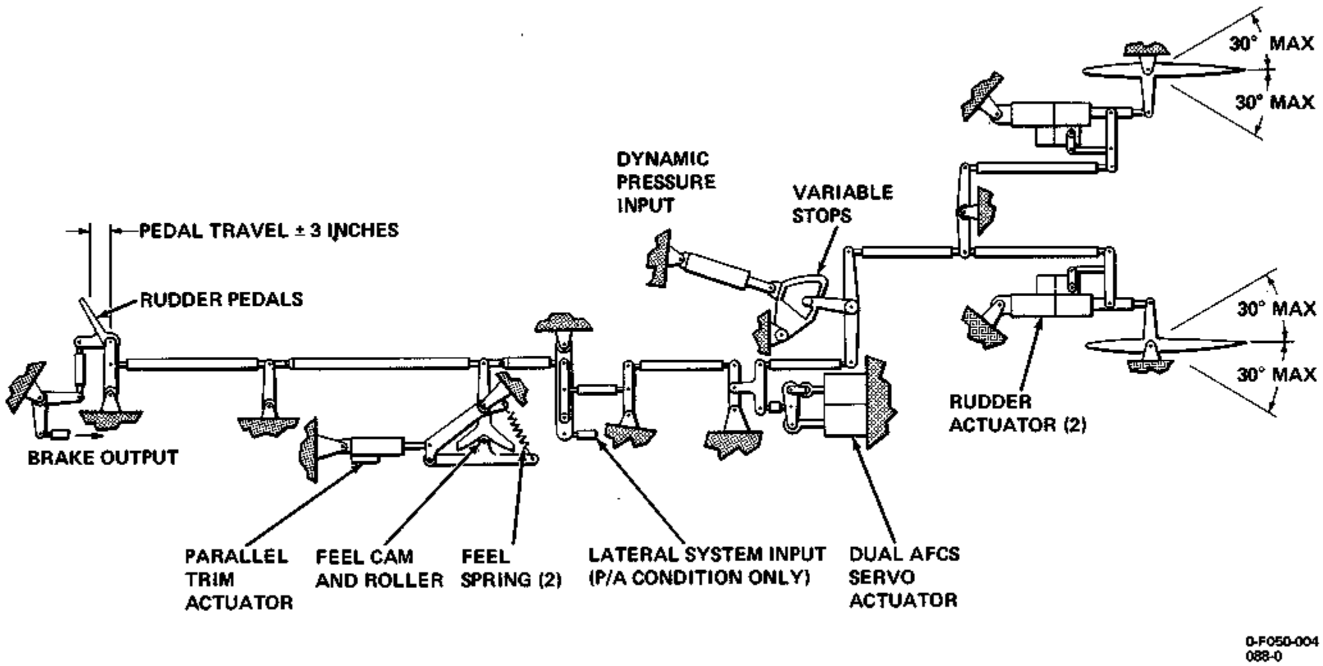


Figure 1-51.

DIRECTIONAL SYSTEM AUTHORITY

COCKPIT CONTROL			RUDDER SURFACE		PARALLEL TRIM	
ACTUATION	MODE	MOTION	AUTH	RATE	AUTH	RATE
Rudder Pedals	Manual (Unres)	3" Lt 3" Rt	±30° Max	106°/Sec	7°	1.13°/Sec
	Manual (Res)	0.7" Lt 0.7" Rt	±9.5° Min	106°/Sec		
AFCS	Series	None	±9.5°	80°/Sec	-	-
LSRI Lat Stick	Manual	3.5" Lt 3.5" Rt	±11°	106°/Sec	±3.25°	0.4°/Sec

ROLL CONTROL

Spoiler roll control, operational at wing sweep angles less than 57 degrees, is effected by spoiler deflection proportional to control stick transducer deflection. Full lateral stick results in 55 degrees of spoiler deflection. The spoiler roll transducers are positioned only by manual stick movements and are insensitive to lateral trim commands. The proportion of spoiler deflection with stick movements varies with flap position and DLC to optimize roll response and maneuverability. A ±1/2 inch stick deadband is provided to preclude spoiler actuation with small lateral stick roll commands. Small roll corrections are made with differential stabilizer positions. The spoilers are flush with the wing when the flaps are up and the stick is centered. When the flaps are extended, the spoilers are mechanically drooped to -4 1/2 degrees to form an aerodynamic slat with the cove door, eyebrow door and flap. In this configuration spoiler deflection for roll control is proportional to lateral stick deflection from -4 1/2 to +55 degrees.

RUDDER AUTHORITY STOPS

Rudder authority control stops are incorporated to limit rudder throws in the high q flight environment. Unlimited rudder throws at high airspeeds would require a substantial increase in structure without a performance increase. Rudder loads are controlled by limiting rudder linkage movement with a cam positioned by an electromechanical screwjack. Rudder deflection limits are scheduled by the ADC which also monitors and compares the commanded limits and actual rudder cam stop position. Disagreement of command and position removes power from the cam motor and lights the RUDDER AUTH caution light. Rudder authority is never less than 9.5 degrees in the closed limits position. This provides more than adequate single engine landing and wave-off control; and ample control for normal air combat maneuvering.

SPOILERS

(U) Four spoiler control surfaces (figure 1-52) are installed on the upper surface of each wing to augment roll control power, provided direct lift control and implement aerodynamic ground roll braking. The spoiler panels are arranged in pairs and powered by independent hydraulic sources. The inboard pairs are powered by the combined hydraulic system. To limit combined system exposure to combat damage in the highly vulnerable wing area, an independent electro-hydraulic module provides power for the outboard spoiler pairs. The spoiler actuators are electro-hydraulic single system, double acting servo actuators. Hydraulic actuation of the servo actuators is controlled by electric servo valves at the actuator and command by control stick transducers. The spoiler actuator operating rod ends are mechanically connected with a bellcrank to the spoiler operating arms to effect spoiler deflection.

(U) The spoiler positions are indicated on the central surface position indicator. Four spoiler window indicators reflect the position of the four spoiler pairs. A pictorial presentation of spoiler shape indicate the UP and DROOP positions. The abbreviation letters DN indicate the spoiler down and flush with wing position.

SPOILER CONTROL SYSTEM

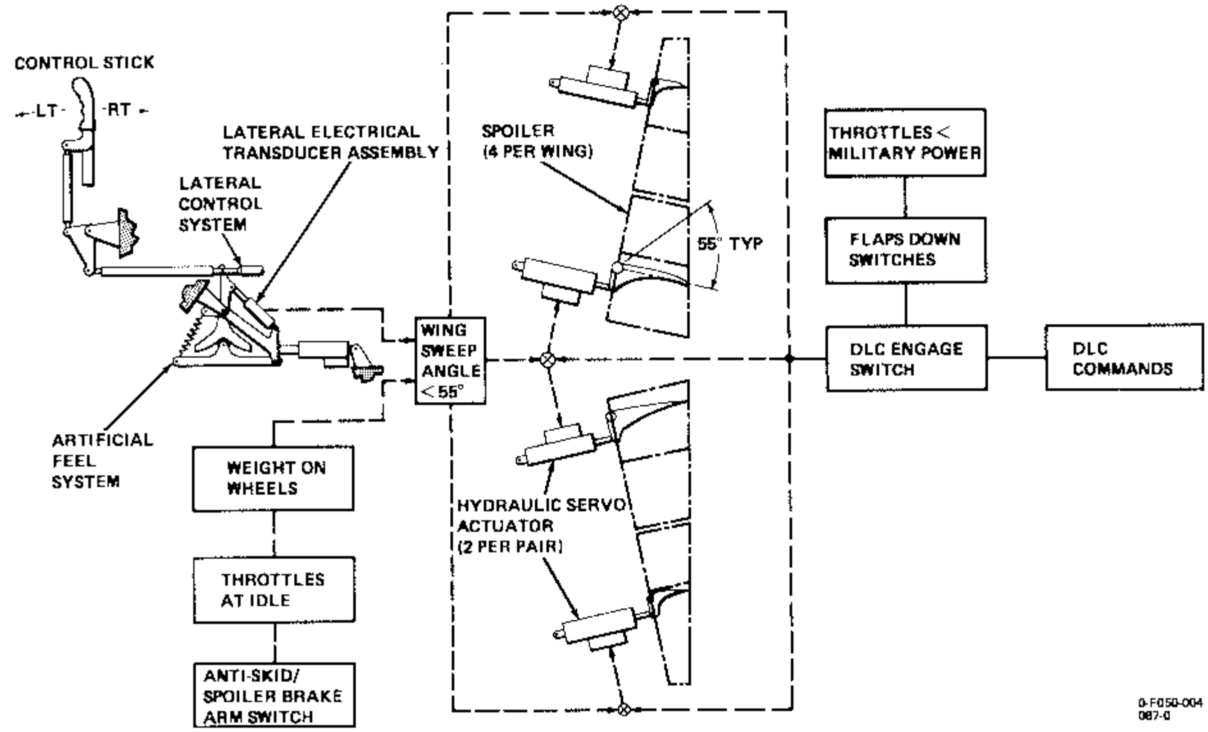


Figure 1-52.

SPOILER POSITION INDICATOR

(U) Spoiler position is displayed on the control surface position indicator located on the pilot's left vertical console. Four flip-flop indicators, two for the left and two for the right spoilers, display three spoiler positions.

(U) The indicator displays spoilers down DN (flush with wing surface); extended more than zero degrees , or dropped (-4 1/2 degrees below wing surface) . Differential stabilizer position is also shown on the control surface position indicator and is depicted as the left or right wing down.

(U) The inner pair of indicators display both number 1 and number 2 spoiler positions while the outer pair show number 3 and number 4 spoiler positions. All indicators are activated by the spoiler actuator. If either spoiler in a pair is at a position greater than zero degrees, the indicator will show an extended flag. However, both spoilers in a pair must be at -4 1/2 degrees before a dropped flag appears.

GROUND ROLL BRAKING

(U) Aerodynamic ground roll braking is provided by symmetric deflection of all spoilers to +55 degrees. Ground roll braking is controlled by the ANTI SKID/SPOILER BK switch on the pilot's left vertical console. The three position switch allows optional select of both spoilers and wheel anti-skid braking without anti-skid or OFF where neither spoilers nor anti-skid is armed. With SPOILER BK selected two conditions are required to actuate the apoilers:

- Weight on wheels
- Both throttles at idle

Failure to satisfy any one of the above conditions will cause the spoilers to return to the down position.

SPOILER LOGIC AND TEST

Wing Sweep

(U) The spoilers are inoperative at wing sweep angles greater than 57 degrees. With increasing wing sweep the spoiler contribution to roll control decreases as the angle of the spoiler chord to the relative wind decreases. Additionally, the geometry of the spoilers and overwing fairing requires the spoilers to be flush for wing sweeps beyond 57 degrees. The spoiler command signal is removed at 57 degrees and the spoilers are hydraulically retracted to 0 degrees. Mechanical spoiler downlocks are inserted at 62 degrees. Roll response is not compromised with wing sweep spoiler lockout as the differential tail provides more than ample roll control power with increasing wing sweep and decreasing roll inertia.

Asymmetric Spoiler

(U) Eighteen degree limit switches are installed on each spoiler panel to detect asymmetric spoiler deflection. The spoiler switches operate in conjunction with stick position so that any spoiler deflection greater than 18 degrees together with an opposite roll command stick position, signals a spoiler failure command. A stick displacement of one inch to oppose the asymmetric spoiler will remove all electrical command and signals and hydraulically power the spoilers down to the -4 1/2 degrees position. In the flap up configuration, the spoilers at the -4 1/2 degrees position will deflect the spring loaded eyebrow doors and be properly positioned for the flaps down configuration.

Symmetric Spoiler Deflection

(U) The eighteen degree spoiler limit switches also control the spoiler system logic to prevent simultaneous deflection of corresponding spoiler pairs on both wings in flight. Simultaneous deflection of both corresponding spoiler pairs greater than 18 degrees will signal a spoiler failure, remove all electrical commands to the affected spoiler set, and hydraulically power the failed spoiler pairs to the -4 1/2 degrees position. Simultaneous deflection of non-corresponding spoiler pairs, i.e., inboard pair on one wing and outboard pair on the other wing, will not signal a symmetric spoiler failure. The probability of this cocurrence, which is essentially a simultaneous double asymmetric failure, is extremely remote and can be easily corrected by a ±1 inch stick movement. This will remove all electrical commands and power the spoilers to the -4 1/2 degrees position. With either or both SPOILER BK or BOTH selected and weight on wheels the symmetric spoiler failure logic is disarmed to permit symmetric 55 degrees spoiler deflection for ground roll braking.

Spoiler Failure

(U) Following a spoiler failure, the SPOILERS and MASTER CAUTION light will illuminate. Transient spoiler failures can be reset by depressing the MASTER RESET button on the pilot's left vertical console. A failure of the AFCS roll computer will fail the inboard spoiler pairs. An AFCS pitch computer failure will render the outboard spoiler pairs inoperative. Passive spoiler failures which do not include the spoiler or stick logic switches will not be displayed to the pilot. DLC cannot be engaged with a spoiler failure.

Spoiler Test

(U) The spoiler test provides a continuity and logic test for the spoiler 18 degrees and stick switches. The test is conducted with the flaps down and engines operating. Test procedure is as follows:

- Select SPOILER BK to deflect spoilers to 55 degrees.
- Move stick one inch left which will initiate a spoiler failure. The SPOILERS and MASTER CAUTION lights will illuminate and the spoiler will retract to the -4 1/2 degrees drooped position.
- Depress MASTER RESET button to restore initial conditions and repeat with a one inch right stick movement.

CONTROL STICK CONTROLS

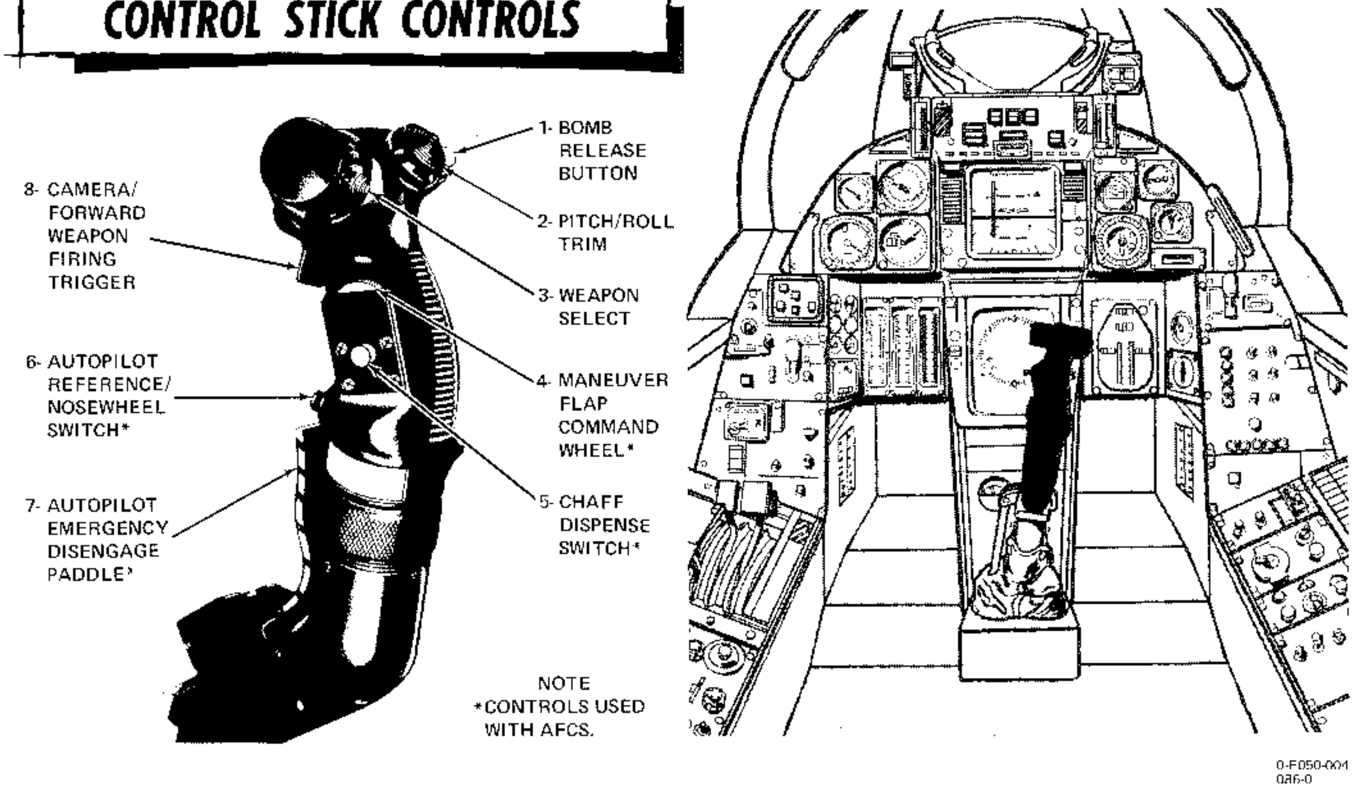


Figure 1-53.

	NOMENCLATURE	FUNCTION
1	BOMB RELEASE BUTTON	Pilot control for release of stores.
2	PITCH/ROLL TRIM BUTTON	Spring-loaded to (center) off position. Up and down positions control pitch trim and left and right positions control roll trim. Manual trim is inoperative during autopilot operation.
3	WEAPON SELECT SWITCH	SP/PH - Selects Sparrow or Pheonix missiles. SW - Selects Sidewinder missiles. GUN - Selects gun. OFF - Inhibits all weapon selection by the pilot.

	NOMENCLATURE	FUNCTION
4	MANEUVER FLAP/DLC COMMAND WHEEL	Spring-loaded to a neutral position. Forward rotation retracts spoilers; aft rotation extends spoilers. DLC must be engaged.
5	DLC ENGAGE DISENGAGE/ CHAFF SWITCH	Momentary depress the switch with flaps down, throttles less than MIL, and on failures in spoiler system engages DLC. With flaps up switch will dispense chaff. DLC is disengaged by momentary depressing switch or advancing throttles to MIL.
6	AUTOPILOT REFERENCE/ NOSEWHEEL SWITCH	With weight on wheels nosewheel steering is engaged. Weight off wheels and autopilot engaged switch engages compatible autopilot modes.
7	AUTOPILOT EMERGENCY DISENGAGE PADDLE	Disengages all autopilot modes and releases all autopilot switch. Momentary disengages (as long as switch is held) Pitch and roll SAS servos; switches remain ON. Yaw SAS is not disengaged.
8	CAMERA/FORWARD WEAPON FIRING TRIGGER	Pilot control of gun camera or forward weapons.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

(U) The automatic flight control system (AFCS) (figure 1-54) augments the aircraft's natural damping characteristics and provides automatic commands for control of attitude, altitude, heading and approach modes selected by the pilot. All AFCS functions except automatic carrier landing features are integrated into the primary flight control system. A built-in-test (BIT) capability provides in-flight monitoring for fail safe operation and an automatic operational readiness test for pre-flight checks.

(U) The AFCS is functionally divided into the stability augumentation system (SAS) and autopilot. Stability augumentation is provided for all three aircraft axis (pitch, roll, yaw). Control surface commands are generated by the roll and yaw computers in response to inputs from the AFCS control panel, aircraft attitude and heading sensors, air data computer (ADC), and data link commands. Computer outputs are fed through the flight control system dual series actuators which drive the control system mechanical linkages to produce surface motion. In an automatic carrier landing mode, pitch attitude command signals drive a separate pitch parallel actuator to drive the horizontal tail, while rod attitude commands drive the spoilers. Pitch and roll SAS axes consists of three completely redundant channels that provide fail-operational, fail-safe operation. Control surface rates and authorities in response to AFCS inputs are summarized in Table IV.

TABLE IV. AFCS RATES AND AUTHORITIES

AXIS	ACTUATOR	SURFACE	AUTHORITY	RATE
PITCH	Dual Series	Stabilizer	±3°	20°/sec
	Parallel (ACL Only)	Stabilizer	10° LEU 30° LED	36°/sec
	Parallel Trim	Stabilizer	6° LEU 16.1° LED	0.1° sec
ROLL	Dual Series	Diff. Stabilizer	±5°	33°/sec
		Spoilers (ACL)	15° MAX	250°/sec
YAW	Dual Series	Rudder	±9.5°	80°/sec

STABILITY AUGMENTATION OPERATION

(U) Stability augmentation is controlled by the three STAB AUG switches located on the upper half of the AFCS control panel (figure 1-55). SAS is engaged by placing these switches to the ON position during normal post start procedures. The PITCH and ROLL STAB AUG switches are solenoid-held and the YAW STAB AUG switch is a manually operated toggle switch. All three STAB AUG switches should normally remain in the ON position throughout the entire flight.

AUTOPILOT OPERATION

(U) The autopilot is controlled by four switches located on the lower half of the AFCS control panel (figure 1-55) and the NWS/REF ENGAGE switch located on the stick grip. Before the autopilot is engaged, first ascertain that all three SAS axes are engaged, then place the ENGAGE/OFF switch to the ENGAGE position. No warmup period is required. The autopilot may be engaged with the aircraft in any attitude. If however, aircraft attitude exceeds ±30° in pitch and ±60° in roll, the autopilot will automatically return the aircraft to these limits.

Pilot Relief and Guidance Modes

(U) With the autopilot engaged, maneuvering the aircraft by control stick steering results in reversion to augmented manual flight. The control stick will command the stabilizer and spoilers as appropriate for pitch and roll control. A force of approximately two pounds in pitch and one pound in roll is required to effect control in this mode.

(U) Attitude Hold - Attitude hold is selected by establishing the desired attitude using control stick steering and then releasing pressure on the stick. The autopilot will hold pitch attitudes up to ±30° and bank angles up to ±60°.

(U) Heading Hold - Heading hold is engaged by setting the HDG-GT switch to the HDG position. After maneuvering the aircraft to the desired reference heading, release the control stick at a bank angle of less than ±5°. The autopilot will then hold the aircraft on the desired heading. The A/P REF NWS switch is not used in the heading hold mode. Heading reference is obtained from either the inertial navigation system or attitude heading and reference (AHRS) system, selected by the NFO.

(U) Ground Track - To engage ground track set the HDG-GT switch to GT. Upon illumination of the A/P REF light (on left side of the vertical display indicator), press the A/P REF NWS button on the control stick grip. When the A/P REF light goes out, the mode is engaged. Disengagement will occur if more than 1 1/4 pounds lateral stick force is applied and will be indicated by illumination of the A/P REF light. The ground track mode may be re-engaged by releasing the stick force and depress the A/P REF switch.

AUTOMATIC FLIGHT CONTROL SYSTEM

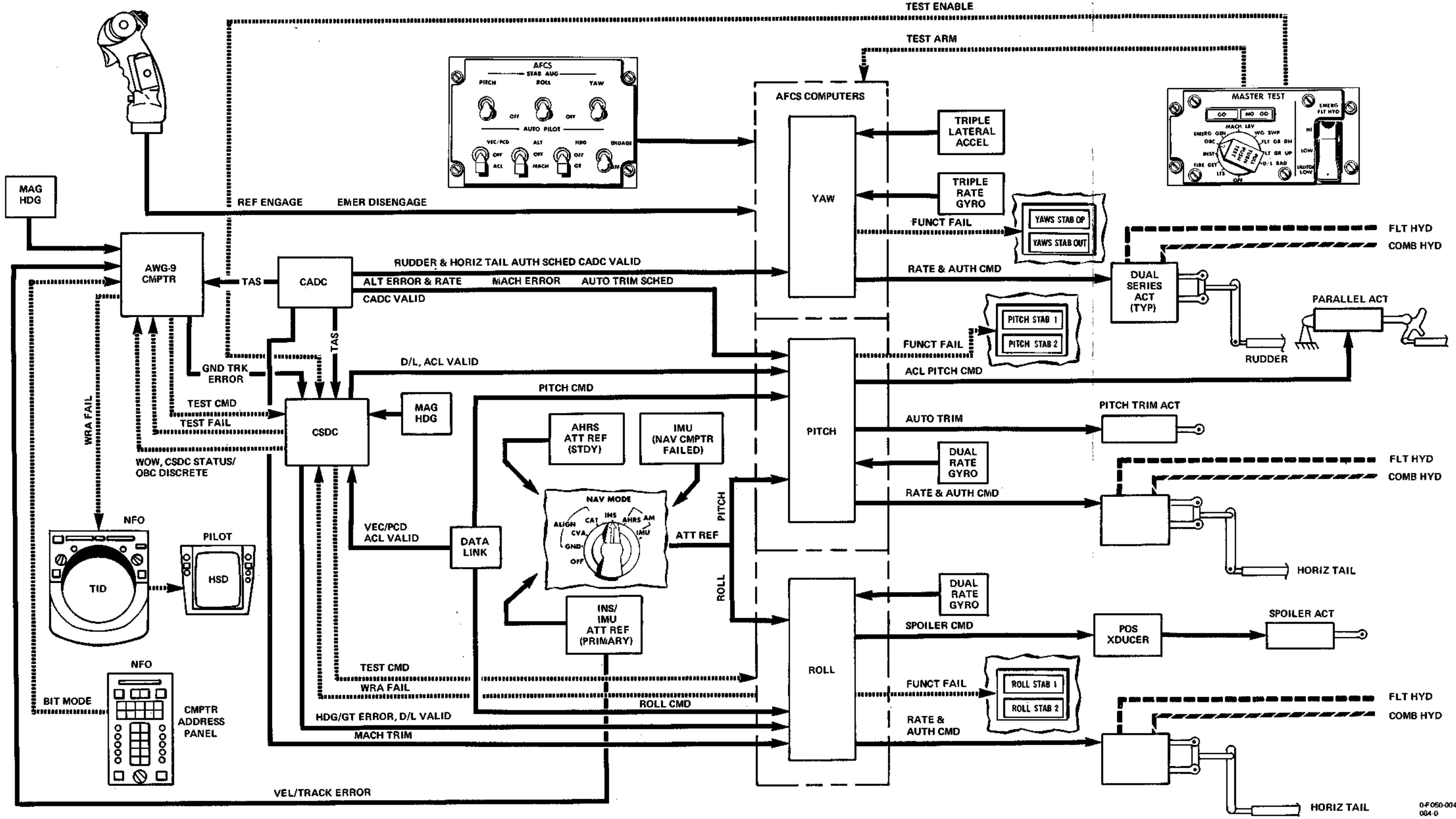


Figure 1-54.

AFCS CONTROLS

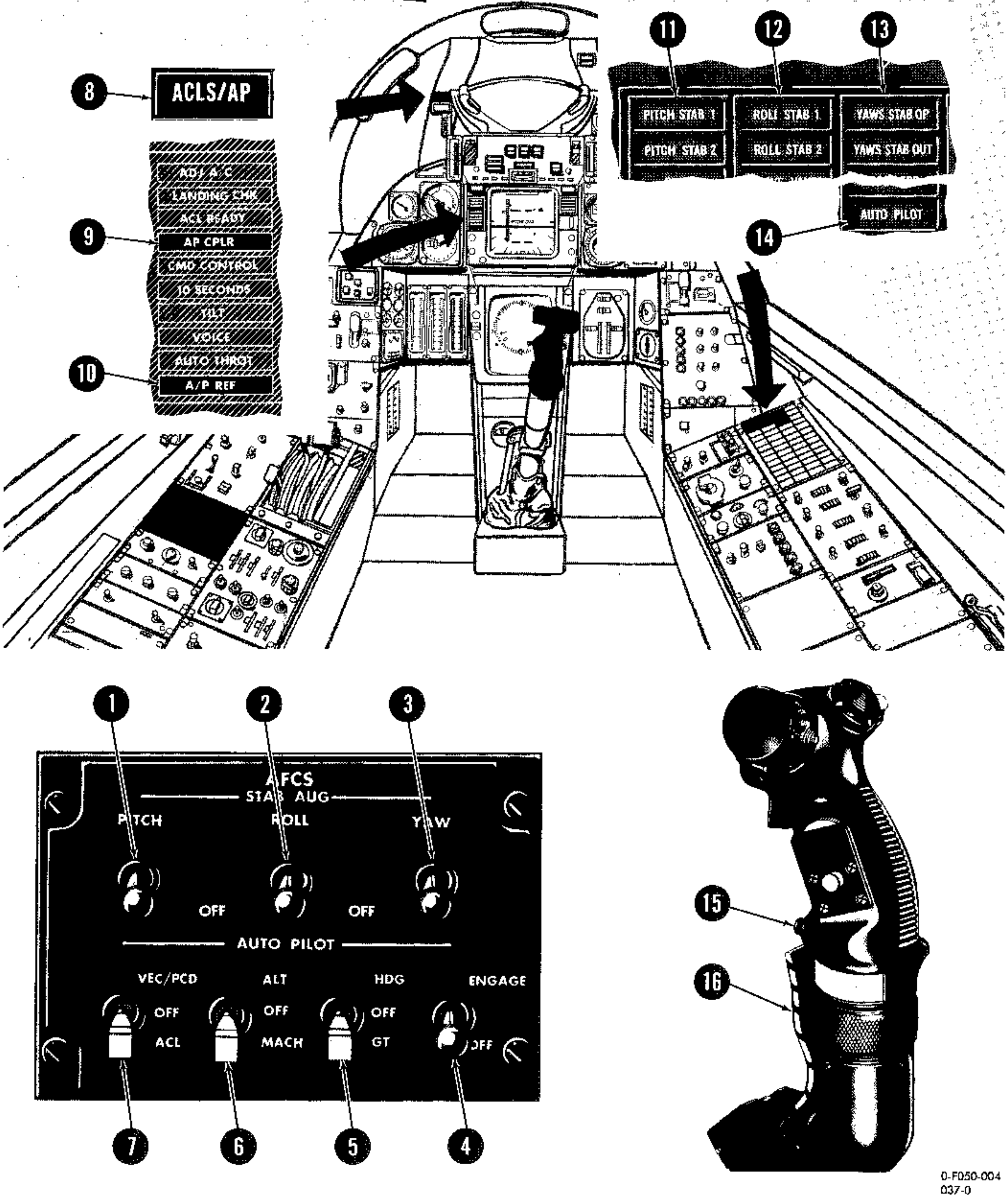


Figure 1-55.

NOMENCLATURE	FUNCTION
① PITCH SAS ENGAGE Switch	Engages dual channel pitch stability augmentation. Solenoid held in ON (up) position. Spring-loaded to OFF. Required to engage autopilot.
② ROLL SAS ENGAGE Switch	Engages dual channel roll stability augmentation. Solenoid held in ON (up) position. Spring-loaded to OFF. Required to engage autopilot.
③ YAW SAS ENGAGE Switch	Engages three channel yaw stability augmentation in the up position. Required to engage the autopilot.
④ AUTOPILOT ENGAGE Switch	ENGAGE - Engages autopilot. PITCH, ROLL and YAW must be engaged. No warm-up required. OFF - Disengages autopilot.
⑤ HDG-GT Switch	HDG - Autopilot will hold constant aircraft heading. Does not require A/P REF NWS switch. OFF - Disengages heading hold and ground track. GT - Selects autopilot ground tracking using data link commands. Engaged by A/P REF NWS switch.
⑥ MACH-ALT Switch	ALT - Autopilot will maintain barometric altitude. Engaged by A/P REF NWS switch. OFF - Disengages mach/altitude modes. MACH - Autopilot will maintain a constant Mach. Engaged by A/P REF NWS switch.

	NOMENCLATURE	FUNCTION
7	VEC/PCD - ACL Switch	VEC/PCD - Autopilot roll axis commands steer aircraft using data link signals for vectoring. If the PCD discrete is present both roll and pitch axis commands are used. Engaged by A/P REF NWS switch. OFF - Disengages VEC/PCD and ACL modes. ACL - Autopilot will accept data link signals for carrier landing using spoiler commands in roll and parallel servo signals in pitch. Only pitch commands are transmitted to stick movement. Engaged by A/P REF NWS switch.
8	ACLS/AP Light	Autopilot and all carrier landing system disengaged.
9	A/P CPLR Light	Indicates the autopilot is coupled to all weather carrier landing system.
10	A/P REF Light	Autopilot mode is selected but is not engaged.
11	PITCH STAB Light	Illuminated channel inoperative. Single channel 50% loss of authority.
12	ROLL STAB Light	Illuminated channel inoperative. Single channel 50% loss of authority.
13	YAW STAB Light	OP - One channel inoperative. Full authority available. OUT - Two channels inoperative. Yaw stabilization inoperative.
14	AUTO PILOT Light	Autopilot on reference failure.
15	A/P REF NWS Switch	Engages the ALT, MACH, HDG, GT, ACL or VEC/PCD mode selected. Autopilot must be engaged and compatible autopilot modes selected. Requires weight off wheels.

NOMENCLATURE	FUNCTION
16 AUTOPILOT EMERGENCY DISENGAGE PADDLE	Disengages all autopilot modes and releases all autopilot switchs. Momentary disengages (as long as switch is held) pitch and roll SAS servos; switches remain ON. Yaw SAS is not disengaged.

(U) Ground track steering computations are performed by the AWG-9 computer based on inputs from the ADC, IMU, and AHRS. The computer output, in the form of ground track error signals, is processed in the CSDC which generates steering commands to the autopilot roll axis. Bank angles are limited to ±30°. Selection of the heading reference (IMU or AHRS) is controlled by the NFO through the NAV MODE select switch located on the TID. The IMU will normally be the primary reference and the AHRS a back-up in the event of IMU failure. IMU or AHRS failure is indicated by illumination of their respective advisory lights on the pilot and NFO caution-advisory panels.

(U) Mach Hold - The autopilot may be used to maintain a desired Mach number by setting the ALT/MACH switch to MACH. When the A/P REF light illuminates depress the A/P REF NWS switch on the stick grip when the desired Mach number is attained. The mode is engaged when the A/P REF light goes out.

(U) Application of more than 2-1/4 pounds longitudinal stick force will result in mode disengagement and illumination of the A/P REF light. To re-engage the mode, depress the A/P REF NWS switch on the stick grip. Re-engagement will be indicated when the A/P REF light goes out. The autopilot maintains the desired Mach number by automatic control of pitch attitude.

(U) Altitude Hold - This mode is engaged by setting the ALT/MACH switch to ALT. When the A/P REF light illuminates depress the A/P REF NWS switch when at the desired altitude. This will engage the altitude hold mode and the A/P REF light will go out. Disengagement of altitude hold is accomplished by applying 10 pounds or more longitudinal stick force, or by placing the ALT/MACH switch in the OFF position. This will cause the A/P REF light to illuminate. The mode may be re-engaged by depressing the A/P REF NWS switch on the stick grip, when at the desired altitude, and observing that the A/P REF light goes out. Altitude hold should not be used for any maneuvers requiring large, rapid pitch trim changes due to limited servo authority and slow auto trim rate.

Data Link Vector - Precision Course Direction (PCD)

(U) This mode is engaged by placing the VEC/PCD - ACL switch in the VEC/PCD position, and depressing the A/P REF-NWS switch. Mode engagement is evidenced by the A/P REF light extinguishing. Disengagement of the mode is accomplished by application of stick forces of 1-1/4 pounds lateral or 2-1/4 pounds longitudinal, or disengagement by placing the VEC/PCD - ACL switch in the OFF position. If the switch is left in the VEC/PCD position the A/P REF light will illuminate and the mode may be re-engaged by depressing the A/P REF NWS switch.

(U) Determination of whether data link or precision course direction signals are present is made in the AFCS pitch and roll computers in response to inputs from the data link converter and CSDC. If the data link discrete is present, the autopilot roll axis will respond to data link heading commands and bank angle authority will be limited to ±30°.

(U) When the PCD discrete is present, the autopilot roll and pitch axes will respond to data link commands.

(U) Automatic Carrier Landing (ACL) - To engage the ACL mode place the VEC/PCD - ACL switch in the ACL position and depress the A/P REF NWS switch. Mode engagement is evidenced by the A/P REF light extinguishing. Disengagement of the mode will occur under the following conditions:

- Application of lateral or longitudinal stick forces in excess of 10 pounds - re-engage by depressing the A/P REF-NWS switch.

- Loss of data link discrete wave-off or TILT discrete from the carrier - ACL mode and autopilot will be disengaged, resulting in the autopilot engage switch releasing to OFF, and illumination of the ACLS/AP caution light on the windshield frame and illumination of AUTO PILOT caution light. The system may be re-engaged by pressing the MASTER RESET switch or by re-engaging the autopilot.

(U) ACL commands received by the data link system will control the aircraft through the autopilot by spoiler commands in roll and by the parallel servo in pitch. Pitch commands will be noticed by movement of the control stick.

Automatic Pitch Trim

(U) Automatic pitch trim is used in all autopilot modes to trim the aircraft upon disengagement of the autopilot functions. The pitch series servo position is monitored in all autopilot modes, except ACL, to drive the aircraft pitch trim motors, at one tenth manual trim rate, in response to changes in aircraft flight condition or center of gravity loads. The pilot's manual trim button on the stick is inactive during all autopilot operation. During ACL operation, the pitch parallel servo force is monitored to drive the aircraft trim motor, to prevent large stick forces from developing subsequent to ACL mode disengagement.

AFCS Emergency Disengage

(U) Operation of the emergency disengage paddle switch on the control stick, (figure 1-55), results in momentary disengagement of the pitch and roll SAS servos while the paddle switch is held depressed. The pitch and roll SAS engage switches will remain ON. Release of the paddle switch will result in re-engagement of the pitch and roll SAS servos if all monitors reflect normal operation. The YAW SAS channels are not effected by operation of the emergency disengage switch and will remain ON at all times unless the YAW SAS switch is placed in the OFF position.

AFCS BUILT IN TEST (BIT)

(U) The AFCS BIT provides the capability to perform complete AFCS tests without external test equipment. An independent test programmer and associated hardware is contained within each axis computer. These are two modes of BIT operation: operational readiness or preflight, and inflight continuous monitoring (PITCH and ROLL SAS only).

Operational Readiness or Pre-Taxi BIT

(U) The operational readiness or pre-taxi check is performed with weight on wheels for maintenance or preflight checks of the AFCS. It is initiated through the MASTER TEST switch on the pilot's MASTER TEST panel.

Pre-taxi Checks:

1. Throttles ---------------------- Less than MIL
2. Attitude reference -------------- IMU/AHRS
3. Data link modes ---------------- OFF
4. Anti-skid spoiler brake switch --- OFF
5. Autopilot switches --------------- OFF
6. Control surfaces ---------------- CLEAR
7. STAB AUG switches ------------- ON
8. Master test switch -------------- Rotate in raised position to OBC and DEPRESS to initiate test.

(U) A complete test sequence encompassing all functions specifically associated with the selected axis (axes) will commence upon test initiation and will cycle to OFF after completion. The PITCH STAB 1 and 2, ROLL STAB 1 and 2, YAW STAB OP, and YAW STAB OUT caution lights will illuminate and serve as an AFCS BIT running indication. All other AFCS caution lights will illuminate momentarily during BIT testing. Approximately 1 minute and 35 seconds is required for BIT checks of all three axes. The functions tested are:

- PITCH AXIS - Pitch damper, pitch autopilot, outboard spoiler control.
- ROLL AXIS - Roll damper, roll autopilot, inboard spoiler control, Mach trim control.
- YAW AXIS - Yaw damper, authority controls.

AFCS Inflight BIT

(U) AFCS inflight BIT provides automatic continuous monitoring, and test initiation for the AFCS pitch and roll axes only. Inflight BIT is not provided for the yaw axis because of this axis fail operational mechanization. If yaw axis failures occur the appropriate caution light will illuminate (figure 1-56), and flight envelope restrictions must be observed as applicable. Inflight testing in the pitch and roll axis is automatically initiated when a failure is detected. When a failure is detected the affected axis will disengage, both caution lights for the affected channel will illuminate, and BIT will automatically run for 9 to 14 seconds depending on the axis affected. Weapon replaceable assembly failure indications can be observed inflight by the NFO on the TID by selecting the BIT category on the computer address panel. Inflight fault isolation is limited to any of four WRA's, the pitch and roll computers, and pitch and roll rate gyros.

ALIGHTING GEAR

(U) In general the alighting gear consists of that hardware which permits the aircraft to operate from carrier or field-based facilities. The aircraft has tricycle-type landing gear with retractable air-oil shock struts and high pressure tires. The main landing gear have multiple-disk hydraulic brakes with an anti-skid system which controls maximum wheel deceleration without inducing a skid. An electro-hydraulic nosewheel steering system facilitates directional steerage during ground operations in addition to serving as a shimmy damper and centering mechanism. The dual wheel nose landing gear incorporates a launch bar and holdback system for nose tow catapult operation. An arresting hook system is provided for carrier and field-based arrested landings. Pilot normal control of the alighting gear is provided from the cockpit with sufficient back-up control provisions to enable a safe landing subsequent to sustaining failures in the primary motive and control systems. The alighting gear are operated by the combined hydraulic system and essential no. 2 dc electrical power in the normal mode with back-up motive control provided by pressurized nitrogen or mechanical means. Control and indication for normal and emergency operation of the alighting gear as provided in the pilot's cockpit are presented in figure 1-56.

LANDING GEAR

(U) The aircraft is equipped with fully retractable tricycle landing gear which are operated by combined hydraulic pressure in the normal mode of operation and use a stored source of pressurized nitrogen for emergency extension. The landing gear retract forward so that airloads and gravity assist on emergency extension. Air-oil shock struts with oil metering pins are used to reduce landing loads transmitted to the airframe, and the struts are fully extended with the gear in the wells. All landing gear doors remain open with the gear extended and are sequenced to operate as a function of gear control module or gear position commands. The lower strut pistons incorporate false axles to assist in changing wheel assemblies. Pilot operation of the landing gear handle mechanically positions the gear control module for normal operation and by pulling the handle mechanically selects emergency extension of the gear using the pneumatic back-up source. Both modes of gear operation can be accomplished without electrical power except for the gear position indication which requires essential No. 2 DC power. Gear downlock actuators incorporate internal mechanical finger locks which maintain the downlock inserted position in the absence of hydraulic pressure. Landing gear tires are qualified to a limit ground speed of 190 KIAS. Design limit

ALIGHTING GEAR CONTROLS AND INDICATORS

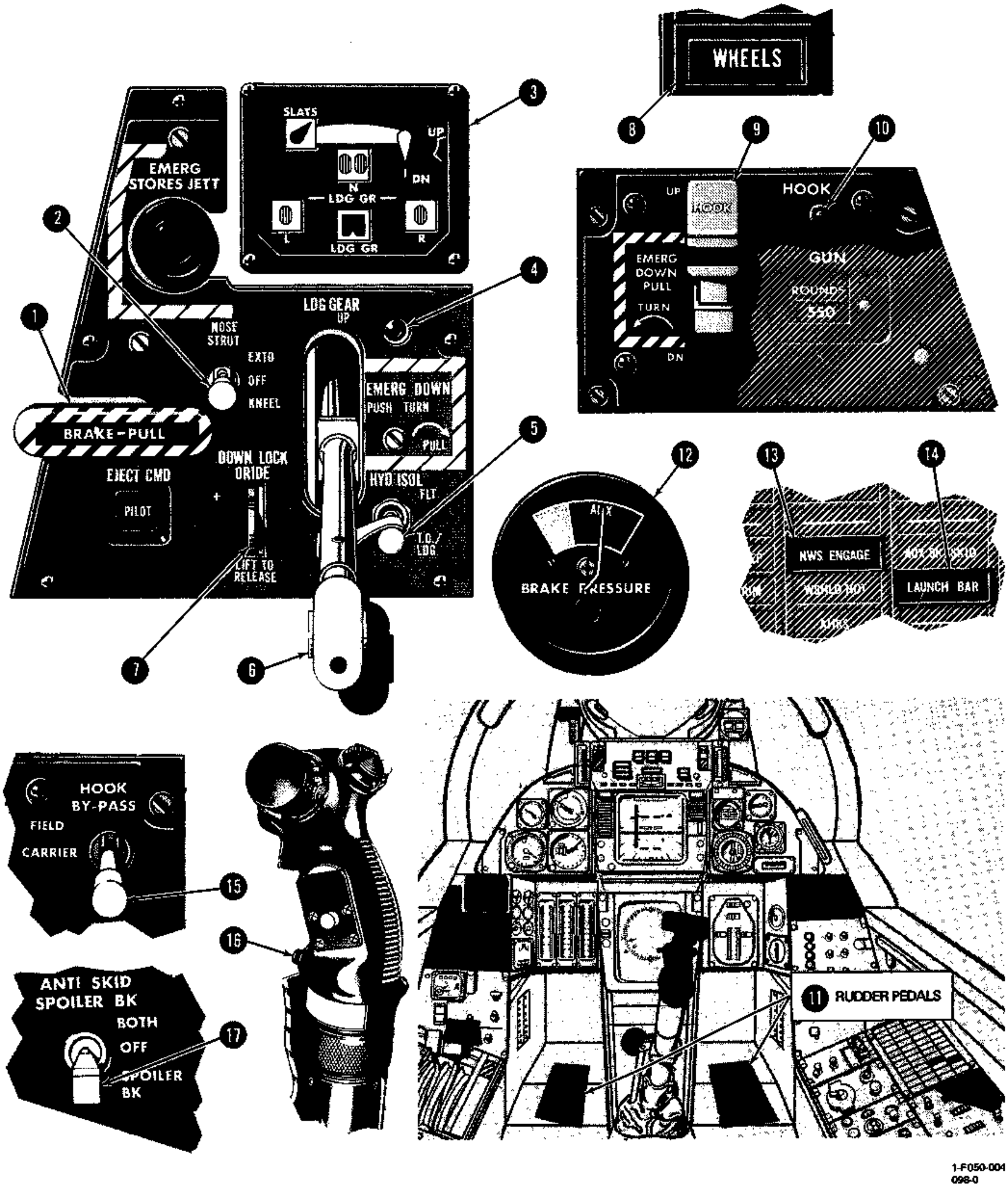


Figure 1-56

NOMENCLATURE	FUNCTION	
(1) PARKING BRAKE Handle	FWD -	Parking brake released. Modulated braking action available with brake pedal depression.
	AFT -	Parking brake set. No modulation of control, locks both main wheel brakes.
(2) NOSE STRUT Kneel switch	EXTD -	Ports combined hydraulic pressure into nose strut transfer chamber causing strut to extend. Combined hydraulic system must be pressurized before switch is activated on ground power. Launch bar is lifted into the uplock position by torque arms as strut extends 14 inches.
	OFF -	Spring-loaded return position of switch.
	KNEEL -	Nose strut transfer control valve releases pressure in the shock strut, which strokes 14 inches. Combined hydraulic system must be pressurized before switch is active on ground power. Launch bar uplock can be released to allow bar to lower to deck by turning nose wheel ± 10°.
(3) LANDING GEAR/FLAP INDICATOR	$\boxed{\text{UP}}$ -	Landing gear down and locked (except main landing gear side-brace actuator).
	 -	Unsafe gear or power off indication.
	 -	Landing gear retracted and doors closed.

	NOMENCLATURE	FUNCTION	
4	LANDING GEAR TRANSITION LIGHT	Illuminates whenever gear and door positions (including main landing gear sidebrace actuators) do not correspond to handle position. Light out when gear and doors are locked in position selected by handle.	
5	HYD ISOL SWITCH	FLT -	Combined system hydraulic pressure is shut-off to the landing gear, nose wheel steering, wheel brakes, and tail hook retraction.
		T. O. /LDG -	Combined hydraulic pressure is available to all components barring automatic isolation.
6	LANDING GEAR HANDLE	NORMAL -	Up-down overcenter action provides normal retraction/ extension by the combined hydraulic system.
		EMERGENCY-	Down-push-turn-pull action provides emergency extension of all gear by a compressed nitrogen change.
7	DOWN LOCK ORIDE LEVER	DOWN -	Weight-on-wheels indication, prevents gear handle being retracted without pilot override (raising lever).
		UP -	Weigh-off-wheels indication, does not inhibit pilot raising gear handle. Automatic operation by electrical solenoid.
8	WHEELS WARNING LIGHT	Illuminates (flashes) with flaps $>10°$ deflection and either or both throttles $< 87\%$ RPM without all landing gear down and locked. Approach lights and indexer will not illuminate unless all gear are down and locked.	
9	ARRESTING HOOK HANDLE	UP -	Electrically energizes hydraulic retract actuator to raise hook into uplock.

	NOMENCLATURE	FUNCTION
		DN - Electrically releases uplock actuator and allows hook to extend by dashpot pressure and gravity.
		EMERG - (Pull-twist) mechanically releases uplock actuator and allows hook to extend by gravity and dashpot pressure.
(10)	HOOK TRANSITION LIGHT	Illuminates whenever arresting hook position does not correspond with handle position. Light will not go out in down position until hook is in full trail angle.
(11)	RUDDER/BRAKE PEDALS	FWD/AFT - Provides ±90° nose wheel steering on deck with system engaged.
		DEPRESSION-Provides normal, anti-skid, and auxiliary brake modes.
(12)	BRAKE PRESSURE GAGE	Provides pilot indication of brake accumulator pressure remaining which is indicative of brake cycles remaining.
(13)	NWS ENGAGE ADVISORY LIGHT	Illuminates when nose wheel steering engaged and will respond as a function of rudder pedal displacement. Nose wheel steering automatically centers with hook down.
(14)	LAUNCH BAR ADVISORY LIGHT	Illuminates when launch bar is not locked up. Light out when bar retracted or locked up with nose gear down.
(15)	HOOK BY-PASS SWITCH	FIELD - Used for non-arrested landings. Bypasses the flashing feature of the approach lights and indexer when landing gear is down and hook retracted.
		CARRIER - Used for arrested landings. Approach lights and indexer flash when landing gear is down and the hook retracted.

NOMENCLATURE	FUNCTION	
(16) NOSE WHEEL STEERING BUTTON	Push to engage/disengage nose wheel steering. Requires weight-on-wheels.	
(17) ANTI SKID SPOILER BK SWITCH	BOTH -	Anti-skid and spoiler brakes operate with weight-on-wheels.
	OFF -	Anti-skid and spoiler brake inoperative with weight-on-wheels.
	SPOILER BK-	Spoiler brakes operate with weight-on-wheels and both throttles IDLE. Anti-skid is deactivated.

landing sink speed for the aircraft is 24.7 feet per second, whereas the nominal landing sink speed is 11 feet per second. Normal and emergency controls and displays associated with operation of the landing gear are presented in figure 1-56.

Main Landing Gear

(U) The main landing gear is housed in the glove-sponson area on the outboard side of the engine nacelles. The resultant wide tread width (16.4 feet) between main wheel tires provides for good lateral stability for ground handling operations. The air-oil shock struts are pinned to supporting trunnions on the landing gear bulkhead that serves as the engine nacelle forward firewall. Each shock strut consists of an upper outer cylinder and a lower internal piston which has a maximum stroke of 25 inches. A hard step (31,000 pounds required for further compression) in the strut air curve provides a consistent 4 inch stroke remaining in the ground static condition. A side brace link pivots from the inboard side of the strut outer cylinder to engage in a nacelle fitting and thus provide additional side load support for ground operations. The side brace link is mechanically extended to engage the nacelle fitting by the unfolding action of the drag brace. A universal fitting at the forward end of the drag brace transmits gear drag loads to the wing box structure, and as the drag brace folds during gear retraction, it causes the wheel assembly to rotate (90 degrees) and lay flat in the glove well forward of the wing box. The path of the wheel assembly is controlled by the drag brace as it folds (jack-knifes upwards) during gear retraction and unfolds during extension. The fully extended shock strut and jack-knifed drag brace retracts forward into the sponson well. Inboard, outboard, and aft main gear hinged doors, which remain open with the gear extended, are individually actuated closed in sequence to provide fairing for the retracted gear. An uplock hook on the shock strut engages a roller in the wheel well to hold the gear in the retracted position. The main landing gear actuator, located on the inboard side of the shock strut, retracts and extends the gear assembly. The gear downlock actuator, mounted at the drag brace knee pin, extends to prevent unlocking (jack-knife) of the drag brace. Hydraulic pressure must be supplied to the downlock actuator in order to retract it against the spring action of the integral locking mechanism. A paint stripe across the drag brace knee pin provides an external visual indication of the drag brace locked condition. A ground lock device clamps onto the downlock actuator rod for safetying the main gear.

(U) Maximum strut extension and wheel alignment are controlled by torque arms which incorporate cam operated micro switches to detect a weight-on-wheels condition (greater than 5 inches of strut compression). The axle on which the wheel and brake assembly is mounted protrudes outboard at the lower end of the strut piston. A false axle is provided to facilitate changing the wheel assembly. The wheel, bearings and false axle are an integral assembly which is keyed to the concentric true axle to prevent rotation. Correct installation of the wheel retaining nut is not possible without proper assembly of the false axle. The main wheel incorporates a multiple disk hydraulic brake assembly and wheel rotation sensor for anti-skid system operation. The single split-type wheel assembly incorporates a tire change counter, thermal fuse blow plugs, and a pressure relief device to prevent over inflation of the tire. Field/carrier inflation pressures for the tubeless tire (37 x 11.50-16, 28 ply rating, rib tread) are 245/350 psi of nitrogen. Additional hardware on the main landing gear include wheel jack pads, tie down rings, hydraulic swivels and lines.

Nose Landing Gear

(U) The nose landing gear is housed in the forward fuselage beneath the pilot's cockpit. The air-oil shock strut outer cylinder is supported by trunnions mounted between two longeron beams at the aft end of the nose wheel well and by an aft acting drag brace. The shock strut consists of an outer cylinder and a lower internal piston which has a maximum stroke of 18 inches. During normal ground operations the strut is fully extended but pilot control is provided to kneel the strut (4 inches stroke remaining) for catapult operations. During retraction the fully extended nose strut is rotated forward by the double acting retract actuator into the well and enclosed by two forward and two aft doors. The forward doors are operated by a separate actuator which also engages the gear uplock whereas the two aft doors are mechanically linked to the shock strut. The drag brace fairing seals the respective well with the gear retracted. An uplock hook actuator, that is activated with strut contact of the timer valve on the overhead of the wheel well, engages a roller on the lower piston to hold the gear and doors in the retracted position. During gear extension the telescoping drag brace compresses so that a double acting downlock actuator inserts a piston through a hole in the aft mount of the drag brace (inner and outer barrel) to form a rigid member for transmission of loads to the airframe. A mechanical detent is provided for locking the downlock actuator in the extend (inserted) position without hydraulic power. Since the drag brace dimension does not provide an indication of gear locked status and the downlock mechanism is concealed in the fuselage, there is no foolproof visual means of ascertaining that the nose gear is locked down and the pilot must rely on his indicator. Insertion of a ground lock pip-pin into the downlock actuator does provide a valid indication of gear locked status.

(U) Maximum strut extension and wheel steering angle are controlled by torque arms interconnecting the steering collar and the lower piston. Dual axles provide a mount for dual nose wheel assemblies with false axles. The single split-type wheel assembly incorporates a tire change counter and a pressure relief device to prevent over inflation of the tire. Field/carrier inflation pressures for the tubeless tires (22 x 6.6-10, 20 ply rating, rib tread) are 105/350 psi of nitrogen. Additional hardware on the nose landing gear include the wheel jack pad, tie down rings, launch bar, holdback fitting, approach lights, nose wheel steering actuator, taxi light, and strut transfer cylinder. The wheel axles incorporate recessed holes for attachment of a universal tow bar with maximum steering angle of ±120 degrees.

Normal Operation

(U) The landing gear handle is mechanically connected to the gear control module which directs combined hydraulic fluid into the gear up and down lines and provides a path for return flow. In the down position the handle mechanically sets the hydraulic isolation switch to provide hydraulic pressure for gear operation. The handle is electromechanically locked in the down position with weight-on-wheels to prevent inadvertent gear retraction. Pilot override of the solenoid operated handle lock can be effected by lifting the downlock latch next to the gear handle. Vertical movement of the gear handle causes a corresponding up/down selection of the landing gear with the combined hydraulic system pressurized. Three flip-flop indicators provide a position display for each of the landing gear and a gear transition light illuminates on the control panel any time the gear position and handle are not corresponding. In addition, a WHEELS warning light will alert the pilot if the landing gear are not down with flaps deflected greater than 10 degrees and both throttles less than 85%RPM. The normal transition time for operation of the landing gear is 9 seconds.

(U) Placement of the landing gear handle to the UP position actuates the gear control module which ports hydraulic pressure to the unlock side of the downlock actuators, gear retract actuators and in sequence to the door and uplock actuators. The gear shock strut and door uplocks are hydraulically operated into a mechanical overcenter position so that combined system hydraulic pressure may be subsequentially manually isolated from these components without danger of an inadvertent unlock condition at high speed or maneuvering conditions. An UP indication is displayed on the gear position indicators when the gear are in the uplock and all doors closed. Placement of the gear handle to the DN position actuates the gear control module to port hydraulic pressure to the unlock side of the door uplocks, subsequently open the gear doors, release the strut uplocks, pressurize the gear extend actuator (nose gear only) and downlock actuators. A gear down symbol (wheel) is displayed on the gear position indicators when the gear downlocks are inserted; however, the gear transition light will not extinguish until the downlocks are inserted and proximity switches on the nacelle side brace fitting verify that the main gear side brace links are engaged. With the main gear downlock inserted but the side brace link not engaged landing sink speed is restricted to 8 feet per second.

EMERGENCY GEAR EXTENSION

(U) Rotation (90° clockwise) and aft extension of the gear handle manually ports pressurized nitrogen from the ground charged emergency gear bottle for a single shot extension of the landing gear by an alternate motive source. The emergency landing gear nitrogen bottle (nonshatterable design, 3,000 psi at 70°F) and filler port (with relief valve to prevent overpressurization) are located in the nose wheel well. Although emergency extension can be initiated with the landing gear control handle in any position, it is preferable that the handle be selected down before actuating the emergency extension system. By rotating to unlock and pulling the gear control handle aft an interconnecting cable trips the emergency gear extension air release valve in the nose wheel well. By this action a secondary spool within the gear module is shuttled by pneumatic pressure to provide a return passage for the gear up and down hydraulic lines, thus disabling nose wheel steering and normal/anti-skid wheel brake modes.

(U) Pneumatic pressure is directed by separate lines to power open the gear door actuators in sequence, release the gear uplock actuators, pressurize the nose gear retract actuator to extend the gear (main gear free fall), and pressurize the downlock actuators. A normal gear down indication is achieved upon emergency gear extension. Subsequent to emergency extension, gear retraction cannot take place until the dump valve is mechanically reset and the handle returned to normal stowed position on the operating shaft. A manual reset plunger enables ground maintenance personnel to reset the control module secondary spool to bleed nitrogen gas from the system after emergency extension. The landing gear must be cycled with the aircraft on jacks to achieve a satisfactory bleed of air from the hydraulic actuators following a gear dump.

NOSE WHEEL STEERING

(U) The electro-hydraulic nose wheel steering system provides for on-deck aircraft directional control, nose wheel shimmy damping and nose wheel centering. The power unit is located on the lower portion of the nose landing gear strut outer cylinder and through a ring gear controls the directional alignment and damping of the lower piston assembly. Steering power is generated by a rotary vane actuator, and an internal mechanical feedback is used in comparison with electrical steering commands to provide positive angular control. Combined hydraulic system pressure is the motive power used for steering and centering as controlled by electrical signals energized from essential No. 2 power. Hydraulic pressure is derived from the gear down line such that steering control is disabled subsequent to emergency extension of the landing gear. Although the hydraulic pressure is also controlled through an isolation valve, hydraulic isolation is removed with weight-on-wheels. The following modes are provided by the steering unit.

Directional Control Mode

(U) Nose wheel steering control during ground operations is energized by the push-to-engage/disengage button on the lower forward side of the pilot's stick grip. The system cannot be engaged or armed in flight unless an interlock malfunction has occurred. Once engaged, the pilot can release the button and the system will remain engaged until an interlock (weight-on-wheels) is broken, electrical power is interrupted, or the control button is depressed. Engagement of the nose wheel steering mode for directional control is indicated by illumination of the NWS ENGAGE advisory light. With the system engaged, the nose gear steering angle is controlled by rudder pedal position. Although the steering signals are generated by a transducer in the directional flight control system feel assembly, centering is unaffected by directional trim conditions. The transducer senses only rudder pedal position and transmits a proportional voltage to control porting of all hydraulic pressure in the steering unit to turn the nose wheel. Maximum power steerage authority is 90 degrees either side of neutral and the nose gear can swivel a maximum of 120 degrees about the centered position. Depending on the weight on the nose gear (function of gross weight and CG), steering torque is normally sufficient to turn the nose wheel ±5 degrees with the aircraft in a static position; however, only a slight forward movement will provide the pilot with full power steering authority. In a full pedal deflection turn, using nose wheel steering, the aircraft pivots about a point located between the main gear such that the inboard main wheel rolls backwards. Under this condition application of either main wheel brake will only serve to increase the radius of turn. Because of the outboard location of the engines the application of thrust in tight turns should be made on the outboard engine to efficiently complement the turning moment of the nose gear.

Centering Mode

(U) The nose wheel is automatically powered to center during arrestment to prevent nose wheel castering on rollback and during gear retraction to center the nose gear before it enters the wheel well. With the hook not stowed the nose wheel steering unit is powered to center and the steering unit will not respond to rudder pedal displacement commands unless the pilot engages the nose wheel steering mode. During gear retraction, hydraulic pressure bypasses the steering unit shut-off valve to power center the nosewheel independent of rudder pedal displacement.

Shimmy Damping Mode

(U) Shimmy damping is provided as the nose wheel turns by directing fluid through orifices to either side of the rotary vane steering actuator. The closed-loop shimmy damping action is accomplished in the steering unit energized as well as deenergized state.

WHEEL BRAKE SYSTEM

(U) The wheel brake system provides power boost hydraulic control of the multiple disk-type main wheel brakes using pressurized fluid in the landing gear down line from the combined hydraulic system. Individual or collective wheel brake control can be modulated by depression of the rudder toe pedals or collective, unmodulated brake control is available with the parking brake. Wheel brake application is controlled from the front cockpit during power on/off ground operations. An anti-skid system is provided to operate electro-hydraulically in conjunction with the normal wheel braking mode. A schematic of the wheel brake system is provided in figure 1-57.

(U) Brake pedal and parking brake control motions in the pilot's cockpit are mechanically transmitted to the power brake module which is located above the left inlet ramps together with the wheel brake accumulators and the brake valve. Separate hydraulic lines transmit normal and emergency fluid pressure from the power brake module to the left and right wheel brake assemblies. At each brake assembly the normal and emergency lines input to the brake shuttle valve which has a single output to the five interconnecting pistons located radially around the brake housing. The shuttle valve is detented so as to provide for automatic switchover of the controlling input as a function of normal or emergency line fluid pressure. Fluid pressure at the shuttle valve is applied to the pistons which move the retractor plate compressing the stator and rotor disks for brake application. Retractor springs relieve disk compression pressure upon the reduction of brake fluid pressure with the porting of excess fluid in the power brake module to the combined hydraulic system return lines. Two wear indicator pins on the brake piston housing measure lining wear for preflight inspection. Normally these pins extend approximately 1/4 of an inch from the inboard brake surface. Three thermal relief plugs are mounted in each main wheel assembly to relieve tire pressure and thus avert a blowout due to hot brakes if the local temperature exceeds 400°F.

Normal Braking

(U) In the normal mode of operation wheel brake application is modulated by brake pedal depression using pressurized fluid from the combined hydraulic system. Depression of a brake pedal causes movement of the corresponding metering valve in the power brake module which directs an amount of pressurized fluid proportional to brake pedal depression to the brake valve. Fluid from the brake valve is directed back through the power brake module and through the normal brake line to the brake assembly. The fluid passes through a shuttle valve and flows to the wheel pistons which extend to exert a force on the brake disks, applying the brakes. Upon brake release, internal springs overcome fluid pressure to release the compression force on the brake disks. In the normal mode of operation the brake cycles remaining gage indication should continue to indicate a full charge on the brake accumulators since this fluid energy is maintained in reserve.

WHEEL BRAKE SYSTEM

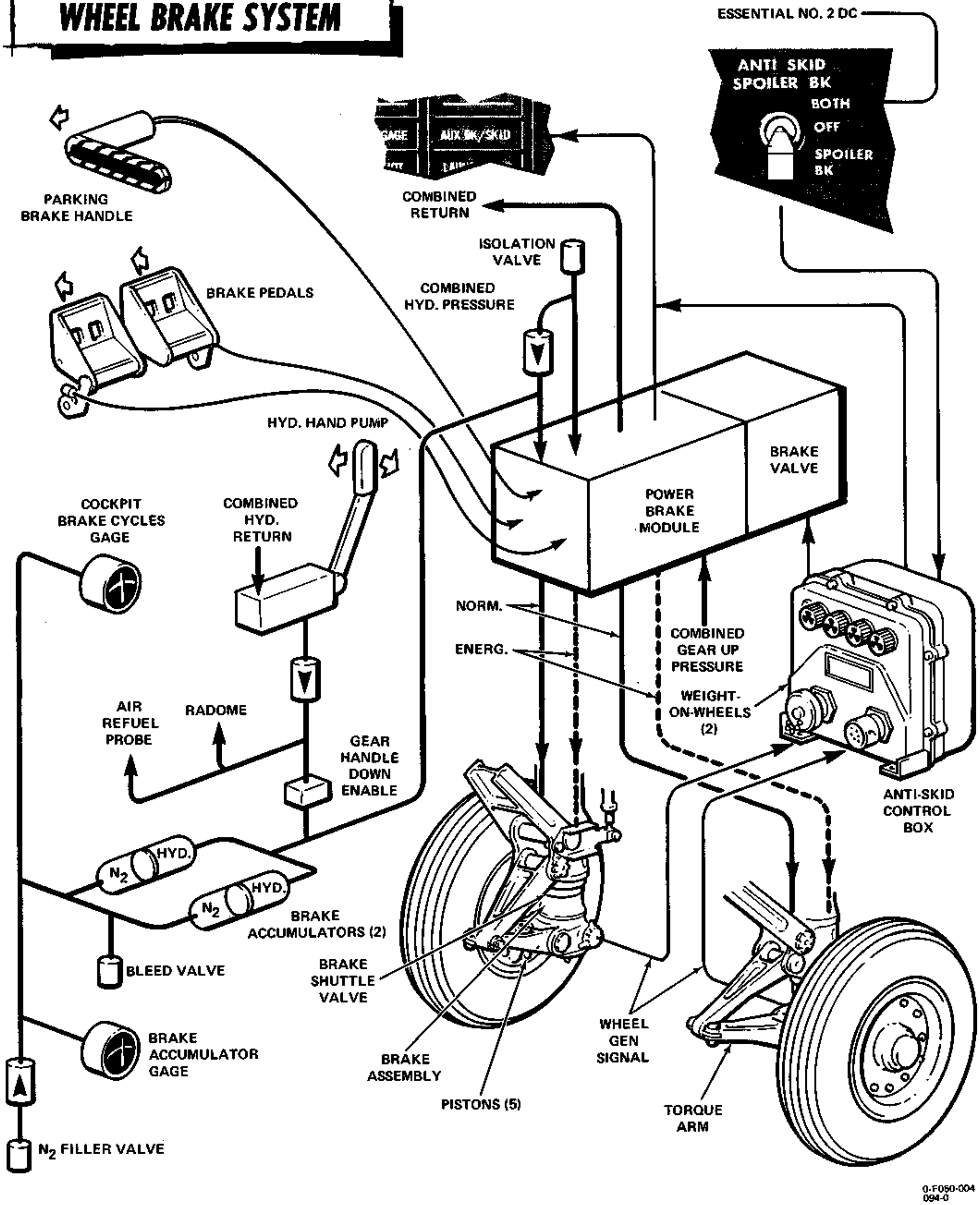


Figure 1-57.

Anti Skid

(U) The anti skid system operates electro-hydraulically in conjunction with the normal mode of wheel brake operation to deliver maximum metered fluid pressure to the wheel brakes upon pilot command without causing a skid. Essential No. 2 power for anti skid operation is supplied through the pilot's right circuit breaker panel and controlled by the ANTI SKID/ SPOILER BK switch on the pilot's left vertical console panel. Individual main wheel rotational velocity is sensed by direct current generators mounted in the wheel hubs which transmit a voltage proportional to wheel speed to the skid control box. The skid control box which is located on the forward bulkhead of the nose wheel well receives and responds to voltages from the main wheel generators. The control box detects changes in wheel deceleration and responds to energize a coil in the brake valve which reduces the supply of brake pressure to both wheel brakes in proportion to the signal from the skid control box. Thus the energized state of the coil in the brake valve may reduce fluid pressure in the normal brake lines to both wheel brakes to prevent a skid. When the wheel stops skidding, the control box provides a signal to the brake valve coil to direct maximum metered pressure to the wheel brakes. With the anti skid system armed in flight, the touchdown circuit in the control box causes the brake valve to direct normal brake pressure to return until the weight is on both main gears. Both main landing gear struts must be partially compressed (weight-on-wheel) before the brakes can be applied in the anti skid mode so that the system prevents braking before touchdown regardless of brake pedal application. With weight-on-wheels, a ground speed of greater than 15 knots must be maintained for the anti skid system to be operative.

(U) When deenergized, the anti skid system has no effect on normal brake operation and no time delay is required for warm-up. The anti skid system is inoperative when the wheel brakes are in the auxiliary or parking modes of operation since the emergency brake lines bypass the brake valve. If an electrical failure occurs in the anti skid system or if hydraulic pressure is removed from either brake for greater than 1.2 seconds, the system automatically becomes inoperative and illuminates the AUX BK/SKID advisory light with the ANTI SKID/SPOILER BK switch in the BOTH position. Failure of the anti skid system will not cause illumination of the AUX BK/SKID advisory light with the control switch in the OFF or SPOILER BK positions. A self-test of the anti skid system can be initiated on the face of the control box with the system energized, parking brake handle released and the aircraft in a ground static condition. Approximately 10 seconds is required for the self-test which checks to a high extent the operational status of the control box, brake valve and wheel sensors and displays the results by the illuminated status of the AUX BK/SKID advisory light or more specifically by the BIT flags on the face of the control box.

Auxiliary Brake

(U) Transfer to the auxiliary mode of brake operation is automatic without the requirement for pilot action upon the loss (less than 2000 psi) of combined hydraulic system pressure. Two brake accumulators provide pressure for auxiliary and parking modes of operation when combined hydraulic system pressure is not available. The floating piston accumulators which deliver 3000 psi when fully charged by the combined hydraulic system or hydraulic hand pump (with gear handle down only) are located above the left inlet ramps in close proximity to the power brake module. When the combined hydraulic system pressure decreases below 2,000 psi, priority valve springs in the power brake module shift the metering system to the emergency mode. With accumulator pressure greater than 1300 psi the auxiliary fluid passes through the metering valves in response to brake pedal depression to discharge the fluid directly into emergency brake lines bypassing the brake valve. Thus accumulator fluid pressure is directed through separate lines to the emergency side of the shuttle valves on the main landing gear with the wheel brake cylinders operating as in the normal mode. The anti skid mode of operation is automatically deactivated in the auxiliary mode. A detented plunger in the wheel brake assembly shuttle valve is shifted by the pressure in the emergency hydraulic line to direct such fluid to the brake pistons and at the same time block off the normal line port at the shuttle valve. Approximately 8 full dual brake pedal applications are available in the auxiliary mode when starting with fully charged accumulators. A full charge in the accumulators is indicated on the front cockpit mounted (center pedestal) brake cycles remaining gage by full clockwise displacement of the needle at the top of the green (AUX) band. Likewise, the corresponding reading on the nose wheel well mounted brake pressure gage is 3000 psi. Approximately 10 strokes of the hydraulic handpump are required to recharge the brake accumulator for each full cycle application of the wheel brakes. Full capacity operation of the brake accumulators in the emergency modes of operation is predicated on the system being serviced with a proper nitrogen precharge (900 psi without combined hydraulic system pressurized). As the brakes are cycled in the auxiliary mode with an attendant reduction in accumulator pressure the brake cycles remaining needle moves to the left in the green band to the red emergency (EMER) band. At approximately 1,300 psi brake accumulator pressure remaining (red band) the braking action by the brake pedals is no longer available because a priority valve in the power brake module blocks flow to the metering valves. In the event that the priority valve in the power brake module does not return to the normal position with combined hydraulic pressure greater than 2000 psi, the AUX BK/SKID advisory light on the pilot's caution-advisory panel will illuminate. The wheel brake accumulators can only be recharged by the hydraulic handpump with the landing gear handle down. Pilot manual isolation or system automatic isolation of the combined hydraulic system cuts off the supply of combined hydraulic pressure to the power brake module so that depression of the brake pedals will cause depletion of the brake accumulator charge.

Parking Brake

(U) The parking brake mode provides a means for pilot relief of braking requirements to maintain a ground static position during normal operations or for collective locking of the wheel brakes during emergency conditions. Aft movement of the parking brake handle provides for unmodulated porting of accumulator fluid pressure to both wheel brake assemblies with the landing gear extended. Pressurized fluid from the accumulator is directed to a single rotary valve in the power brake module by-passing the normal/aux mode metering valves because of the 1,300 psi priority valve. Pulling the parking brake handle rotates the rotary valve to direct accumulator fluid pressure through the emergency lines (bypassing the brake valve) to the shuttle valve at the wheel brake assembly. In the parking brake mode the rudder pedals have no effect on wheel brake operation and with greater than 2,000 psi combined hydraulic system pressure at the power brake module the AUX BK/SKID advisory light will be illuminated. Pushing the parking brake handle forward relieves wheel brake pressure and the power brake module reverts to the normal/auxiliary braking mode. When auxiliary mode braking action is no longer available by depression of the rudder pedals sufficient accumulator fluid pressure remains for a minimum of one parking brake application. In the absence of a pressurized combined hydraulic system the wheel brake accumulators can only be recharged by the pilot's hydraulic handpump with the landing gear handle in the down position. Upon post flight shutdown after field braking operations the parking brake should not be set to avoid fusing the brake disks in the event of hot brakes.

Wheel Anti-Rotation

(U) During the initial phase of the landing gear retraction cycle pressurized fluid from the gear up lines is directed to the power brake module to displace the normal/auxiliary metering valves to achieve the same result as rudder pedal depression. The purpose of this automatic feature is to stop main wheel rotation before the wheels enter the wells. No such feature is provided for the nose wheels.

Wheel Brake Performance

(U) The capacity of the wheel brake assemblies is sufficient to restrain the aircraft in a static condition on a dry surface with Zone No. 1 afterburner (nozzle position 2) set on both engines. During rollout conditions the maximum recommended speeds for wheel brake application in the normal and anti skid modes is presented in figure 1-58. The minimum hydroplaning speed for the main tires on a wet runway is approximately 140 knots.

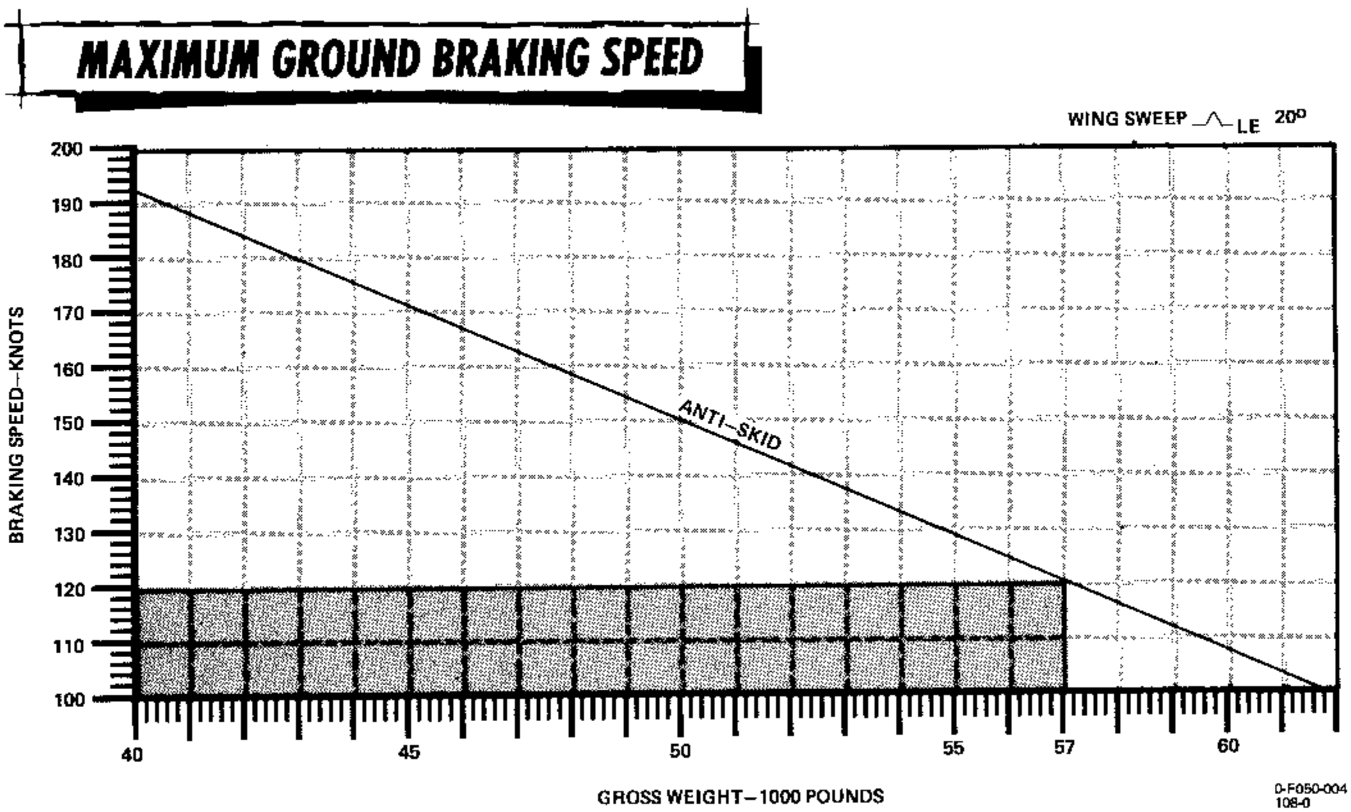


Figure 1-58.

NOSE TOW CATAPULT SYSTEM

(U) Catapult interface components mounted to the nose landing gear shock strut piston provide the aircraft with a nose tow catapult capability. A launch bar attached to the forward face of the nose gear steering collar guides the aircraft onto the catapult track and serves as the tow link which engages the catapult shuttle. A holdback fitting at the apex of the torque arms secures the trail bar for holdback restraint incident to launch. In addition to these components, the nose landing gear shock strut stroke and air curve are conditioned for catapulting by a transfer cylinder and control valve. The nose strut uses the stored energy catapult principal to impart a positive pitch rotational moment to the airplane at shuttle release thus providing for a hands-off launch fly-away technique.

Nose Strut Transfer Cylinder

A transfer control valve on top of the shock strut causes the strut piston to stroke 14 inches and coincidentally change the strut air curve characteristics for catapult launch. Pilot control of the transfer control valve is provided by the NOSE STRUT switch on the landing gear control panel. The three position (EXTD-OFF-KNEEL) toggle switch is spring loaded to return to the detented center OFF position. Control interlocks for the switch differ between aircraft power and ground electrical power to eliminate a potential ground maintenance hazard. The position of the valve remains in the last commanded position independent of electrical or hydraulic power interruptions. In both cases the transfer control valve source of electrical power is the essential No. 2 bus and combined hydraulic system fluid is used as the transfer medium. With ground electrical power on the aircraft, the combined hydraulic system must be pressurized (>500 psi) before the control switch can command a position change of the transfer control valve. The control switch need only be held to an extreme position momentarily to effect a change in transfer control valve position.

(U) Selection of KNEEL releases hydraulic fluid from the shock strut transfer cylinder to the combined hydraulic system return line thus causing the weight of the aircraft to stroke (compress) the shock strut 14 inches. Stroking of the nose strut causes the aircraft to rotate about the main wheels. The aircraft may be taxied or towed in the strut kneeled position except for the nuisance trip of the launch bar at greater than 10 degrees steering angle; this is the position used for taxiing onto the catapult and enhances accessibility to the forward fuselage compartments during ground maintenance. The kneeling action also changes the strut air curve (figure 1-59) to place the nose gear on a hard step which requires a static load of nominally 30,000 pounds to commence further stroking (compression). Since the nose strut is bottomed during the catapult launch stroke, the energy stored in the last 4 inches of strut piston stroke is released upon shuttle release at the end of the catapult stroke to impart a nose-up pitching moment to rotate the aircraft to the fly-away attitude without any control programming required by the pilot. All of the stored energy is expended before the main wheels leave the deck edge. The aircraft angle of attack during the launch stroke is nominally -1.5 degrees so that nominally 12 to 15 degrees (dependent on desired flight path angle) of positive pitch attitude rotation is necessary to achieve the correct fly-away attitude. Full extension of the nose strut after launch and weight-off-wheels provide a redundant and automatic transfer of the control valve to the EXTD position without pilot action required. With weight-off-wheels the NOSE STRUT control switch is inoperative. Transfer of the control valve to the EXTD position ports pressurized combined hydraulic fluid into the shock strut transfer cylinder which conditions the strut air curve (see figure 1-59) to that used for landing, taxiing and field take-off operations. In the strut extended position (18 inch stroke) approximately 9,000 pounds of static load on the gear is required to initiate compression so that the strut remains extended for taxiing and compresses upon brake application, during engine high power run-ups in the static position and upon nose wheel touchdown during landing.

NOSE GEAR AIR CURVES

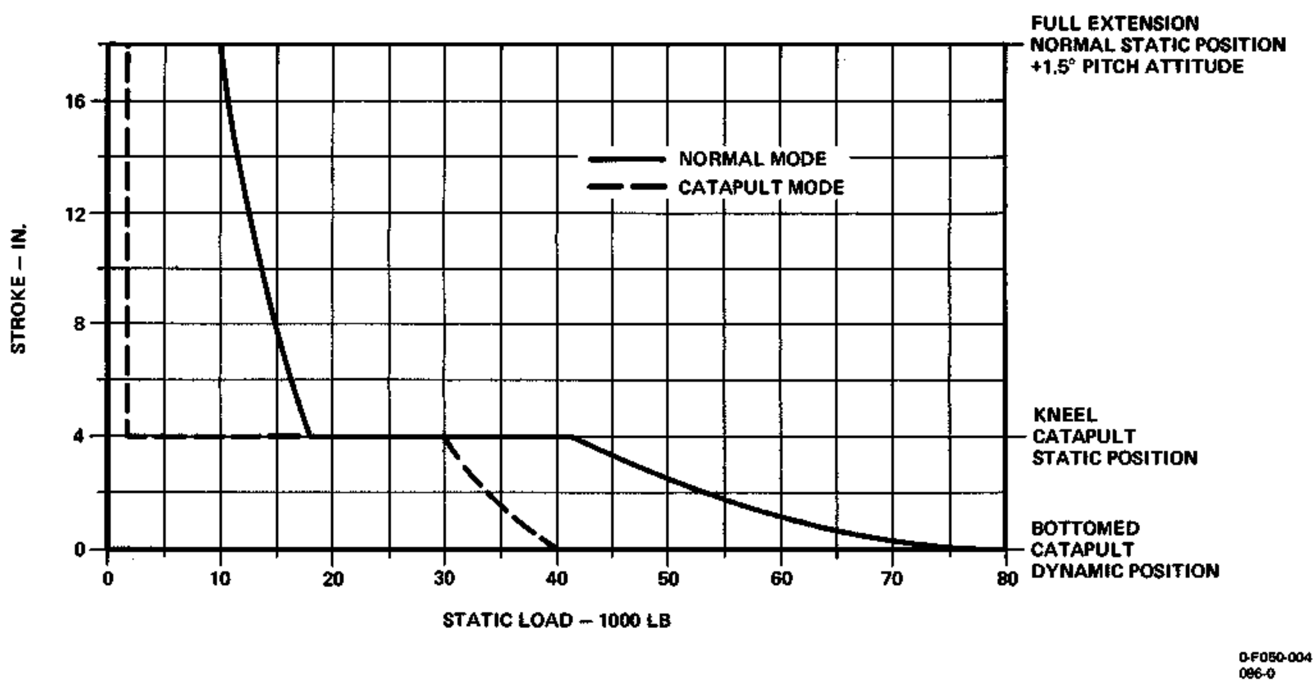


Figure 1-59.

Launch Bar

(U) The launch bar is attached to the forward face of the nose gear steering collar and serves as the tow link for catapulting the aircraft. With the nose strut extended the launch bar is held in the retracted position by the uplock and two compressed bungee arms which are connected to the upper torque arm. Unlocking of the strut in the extended position does not cause the bar to lower because of the compressed bungees which act as fixed links (see figure 1-60). Kneeling of the nose strut causes the bungee springs to extend and by turning the nose gear greater than ±10 degrees steering angle from neutral the uplock latch is released by a roller and cam plate. A proximity sensing switch on the uplock detects the latch out of the locked position and causes the LAUNCH BAR advisory light to illuminate. With the spring bungees extended, a force is applied to lower the launch bar and maintain a holddown force to insure that deck contact is maintained. The launch bar installation is compatible with conventional and flush deck catapult shuttle installations. Ears on the head of the launch bar engage under the lip of the catapult lead-in track and the head serves as a guide to steer the nosewheel onto the catapult track and to engage the shuttle for an abort, the launch bar cannot be raised until disengaging the shuttle and extending the nose strut. Engine thrust must be reduced below about 85% RPM in order for the strut to extend sufficiently to allow the launch bar to clear the shuttle. The LAUNCH BAR advisory light is interlocked to extinguish when both throttles are at the MIL position even though the launch bar position and mechanism remains unchanged; this action is effected to establish a "lights-out" criteria for launch.

(U) During the launch stroke the catapult tow force causes the strut to bottom and at shuttle release the stored energy strut dissipates its energy to fully extend the strut. Extension of the strut and torque arms mechanically cams the launch bar up to the retracted and locked position. The logic for the LAUNCH BAR advisory light will cause it to illuminate if the launch bar is not engaged in the uplock with weight-off-wheels. During nose gear retraction the wheel on the head of the launch bar rolls on a cam plate on the overhead of the nose wheel well to stow the launch bar upward against the shock strut outer barrel with the gear retracted. Upon nose gear extension the launch bar lowers from the stowed position to the uplock position and again is restrained from lowering by the uplock and two bungee arms. Thus, the operation of the launch bar is controlled from the cockpit indirectly, by control of the strut transfer valve and nose wheel steering, avoiding the need for additional motive hardware. With the strut kneeled the launch bar uplock latch can also be manually tripped externally be a ground crewman.

Holdback Fitting

The holdback fitting for insertion of the trail bar consists of a universal fitting at the apex (knee) of the nose strut torque arms. The mounting prevents the trail bar from being adversely loaded should the bar contact a deck obstacle while taxiing. A ground crewman must manually attach the trail bar prior to taxiing the aircraft into the catapult lead-in track. Proper positioning of the aircraft on the catapult relative to the shuttle is provided by a snubbing action of the trail bar. The trail bar consists of a reusable holdback bar that provides for repeatable releases at a tow force of 76,000 pounds. Exceedance of this tow force on launch causes the trail bar to release the aircraft holdback fitting and remain engaged in the catapult snubber mechanism.

LAUNCH BAR (CATAPULT)

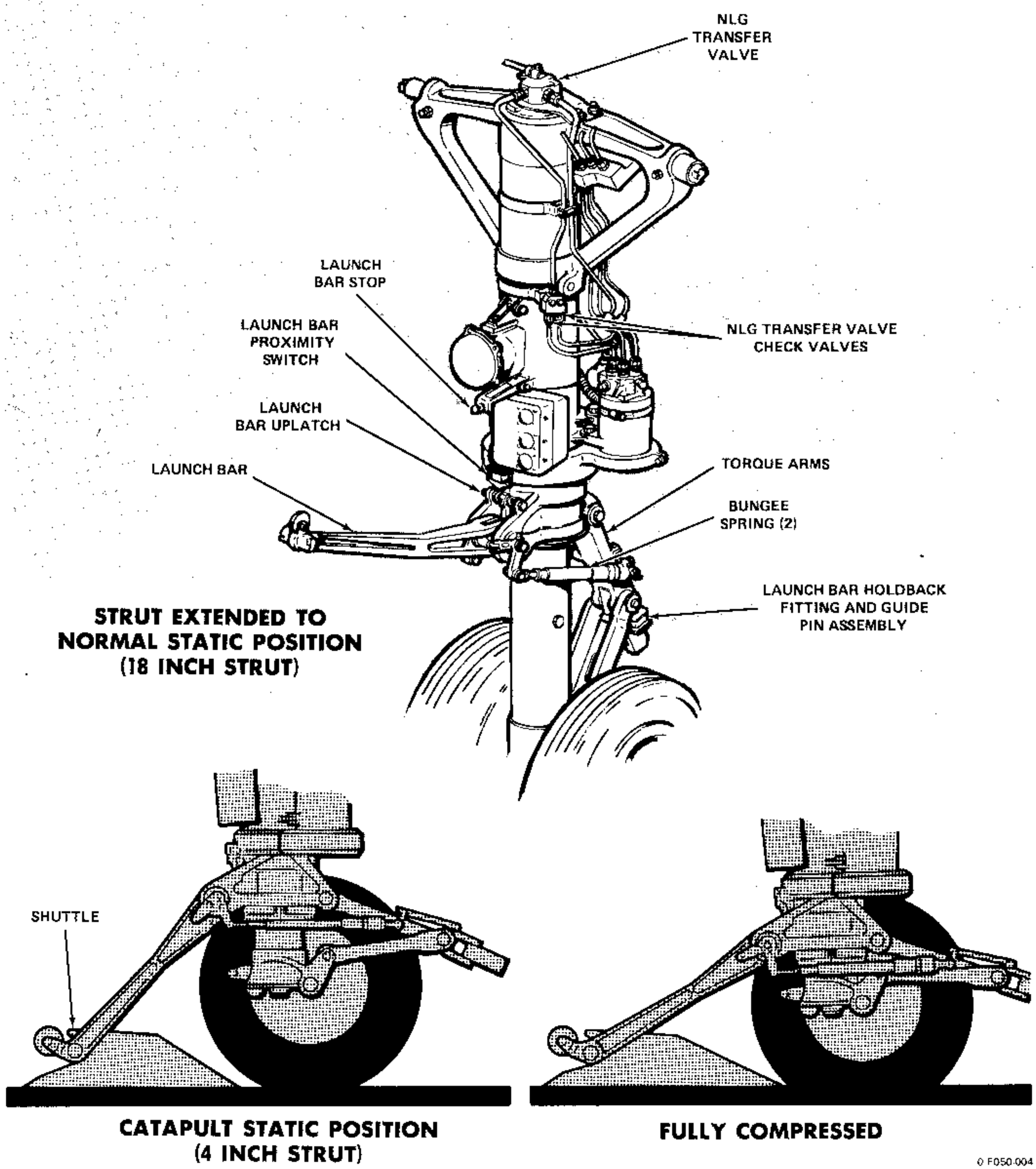


Figure 1-60.

(U) High power engine turn-up operations can use the holdback fitting to attach aircraft restraining hardware to deck secured fittings. Prior to the application of high power the nose strut should be kneeled and slack taken out of the holdback mechanism, otherwise dynamic loads may exceed mechanism design strength conditions.

ARRESTING HOOK

(U) The arresting hook system consists of a stinger type tail hook and associated control mechanism mounted to the underside of the center fuselage boattail which enables the aircraft to perform field or carrier arrestments. Structural members in the protruding tail hook attachment fairing transmit arrestment loads to the fuselage consistent with the limits defined in figure 1-61. The arresting hook shank is free to pivot at the attachment point in the aircraft vertical plane with a pneumatic dashpot preloading the hook down to minimize hook point bounce upon deck contact. Additionally, the hook shank is free to pivot in the aircraft lateral plane within a ±26 degrees sway angle with positive centering action provided by a pneumatic damper housed inside the tail hook shank. Aircraft geometric dimensions for pilot's hook-to-eye distance are presented in Table V for various approach pitch attitudes. The trail angle of the arresting hook provides for hook point-deck contact with the main landing gear in the static position and nose landing gear strut collapsed and tires flat. Hook point replacement can be effected without shank removal. The throat of the hook point should be appropriately greased to reduce wear and the throat design minimizes the probability of a double wire engagement. Nacelle ventral and nozzle ground clearance is such that arresting hook pendant dynamics excited by gear contact which exists at wire pick-up will not reasonably cause airframe contact. Normal operation of the arresting hook requires combined hydraulic system pressure, dashpot charged and essential dc No. 2 electrical power; because of a redundant means of pilot control (electrical and mechanical) emergency extension of the arresting hook can be accomplished without the aforementioned sources of power. Hook retraction requires electrical and combined hydraulic power.

(U) A redundant means of pilot control (electrical and mechanical) is provided for lowering the arresting hook, whereas electrical and combined hydraulic power are required for hook retraction. As illustrated in figure 1-61 normal operation of the pilot's hook control consists of a straight down-up movement of the hook control handle. This action actuates switches that provide electrical command signals to the hook control valve. For lowering the hook the control box causes the unlock side of the uplock actuator to be pressurized and vent fluid from the retract side of the lift cylinder. Dashpot pressure assisted by gravity causes the arresting hook to lower. With the hook down, the pneumatically charged dashpot provides holddown contact force to minimize hook bounce and maintain the hook in the trail position for arrestment. With the hook out of the uplock and weight-on-wheels, a nose wheel steering centering command is automatically provided to prevent nose gear castering on arrestment rollback. Pilot deflection of the rudder pedals when in the automatic nosewheel centering mode will not override the centering command until nose wheel steering is engaged by the pilot. The arresting hook not stowed switch is interlocked with the approach lights and indexer circuits to cause the lights to flash with gear down, weight-off-wheels and the HOOK BY-PASS switch in the CARRIER position. For hook retraction the control valve pressurizes the retract side of the lift cylinder and the lock side of the uplock actuator.

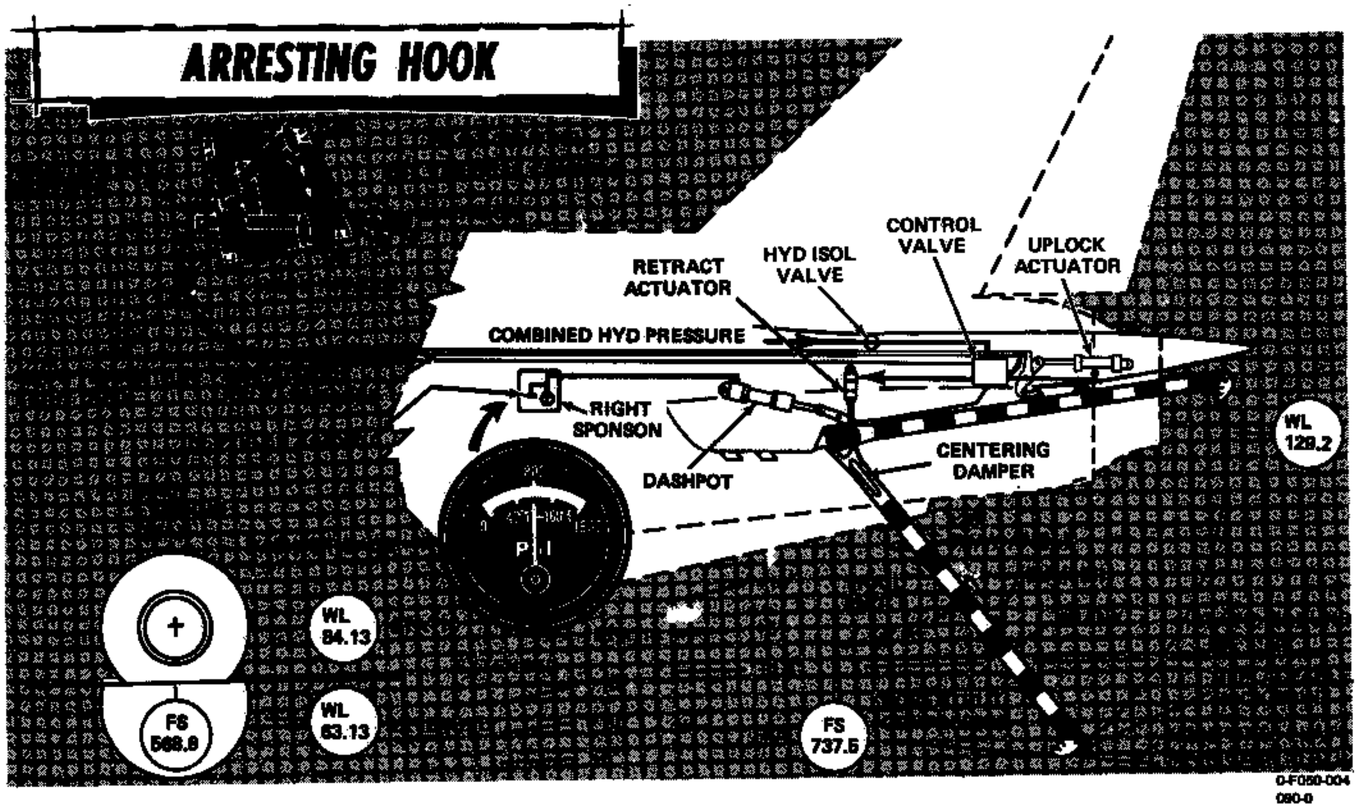

ARRESTING HOOK
CONTROL VALVE
UPLOCK ACTUATOR
RETRACT ACTUATOR
HYD ISOL VALVE
COMBINED HYD PRESSURE
RIGHT SPONSON
DASHPOT
CENTERING DAMPER
WL 129.2
WL 84.13
WL 63.13
FS 568.8
FS 737.5
0-F050-004
080-0

Figure 1-61

TABLE V. PILOT'S HOOK-TO-EYE DISTANCE

PITCH ATTITUDE DEG	HOOK-TO-EYE DISTANCE ~ FT	
	VERTICAL	HORIZONTAL
5°		
6°		
7°		
8°		
9°		
10°		

Note

(1) Pilot's eye F. S. ____________, W. L ____________

(2) Hook point in trail position F. S. ________, W. L. ________

When the arresting hook roller engages the uplock mechanism it triggers the uplock closed but the retract actuator remains pressurized unless the hydraulic isolation valve is closed. On-deck hook retraction time is approximately 3 seconds. The hook transition light illuminates as long as a discrepancy exists between the hook down and stow switches and cockpit handle positions. The transition light remains illuminated during on-deck extension which requires approximately 1 second.

(U) Normal and emergency hook extension controls are integrated into a single handle. The emergency control system lowers the hook by mechanically (via cable) tripping the uplock and venting hook lift cylinder pressure. Emergency hook extension is initiated by pulling the hook handle aft approximately 4 inches with the handle in the DN position.

(U) Although the hook retract system is automatically isolated from combined hydraulic system pressure with loss of the engine driven flight hydraulic pump, with weight-on-wheels the isolation valve is opened to enable hook retraction. There are no provisions for hook retraction in the event of a combined hydraulic system failure or loss of essential dc No. 2 power. The weight of the arresting hook (approximately 89 pounds) combined with dashpot pressure make it impractical to manually lift the hook without first bleeding off dashpot pressure. The hook can be temporarily suspended off the deck for towing by a chain attached to the hook uplock latch. A single pin inserted into the uplock serves as the safety device with the hook retracted.

ENVIRONMENTAL CONTROL SYSTEM

(U) The environmental control system regulates the environment of the aircrew and the electronic equipment. The system provides temperature controlled and pressure regulated air for the cockpits, canopy defogging, windshield anti-icing, rain removal, and electronic equipment. In addition, pressurized air is furnished to auxiliary systems such as suit vent air, anti-g suits, wing air bag seals, canopy seals, external fuel drop tanks, ammunition drum, and gun gas purging. A large portion of the refrigerated bleed air is also used by the airborne weapons control system (AN/AWG-9) and the Pheonix missile system (AN/AIM-54). Operating temperature is maintained in the radar and missile systems by using a liquid coolant (Coolanol 25) as a heat sink, processed through both ram air and refrigerated air heat exchangers.

(U) The pilot can shut down the air conditioning system and has control of admitting emergency ram air to the cockpits to clear them of smoke or fumes. Forced air cooled electronic equipment can also be cooled by emergency ram air.

(U) A provision for connecting a ground air conditioning unit is located on the lower right side of the fuselage aft of the nose gear wheel well. A schematic of the environmental control system is illustrated in figure 1-62.

REFRIGERATION AND PRESSURIZATION

(U) Refrigeration and pressurization is accomplished by a bootstrap cooling turbine and two ram air cooled heat exchangers. The system consists of three manifolds: a high temperature manifold, a 400° F manifold, and a cold air manifold.

(U) The high temperature manifold ducts air (1070° F and 400 psi, maximum) from each engine's 16th stage bleed air manifold to the primary heat exchanger. Either or both engine 16th stage manifolds are opened or closed by individual bleed air shutoff valves. Cockpit control of these valves is affected through the AIR SOURCE selector pushbuttons on the air conditioning control panel. The OFF button closes both bleed air shutoff valves and the dual pressure regulating and shutoff valve. The L ENG button closes the right engine shutoff valve which allows left engine bleed air to be supplied; and the R ENG button closes the left valve, which allows right engine bleed air to be supplied.

(U) The dual pressure regulating valve, located upstream of the primary heat exchange, regulates downstream pressure to 80 psi. If the primary regulator of the dual check valve fails, the secondary regulator limits downstream pressure to 90 psi. If both the primary and secondary regulators of the dual check valve fail, the valve automatically limits outlet pressure to 70 psi. A complete failure of the dual valve will close it, causing a complete shutdown of the pressurization and air conditioning system. If the valve fails in an open position, a pressure sensor downstream of the primary heat exchanger closes both engine bleed air shutoff valves when pressure reaches 150 psi.

(U) When the high temperature manifold is shutdown, either automatically or by the pilot, the AWG-9, IR and TV systems will automatically shutdown when radar pressurization is reduced. After an automatic shutdown, the OFF or RAM pushbutton must be depressed prior to opening the emergency ram air door so that the forced air cooled electronics and the cockpit will be cooled.

(U) Fire detection circuits in the engine compartments will detect a leak in the high temperature duct and illuminate the appropriate L or R FIRE warning light. Between each engine fire wall and the primary heat exchanger, bleed air leak sensing elements are placed along the outside of the duct. When these sensing elements detect temperatures in excess of 575° F, the BLEED DUCT caution light is illuminated.

(U) The primary and secondary air-to-air heat exchangers are located between the right and left engine inlets and the fuselage. At airspeeds above .25 Mach, ram air across the heat exchangers is used for cooling. During ground operations and at airspeeds less than .25 Mach, airflow across the heat exchangers is provided by air turbine powered fans. Air from the high temperature manifold is used to power the turbofan of the primary heat exchanger. The secondary heat exchanger turbofan is powered by air from the compressor side of the bootstrap turbine. Air inlets for these fans are located on the underside of the fuselage directly below each heat exchanger. When the fans are operating, spring-loaded butterfly valves open to allow airflow to enter the same inlet duct used by the ram air to the heat exchangers. The ram air inlets are closed by spring-loaded inward flapper doors when the fans are operating. When ram air forces the flapper doors open, a projection above the inlet doors forces the fan butterfly doors closed and prevents reverse airflow through the fans. The air turbofans have automatic overspeed protection only in the normal direction of rotation. After passing through the heat exchangers, the cooling air exits through open louvers on top of the left and right wing roots.

The 400° F Manifold

(U) Discharge air from the primary heat exchanger (500° F maximum) is used to pressurize the external fuel tanks and wing seals. Another branch of this manifold mixes primary heat exchanger discharge air (500° F maximum) and uncooled engine bleed air to provide the heating requirements for the radar/missile cooling system, pressure/ventilation suit, electronic cooling, cockpit heating, and windshield rain repellant system. Overheating of the 400° F manifold is prevented by actuation of the 475° F overtemperature switch, which closes the 400° F modulating, regulating and shutoff valve stopping the flow of engine bleed air. The remaining 400° F manifold air is cooled by passing through the service air-to-air heat exchanger which normally uses cold air manifold air as a heat sink. If the cold air manifold is inoperative, the pilot can open the emergency ram air door which will provide cooling air for the service heat exchanger. Air from the service heat exchanger is used for the pressure/ventilation suit, g-suit, and canopy seal.

(U) The air temperature to the pressure/ventilation suit can be controlled by the pilot modulating the temperature control thumbwheel on the air conditioning panel. A temperature switch in the pressure vent suit line closes the suit hot air modulating valve when temperatures exceed 120° F. In addition, each crewman can regulate the amount of airflow through their individual vent suit system by modulating the vent airflow thumbwheel on the oxygen panel. If the cold air manifold is shutdown, there is no cooling air across the service heat exchanger until the emergency ram air door is opened.

(U) Air from the service heat exchanger also provides servo air, reduced to 16.5 psi to operate temperature modulating valves. Additional air from the service heat exchanger is passed through two atmosphere pressure regulators which reduces the pressure to 30 psi and then through a dehydrator for moisture removal. Air from the dehydrator is used to pressurize the radar wave guide, antenna, transmitter, and AWCS/ missile liquid cooling loop expansion tanks. This air is again reduced to 15 psi and used to pressurize the 1R receiver/TV optical sight and the 1R scanner.

Cold Air Manifold

(U) Conditioned air from the primary heat exchanger flows through the modulating, regulating, and shutoff valve which regulates the air pressure and modulates the air flow to the compressor section of the bootstrap turbine-compressor. This valve automatically closes if compressor inlet temperature reaches 550°F or if temperature at the turbine side reaches 475°F. When the RAM pushbutton is selected, this valve is closed and the bootstrap turbine-compressor is shutdown. Prior to entering the compressor section, air is tapped off and bypasses the turbine to provide anti-ice temperature control of the cold air downstream of the bootstrap unit.

(U) The compressor increases the air pressure and temperature and ducts the air through the secondary heat exchanger. The temperature rise caused by the compression process is then removed in the secondary heat exchanger by heat transfer to ram cooling air. After leaving the secondary heat exchanger, the air enters the turbine section where it is cooled by expansion through a radial inward-flow turbine. As the air passes through the turbine, energy is extracted, pressure and temperature decrease, and cold air exits from the bootstrap turbine.

(U) The turbine incorporates a dual nozzle, allowing it to perform efficiently over wide operating conditions. When the air pressure entering the turbine drops below 60 psi, the turbine nozzle selector valve opens and allows air flow to both nozzles. When the pressure increases to 65 psi, the valve closes limiting air flow to one nozzle.

(U) Downstream of the turbine, a turbine discharge relief valve opens at 15 psi to relieve excess pressure in the cold air manifold. Cold air temperature is regulated by a temperature sensor downstream of the water separator to prevent icing in the water separator. The air is regulated to 35°F below 30,000 feet and 0°F above 30,000 feet. If the turbine discharge hot air valve cannot maintain 35°F with the turbofan operating, the turbofan in the primary heat exchanger slows down.

(U) Upstream of the water separator the cold air manifold branches. One branch goes to the AWCS/missile cooling system and another goes to the water separator. Check valves in each duct prevent reverse flow of ground cooling air. The water separator swirls the air and removes approximately 70% of the entrained moisture. From the water separator, the cold air flows to the cockpit for air conditioning, the gun drum for cooling and gas exhaust, through the service heat exchanger, and to the avionic equipment.

External Air Conditioning

(U) A provision for connecting a ground air conditioning unit is located under the fuselage, aft of the nose wheel well. An additional provision for connecting an external source of servo air is located in this same area.

(U) Because the standard ground air conditioning unit is not capable of providing sufficient cooling for the cockpits, avionics, and AWG-9/AIM-54 simultaneously, an electrically operated servo air controlled Y-duct diverter valve is located just downstream of the external air inlet. Cockpit control of the diverter valve is provided by the GND CLG switch on the NFO's right console. If the OBC/CABIN position is selected, ground cooling air will go to the cockpit, all the forced air cooled avionics, and a controlled amount of air will be available to cool the AWG-9 liquid loop at a reduced heat load. If AWG-9/AIM-54 position is selected, ground cooling air will go to the AWG-9/AIM-54 liquid cooling loop and all forced air cooled avionics associated with the system, which includes the IMU, transformer rectifiers, CICU, instrument power supply, and UHF equipment.

COCKPIT AIR CONDITIONING

(U) Cockpit air conditioning uses cool dry air from the cold air manifold and hot air from the 400°F manifold to provide temperature and pressure regulated air. Prior to entering the cockpit the hot air cockpit modulating valve mixes 400°F air with the conditioned air. The conditioned air passes through a flow sensing venturi which controls the modulating, regulating and shutoff valve.

(U) The pilot can control cockpit temperature by selecting either a manual (MAN) mode or automatic (AUTO) mode with the temperature mode selector switch. In the AUTO mode, temperature is determined by the number selected by the pilot with the temperature control thumbwheel. This desired temperature is maintained by a cabin temperature sensor in the forward left side of the cockpit. In the MAN mode, the temperature control thumbwheel manually positions the cockpit hot air modulating valve to maintain cockpit and suit air temperatures. If cockpit airflow temperatures reach 250°F, a cockpit over-temperature protection switch closes the hot air modulating valve.

(U) The conditioned air entering the cockpit is divided forward and aft, with 50% of the air going to each cockpit. A defog lever located on the right console in each cockpit individually controls the percentage of airflow through the cockpit diffusers and the canopy defog nozzles. When the lever is in the CABIN AIR position (full aft), 70% of the air is directed through the cockpit diffusers and 30% through the canopy defog nozzles. In the DEFOG position, 100% of the air is directed through the canopy defog nozzles.

AMMO COOLING SYSTEM

(U) The ammo cooling system uses dry conditioned air after it has passed through the water separator to ventilate the ammo drum. Immediately prior to gun firing, the gun control unit electrically opens a modulating valve allowing conditioned air to enter the ammo compartment. A hydraulically operated gun gas exhaust door at the bottom of the gun drum compartment is opened. This draws the conditioned air through the gun compartment, expelling the unburned hydrogen and carbon monoxide gases overboard. Ram air passing through and around the gun barrels also help expell gun gases. To ensure complete ventilation and purging of the gun breech, the cool air shutoff valve and the exhaust door remain open for approximately 35 seconds after the gun firing has ceased. A slot aft of the gun exhaust door is fixed open to allow ventilation while in flight.

(U) During emergency ram air operations when the air conditioning system is completely shut down, ram air will be ducted to the ammo and gun compartments.

RAM AIR OPERATIONS

(U) If the environmental control system is shut down for any reason, emergency ram air can be used to ventilate the cockpits, ammo drum and gun exhaust, and provide cooling air to the service/suit heat exchanger and those electronic subsystems requiring forced air cooling. The emergency ram air door is on the lower right side of the fuselage, in-board of the right glove vane. To activate the emergency ram air door actuator the pilot must select either the OFF or the RAM air select pushbutton on the air conditioning control panel. Selecting the L ENG, R ENG, or BOTH ENG pushbuttons automatically closes the emergency ram air door. Opening and closing the ram air door is controlled by the pilot modulating the RAM AIR lever on the air conditioning control panel. To increase cockpit air flow, the cockpit pressurization must be dumped.

(U) When the RAM pushbutton is selected the modulating, regulating, and shutoff valve (between the primary heat exchanger and the turbine compressor) is closed, stopping the flow of high-pressure bleed air to the bootstrap turbine compressor. Warm air from the primary heat exchanger is cooled by the emergency ram air in the service/suit heat exchanger. The emergency ram air then flows to the cockpit and gun ammunition drum.

(U) However, the emergency ram air does not provide cooling to the AWG-9/AIM-54 liquid cooling loop. The AWG-9 liquid cooling loop incorporates a ram air - liquid heat exchanger. This ram air door is located under the right glove forward of the primary heat exchanger inlet. There are no cockpit controls for this ram air door; it is controlled by the AWG-9/AIM-54 controller and is independent of the air conditioning and pressurization system. Therefore, loss of the ECS air pressurization and conditioning cold air manifold will not affect operation of the radar, IR or TV cooling loop. The AWG-9 system ram air heat exchanger will maintain the liquid temperature within operating limits over a wide range of the aircrafts operating envelope. However, loss of the 400°F manifold which is used to pressurize all electronic equipment will require the aircrew to shut down the AWG-9, IR or TV systems. The 400°F manifold is shut off anytime the OFF AIR SOURCE pushbutton is selected or the dual pressure regulating valve fails. If loss of aircraft pressurization is suspected, the flight crew should check for G-suit and vent suit operation. If suit pressurization is available AWG-9 and IR/TV operation can be continued.

CAUTION

> Before allowing ram air to enter the aircraft reduce airspeed to 350 KIAS or 1.5 Mach, whichever is lower, to prevent ram air temperatures above 110°F from entering the aircraft.

COCKPIT PRESSURIZATION

(U) From sea level to 8,000 feet altitude, the cockpit remains unpressurized. Between altitudes of 8,000 feet to 23,000 feet the system maintains a constant cockpit pressure altitude of 8,000 feet. At altitudes above 23,000 feet, the cockpit pressure regulator maintains a constant 5 psi pressure differential greater than ambient pressures. A graphic illustrating of the aircraft's cabin pressure schedule is provided in figure 1-63.

(U) A cockpit pressure altimeter (figure 1-64) is provided on the right side of the pilot's instrument panel so that he may monitor cockpit pressure altitude. In the rear cockpit, the NFO is provided a low pressure warning light on the caution/advisory light panel. This pressure caution light, placarded CABIN PRESS will come on if cockpit pressure drops below 5 psia or cockpit altitude is above 27,000 feet.

(U) If the cockpit pressure regulator malfunctions, the cockpit safety valve will open to prevent a cockpit pressure differential from exceeding a positive 5.5 psi or a negative differential of 0.25 psi. The cockpit pressure regulator and the safety valve are pneumatically operated and function independently through separate pressure sensing lines.

CABIN PRESSURE SCHEDULE

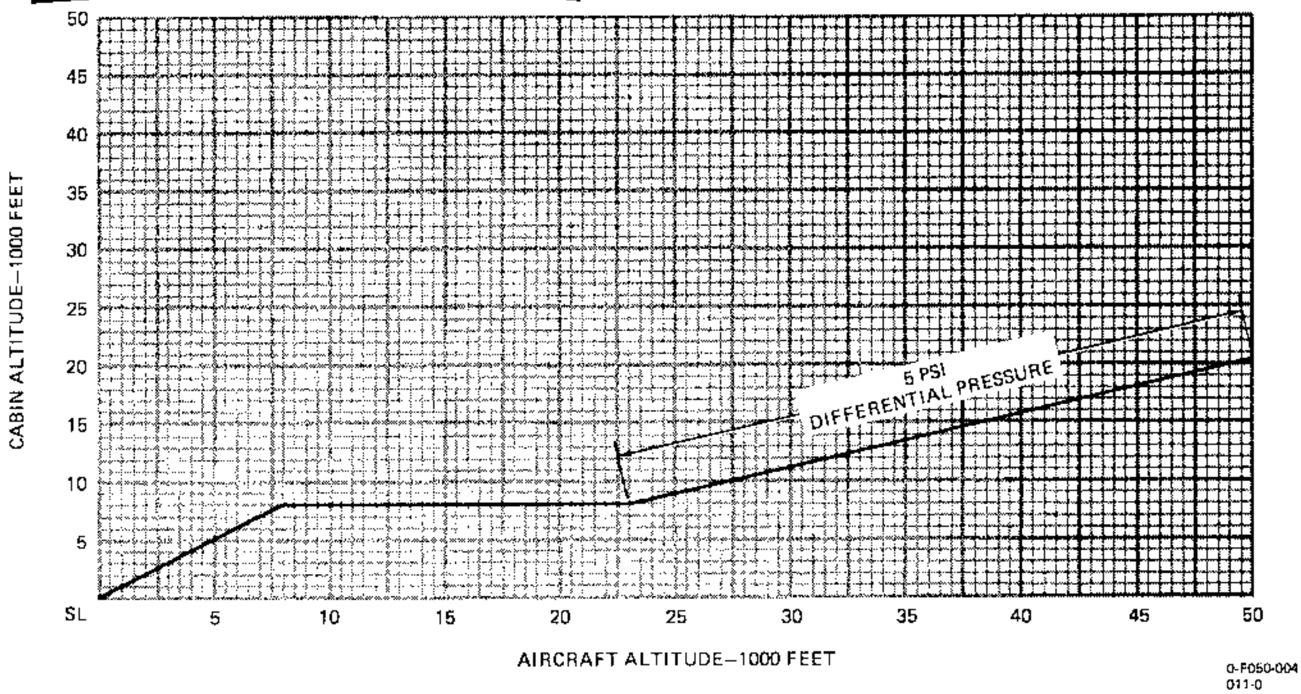


Figure 1-63

(U) Cockpit pressurization can be dumped by the pilot selecting the DUMP position with the cabin pressurization switch. When the DUMP position is selected the safety valve is immediately opened and the cockpit is depressurized.

PRESSURE/VENTILATION SUIT

(U) The pressure/ventilation suit system provides temperature controlled, pressure regulated engine bleed air to both vent suits or to the ejection seat ventilated cushions when pressure suits are not worn. The pilot can select either a manual or an automatic temperature control mode and can regulate a desired suit or seat cushion temperature. Each individual crewman can control the volume of air flow through the pressure suit or seat cushion by modulating the VENT AIR FLOW thumbwheel on his left inboard console.

(U) Conditioned air from the service heat exchanger, augmented with 400 F air is supplied at the temperature selected by the pilot with the temperature control thumbwheel on the air conditioning control panel. The air pressure is regulated to 3 psi above cockpit pressure and the air flow controlled by the individual crewman by the VENT AIR FLOW thumbwheel.

ANTI-G-SUIT

(U) Each anti-g suit is connected to the aircraft pressurization system by an anti-g suit hose which delivers pressurized air to the suit control valve and then to the suit through a composite disconnect. Below 1.5 g's the suit remains deflated. A spring balanced anti-g valve automatically opens when g forces exceed 1.5 g's to overcome the valve spring tensions. Operation of the anti-g suit valve may be checked by depressing the test button marked "G" VALVE on each crewman's left console. Depressing the test button will inflate the anti-g suit bladders. When the test button is released the bladders will deflate.

AIR CONDITIONING AND PRESSURIZATION CONTROLS

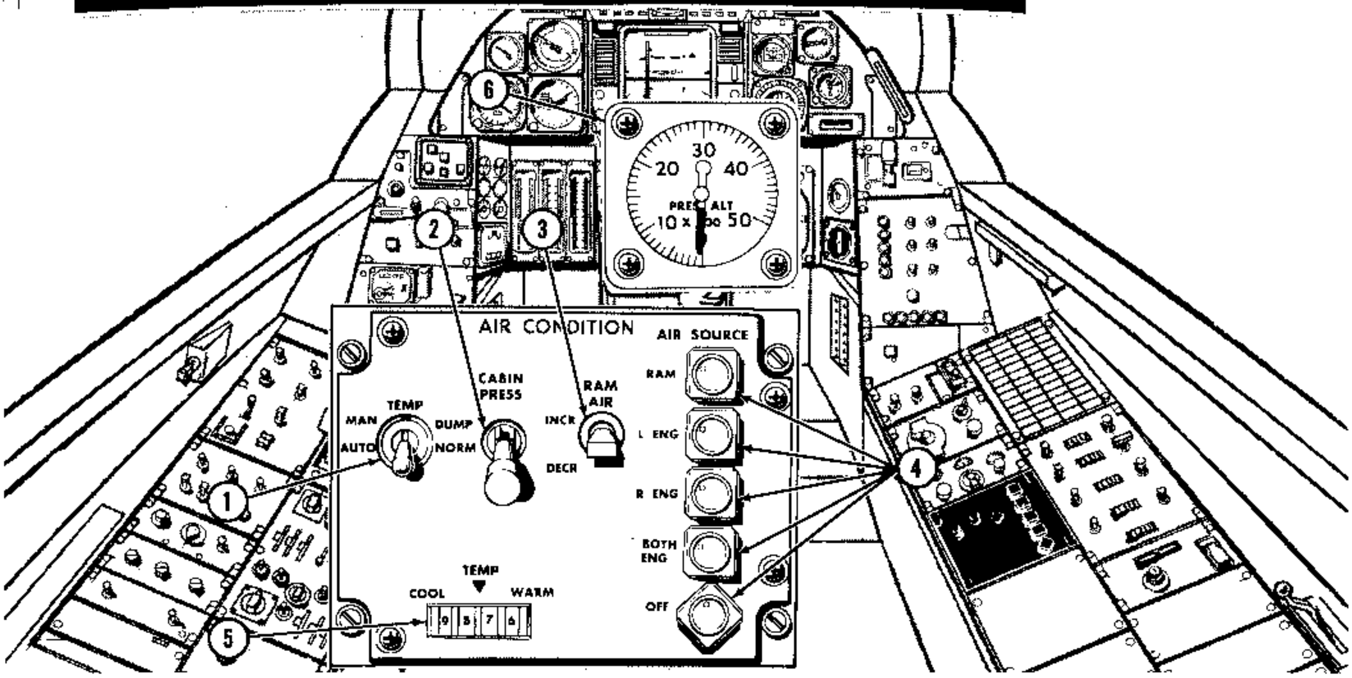


Figure 1-64

	NOMENCLATURE		FUNCTION
1	TEMP MODE SELECTOR SWITCH	AUTO -	Cockpit and pressure suit temperature is automatically maintained at that comfort level selected on the temperature control selector.
		MAN -	Cockpit temperature must be manually selected as airspeed and altitude change to maintain a desired temperature.
2	CABIN PRESS SWITCH Two position lever-lock switch. Must be lifted to be moved to the DUMP position.	NORM -	Cockpit pressure will be maintained at an altitude of 8,000 feet up to 23,000 feet, above which the regulator maintains a 5 psi pressure differential.
		DUMP -	The pressure safety valve is opened, depressurizing the cockpit, and activates ram air system.

NOMENCLATURE		FUNCTION
3	RAM AIR CONTROL	Manually opens the ram door and regulates the amount of ram air after the RAM or OFF air source button is selected.
4	AIR SOURCE SELECTOR BUTTONS	RAM - Closes the modulating, regulating, and shutoff valve. The ram air control switch is activated. Combined ram air and regulated 400° F bleed air are available to the cockpits and air cooled electronic equipment. The RAM AIR control selector must be used to regulate the ram flow entering the cabin. When either BOTH ENG, L ENG or R ENG are selected, the ram air door automatically closes. L ENG - The left engine is the source of bleed air for the environmental system and the right engine bleed air shutoff valve is closed. R ENG - The right engine is the source of bleed air for the environmental system and the left engine bleed air shutoff valve is closed. BOTH ENG - The right and left engine bleed air shutoff valves are open and both supply bleed air to the environmental control system. This is the normal position. OFF - Both the left and right engine bleed air shutoff valves and the dual pressure regulator valve are closed. Pressurization and air conditioning are not available. Activates the ram air control switch.
5	TEMPERATURE CONTROL THUMBWHEEL	Used to select cockpit and suit air temperature. It can be rotated through a 300 degree arc with mechanical stops at each end placarded COOL and WARM. A mid position temperature is approximately 70°F in the automatic mode. With the mode selector switch in AUTO the temperature selected is automatically maintained by the modulating temperature control valves. In the MAN position, the temperature control thumbwheel must be repositioned to maintain cockpit and suit air temperature with changes in airspeed and altitude.
6	CABIN ALTITUDE INDICATOR	Displays cabin altitude in one thousand foot increments from 0 to 50,000 feet.

CANOPY SEAL PRESSURIZATION

(U) Pressurized air from the air conditioning system is ducted through the cockpit to the canopy seal. The seal is automatically inflated when the canopy actuator is moved to the closed position. A check valve in the canopy pressure regulating valve prevents the loss of canopy seal pressurization if the conditioned air manifold is depressurized. The canopy pressure regulating valve maintains the seal pressurization at 25 ± 5 psig. Initial movement of the canopy actuator automatically deflates the seal and vents the air pressure through the canopy pressure regulating valve.

EQUIPMENT PRESSURIZATION

(U) Pressurized air is passed through a regenerative dual desiccant system (dehydrator) to assure dry air and prevent internal arcing of the high power AWCS components and to pressurize both expansion tanks of the liquid cooling system. Air pressure at 30 psia from the two atmosphere pressure regulator is directed through the desiccant selector valve which is controlled by a timer. The timer cycles every twelve minutes of operation to alternate the flow of drying air desiccant beds. This dry 30 psia air pressure is supplied directly to the radar wave guide and expansion tanks. Downstream of the desiccant dryer, the 30 psia air is further reduced to 15.75 psia required for pressurization of the IR equipment.

ELECTRONIC EQUIPMENT COOLING

(U) Ambient cooled electronic equipment located in the electronic bays is cooled by the air exhausted from the cockpits. Equipment incapable of being cooled by free convection is force-air cooled from the cold air manifold. Below 30,000 feet, this air is maintained at 62°F to eliminate entrained moisture. At higher altitudes, where moisture is not present, the air temperature is controlled at 42°F. If the ambient air temperature is inadequate to cool the forced air cooled electronic equipment, the AWG-9 OVHT advisory light on the NFO's CAUTION-ADVISORY panel will illuminate. Exhaust air from the forced air cooled electronics is also discharged into the electronic bays before it is dumped overboard through the nose wheel compartment. A schematic of the AWG-9/AIM-54 and electronic equipment is illustrated in figure 1-65. Controls and indicators are shown in figure 1-66.

AWG-9 Electronic Equipment Cooling

(U) The electronic equipment cooling (figure 1-65) is necessary to dissipate the heat generated by the AWG-9 equipment. This is accomplished by circulating 8 gpm of coolant fluid through the electronics and rejecting the heat picked up by the coolant in the ram-air heat exchanger and/or the AWCS/missile heat exchanger.

(U) When the ram-air temperature is below 110°F, (with the radar system transmitting, or 88 F when the radar is not transmitting), the liquid cooling temperature exiting the ram-air heat exchanger is reduced to 70°F by modulating the ram-air flow through the ram-air heat exchanger.

(U) If the heat exchanger liquid coolant exit temperature exceeds 82°F, the excess heat is rejected to ECS air in the AWG-9/AIM-54 heat exchanger. Under these conditions, the liquid coolant is maintained at 82°F by modulating the cold air flowing through that section of the AWG-9/AIM-54 heat exchanger associated exclusively with the AWG-9 fluid circuit.

AW6-9/AIM-54 AND AVIONIC EQUIPMENT COOLING

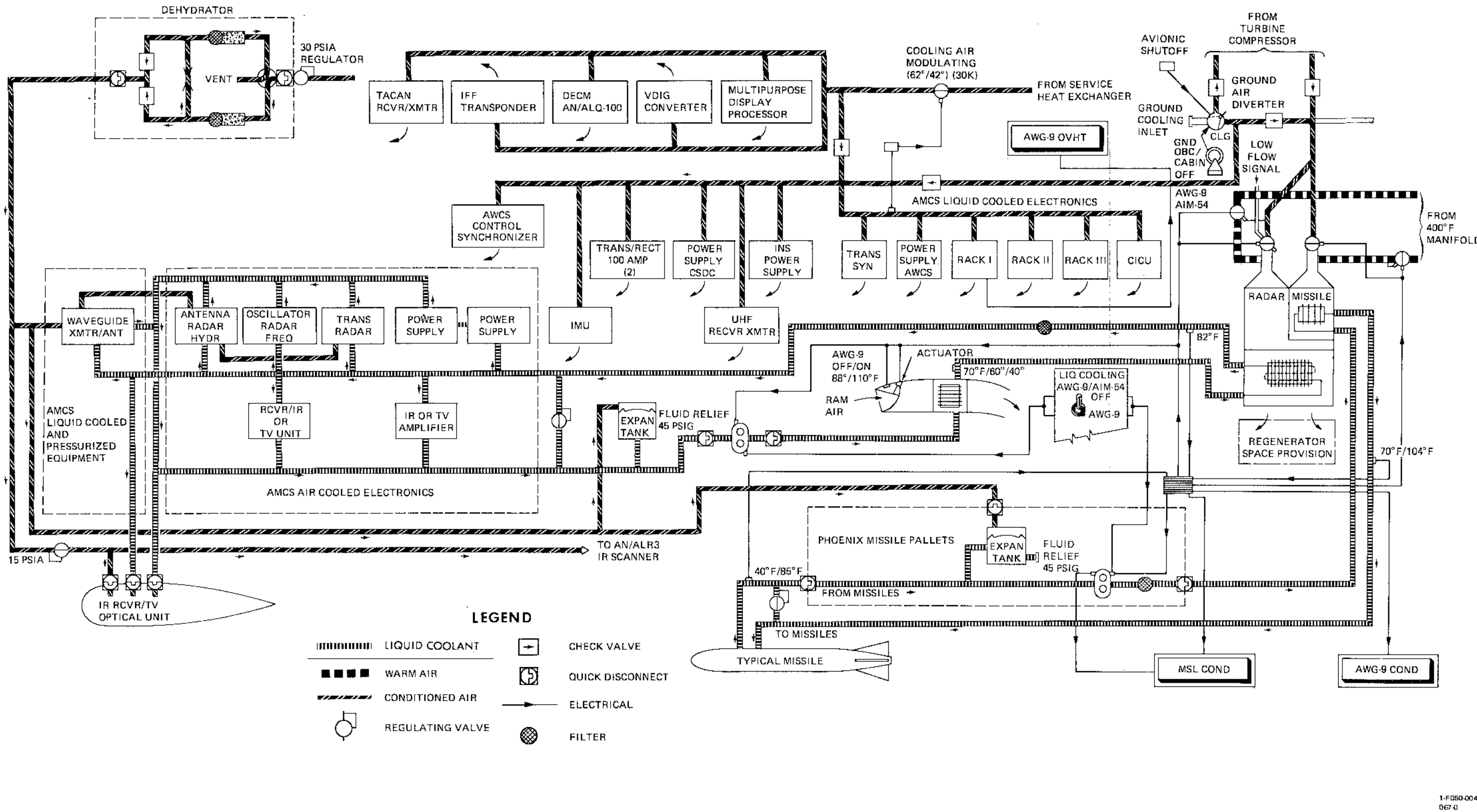


Figure 1-65.

When the ram-air temperature exceeds 110°F or 88°F, depending on radar transmitter mode, the ram-air door is closed and the heat of the AWG-9 system is rejected to the ECS air. The cold air valve accepts a pneumatic low air flow external signal from the cabin flow sensor which will modulate this valve toward the closed position until the minimum cockpit air-flow requirements are satisfied.

(U) The AWG-9 electronic equipment warmup is accomplished by bleed air. When the temperature of the coolant returning from the AWG-9 electronics falls below 40°F, the system is placed in the warm-up mode. Under this condition the cold air valve is closed and hot bleed air is introduced upstream of the AWCS heat exchanger by a modulating hot air valve to achieve an 82°F fluid temperature exiting the AWG-9/AIM-54 heat exchanger. This mode of operation is maintained until the return temperature from the AWG-9 electronics reaches 60°F, at which time the system reverts to normal cooling operation.

(U) The AWG-9 cooling loop is activated by the LIQ COOLING switch on the NFO's left outboard console. In the AWG-9/AIM-54 switch position, both the AWG-9 radar and AIM-54 missile cooling loops are activated for airborne operation. If the AWG-9 position is selected, only the radar cooling pump for airborne thermal conditioning is available. A temperature sensor in the AWG-9/AIM-54 heat exchanger outlet line will illuminate the AWG-9 COND advisory light when the liquid temperature exceeds 104°F. In addition, a pressure switch in the AWG-9 pump will illuminate the AWG-9 COND advisory light when pump output pressure is less than 60 psi. All of the above functions are controlled and automatically switched between the various operating modes by the system controller using the temperature signals from the four liquid temperature sensors, and the ram air sensor. The AWG-9 OVHT advisory light will illuminate after a time delay between 25 to 40 seconds when an over temperature condition exists.

(U) During ground operation of the AWG-9 radar system the thermal conditioning loop is activated by the GND CLG switch on the IFF ANTENNA/TEST panel located on the NFO's right outboard console. With an external air conditioning unit attached to the aircraft, and the GND CLG switch in the AWG-9/AIM-54 position, the ground cooling diverter valve directs the external conditioned air to the AWCS and missile systems. Figure 1-66 illustrates the controls and indicators associated with the AWG-9 cooling loop.

AWG-9/AIM-54 LIQUID COOLING CONTROLS AND INDICATORS

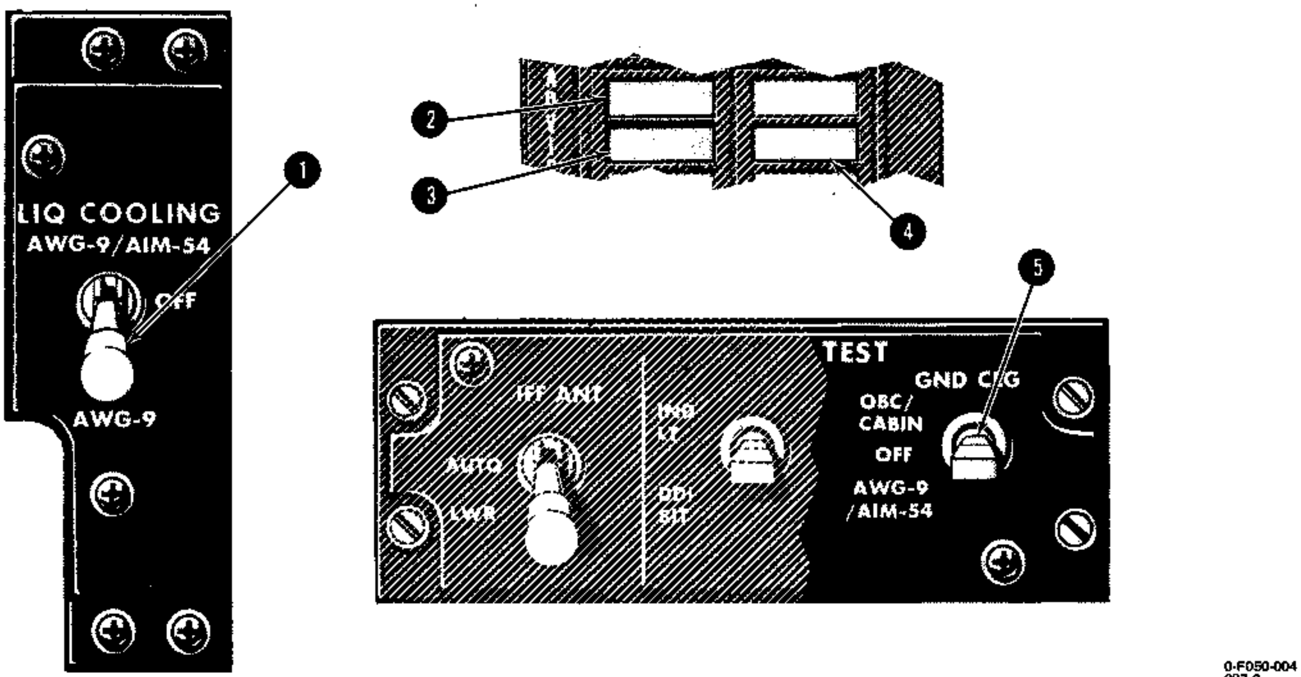


Figure 1-66.

NOMENCLATURE	FUNCTION		
(1) LIQ COOLING SWITCH	AWG-9/AIM-54	-	Activates the AWCS cooling pump motor and the AIM-54 cooling pump motor to provide ground and airborne thermal conditioning of the AWCS and Phoenix missiles.
	OFF	-	Deactivate the cooling systems and shuts down AWCS and AIM-54 cooling pumps.
	AWG-9	-	Activates the AWCS (AWG-9 radar) cooling pump for ground and airborne thermal conditioning.
(2) AWG-9 OVHT LIGHT (advisory light)	Illuminates after a time delay of 25 to 40 seconds when an over temperature condition exists in the electronic forced air cooling system.		
(3) MSL COND LIGHT (advisory light)	Illuminates when temperature of the coolant flow to the missiles exceeds 115° ± 3 F; also indicates pump output pressure is below 60° ± 5 psi.		

	NOMENCLATURE	FUNCTION	
4	AWG-9 COND LIGHT (advisory light)	Illuminates when the coolant exiting the heat exchanger reaches 104° ± 3 F; also indicates pump output pressure is below 60 ± 5 psi.	
5	GND CLG SWITCH	OBC/CABIN	- Positions ground cooling diverter to allow external cooling air for all forced air cooled electronic equipment. Reduced supply to AWG-9 liquid loop heat exchanger. Deactivates cockpit low flow override signal and removes primary power to AWG-9 electronics.
		OFF	- Deactivates the ground air diverter. Activates the cockpit low flow override signal.
		AWG-9/AIM-54	- The ground cooling diverter is positioned to allow external cooling air for the AWCS/missile liquid cooling system and all forced air cooled avionics associated with the system. Also cools IMU, transformer rectifiers, CICO, instrument power supply, and UHF radio. Deactivates the cockpit low flow override signal.

AIM-54A Missile Cooling

(U) The AIM-54A cooling (figure 1-65) is required to dissipate heat generated in the AIM-54A missile electronic systems. This is accomplished by circulating 18 gallons of coolant fluid per minute through one to six missiles (3 gpm per missile) at a controlled inlet fluid temperature to the missiles of 70°F. Heat picked up in the missiles is rejected to ECS air in the AWCS heat exchanger section associated with the AIM-54A liquid circuit. A liquid temperature sensor monitors the fluid outlet temperature of the AWCS heat exchanger and provides a signal to the controller, which in turn continuously positions the electrically controlled, pneumatically actuated cold-air valve to maintain the coolant exit temperature at the desired 70°F.

(U) If the temperature of the coolant delivered to the missiles exceeds 115°F, the MSL COND advisory light on the NFO's CAUTION-ADVISORY panel illuminates. A pressure switch in the AIM-54 cooling loop pump will illuminate the MSL COND advisory light on the CAUTION-ADVISORY when pump output pressure is less than 60±5 psi. When the pressure switch is actuated, the cold-air or hot-air valve will close.

(U) Whenever the temperature of the fluid returning from the missiles falls below 40°F, the coolant is automatically put into the warm-up mode. The controller will close the cold-air valve and modulate the hot-air valve to maintain the fluid exiting the AIM-54A section of the AWCS/missile heat exchanger at 104°F. This condition will continue until the return fluid temperature reaches 85 F at which time the system reverts to the normal 70°F cooling mode of operation.

CANOPY DEFOG AND CABIN AIR

(U) Conditioned air for canopy defogging and cockpit air distribution is controlled by the cabin air control lever (figure 1-67) on the right console of each crew position. The CABIN AIR Position is the normal position. This allows half the conditioned air to flow through the cockpit air diffusers and half through the canopy air diffusers. When the temperature selector toggle switch is in AUTO, cockpit temperature is automatically maintained by the cabin temperature sensor in the cockpit. The pilot must select a temperature range (60°to 80°F) using the thumbwheel temperature control. In the MAN position a maximum temperature of 250°F could be selected by rotating the thumbwheel to full WARM. A thermal safety switch closes the cabin hot air modulating valve at 250°F.

(U) When the DEFOG position is selected the air flow is directed through the canopy air diffusers only. If canopy fogging is excessive, select the MAN position and a warmer temperature with the temperature control will defog the canopy.

WINDSHIELD HEAT

(U) Windshield heating is provided by electrical elements implanted between the windshield panes. The temperature of the windshield is maintained automatically to provide 105°F whenever electrical power is supplied to the aircraft. There are no cockpit controls for windshield heating. A test switch on the SYS TEST/SYS PWR ground check panel is provided to ground test the system. The electrical elements are energized by essential DC No. 2 power.

WINDSHIELD ANTI-ICE SYSTEM

(U) The windshield anti-ice system provides a blast of hot air (approximately 390°F) over the exterior surface of the windshield. A temperature overheat sensor located at the base of the windshield protects the windshield from overheating. When the sensor detects an overtemperature condition (300°F), a signal closes the pressure regulating valve and illuminates the WSHLD HOT advisory light on the pilot's caution/advisory light panel. The windshield anti-ice system is activated by selecting the AIR (center) position with the WINDSHIELD lock lever switch on the EXT ENVIRONMENT PANEL (figure 1-67).

WINDSHIELD RAIN REMOVAL/REPELLENT SYSTEM

(U) The windshield rain removal system is provided to maintain the windshield clear of impinging rain. Compressor bleed air at approximately 390°F and at a pressure of 18 psi is directed over the outside of the windshield through a fixed area nozzle. This blast of hot air over the windshield will evaporate rain and prevent its further accumulation. The system is effective for rain rates of .6 inches per hour at normal landing speeds. A chemical rain repellent spray is provided to supplement the hot air system. It operates in conjunction with the rain removal system and improves visibility at rain rates above .6 inches per hour. However, the rain repellent should not be used at low rain rates or on a dry windshield.

(U) A self-contained chemical rain repellant system supplies a metered spray to the surface of the windshield in conjunction with the air blast to improve visibility during heavy rain conditions. When the RAIN REPEL position (figure 1-67) is selected a time delay opens a shutoff valve for approximately 0.7 seconds and releases a spray of fluid over the windshield surface. The windshield rain removal system is supplied electrical power from the essential dc No. 2 bus and protected by a circuit breaker on the NFO's circuit breaker panel.

NOTE

The chemical rain repellant is not a windshield washer and should not be applied to a dry surface or during light rain conditions. If used on a dry windshield or during light rain, a milky haze will form on the windshield surface.

WINDSHIELD RAIN REMOVAL AND DEFOG CONTROLS

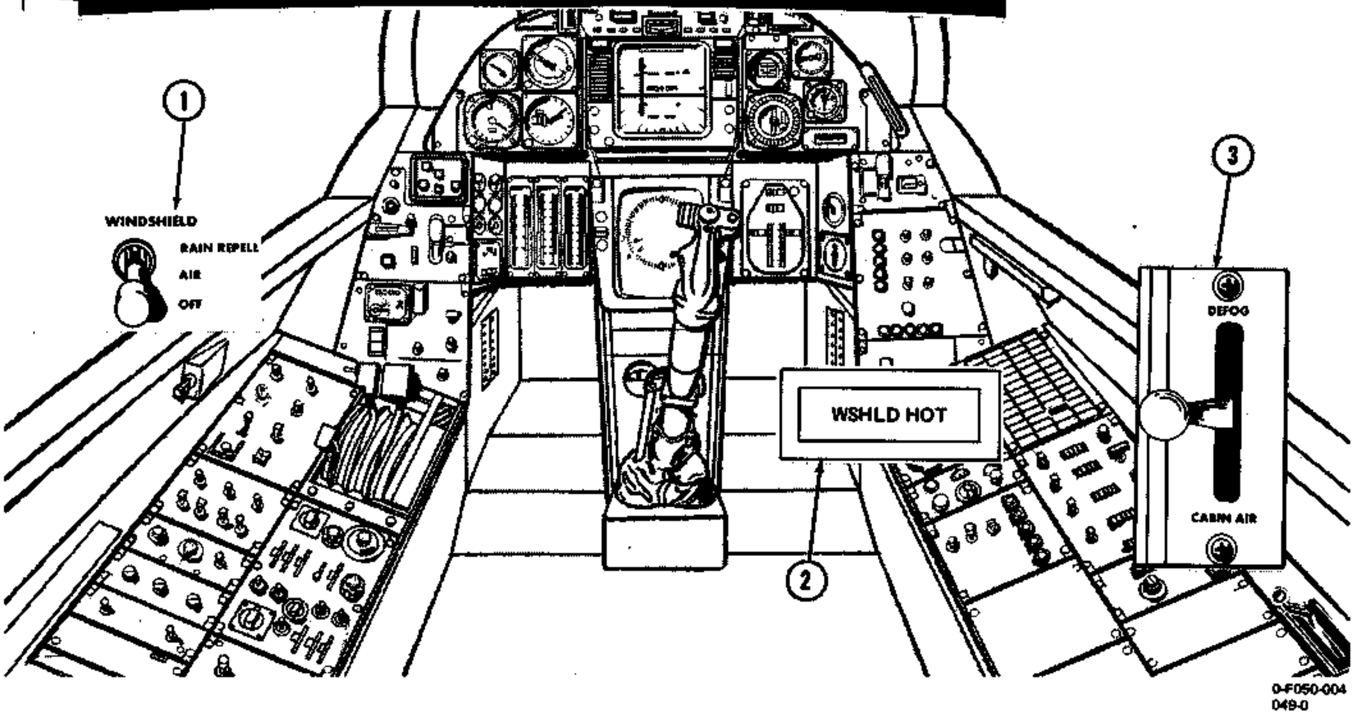


Figure 1-67.

	NOMENCLATURE	FUNCTION		
1	WINDSHIELD RAIN REMOVAL/REPEL SWITCH	AIR/RAIN REPEL	-	Momentary switch position. Lock lever switch must be pulled out and moved up to select rain repel. Provides a chemical spray and a blast of hot air over the exterior windshield. Should be used only during heavy rain and low air-speed (takeoff and landing).
		AIR -		Provides a hot blast of air (390°F) over the exterior windshield. Used for windshield anti-ice. Switch will spring return to this position from AIR/RAIN REPEL.
		OFF -		Closes the shutoff valve after a 5 second delay. The system is deenergized.

NOMENCLATURE		FUNCTION
(2) WSHD HOT LIGHT		Windshield hot advisory light will come on when windshield overheat condition exists (300°F).
(3) CABIN AIR/DEFOG LEVER (Both Cockpits)	CABIN AIR -	70% of the conditioned air directed through the cockpit air diffusers and 30% is through the canopy air diffusers. This is NORMAL position.
	DEFOG -	Air flow is directed through the canopy air diffusers only.

OXYGEN SYSTEM

(U) Breathing oxygen is provided each crewman by converting liquid oxygen to gaseous oxygen within either one or two liter liquid oxygen converters. Emergency oxygen is available to both crewmen through a high pressure gaseous oxygen bottle contained in each ejection seat pan.

NORMAL OXYGEN SYSTEM

(U) The normal oxygen system provides crewmen with 100% pressure regulated, temperature controlled oxygen. The system is designed for use with a pressure-demand type regulator which may be mask mounted or chest mounted. It is also compatible with the the full pressure suit oxygen regulator. Normally one 10 liter liquid oxygen converter is installed. Provisions are available for installation of a second converter if required. Liquid oxygen in the converter is changed to gaseous oxygen at a pressure of 70 to 80 psi in normal use. From the converter, gaseous oxygen is routed through a heat exchanger where its temperature is increased to a level that is comfortable for breathing. From there, oxygen passes through a demand type pressure regulator and into the oxygen mask or full pressure suit helmet. The liquid oxygen converters are mounted in the right side of the fuselage adjacent to and beneath the forward cockpit. An access door in the side of the fuselage is provided for servicing the system. Oxygen quantity duration chart is shown in figure 1-68.

OXYGEN PANELS

(U) Oxygen panels are located aft on the left console in each cockpit (figures 1-3 and 1-4). A single switch on each panel placarded OXYGEN has two positions, ON and OFF. In the ON position oxygen is free to flow from the converter through the regulator to the mask. In the OFF position a dual shut off and relief valve at the control panel closes to shut off flow to the regulator.

OXYGEN QUANTITY INDICATOR

(U) An oxygen quantity indicator is located on the right side of the pilot's right knee panel. The indicator is calibrated in liters and indicates quantity remaining in increments of one liter. The indicator operates on 115V power from essential bus No. 2. If a loss of electrical power occurs the indicator will indicate less than zero and the OFF flap on the lower face of the indicator will appear. The liquid oxygen indicator is tested through the built-in-test (BIT) system. When the master test switch on the pilot's MASTER TEST panel is set to INST and depressed, the indicator pointer should read 2-liters, and the pilot's and NFO's OXY low caution lights illuminate.

OXYGEN CAUTION LIGHT

(U) An oxygen low, OXY LOW caution light is located on the pilot's and NFO's caution advisory panels, located forward on the right consoles. The amber light will come on when oxygen remaining is less than 2 liters or pressure downstream of the heat exchanger is less than 4.2 psi. When the light comes on, a cross check with the oxygen quantity indicator will disclose whether the oxygen quantity is low or pressure is low. The light is powered by the essential dc bus No. 2 through the OXY/FUEL LOW CAUTION circuit breaker.

OXYGEN DURATION CHART

MAN HOURS OF OXYGEN REMAINING–100%

WARNING

- CONSUMPTION RATES SHOWN ARE FOR 1 MAN
- WHEN TWO PERSONS ARE USING OXYGEN, DIVIDE THE NUMBER OF HOURS REMAINING BY TWO.

NOTE

- DURATION DATA SHOULD BE USED AS A GUIDE ONLY SINCE OXYGEN CONSUMPTION VARIES WITH THE INDIVIDUAL.
- CONVERSION OF LIQUID O_2 TO GASEOUS O_2 IS 860 LITERS OF GASEOUS TO 1 LITER OF LIQUID O_2
- CONSUMPTION RATES ARE BASED ON MIL-D-19326D

CABIN ALTITUDE	LITERS LIQUID O_2 20	15	10	5	4	3	2	1	LESS THAN 1
35,000 FEET and above	123.7	92.8	61.8	30.9	24.7	18.5	12.3	6.1	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
30,000 FEET	90.5	67.8	45.2	22.6	18.1	13.5	9.0	4.5	
25,000 FEET	69.9	52.4	34.9	17.4	13.9	10.4	6.9	3.4	
20,000 FEET	52.7	39.5	26.3	13.1	10.5	7.9	5.2	2.6	
15,000 FEET	42.4	31.8	21.2	10.6	8.4	6.3	4.2	2.1	
10,000 FEET	34.1	25.5	17.0	8.5	6.8	5.1	3.4	1.7	
8,000 FEET	31.0	23.2	15.4	7.7	6.2	4.6	3.1	1.5	
5,000 FEET	27.0	20.2	13.5	6.7	5.4	4.0	2.7	1.3	
SEA LEVEL	21.8	16.3	10.9	5.4	4.3	3.2	2.1	1.0	

0-F050-004
073-D

Figure 1-68.

EMERGENCY OXYGEN SYSTEM

(U) A limited supply of gaseous oxygen is available for ejection and for use in the event of failure or depletion of the normal system. An oxygen bottle is contained in the survival kit of each ejection seat. The bottle is charged to 1,800 to 2,100 psi and is activated automatically on ejection. Flow from the emergency bottle is routed through a pressure reducer where pressure is reduced to 50 to 90 psi. It then follows the path of the normal oxygen system, flowing through the oxygen regulator to the face mask. The supply of oxygen available from the emergency bottle is adequate for approximately ten minutes. An emergency oxygen gage is located under the right rear of the seat cushion.

Emergency Oxygen Handle

(U) The emergency oxygen handle is a green ring provided for use in the event of failure or depletion of the normal system. The handle is located under the left side of the ejection seat cushion. Pulling the handle releases the oxygen, providing approximately a fifteen minute supply. The duration will depend upon altitude, breathing rate, and regulator diluter setting.

PITOT-STATIC SYSTEM

(U) The pitot and static pressure system supplies impact (pitot) and atmospheric (static) pressure to the pilot's and NFO's flight instruments, to the air data computer (ADC), and to the engine air inlet control system programmers. Some systems require static pressure only; others require static and pitot pressure. Figure 1-69 illustrates those systems utilizing the pitot-static and other air sensor systems.

AIRSTREAM SENSORS

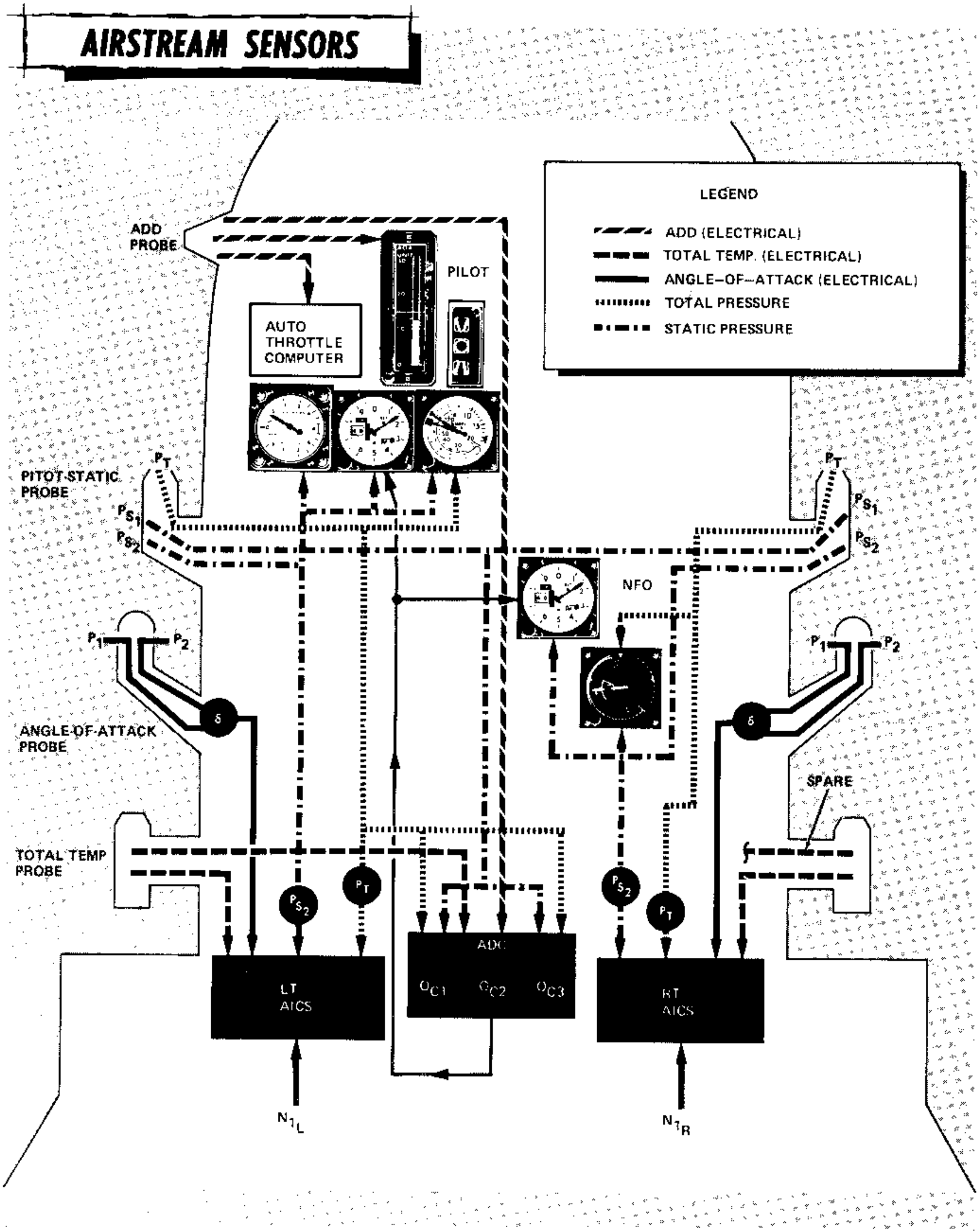


Figure 1-69.

(U) The pitot and static system is composed of two separate systems with individual pitot-static probes, one on each side of the forward fuselage. The left probe supplies the pilot's airspeed/Mach indicator, servoed barometric altimeter, vertical velocity indicator, and the left air inlet programmer. The right probe supplies the NFO's airspeed/Mach indicator, servoed barometric altimeter, and the left air inlet programmer. Both static probes and the left pitot probe direct pressure to the ADC where they are calibrated and corrected. This data is integrated and processed by the ADC to provide various instruments and systems requiring air data derived information.

(U) Each pitot-static probe is equipped with electrical heating elements to prevent icing. Pitot-static heat is controlled by the pilot through the ENG-PROBES ANTI ICE switch on the pilot's left console. In the AUTO position, pitot probe heat is available only with weight off wheels. The ORIDE position activates the probe heat elements independent of the weight on wheels switch and illuminates the INLET ICE caution light on the CAUTION-ADVISORY panel.

Note

> The ENG-PROBES ANTI ICE switch should only be positioned to ORIDE or AUTO during flight and ground operations when icing conditions exist or are anticipated. Engine anti-icing has adverse effects on engine stall during take-off and wave-off performance.

AIR DATA COMPUTER (ADC)

(U) The air data computer (ADC) is a dual digital computer capable of making yes and no decisions, solving mathematical problems (add, subtract, multiply and divide) and converting outputs to either digital or analog form as required by each aircraft system. It gathers, stores, and processes pitot pressure, static pressure, total temperature, and angle of attack data from the aircraft airstream sensors (figure 1-69). The computer also receives electrical feedback signals from servo outputs and actuator positions in the wing sweep and high lift systems. It performs schedule computations, limit control and electrical interlocks, failure detection, and system test logic. Major systems that depend on all or part of these ADC functions include AWCS, VDIG, data link, AFCS, barometric altimeter, engine fuel control system, power trim indicators, wing sweep control, maneuver flap control, glove vane scheduling, IFF, and environmental control system. A typical flow of data through the ADC is shown in figure 1-70.

(U) Computations are performed in two parallel and independent computing channels (I and II). Each channel is continuously monitored by on-line monitoring, cross channel comparisons with a third independent Qc sensor for system built-in-test (error detection) and self-test features. Computer outputs are normally provided from channel I. However, if the comparators between channel I and II or the independent Qc sensor detect a failure in channel I, and automatic channel switching is made to channel II. Both channels continue to perform identical computations, only the computer output source becomes channel II. If all three disagree, both channels I and II are disabled and the CADC caution light will illuminate. Nuisance failure interruptions are prevented by time delays in the error detectors. Transient failures in the ADC may be reset by momentarily depressing the MASTER RESET button on the pilot's left vertical console. This will reinstate channel I as the primary computer output source.

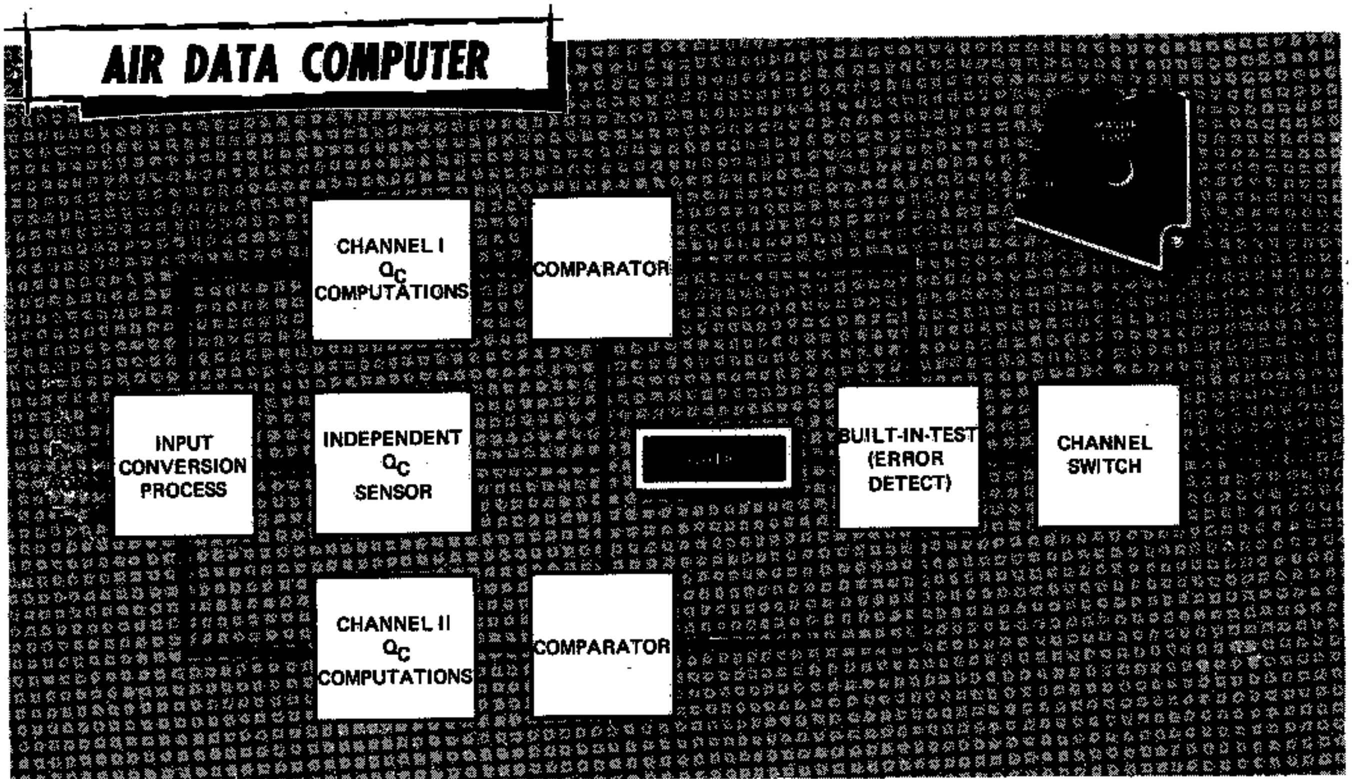


Figure 1-70.

0-F050-004
102-0

ADC Built-in-Test

(U) Built-in-test (BIT) capabilities provide continuous inflight monitoring of computer operations and servo outputs and actuator positions for the variable geometry systems. These BIT features consist of a combination of computer memory software (diagnostic program or test problems) and built in test equipment. Failures detected by the computer in the built-in-test processes can only be cleared if the detected fault disappears and the pilot initiates a manual reset command by pressing the MASTER RESET button.

(U) Computer data processing monitoring is accomplished by comparison of pitot (impact) pressure and also by a diagnostic program which is integral to the computer program. Each channel of the ADC contains a data processor which performs all arithmetic operations for that channel. During computer operation, each processor receives both air-data information and memory inputs. The memory data consists of constants and instructions that control the sequence of operations which the processor performs in computing air data outputs.

(U) By implementing test problems and instructions within the computer memory, the processor sections of the computer are tested. The test problems from the memory are supplied to the processor in the same manner as are the actual air data inputs, thus simulating actual computer functions during processor BIT mechanizations. At specific intervals during computer operation, a test problem is routed from the computer memory to the processor.

(U) Air-data inputs to the processor are inhibited at this time so that the processor inputs may be time-shared. When the test problem has been entered, the memory supplies instructions to the processor, which then computes the test problem by arithmetic computations and provides the computation as an output, which is used as an input to a comparator.

(U) The other comparator input is the actual solution to the test problem which is stored in the memory. When the processor computation is available, it is compared to the memory test problem solution. If a discrepancy exists longer than the period of the time delay, a failure signal is generated.

ADC Self-Test (OBC)

(U) The ADC self-test feature is part of the on-board check (OBC) process to test the computer operation. Self-test can only be performed when the aircraft is on the ground and must be manually commanded by selecting the OBC position on the master test panel.

(U) When self-test check of the computer is initiated, normal air data inputs are locked out and in their place constants from the computer memory are received. Failures detected in the computer are indicated by illumination of the NO GO light on the master test panel. Self-test detected failures must be manually reset by pressing the MASTER RESET button. With the landing gear down, the self-test feature is enabled.

FLIGHT INSTRUMENTS

ATTITUDE DIRECTOR INDICATOR (Aircraft 1, 2, 3, and 7 only)

(U) The attitude director indicator on pilot's instrument panel is installed in place of the vertical display indicator group. It displays pitch, roll and heading information on an attitude sphere in relation to a miniature aircraft. Pitch, roll and heading signals are received from attitude heading reference system (AHRS). The indicator receives 115V ac power from the essential ac bus. An OFF warning flag will appear on lower left face of the indicator, if power fails or AHRS has a malfunction. The glide slope (G/S) alarm flag has no operational function in this aircraft installation. A pitch trim knob on lower right side of instrument is used to adjust horizon line to correct for changes of aircraft pitch attitude. The turn and slip indicator is not operational.

STANDBY ATTITUDE INDICATOR

(U) A small (two inches diameter) standby attitude indicator on the right side of the pilot's instrument panel is for emergency use if the vertical display indicator attitude information is unreliable. In aircraft 1, 2, 3, and 7 the standby attitude indicator is a backup for the ADI. The indicator is a self contained vertical gyro which displays aircraft roll through 360 degrees and pitch through ± 92 degrees from the horizontal. Electrical power is supplied by the essential ac buses. A power failure during operation will cause an OFF flag to appear on the face of the indicator.

AIRSPEED/MACH INDICATOR

(U) The airspeed/Mach/indicator on the left side of pilot's instrument panel provides values of airspeed, Mach number, command airspeed, command Mach number, and maximum safe Mach number on a single presentation. Pneumatic pressures from pitot-static system and the air data computer activate airspeed and Mach mechanisms in indicator. Command Mach or command airspeed signals are provided by the digital data link (DL) system. The vertical display indicator group (VDIG) is provided with a command scale change from the airspeed/Mach number indicator. The range and derivation of each indicator display are as follows:

INDICATED AIRSPEED - 80 to 850 knots, is obtained from pitot-static-operated airspeed mechanism that drives airspeed pointer.

MACH NUMBER - 0.4 to 2.8 Mach, are derived from static-operated altitude mechanism that drives a moving scale (subdial) to indicate Mach number against airspeed pointer.

COMMAND AIRSPEED - 80 to 850 knots, and command Mach, 0.4 to 2.8 Mach, indications are obtained from a servo-driven index marker that presents Mach against airspeed (Mach dial). D/L power switch must be ON.

MAXIMUM SAFE MACH - at altitudes of 1,000 to 80,000 feet, is obtained from a servo-driven pointer that indicates maximum safe Mach against Mach subdial. The air data computer computes a maximum safe Mach number as a function of impact pressure corrected for standard day temperature deviations. When a safe Mach is exceeded, the REDUCE SPEED advisory light on the right side of the pilot's VDI will illuminate.

NOTE

When power failure occurs, the safe mach marker drives to the 12 o'clock position.

SERVOED BAROMETRIC ALTIMETER

(U) The servo barometric altimeter on left side of pilot's and the NFO's instrument panels displays accurate pressure altitude in feet. The altimeter is electrically operated by a synchro signal received from air data computer for normal operation. An integral standby mechanism, of the aneroid type, is incorporated and presents pressure altitude with normal barometric setting correction. The face of the altimeter is marked in 50-foot increments around the periphery of the dial and numerals ranging from 1 to 10 indicate the 100-foot increments. A single-dial pointer indicates correct altitude from 0 to 1,000 feet a complete revolution. A single drum-type counter, in the center of the instrument, also shows altitude in 100-foot increments. A dual digital counter to left of the 100-foot counter shows altitude in 1,000-foot increments. A four digital counter placarded IN. HG. is used in conjunction with an altimeter setting selector control on lower left side of the instrument.

(U) A selector level is on lower right corner of indicator and marked RESET/STBY. Normally the selector lever will be in the center position and the altimeter will receive its electrical inputs from the air data computer. However, if this system becomes unreliable, the altimeter will automatically revert to a standby mode which uses only inputs from the pitot static system. A failure warning flag marked STBY will appear on the dial face, indicating altimeter is in mechanical or standby mode of operation. The STBY flag will be hidden when the instrument is in normal electrical mode of operation. The electrical mode of operation may be selected by manually positioning the standby mechanism to RESET. Conversely, standby mode of operation may be selected by positioning the selector lever to STBY. Operating power is 115V ac through the essential No. 2 bus.

VERTICAL VELOCITY INDICATOR

(U) The vertical velocity indicator on left side of pilot's instrument panel is contained within a sealed case, and connected to a static pressure line through a calibrated leak. The reaction of a diaphragm inside the case to changing pressure is represented on face of indicator by a linkage system of gears and levers. The instrument will automatically compensate for changes in temperature. The immediate response of the diaphragm to atmospheric pressure is measured against trapped static pressure inside the case. When climbing or descending at a constant rate, a definite ratio between the diaphragm pressure and the case pressure is maintained through the calibrated leak. When aircraft is leveled, calibrated leak requires approximately 6 to 9 seconds to equalize two pressures, causing a lag in proper reading. When establishing a climb or descent, this lag is again apparent. Sudden or abrupt changes in attitude may cause erroneous indications due to sudden change of air flow over static probe.

TURN AND SLIP INDICATOR

(U) The turn and slip indicator on pilot's instrument panel gives information on rate of turn of aircraft around its vertical axis and turn coordination. The driving mechanism for the pointer is a dc meter movement receiving its input from a rate of turn transmitter. A needle-width deflection of pointer will initiate a 360-degree turn in 4 minutes. The inclinometer portion of the instrument contains damping fluid and a ball that moves from center in an uncoordinate turn.

ACCELEROMETER

(U) The accelerometer on pilot's instrument panel is a direct reading instrument used to measure the accelerations of the aircraft along its vertical axis. The dial is graduated in g units from -5 g's to +10 g's. The normal reading of the instrument at rest is +lg. The instrument has three pointers of which one continuously indicates vertical acceleration of aircraft. The other two pointers, one of which will stop and remain at maximum positive acceleration value attained, while the other will function in the same manner for negative acceleration values. These two pointers will remain at the highest values reached until reset by depressing a knob on lower left corner of instrument.

STANDBY COMPASS

(U) A conventional standby compass is above the pilot's instrument panel. It is a semi-float-type compass suspended in compass fluid. A pair of magnets attached to compass card align with earth's magnetic field to present magnetic heading indications.

CLOCK

(U) A mechanical clock is provided on the instrument panel in each cockpit. It is an 8-day clock and incorporates a 1-hour elapse time capability. A winding and setting selector is in lower left corner of instrument face. The knob is turned in a clockwise direction to wind the clock, and when selector is pulled out it is used to set hour and minute hands. An elapsed time selector in upper right corner controls elapsed time mechanism. This mechanism starts, stops, and resets the sweep second and elapsed time hands.

ANGLE-OF-ATTACK SYSTEM

(U) The angle-of-attack system measures the angle between the longitudinal axis of the aircraft and the relative wind. The indicating system provides a visual indication of angular position of wing chord in relation to aircraft flight path. This indication is used for approach monitoring and to warn of an approaching stall. Aerodynamically, angle-of-attack is airspeed. Optimum angle-of-attack is not affected by gross weight, bank angle, density altitude, or load configuration. Variations of these factors require pitch and throttle adjustments to maintain the angle of attack.

(U) The system includes a van-type transmitter, approach lights, an indicator, and an indexer. The indexer and the indicator are electrically slaved to the sensor vane transmitter. In flight, the vane which is on left side of the fuselage will align itself with the relative airflow like a weather vane. Rotation of the vane generates an indicated angle of attach signal to the angle of attack indicator, and also lights the angle-of-attack indexer and approach lights. A damper assembly prevents rotational overshoot and flutter of the vane due to turbulence.

(U) Probe anti-icing is provided by means of a 115 volt ac heating element in the leading edge of the vane. The heating element is controlled by the ENG-PROBES ANTI ICE switch on the pilot's left vertical console. Inflight probe heating is always activated regardless of the position of the ENG-PROBES ANTI ICE switch. During ground operation probe heat is on with the landing gear handle down and the switch in the AUTO or ORIDE position. With weight on wheels, the OFF position deactivates the probe heating element.

NOTE

The ENG-PROBES ANTI ICE switch should only be used during flight and ground operations when icing conditions exist or are anticipated. Engine anti-icing has adverse affects on engine stall during take-off and wave-off performance.

(U) A safety of flight check of the angle-of-attack indicator and other aircraft instruments can be performed while in flight or on the deck. When the INST position on the pilot's MASTER TEST panel is selected and the master test switch is depressed, the reference bar on the angle-of-attack indicator should indicate 18 units. A simular check of the indexer can be made by selecting the LTS position of the MASTER TEST panel. When the master test switch is depressed, the indexer and the GO indicator light on the MASTER TEST panel should light verifying proper continuity. If the indexer does not light and the NO GO indicator on the MASTER TEST panel lights, a malfunction exists in the circuit.

Angle-of Attack Indexer

(U) The angle-of-attack indexer on the pilot's glare shield (figure 1-71) has two arrows and a circle illuminated by colored lamps to provide approach information. The arrows are positioned vertically with the circle located between the two. The cam-operated switches in the angle of attack indicator also control the angle of attack indexer. The upper arrow is for high angle of attack (yellow), the lower arrow is for low angle of attack (red), and the circle is for optimum angle of attack (green). When both an arrow and a circle appear, an intermediate position is indicated. The indexer lights function only when the landing gear is down. A flasher unit causes the indexer lights to pulsate when an unsafe angle-of-attack exists, or the arresting hook is up with the HOOK BY-PASS switch in the CARRIER position. The intensity of the indexer lights is controlled by the INDEXER thumbwheel control on the pilot's MASTER LIGHT panel.

Angle-of-Attack Indicator

(U) This indicator (figure 1-71) displays the aircraft angle of attack (wing sweep 20 degrees), provides a stall warning reference marker, a climb bug, cruise bug, and an angle of attack approach reference bar for landing approach.

ANGLE OF ATTACK DISPLAYS

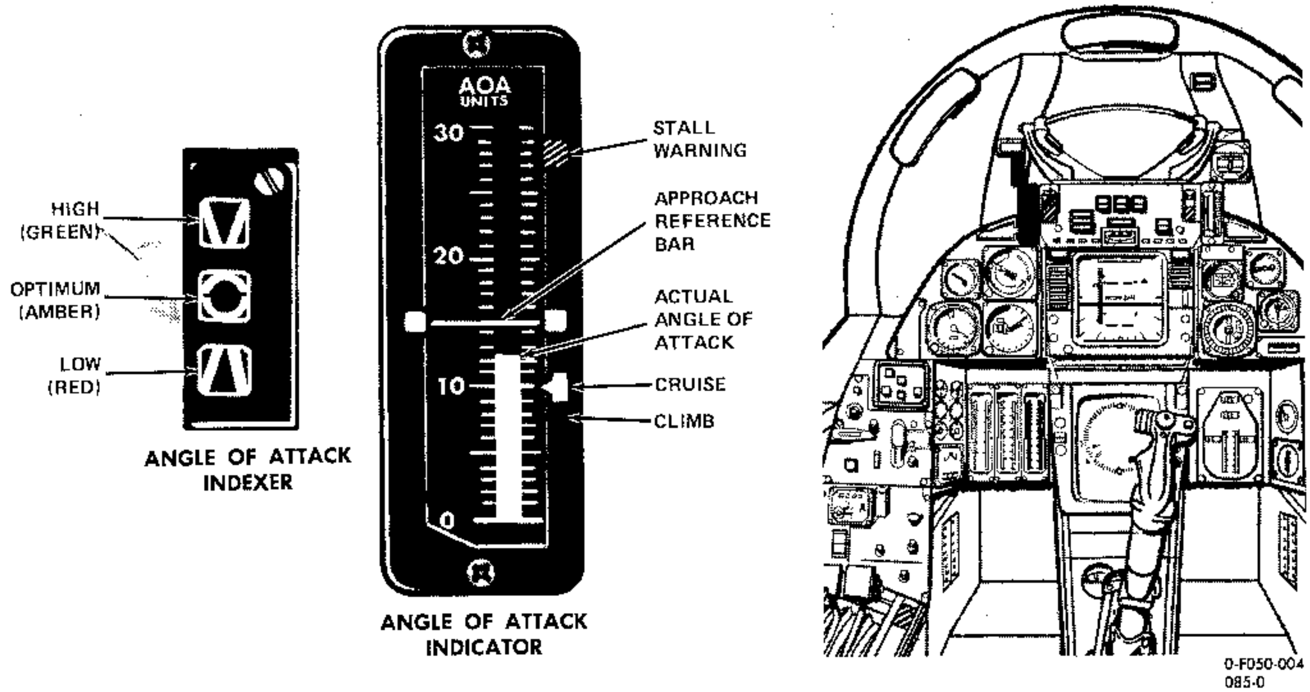


Figure 1-71.

(U) Angle of attack is displayed by a vertical tape on a calibrated scale from 0 to 30 units, equivalent ot a range of -10 to +40 angular degrees of rotation of the probe. The approach reference bar is provided for approach (on speed) angle of attack. This bar will position itself at 15 units during all normal operating modes when the landing gear handle is down and wing sweep is at 20 degrees. When direct lift control (DLC) is selected the approach reference bar will change its position to 18 units. The angle of attack indexer and approach lights will automatically follow the indicator schedule. If a failure occurs in DLC, the optimum approach bar will revert to 15 units and the indexer and approach lights will follow the 15 units optimum approach schedule. When the landing mode is selected with DLC engaged, the error symbol on the HUB display is faulty, and should be disregarded. When operating in DLC mode, use only the angle of attack indicator readings. The climb reference marker is set at 8.5 units, the cruise marker at 9.5 units, and the stall warning marker at 16 units. These reference markers are preset to the optimum angle of attack values and cannot be changed by the pilot.

Approach Lights

(U) The approach lights consists of red, amber, and green indicator lights above the nose gear strut. The lights are controlled by switches actuated by the angle of attack indicator and provides qualitative angle-of-attack information to the LSO during landing approaches. A flasher unit in the angle of attack system will cause the approach lights to pulsate when the arresting hook is up with the landing gear down and the HOOK BY-PASS switch in the CARRIER position. When the FIELD position of the HOOK BY-PASS switch is selected the flasher unit is disabled.

(U) A green approach light indicates a high angle-of-attack, low air speed; an amber light indicates optimum angle-of-attack; and a red approach light indicates a low angle-of-attack, high airspeed.

CANOPY SYSTEM

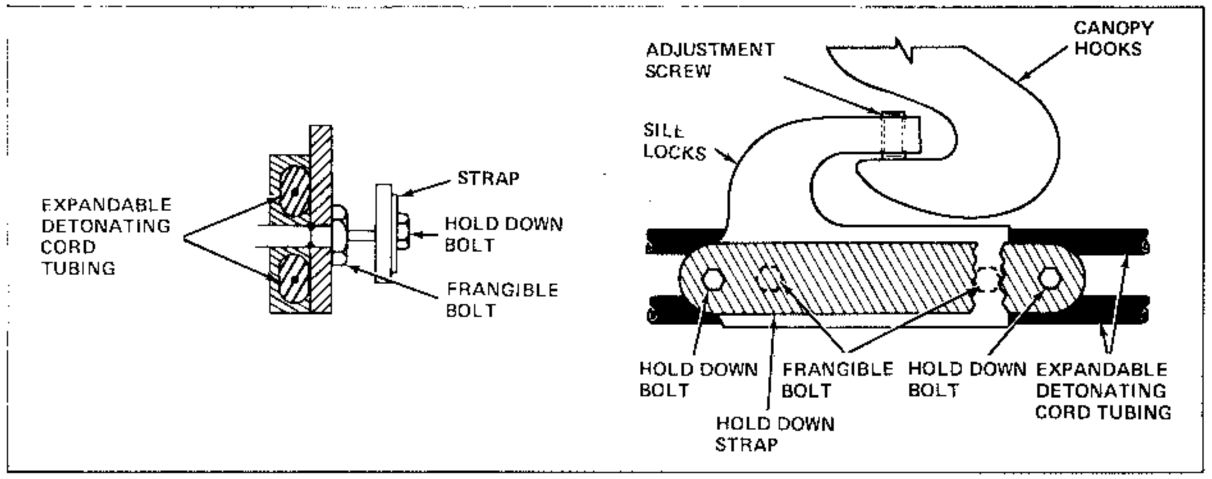
(U) The cockpit is enclosed by a one piece, clamshell, rear hinged, transparent canopy. It consists of two acrylic panels in a metal frame. Normal opening and closing of the canopy is provided by a pneumatic/hydraulic actuator, and a separate pneumatic actuator for locking and unlocking. The canopy can be opened to approximately 25 degrees for ingress and egress with a tailwind up to 60 knots. Approximately 8 to 10 seconds is required to complete a normal open or close cycle. In emergencies the canopy can be jettisoned from either crew position, or externally from either side of the forward fuselage. For rescue procedures, refer to Part 2, Section V, EMERGENCY PROCEDURES.

(U) The canopy system is controlled with the canopy control handle located under the right forward canopy sill at each crew position. An external canopy control handle is provided on the left side of the fuselage directly below the boarding ladder. A CANOPY caution light is provided on the caution-advisory panel at each crew position. Illumination of the light is a warning that the canopy is unlocked. The light is actuated by a switch connected to the mechanical lock lockpin on the canopy locking actuator. Electrical power for the caution light is supplied from the essential DC bus No. 2, through the CAN/LAD/CAUT/EJECT CMD IND circuit breaker.

(U) Pneumatic pressure for normal canopy operation is stored in a 225 cubic-inch high-pressure dry nitrogen reservoir. Servicing is accomplished externally through the nose wheel well. The filler valve is a common servicing point for both the canopy pneumatic reservoir and the emergency landing gear reservoir. Normal pressure should be serviced to 3,000 psi. A pressure gage is provided in the nose wheel well and should be checked by the aircrew during pre-flight inspection. A fully charged nitrogen bottle will provide approximately 10 complete cycles (open and close) of the canopy before the system is reduced to a minimum operating pressure of 225 psi. If pneumatic pressure drops below 225 psi, the canopy control module automatically prevents further depletion of the main reservoir and the canopy must be opened manually. To assist opening the canopy manually an auxiliary pneumatic bottle is connected to the unlooking actuator to move the canopy aft out of the locks. The pneumatic pressure remaining in the normal system acts against the transfer cylinder piston to counterbalance the weight of the canopy allowing the aircrew to raise the unlocked canopy. The canopy can be opened and closed with nominal effort from either aircrew position. However, when the auxiliary unlocking bottle is activated, the canopy cannot be locked closed until the auxiliary pneumatic unlock valve is reset, even after the main canopy reservoir is recharged. The auxiliary canopy pneumatic bottle, recharge valve, and reservoir gage are located below inspections plates behind the turtle deck.

(U) The canopy actuators are supplied metered nitrogen pressure through a selector module and associated timing valves, which control the positioning of the canopy. Figure 1-72 illustrates the canopy pneumatic and pyrotechnic systems. Under normal operation conditions (30 knots headwind), the selector module regulates pressure to 325 psi. When required by high loads (30 to 60 knots headwind), the selector module will increase pressure to 790 psi.

CANOPY PNEUMATIC AND PYROTECHNIC SYSTEMS



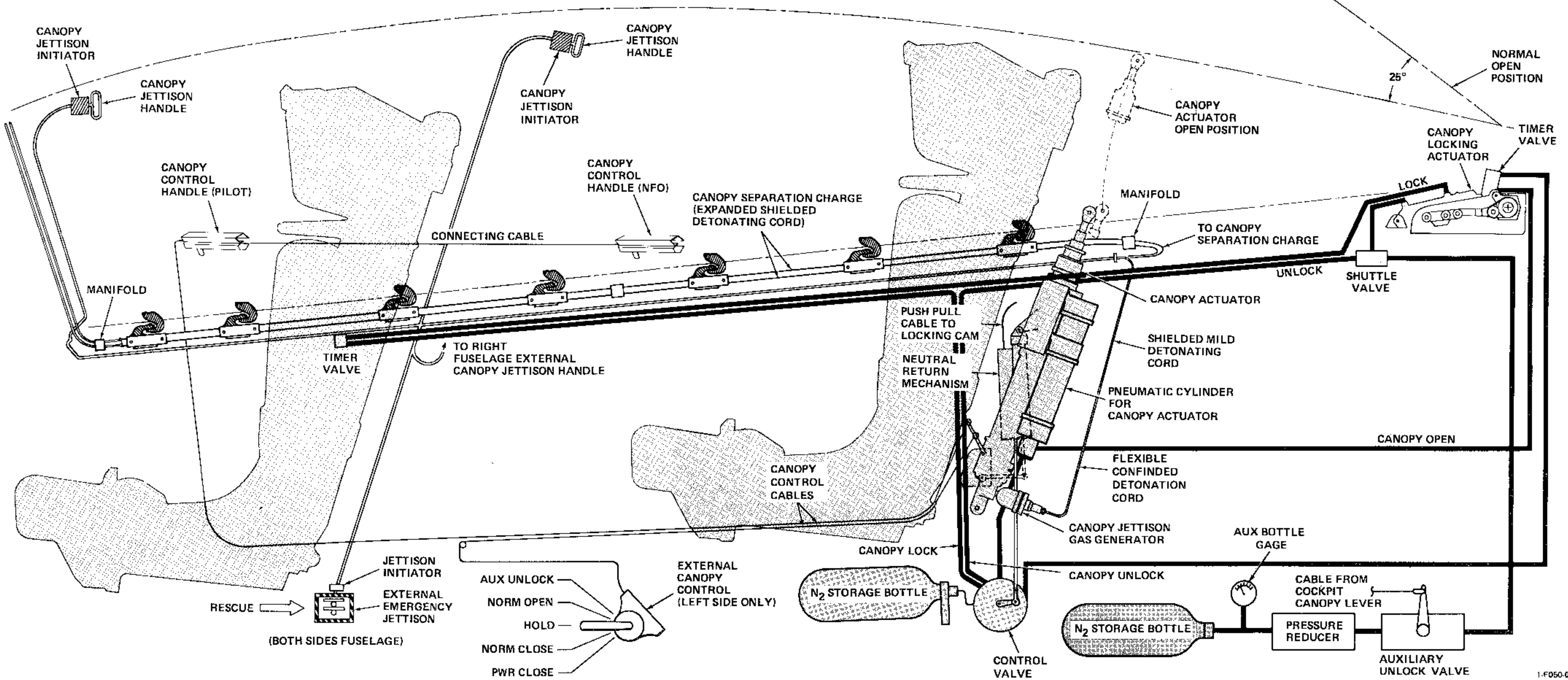


Figure 1-72.

The canopy actuator is a double acting hydraulic cylinder with two transfer cylinders. Pneumatic pressure admitted to the air side of the transfer cylinder, causes the actuator to extend or retract which opens or closes the canopy. The locking actuator is a double acting pneumatic cylinder which unlocks the canopy by sliding it aft out of the fixed sill locks, or forward into the sill locks. When the canopy is closed and locked, a rubber seal in the canopy frame inflated by conditioned air from the environmental control system, forms a pressure-tight fit with the canopy sill. If the canopy is separated by emergency jettisoning, the canopy locks are broken by explosive charges and the canopy is jettisioned by a cartridge actuator piston built into the actuator piston rod.

Normal Canopy Operation

(U) The canopy pneumatic/hydraulic system is operated by actuation of either of the cockpit control handles (figure 1-73), or the external control handle, which positions valves within the pneumatic control module to open or close the canopy. Normal modes of operation available are: NORM OPEN, AUXILIARY UPLOCK, HOLD, NORM CLOSE, and BOOST CLOSE.

(U) When the NORM OPEN position is selected, 325 psi of nitrogen is ported to the locking actuator through the control module. By means of a torque tube, cranks, and links, the canopy is moved aft disengaging the canopy hooks from the sill locks. Pneumatic pressure is then ported to the canopy actuator to raise the canopy.

(U) In all operating modes, the control handle must be returned to the HOLD position. With the canopy open and the handle in the HOLD position, hydraulic lock valves within the canopy actuator hold the canopy in any desired position. However, when the canopy is closed and the control handle is in the HOLD position the lock valves are open. This feature ensures that all of the flight loads are taken by the canopy locks and that no loads are taken by the canopy actuator.

(U) The NORM CLOSE position allows the canopy to close under normal conditions, (30 knots headwind) using its own weight without an expediture of stored nitrogen. When the control handle is set to NORM CLOSE, both sides of the canopy actuator are vented to the atmosphere allowing the canopy to lower itself. The final closing motion actuates a pneumatic timer which directs 325 psi pressure from the control module to the locking actuator. Through a torque tube, cranks, and links, the canopy is moved forward to engage the canopy hooks in the sill locks.

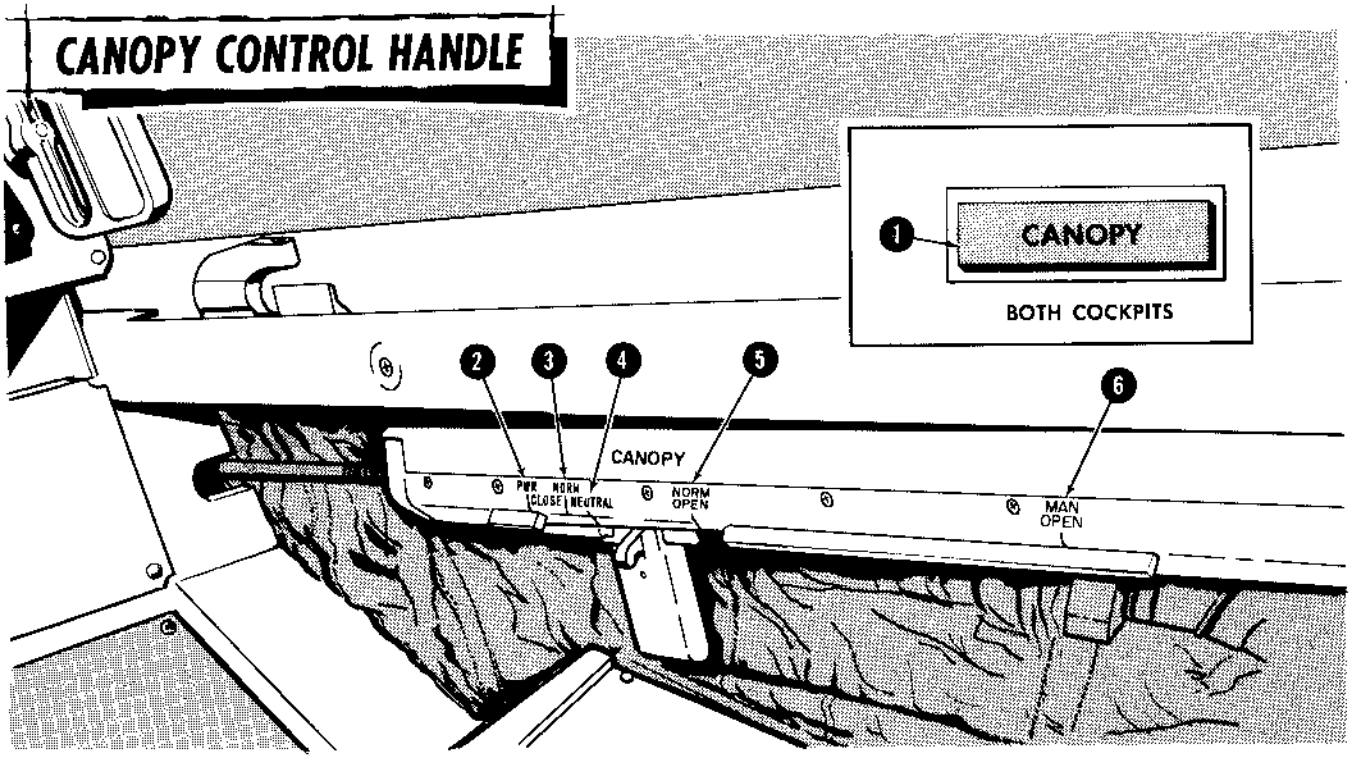
(U) To close the canopy under high headwind conditions (30 to 60 knots) the BOOST CLOSE position is provided. With the control handle in this position, the control module ports 790 psi regulated pneumatic pressure to the close side of the canopy actuator. Hydraulic fluid on the opposite side of the transfer cylinder piston causes the actuator to retract closing the canopy. Hydraulic flow regulators in the canopy actuator control the opening or closing time which is approximately 8 to 10 seconds under all load conditions.

Auxiliary Canopy Opening

(U) When the main pneumatic reservoir pressure is reduced to 225 psi, the canopy control module automatically prevents further depletion of reservoir pressure and the canopy must be opened manually. Actuation of the auxiliary mode can be effected from either the pilot's or NFO's canopy control handle or from the ground external canopy control. To open the canopy in this mode, a latch on the control handle must be depressed to move the handle past the NORM OPEN detent to the AUXILIARY UNLOCK position. This actuates a pneumatic valve, which admits regulated pneumatic pressure at 325 psi from an auxiliary N_2 bottle to the locking actuator and moves the canopy aft out of the sill locks. When the canopy is unlocked, pneumatic pressure from the main reservoir is ported to the open side of the canopy actuator to counter-balance the weight of the canopy allowing the canopy to be manually opened or closed. The canopy can be positioned at any intermediate position by returning the control handle to the HOLD position.

(U) Before exiting the cockpit the control handle should be returned to the HOLD position. If left in the AUX OPEN position the canopy's own weight or a tailwind could force the canopy down with low pressure in the main reservoir. The canopy control handle does not automatically return to the HOLD position from AUXILIARY UNLOCK. To return the handle to HOLD, the latch must be depressed to release the handle from the AUXILIARY UNLOCK detent. Once the auxiliary canopy unlock bottle is used, the canopy will not return to the normal mode of operation, and cannot be locked closed until the auxiliary pneumatic selector valve on the aft canopy deck is manually reset (lever in vertical position).

(U) The auxiliary canopy N_2 bottle is located on the turtleback behind the canopy hinge line (figure 1-72). Servicing of the auxiliary bottle is performed through the small access panel immediately behind the canopy on this turtleback. A fully charged bottle will provide approximately 20 operations in the auxiliary open mode.



0-F 050 004
050-0

Figure 1-73

NOMENCLATURE		FUNCTION
1	CANOPY CAUTION Light	Advises crew that the canopy is not in a down and locked position. Lights in both cockpits.
2	BOOST CLOSE	Used to close the canopy when headwinds are greater than 30 to 60 knots.
3	NORM CLOSE	Used under normal conditions (headwinds 30 kts. or less).
4	HOLD	Used when the canopy is not being operated or to stop the canopy at an intermediate position.
5	NORM OPEN	Used to open the canopy under normal conditions.
6	AUX OPEN	Used to open canopy manually which is required when nitrogen bottle pressure drops below 225 psi.

EXTERNAL CANOPY CONTROLS

(U) Access to the external canopy control is obtained through an access door located on the left fuselage directly below the boarding ladder. Pulling the handle out to a first detent and rotating it counter-clockwise closes the canopy while clockwise opens it under normal operating conditions. Pulling the handle out to a second detent and rotating it counter-clockwise, performs the POWER position functions and clockwise, performs the MANUAL position functions.

CANOPY JETTISON

(U) The canopy can be jettisoned from either cockpit or from external controls on each side of the fuselage. An internal control handle in each cockpit is located on the forward right side of each crewmember's instrument panel, is painted yellow and black for ease of identification. Two external control handles, one on either side of the fuselage directly below the pilot's cockpit, are appropriately marked with operating instructions (Refer to Section V, EMERGENCY PROCEDURES). The handles are a squeeze/pull pull type mechanism which comes free of the initiator when actuated. Each handle is attached directly to its own individual initiator. Actuation of an initiator, ignites a shielded mild detonating cord which follows a path to an expanding shielded detonating cord that severs the sill locks, releasing the canopy hooks. Simultaneously, the shielded mild detonating cord ignites a gas generator at the base of the canopy actuator which pressurizes the outer barrel of the canopy actuator, shearing the connection between the barrel and inner housing of the actuator, in turn propelling the inner housing, piston, and canopy away from the aircraft. In addition to the four canopy jettison handles, the canopy is jettisoned automatically prior to ejection during an ejection cycle.

(U) Ejection is not performed through the canopy, therefore, the canopy is jettisoned as a part of the normal ejection sequence. A downward pull on the face curtain or an upward pull on the alternate ejection handle will jettison the canopy prior to ejection. The canopy and ejection seat jettison system is illustrated in figure 1-50.

INTERNAL CANOPY JETTISON HANDLES

(U) There are two internal canopy jettison handles, located on the right side forward in each cockpit, painted with yellow and black stripes for ease of identification. Each jettison handle has its own initiator but the function of either handle is the same. When either handle is pulled an initiator ignites the detonating cords and canopy thrust gas generator which will jettison the canopy from the aircraft. The jettison control handles require squeezing the outer face of the handle then pull for actuation. The length of pull is approximately 1/2 to 3/4 inch and comes free of the aircraft when actuated.

EXTERNAL JETTISON CANOPY HANDLES

(U) There are two external emergency jettison handles located on the lower left and right fuselage below the pilot's cockpit, appropriately marked for rescue. Opening the access door and pulling either T handle fires an initiator (each has its own initiator) which detonates the shielded mild detonating cord which in turn detonates the expanding detonating cord severing the sill locks and disengaging the canopy. Simultaneously, the detonating cord initiates the canopy thruster generator, jettisoning the canopy from the aircraft. The jettison control handles require squeezing the inner face of the handle and then pulling for actuation. The length of pull is approximately 1/2 to 3/4 inch and the T handle comes free on the aircraft when actuated. See Section V, Part 2, GROUND EMERGENCIES for canopy external jettisoning procedures.

EJECTION SYSTEM

(U) The aircraft is equipped with an automatic sequenced command escape system incorporating Martin-Baker MK GRU-7A ejection seats. When either crew member initiates the command escape system, the canopy is ballistically jettisoned and each crew member is ejected in a preset time sequence. Ejection trajectories are canted laterally to provide additional separation of the seats. The NFO is ejected to the right and the pilot to the left.

(U) A command ejection lever above the left outboard console allows the NFO to select either pilot or NFO control of the command ejection system. An EJECT COMD flip-flop type indicator on the landing gear panel indicates the command mode selected. The NFO may eject individually when the command ejection lever is in the pilot control position. When the command ejection lever is in the NFO command position, the NFO can initiate ejection of both seats. Regardless of the position of the command ejection lever, an ejection initiated by the pilot will always eject the NFO. Command ejection by either crew member will eject the NFO first and the pilot 0.4 second later.

(U) The ejection system is capable of safe ejection on the deck at zero airspeed. Preflight procedures and location of safety pins are shown in Section III of this manual, ejection procedures are discussed in Section V. The canopy and seat jettison system is illustrated in figure 1-74.

MK GRU-7A EJECTION SEAT

(U) The MK GRU-7A ejection seat (figure 1-75 sheets 1 and 2) consists of a main beam assembly, two vertical and three horizontal members, plus the following components and assemblies: a seat-stabilizer drogue-parachute assembly within the headrest; primary firing mechanism, and attached face curtain; alternate firing handle; parachute withdrawal-line guillotine; drogue gun; time-release mechanism; ballistically powered inertia-reel mechanism and dual attached leg-restraint cord assemblies; ventilated back pad; ventilated seat pad; RSSK-7 survival kit containing an emergency oxygen bottle, and the seat

CANOPY/SEAT JETTISON SYSTEM

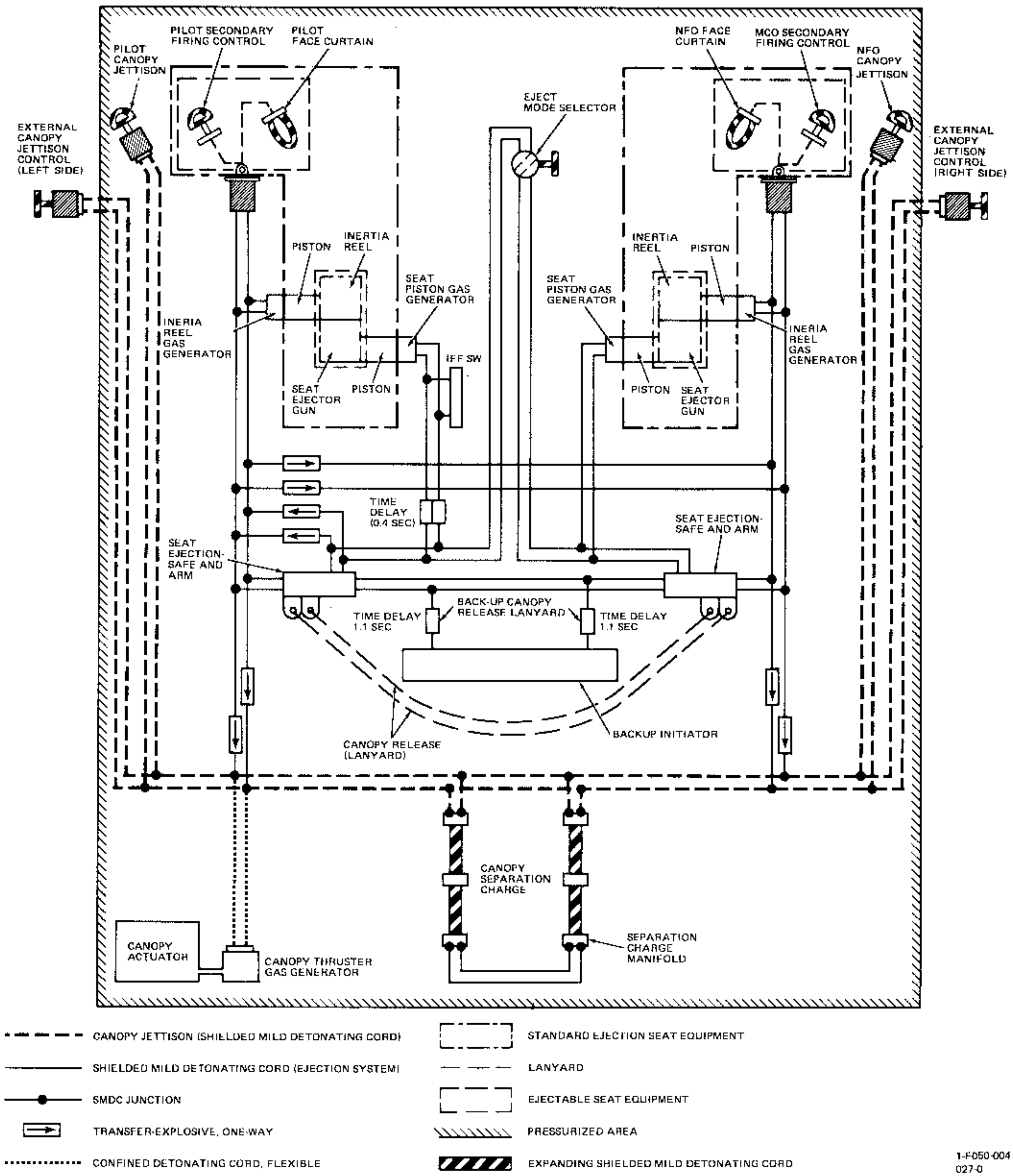


Figure 1–74

EJECTION SEAT

MK GRU 7A

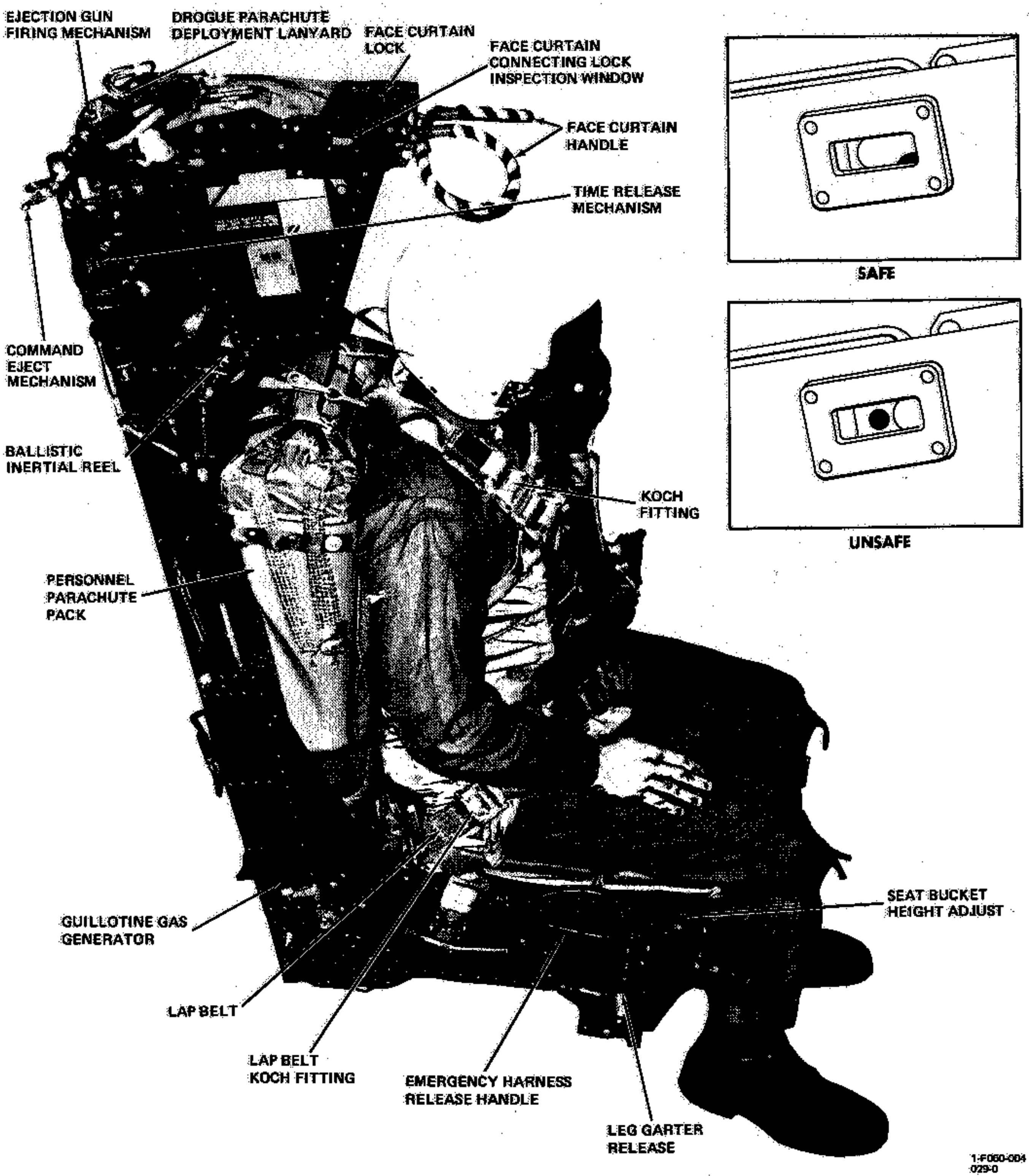


Figure 1-75 (Sheet 1)

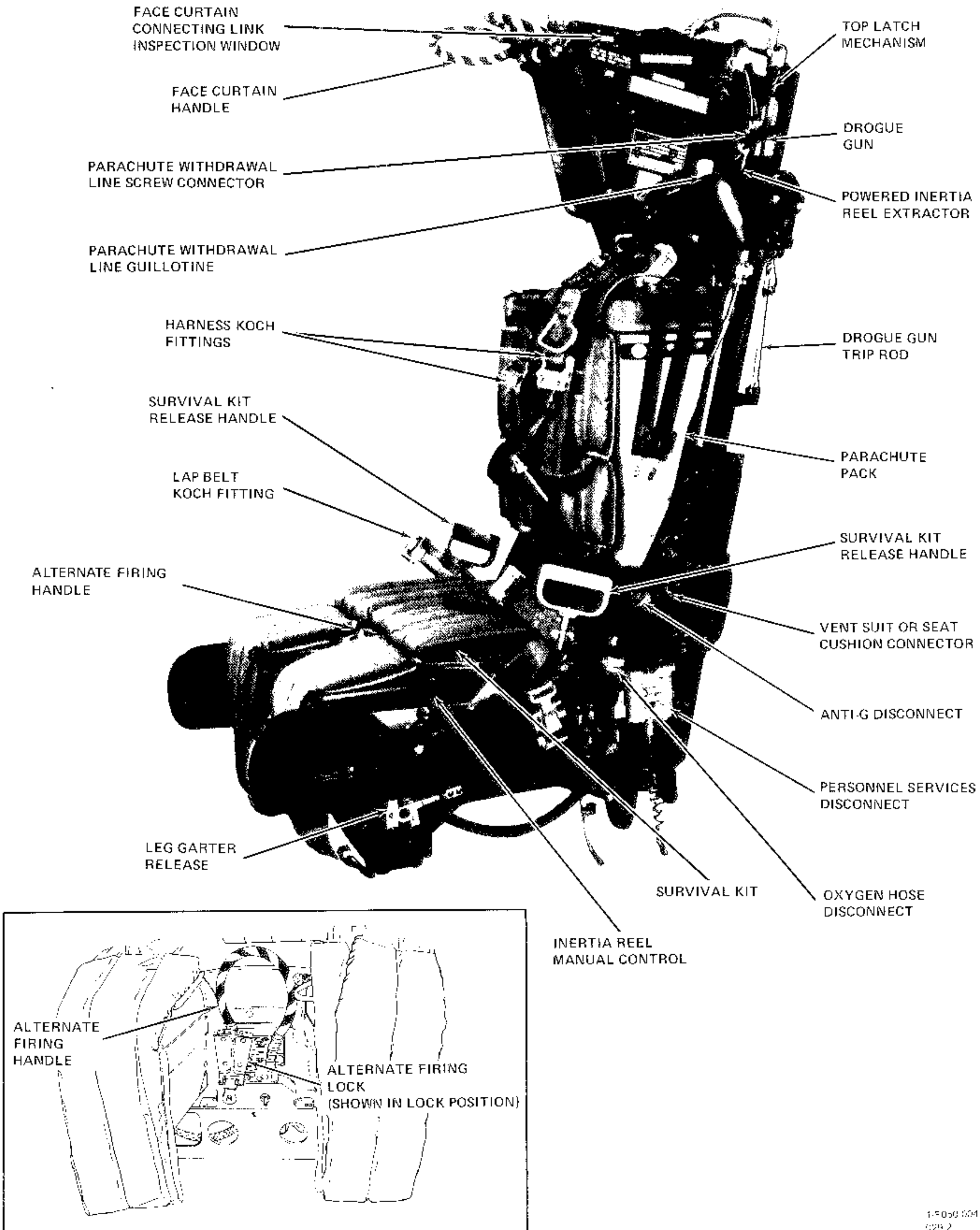


Figure 1-75 (Sheet 2)

bucket height-adjustment actuator. Each seat contains an emergency harness release handle, a 28 foot personal parachute and container, a rocket motor triggered by a gas generator, and a sear extractor that is actuated by the command eject gas-generating initiator.

(U) Ballistically powered inertia reels on each seat are triggered by a gas generating initiator which are actuated by pyrotechnics to ensure that the crewmember is pulled back into the seat prior to ejection. The pyrotechnic train of the pilot's seat actuates the IFF transponder and ECM destruct switch.

EJECTION SEAT OPERATION

(U) The ejection sequence is initiated by pulling either the face-curtain firing handle or the alternate firing handle. When either firing handle is pulled the ejection pyrotechnic system (figure 1-74) is actuated and the following sequence of events occur:

- The seat shoulder harness power retraction reel is fired.
- The canopy hooks are cut and the canopy is ballistically jettisoned.
- The seat safe and arm device is armed.

(U) When canopy jettison is complete the safe and arm device is fired by an 8 foot canopy release lanyard attached to the canopy. During a command sequence ejection this pyrotechnic train return fire instantaneously triggers the NFO's seat and the pilot's seat is delayed 0.4 second. To circumvent failure of the 8 foot canopy release lanyard to fire the safe and arm device, a 10 inch backup canopy release lanyard attached to the canopy activates an initiator and a 1.1 second time delay as a backup to fire the safe and arm device.

(U) The powered inertia reel pulls the occupant back into the seat to an upright position within 0.25 second. When catapult gun fires, the expanding gases drive the catapult tubes upward and eject the seat and occupant from the aircraft. As the seat rises, the occupant's legs are pulled back into the leg pads and the restraint cords are pulled tight. As the seat continues to rise, tension built up in the restraint cords shear rivets (securing the cords to the deck), freeing the cords from the aircraft. The occupant's legs remain secured to the seat by snubbers wedging against the restraint cord, preventing any flailing of the occupant's legs. The occupant's legs remain secured to the seat until the time-release mechanism actuates a plunger freeing the occupant's legs prior to separation from the seat. While the seat is going up the rail, the following events occur:

- The drogue-gun trip rod is pulled.
- The time-release mechanism trip rod is pulled.
- The emergency oxygen bottle is activated.
- The rocket motor gas generator lanyard is pulled.

(U) When fully extended, the rocket motor lanyard fires the gas generator which fires the rocket motor under the seat.

SEAT CATAPULT/ROCKET FIRING

(U) One-half second after the drogue-gun trip rod is extracted, a small controller drogue chute is drawn from the pack by a metal piston fired from the drogue gun. The controller drogue chute then tows the stabilizer chute out of its container. The stabilizer chute is secured to the seat by a scissor shackle until released by the time-release mechanism.

(U) During ejection, the occupant is held in the seat by upper restraint straps, lap belt, and leg restraint cords. Deployment of the personnel parachute and separation of the occupant from the seat are delayed by the time-release mechanism until the occupant has descended from the upper atmosphere, or until expiration of a 2.0 second time daley.

(U) When the seat is below an altitude of 11, 500 to 14, 500 (barostat setting), the time-release mechanism starts a 2.0-second time delay, after which the time-release mechanism releases the drogue parachute from the scissors shackle, allowing the drogue chute to pull the personnel parachute withdrawal line out of the guillotine and deploy the personnel parachute. At the same time, the harness restraints, lap belt, leg restraints, survival kit, upper block of the personnel services disconnect, and face blind are unlocked. The occupant is then free to be pulled from the seat sticker clips by the line stretch of the main parachute as the seat rotates away. When the occupant is free from the seat, he makes a normal parachute descent with his (RSSK-7) survival kit connected to him by the integrated torso harness.

MANUAL SEAT SEPARATION

(U) If the time-release mechanism fails, the crewmember can manually separate from the seat by pulling up and aft on the emergency harness release handle (front right side of seat bucket). Rotating the emergency harness release handle fires the guillotine severing the personnel parachute withdrawal line and releasing the lap belt, the leg-line straps, the torso harness, parachute container, survival kit, and personnel services disconnect. The occupant must kick free of the seat and manually deploy the personnel parachute by pulling the D-ring.

WARNING

The personnel parachute must be deployed manually if the emergency harness release handle has been pulled.

EJECTION CONTROLS

Face-Curtain Handle

(U) The automatic ejection sequence is initiated by pulling the face-curtain handle located immediately aft and above the occupant's head. When the face-curtain handle is pulled, the system initiator is mechanically actuated. (After the time delay, the primary cartridge is fired.) As the pilot's seat rises, the drogue-gun and time-release mechanism sears are extracted by trip rods, the emergency IFF transponder and ECM destruct switch are activated.

NOTE

The seat is fired after approximately 7 inches of face-curtain travel.

Alternate Firing Handle

(U) The alternate firing handle is on the front of the seat bucket, between the occupant's legs. The sequence of operation remains the same, since both the face-curtain handle and the alternate firing handle initiate seat firing by mechanically actuating the escape system initiator.

NOTE

- If the face-curtain handle does not actuate the ejection seat, the face-curtain handle should be held while the alternate firing handle is actuated, to prevent the possibility of entanglement with the drogue gun when it fires.

- The seat initiator is fired after 2 inches of alternate firing handle travel.

PARACHUTE

(U) The personnel parachute is contained in a rectangular back pack aft of the occupant. It consists of a 28 foot canopy, suspension lines, and personnel harness connections. The parachute pack is attached to the seat by two lock in fittings. On ejection, the locks are automatically released by the time-release mechanism at both points. The parachute pack is attached to the occupant by two straps connected to the top Koch fittings of the integrated torso harness.

TIME-RELEASE MECHANISM

(U) The time-release mechanism is on the right side of the ejection-seat headrest and is armed upon ejection by a time-release mechanism trip rod which is secured to the trip rod bracket.

(U) The time-release mechanism contains an altitude-sensing barostat to prevent premature deployment of the parachute at high altitudes. At approximately 13,000 (±1,500) feet or less, the time-release mechanism releases the drogue parachute from the scissors shackle, allowing the personnel parachute withdrawal line out of the guillotine to deploy the occupant's personnel parachute. At the same time, the torso harness restraints, lap belt, leg restraints, upper block of the personnel services disconnect, survival kit, parachute pack and face-curtain are unlocked.

(U) The occupant is then free to be pulled from the seat sticker clips by the line stretch of the main parachute.

PARACHUTE WITHDRAWAL LINE

(U) The parachute withdrawal line is connected from the personnel parachute to the seat drogue parachute with an acorn screw connector. In normal ejection, this line is pulled out by the drogue parachute after the time-release mechanism fires to deploy the personnel parachute. (The line is routed through a ballistic guillotine that severs the line for manual seat separation).

Guillotine

(U) The guillotine on the left side of the seat is fired by the initial travel of the harness equipment release handle. The guillotine initiator is mounted on the right aft side of the seat bucket. This severs the personnel parachute from the seat and drogue parachute; the personnel parachute must then be manually deployed by pulling the D-ring.

EMERGENCY HARNESS RELEASE HANDLE

(EMERGENCY OVERRIDE LEVER)

(U) This control is the black-and-yellow-striped handle forward on the right side of the seat bucket. In the forward (locked) position, the occupant and his personal equipment are secured to the seat. When the emergency harness release lever is lifted and rotated to the aft (unlocked) position, the occupant and his personal equipment are disconnected from the seat except for the sticker clips. The initial travel of this handle fires the parachute withdrawal-line guillotine, and severing the personnel parachute from the drogue parachute. The personnel parachute must be manually deployed by pulling the D-ring. The emergency harness release handle simultaneously releases the lap-belt harness locks, personal parachute container retention straps, survival kit attachment lugs and sticker clips, inertia-reel straps, dual leg-restraint cords, and personnel services disconnect.

WARNING

When the emergency harness release handle is actuated, the face-curtain handle and the alternate firing handle are automatically locked and ejection cannot be accomplished. There is no provision for resetting this control in flight.

(U) This control will normally be in the forward position. Emergencies such as ditching, over-the-side bailout, or failure of the time-release mechanism will require operation of this control.

POWERED INERTIA REEL

(U) Each seat has a ballistically powered inertia reel that is fired when ejection is initiated. The initiator is fired by a gas generator which extracts the sear pin. The burn time is set to position the occupant to an upright position prior to ejection (less than 0.25 second) with little or no shock. The inertia-reel straps are connected to the parachute risers and to the upper torso harness.

Inertia-Reel Manual Control

(U) This control is on the left side of the seat. In the forward position, forward movement of the occupant is restricted and any slack created by rearward movement is taken up by the inertia reel. The control is locked in this position by using the detent. In the spring-loaded center position, the occupant can move forward freely, unless the reel locks owing to excessive forward velocity. When the forward velocity decreases sufficiently, the inertia straps are released without the necessity of repositioning the manual control. Both straps feed from the same shaft and it is impossible for one to lock without the other. If the reel is locked manually, the control must be positioned full aft to the unlock position to release the straps.

LEG GARTER

(U) The leg garter and restraint cords are used to keep the occupant's legs firmly against the leg rests during ejection. The garter is placed around the leg above the calf and above the ankle. It should be tight enough so it does not slip down over the calf.

(U) The leg-restraint cords are attached to the aircraft deck and routed through the snubber box seat structure. They are then passed through garter rings and snapped into the leg line locks. The garter rings are snapped into the bayonet fitting when strapping in. Leg-line snubber release is accomplished by pulling the release lever located on the outer side of the leg-line snubber boxes.

PERSONNEL SERVICES DISCONNECTS

Oxygen Hose Disconnect

(U) The oxygen hose disconnect is located on the left side of each seat bucket. The disconnect terminates in a block that also terminates the aircraft oxygen system line. The disconnect separates from the seat at ejection, actuates the emergency oxygen system, and disconnects the communications line from the seat occupant.

Vent Air/Anti-G Disconnect

(U) A vent line (exposure suit and back pad) and anti-g line personnel services disconnect is mounted on the left side of the seat. The lower block of the disconnect is separated by the extended lanyard. The upper block is released prior to seat separation by the emergency harness release handle actuation of the seat release mechanism. The disconnect is unlocked by the extended lanyard tripping the release mechanism and pulled free when the occupant separates from the seat upon deployment of the personnel parachute.

SEAT ADJUSTMENT

(U) Seat adjustment is controlled by a three-position, momentary-contact switch on the right side of the right thigh support. Moving the switch forward (FWD) lowers the seat and the AFT position raises the seat. Seat adjustment is limited to 5 inches of vertical movement.

CAUTION

> The seat height-adjustment actuator is an intermittent-duty motor with duty cycle of 30 seconds on and 10 minutes off.

SURVIVAL KIT (RSSK-7)

(U) An RSSK-7 survival kit is in the seat bucket of each crewman's ejection seat. The upper half of the kit is a contoured lid assembly covered by a ventilated seat pad and contains an emergency oxygen bottle, which can be activated manually or automatically by seat ejection. The lower half of the kit contains the following:

- Life Raft
- Disalter Kit
- Water Storage Bag
- Survival Radio
- Signal Mirror
- Signal Flares
- Morse Code Card
- Dye Markers
- Bailing Sponge
- Sunburn Ointment
- Shark Repellant
- Nylon Cord
- Rations

(U) After the seat clears the aircraft during the ejection sequence, a time-release mechanism releases the crewman's harness and survival kit from the seat. During descent, when the crewman pulls the survival kit release handle (one on each side of Kit), the lower half of the kit falls away but remains connected to the upper half by a lanyard approximately 23 feet long. As the lower half reaches the end of its free fall, the life raft inflation mechanism is actuated and the raft inflates.

(U) When ejecting over water, the life raft should be inflated before entering the water. If the crewman enters the water before the survival kit release handle is pulled, the life raft can only be inflated by first pulling the release handle and then pulling the life raft lanyard until the raft is inflated.

LIGHTING SYSTEM

EXTERIOR LIGHTS

(U) The exterior lights include position lights, formation lights, anti-collision lights, a taxi light, and an air refueling probe light. All exterior lighting except the air refueling probe light is controlled by the master light panel on the pilot's right console, and the exterior lights master switch on the outboard throttle. The pilot's light control panel is illustrated in figure 1-76 and the NFO's light control panel in figure 1-77.

Position Lights

(U) The position lights consist of a red light on the left wing tip, a green light on the right wing tip, and a white light on the aft center of the fuselage. Supplement position lights include upper and lower red lights on the left wing glove vane, and upper and lower green lights on the right wing glove vane. When the wing sweep angle is forward of 25 degrees the wing tip position lights are operational. When the wings are swept aft of 25 degrees the wing tip position lights are disabled and the supplement position lights are operational. The reverse will occur when the wings are swept forward of 25 degrees. The position lights are powered by the No. 2 essential bus through circuit protection on the NFO circuit breaker panel.

NOTE

> When the anti-collision lights are on, the flasher unit for the position lights is disabled.

ANTI-COLLISION LIGHTS

(U) There are three red, rotating-type anti-collision lights. One anti-collision light is installed in the bottom of the IR pod located on the lower forward fuselage. This provides a flashing beacon directed forward and down. Another anti-collision light is installed in the top forward part of the left vertical stabilizer and directs its beacon forward and above the aircraft. The third anti-collision light is installed on the top aft part of the right vertical stabilizer and directs its anti-collision beacon up and down.

(U) The anti-collision lights are powered through the right main bus with circuit protection of the NFO's ac right main circuit breaker panel.

FORMATION LIGHTS

(U) The formation lights consist of wing tip lights on each wing, and forward and aft fuselage lights on both sides of the aircraft. All formation lights are green. Intensity of the lights are controlled by a thumbwheel on the master light panel. Electrical power is supplied through the right main bus with circuit protection on the NFO's ac right main circuit breaker panel.

TAXI LIGHT

(U) The taxi light installed on the nose wheel is a fixed position light. A limit switch on the nose gear door will turn the light off when the gear is retracted. A two position, ON/OFF toggle switch is on the master light panel. Electrical power is supplied through the right main bus with circuit protection on the NFO's ac right main circuit breaker panel.

PILOT'S LIGHT CONTROLS

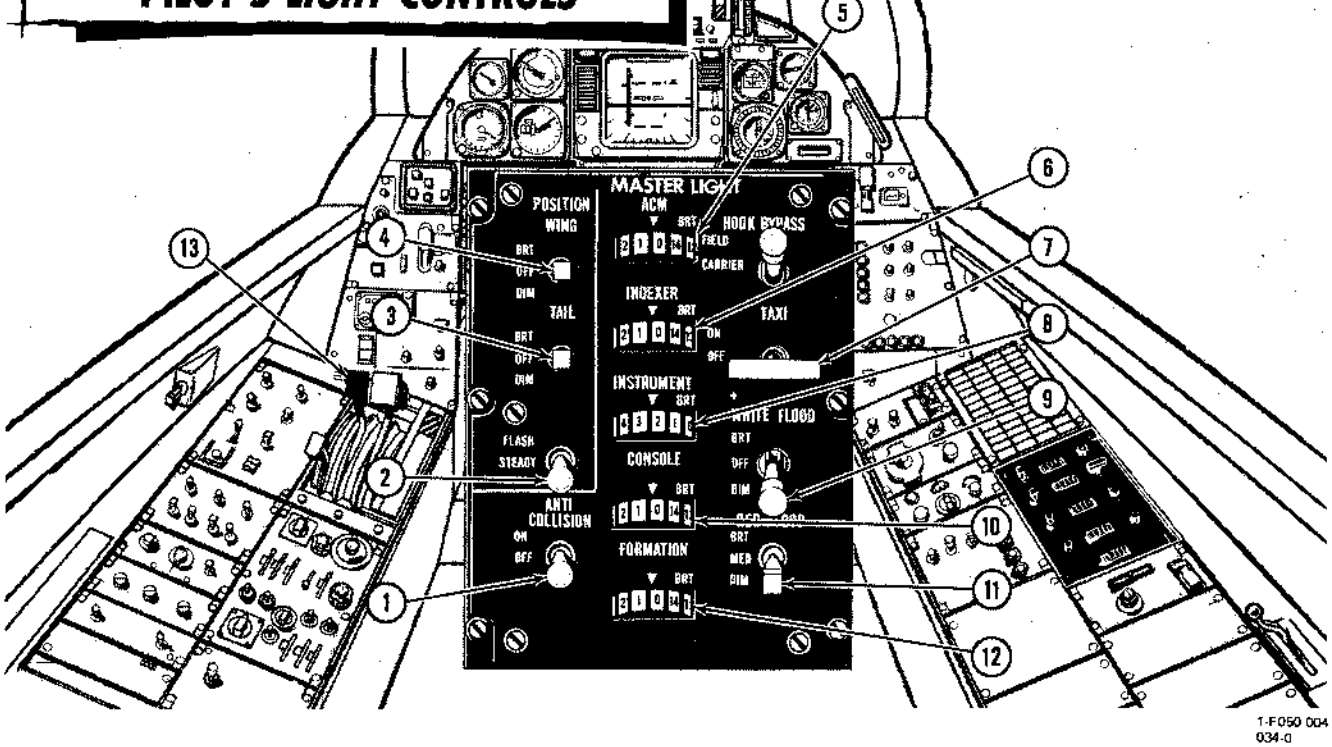


Figure 1-76

NOMENCLATURE		FUNCTION	
1	ANTI COLLISION LIGHT SWITCH	ON/OFF -	Energizes or deenergizes the anti-collision lights. When anti-collision lights are on the flasher unit for the position lights is disabled.
2	POSITION LIGHTS FLASHER SWITCH	FLASH -	With the wing and tail positions (or either) lights on, causes them to alternate off and on.
		STEADY -	with the wing and tail (or either) position lights on, lights are on steady.
3	TAIL POSITION LIGHT SWITCH	BRT -	High intensity of tail lights
		OFF -	Deenergizes tail position lights
		DIM -	Low intensity of tail lights

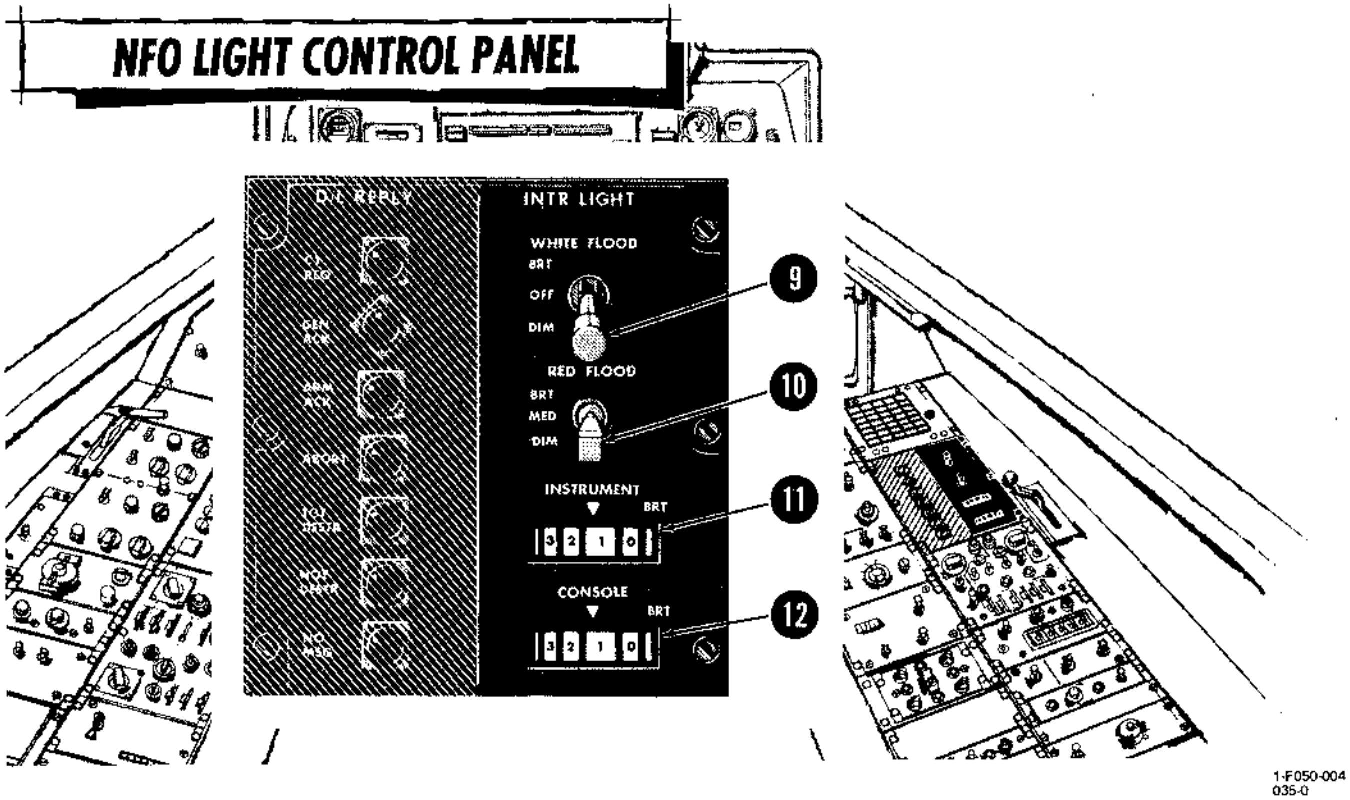


Figure 1-77

	NOMENCLATURE	FUNCTION	
(4)	WING POSITION LIGHT SWITCH	BRT - OFF - DIM -	High intensity of wing lights Deenergizes wing lights Low intensity of wing lights
(5)	ACM PANEL LIGHT THUMBWHEEL	0 to 1 1 to 13 14	turns lights on night variable intensity day bright (maximum intensity).
(6)	INDEXER THUMBWHEEL	0 to 14	variable increase in intensity of indexer lights and angle-of-attack approach lights.
(7)	TAXI LIGHT SWITCH	ON/OFF -	Nose gear must be down and the master exterior light switch must be on. Switch will return to OFF when nose gear is raised and door is closed.
(8)	INSTRUMENT LIGHTS THUMBWHEEL	0 to 1 - 1 to 14 -	turns instrument panel lights on. variable increase of intensity to a maximum brightness at 14.

NOMENCLATURE		FUNCTION
(9) WHITE FLOOD LIGHTS SWITCH		Switch must be pulled out to be moved to BRT or DIM.
	BRT -	High intensity brightness
	DIM -	Low intensity light.
(10) CONSOLE LIGHTS THUMB-WHEEL	0 to 1 -	turns console lights on
	1 to 14 -	variable increase of intensity to maximum brightness at 14.
(11) RED FLOOD LIGHT SWITCH	BRT -	High intensity red lights.
	OFF -	Red flood lights off.
	DIM -	Low intensity red lights.
(12) FORMATION LIGHTS THUMBWHEEL	0 to 1 -	turns formation lights on master exterior light switch on out-board throttle must be on.
	1 to -	variable increase of light intensity to maximum at 14.
(13) EXTERIOR LIGHTS MASTER SWITCH	ON -	(guarded switch) allows the pilot to signal for night catapult and permits him to turn off exterior lights after a night carrier landing.
	OFF -	

INTERIOR LIGHTS

(U) The interior lighting of both cockpits consists of red instrument panel and console panel lights, red and white flood lights for additional console and instrument panel lighting, and a utility/map light for each crew station. In the pilot's cockpit the interior lights are controlled from the master light panel on the right outboard console. The NFO can control his interior lighting from the interior light (INTR LIGHT) panel on his right outboard console. All white daylight lighting is powered from the main ac buses; only the red lighting is available when operating from the emergency ac generator.

INSTRUMENT AND CONSOLE PANEL LIGHTS

(U) All flight instrument in the pilot's and NFO's instrument panel and console panel lights are lighted by aviation red lighting. Individual thumbwheel controls are provided for instrument and console lighting controlled by the pilot and NFO. The Thumbwheels have fourteen variable selections from 0 to 14. Initial rotation from 0 to 1 activates the circuitry and provides a low intensity light. Further rotation up to a maximum intensity (13) increases the brightness. Position 14 corresponds to "day" mode. The day mode (14) turns out all instrument and console panel lights and increases the intensity of the warning, caution and advisory lights. The INSTRUMENT thumbwheel also controls the intensity of the caution/advisory panels, which consists of high and low intensity lighting. The pilot's and NFO's instrument and console lights are protected by circuit breakers on the NFO's right ac main circuit breaker panel.

FLOODLIGHTS

(U) The floodlights consist of 4.2-watt red and white and 20-watt white lights that illuminate the instrument and console panels. The pilot's and NFO's instrument panels are illuminated by white floodlights mounted on each side or above the consoles. The console panels are illuminated by white floodlights mounted aft of each side console and directed down toward the consoles. When navigating around thunderstorms the white floodlights should be turned on bright to assist in preventing temporary blindness from lightning. The WHITE FLOOD toggle switch on the pilot's master light panel and another on the NFO's light panel are safety interlock switches that must be pulled up to be positioned to BRT or DIM.

NOTE

> When the white floodlights are on (BRT or DIM), the intensity of the caution/advisory panel lights is increased to a day (bright) illumination mode.

(U) All red instrument panel lights are mounted in the pilot's and NFO's glare shields above the instruments. The red console lights are mounted serially under the longerons below the canopy locks and illuminate all console panels.

NOTE

> The WHITE FLOOD switch must be in the OFF position to see the red console or instrument lights.

(U) The red floodlights are protected by a circuit breaker on the NFO's ac essential No. 1 circuit breaker panel. The white floodlights are protected by a circuit breaker on the NFO's right ac main circuit breaker panel.

UTILITY/MAP LIGHTS

(U) The pilot's utility/map light is located on a bracket on the cockpit longeron above the aft of the left outboard console. The NFO's utility/map light is located in a bracket on the cockpit longeron above and mid-way along the left console. Each light has a rheostat control including an ON-OFF position on the rear of the lamp. Depressing the locking button on top of the lamp permits rotating the face of the lamp to select either a red filter or white light with a flood or spot illumination option. An alligator clip and swivel mounting allow the light to be positioned on a clipboard or other convenient location. A flasher button on the heel of the lamp allows either crewmember to use the light as a signal lamp. The utility/map lights are supplied electrical power from the essential No. 2 ac bus and are protected by a circuit breaker on the NFO's essential ac No. 2 circuit breaker panel.

WARNING AND INDICATOR LIGHTS

(U) Warning, caution, and advisory lights (figures 1-78 and 1-79) are provided in both cockpits to alert the pilot and NFO of aircraft equipment malfunctions, unsafe operating conditions, or that a particular system is in operation.

(U) The warning lights are illuminated red with black letters to warn of hazardous conditions that require immediate corrective action. All caution lights are located on the pilot and NFO CAUTION - ADVISORY panels. These lights illuminate yellow letters on an opaque background to indicate an impending dangerous condition that requires attention but not immediate action. The lower half of the CAUTION -ADVISORY panels consists of advisory lights that illuminate green letters on opaque background. Advisory lights indicate non-hazardous conditions which may not require corrective action. In addition, the digital data indicator (DDI) on the NFO's right console contains forty advisory lights associated with data link communications. Repeater data link advisory lights are located on the left side of the pilot's VDI. A functional description of each light is included in the applicable system description.

MASTER CAUTION Light

(U) The pilot's MASTER CAUTION light is centrally located on the air combat maneuver panel, and in the aft cockpit the NFO MASTER CAUTION light is on the right instrument panel. When the lights are illuminated yellow letters are illuminated on an opaque background. These lights will give a flashing indication whenever a caution light on the respective CAUTION - ADVISORY panel illuminates. A MASTER CAUTION light may be extinguished by depressing its lens. This will activate a reset switch which rearms the master circuit for a subsequent caution light. The illuminated caution light on the CAUTION-ADVISORY will not be extinguished by reseting the master caution light.

Indicator Lights Test

(U) A safety of flight check of all indicator lights can be performed while airborne or during on deck operations. The test is initiated by selecting the LTS position on the pilot's MASTER TEST panel and depressing the master test knob. Illumination of each indicator light and the GO light on the MASTER TEST panel verifies proper continuity of the indicator lights. A malfunction is indicated by a NO GO light and the associated indicator light does not illuminate.

(U) Depressing the master test switch energizes the L and R FIRE warning lights, the approach indexer, the emergency stores jettison button, and the caution-advisory panel lights which activates the MASTER CAUTION light. This light flashes unless there is a

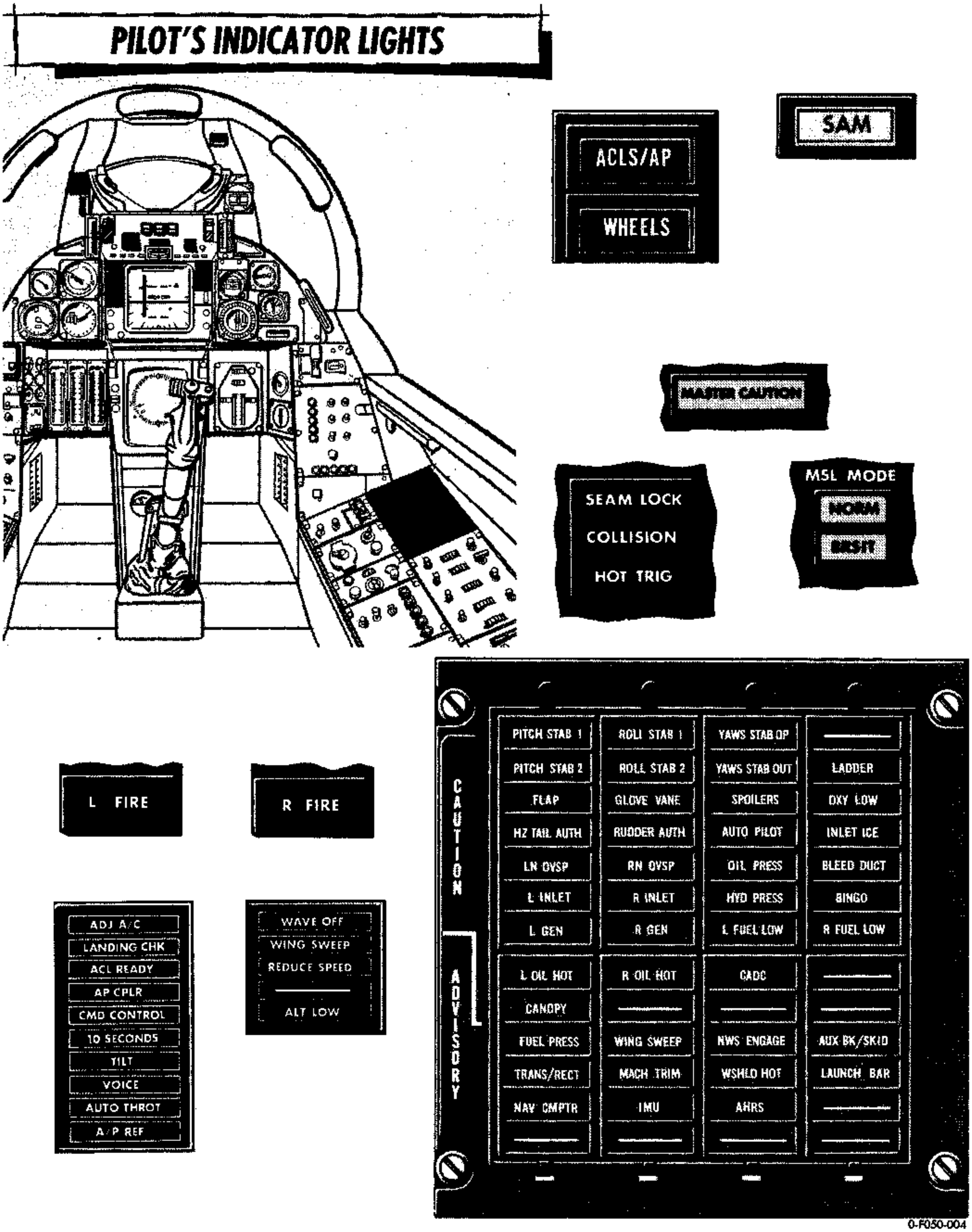
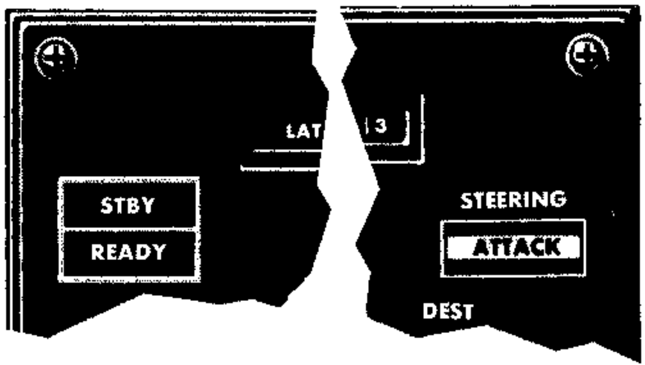
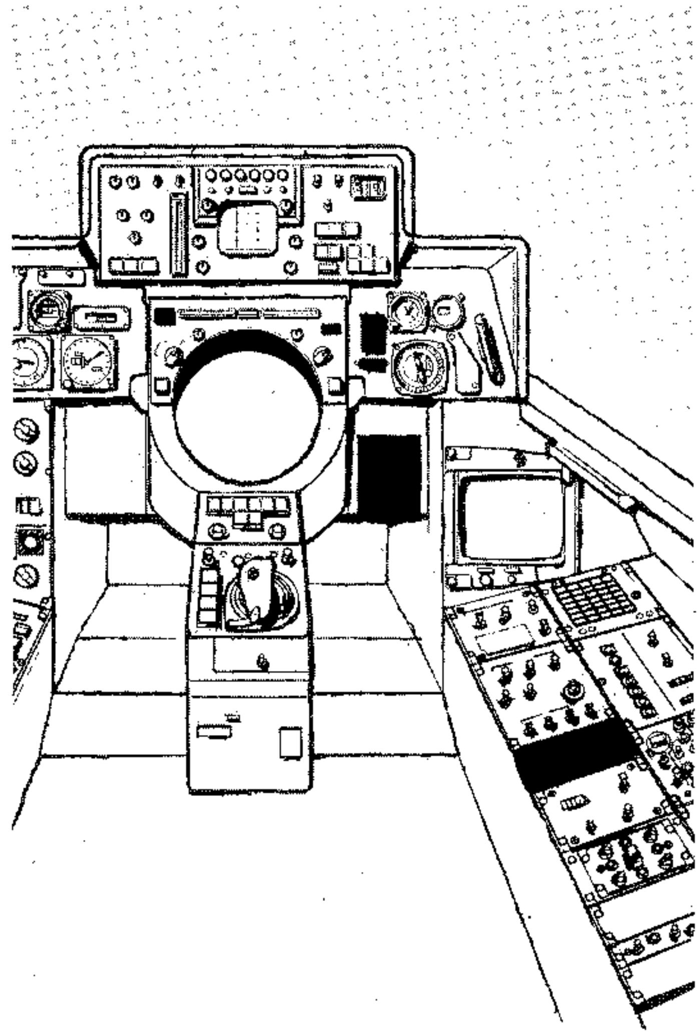
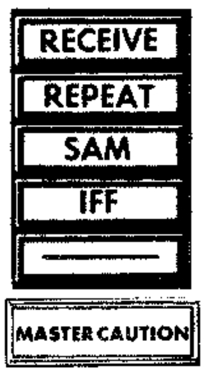

PILOT'S INDICATOR LIGHTS
ACLS/AP
WHEELS
SAM
MASTER CAUTION
SEAM LOCK
COLLISION
HOT TRIG
MSL MODE
L FIRE
R FIRE
ADJ A/C
LANDING CHK
ACL READY
AP CPLR
CMD CONTROL
10 SECONDS
TILT
VOICE
AUTO THROT
A/P REF
WAVE OFF
WING SWEEP
REDUCE SPEED
ALT LOW
CAUTION
PITCH STAB 1
ROLL STAB 1
YAWS STAB OP
PITCH STAB 2
ROLL STAB 2
YAWS STAB OUT
LADDER
FLAP
GLOVE VANE
SPOILERS
OXY LOW
HZ TAIL AUTH
RUDDER AUTH
AUTO PILOT
INLET ICE
LN OVSP
RN OVSP
OIL PRESS
BLEED DUCT
L INLET
R INLET
HYD PRESS
BINGO
L GEN
R GEN
L FUEL LOW
R FUEL LOW
ADVISORY
L OIL HOT
R OIL HOT
CADC
CANOPY
FUEL PRESS
WING SWEEP
NWS ENGAGE
AUX BK/SKID
TRANS/RECT
MACH TRIM
WSHLD HOT
LAUNCH BAR
NAV CMPTR
IMU
AHRS
0-F050-004
092-0

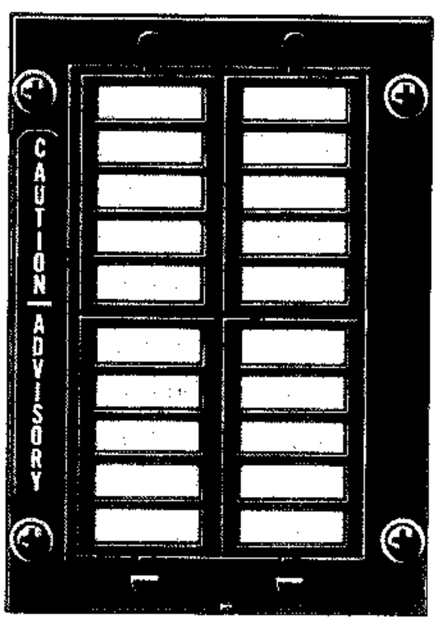
Figure 1-78

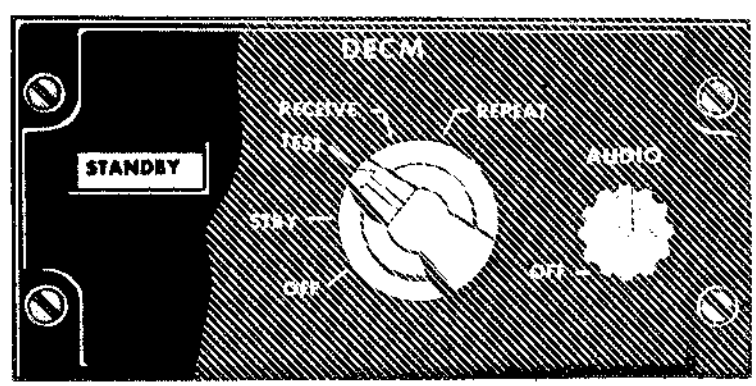
MCO'S INDICATOR LIGHTS











0-F050-004
093-0

Figure 1-79

circuitry failure within a caution-advisory indicator, in which case the light is on steady. The caution-advisory panel also supplies power to turn on the following indicator lights:

SAM	Landing gear transition
WHEELS	Hook transition light
ALT LOW	AUTO THROT
ACLAS/AP	A/P REF
REDUCE SPEED	WING SWEEP

BOARDING LADDER

(U) The boarding ladder (figure 1-80), located midway between the two cockpits consists of three folding sections housed within the left fuselage. It is held in the closed position by two mechanical locking pins which are actuated by the ladder control handle located in the face of the boarding ladder. The ladder must be released or stowed from ground level. The boarding ladder door will open to a semi-stow position initially and is held in this position by a spring-ball lock mechanism. The semi-stow position cannot be utilized as a step, but is intended to permit operation of the manual ammunition drive without holding the ladder. The ladder can be lowered to the down position by applying pressure to the ladder overcoming the ball lever spring mechanism. Unfolding the remaining two sections will place the ladder in a fully extended position. The bottom rung of the ladder is approximately 26 inches above the deck when in a fully extended position, with the nose gear unkneeled, and 12 inches above the deck if the nose gear is kneeled. If the nose gear is kneeled and the bottom section of the ladder is folded, the bottom rung is approximately 24 inches above the deck. A LADDER caution light is located on the pilot's caution advisory panel to advise the pilot that the boarding ladder is not in a full up and locked position.

BOARDING STEPS AND HANDHOLD

(U) There are two positive locking boarding steps, one on either side of the boarding ladder directly below each cockpit. They may be opened or closed from either cockpit or standing on the boarding ladder. The dimension of each step is approximately 9 inches by 13 inches. A single handhold is located directly above the boarding ladder. It is a spring-loaded door which returns to fuselage form when released. The dimensions of the handhold are approximately 3 inches by 5 inches.

NOSE RADOME

(U) The nose radome is a 135 pound fiberglass and aluminum structure, conical in shape with a maximum diameter of 45 inches and a length of 83 inches. It is attached to the aircraft by a top hinge and bottom mounted latches, permitting it to be rotated up for access and maintenance. In the open position the bottom of the dome is approximately 9 feet above the deck. A jury strut attached to the lower part of the dome can be fastened to the aircraft bulkhead to support the dome in an open position.

NOTE

After the nose radome is raised and the jury strut fastened in position, release hydraulic pressure to take load off hydraulic system.

BOARDING LADDER

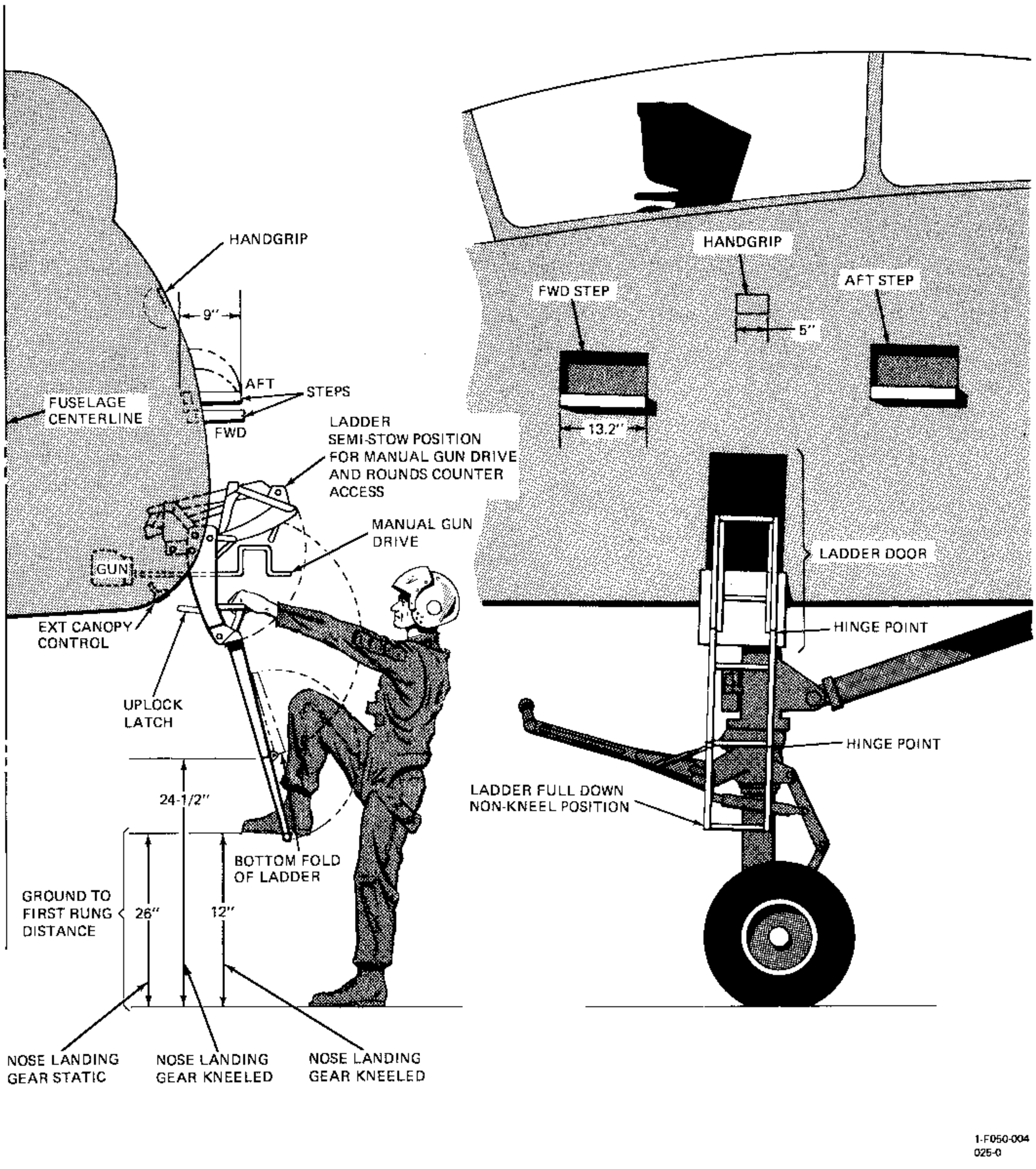


Figure 1-80

part 3 Aircraft Servicing

(U) The following servicing data is provided to lend assistance if the aircraft lands at a strange field or maintenance crews are unfamiliar with the aircraft. When operating in and out of military airfields, consult the current FLIP (ENROUTE) IFR- or VFR-Supplement for compatible servicing units, fuel, etc.

Fuel . . . . MIL-T-5624 (JP-5 or JP-4)
Hydraulic Fluid MIL-H-5606B
Liquid Oxygen, (type II) MIL-O-27210
Oil . . . . MIL-L-7808D or MIL-L-23699
Rain Repellent MIL-R-81261
Nitrogen (Type I) BB-N-411b

CAUTION

When ground temperature is -40°C (-40°F), MIL-L-7808D oil should be used.

POWER UNITS			
	PNEUMATIC STARTING	ELECTRICAL POWER	AIR CONDITIONING
SHORE BASED	GTC85-72	NC-10 NC-10A	NR-10 NR-5C
SHIPBOARD	GTC85-72	DECK EDGE OR NC-10	NR-5
REQUIREMENTS	60-70 LBS PER MIN @ 40-50 PSI 100° TO 425°F	115 ± 10 VAC 400 ± 25 HZ 3 PHASE ROTATION A,B,C	75 LBS PER MIN @ 3 PSI AND 45°F

GROUND REFUELING

(U) Single-point refueling is provided for pressure filling of all aircraft fuel tanks through a standard refueling receptacle located on the lower right side of the forward fuselage. Ground refueling is controlled by two precheck selector valves and a vent pressure gage adjacent to the refuel receptacle on the ground refuel/defuel panel. Positioning of these valves can be used for selective ground refueling of either the fuselage or wing and drop tanks. The two manually set valves separately check the high level pilot valves in the forward/aft tanks and the wing/drop tanks. The direct reading vent pressure gage indicates pressure in the forward, aft, and wing tank vent lines. When aircraft fuel tanks are full, fueling will stop automatically. For HOT REFUELING procedures, refer to Section III.

WARNING

Ensure that both the fueling unit and the aircraft are properly grounded, bonding cable is connected between aircraft and refueling source, and that fire extinguishing equipment is readily available.

CAUTION

During ground refueling operations the direct reading vent pressure indicator must be observed and refueling stopped if pressure indicates in the red band (above 4 psi).

NOTE

If the aircraft is serviced with JP-4 type fuel, the fuel-control fuel-grade (specific gravity adjustment) selector on each engine must be reset to the JP-4 position. Satisfactory engine performance is dependent upon trimming of the engine fuel controls to ensure rated thrust, prevent limit exceedance, and ensure airflow compatibility with the air inlet duct opening.

(U) PNEUMATIC PRESSURES

NOTE

Dry nitrogen, specification BB-N-411b, type 1, grade A, is preferred for charging pneumatic systems.

Wheel Brake Accumulators (2)	900 psi
Emergency Landing Gear System	3,000 psi at 70° F
Canopy (NORMAL) System	3,000 psi at 70°F
Nose Gear Shock Strut	1180 psi
Windshield Rain Repel	75 psi
Nose Wheel Tires (2)	
Ashore	105 psi
Afloat	350 psi

Main Gear Shock Struts (2)	placarded on main gear strut
Main Wheel Tires (2)	
Ashore	240 psi
Afloat	350 psi
Arresting Hook Dashpot	800 psi

(U) RESERVOIRS

Compound Hydraulic	placarded on access door
Flight Hydraulic	placarded on access door
Outboard Spoiler Hydraulic	placarded on module compartment
Engine Oil (2)	20 quarts
Integrated Drive Generator (2)	pints
Liquid Oxygen Converters (2)	10 liters
Emergency Oxygen Bottle	1800 psi at 70°F (minimum)
Windshield Rain Repel Tank	2 quarts

GROUND HANDLING

DANGER AREAS

(U) Engine exhaust and intake danger areas are shown in figure 1-81, and noise danger areas in figure 1-82.

RADIATION HAZARD AREAS

(U) Radiation hazards from high power radio and radar radiation are included in the Supplemental Flight Manual, NAVAIR 01-F14A-1A.

DANGER AREAS

WARNING

- At high thrust settings, the danger area around intake ducts may extend as far as four feet aft of the duct lip.
- Ear protection shall be worn at all times.

NOTE

This illustration contains estimated temperature and wake velocity distribution of P&W TF30-P-412 Afterburning Turbofan Jet Engine as per P&W Specification N-6130 with afterburners at maximum nozzle opening for MAXIMUM POWER (zone 5) and with afterburner nozzles fully closed for INTERMEDIATE and IDLE POWER.

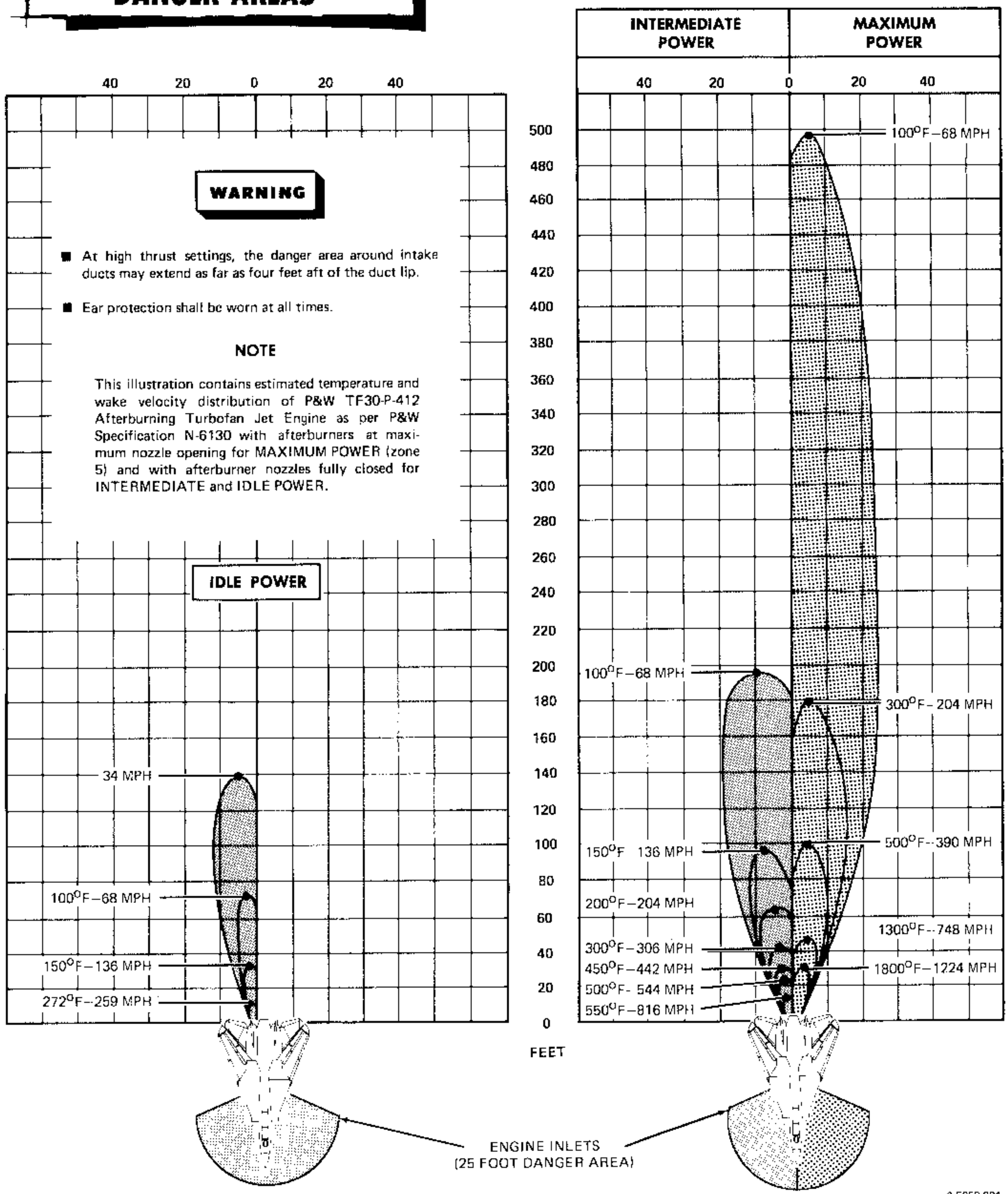


Figure 1-81

NOISE DANGER AREAS

ALLOWABLE NOISE EXPOSURE
SOUND LEVEL IN dBA (Slow Response)

EXPOSURE TIME, HRS*	1/4	1/2	1	2	4	6	8
NO PROTECTION	115	110	105	100	95	92	90
EAR PLUGS WITH AVERAGE SEAL	127	122	117	112	107	104	102
EAR PLUGS AND EARMUFFS	135	130	125	120	115	112	110

*Duration of exposure per day
Ref. BUMED Inst 6260.6B, 5 March 1970

NOTE

- TF30-P-412 engine noise danger areas are symmetrical about engine centerlines during dual engine operation at maximum power.
- This illustration contains the estimated sound pressure levels for near field and a 275 foot arc about the P and W TF30-P-412 AFTERBURNING TURBOFAN Jet Engine as per P and W Specification N-6130 with afterburners at maximum nozzle opening (zone 5).
- If engines are run up in front of blast deflector, sound is reflected to sides resulting in a distortion of pattern illustrated.

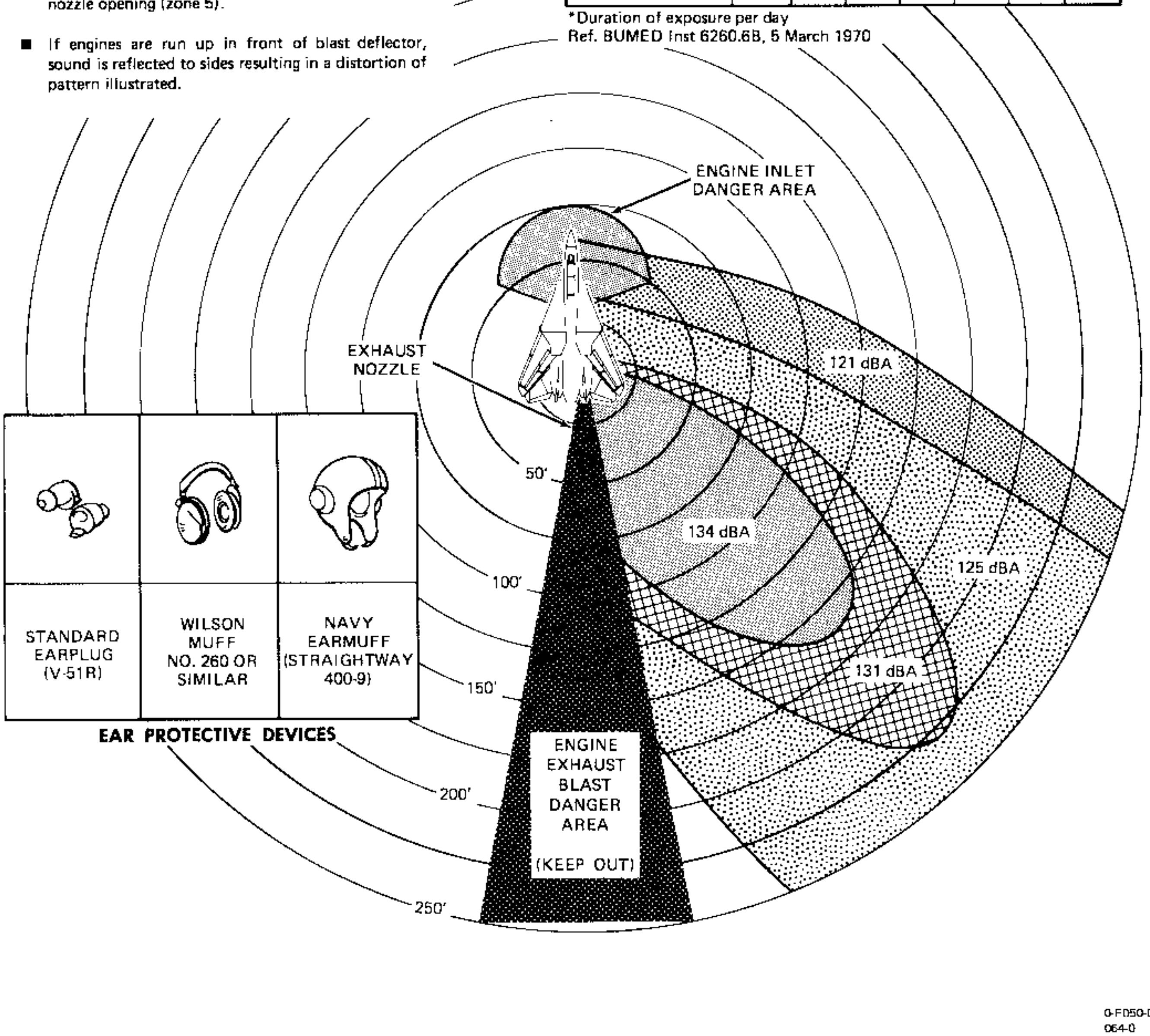
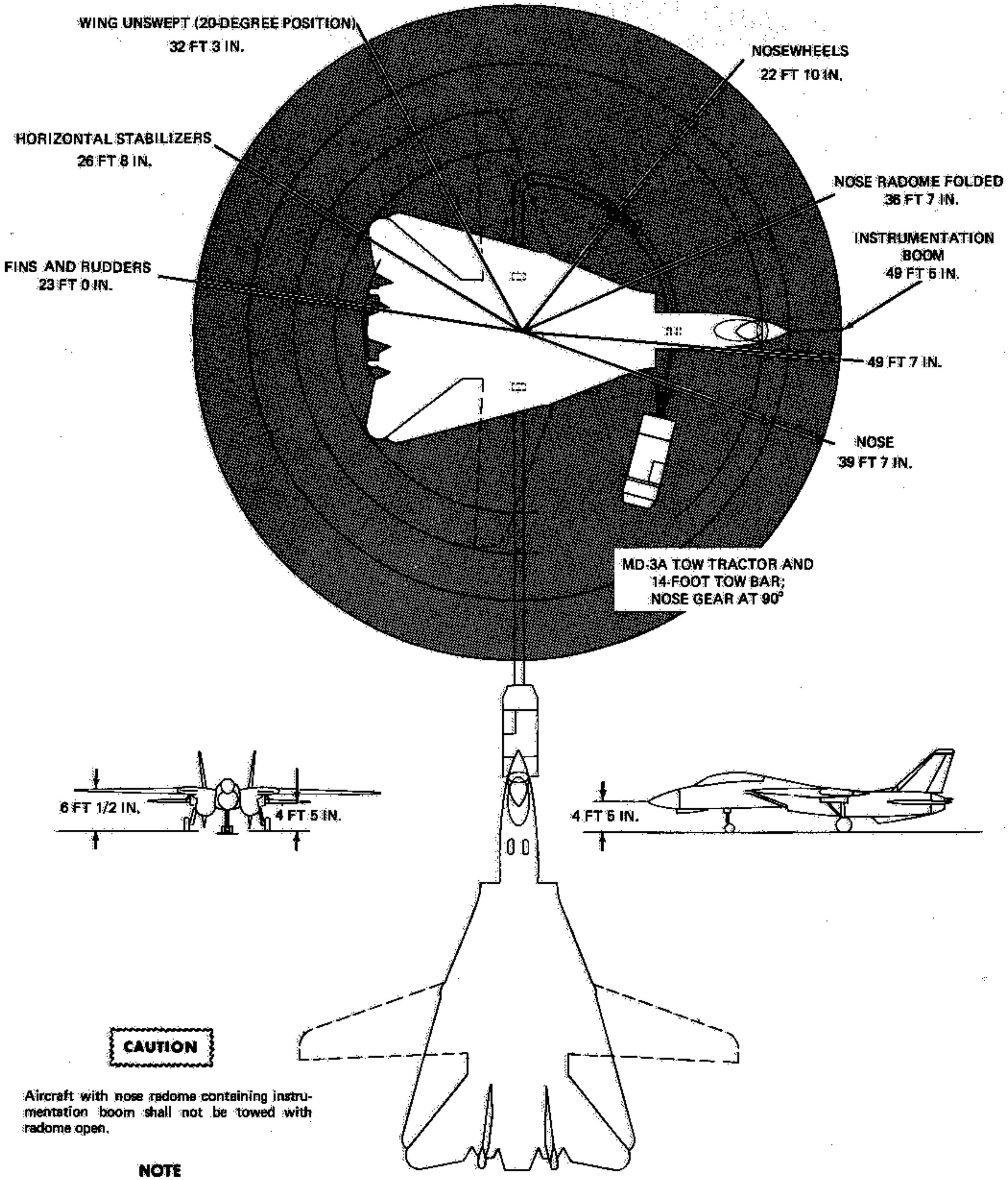


Figure 1-82

TOWING TURN RADII



CAUTION

Aircraft with nose radome containing instrumentation boom shall not be towed with radome open.

NOTE

Normally, aircraft shall not be towed with nose radome open.

1-F113-004
03700-018-0

Figure 1-83

part 4 Aircraft Operating Limitations

TO BE SUPPLIED

section III

NORMAL PROCEDURES

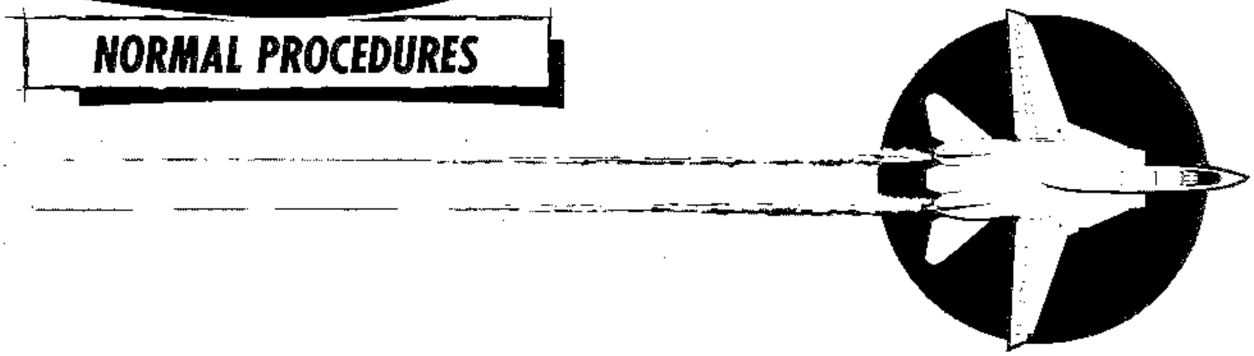


TABLE OF CONTENTS

TABLE OF CONTENTS (CONTINUED)

PART 5 CARRIER-BASED PROCEDURES

*To be supplied

part 1 Briefing/Debriefing

PRE-FLIGHT BRIEFING

(U) These briefings are presented immediately before the launching of scheduled flights, and therefore must be carried out in the most expeditious manner. It is imperative that all pilots and naval flight officers be in flight gear and ready for the briefing at the designated time. The briefing shall include, but not be limited to, the following:

Note

> Information marked with an asterisk (*) shall be displayed on a status board in the briefing or ready room, and should be copied by pilots before commencement of the briefing.

1. SCHEDULING

   a. Event Number*

   b. Takeoff, recovery times, and recovery order*

   c. Aircraft-pilot lineup*

   d. Mission assigned to each aircraft*

   e. Marshal information

2. MISSION

   a. CAP station assignment and control*

   b. Target aircraft rules of engagement

   c. Ground target description and procedures

3. COMMUNICATIONS

   a. Channels and frequencies*

   b. Navigational aids*

   c. Lost communications procedures

   d. Reports required

   e. Authenticators, IFF

   f. EMCOM conditions

4. PARTICIPATING UNITS

   a. Voice calls and side numbers*

   b. Disposition

   c. Utilization

   d. Friendly subs and surface units

5. OPERATIONS

   a. Instructions for coordinating other units

6. ORDNANCE

   a. Ordnance carried*

   b. Restrictions on use

7. WEATHER - BASE, ENROUTE, TARGET, AND DIVERT FIELD

   a. Wind: direction and velocity at surface and at applicable altitudes*

   b. Cloud coverage: present and forecast*

   c. Visibility*

   d. Sea state*

   e. Water and air temperature (cold weather)*

   f. Target weather*

   g. Divert weather*

8. MISCELLANEOUS

a. Other units in the area

b. Restricted or danger areas

c. Current NOTAMS, bulletins, and safety-of-flight information

d. Flight leader brief on takeoff, rendezvous frequency switch, landing procedures, etc.

e. SAR - participating units and procedures

f. BINGO fuel

g. Tanker aircraft information

part 2 Mission Planning

TO BE SUPPLIED

part 3 Check Lists

INTRODUCTION

(U) The normal procedures in this manual are promulgated to standardize general field and carrier-based operations for the F-14A aircraft and thereby ensure a greater degree of safety and operational effectiveness. These procedures are intended to govern those situations most frequently encountered and are not presumed to cover every conceivable situation. Refer to base operating instructions for the handling of live ordnance.

CHECKLISTS

(U) F-14 aircraft check lists are available in three forms based on the degree of aircrew familiarization; since the sequence remains the same, the only differences are in the degree of amplification. As the aircrewman becomes more proficient in type, a more abbreviated form is available to promote operational efficiency, and safety is not compromised since in all instances the thoroughness of checks remains the same. The placarded take-off and landing check lists on the cockpit instrument panels are a fundamental element in all instances. In the interest of procedural standarization, the shore-based and carrier-based check lists are maintained the same except for the response relative to the checks. The expanded check lists presented in this flight manual describe in detail those items which should be checked on each flight. Adherence to these check lists will provide the flight crew with a detailed status of weapon system performance incident to flight. However, it is encumbant on the flight crew to expand the checks as necessary to verify the corrective status of previously reported discrepancies. Reference should be made to maintenance test flight procedures for more detailed tests that can be performed on the aircraft and weapon systems if deemed necessary. The flight crew should be thoroughly familiar with the details of the check list procedures outlined herein so that abbreviated forms of the check lists may be safely employed. As the first level of simplification the pocket check list contains a reprint of the normal procedures check list less amplifying information. Lastly, after the expanded check list procedures herein, a much abbreviated check list is provided in suitable form for reprinting on a standard kneeboard size card.

EXTERIOR INSPECTION

(U) The exterior inspection to be performed by the flight crew has been divided into seven areas as shown below. Checks peculiar to only one side are so designated by (L) and (R) for the left and right side, respectively. Throughout the inspection the crew should check the surface condition, security of panels/fasteners, leaks (fuel, oil or hydraulic), external store/rack security and safety, servicing, ground safety cover/locks removed, FOD hazards, and configuration status.

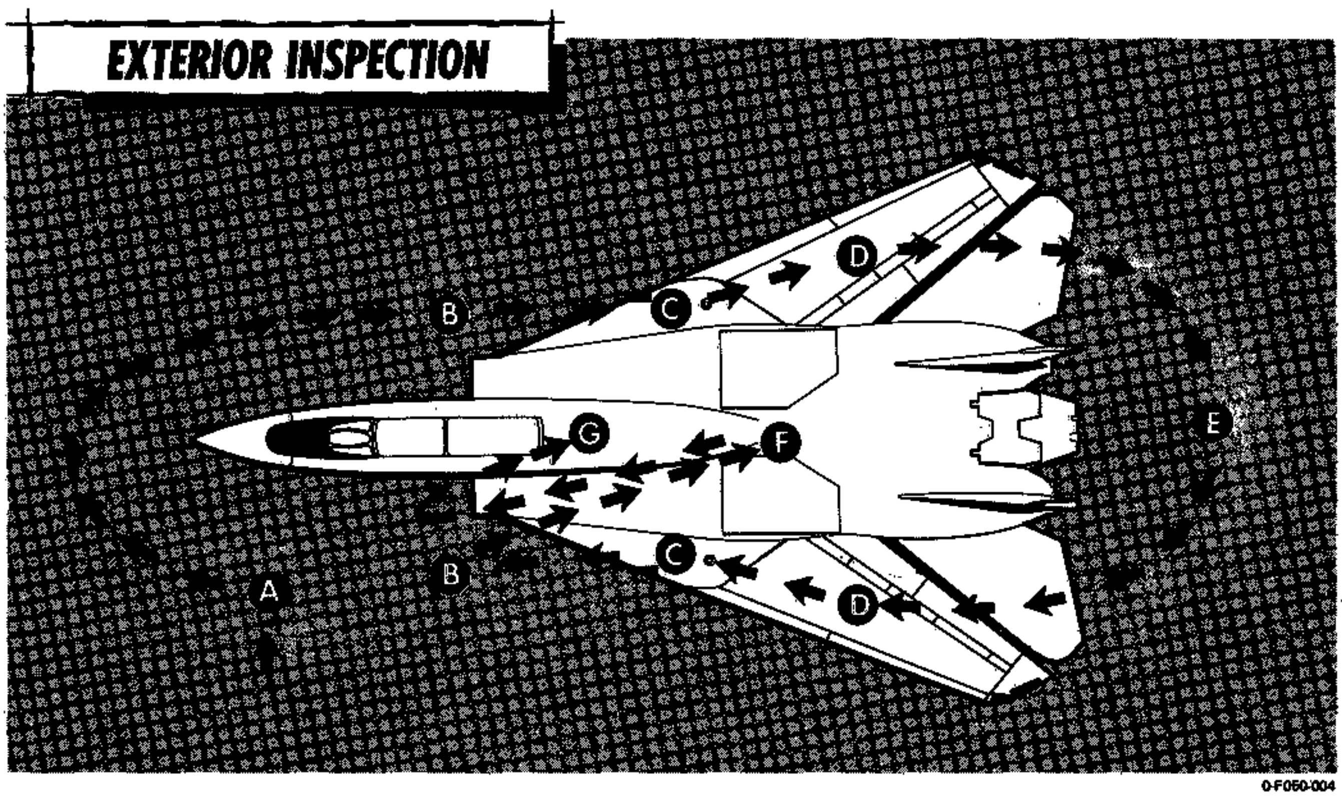


Figure 3-1.

(A) FOREWARD FUSELAGE

- Gun (L)
- Probes
- Nose wheel well
  - Doors
  - Air bottle gages
  - Retract actuator
- Nose strut
- Steering actuator
- Launch bar
- Nose wheels/tires
- Drag brace
- Approach lights
- IR/TV pod
- Radome

(B) INLET

- Ramps and seals
- Eng compressor face
- Ice detector (L)
- ECS heat exchanger inlet

(C) NACELLE & SPONSON

- Outboard spoiler module (L)
- Sta 2 & 7 racks/stores
- Main wheel well
  - Doors
  - Hydraulic lines
- Drag brace
- Side brace
- Main struts
- Brakes
- Main wheels/tires
- Hydraulic reservoirs
- Hook dashpot (R)
- Nacelle doors
- Ventrals

(D) GLOVE & WING

- Glove vane
- Sta 1 & 8 racks/stores
- Slats and flaps
- Wing cavity seal

(E) AFT FUSELAGE

- Horizontal tails
- Vertical tails/rudders
- Exhaust nozzles/fairings
- Fuel vent
- Tail hook
- Back-up flight module
- Speed brakes
- Fuel dump

(F) FUSELAGE UNDERSIDE

- Sta 3-6 racks/stores
- Fuel cavity drains

(G) FUSELAGE TOP DECK

- Overwing fairings
- ECS heat exchanger exhausts
- Bleed exit doors
- Canopy lanyards
- Aux canopy bottle reset

EJECTION SEAT INSPECTION

(U) The pilot and NFO shall perform the following checks on their respective ejection seats prior to flight. Although the ejection seat ground safety pins (figure 3-2) should have been removed by the plane captain, the crew should visually doublecheck that all pins are removed prior to flight. Ground safety pin removal should be confirmed on all actuation devices where a pin can be inserted to safety the mechanism for ground maintenance. Abbreviated preflight checklists for the ejection seat are provided in the pocket checklist and on the ejection seat headbox.

1. Face curtain .... SECURE

   Check that face curtain cross bar is securely restrained in the headbox.

2. Face curtain locking tab .... UP (Locked), PIN REMOVED

3. Face curtain connecting link (L) .... SAFE INDICATION

   Tabs adjacent as viewed through inspection window.

4. Drogue chute link lines .... CLIPS ENGAGED

5. Scissor shackle .... STOWED, DROGUE LANYARD CONNECTED

6. Ejection gun sear .... CONNECTED & PIN REMOVED

7. Ejection gun sear withdrawal links . . . . CONNECTED

   Check that withdrawal links are connected to sear.

8. Time release mechanism rod .... SCISSOR RELEASE PIN PROTRUDING

   Check that pin is protruding above housing so as to inhibit plunger movement and scissor release.

9. System initiator .... CONNECTED & PIN REMOVED

   Check that initiator is connected to withdrawal arms and inner and outer torque shaft.

10. Top latch mechanism .... FLUSH

    Check that top latch dowel is flush with or slightly recessed from the end of the housing.

EJECTION SEAT INSPECTION - (Continued)

11. Drogue withdrawal line .... CONNECTED TO DROGUE SLUG

    Check that shielded withdrawal line is free for the slug to deploy the stabilizing drogue chute.

12. Parachute withdrawal line screw connector .... FINGER TIGHT

    Check that the screw connector is torqued with essentially no gap.

13. Guillotine cutter ... WITHDRAWAL LINE ROUTED THROUGH GUILLOTINE

    Check that the sheathed parachute withdrawal line is routed through the guillotine cutter under the yellow protective guard.

14. Parachute premature deployment lanyard .... SECURED

    Check that the lanyard is anchored to the inertia reel housing.

15. Shoulder harness .... SECURED

    Check that inertia reel straps are anchored to inertia reel housing.

16. Parachute container .... SECURED

    Check that the container is secured to the seat back plate.

17. Power inertia reel sear .... CONNECTED

    Check that actuating link is connected to the power inertia reel gas generator sear.

18. Rocket motor initiator sear .... CONNECTED & PIN REMOVED

19. Drogue gun trip rod .... CONNECTED & PIN REMOVED

    No red indication showing on inner rod and rod attached to ejection gun cross beam.

EJECTION SEAT SAFETY PINS-MK GRU 7A

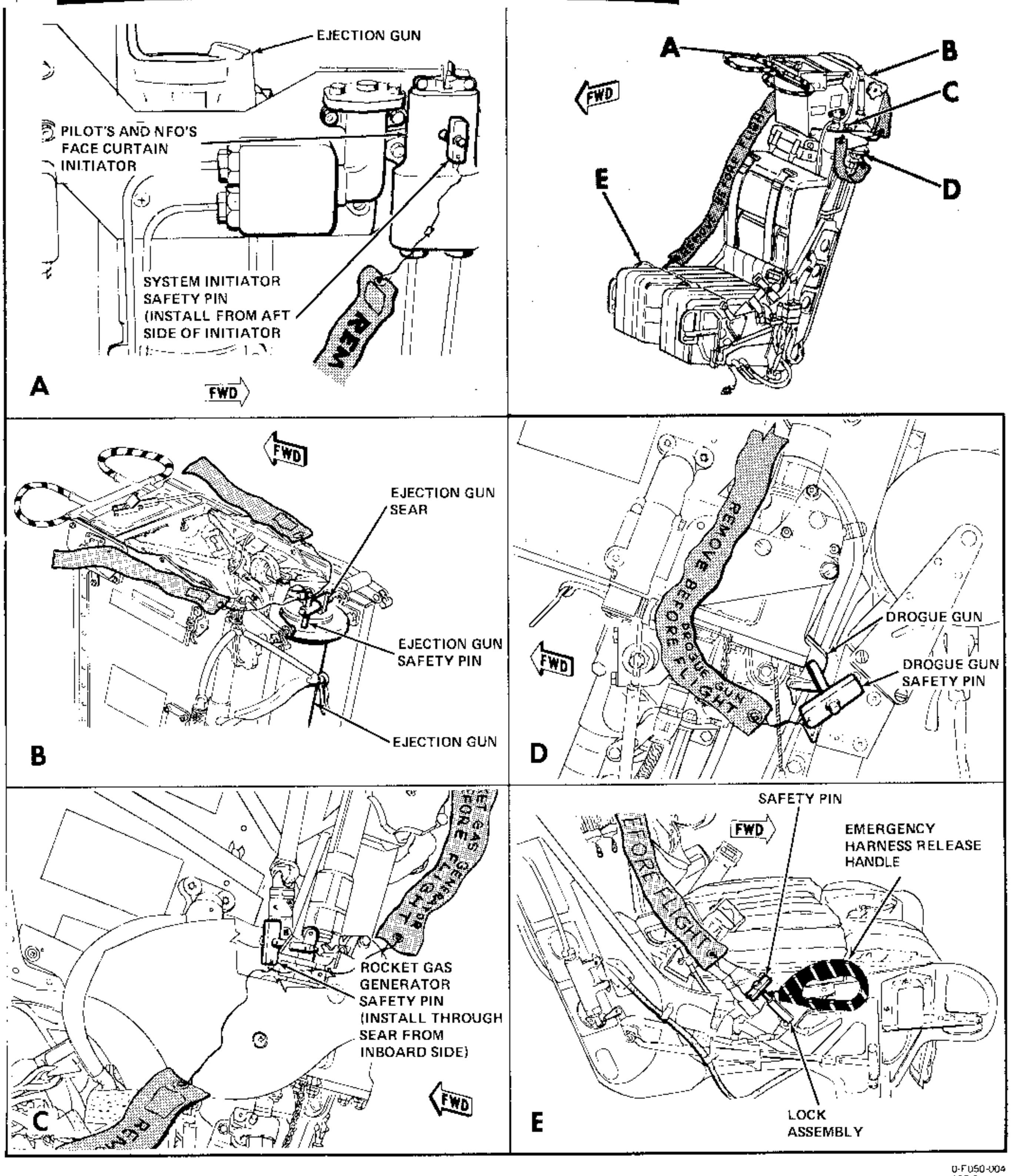


Figure 3-2.

EJECTION SEAT INSPECTION- (Continued)

20. Rocket motor initiator sear extraction lanyard .... ATTACHED

    Check that the lanyard is attached to the sear actuating link and the drogue gun trip rod.

21. Ventilation hose .... CONNECTED TO SEAT OR DISCONNECTED

    Connect as appropriate depending on flight suit configuration.

22. Personal services blocks..... VERIFY CONNECTION TO SEAT BUCKET DISCONNECT LANYARDS (2) SECURED TO DECK

23. Emergency oxygen trip cable .... SECURED

    Check that the seat kit cable is connected to the lower block so as to enable automatic oxygen actuation upon ejection.

24. Survival kit sticker clips and lugs .... SECURED/ENGAGED

    Check slips secured to seat bucket fitting and release linkage attached.

25. Lap belt restraint ....SECURED

    Tug lap restraint fittings to assure that end fittings are secured to seat bucket.

26. Alternate ejection handle ....GUARD UP (Locked), PIN REMOVED

27. Emergency oxygen lanyard....................................STOWED

28. Survival kit forward restraint pin .... ENGAGED

    Pull upward on forward edge of survival kit to ensure that restraint pin is engaged.

29. Leg restraint cords .... SECURED TO DECK, ROUTED THROUGH SNUBBER, ANCHORED TO SEAT RELEASE FITTING

30. Emergency harness release handle .. DOWN & SECURED, PIN REMOVED

    Check that handle is full down and trigger protrudes from grip.

31. Emergency oxygen supply bottle .FULL INDICATION, 1800 PSI (IN BLACK)

32. Guillotine initiator sear ....CONNECTED & PIN REMOVED

    Check that emergency release handle actuating linkage is connected to sear and leg restraint release linkage.

33. Time release mechanism trip rod ....CONNECTED

    No red indication showing on inner rod and rod attached to ejection gun cross beam.

34. Face curtain connecting link (R) ....SAFE INDICATION

    Tabs adjacent as viewed through inspection window.

INTERIOR INSPECTION

(U) The interior inspection provides for a systematic coverage of all cockpit controls to ensure proper set-up prior to the application of external power, assuming no external air conditioning source will be used prior to engine start. These checks correspond to the condition which the plane captain should set-up the cockpit as part of his pre-flight. Each cockpit set-up consists of a sequential sweep of controls on the left hand console, instrument panel and right hand console. After familiarization, the interior inspection set-up procedures should not require crew reference to a check list.

INTERIOR INSPECTION-PILOT

1. Harnessing .... FASTEN

   a. Leg restraint lines/garters - CONNECT

      Connect leg line bayonet fitting to leg garter quick release buckle on the respective side. Assure that leg lines are not twisted.

   b. Personal services (vent and anti-g hoses) - CONNECT

      Insert personal vent and anti-g hoses into seat block fittings. In the absence of a suit vent requirement the seat block vent hose should be inserted into the seat back fitting to provide ventilation air through the seat and back cushions.

   c. Lap belt - ATTACH

      Attach lap belt fittings and pull bucket straps snug so as to provide secure lap restraint for flight and seat kit suspension in the event of emergency egress or ejection.

   d. Shoulder harness - ATTACH

      Attach shoulder harness release fittings to torso fittings. Check for proper positioning of parachute D-ring on left riser.

   e. Inertia reel - CHECK

      Position the inertia reel manual control knob forward to the lock position and check that both shoulder straps lock evenly and securely when leaning back. Position control knob full aft to unlock harness and release knob to the neutral position.

INTERIOR INSPECTION-PILOT - (Continued)

f. Oxygen/audio connection - ATTACH

Attach composite fitting without causing unnecessary twisting of hard hose.

2. Oxygen .... CHECK

Turn oxygen switch on, purge with mask held away from face. Place mask to face and check for normal breathing and regulator/mask operation. Turn oxygen switch off, check no breathing.

3. Vent air flow thumbwheel .... SET

Set thumbwheel as desired to control vent airflow anywhere between no flow (0) and full flow (15).

4. Tone volume controls .... SET

5. ICS control panel .... SET

a. Interphone volume - CLOCKWISE

b. Amplifier selector - NORM

c. Function selector - COLD MIC

6. UHF/DL control panel .... SET

a. UHF function selector - OFF

b. Main UHF squelch disabled - OFF

c. Main UHF volume - MID RANGE

d. Main UHF channel - SET

e. Main UHF manual frequency - SET

f. Main UHF channel/frequency selector - PRESET

g. Aux receiver channel - SET

h. Aux receiver sensitivity - MID RANGE

INTERIOR INSPECTION-PILOT - (Continued)

i. Aux receiver volume - MID RANGE

j. Communication antenna switch - AUTO

k. D/L antenna - UPPER

l. D/L address - SET

m. D/L forced reply - NORM

n. D/L anti-jam - OFF

o. D/L channel - SET

p. D/L power - OFF

q. D/L manual frequency - SET

7. Emergency wing sweep handle .... EMERG/CORRESPONDING

   Wing sweep control should not be in the AUTO or MANUAL modes with the wings swept aft.

8. L/R throttles .... OFF

9. Wing sweep control button .... AUTO

10. Speed brake switch .... HOLD (Centered)

11. Exterior lights master switch .... SET

    Position switch in accordance with standard procedures for day/night and field/carrier operations.

12. Flap handle .... CORRESPONDING

13. Throttle friction lever .... OFF (Aft)

14. Air temperature switch .... SET

    Ambient air temperature: $>80°F$ - HOT
    $40° - 80°F$ - NORM
    $<40°F$ - COLD

15. Throttle mode switch .... BOOST

INTERIOR INSPECTION-PILOT - (Continued)

16. Engine/probe anti-ice switch .... AUTO

17. Windshield air switch .... OFF

18. L/R inlet ramp switches .... AUTO

19. Anti-skid spoiler brake switch .... OFF

20. Fuel control panel .... SET

    a. Wing/external transfer switch - AUTO

    b. Refuel probe switch - RET

    c. Dump switch - OFF

    d. Feed switch - NORM (guard down)

21. Landing gear handle .... DN

22. Parking brake handle .... SET (Aft)

23. Radar altimeter .... OFF

24. Altimeter .... SET

    Set to field/carrier elevation

25. L/R fuel shutoff handles .... IN

26. ACM panel .... SET

    a. ACM switch - OFF (guard down)

    b. Master arm switch - SAFE (guard down)

27. Weapon select .... OFF

28. VDIG/HSD retaining locks .... ENGAGED

    Visually check that the retaining locks for the pilot's displays are engaged to assure safe back-up mounting for catapulting and flight.

INTERIOR INSPECTION-PILOT - (Continued)

29. HUD filter .... SET

    Filter control knob on right side of VDI should be pushed in for day flights and pulled out (filter in place) for night operations.

30. G-meter .... RESET

31. Clock .... WOUND & SET

32. Fuel BINGO .... SET

    Set total fuel remaining valve for initial activation of fuel BINGO caution reminder consistent with mission profile to be flown.

33. L/R circuit breakers .... IN

34. Brake accumulator pressure .... CHECK

    Check brake pressure indication relative to reading observed on nose wheel well gage during exterior inspection.

35. Hydraulic handpump .... CHECK

    Extend handpump handle and stroke to check firmness of pumping action and an indication of pressure buildup on the brake pressure gage. Stow handpump handle in a convenient position for ready access.

36. Hook handle .... CORRESPONDING

37. Display control panel .... SET

    a. Mode - TO

    b. HUD declutter - OFF

    c. HUD AWL - ACL

    d. VDI mode - NORM

    e. VDI AWL - ACL

    f. HSD mode - NAV

INTERIOR INSPECTION-PILOT - (Continued)

g. HSD ECM - OFF

h. Steer cmd - TACAN

i. Display power - OFF

38. Elevation lead .... SET

39. L/R generator switches .... NORM

40. Emergency generator switch .... NORM (guard down)

41. TACAN control panel .... SET

a. Channel selector - SET

b. Function selector - OFF

c. Volume control - COUNTER CLOCKWISE

42. ARA-63 control panel .... SET

a. Channel selector - SET

b. Power switch - OFF

43. Compass control panel .... SET

a. Mode control - SLAVED

b. Hemisphere switch - SET

c. Latitude knob - SET

44. Air condition control panel .... SET

a. Temperature mode switch - AUTO

b. Temperature control thumbwheel - SET (5-7 mid range)

c. Cabin pressure switch - NORM

d. Air source selector - BOTH ENG

INTERIOR INSPECTION-PILOT - (Continued)

45. Master light control panel .... SET

    Set external and interior lighting controls consistent with day/night and field/carrier operating conditions.

46. Master test selector knob .... OFF

47. Emergency flight hydraulic switch .... AUTO (guard down)

48. Canopy defog lever .... CABIN AIR (aft)

49. Storage case .... INSPECT

    Check adequacy of flight planning documents and storage of loose gear.

INTERIOR INSPECTION-FLIGHT OFFICER

1. Circuit breakers left and right .... IN

2. Harnessing .... FASTEN

   a. Leg restraint lines/garters - CONNECT

      Connect leg line bayonet fitting to leg garter quick release buckle on the respective side. Assure that leg lines are not twisted.

   b. Personal services (vent and anti-g hoses) - CONNECT

      Insert personal vent and anti-g hoses into seat block fittings. In the absence of a suit vent requirement the seat block vent hose should be inserted into the seat back fitting to provide ventilation air through the seat and back cushions.

   c. Lap belt - ATTACH

      Attach lap belt fittings and pull bucket straps snug so as to provide secure lap restraint for flight and seat kit suspension in the event of emergency egress or ejection.

   d. Shoulder harness - ATTACH

      Attach shoulder harness release fittings to torso fittings. Check for proper positioning of parachute D-ring on left riser.

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

e. Inertia reel - CHECK

Position the inertia reel manual control knob forward to the lock position and check that both shoulder straps lock evenly and securely when leaning back. Position control knob full aft to unlock harness and release knob to the neutral position.

f. Oxygen/audio connection - ATTACH

Attach composite fitting without causing unnecessary twisting of hard hose.

3. Oxygen .... CHECK

Turn oxygen switch on, purge with mask away from face. Place mask to face and check for normal breathing and regulator/mask operation. Turn oxygen switch off, check no breathing.

4. Vent airflow thumbwheel .... SET

Set thumbwheel as desired to control vent airflow between no flow (0) or full flow (15).

5. ICS control panel .... SET

a. Interphone volume - MID-RANGE

b. Amplifier selector - NORM

c. Function selector - COLD MIC

6. UHF/DL control panel .... SET

a. UHF function selector - OFF

b. Main UHF squelch disable - OFF

c. Main UHF volume - MID-RANGE

d. Main UHF channel - SET

e. Main UHF manual frequency - SET

f. Main UHF channel/frequency selector - PRESET

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

g. Aux receiver channel - SET

h. Aux receiver sensitivity - MID-RANGE

i. Aux receiver volume - MID-RANGE

j. Communications antenna switch - AUTO

k. D/L antenna - UPPER

l. D/L address - SET

m. D/L forced reply - NORM

n. D/L anti-jam - OFF

o. D/L channel - SET

p. D/L power - OFF

q. D/L manual frequency - SET

7. TACAN control panel .... SET

a. Channel selector - SET

b. Function selector - OFF

c. Volume control - MID-RANGE

8. Liquid cooling switch .... AWG-9/AWG-9 AIM-54 as applicable

9. Ejection seat command control .... PILOT

Position in accordance with squadron policy.

10. Sensor control panel .... SET

a. RADAR .... SET

(1) STAB - IN

(2) AZ CNTR - CENTERED

(3) EL CNTR - CENTERED

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

(4) VSL - OFF

(5) AZ SCAN - ±65°

(6) EL BARS - (1) BAR

b. IR/TV .... SET

(1) SLAVE - INDEP

(2) TRACK - AUTO

(3) AZ CNTR - CENTERED

(4) EL CNTR - CENTERED

(5) FOCUS - INF

(6) AZ SCAN - ±65°

(7) EL BAR - (1) BAR

11. Computer address panel .... SET

a. CATEGORY - TAC DATA

12. Armament control panel .... SET

a. WPN type - OFF

b. ATTK mode - MAN

c. STEP/RPL - STEP

d. SGL/PRS - SGL

e. ELECT FUZE - SAFE

f. A/G GUN - OFF

g. MECH FUZE - SAFE

h. SEL JETT - SAFE

i. JETT options - MER/TER

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

j. INTERVAL - SET

k. QTY - SET

l. STA sel 1-8 - SAFE

m. MSL options - NORM

n. MSL speedgate - NAR

13. Ground check panel .... CLOSED

14. Attitude gyro .... CAGED

15. Altimeter .... SET

Set for field carrier elevation

16. Detail Data Display .... SET

a. PD THRLD CLEAR - NORM

b. PD THRLD CLUTTER - NORM

c. AGC - NORM

d. PARAMP - ON

e. IR AUDIO THRLD - COUNTER-CLOCKWISE

f. IR AUDIO VOL - COUNTER-CLOCKWISE

g. ACM THRLD - COUNTER-CLOCKWISE

h. PULSE GAIN - COUNTER-CLOCKWISE

i. IR GAIN - COUNTER-CLOCKWISE

j. PULSE VIDEO - COUNTER-CLOCKWISE

k. BRIGHT control - COUNTER-CLOCKWISE

l. ERASE - COUNTER-CLOCKWISE

m. JAM/JET - COUNTER-CLOCKWISE

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

n. ASPECT switch - AS DESIRED

o. MLC - AUTO

p. TGTS - NORM

q. XMTR CHAN - SET

r. MSL CHAN - SET

17. TID controls .... SET

a. CONTRAST - COUNTER-CLOCKWISE

b. NAV MODE - OFF

c. BRIGHT - COUNTER-CLOCKWISE

d. DESTINATION - HB

e. TID MODE - A/C STAB

18. Clock ....WOUND & SET

19. ECMD .... SET

a. BRIGHTNESS CONTROL - COUNTER-CLOCKWISE

20. Hand control panel .... SET

a. IR/TV switch - OFF

b. WCS switch - OFF

c. EL CONTROL - DETENT

d. MISSION RECORDER - OFF

21. EDM display panel .... SET

a. CORR - OFF

b. ORIDE - OFF

c. MODE - NAV

d. DATA/ADF - OFF

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

22. ECM control panel .... SET

    a. APR 25 BAND - OFF (3 switches)

    b. E-AAA - OFF

    c. APR 25-27 PWR - OFF

    d. AUDIO (APR 25-27) - OFF

23. DECM control panel .... SET

    a. Control switch - OFF

    b. AUDIO - OFF

24. CHAFF/FLARE Dispense panel .... SET

    a. MANUAL dispense - OFF

    b. PWR MODE - OFF

    c. CHAFF/flare programmer - SET

25. D/L reply .... SET

    a. Select - NO MSG

26. INTR LIGHT control panel .... SET

    a. WHITE FLOOD - AS DESIRED

    b. RED FLOOD - AS DESIRED

    c. INSTRUMENT brightness - AS DESIRED

    d. CONSOLE brightness - AS DESIRED

27. IFF control unit .... SET

    a. MASTER cont - OFF

    b. MODE select - ON (four switches)

    c. CODE select - SET

INTERIOR INSPECTION-FLIGHT OFFICER - (Continued)

d. IDENT - OUT

e. MODE 4 - OUT

f. RAD TEST - OUT

28. Air-to-air interrogator .... SET

a. CODE - SET

29. IFF antenna .... AUTO

30. GND cooling switch .... OFF

31. Radar beacon control panel .... SET

a. MODE - AS BRIEFED

b. POWER switch - OFF

32. KY-28 control panel .... SET

a. Language mode - PLAIN

b. POWER switch - OFF

33. Cabin defog lever .... CABIN AIR (aft)

PRE-START-PILOT

(U) The following checks are performed by the pilot once ground electrical power is applied to the aircraft incident to starting the engines. Prior to engine start, temperature critical avionics (i.e., UHF/DL, TACAN, CADC, CSDC, VDIG, HSD and AWG-9 related software must not be electrically energized without the use of a forced air ground cooling unit. Therefore, the pre-start checks should be performed expeditiously to prevent induced failures of temperature critical avionics components ... otherwise, a ground cooling air source should be used.

1. Seat/pedals .... ADJUST

   Adjust vertical seat height such that vision over-the-nose is not impaired by the glare shield and helmet can be placed back against headbox without face curtain handle interference. The eye position of the pilot is important for viewing the displays as well as external field of view. Subsequent to seat bucket height adjustment, pull pedal adjust lever on center pedestal to collectively adjust rudder pedal fore-aft position for sitting comfort and capability of achieving combined full rudder pedal/brake pedal deflection to either side. Release adjust lever to lock pedal adjustment in selected detent.

2. ICS .... CHECK

   Verify two-way intercommunication between crewmembers and adjust volume to comfortable level.

3. DL/TACAN/UHF command control .... SET

   Set command control (light illuminated) in accordance with mission and crew operating procedures.

4. Landing gear .... DN

   Check gear position indicating DN and transition light extinguished.

5. Standby altitude gyro .... UNCAGED

   Pull and rotate caging knob counter clockwise to uncage gyro. Set approximate pitch attitude. Ground static attitude is +1.5° in nose strut extended condition on level deck.

PRE-START-PILOT - (Continued)

6. Fuel quantity/distribution .... CHECK

Verify level of individual tank quantities and that summation equals value displayed on the totalizer. Use fuel quantity select switch to ascertain wing and external tank fuel quantities. Check that L/R FUEL LOW caution lights are extinguished.

7. Oxygen quantity .... CHECK

Verify quantity remaining is ample to support planned mission. Check that OXY LOW caution light is extinguished.

8. Compass .... ALIGNED

Push synchronize button to rapid align AHRS. Check for reasonable heading alignment with standby compass. AHRS advisory light should extinguish within 2 minutes.

9. Master Test .... CHECK

Raise and rotate the master test selector knob to each of the following positions. Depress knob to initiate checks and raise knob to terminate same.

a. LTS

Check for illumination of all lights in the cockpit. The brightness level of the ACM panel and indexer lights should be set during this test.

b. FIRE DET

L/R FIRE warning lights should illuminate to verify continuity of respective systems.

c. INST

Check for the following responses:

- RPM - 80 %
- TIT - 960°C
- Fuel flow - 4400 pp

nominal mid-scale tape readings

PRE-START-PILOT - (Continued)

- Angle of attack - 18 units (reference and indication)
- Wing sweep - 45° (program, command and position)
- Fuel quantity - 2,000 pounds
- Oxygen quantity - 2 liters
- LN/RN, OVSP caution lights - illuminated
- L/R FUEL LOW caution lights - illuminated
- OXY LOW caution light - illuminated

d. MACH LEV

Approximately 10 seconds are required to achieve a GO(NO GO) indication.

e. WG SWP

Immediately prior to initiation of test, depress MASTER RESET button to extinguish WING SWEEP advisory light - initiate test within 5 seconds. Wing sweep program index goes to 44°, then 68° and returns to 20°. Illumination of WING SWEEP advisory light during the test indicates a discrepancy in control drive servo loop.

10. ACM control panel . . . . SET

a. Gun rate - SET/CHECK ROUNDS REMAINING

b. SW cool - OFF

c. MSL prep - OFF

d. MSL mode - NORM

e. Station loading status windows - CHECK

Verify proper indication consistent with external store/rack loading condition.

f. Emergency stores jettison button - LIGHT OUT

PRE-START-PILOT - (Continued)

11. Emergency flight hydraulic switch . . . . LOW/HI/AUTO

    LOW - Check ON flag in EMERG FLT LOW hydraulic pressure window. Verify control over horizontal tail and rudder surface positions as viewed on surface position indicator.

    HI - Check ON flag in EMERG FLT HI hydraulic pressure window. Verify control over empennage flight control surfaces and higher surface deflection rate.

    AUTO - Check both EMERG FLT HI and LOW flags on the hydraulic pressure windows display OFF.

12. Ejection seats . . . . ARMED

    Arm ejection seat firing handles by depressing locking tab on top of headbox and rotating lower ejection handle safety guard downward. Verify seat armed condition with NFO.

13. Canopy control . . . . CLOSE

    Alert NFO and receive verbal clearance from NFO prior to initiating canopy close mode. If NORM close mode is insufficient due to high velocity surface wind conditions, depress grip detent release and push handle further forward into the BOOST close mode. Check that the CANOPY caution light extinguishes with full forward translation of the canopy into the sill locks. Handle may remain in the CLOSE or HOLD position for flight.

14. Boarding ladder . . . . STOWED

    Plane captain will stow boarding ladder and steps. Check LADDER caution light extinguished.

PRE-START-FLIGHT OFFICER

(U) The following checks are performed by the Flight Officer once ground electrical power is applied to the aircraft incident to starting engines. Prior to engine start temperature critical avionics (i.e. UHF/DL, TACAN, CADC, CSDC, VDIG, MDIG, AWG-9 and related software must not be energized without the use of a forced air ground cooling unit.

1. Seat/ICS/UHF foot switches . . . . ADJUST

   Adjust seat height such that the helmet can be placed against the headbox without face curtain handle interference. Subsequent to seat height adjustment, adjust ICS/UHF foot pedal fore-aft position for sitting comfort.

2. ICS . . . . CHECK

   Verify two-way communications between crewmembers and adjust volume to a comfortable level.

3. DL/TACAN/UHF . . . . SET

   Set command control in accordance with mission and crew operating procedures.

4. Standby attitude gyro . . . . UNCAGED

   Pull and rotate caging knob counterclockwise to uncage gyro. Set approximate pitch attitude. Ground static attitude is +1.5° in nose strut extended conditions on level deck.

5. LTS . . . . CHECK

   Check for illumination of console and instrument lighting.

6. LT test . . . . CHECK

   Check for illumination of all caution and advisory lights, ECM lights and DDI lights.

7. Ejection seats . . . . ARMED

   Arm ejection seat by depressing locking tab on top of headbox and rotate alternate ejection handle safety guard downward. Visually check pilot's locking tab depressed.

PRE-START-FLIGHT OFFICER - (Continued)

8. Canopy control .... CLOSE

   Pilot will normally close canopy, however, if flight officer closes canopy insure verbal clearance from pilot. If NORM close mode is insufficient due to high velocity surface wind conditions, depress grip detent release and push handle further forward into the BOOST close mode. Check that CANOPY caution light extinguishes with full forward translation of canopy into the sill locks. Handle may remain in the CLOSE or HOLD position for flight.

ENGINE START-PILOT

(U) Prior to engine start the pilot in conjunction with the plane captain should ascertain that the turn-up area is clear of FOD hazards, adequate fire supression equipment is readily available, and engine intakes and exhausts are clear. Although the engines cannot be started simultaneously, either engine can be started first. The following procedure establishes starting the right engine first in which case all ground support equipment should be positioned on the left side of the aircraft insofar as practical. Whenever possible the aircraft should be positioned so as to avoid tailwinds which can increase the probability of engine hot starts. Prior to starting either engine the following caution/advisory lights will be illuminated: L/R GEN, L/R FUEL PRESS, OIL PRESS, HYD PRESS, WING SWEEP, IMU, NAV CMPTR and LADDER. Subsequent to connecting the low pressure start air source and coming up to pressure at the plane captain's signal:

1. Engine crank switch .... R (right engine)

   Place the crank switch to the R position where the switch is solenoid held until automatically released to the OFF position at the starter cut-out speed of 45% RPM. Pilot manual deselection of the switch to OFF will interrupt the crank mode at any point in the start cycle. Oil pressure rise and flight hydraulic pressure rise will be evident at rotor speed of 10% RPM.

2. R throttle .... IDLE AT 20% RPM

   Advance the R throttle around the horn from OFF to IDLE when the rotor speed exceeds 20% RPM; this action automatically actuates the ignition system. Attempting a ground start at lower engine rotor speeds will aggravate hot start tendencies. Exceedance of 750°C TIT constitutes a hot start. An immediate indication of fuel flow (≈500-700 pph) will be exhibited and light off (TIT rise) should be achieved within 5 seconds. The

ENGINE START-PILOT - (Continued)

rapid rise in TIT should be carefully monitored for overtemp tendencies. Peak starting temperatures will be achieved in the 30% to 40% RPM range, when, after a slight hesitation, a reduction will return the TIT to the nominal 600°C level. During the initial starting phase the nozzle should expand to a full open position indication of 5.

3. IDLE engine instrument readings . . . . CHECK

   a. Engine crank switch - OFF

      Switch should automatically return to OFF at ≈ 45% RPM and conditioned airflow from the ECS will be directed to the cockpit and avionics cooling loops.

   b. R GEN caution light - OUT

      The right generator should automatically pick up the electrical load on the right main AC bus, as indicated by R GEN caution light extinguishing, at approximately 50% RPM.

   c. R FUEL PRESS advisory light - OUT

      The fuel pressure light should extinguish by the time the engine achieves idle RPM. Failure of the light of extinguish indicates a fuel motive flow malfunction (motive flow fuel pump or turbine driven boost pump) in which case the engine should be shut down.

   d. RPM - 63% to 74%

   e. TIT - nominal 600°C

   f. Fuel flow - nominal 1000 pph

   g. Nozzle position - 5 (Open)

      - Advance throttle slightly and check nozzle closure to position 0.

      - Retard throttle to idle and check nozzle position returns to 5.

   h. Oil pressure - Greater than 30 psi

   i. Hydraulic pressure - 3000 psi flight system
      0 psi combined system

ENGINE START - (Continued)

4. Ground power/air . . . . DISCONNECT

   Removal of ground electrical power causes:

   - Right generator to supply power to right and left main AC electrical busses

   - Hydraulic shutoff valves open to activate the hydraulic transfer pump in the flight-combined direction. Combined hydraulic pressure should indicate approximately 2400 psi.

   - AUX BK/SKID light illuminates with parking brake set to indicate brake valve shuttle is in AUX mode with 2000 psi combined system pressure.

5. Ground safety pins . . . . REMOVED AND STOWED

   Once the combined hydraulic system is pressurized the plane captain can remove the landing gear downlock pins, store the pins in the nose wheel well.

6. Cross bleed start of left engine . . . . R THROTTLE TO 85%

7. Repeat steps 1-3 for the left engine

   - Check combined hydraulic pressure 3000 psi

   - Following start of both engines all caution/advisory lights except IMU, NAV CMPTR, WING SWEEP and AUX BK/SKID should be extinguished.

PRE-TAXI

(U) The sequence of pre-taxi checks accomplishes all in-place checks necessary to be accomplished before setting the aircraft in motion. During this time the plane captain is afforded an opportunity to check the exterior in a fully powered condition for leaks, bleed air ejector operation, ground cooling fan operation, etc.

PRE-TAXI-PILOT

1. Communications .... ON
   a. UHF function selector - TR+G
   b. Data link power - ON
2. TACAN function selector .... T/R
3. ARA-63 power switch .... ON
4. Displays control power .... ALL
5. Radar altimeter .... ON

   Rotate knob clockwise to turn radar altimeter ON and set minimum altitude warning index at 50 feet.
6. Pitch, roll and yaw stability augmentation switches .... ON
7. Pitch, roll and rudder trim .... SET 0°
8. Master test .... CHECK
   a. INST

      Adjust display brightness controls and set HUD/VDI pitch trim to +1.5° (nose strut extended) on AHRS reference. Check display test patterns in TO, CRUISE, A/A, A/G and LDG modes. Return display mode switch to TO.

PRE-TAXI-PILOT - (Continued)

b. EMERG GEN

Check that NO GO light illuminates for about 1 second until emergency generator power is connected to essential buses and GO light illuminates. When discontinuing this test the resultant power interruption causes the AHRS advisory light to illuminate momentarily.

c. OBC

The following systems are automatically exercised during the 1-1/2 minutes required to complete the OBC tests. Failures are displayed on the TID display.

- AICS

  The AICS self-test turns on hydraulic power and exercises the ramps through full cycle STOW-EXTD-STOW. During the test the respective RAMP caution light will be illuminated until the ramps return to the fully stowed position and hydraulics are shut off. A failure is indicated by an INLET caution light and/or OBC readout.

- PITCH, ROLL, YAW STAB AUG
AUTHORITY STOPS
MACH TRIM COMPENSATOR
AUTOPILOT, SPOILER } AFCS COMPUTERS

  During the course of the test the STAB AUG caution lights will remain illuminated until the test is satisfactorily completed. All lights should be extinguished at the termination of the test.

- CADC

  - Engine power trim indicators display in-band condition

  - Wing sweep program bug drives to $>68^\circ$

PRE-TAXI-PILOT - (Continued)

- AUTO THROTTLE

  This test is a computer self-test with output commands inhibited to prevent throttle movement.

d. FLT GR DN

Overrides weight-on-wheels interlock so that autothrottle can be engaged on the deck if it is desired to check the servo drive loop. Have plane captain rotate ADD probe clockwise to give low ADD reading prior to engagement.

e. FLT GR UP

Simulates landing gear UP configuration for pressurization check of external fuel tanks. GO light should illuminate with external fuel tanks aboard. Check is brief so that actual fuel transfer is not necessary to verify external fuel tank transfer capability.

f. D/L RAD

g. OFF

Return master test knob to OFF at termination of checks.

9. Component cycles .... CHECK

Cycle the following components in response to plane captain's hand/wand signals.

a. Arresting Hook - DN-UP

Lower arresting hook handle and after checking illumination of the hook transition light raise handle. Transition light should be extinguished to indicate hook in a safe uplocked condition for flight.

b. Speed Brakes - EXTD - IN

Cycle speed brakes EXTENDED then IN while checking for proper indication in the retracted, partially extended, and fully extended positions.

PRE-TAXI-PILOT - (Continued)

c. Air refueling probe - EXTD-RET

Cycle the probe to the extend position noting illumination of the probe transition light with switch-probe position disparity. Check probe nozzle head for condition. Retract probe and again check transition light extinguished when fully retracted/doors closed.

d. Wing sweep . . . . 68° EMERG

Spider detent should be at 20° so that WING SWEEP lights remain illuminated. HZ TAIL AUTH caution light illuminates after bringing handle forward to 68°. Light extinguishes when OVSP stops removed.

e. Flight controls . . . . CYCLE

Complete full cycle sweep of longitudinal, lateral, directional and combined longitudinal-lateral controls while checking for full authority on surface position indicator.

- longidutinal 30° UP to 10° DN horizontal tail
- lateral ±24° total differential tail
- directional ±30° rudder
- long-lat combined 35° UP to 15° DN horizontal tail

10. Radar altimeter BIT . . . . TEST

Depress set knob; check radar altitude displays 100 ft. and indicator green light illuminated. Release knob and pointer should display ≈ 0 ft, warning tone signal and ALT LOW warning light illuminated momentarily.

11. ARA-63 BIT . . . . TEST

12. Flight instruments . . . . CHECK

a. Altimeter

Barometric setting and error determined. Check in RESET mode.

PRE-TAXI-PILOT - (Continued)

b. Turn & slip indicator

Check alignment of turn rate needle and centering of ball.

c. INS ALIGNED

Check NAV CMPTR and IMU caution lights extinguished. Verify HUD/VDI pitch trim +1.5° (nose strut extended) on IMU reference.

d. Compass

Validate IMU heading alignment on HUD, VDI, HSD and BDHI. Cross check with known reference and standby compass.

e. Navigation

Check TACAN, VHF/ADF and IMU navigation reference indications.

13. Anti-skid/Spoiler brake switch . . . . BOTH (FIELD)
. . . . OFF (CARRIER)

PRE-TAXI-FLIGHT OFFICER

1. Liquid cooling switch . . . . AWG-9/AIM-54-AWG-9 as applicable

2. WCS power switch . . . . STBY

   a. Verify BIT 2

3. Computer address panel . . . . INSERT HOME BASE

4. NAV MODE switch . . . . ALIGN

   Select appropriate align mode (GND, CVA). Note time.

5. IR/TV power switch . . . . STBY as applicable

PRE-TAXI-FLIGHT OFFICER - (Continued)

6. Communication .... ON/SET
   a. UHF functions selector - TR&G
   b. Data Link power - ON
7. TACAN function selector .... T/R

   After two minutes, actuate self-test and check
8. Computer address panel .... SET
   a. Enter desired data, WP, FP, etc.
9. Detail Data Display
   a. PD THRLD Clear - NORM
   b. PD THRLD Clutter - NORM
   c. AGC - NORM
   d. PARAMP - ON
   e. IR AUDIO THRLD -
   f. IR AUDIO VOL -
   g. ACM THRLD -
   h. MANVR - OFF
   i. ALT DIFF - OFF
   j. VGS - OFF
   k. Range Buttons - 10
   l. PULSE VIDEO - SET
   m. IR GAIN - SET
   n. JAM/JET - COUNTER-CLOCKWISE
   o. BRIGHT - SET

PRE-TAXI-FLIGHT OFFICER - (Continued)

p. SYM INT - SET

q. ERASE - SET

r. ASPECT switch - NOSE

s. MLC - AUTO

t. TGTS - NORM

u. XMTR CHAN - SET (TRAINING/TACTICAL)

v. MSL CHAN - SET

w. DISPLAY - RDR

x. WCS MODE - PD SRCH

10. TID controls .... SET

a. CONTRAST - SET

b. BRIGHT control - SET

c. DEST - HOME BASE (HB)

d. CLSN - OFF

e. TID MODE - A/C STAB

f. TID EXP - OFF

g. ALT NUM - ON

h. SYM ELEM - ON

i. DATA LINK - OFF

j. JAM strobe - OFF

k. Non ATTK - OFF

l. Launch zone - ON

m. VEL Vect - OFF

n. RANGE scale - SELECT

PRE-TAXI-FLIGHT OFFICER - (Continued)

11. ECM display panel .... SET
    a. Brightness control - SET
    b. TEST button - DEPRESS-CHECK
    c. MODE - NAV
    d. Data ADF - BOTH
    e. CORR - OFF
    f. OVERRIDE - OFF
12. Hand control panel .... SET
    a. Light test (All AWG-9 lights come on) - PRESS/CHECK
    b. El Vernier (Category - radar position) - SET 0° EL.
13. ECM .... OFF
14. DECM .... OFF
15. CHAFF/FLARE dispenser .... OFF
16. Cabin defog lever .... CABIN AIR (aft)
17. IFF .... SET/ON
    a. SET code - TEST
18. Air-to-air interrogator .... SET
    a. SET code - TEST
19. IFF antenna switch .... AUTO
20. Indicate lights test .... TEST

    Hold switch forward.
21. DDI BIT test .... TEST

    Hold switch aft.

PRE-TAXI-FLIGHT OFFICER - (Continued)

22. After align completed .... INS

    a. READY light - ON

    b. STANDBY light - OUT

    c. NAV mode switch - INS

Note

The IMU should be aligned prior to releasing parking brake otherwise the alignment process will be automatically interrupted until parking brake is reapplied.

23. DEST data .... VERIFY

24. BRG/Dist to DEST .... CHECK

25. OWN A/C GND SPD .... CHECK

TAXI-PILOT

1. Parking brake .... RELEASE

    a. IMU - ALIGNED

       Prior to releasing the parking brake the IMU should be aligned otherwise the alignment process will automatically be interrupted until parking brake is reapplied.

    b. Anti-skid - TEST (field)

       After releasing the parking brake and while depressing the toe brakes, the plane captain should depress the test button on the anti-skid control box in the nose wheel well. The test requires approximately 10 seconds to complete after which the AUX BK/SKID caution light should remain extinguished. Illumination of the light is indicative of a malfunction after which the anti-skid should not be used.

2. Nose wheel steering .... CHECK

   NWS advisory light illuminates upon engagement. Check control and polarity in static position before commencing to taxi.

TAXI-PILOT - (Continued)

3. Brakes .... CHECK

   Check for proper operation by applying L/R brake individually and observing brake pressure recovery to the fully charged condition.

4. Wing sweep .... 20° AUTO

   a. EMERG wing sweep handle - 20°

      Move the emergency wing sweep control handle to the 20° (full forward) spider detent and stow the handle and guard.

   b. Master reset button - DEPRESS

      This action extinguishes the WING SWEEP warning and advisory lights and enables AUTO/MAN control modes.

5. Maneuver flap/slats .... CHECK

   Roll maneuver flap thumbwheel aft and hold for 3 seconds. Check for full deflection (10° flaps) of maneuver devices (except glove vane).

6. Flap handle .... DOWN

   Check for full deflection of the flaps and slats to the DN position and automatic activation of the outboard spoiler module.

7. Spoilers .... CHECK

   a. Check all spoilers indicating the drooped position.

   b. Spoiler brakes - CHECK (field)

      Check with both throttles at IDLE that all 8 spoiler segments indicate and visually check UP. All spoilers should return to the drooped condition with advancement of either throttle forward of IDLE.

TAXI-PILOT - (Continued)

c. Spoiler lateral control - CHECK

For field operations advance one throttle forward slightly to drop out spoiler brake mode. Deflect stick full left and right and check for full spoiler deflection (indication and visual) on the side corresponding to stick deflection.

8. Ordnance .... CHECK/ARM

Perform the following functions at prescribed location prior to take-off in accordance with base operating procedures:

a. Missile seeker/tuning - CHECK

b. Gun/external stores - GROUND SAFETY PINS REMOVED/ARMED

TAXI-FLIGHT OFFICER

(U) The following sequence is with the assumption that the INS has completed alignment and in the INS mode of operation. OBC/BIT and BIT sequences may be performed at any time after INS alignment.

1. Perform OBC - If not previously checked.

OBC/BIT initiate for Class IIA equipment (AFCS, APC, AICS, CADC) must be coordinated with pilot. For NFO portion of OBC/BIT:

a. Category switch - SPL

b. OBC/BIT on CAP - DEPRESS

- Observe "M" on TID - 30 sec. approx.
- Observe OBC flashing on TID - 95 sec. approx.
- Observe FAILED acronyms TID
- Observe test complete TID

c. Category switch - DESELECT SPL

TAXI-FLIGHT OFFICER - (Continued)

2. Perform BIT sequency 1 thru 4

   If not previously checked.

   a. Category switch - BIT

   b. BIT sequences - SELECT

      - Observe failures

   c. Category switch - SELECT NAV

3. OWN A/C GND SPD .... CHECK

TAKE-OFF-PILOT/NFO

(U) Prior to take-off the contents of the placarded aircraft take-off checklist will be completed by the challenge (NFO) and reply (pilot) method with ICS on HOT MIC as a double check of the aircraft configuration status.

	NFO Challenge		Pilot Reply (Typical)
1.	BRAKES	-	"Check OK, accumulator pressure up, brake light out, anti-skid on/off (field/ carrier)."
2.	FUEL	-	"Normal feed, auto transfer, wing/ external tank transfer checked, total________, distribution____________ aft and left and ________ fwd and right, feed tanks full."
3.	CANOPY	-	"Closed, locks engaged, canopy light out, seal inflated, handle closed."
4.	STAB AUG	-	"On"
5.	SEAT - "Armed Aft, "Command Eject Forward" (Field) Aft (Carrier)	-	"ARMED FWD"
6.	TRIM	-	"Set 0°."
7.	WINGS	-	"20°, Auto, Wing Sweep lights out."

TAKE-OFF-PILOT/NFO - (Continued)

NFO Challenge	Pilot Reply (Typical)
8. FLAPS	- "Full down, aux flap down, spoilers drooped (when not in spoiler brake mode)."
9. CONTROLS	- "Authority check OK, hydraulic pressure 3000 psi."
10. HARNESS - "Locked Aft," "Lights out, ready for takeoff."	- "Locked forward, lights out except ______ (acknowledge exceptions)."

DESCENT-PILOT

1. Canopy defog lever .... DEFOG (forward)
2. Fuel quantity and distribution .... CHECK

   Set BINGO counter to desired direct fuel state.

3. Wing sweep mode .... AUTO
4. Altimeter .... RESET

   Reset altimeter to local barometric setting as biased by altimeter error,

DESCENT-FLIGHT OFFICER

1. Canopy defog lever .... DEFOG (forward)
2. Altimeter .... RESET

   Reset to local barometric setting

3. Challenge pilot radar altimeter .... ON
4. Fuel quantity .... NOTE
5. Flight instruments .... CHECK
6. ECM .... OFF
7. DECM .... OFF
8. Chaff flare dispenser .... OFF

LANDING-PILOT

(U) The landing checklist is designed to be performed in sequence upon returning to base for landing. The initial steps are performed prior to entering the landing pattern whereas the final steps are essential elements to be checked prior to each landing when remaining within the pattern. With the ICS on HOT MIC the pilot should call the accomplishment of each step so that the NFO can double check that all essential steps have been performed.

1. Armament .... SAFE

   A safe condition consists of:

   - Master Arm Switch - SAFE (guard down)
   - ACM switch - OFF (guard down)
   - Weapon select switch - OFF
   - NFO armament controls - SAFE

2. Fuel .... CHECK

   Check fuel quantity to assure acceptable distribution, gross weight and CG for landing. Fuel switches should be set to transfer all available fuel to the feed tanks. Dump switch should be OFF.

3. Hook .... UP (field)
   DN (carrier)

   Transition light should be extinguished.

4. Anti-skid .... BOTH (field)
   OFF (carrier)

5. Wings .... 20° AUTO

   Check wings in AUTO sweep control mode and set at 20° (or appropriate for Mach).

.... Entry to Landing Pattern ....

6. Speed brakes .... EXTEND

   Place switch in EXTD position and hold until indicator displays full speed brake extension condition.

LANDING-PILOT - (Continued)

7. Wheels .... DN

Place gear handle DN and check for wheels DN indication on all three gear and gear transition light extinguished. Check brake accumulator pressure fully charged.

8. Flaps .... DN

With the wings at 20° sweep, place flap handle DN and check for main flap and slat full DN indication and auxiliary flap extension (no FLAP caution light).

9. Autopilot .... SET

Set autopilot switches OFF (depress emergency disengage paddle switch) if not using ACL mode for landing. For ACL have the autopilot ENGAGED and ACL switch armed.

10. Auto throttle .... SET

If using auto throttle for landing, put throttle friction OFF, match throttles and set the throttle mode switch to AUTO.

11. Harness .... LOCKED

12. Annunciator lights .... OFF

Except for known exceptions, receive confirmation of same from NFO.

LANDING-FLIGHT OFFICER

(U) The landing checklist is designed to be performed in sequence upon returning to base for landing. The initial steps are performed prior to entering the landing pattern whereas the final steps are essential elements to be checked prior to each landing when remaining within the pattern. Crew should be on Hot Mic with pilot calling accomplishments of each step so the NFO can double check that all essential steps have been performed.

1. Armament .... CHECK

   a. ACP switches - SAVE

   b. Insure pilot has checked following switches:

      (1) Master arm - OFF

      (2) ACM switch - OFF (guard down)

      (3) Weapon select - OFF

2. Radar .... STBY

3. Harness .... LOCKED

4. Annunciator lights .... OFF

   (except for known exceptions)

POST LANDING-PILOT

1. Anti-skid .... OFF (field)

   Place anti-skid/spoiler brake switch to OFF.

2. Flaps/slats .... UP

   Move flap handle forward and check for complete retraction of main flaps/slats and auxiliary flaps (no FLAP caution light). Check auto deactivation of the outboard spoiler module.

3. Speed brakes .... RETRACT

POST LANDING-PILOT - (Continued)

4. Wing sweep .... 68° OVSP ENABLE

   As soon as the auxiliary flaps are retracted(______seconds), aft wing sweep command may be effected as follows:

   a. Wing sweep button - AFT

      Hold button aft until sweep command bars slew to 68°.

   b. Wing sweep handle - OVSP ENABLE

      Raise guard to gain access to emergency wing sweep handle. Handle will be at 68° position; raise handle grip to full up extension position (spring held over-sweep enable position).

5. Avionics .... OFF

   Turn off all avionics (data link, radar altimeter, displays, TACAN, ARA-63) except UHF radio. Cage standby attitude gyro.

6. Wing sweep .... 75° OVSP

   When HZ TAIL AUTH caution light extinguishes, move emergency wing sweep handle full aft (75° sweep position) and stow. Rotate handle guard to stowed position.

7. Right throttle .... OFF

   Check hydraulic transfer pump operation in the combined-flight direction with the HYD PRESS, OIL PRESS, R GEN and R FUEL PRESS caution lights illuminated.

8. Ejection seats .... SAFE

   Raise guards on face curtain and secondary firing handles to lock seat actuation devices.

9. Ordnance .... DEARM (field)

   Dearm and safety ordnance in accordance with local field operating procedures.

.... WHEELS CHOCKED ....

POST LANDING-PILOT - (Continued)

10. Parking brake handle .... SET (aft)

    Do not set parking brake subsequent to a field landing if the brakes have been used extensively.

11. UHF radio .... OFF

12. Oxygen .... OFF

    After removing mask turn oxygen OFF.

13. Left throttle .... OFF

    Upon signal from plane captain secure left engine. Check emergency generator auto operation upon shutdown.

14. Lights .... OFF

    Turn off internal and external light switches.

15. Canopy .... OPEN

POST LANDING-FLIGHT OFFICER

1. Ejection seat .... SAFE

   Raise guards on face curtain and secondary firing handles.

2. Oxygen .... OFF

3. UHF/data link .... OFF

4. TACAN .... OFF

5. IFF .... OFF

6. Attitude gyro .... CAGE

7. Mission recorder .... ON

8. OBC/BIT .... CHECK

   Note discrepancies of system.

POST LANDING-FLIGHT OFFICER - (Continued)

9. BIT run sequences (1) thru (4) .... CHECK

   Note discrepancies of system.

10. Mission recorder .... OFF

11. NAV mode .... OFF

   Check present position prior to turning INS to off.

12. ECM display panel .... SWITCHES OFF

13. Radar beacon .... OFF

14. ALQ-28 .... OFF

15. Console/instrument lighting .... OFF

POST FLIGHT INSPECTION

(U) Except for carrier operations the aircrew should inspect the aircraft for notable discrepancies after the flight. The aircrew will ensure that all aircraft and weapons systems discrepancies are completely and clearly reported on prescribed forms to facilitate maintenance corrective action. All failures indicated by OBC/BIT legends should be likewise reported.

HOT REFUELING

(U) Prior to commencing ground hot refueling operations a qualified ground crewman shall inspect the exterior of the aircraft for any discrepancies that might be hazardous to refueling or further flight operations. One ground crewman shall remain in a position on the starboard side of the aircraft within view of both the pilot and refueling crew. The occurrence of any hazardous condition requires the immediate termination of refueling operations.

(U) Subsequent to refueling the aircrew should refer to check lists appropriate to configure the airplane for take-off or shutdown depending on intentions.

1. Fire extinguisher equipment .... AVAILABLE
2. All emitters .... STBY/OFF
3. Wing sweep .... 68°
4. Right throttle .... OFF
5. Left throttle .... IDLE
6. Wheels ....CHOCKED
7. Parking brake .... SET (Aft)
8. WING/EXT TRANS switch .... AUTO

F-14A KNEEBOARD CHECKLIST - PILOT

PRE-START

LANDING GEAR - DN
WING SWEEP - EMERG/CORR
STBY ATTITUDE - SET
COMPASS - ALIGNED
MASTER TEST
- LTS
- FIRE DET
- INST
- MACH LVR
- WG SWP

ACM PANEL - SET
EMERG FLT HYD - CHECK

ENG START

ENG CRANK - R
20% RPM-R THROT IDLE
- ENG CRANK
- R GEN
- R FUEL PRESS
- RPM
- TIT
- FUEL FLOW
- NOZZLE
- OIL PRESS
- HYD PRESS

GND PWR/AIR - DISCONNECT
- HYD PRESS
- AUX BK/SKID

GEAR PINS - REMOVED
X-BLEED START - L ENG
- REPEAT ENG CRANK - L
- HYD PRESS
- LIGHTS

PRE-TAXI

COMM/DL/NAV - ON
DISPLAYS - ON
RADAR ALT - ON
P/R/Y STAB AUG - ON
TRIM - 0°
MASTER TEST
- INST
- EMERG GEN
- OBC

PRE-TAXI CONT'D

- FLT GR DN
- FLT GR UP
- D/L RAD
- OFF

COMPONENT CYCLES
- HOOK
- SPEED BRAKES
- AR PROBE
- WING SWEEP - 68° EMERG
- FLIGHT CONTROLS

RADAR ALT - TEST
FLIGHT INSTRUMENTS - CHECK
ANTI-SKID/SP BK - SET

TAXI

PARKING BRAKE - RELEASE
ANTI-SKID - TEST
NWS/BRAKES - CHECK
WING SWEEP - 20° AUTO
MAN FLAPS - 10°
FLAPS/SLATS - DN
SPOILERS - CHECK

TAKE-OFF CHECKLIST

LANDING CHECKLIST

POST LANDING

ANTI-SKID - OFF
FLAPS/SLATS - UP
SPEED BRAKE - RETRACT
WING SWEEP - 68° OVSP ENABLE
AVIONICS - OFF
WING SWEEP - 75° OVSP
R THROTTLE - OFF
SEATS - SAFE
– – WHEELS CHOCKED – –
PARKING BRAKE - SET
UHF - OFF
L THROTTLE - OFF
LIGHTS - OFF

1 April 72

part 4 Field-Based Procedures

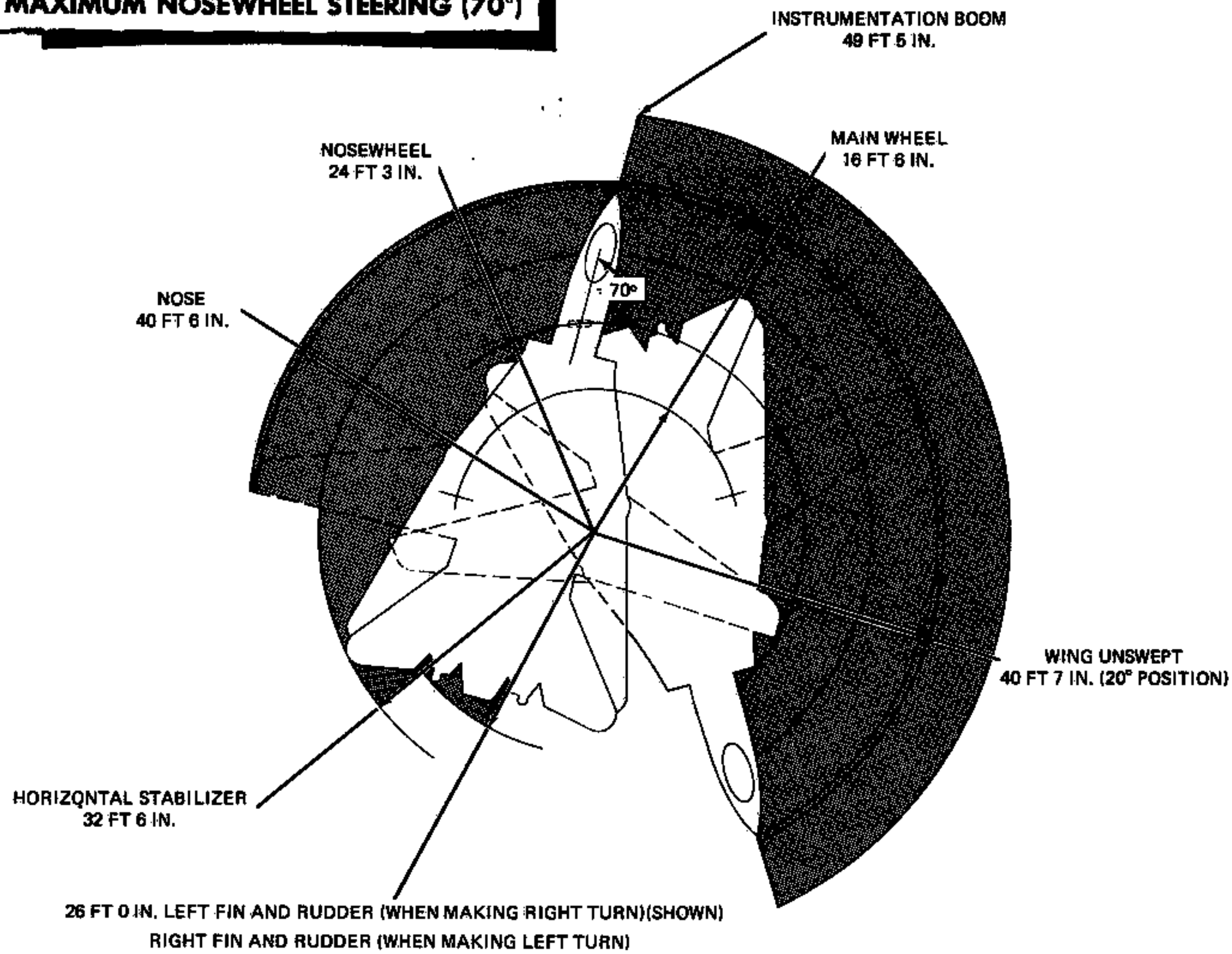
TAXIING

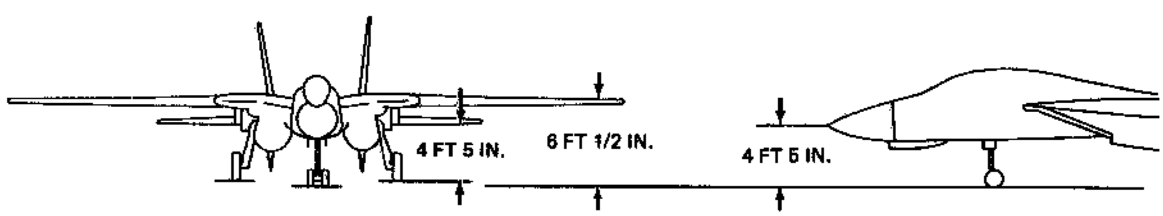
(U) Prior to releasing the parking brake the aircrew should ensure that the inertial platform is aligned. Subsequent to releasing the parking brake and while depressing the toe pedal brakes the plane captain should depress the anti-skid test button on the skid control box in the nose wheel well. Engagement of nose wheel steering in the static position should be effected to check steering control and polarity. At a static position only small steering deflections of the nose wheels can be effected because of limited steering actuator torque. Proper operation of the wheel brakes should be individually checked prior to setting the aircraft in motion and ensure the brake accumulator pressure recovers to the fully charged condition. The canopy should be closed during taxi operations to eliminate canopy lateral shimmy and attendant consequences. To set the aircraft in motion starting from a static position requires the initial application of about 80% RPM. While departing the line area both aircrewmen should clear the extremities of the aircraft, and the wings should remain at the 68° sweep angle to minimize the span clearance. Once in motion IDLE thrust is normally sufficient to sustain taxi speeds and full nose wheel steering authority may be realized.

(U) Taxi speed should be maintained at a reasonable rate consistent with traffic, lighting and surface conditions. Do not ride the wheel brakes so as to prevent overheating. The taxi interval should be sufficient to avoid taxiing in another aircraft's jet wash which presents an additional FOD potential. Although the anti-skid system is armed at speeds less than 15 knots, the anti-skid system is not operative. The nose wheel steering can remain engaged throughout the taxi phase. Application of wheel brakes in conjunction with nose wheel steering should be performed symmetrically to minimize nose tire side loads. In minimum radius turns (figure 3-3) using nose wheel steering, the inboard wheel rolls backwards as the axis of rotation is between the main gear. Because of the distance from the cockpit to the main landing gear the pilot should make allowance for such in turns to prevent turning too soon and cutting corners short.

(U) Crew comfort during taxi operations is affected by the nose strut air curve characteristics which maintains the strut in the fully extended (stiff strut) position except during deceleration. Because of the wide stance of the main gear, differential application of wheel brakes is effective for turning the aircraft without the use of nose wheel steering. Subsequent to flight while returning to the line at light gross weights, one engine may be shut down to prevent excessive taxi speeds at IDLE thrust. While operating on one engine

MAXIMUM NOSEWHEEL STEERING (70°)





1-F113-004
03700-019-0

Figure 3-3

TAXIING - (Continued)

at IDLE, do not sweep the wings or command high rate horizontal tail deflections to prevent starving the wheel brakes of normal hydraulic supply pressure.

(U) On deck engine operations for extended periods of time can result in an unacceptable build-up in fluid (hydraulic, engine oil and IDG oil) temperatures by taxing heat exchanger capacities. Since the left IDG supplies the majority of the electrical power load it is more susceptible to overheating than the right. Tailwind and/or high power conditions at high ambient air temperatures increase the changes of fluid overtemperature. The pilot should take appropriate precautions during extended ground operating period to avert fluid overtemperature conditions. Since the outboard spoiler module is automatically energized with the flap handle down with weight-on-wheels it is incumbent on the pilot to restrict the amount of flaps down operation on the deck to prevent module fluid overheating. Approximately 85% RPM is optimum for cooling the engine after extended high power operations.

TAKE-OFF

(U) An engine run-up prior to applying power to take-off is not necessary unless incident to a maintenance test flight or to check a previous flight discrepancy. In such an event, care should be exercised to perform high power turn-ups in a position to avoid jet blast damage to the ground or expose landing traffic to the high velocity wash. The placarded aircraft take-off checklist should be completed prior to calling for take-off clearance and all annunciator lights should be extinguished except NWS ENGAGED. Full flaps/slats are used for all take-offs regardless of thrust or gross weight conditions. Both crewmen should be operating in HOT MIC during this phase of flight to enhance communications in event of an emergency. Upon tower clearance and after visually clearing the approach zone, the pilot should taxi onto the runway (take downwind side if another aircraft to follow) and roll straight ahead to align the nose wheel and check compass alignment.

(U) Hold in position for take-off using the toe pedal brakes with nose wheel steering engaged; do not use the parking brake to restrain the aircraft under high power conditions—tire skid might result. Anti-skid and spoiler brakes should have been previously armed so check spoilers in the drooped position upon throttle advancement for take-off. The wheel brake effectiveness is sufficient to restrain the aircraft in a static condition with both engines in minimum zone afterburning (nozzle position 2); however, the braking effectiveness is dependent on surface condition and aircraft gross weight. Upon receiving tower clearance for take-off and after clearing the area to the rear, advance both throttles to MIL and check engine instruments. The power trim indicators provide an indication of engine trim condition based on ambient air temperature and are the prime reference to ascertain that the engines are developing rated thrust. Cross check other engine instruments (RPM, TIT, fuel flow and oil pressure) to ascertain that all are within limits. For heavy weight MIL thrust take-offs where single engine flyaway performance at MIL is marginal, the ability to achieve an afterburner light on both engines should be verified just prior to brake release. For an afterburner take-off, both afterburners should be lit (nozzle position 2) before brake release. Engine instruments should be re-verified for afterburner

TAKE-OFF - (Continued)

operating conditions; the power trim indications are not valid for afterburner operating conditions. Ground engine power checks should be performed as expeditious as possible to conserve fuel. The nose strut will compress to the kneel position at high power settings when statically restrained but will recover to the static extended position upon throttle retard or brake release.

(U) Subsequent to take-off power checks and with a safe interval on the preceding aircraft, release the toe pedal brakes and use nose wheel steering to maintain directional alignment during the initial phase of take-off roll. If performing a MAX A/B take-off, immediately advance the throttles to MAX after brake release and check nozzle position indications staging towards full open (position 5). The nose strut should return to the fully extended position (+1.5° pitch attitude) upon brake release; failure to do so will increase the take-off ground roll. Use of differential braking to control directional alignment should be avoided due to its attendant effect on ground roll distance. Although rudder effectiveness for directional control is adequate at 60 KIAS, nose wheel steering can be used throughout the entire ground roll phase and the system automatically disengages upon lift-off without any pilot action.

(U) Minimum ground roll take-off procedures do not differ from the normal procedures except for the use of MAX A/B thrust. Maintain the control stick at the trimmed condition during the pre-rotation ground roll phase to minimize aircraft drag. At about 10 knots before the predicted nose wheel lift-off speed pull the control stick straight aft so that full symmetrical deflection of the horizontal tails is achieved at the predicted rotation speed. Hold the control stick full aft until achieving lift-off which will occur at approximately a 6° nose-up pitch attitude. With the wings swept forward the airplane seems to "balloon" from the runaway in a near level nose attitude with a more docile transition to flight than characteristic of swept wing aircraft. Although on-deck pitch attitude rotation in excess of 10° provides marginal tail-ground clearance, the airplane is airborne well before such a phenomenon becomes a limiting factor.

(U) After lift-off relax the aft stick force as the aircraft accelerates towards an in-trim condition. Raise the landing gear control handle after ensuring that the aircraft is definitely airborne. Pitching moments associated with gear retraction are negligible and a gear UP indication should be achieved in about 9 seconds after initiation. At approximately 160 KIAS (dependent on longitudinal acceleration) the flap handle can be placed in the UP position. A moderate nose-up pitching moment occurs during the flap/slat retraction phase which takes approximately 8 seconds. On heavy gross weight MIL thrust take-offs, the maneuvering flap thumbwheel can be held aft approximately 3 seconds after flap handle retraction to cause the high-lift devices to retract to only the 10° flap position; in this manner angle of attack does not become excessive with slow longitudinal accelerations. As the airplane accelerates to 220 KIAS the thumbwheel can be rotated forward to achieve a clean wing configuration. Do not attempt to correct immediately after lift-off to counter a lateral drift due to a crosswind condition. The use of large lateral control deflection should be avoided to keep from braking out the wing spoilers which have a negative effect on lift and drag. Differential tail authority within the spoiler deadband (±1/2 inch lateral stick deflection) is

TAKE-OFF - (Continued)

adequate for maintaining wings level flight or effecting gradual turns with symmetric thrust. Prior to reaching the flap (225 KIAS for >10° flaps) and gear (280 KIAS) limit speeds the pilot should ascertain that all devices are properly configured for higher speed flight. The combined hydraulic system non-flight essential components (gear, brakes, nose wheel steering and hook retract) may be isolated by selecting FLT on the hydraulic isolation switch. A gradual climbout pitch attitude should be maintained until intercepting the optimum climb speed. A recheck of engine instruments and configuration status should be performed after clean-up during the climbout phase.

(U) In preparation for an abort situation during the take-off roll the pilot should be familiar with the placement and capacity of all field arresting gear. To execute an abort during the take-off ground roll the pilot need only retard the throttles to IDLE (spoiler brakes automatically deploy); full speed brake deployment and full aft stick may be employed to increase drag at high abort speeds. The extent of braking should be tailored to ground velocity and runway distance remaining. Refer to Section I for maximum recommended brake application speeds. Loss of an engine during the take-off phase can be adequately compensated for with rudder and lateral control at speeds down to 15 units AOA with MAX A/B thrust on the operating engine.

PATTERN ENTRY

(U) Entry to the field traffic pattern will be at the speed and altitude prescribed by local course rules. The initial portion of the placarded landing checklist should be completed before arriving at the break position. A pattern entry speed of 250 KIAS provides for a sharp break and deceleration to the flaps down limit speed (220 KIAS) by the time the airplane is rolled out on the downwind leg. Maneuver flaps/slats and speed brakes may be deployed in the break at the discretion of the pilot. The remainder of the placarded landing checklist should be completed prior to arriving at the 180° position abeam the touchdown point with the aircraft stabilized at the recommended approach angle of attack.

LANDING

(U) At the abeam position for landing, the aircraft should be on the prescribed altitude and approach angle of attack with the landing checklist completed. The recommended approach angle of attack should be crosschecked with gross weight and indicated airspeed in wings-level flight to verify approach reference accuracy (figure ____). The turn off the 180° position should be judged based on surface wind conditions and interval traffic (type, pattern, touch-and-go or final landing, etc.) so as to allow sufficient straightaway on final prior to touchdown.

(U) Maintain the approach reference angle of attack in the approach pattern until touchdown regardless of auto throttle usage. In gusty air conditions it is prudent to increase the approach speed to compensate for peak variations in wind velocity. The quality of the approach and touchdown is enhanced by starting from an on speed and altitude condition. The low

LANDING - (Continued)

thrust required in the landing approach leaves little margin for rapid deceleration or altitude loss and it is incumbent on the pilot to precisely control these parameters from the onset of the approach to touchdown point.

(U) Control surface effectiveness in the final approach is more than adequate to cope with all exigencies, however, the control system gearing provides for small precise commands without overcontrol tendencies. Inertia and control moment arm for pitch attitude control in conjunction with engine thrust response and authority characteristics dictate that the pilot make small precise corrections on the glide slope for the most effective control technique. Overcontrol tendencies may produce the same end result as the pilot attempts to quicken the airframe response but in the final analysis he is needlessly perturbating the airplane and increasing his workload. Lateral overcontrol produces spoiler breakout which upsets the equilibrium in the pitch axis. Directional control coordination is not specifically required and on final the pilot should rest his feet on the rudder pedals and not attempt to make directional axis corrections except under crosswind or single engine approach conditions. Because of the low approach speed range the crab angle to offset lateral drift in a crosswind landing is larger than for aircraft approaching at higher speeds; just prior to touchdown the pilot should kick out the crab so as to be directionally aligned with the runway at touchdown.

(U) The landing should be planned for the downwind side of the runway with traffic behind, the opposite side of nearest traffic on landing rollout, or on the turn-off side of the runway when the surface wind condition is of little consequence. Pilots should practice flying on the field optical landing aid system whenever possible. Fly the airplane down to the deck and refrain from flaring so as to accurately establish a touchdown point and achieve initial compression of main gear struts to arm the anti-skid and spoiler brakes. With the wings forward ground effect is pronounced, and sufficient controllability is provided so that corrections can be made in ground effect. Pitch attitude at touchdown or during rollout should not exceed 10° to avoid tail-ground clearance problems. Since the minimum nose wheel lift-off speed is less than the recommended approach speed, pitch attitude can be maintained after touchdown without the nose falling through. The automatic deployment of spoiler brakes upon initial main strut compression causes a slight nose up pitching moment at high pitch attitudes. For minimum ground rollout apply large symmetric forces on the brake pedals and concurrently pull the control stick full aft after spoiler brake deployment with the throttles chopped to IDLE. Directional control during rollout requires some differential braking; do not pump the brakes, but modulate the pressure about the depressed condition. Nose wheel steering may be used during the rollout but it must be engaged with the rudder pedals centered to avert a directional swerve upon engagement. Brake pressure should be eased as the aircraft decelerates to a stop as the anti-skid is not operative at wheel rotation speeds less than 15 knots. Follow the post-landing checklist for proper configuration clean-up procedures. Clear the area behind before turning off across the runway. With the wings at 75° the right engine may be shut down to reduce residual thrust during low gross weight taxiing.

(U) For touch-and-go landings, MIL thrust is applied subsequent to touchdown and automatic retraction of the speed brakes occurs to configure the aircraft for a go-around.

LANDING - (Continued)

Control for rotation is greater than experienced on take-off although the aircraft has the same basic lift-off characteristics. Fuel required per pass is nominally 300 pounds contingent on traffic pattern congestion. Procedures for a single engine approach are the same as twin engine except that pilot directional control inputs are necessary to counteract the asymmetric thrust moments to maintain directional alignment. For a single engine go-around the asymmetric thrust moment is more pronounced because of the high thrust involved; significant rudder pedal deflections are required to maintain directional alignment and a wing down (into operating engine) technique can be employed to enable the lift vector to provide some compensation if a straight flight path is desired.

section VII
COMMUNICATIONS EQUIPMENT AND PROCEDURES

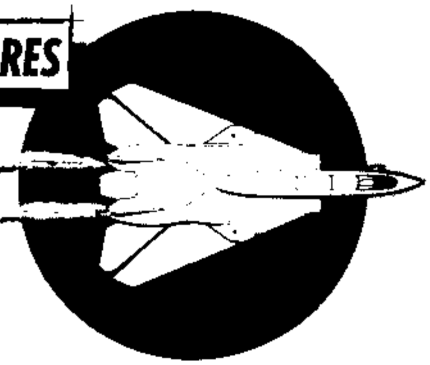


TABLE OF CONTENTS

COMMUNICATIONS AND ASSOCIATED EQUIPMENT

(U) Figure 7-1 (sheets 1 and 2) lists the communication-navigation-identification (CNI) equipment associated with the aircraft weapon systems. For information concerning defensive electronic countermeasure (DECM) equipment, refer to Supplementary Flight Manual NAVAIR 01-F14A-1A, Section VIII, Part 4.

CAUTION

Operation of electronic equipment for more than 5 minutes without adequate cooling air will result in permanent damage to equipment.

COMMUNICATIONS AND ASSOCIATED EQUIPMENT

Type and Designation	Function	Range	Operator	Location of Controls
Intercom	Provides voice communications between crew members, applies radio audio signals and various warning tones to the crews' headsets.	Within the aircraft	Pilot and NFO	Pilot and NFO left side consoles
TACAN AN/ARN-84(V)	Provides bearing and distance information.	Line-of-sight up to 300 miles, depending on altitude	Pilot and NFO	Pilot right side console NFO left side console
UHF Data Link AN/ASW-27B	Provides two-way digital message communication.	Line-of-sight up to 180 nautical miles	Pilot and NFO	Pilot and NFO left side consoles
UHF Communications Set AN/ARC-51A	Provides two-way voice communications and ADF receiver.	Line-of-sight up to 180 nautical miles	Pilot and NFO	Pilot and NFO left side consoles
UHF Auxiliary Receiver AN/ARR-69	ADF and auxiliary voice receiver.	Line-of-sight up to 180 nautical miles	Pilot and NFO	Pilot and NFO left side consoles

Figure 7-1. (Sheet 1 of 2)

COMMUNICATIONS AND ASSOCIATED EQUIPMENT

Type and Designation	Function	Range	Operator	Location of Controls
UHF Direction Finder AN/ARA-50	Provides bearing information to selected UHF stations.	Line-of-sight up to 180 nautical miles	Pilot and NFO	Pilot and NFO left side console
Voice Security Equipment KY-28	Cryptographic encoding/decoding of voice communications. Used with Main UHF communications radio.	Line-of-sight up to 180 nautical miles depending on altitude	NFO	NFO right side console
IFF Transponder AN/APX-72	Responds to interrogations by other aircraft or ground stations.	Line-of-sight up to 300 nautical miles	NFO	NFO right side console
IFF Interrogator AN/APX-76A	Requests identification from other aircraft.	Line-of-sight	NFO	DDD and NFO right side console
Receiver Decoder AN/ARA-63	Provides glide-slope signals for carrier landing system.	Line-of-sight up to 20 nautical miles	Pilot	Pilot right side outboard console
Radar Altimeter AN/APN-194	Displays height above earth's surface.	0 to 5,000 feet	Pilot	Radar altimeter indicator on pilot's instrument panel
Radar Beacon AN/APN-154	Aids in tracking by ship and ground-based X-band radars. Provides down-link for automatic carrier landing system.	Line-of-sight up to 300 nautical miles	NFO	NFO right side outboard console

Figure 7-1 (Sheet 2 of 2)

COMMUNICATIONS ANTENNAS

(U) Four UHF/L-band dual-blade antennas provide omnidirectional coverage for UHF voice, UHF data link, TACAN, and IFF/SIF transponder (APX-72) operation. TACAN and UHF voice communications use one set of antennas; the data link and IFF transponder, another set of antennas. (Refer to GENERAL ARRANGEMENT illustration for antenna locations.) The IFF interrogator antenna (APX-76) is an integral part of the AWCS antenna.

(U) Each individual system is connected to the appropriate portion of an upper or lower antenna through a coaxial switch and diplexer. For UHF voice communications, the COMM ANT switch on the integrated control may be used to select the upper or lower antenna manually, or to select automatic actuation and lock-on of the first antenna to receive a usable signal. The data link antennas are selected manually; an upper or lower antenna is selected by means of the D/L ANT switch on the integrated control.

(U) The upper TACAN antenna is located aft of the canopy on the turtle deck, and the lower antenna is imbedded in the left vertical fin. Only one antenna is used at a time. Automatic switching between antennas prevents loss of TACAN information. If a signal is lost or is too weak to hold receiver lock-up, the TACAN automatically cycles between the two antennas every 6 seconds seeking a stronger signal. During this cycling and search period, memory circuits retain range tracking for 8 to 12 seconds and bearing tracking for 8 seconds.

(U) The IFF antenna lobing switch is controlled by a IFF ANT switch on the NFO right outboard console. In the AUTO position, the lobing switch cycles the receiver-transmitter between upper and lower antenna pattern coverage. In the LWR (lower) position, only the lower antenna pattern is used to receive and transmit interrogation signals. The upper antenna pattern has a slight forward tilt; the lower pattern a slight aft tilt.

COMMUNICATIONS

INTERCOMMUNICATIONS (ICS)

(U) The ICS provides normal, backup, or emergency communications between crewmembers. It also combines and amplifies audio signals received from other electronic receiving equipment; ie, ECM, Sidewinder tone, IFF/SIF, radar altimeter, and voice radios.

(U) Identical ICS control panels (figure 7-2) are on the pilot and NFO left side consoles. The ICS consists of four amplifiers, two at each cockpit station, which permit duplex operation during normal operation. If one amplifier fails, it may be bypassed by selecting either the B/U (backup) or EMERG (emergency) position on the ICS control panel. This permits continued ICS operation. If two amplifiers fail at the same station, intercommunication is impossible.

(U) An external interphone connection is installed on the right side of the forward bulkhead in the nosewheel well of aircraft numbers 1, 2, 3, 7 and 9 (flight test) only. Ground personnel can communicate with the cockpit stations, when the NFO ICS switch is set to HOT MIC.

INTERCOMMUNICATIONS

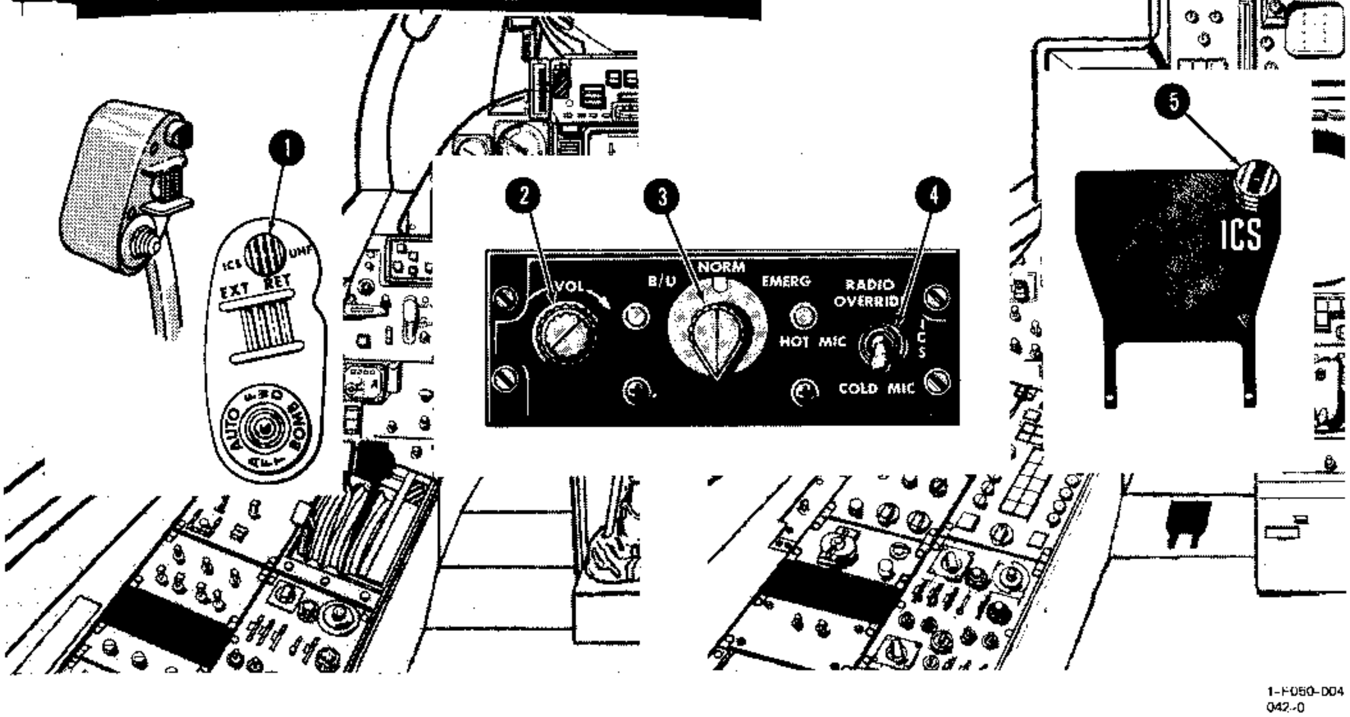


Figure 7-2

NOMENCLATURE	FUNCTION
(1) PILOT INTERCOM KEYING momentary switch	Aft position permits intercom when COLD MIC is selected on the Function Selector Control.
(2) VOLUME CONTROL	Clockwise rotation increases intercommunications audio level at cockpit station where the control is actuated. Audio level at other station not affected.
(3) AMPLIFIER SELECTOR switch	B/U - (Backup) used to bypass a faulty amplifier. NORM - (Normal) used when all amplifiers are functioning properly. EMERG - (Emergency) used to bypass faulty amplifier.

NOMENCLATURE		FUNCTION	
4	Function Selector control	OVERRIDE -	Attenuates non-critical (see Fig. 7-4) radio audio to permit intercom when intercom is urgent.
		HOT MIC -	Permits intercom without keying.
		COLD MIC -	Permits intercom only when crewmen actuates ICS keying switch.
5	NFO INTERCOMM KEYING momentary button		Permits intercom if COLD MIC is selected on the Function Selector Control.

AUDIO SIGNALS

(U) Audio warning signals associated with the weapon system are available to either or both crewmen through the ICS. Each signal has a distinct tone. A visual display is associated with every audio signal so that the aircrew can expect the tone and thus interpret its meaning. Most audio signals may be attenuated or turned off if not required, allowing the aircrew to concentrate on more critical tones. Critical warning tones cannot be attenuated by any mode of ICS operation, except by means of the volume control on the ICS panel, which attenuates all audio signals together.

(U) The TONE VOLUME control panel (figure 7-3) on the pilot left side console contains three volume controls for regulating audio signals from the APR-25, APR-27, and a Sidewinder missile release (SW). Clockwise rotation of a knob increases the audio signal in the pilot's headset. Full counterclockwise rotation turns it off. The NFO has corresponding volume controls available to him on the control panels for these three equipments.

(U) Figure 7-4 provides a glossary of audio warning signals available within the aircraft weapon systems.

TONE VOLUME CONTROL PANEL

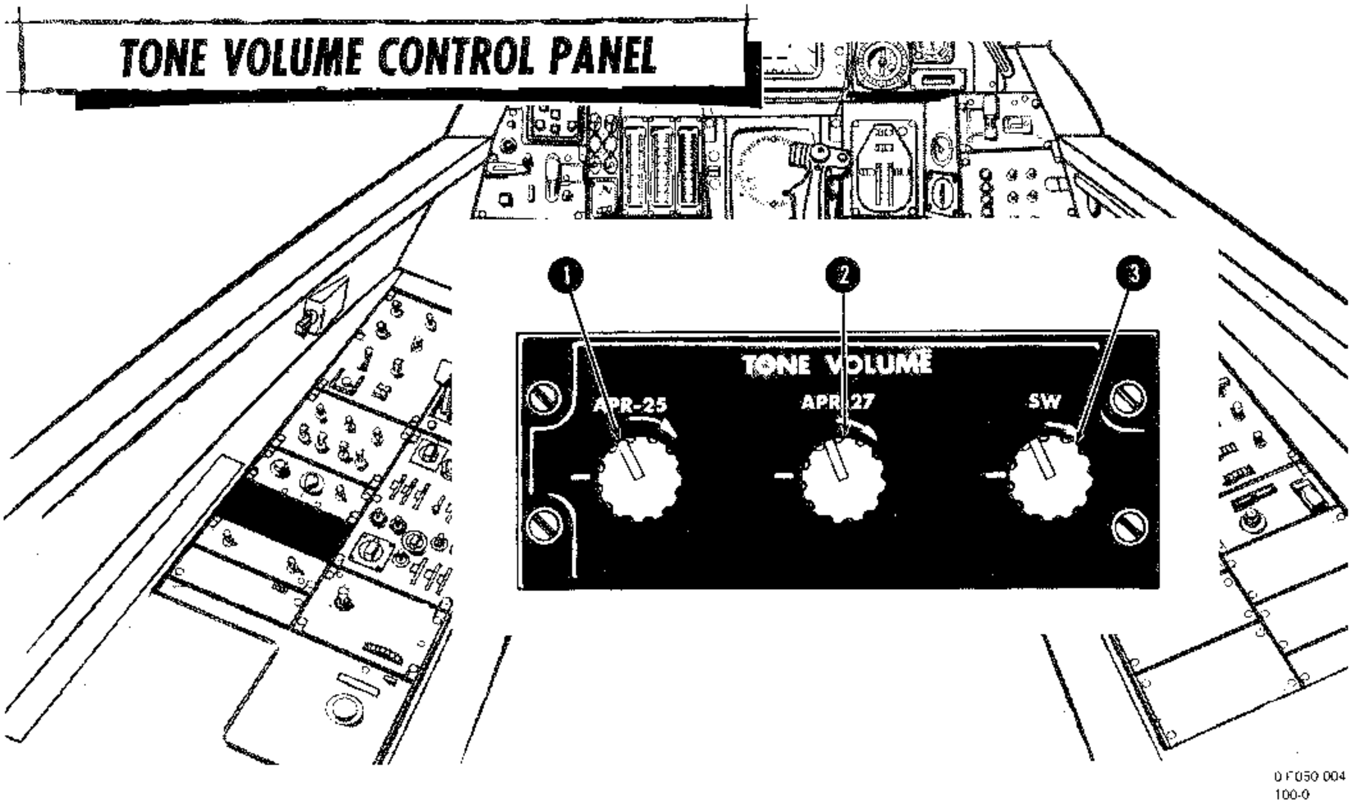


Figure 7-3

NOMENCLATURE	FUNCTION
(1) APR-25 VOLUME CONTROL	Clockwise rotation increases tone in pilot's headset. Full counterclockwise rotation turns tone off.
(2) APR-27 VOLUME CONTROL	Clockwise rotation increases tone in pilot's headset. Full counterclockwise rotation turns tone off.
(3) SW VOLUME CONTROL (SIDEWINDER)	Clockwise rotation increases missile tone in pilot's headset. Full counterclockwise rotation turns tone off. MASTER ARM switch must be ON for tone to be available.

GLOSSARY OF TONES

Tone	Critical	Crew Position	Controls	Function	Characteristics
AWCS IR	NO	NFO	AWCS control panel Volume control on DDD panel	Presence of IR target	Low to high frequency rising in pitch with amount of heat detected
SIDEWINDER	NO	PILOT	Tone volume panel OFF switch on armament jettison panel	Missile acquisition	High frequency, increases in intensity with positive lock-on
AN/APR-25	NO	PILOT NFO	Tone volume panel (Pilot) ECM panel (NFO)	Threat radar detection	Low to high frequency, determined by scan rate and prf of threat radar
AN/APR-27	YES	PILOT NFO	Tone volume panel (Pilot) ECM panel (NFO)	Missile launch is imminent	Low to high frequency warble when tone is present
AN/ALQ-100 (MOD)	NO	NFO	DECM control panel	Identifies presence and type of radar	Raw prf sound
RADAR ALTIMETER	YES	PILOT NFO	Radar Altimeter indicator	Below preset altitude	1,000 Hz tone, modulated at 2 pulses per second, lasting for 3 seconds.
APX-72	NO	NFO	IFF control panel	Valid mode 4 interrogation	PRF of interrogation pulse. 2,000 and 6,000 Hz
APX-76 (A/C 1 thru 20 only)	NO	NFO	M4 alarm override switch	Indicates interrogated transmission	400 Hz
TACAN	NO	PILOT NFO	TACAN control panel	Station identification	International Morse code with three-letter designation

Figure 7-4

UHF RADIO (AN/ARC-51A)

(U) The main UHF radio provides air-to-air and air-to-surface voice communications. Radio frequency range extends from 225.00 to 399.95 MHz. The equipment allows transmission and reception on any one of 20 preset channels and a guard channel (243.0 MHz). Guard frequency may be monitored simultaneously with any other selected frequency. Manual selection of 3,500 channels in 50 KHz increments is also provided. Dual antenna installations provide reliable line-of-sight communications up to 180 nmi depending on altitude, with an average output of 20 watts. The UHF radio may also be used in conjunction with the automatic direction finder (AN/ARA-50) and with the KY-28 for secure voice communication. Plain or secure voice capability is provided by selecting either plain or cipher mode on the KY-28 panel. A jumper is provided for enabling the UHF radio when the KY-28 is not installed. When the KY-28 is installed, the UHF radio will not operate if the KY-28 power switch is OFF.

(U) Identical control panels (figure 7-5) for the UHF radio are located on the pilot and NFO left side consoles. A remote control indicator on each instrument panel indicates the preset channel selected by the crewman who has command. The pilot's UHF keying switch is located on the inboard throttle grip; the NFO's keying switch, is in the right foot well. The UHF control panel is an integrated control panel, which controls the UHF receiver-transmitter, automatic direction finder, UHF auxiliary receiver, UHF data link transceiver, and the antenna system associated with these units.

(U) The UHF command pushbutton on the COMM/NAV CMD panel transfers control of the UHF function from one cockpit station to the other. Each crewmember may either give or take control. The active frequency is the frequency selected by the crewman who has control. The remote control indicator at each cockpit station displays the preset channel being used, G for guard channel, or OFF for manual frequency. The crewmember who has UHF control is indicated by the green-lighted UHF pushbutton on his COMM/NAV CMD panel. Actuation of controls on the integrated control panel is effective only if that cockpit has control; the volume control, however is individually controlled for each cockpit station. UHF and ICS volume must be adjusted to permit reception of both simultaneously.

UHF AUXILIARY RECEIVER (AN/ARR-69)

(U) The UHF auxiliary receiver is used primarily for ADF reception; secondarily, for emergency voice reception. The AUX receiver provides ADF and voice on any one of 20 preset channels in 265.0 to 284.9 MHz frequency range plus guard channel (243.0 MHz). It has line-of-sight range, varying with altitude. Controls are located in the integrated control panels (figure 7-5). The crewmember who has main UHF control has control of the auxiliary receiver, however, the volume controls are adjustable by each crewmember. Cipher audio cannot be received with the UHF auxiliary receiver; it is incompatible with the KY-28.

UHF AUTOMATIC DIRECTION FINDER (AN/ARA-50)

(U) The UHF automatic direction finder may be used in conjunction with the MAIN UHF radio or the AUX Receiver, depending on the setting of the MAIN/AUX UHF function selector switch. ADF provides relative bearings to transmitting ground stations or other aircraft. It can receive signals on any one of 20 preset channels or on any one of 3,500 possible manual set frequencies in the 225 to 400 MHz range, depending on which receiver is used. The system is integrated with the auxiliary receiver. Controls are located on the integrated control panels (figure 7-5).

(U) The system has a line-of-sight range, varying with altitude. Operating power is 115 volts from the essential No. 2 bus, 28 volts from the essential No. 2 bus, and 26 volts ac through the NFO circuit breaker panels. The system uses the ARA-48 ADF antenna. Bearing to transmitting stations is displayed on the pilot/NFO BDHI (No. 1 needle), pilot HSD, and NFO ECMD. Operation of the ADF is automatic, depending on the position of the function selector switch on the MAIN UHF control panel (figure 7-5). The ADF signal is interrupted during UHF transmissions.

DATA LINK SYSTEM (AN/ASW-27B)

(U) The Digital Data Communications Set AN/ASW-27B is an airborne high speed, two-way digital communications terminal which is part of the Digital Data Communications/Control Network, AN/USC-2. Control of this network rests with the Naval Tactical Data System (NTDS) on board a carrier, the Airborne Tactical Data System (ATDS) on board an Early Warning aircraft, such as the E-2; or the Marine Tactical Data System (MTDS) installed in a wheeled van along with the TRQ-27. The AN/ASW-27B receives and transmits computer generated digital messages over FM, UHF radio, linking the F-14A computers with the controlling computers of the NTDS, ATDS or MTDS. Each control message received by the F-14A can trigger an automatic reply message back to the controlling system. The control messages contain information for single and multiple intercept vectoring, precision course direction and automatic carrier landings. Reply messages contain aircraft status (weapons, fuel, etc.) aircraft position (INS or TACAN data) and target data.

(U) One Tactical Data System (TDS) can control many F-14A aircraft on a single frequency since each aircraft has a unique address and only control messages containing the correct address will be accepted by a particular aircraft. At the most rapid message rate, a controlled aircraft will receive one control message and transmit one reply message every 96 milliseconds or a two-way message rate of 10 per second. At this rate, three aircraft can be controlled by a single TDS. If, for example, 30 aircraft are to be controlled by one TDS, each controlled aircraft will receive one message per second. In this manner Data Link communication can use any one of 250 frequency channels spaced at 100 KHz intervals from 300.0 MHz to 324.9 MHz. Communication can be maintained at line-of-sight ranges up to 180 nautical miles.

(U) To intercept multiple targets, the F-14A is sent control messages containing air target information. Two control messages (No. C1 and C2) are required to give complete information on one target. This information is routed directly to an AWG-9 computer target track file and displayed on the Tactical Information Display (TID) with a Data Link (D/L) symbol to differentiate it from AWG-9 radar received targets. Eight Data Link track files are available in the AWG-9 computer. Therefore as many as eight (8) single targets in successive C1 and C2 control messages may be sent to the F-14A: hence, multiple targets are given to the F-14A. Reply message R3 is returned in response to control message C1. This reply message contains target information stored in the F-14A AWG-9 computer in the particular track file number requested in the control message C1. Also, F-14A radar targets may be reported to the ATDS with this reply message by selecting D/L and pushing the REPORT button on the computer address panel. This target will then appear on the ATDS display with a symbol indicating it was reported by D/L controlled aircraft. In this manner rapid exchange of target information is accomplished providing a capability of conducting tactical operations in a complex threat environment against multiple targets.

(U) The TDS may send information to vector the F-14A to a single target or intercept point by means of control messages C3 and C9. These messages contain information on the course, altitude and speed that the F-14A should fly to reach an intercept point specified in range and bearing. This information is routed to the pilot's displays (VDIG) which are programmed to accept it if the pilot selects CRUISE or A/A mode and VEC steering command on the pilot's displays control panel. The F-14A replies with reply message R0 and R1 alternately. The R0 message contains F-14A magnetic heading, weapons on board, altitude and fuel quantity. The R1 message sends F-14A position in relation to a TACAN station.

(U) Control message, C19 sends vectoring information to the F-14A to guide it to an intercept point over land where a TPQ/27 MTDS can acquire control of the F-14A sending control message C-18. This message sends vertical and lateral glide slope error and pitch and bank commands to fly the F-14A on a precision course for a bombing run. For Data Link bombing the AWL-PCD steering command button and the A/G mode button must be selected on the pilots displays control panel and the ATTACK MODE selector on the NFO's armament panel must be in D/L BOMB position. The auto pilot (AFCS) may be engaged for automatic aircraft response to the D/L pitch and bank commands. The F-14A replies with R0 and R1 alternately.

(U) All weather carrier landing (AWCL) can be accomplished in either an automatic or manual mode using data link. Control message C5 vectors the F-14A to the landing window where the AN/APN-42 radar acquires the aircraft. The NTDS then transmits control message C6 which contains vertical and lateral glide slope error and pitch and bank commands for the autopilot. The glide slope errors are displayed for the pilot when he selects LDG mode and AWL-PCD steering command on the displays control panel. The autopilot may be engaged to the D/L pitch and bank commands for an automatic "hands-off" landing or the pilot may fly the F-14A to nullify the displayed glide slope errors for a manual landing.

UHF AND DATA LINK CONTROL PANEL

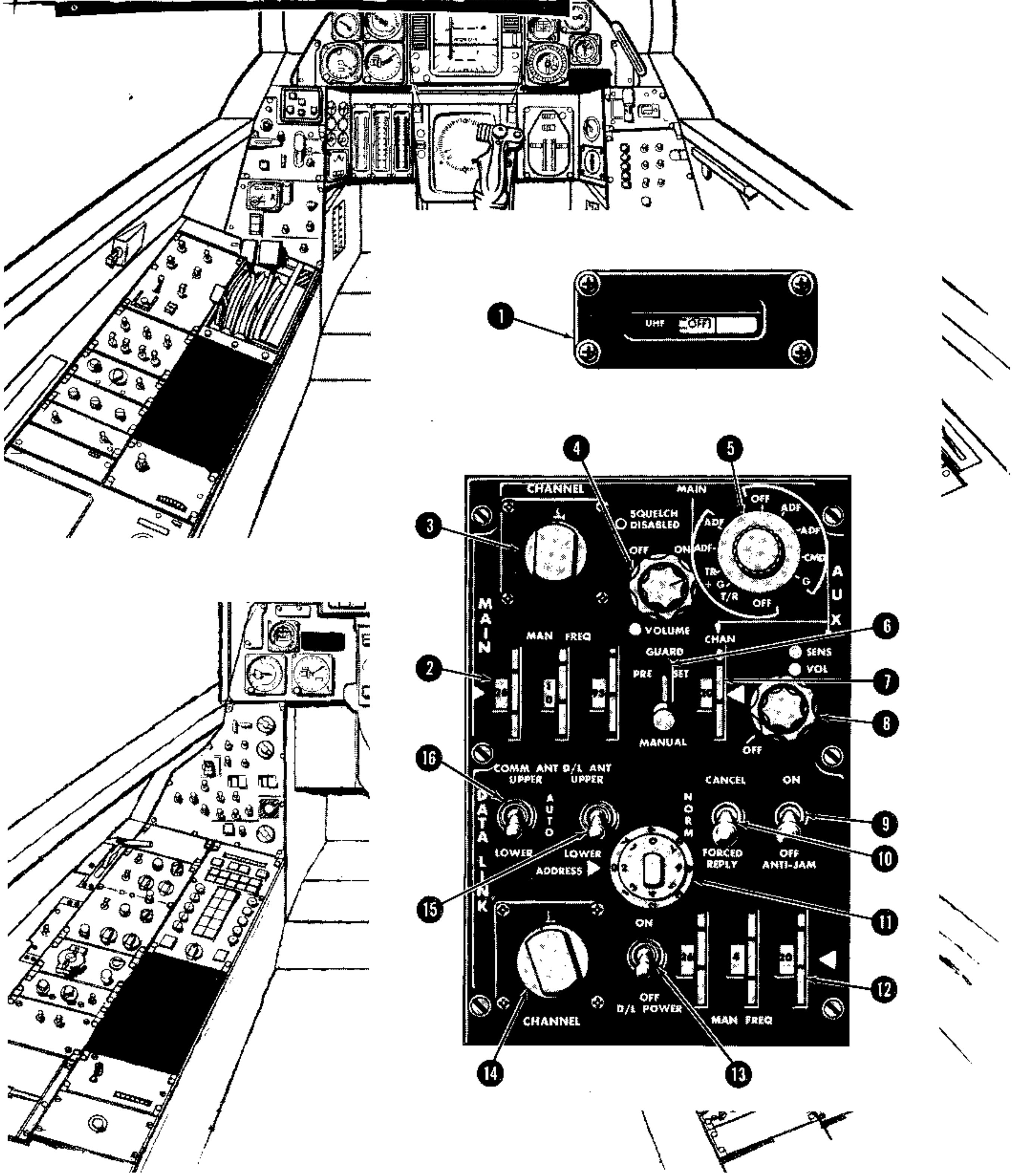


Figure 7-5

UHF and Data Link Integrated Control Functions

NOMENCLATURE		FUNCTION
1	REMOTE CHANNEL INDICATOR	Displays channel selected by crew member who has UHF command if preset mode is used or OFF in manual mode or G if guard channel is used. Readout will be identical at both cockpit stations.
2	MANUAL FREQUENCY SELECTOR	Manual frequency set in with rotation of three thumbwheels, provides 3500 possible frequencies. GUARD-PRESET-MANUAL switch must be in MANUAL.
3	PRESET CHANNEL SELECTOR	Permits selection of one of 20 preset channels. GUARD-PRESET-MANUAL switch must be in PRESET.
4	UHF VOLUME CONTROL	Rotary control, inner knob; clockwise rotation increases main UHF audio level.
4	SQUELCH DISABLE CONTROL	Rotary control, outer knob, clockwise rotation increases UHF receiver sensitivity.

		POSITION		CAPABILITY	
		MAIN	AUX	MAIN	AUX
5	FUNCTION SELECTOR CONTROL (MAIN and AUX)	OFF	OFF	None	None
		T/R	ADF	Transmit and receive	Receive ADF
		T/R +G	ADF	Transmit and receive; guard receive	Receive ADF
		ADF	CMD	ADF and guard receive; UHF transmit with ADF interrupt	Receive ADF voice receive within frequency range
		ADF	G	ADF receive; UHF transmit with ADF interruption	Guard voice and receive ADF

NOMENCLATURE		FUNCTION	
6	GUARD-PRESET-MANUAL FREQUENCY SELECTOR	GUARD -	Permits rapid selection of guard channel.
		PRESET -	Permits selection of one of 20 preset UHF channels on the preset channel selector.
		MANUAL -	Permits selection of one of 3500 possible frequencies set on by the manual frequency selector.
7	AUX CHANNEL SELECTOR		Rotation of thumbwheel selects one of 20 preset AUX channels in 265 to 285 MHz range.
8	AUX VOLUME CONTROL		Clockwise rotation of inner knob increases AUX receiver audio level.
8	AUX SENS CONTROL		Clockwise rotation of outer knob increases AUX receiver sensitivity.
9	ANTI-JAM SWITCH	ON -	All incoming messages are received including those with incorrect parity.
		OFF -	Messages with incorrect parity are rejected.
10	REPLY SWITCH	CANCEL -	Inhibits reply message transmission.
		FORCED -	Forces reply message transmission.
		NORM -	Allows control message to decide whether reply message is required.
11	ADDRESS SELECTOR		Permits selection of two least significant octal digits. Most significant digits are preselected at installation.
12	MANUAL FREQUENCY SELECTOR		Permits manual frequency selection for data link in 300 to 324.9 MHz range.

NOMENCLATURE		FUNCTION	
(13)	D/L POWER SWITCH	ON -	Applies power to converter, transceiver, DDI, and command airspeed index marker on airspeed/Mach/indicator.
		OFF -	Removes power from the system. Inhibits sending or receiving data link messages.
(14)	PRESET CHANNEL SELECTOR		Permits selection of one of 19 preset channels for data link or position (M) for thumbwheel selection.
(15)	D/L ANT SWITCH	UPPER -	Selects upper data link antenna.
		LOWER -	Selects lower data link antenna.
(16)	COMM ANT SWITCH	UPPER -	Selects upper UHF antenna.
		LOWER -	Selects lower UHF antenna.
		AUTO -	Automatically selects usable rf signal.

(U) Controls for the data link are at both cockpit stations on the integrated control panel on the left side inboard console. The (DDI) on the NFO right side outboard console, decodes and displays individual message discretes in abbreviated words. In addition, certain discretes are repeated by advisory lights located on each side of the pilot's VDI. Electrical power for the data link receiver and transmitter is supplied from the left main ac bus, with circuit protection on the NFO left main ac circuit breaker panel.

Operation

(U) To place the data link system in operation, proceed as follows:

1. Address selector . . . . AS BRIEFED
2. Channel selector/manual frequency . . . . SET
3. D/L power switch . . . . ON
4. Reply switch . . . . NORM
5. D/L antenna switch . . . . AS REQUIRED
6. Anti-jam switch . . . . OFF

NOTE

If the TILT legend on the DDI is illuminated, recycle anti-jam switch or select the other antenna position.

DIGITAL DATA INDICATOR (DDI)

(U) The digital data indicator (DDI) (figure 7-6), an advisory light panel on the NFO right side outboard console, displays 40 legends that advise the aircrew of actions to be taken during a data link flight mission. The DDI accepts signals from the data link, translates them into words or abbreviations and illuminates the proper display on the advisory light. Since the pilot requires only the most important messages relative to aircraft operation, repeater indicators on the left side of the VDI provide legends concerned with traffic control and ACL. These signals are derived from data link coded discrete messages in the form of a four-bit binary code.

(U) The messages to be displayed are divided into three groups. Discrete legends, their assigned group, and message definition are listed in figure 7-7. Only one discrete message in group 1 can be displayed at a time. It is cancelled when another group 1 message is received, or when power to the DDI is interrupted.

(U) Group 2 is subdivided into three groups: 2A, 2B, and 2C. One discrete message from each of these groups may be simultaneously displayed. Each message in these groups are individually self cancelled after an interval of 30 ± 5 seconds, or are mutually cancelled if a different message within a subgroup is received.

(U) The third group is a discrete independent group. In this group all messages other than those related to the autopilot coupler can be displayed simultaneously. Only one of two messages in the autopilot coupler group (A/P CPLR and MANUAL) may be displayed at a time.

DIGITAL DATA INDICATOR

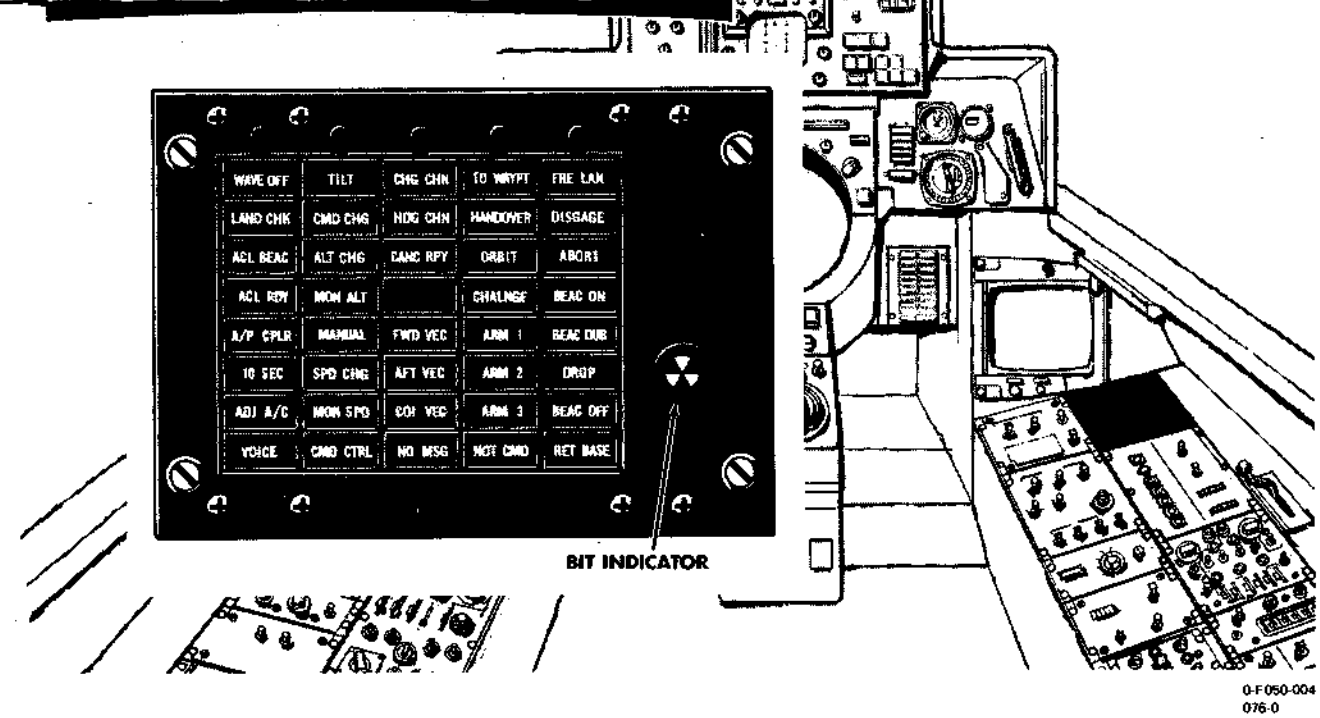


Figure 7-6

Legend	Group	Message
WAVE OFF	1	Wave-off
LAND CHK	1	Landing Checklist
ACL BEAC	1	Beacon On
ACL RDY	1	ACL Ready
A/P CPLR	INDEPENDENT	Autopilot May Be Used
10 SEC	2C	10 Seconds to Go
ADJ A/C	2C	Adjacent Aircraft
VOICE	2C	Revert to Voice
TILT	INDEPENDENT	No Message for 10 Seconds, Change Antenna Position
CMD CHG	2A	Command Change Warning
ALT CHG	2A	Altitude Change Warning
MON ALT	2A	Monitor Altitude
MANUAL	INDEPENDENT	Autopilot May Not Be Used
SPD CHG	2B	Speed Change Warning
MON SPD	2B	Monitor Speed
CMD CTRL	1	Under ACL Control Command Control
CHG CHN	1	Change Channel
HDG CHN	1	Heading Change Warning
CANC RPY	INDEPENDENT	Cancel Reply
FWD VEC	1	Destroy With Arm 3
AFT VEC	1	Destroy With Arm 2
COL VEC	1	Destroy With Arm 3
NO MSG	1	No Statement
TO WAYPT	1	Vector To Waypoint
HANDOVER	1	Handover
ORBIT	1	Orbit
CHALNGE	1	Challange
ARM 1	1	Arm 1
ARM 2	1	Arm 2
ARM 3	1	Arm 3
NOT CMD	1	Not Under Control
FRE LAN	1	Free Lance
DISGAGE	1	Disengage
ABORT	1	Abort
BEAC ON	2B	Beacon On
BEAC DUB	2B	Beacon Double
DROP	INDEPENDENT	Drop
BEAC OFF	2B	Beacon Off
RET BASE	1	Return to Base

Figure 7-7

BIT TEST indicator

(U) The built-in test (BIT) indicator is located on the DDI panel. When a BIT test is performed, a normal function is displayed by black indication; a malfunction, by a black and white cloverleaf pattern. The indicator display is maintained with a magnetic latching and will show a BIT failure after power has gone off and then back on. The indicator must be manually reset by rotating the knurled ring in a clockwise direction and then in a counterclockwise direction.

Data Link Radiate Test

(U) Selecting the D/L RAD position on the pilot's MASTER TEST panel and depressing the master test switch, initiates a test sequence of the data link converter, which locks out tactical control messages and allows entry of test messages TM-10 and TM-21. A ground circuit signal to the converter routes test messages to the VDI, the DDI, the HSD, and the Mach-airspeed indicator.

VOICE SECURITY EQUIPMENT (TSEC/KY-28)

(U) The KY-28 is integrated and operates in conjunction with the UHF communications set to permit secure voice in a hostile environment. It shall be operated as directed by appropriate authority. Theory of operation and practical application are covered in the operation manual KAO-1243/TSEC.

(U) The KY-28 control panel (figure 7-8) on the NFO right side console, has the only cockpit controls for operating the UHF receiver-transmitter in cipher or plain language modes. Electrical power is provided from the dc essential bus No. 2 with circuit protection on the NFO dc essential No. 2 circuit breaker panel.

(U) The KY-28 has two basic modes of operation: plain (P) and cipher (C). The plain mode is used during normal UHF communications. The cipher mode is used when secure voice communications are desired. The KY-28 power switch must be ON to attain normal UHF communications if KY-28 is installed. The UHF set power must be ON to attain KY-28 operation. The receiving station must be properly equipped to receive transmissions which are transmitted in the cipher mode.

(U) Approximately 2-minutes are required for warmup before operation. Transmitting keys, frequencies, etc. used are the same as for the basic UHF system (ARC-51). When operating in the plain mode, the first actuation and release of a transmitting button causes a 2-second double pitched broken tone in the headset. When the tone ceases, plain transmissions may be performed. When operating in the cipher mode, actuation of a transmitting button causes a single beep tone in the headset, after approximately 1-1/2 seconds, (on every transmission); the set is then ready for transmission in the cipher mode.

(U) The operating code for the KY-28 is set in by ground personnel as directed by appropriate authority. The code cannot be changed in flight. The ZEROIZE switch is used to clear the code in the KY-28 when directed or considered appropriate by the aircrew. Care must be taken to prevent inadvertent actuation of the ZEROIZE switch while airborne; the aircrew cannot reenter the code.

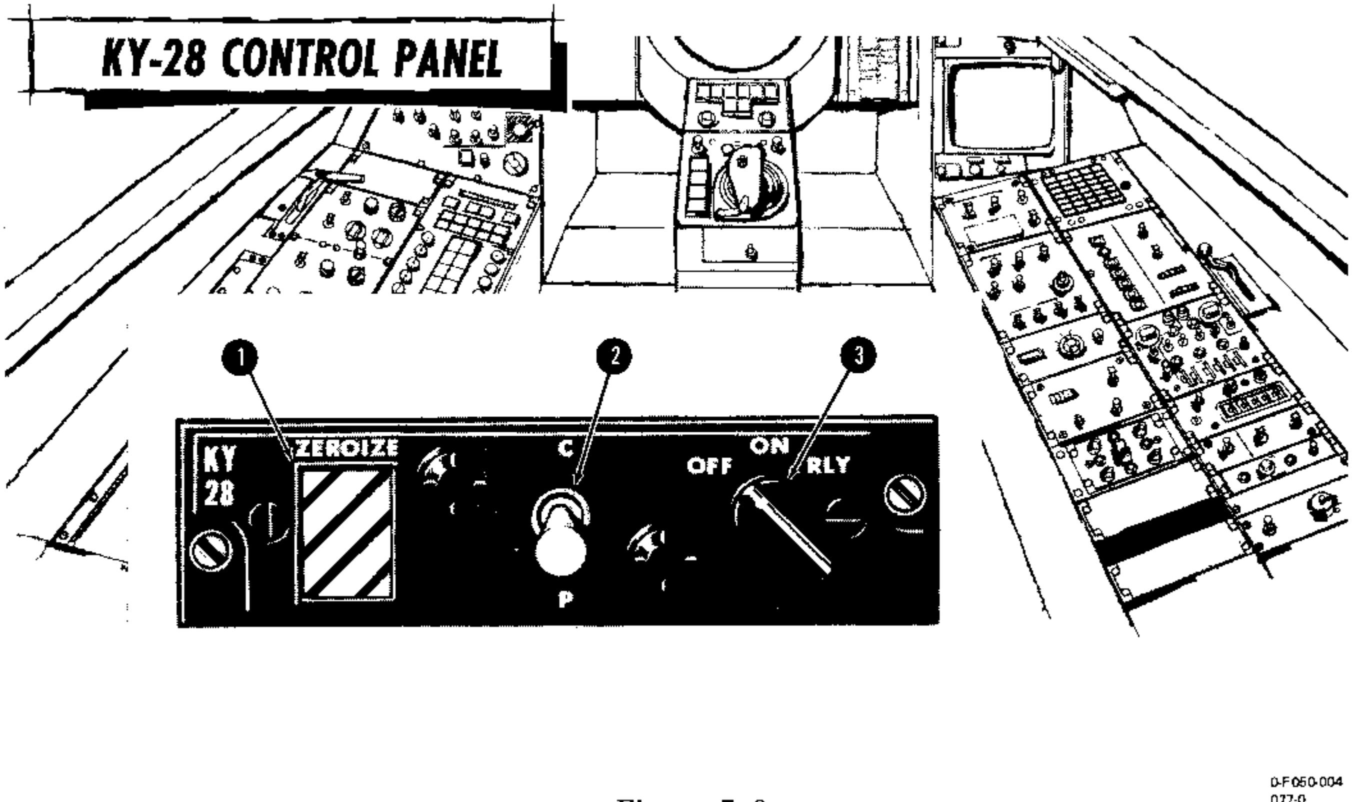


Figure 7-8

NOMENCLATURE		FUNCTION	
(1)	ZEROIZE SWITCH	ZEROIZE -	Guard lifted. The preset codes are erased and must be reset on the ground by qualified personnel before the cipher (C) mode can be used.
(2)	CIPHER-PLAIN SWITCH	C -	Used to transmit and receive secure voice communications over the UHF radio in accordance with the preset codes.
		P -	UHF radio is used as a plain language receiver/transmitter.
(3)	POWER SWITCH	OFF -	Removes power from the system and inhibits use of UHF radio.
		ON -	Applies operating power to KY-28 system.
		RLY -	Relay position, retransmits information between other facilities. (Not operational)

COMM/NAV CMD CONTROL PANEL

(U) Communication/navigation command (COMM/NAV CMD) control panels (figure 7-9) on the left side console at each cockpit station transfer TACAN, data link, and UHF control between the pilot and NFO. Three pushbuttons allow either crewman to take or give command of a particular radio. Control is indicated by a green illumination in the center of the pushbutton selected. The frequency or radio channel selected by crewmen in control of that particular radio is always available to both aircrew positions. Thus, either crewman can switch between the two radio channels by changing the command position without reselecting radio frequencies.

IN-FLIGHT VISUAL COMMUNICATIONS

(U) Communications between aircraft are visual whenever practicable, operations permitting. Flight leaders shall ensure that all pilots in the formation receive and acknowledge signals when given. The visual communications section of NWP-41 should be reviewed and practiced by all pilots and naval flight officers. Common visual signals applicable to flight operations are in figure 7-10 (sheets 1 and 2).

GROUND HANDLING SIGNALS

(U) Communications between aircraft and ground personnel are visual whenever practicable, operations permitting. The visual communications section of NWP-41 should be reviewed and practiced by all aircrew and groundcrew personnel. During night operations flashlights or wands shall be substituted for hand and finger movements.

CARRIER FLIGHT DECK PERSONNEL COLOR CODING

RED SHIRTS - Ordnance, and crash crew
YELLOW SHIRTS - Plane directors, catapult officer, and arrestment officer
BLUE SHIRTS - Plane handlers, messengers, elevator operators, tractor drivers
GREEN SHIRTS - Aircraft maintenance, catapult crew, arrestment crew
BROWN SHIRTS - Plane captains
WHITE SHIRTS - Medical, plane inspector, transfer officer, safety officer
PURPLE - Fuel handling

COMM/NAV CMD CONTROL PANEL

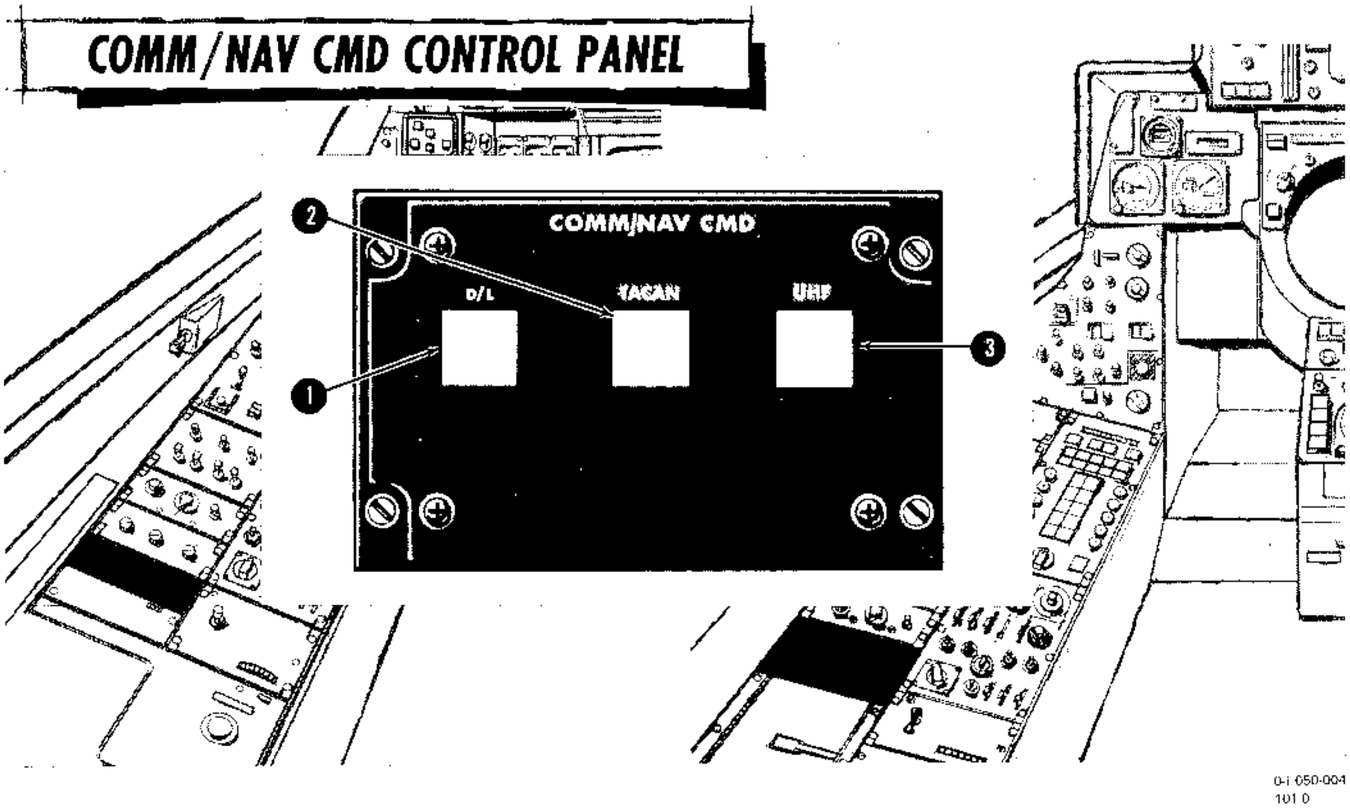


Figure 7-9

NOMENCLATURE	FUNCTION
(1) D/L BUTTON (DATA LINK)	Alternating pushbutton to give and take control of data link; green illumination of button indicates cockpit station having control.
(2) TACAN BUTTON	Alternating pushbutton to give or take control of TACAN; green illumination indicates cockpit station having control.
(3) UHF BUTTON	Alternating pushbutton to give or take control of main UHF radio; green illumination of button indicates cockpit station having control.

VISUAL COMMUNICATIONS

GENERAL CONVERSATION

MEANING	SIGNAL	RESPONSE
Affirmative (I understand). Negative (I do not know).	Thumb up or nod of head. Thumb down, or turn of head from side to side.	
Question (repeat). Used in conjunction with another signal, this gesture indicates that the signal is interrogatory.	Hand cupped behind ear as if listening.	As appropriate.
Wait.	Hand held up with palm outward.	
Ignore last signal.	Hand waved in an erasing motion in front of face, with palm turned forward.	
Numerals, as indicated.	With forearm in vertical position, employ fingers to indicate desired numerals 1 through 5. With forearm and fingers horizontal, indicate number which, added to 5, gives desired number from 6 through 9. A clenched fist indicates zero.	A nod of the head (I understand). To verify numerals, addressee repeats. If originator nods, interpretation is correct. If originator repeats numerals, addressee should continue to verify them until they are understood.

CONFIGURATION CHANGES

MEANING	SIGNAL	RESPONSE
Lower landing gear.	Rotary movement of hand in cockpit, as if cranking wheels.	Execute
Lower arresting gear hook.	Leader lowers hook.	Wingman lowers arresting gear hook. Leader indicates wingman's hook is down with thumb up signal.
Extend or retract flaps or speed brakes as appropriate.	Open and close four fingers and thumb. Execute.	

FUEL AND ARMAMENT

MEANING	SIGNAL	RESPONSE
How much fuel have you?	Raise fist with thumb extended in a drinking position.	Indicate fuel in tens of gallons or hundreds of pounds by finger numbers.
Arm or safety missiles as applicable.	Pistol cocking motion with either hand.	Execute and return signal.
1—Arm or safety tanks as applicable 2—how many tanks do I have? 3—I am unable to drop.	1—Shaking fist 2—followed by question signal 3—followed by nose-held signal	1—Execute and return signal 2— Indicate with appropriate finger numerals 3—nod head (I understand)

0-F050-004
068-1

Figure 7-10 (Sheet 1 of 2)

FORMATION		
MEANING	**SIGNAL**	**RESPONSE**
1) I have completed my takeoff check list and am, in all respects ready for takeoff. 2) I have completed my takeoff check list and am, in all respects, ready for a section takeoff 3) Takeoff path is clear, I am commencing takeoff.	1) Section takeoff leader raises arm (either) over head. 2) Wingman raises arm over head 3) Leader lowers arm.	1) Stands by for reply from wingman, holding arm over head until answered. 2) Wingman lowers arm and stands by for immediate section takeoff. 3) Executes section takeoff.
Leader shifting lead to wingman. (Day)	Leader pats self on head points to wingman.	Wingman pats head and assumes lead.
Leader shifting lead to wingman. (Night)	1) Two aircraft - lead aircraft puts external lights on bright and flash. 2) More than two aircraft - leader places flight in echelon and then use two aircraft procedures.	Wingman turns external lights to DIM and STEADY and assumes lead.
	3) With external light failure - leader shines flashlight on hardhat, then shines at Wingman.	Wingman turns external lights to DIM and STEADY and assumes lead. With external light failure — Wingman shines flashlight at leader, then on his hardhat.
Leader shifting lead to division designated by numerals.	Leader pats self on head points to wingman and holds up two or more fingers.	Wingman relays signal; division leader designated assumes lead.
Take cruising formation.	Thumb waved backward over the shoulder.	Execute.
I am leaving formation.	Any pilot blow kiss.	Nod (I understand).
Aircraft pointed out leave formation.	Leader blows kiss and points to aircraft.	Execute.
Directs plane to investigate object or vessel.	Leader beckons wing plane, then points to eye, then to vessel or object.	Wingman indicated blows kiss and executes.
Refers to landing of aircraft, generally used in conjunction with another signal: 1) I am landing; 2) Directs indicated aircraft to land.	Landing motion with open hand: 1) Followed by patting head; 2) Followed by pointing to another aircraft.	1) Execute; 2) Execute.
a) Join up or break up, as appropriate. b) On GCA/CCA final: Leader has runway/ ship in sight.	Flashing external lights.	a) Comply. b) Wingman repeats, indicating runway/ ship in sight. Ship: Leader waves-off wingman lands. Field: When runway conditions preclude a safe section landing leader will wave-off.
Leader desires wingman to cross under to other side.	Arm held vertically with clenched fist against canopy.	Wingman executes. Section lead — maintains position or allows Number 2 to slide into position as appropriate.
Leader desires section to cross to other side.	Arm held vertically with clenched fist and pumps up and down.	Number 2 wingman maintains position. Section executes.

FORMATION SIGNALS—MADE BY AIRCRAFT MANEUVER COMBAT OR FREE CRUISE		
MEANING	**SIGNAL**	**RESPONSE**
Single aircraft cross under in direction of wing dip.	Single wing dip.	Execute.
Section cross under.	Double wing dip.	Execute.
Close up.	Series of small zooms.	Execute.
Join up; join up on me.	Series of pronounced zooms.	Expedite join-up.

0-F050-004
068-2

Figure 7-10 (Sheet 2 of 2)

NAVIGATION

TACAN SYSTEM (AN/ARN-84(V))

(U) The TACAN system provides continuous indications of slant range and bearing for any selected surface station and only slant range information relative to a cooperating aircraft in an air-to-air mode. Operating range is a line-of-sight distance of approximately 300 nautical miles.

(U) The system provides 126 operating channels in each of two modes. (The current aircraft antenna complement permits the use of the NORMAL/X mode only.) Receiving frequencies for surface-to-air operation are 962 to 1024 MHz and 1151 to 1213 MHz; for air-to-air operation are from 1025 to 1150 MHz. All transmitting frequencies are within 1025 to 1150 MHz. TACAN uses two aircraft antennas, which are time-shared with the UHF radio.

(U) The system, when operating in the REC or T/R modes, is capable of receiving valid signals from a ground station simultaneously with 99 other aircraft. When in the A/A mode, the system is capable of transponding with each of five cooperating aircraft, providing slant range information to each; however, the system will interrogate and lock-on to only one of the five cooperating aircraft. The system is incapable of determining to which one of the five it is locked-on.

(U) Identical TACAN control panels (figure 7-11) are located on the pilot right side console and NFO left side console. The crewman who has navigation command control has control of the TACAN. However, each crewman may adjust the audio level of the identification signal. (In the air-to-air (A/A) mode, the cooperating aircraft must be 63 channels apart. Only range is received however, because the aircraft antenna complement is not configured to receive or transmit bearing information.)

(U) TACAN information is displayed on identical BDHI's (Bearing, Distance, Heading Indicator) on the pilot and NFO right upper instrument panels. The bearing and distance functions of the BDHI are activated when the TACAN mode select switch is at any position except OFF. In the REC and T/R modes, magnetic bearings are displayed by the No. 2 (large) needle. The needle unlocks and enters a search mode whenever bearing information is unreliable. Range information, which is received in the T/R mode or, when operating with a cooperating aircraft, in the A/A mode, is displayed in nautical miles on the direct readout counter. An OFF flag is displayed over the counter window at any time that range information is unreliable. TACAN information is also displayed in conjunction with other system modes on the pilot HSD, HUD, and VDI and the NFO ECMD.

(U) The TACAN system takes approximately 90 seconds to warm-up. Power required by the TACAN system is 115 volt ac, 28 volt dc and 26 volt ac, provided from the essential buses through the NFO circuit breaker panels. If, after the warm-up period, the range and bearing indicators continue to search with a suitable station selected, circuit breakers should be checked, command should be given to the other crewman, or another station should be tuned.

(U) The system is equipped with a memory feature so that tracking will not be interrupted by momentary disruption of received signals. A range signal that is lost after at least 10 seconds of tracking will be sustained for a memory period of 8 to 12 seconds after signal loss. A bearing signal that has been tracked for at least 15 seconds will be retained for 3 to 8 seconds after signal loss.

Built-in-Test (BIT)

(U) The TACAN system contains a built-in-test feature that provides for continuous automatic monitoring and interruptive self-test. The TACAN control unit is equipped with a momentary push button (BIT switch) for initiation of a 22 seconds interruptive self-test sequence, and two status lights labeled GO (green) and NO-GO (amber). The NO-GO light illuminates whenever any of the continuous monitor functions are NO-GO, or if the results of the self-test cycle are NO-GO. The GO light illuminates momentarily (6 to 18 seconds) only at the completion of a satisfactory interruptive self-test cycle. Light circuitry is tested at the initiation of the self-test cycle. When the BIT switch is depressed, both GO and NO-GO lights illuminate; they go out when the BIT switch is released.

SIGNAL DATA CONVERTER (CV-2837/ARN-84(V))

(U) The TACAN system signal data converter furnishes analog signals for display on the pilot's BDHI, HUD, VDI, and HSD and the NFO's BDHI and ECMD. In addition, digital range and bearing signals are provided for use by the AWG-9 computer. The converter also interfaces with the ASW-27B data link system, providing that system with range and bearing in analog form, station number in binary form, and discretes.

(U) To display TACAN information on the HSD, the pilot must select the NAV mode on the his display control panel. Selecting either DEST, VEC, or MAN steering command options will display TACAN relative bearing only. When the TACAN steering command button is selected, all TACAN information displayed the BDHI's will be repeated on the pilots HSD and the NFO's ECMD. To display TACAN information the NFO must select the NAV mode on his ECM display control panel. If range information is not available, range numbers will not appear; if bearing information is not available, the TACAN bearing pointer and the deviation bar will not appear.

(U) When the pilot selects the TACAN steering command option, and has selected either T.O., CRUISE or LDG mode, the VDI and the HUD will display a precision course vector symbol. On the HUD, a solid bar indicates "to" the TACAN course and a dashed line indicates "from" the TACAN course. On the UDI a dark solid bar indicates "from" and a bright solid bar indicates "to".

BEARING DISTANCE AND HEADING INDICATOR (BDHI)

(U) A bearing-distance and heading indicator (BDHI) is located on the right side of the pilot and NFO instrument panels (figure 7-11). It is a remote type heading indicator, which displays aircraft magnetic heading with navigational bearing data and range information. The compass card receives heading reference from the attitude heading reference set (AHRS). Controls on the compass panel permit the BDHI compass card to operate in a slaved or nonslaved (FREE DG) compass mode. A fixed index marker at the 12-o'clock position indicates magnetic heading.

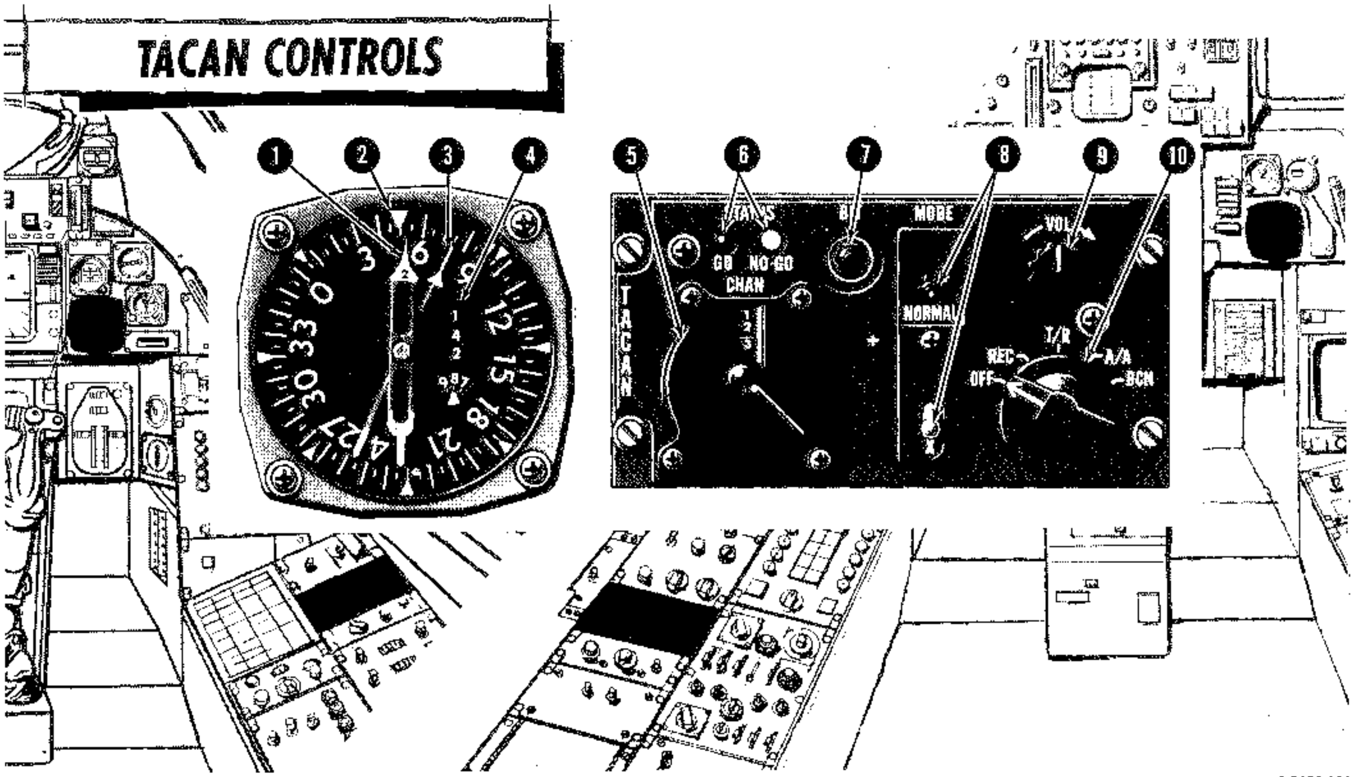


Figure 7-11

NOMENCLATURE		FUNCTION
1	NO. 2 BEARING Pointer	Indicates magnetic course to selected TACAN station.
2	COMPASS ROSE	Top of instrument indicates present magnetic heading.
3	NO. 1 BEARING pointer	Indicates bearing to selected UHF/ADF station.
4	DISTANCE INDICATOR	Indicates slant range (nautical miles) to selected TACAN station, or co-operating aircraft.
5	TACAN CHANNEL SELECTION	Outer control selects first two numbers and inner control selects last digit of desired TACAN channel (126 channels available).

NOMENCLATURE		FUNCTION
6 STATUS lights		Indicate GO/NO-GO status of TACAN system.
7 BIT Switch		Initiates interruptive self-test portion of Built-in-Test feature
8 MODE Selector	X -	Normal TACAN functions
	Y -	Inverse TACAN functions
9 VOLUME CONTROL		Regulates audio level of station identification signal through ICS.
10 FUNCTION SELECT SWITCH	OFF -	Set power off.
	REC -	Provides only bearing information from selected surface station (no range information).
	T/R -	Permits reception of bearing and slant range from selected surface station.
	A/A -	Permits cooperating aircraft to determine only distance from each other, if such aircraft have selected TACAN frequencies 63 channels apart (no bearing information on ID).
	BCN -	Provides bearing and slant range from the aircraft. (Presently not operational.)

(U) Two servo-driven bearing needles indicate magnetic bearings to selected UHF (ADF) and TACAN stations. The No. 1 (single bar) needle receives signals from the UHF (ADF) system; the No. 2 (double bar) needle receives signals from the TACAN coupler. If the compass card is mis-aligned, or a malfunction exists in the compass system, the No. 1 needle will continue to point toward the signal source; however, the bearing to the station is displayed on the indicator as a relative bearing, the top of the indicator bezel being 000°. Under the same circumstances, the No. 2 needle will continue to indicate magnetic bearings to the selected station, or revert to the search mode.

ATTITUDE AND HEADING REFERENCE SET (AHRS)

(U) The attitude heading reference set (AHRS) provides magnetic heading for the BDHI; backup pitch and roll information to the ADI; and auxiliary pitch, roll, and magnetic heading to the AWCS computer in the absence of attitude and direction information from the inertial navigation system. Basic components of the AHRS include a two-gyro (vertical and directional) three-gimbal platform; a remote compass transmitter (flux valve); a compass control panel, and an advisory caution light. The COMP control panel (figure 7-12) has the operating controls for the attitude heading reference set.

(U) The platform is unlimited in roll, but limited to ± 82 degrees in pitch. Changes in aircraft attitude with respect to the vertical reference are detected by the platform and transmitted to the ADI if the inertial measuring unit fails. The directional gyro and flux valve operate together to provide heading signals to the BDHI and the AWCS.

(U) As a compass set, the AHRS operates in three modes; the directional gyro (DG) mode provides a free-gyro heading reference with earth rate correction; the SLAVED mode provides a gyro-stabilized magnetic heading; and the compass (COMP) mode provides an emergency magnetic heading from the compass transmitter. When starting a turn in the COMP mode, the compass tends to move opposite to the turn, lag, catch up with the aircraft heading, then overshoot the rollout (as with the standby or wet compass). The random drift (precision rate) of the gyro in the DG mode does not exceed ± 1.5 degrees per hour. This mode may be used at all latitudes, but is more useful when operating in polar regions or when the magnetic field is weak or distorted. The AHRS operates on 115 volt ac power from the essential ac No. 2 bus and 28 volts dc power from the essential dc No. 2 bus, through the NFO circuit breaker panel. When the COMP mode is selected the AFCS is automatically disengaged to prevent erratic steering commands. The COMP mode does not provide a sufficiently stable heading signal for AFCS operation.

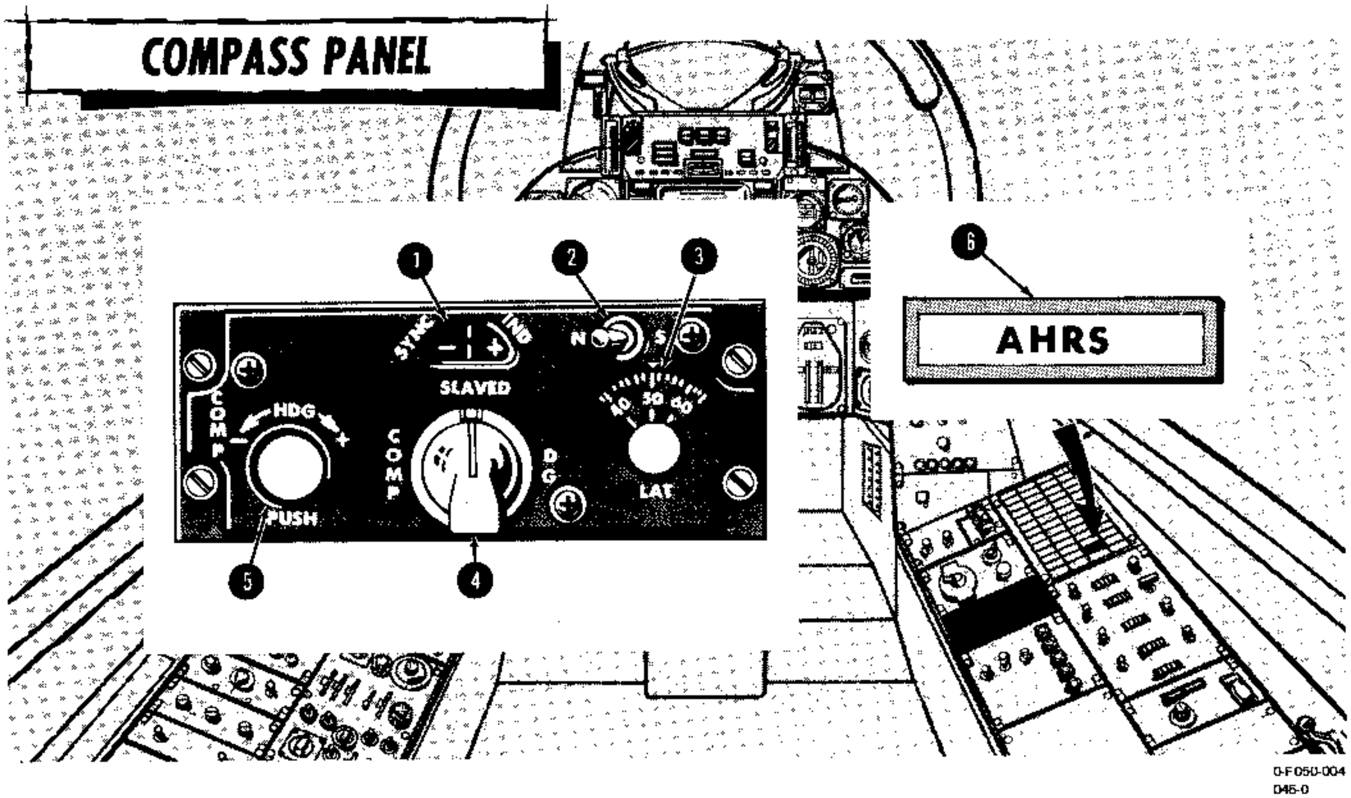


Figure 7-12

NOMENCLATURE	FUNCTION
(1) SYNCHRONIZATION INDICATOR	Indicates synchronization between AHRS gyro and remote compass transmitter. Indicator is deactivated in DG mode.
(2) HEMISPHERE SELECT Switch	In DG mode, hemisphere (N or S) in which aircraft is operating must be selected to provide proper earth rate correction.
(3) LATITUDE CORRECTION Control	Used for DG mode; latitude from 0 to 90 degrees must be selected to allow for earth rate corrections.

NOMENCLATURE	FUNCTION
(4) MODE SELECTOR	COMP - Compass heading is obtained directly from remote compass transmitter without stabilization by directional gyro. This position is used if gyro malfunctions. SLAVED - Gyro-stabilized magnetic heading is from remote compass transmitter. Normal position. DG - Directional gyro mode uses arbitrary gyro heading. Remote compass transmitter is not used and system gyro operates as true gyro.
(5) HEADING SET Button	In SLAVED mode, switch synchronizes directional gyro with remote compass transmitter and sets desired heading on the BDHI. Button must be held depressed until synchronization indicator is centered. Switch has no function in COMP mode.
(6) AHRS Advisory Light	Green light goes on when attitude or heading information from AHRS is unreliable.

INSTRUMENT LANDING SYSTEM (ILS)

(U) The aircraft instrument landing system (ILS) uses the AN/ARA-63 receiver decoder with the carrier-based AN/SPN-41 transmitters. This system is used for manual instrument landing approaches or as an independent monitor during final approach with the all-weather carrier landing system (AWCLS). The system operates in the Ku-band, between 15.4 and 15.7 GHz, on any of 20 channels selected by the pilot. Channel selection is controlled from the ARA-63 CONTROL panel (figure 7-13) on the pilot's right side outboard console.

(U) The aircraft system receives and decodes glideslope azimuth and elevation signals, which are converted into command fly-to indications in the CICU and displayed on the VDI and the HUD in the landing mode. The transmitted azimuth signal produces a 2 degree beam, which is scanned ± 20 degrees from the deck centerline. The transmitted elevation signal produces a 1.3 degree beam with a scan pattern from 0 to 10 degrees looking up. A proportional azimuth angle for steering is ± 6 degrees, right or left of centerline; proportional elevation angle for steering is ± 1.4 degrees from a 3 degree glideslope (above and below). Operating range is approximately 20 nautical miles, 4 miles high and 16 miles wide.

(U) During all-weather landing, the pilot normally displays ILS information on the HUD and select automatic carrier landing (ACL) symbology on the VDI. If the ILS or ACL landing submodes selected on the pilot's display control panel become invalid, the VDIG will revert to a basic landing mode. The basic landing mode does not display the vertical and lateral cross-bars for glideslope error and does not provide a waveoff signal.

AN/ARA-63 DECODER PANEL

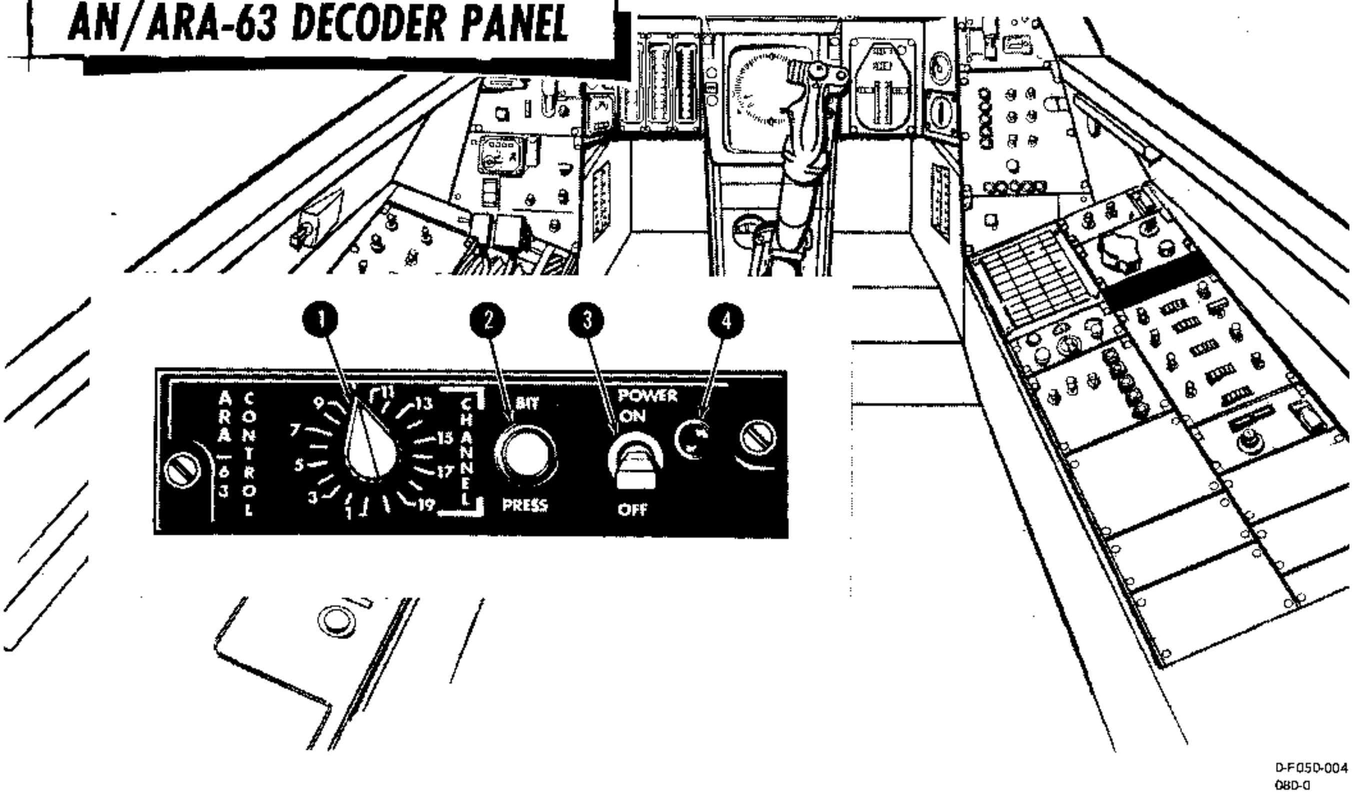


Figure 7-13

NOMENCLATURE		FUNCTION
(1) CHANNEL SELECTOR		Twenty possible channel selections by rotation of selector knob.
(2) BIT PRESS-TO-TEST BUTTON		Depressing button activates BIT test circuitry. Landing symbols available on VDIG.
(3) POWER SWITCH	ON -	Activates receiver decoder for all-weather carrier landing.
	OFF -	Turns system off.
(4) INDICATOR LIGHT		Illuminates when ARA-63 is on and receiving.

RADAR BEACON (AN/APN 154)

(U) The radar beacon enhances aircraft tracking by ship and/or ground-based X-band radars. Pulsed X-band signals, transmitted by the surface radar station are received by the beacon and decoded. If received pulses match the mode selected by the NFO, the beacon responds with a return pulse to the radar site. The reply signal, considerably stronger than a normal radar echo, enhances the radar acquisition and tracking capability of the surface station. Vectors to the aircraft are transmitted by normal UHF voice. Mode of operation is as briefed, or by radio. The beacon responds to 8500 to 9600 MHz; It uses a monopole antenna located on the IR fairing. Future systems will use an ACL augmentor (Automatic Carrier Landing); an additional position (ACL) will be on the control panel.

(U) The beacon augmentor eliminates radar scintillation by providing a large source of reply energy from one point on the aircraft. The beacon augmentor, receives interrogations from the SPN-42 carrier based radar in the Ka band, processes them, and retransmits modulated X-band pulses to the SPN-42, which has an X-band receiving system mounted contiguous to the basic Ka-band radar transmitting antenna. The unique feature of the crossband transponder is that it uses the APN-154 beacon as its X-band transmitter. This is accomplished by coupling the output of the augmentor to the APN-154 and triggering its modulator and transmitter. During the landing phase, it is necessary to manually place the APN-154 into its ACL position. In this mode the APN-154 receiver is disabled to ensure that extraneous X-band signals in the area will not trigger the APN-154 transmitter during landing.

(U) The radar beacon control panel (figure 7-14) on the NFO right side console. Code selection coordination and voice vectoring information must be accomplished by UHF voice radio. The POWER or STBY position can be used for radar beacon warm-up. However, to preclude response to a premature or unintentional interrogation, the STBY position should be used.

(U) The BIT indicator on the control panel will light whenever the system is operating normally and is receiving a Ka band signal from the AN/SPN-42.

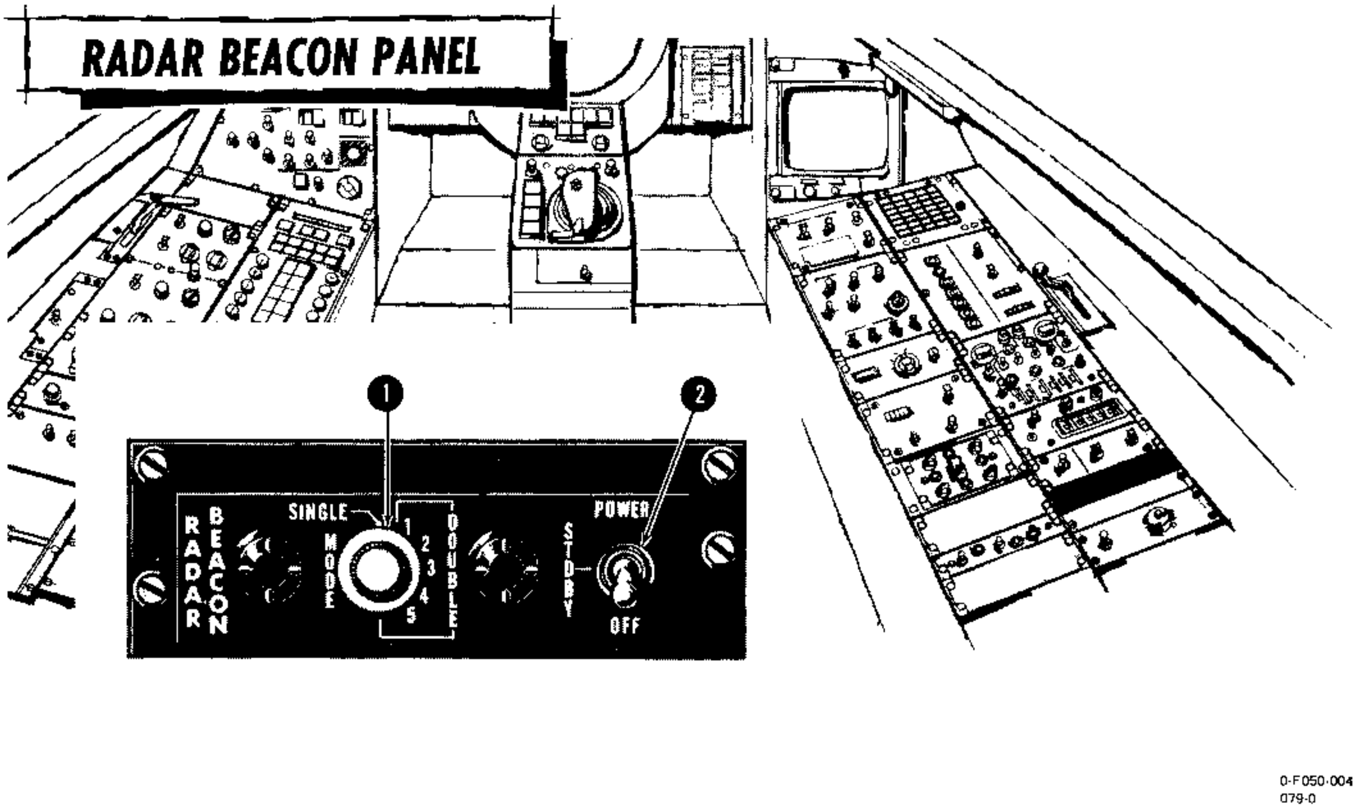


Figure 7-14

NOMENCLATURE		FUNCTION	
(1)	RADAR BEACON MODE Switch	SINGLE -	Limits beacon response to single pulse of any code group received.
		DOUBLE -	Beacon response to one of five double pulse interrogations. Codes 1 thru 5 correspond to switch position 1 thru 5.
		ACL -	Position to be added in the future which will be used for automatic carrier landings.
(2)	RADAR BEACON POWER Switch	POWER -	Places beacon in full operation for interrogations.
		STBY -	Places beacon in warmup and will not reply to interrogations. Allow 5 minutes for optimum performance.
		OFF -	Turns off power to beacon.

RADAR ALTIMETER SYSTEM (AN/APN-194)

(U) The radar altimeter is a low-altitude, pulsed, range-tracking radar that measures the surface or terrain clearance below the aircraft. Altitude information is developed by radiating a short-duration RF pulse from the transmit antenna to the earth's surface and measuring elapsed time until RF energy returns through the receiver antenna and is detected in the receiver. The altitude information is continuously presented to the pilot, in feet of altitude, on an indicator dial.

(U) The radar altimeter has two modes of operation. In the search mode, the system successively examines increments of range until the complete altitude range is searched for a return signal. When a return signal is detected the system switches to the track mode and tracks the return signal, to provide continuous altitude information.

(U) When the radar altimeter drops out of the track mode, an OFF flag appears and the pointer is hidden behind a mask. The altimeter remains inoperative until a return signal is received, at which time the altimeter will again indicate actual altitude above terrain.

(U) Reliable system operation in the altitude range of 0 to 5,000 feet permits close altitude control at minimum altitudes. The system is inoperative in banks of more than 40 degrees, in climbs or dives of more than 40 degrees and when the reflected signal is otherwise too weak.

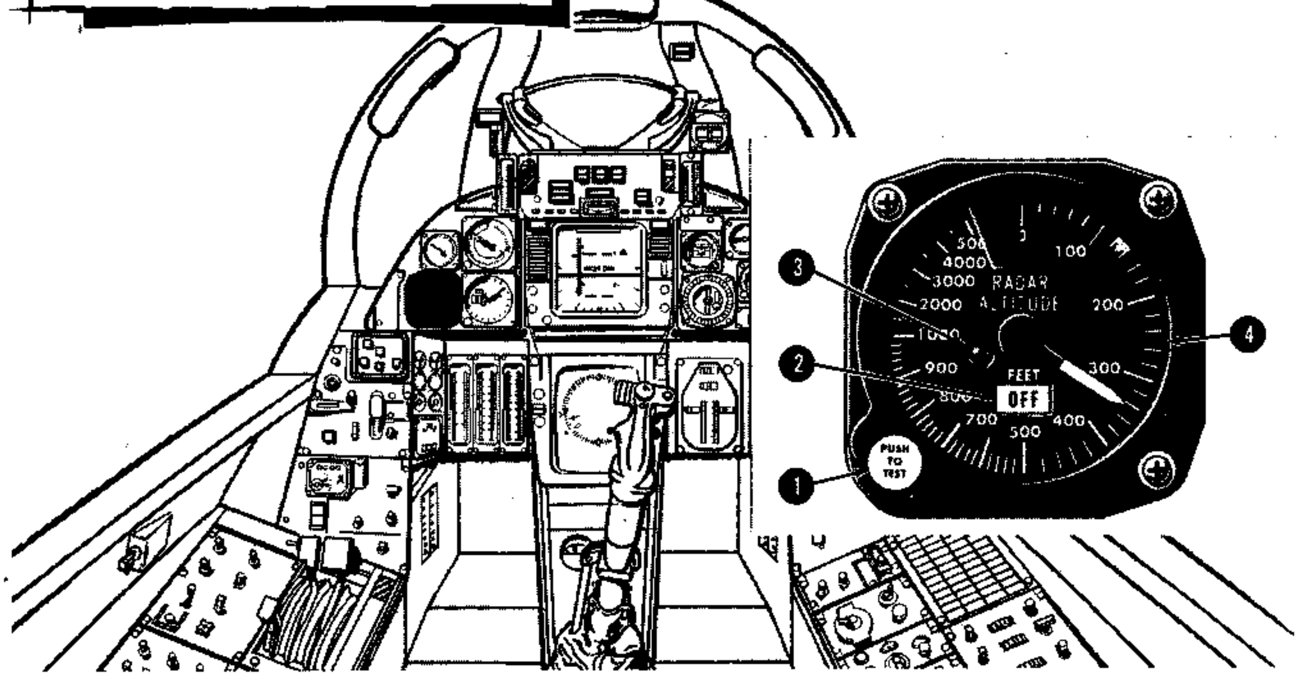
(U) The system includes a height indicator, a test light on the indicator, a radar receiver-transmitter under the forward cockpit, and two antennas (transmit and receive) one on each side of the IR faring, in the aircraft skin. During descent, the test light illuminates momentarily as the aircraft passes through the altitude set or the limit index.

Radar Altimeter Height Indicator

(U) The radar altimeter height indicator (figure 7-15) on the pilot's instrument panel has the only controls for the system. The indicator displays radar altitude above the earth's surface on a single-turn dial that is calibrated from 0 to 5,000 feet in decreasing increment to provide greater definition at lower altitudes. The control knob located in the lower left corner of the indicator, is a combination power switch, self-test switch, and positioning control for the low altitude limit index (limit bug).

(U) Depressing and holding the control knob energizes the self-test circuitry; the test light should go on and the indicator should read 100 ± 10 feet. If the indicator reads correctly, but the test light does not go on, the receiver-transmitter is faulty. If the indicator reads incorrectly, but the light goes on, the indicator is faulty. Normal operation is resumed by releasing the control knob.

RADAR ALTIMETER



0-F050-004
078-0

Figure 7-15

NOMENCLATURE		FUNCTION
1	RADAR ALTIMETER CONTROL SWITCH	Initial clockwise rotation turns set on and continued rotation sets altitude limit index marker. Pushed in tests system with weight off gear. Counterclockwise rotation decreases altitude limit setting, and fully counterclockwise turns set off.
2	OFF FLAG	OFF appears if altimeter is turned off; power is lost, or radar signal unreliable.
3	TEST LIGHT	Momentarily illuminates as a warning as aircraft passes altitude set by index bug. When control knob is depressed and held light should come on and pointer read 100 ± 10 feet.
4	LIMIT INDEX BUG	Can be preset to low altitude limit desired by turning control switch.
5	ALTITUDE LIMIT INDEX REFERENCE	Indicates selected altitude reference set by control switch.

IDENTIFICATION

IDENTIFICATION TRANSPONDER (IFF/SIF), AN/APX 72

(U) The IFF/SIF identification system provides automatic identification of the aircraft in which it is installed, when properly challenged by surface or airborne radar sets. The system operates in the UHF band; it transmits on 1090 MHz and receives on 1030 MHz. Operating range is line of sight and the system is capable of operating to an altitude of 70,000 feet. The system also provides momentary identification of position upon request, and transmits a specially coded response to indicate an emergency.

(U) The system receives coded interrogation signals and processes them through a decoder which checks for a valid code and proper mode. If valid, the decoded signal is transferred to an encoder which prepare coded reply signals for transmission to the interrogating source. The proper reply identifies the target as friendly, and with prior coordination, SIF identifies its aircraft. The system uses two omnidirectional antennas, one located on top of the fuselage and the other on the right ventral fin. Antenna selection is controlled by the NFO, using the IFF ANT switch. LWR position, selects lower antenna and AUTO selects the antenna with strongest signal. An emergency IFF switch automatically activate modes 1, 2 and 3/A when the pilot's seat is ejected, if the transponder is in ON or STBY.

(U) Five independent coding modes are available. The first three modes may be used independently or in combination. Mode 1 provides 32 possible code conbinations, any one of which may be selected in flight. Mode 2 provides 4,096 possible code combinations, but only one is available because the selection dials for the mode are on the receiver-transmitter and not accessible in flight. Mode 2 code combinations must be preset before takeoff. Mode 3/A provides 4,096 additional codes, any one of which may be selected in flight. Mode C furnishes the pressure altitude of the aircraft from the ADC when properly interrogated. Mode 4, which is interfaced with the KIT-1A computer, can be selected to reply to any one of the many classified operational codes for security identification. The Ident-Out-MIC switch effects modes 1, 2 and 3A. It permits the controller to identify the aircraft position on demand. The Ident reply will continue for 15 to 30 seconds after release of the spring loaded switch. The MIC position transmits Ident replies when the radio keying switch is closed.

(U) The radar identification control panel placarded IFF is on the NFO right side console. The control panel (figure 7-16) has the system master switches, code selector switches, and an identification (IDENT) switch. The antenna control panel (figure 7-16) is also located on the NFO right side console.

(U) Power requirements include 115 volts from the essential No. 1 bus, 28 volts from the essential ac No. 2 bus, and 28 volts from the left main bus for test only. Power output of the set is approximately 500 watts.

Operation

(U) To place the IFF in operation, proceed as follows:

1. Master switch - STBY for 1 minute under standard temperature conditions (5 minutes under extreme ranges of operating temperatures), then to NORM.
2. Code selectors - POSITION DETERMINED BY MISSION
3. Mode switches - POSITION DETERMINED BY MISSION
4. Ident - Out - MIC switch - OUT

Mode 4 Controls/Reply Light

(U) The mode 4 controls and indicator light are grouped on the left side of the control panel. The MASTER rotary switch controls transponder operation in Mode 4 as well as in the other modes of operation. Mode 4 will operate normally, when selected, in either the NORM or EMER positions, and at reduced receiver sensitivity in LOW position. The Mode 4 function will be inoperative in either STBY or OFF positions.

(U) With the transponder functioning, Mode 4 operation is selected by placing the Mode 4 ON-OUT toggle switch to ON. Placing the switch to OUT disables Mode 4 operation.

(U) The Mode 4 CODE control selects either of the two (A or B) Mode 4 codes. It has two additional positions, HOLD and ZERO. The switch is spring-loaded to return from HOLD to the A position. At a designated daily time and/or prior to the day's mission, maintenance or operations personnel mechanically set in the Mode 4 code for the present code period in position A, and the code for the succeeding code period in position B, with a single insertion of the KIK-18/TSEC Code Changer Key. Both code settings will be zeroized when power is turned off after the aircraft has landed. The code settings can be retained by activating the HOLD function. This will normally be done after the aircraft has landed, (landing gear must be down and locked) and before power is removed from the transponder. Place the Mode 4 CODE control to HOLD, and release. Allow transponder power to remain on for at least 15 seconds, and then turn off. The code setting is now mechanically latched and will be retained when aircraft power is turned off.

NOTE

> If power is removed from the transponder less than 15 seconds after selecting HOLD, either by turning the transponder off or by turning off aircraft electrical power, the code setting may not be mechanically latched and will zeroize.

(U) With aircraft power on and the MASTER rotary switch in any position except OFF, both code settings can be zeroized at any time by placing the CODE switch to the ZERO position. Both code settings will also be zeroized, if the HOLD function has not been properly actuated before the MASTER switch is turned to OFF. (Inadvertent selection of OFF is prevented by switch design which requires that the rotary knob be pulled out before it can be turned to OFF.) When the CODE switch is placed in the A position, the aircraft transponder will respond to Mode 4 interrogations from an interrogator using the same code setting as that set into the aircraft's Code A position. In the B position, interrogations from an interrogator using the same code setting as that set into the aircraft's code B position will be answered. The changeover time from Code A to Code B use is operationally directed.

(U) The Mode 4 AUDIO-LIGHT switch selects the aircraft indication for Mode 4 replies. In the LIGHT position, the Mode 4 REPLY light (green) will illuminate when Mode 4 replies are transmitted. In the AUDIO Position, an audio signal in the pilot's headset indicates Mode 4 interrogations are being received and illuminations of the Mode 4 REPLY light indicate when replies are transmitted. Mode 4 audio volume can be adjusted by the appropriate aircraft intercom audio volume control. In the OUT position, both light and audio indications are inoperative.

(U) An IFF caution light, located on the caution light panel near the pilot, illuminates whenever the IFF caution light circuitry detects an inoperative Mode 4 capability, provided that: the KIT-1A/TSEC computer is installed, aircraft power is on, and the IFF MASTER switch in <u>not</u> OFF. Specific discrepancies monitored by the IFF caution light circuitry are: (1) Mode 4 codes zeroized, (2) Transponder failure to reply to proper interrogation, <u>or</u> (3) Automatic self-test function of the computer reveals a faulty computer. Illumination of the caution light indicates the equipment will not respond to Mode 4 interrogations, and that operation in a known Mode 4 interrogating environment should be avoided or, if already in one, take appropriate corrective or emergency action. To attempt correction, place the IFF MASTER control switch to NORM (if in STBY or LOW), check to see that the Mode 4 ON-OUT toggle switch is ON, and that the proper A or B code has been selected for the current code time period. If the caution light remains illuminated employ the applicable flight procedures which are operationally directed for an inoperative Mode 4 condition.

NOTE

Since power for the caution light is routed through the KIT-1A/TSEC Transponder Computer, this unit must be physically installed in the aircraft to render the caution light operative.

IFF CONTROL PANELS

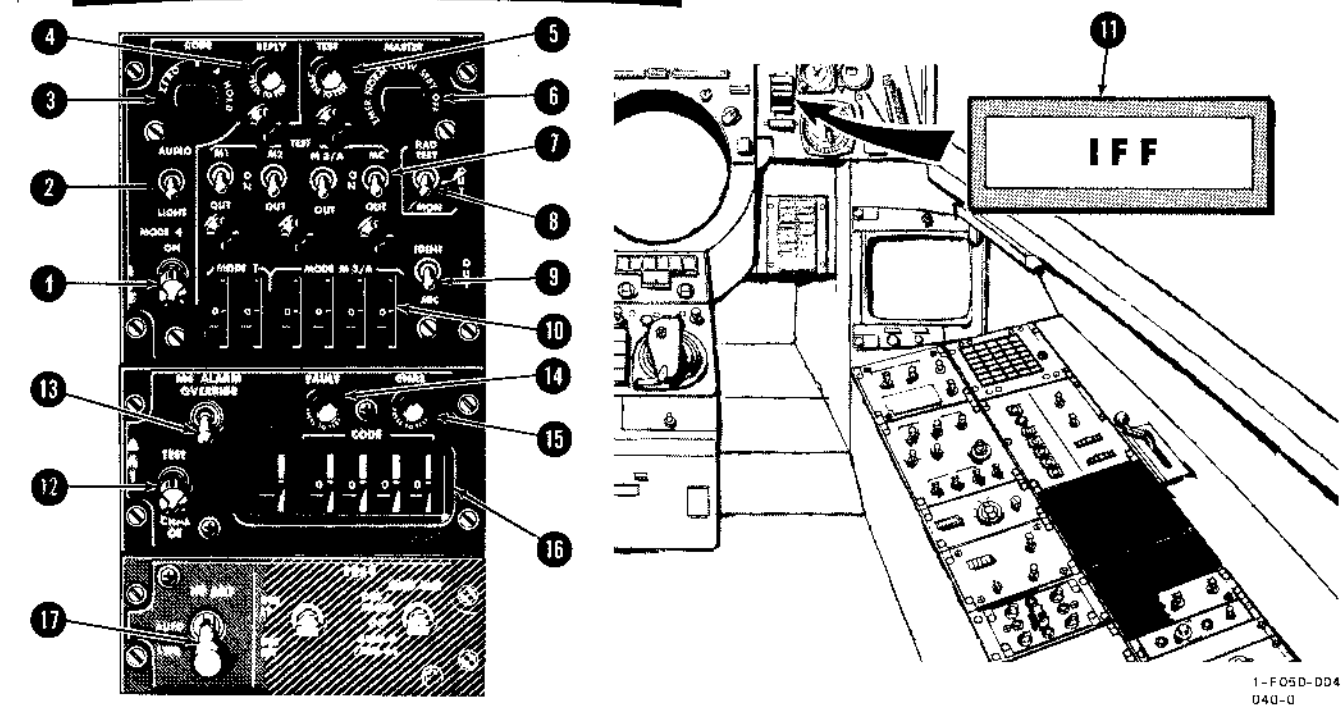


Figure 7-16

NOMENCLATURE		FUNCTION	
(1)	MODE 4 Switch	ON -	Enables mode 4.
		OUT -	Disables mode 4.
(2)	MODE 4 AUDIO/OUT Light Switch	AUDIO -	Selects aural monitoring when mode 4 is being interrogated and REPLY light monitoring of MODE 4 replies.
		OUT -	Disables light and aural monitoring of mode 4.
		LIGHT -	Selects mode 4 reply light montoring.
(3)	MODE 4 CODE Switch	ZERO -	Erases code 4 from KIR-1A and KIT-1A computers.
		B -	Selects KIT-1A computer B code.
		A -	Selects KIT-1A computer A code.
		HOLD -	Retains code in KIR-1A and KIT-1A computers when landing gear is down.

NOMENCLATURE		FUNCTION	
4	MODE 4 REPLY Light		Goes on to indicate mode 4 reply to interrogation.
5	TEST Light		Illuminates when respective test switch is actuated; indicates proper operation of modes 1, 2, 3/A, and C.
6	MASTER Switch	OFF -	Deenergizes set.
		STBY -	Energizes receiver-transmitter for immediate operation upon switching to ON operating position.
		LOW -	Provides reduced receiver sensitivity for response only to strong nearby interrogations.
		NORM -	Allows full receiver sensitivity for response maximum range interrogations.
		EMER -	Provides full receiver sensitivity and generates emergency replies to mode 1, 2, and 3/A and a normal reply to mode C, when interrogated, whether mode switches are on or off.
7	MODE Switches	TEST -	TEST light illuminates if system is functioning properly.
		ON -	Permits selection of interrogating modes to which the transponder will respond.
		OUT -	Deenergized position.
8	RAD TEST/OUT/MON Switch	RAD TEST-	Not operational. (For ground testing only)
		OUT -	Deenergized position.
		MON -	Permits transponder tester to external interrogator. Test light comes on.
9	IDENT-OUT-MIC Switch	IDENT -	Momentary position provides IDENT reply for 10 to 30 seconds after releasing switch; replies to interrogations in modes 1, 2, 3/A.
		OUT -	Deenergizes circuit.
		MIC -	Transfers IDENT reply activation switch from Ident to radio microphone switch.

NOMENCLATURE		FUNCTION
(10) CODE Selectors (MODE 1 and 3/A)		CODE SELECTORS are rotatable drums with imprinted numbers that appear in code selector windows, permitting selection of codes for mode 1 and 3/A.
(11) IFF CAUTION LIGHT		Indicates mode 4 interrogation was received, but system has not generated reply.
(12) TEST/CHAL CC SWITCH		Momentary two position center-return switch.
	TEST -	Onboard transponder is triggered by onboard interrogator. Both sets must have same code settings. IFF reply is displayed on DDD at approximately 4 miles range.
	CHAL CC -	an SIF interrogation cycle starts the 5 to 10 second challenge period. Only correct code replies are displayed.
(13) M4 ALARM OVERRIDE SWITCH		Disables the mode 4 tone alarm to the NFO's ICS.
(14) FAULT LIGHT		Indicates a malfunction of APX-76 receiver-transmitter, caused by receiver, video, or transmitter signals.
(15) CHAL LIGHT		Remains illuminated for the duration of a challenge period indicating correct operation.
(16) CODE Selectors		First thumbwheel selects mode, 1, 2, 3A, 4A, or 4B. Last four thumbwheel rotatable drums with imprinted numbers appearing in code selector windows, permitting selection of desired interrogation code.

NOMENCLATURE	FUNCTION	
17 IFF ANTENNA SWITCH	AUTO -	Antenna lobing switch cycles receiver-transmitter between upper and lower antenna pattern coverage.
	LWR -	Only lower antenna pattern below aircraft is used.

IDENTIFICATION INTERROGATOR (AIR-TO-AIR IFF), AN/APX-76 (V)

The air-to-air interrogator (AAI) provides radar identification of friendly and hostile airborne targets. Decoded IFF video signals are displayed on the detail data display (DDD) when an IFF display is selected or the CHAL CC switch is activated. The interrogator operates in conjunction with the AWG-9 fire control radar. A KIR-1A computer generates mode 4 interrogations and interpolates mode 4 replies. Figure 7-17 shows mode 4 caution/reply light logic. Correct code challenge enables interrogations for which IFF displays are generated only if the received SIF reply code is identical to the code setting on the interrogator control panel. The interrogator includes a self-checking monitor and a fault isolation circuit. The transmitter operates at a fixed frequency of 1030 MHz and the receiver, at a fixed frequency of 1090 MHz.

(U) When the AWCS is operating in any tracking mode, the NFO can challange any detected targets. Using the expanded display format on the DDD, the IFF signal can be used to classify tracked targets as friendly, hostile, or unknown. This information may be entered into the AWG-9 computer for attack computation and data link transmission. When operating the AWCS in a pulse doppler mode, the AWG-9 computer initiates IFF challenges for normal IFF display.

MODE 4 CAUTION/REPLY LIGHT LOGIC

Transponder (APX-72)	Interrogator (APX-76)	Caution	Reply (APX-72)
4 OUT (A) STBY	A	ON	OFF
4 ON (A) STBY	A	ON	OFF
4 ON (A) NORM	A	OFF	ON
4 ON (A) NORM	B	ON	OFF
4 OUT (A) NORM	A	ON	OFF
4 ON (B) NORM	A	ON	OFF
4 ON (B) NORM	B	OFF	ON
4 ON (B) STBY	B	ON	OFF
4 OUT (B) STBY	B	ON	OFF
4 ON (A) NORM RAD TEST	VER BIT 1 (A)	OFF	ON
4 ON (A) NORM	VER BIT 1 (A)	ON	OFF
4 ON (A) STBY	VER BIT 1 (A)	ON	OFF
KIT ZERO	A OR B	ON	OFF

Figure 7-17

section XI
PERFORMANCE DATA

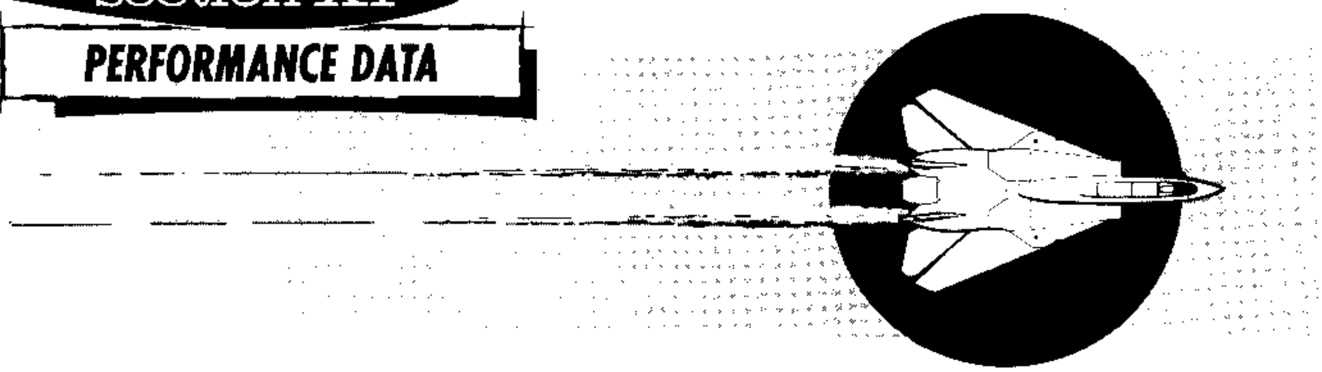


TABLE OF CONTENTS

*These charts to be supplied.

TABLE OF CONTENTS (Cont)

part 1 Standard Data

INTRODUCTION

(U) This section predicts the performance capabilities of the aircraft. It is divided into 10 parts to present performance data in proper sequence for preflight planning. The scope of this section has been limited for this Preliminary F-14A Flight Manual; however, a continuing effort will be made to update and expand the performance data as additional information is derived during the flight test program. All data is applicable to F-14A aircraft equipped with TF 30-P-412 engines. Data on each chart is based on the contractor's estimate, ICAO standard day conditions with provisions to correct for non-standard temperatures, and JP-5 fuel with a density of 6.8 pounds per gallon.

(U) The aerodynamic and non-afterburner engine source data used in preparing these charts was extracted from the F-14A Basic Aerodynamic Data Report, A51-335-R-69-1, dated August 1969, revised February 1971. Data to be presented at a later date which requires afterburner engine performance will be based on the TF-30 "Real" engine.

GLOSSARY OF TERMS

(U) DENSITY ALTITUDE - Pressure altitude corrected for temperature. When conditions are standard, pressure altitude and density altitude are the same. Consequently, if the temperature is above standard, the density altitude will be higher than the pressure altitude. If the temperature is below standard, the density altitude will be lower than the pressure altitude.

(U) PRESSURE ALTITUDE - The height or vertical distance from the standard datum plane. This is a theoretical plane where air pressure (corrected to 15°C) is equal to 29.92 inches mercury (Hg). The indicated pressure altitude may not be the actual height above sea level due to variations in temperature, lapse rate, atmospheric pressure, and errors on the sensed pressure.

(U) INDICATED AIRSPEED - Indicated airspeed (IAS) is equal to the pitot static airspeed indicator reading, as installed in the aircraft, without correction for airspeed indicator system errors.

(U) CALIBRATED AIRSPEED - Calibrated airspeed (CAS) is equal to the airspeed indicator reading corrected for position error. Position error is primarily confined to the static source in that the actual static pressure sensed at the static port is different from the free airstream static pressure.

(U) EQUIVALENT AIRSPEED - Equivalent airspeed (EAS) is equal to the calibrated airspeed corrected for adiabatic compressible flow for the particular altitude. EAS is equal to CAS at sea level in standard air.

(U) TRUE AIRSPEED - True airspeed (TAS) is the aircraft speed over the ground in no wind conditions. True airspeed is EAS corrected for density altitude.

(U) REFUSAL SPEED - Maximum speed to which the aircraft can accelerate and then stop in the remaining available runway length.

(U) REFUSAL DISTANCE - The distance required to accelerate to the refusal speed under normal conditions.

(U) CRITICAL ENGINE FAILURE SPEED - The speed (V_1) at which an engine failure permits acceleration to takeoff in the same distance that the aircraft may be decelerated to a stop.

(U) CRITICAL FIELD LENGTH - The total length of runway required to accelerate on all engines to critical engine failure speed, experience an engine failure, and then continue the takeoff or stop.

(U) TAKEOFF SPEED - Speed at which the main gear lifts off the ground.

(U) TAKEOFF DISTANCE - Distance from start of takeoff to clear 50 feet with both engines operating at military power.

(U) ROTATION SPEED - Nose wheel liftoff speed (V_R).

(U) OPTIMUM CRUISE ALTITUDE - The altitude/Mach number combination which yields maximum cruise capability, i.e., nautical miles per pound of fuel.

(U) MAXIMUM ENDURANCE ALTITUDE - The altitude which yields maximum time/minimum fuel flow.

(U) CRUISE CEILING - The altitude where the rate of climb is 300 feet per minute at normal rated power.

(U) SERVICE CEILING - The altitude where the rate of climb is 100 feet per minute at military rated power.

DRAG INDEX SYSTEM

(U) The DRAG INDEX chart (figure 11-1) presents, in tabular form, the drag count for each externally carried store configuration and its associated suspension equipment. Drag numbers for additional stores will be added to the chart as they become available. The total drag index may then be used to enter the planning data charts. Charts applicable for all loads and configurations are labeled ALL DRAG INDEXES. Charts labeled INDIVIDUAL DRAG INDEXES contain data for a range of drag numbers; i.e., individual curves for a specific drag number. For a listing of the aircraft's station carriage capability, refer to F-14 Tactical Manual, NAVAIR 01-F14A-1T.

STORES STATIONS

DRAG INDEX

- The DRAG INDEX of the clean airplane is zero.
- DRAG INDEX is the total drag level for all stores including mounting hardware and interference.
- DRAG INDEX represents the incompressible drag level.

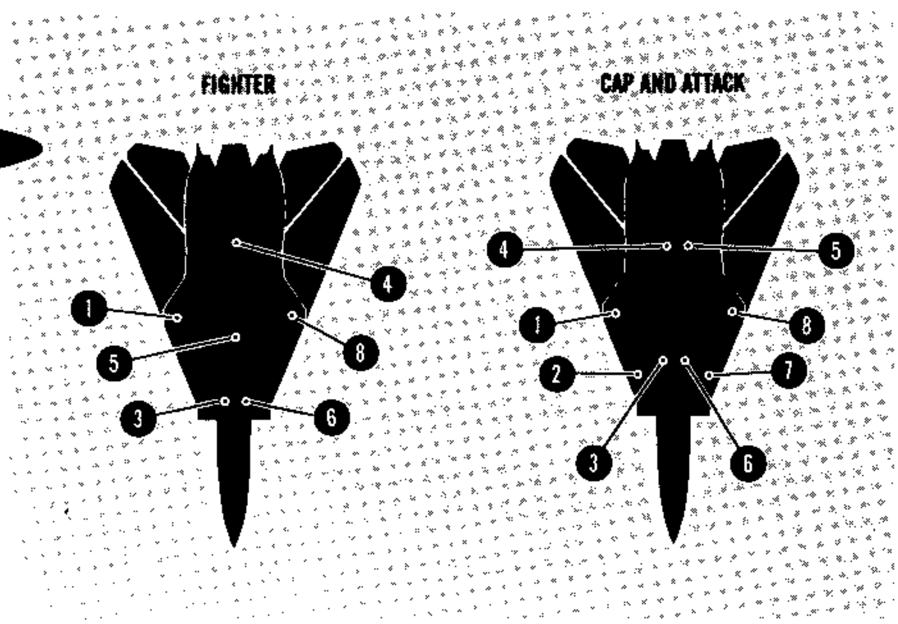


Figure 11-1

(U) STORES CONFIGURATION	DRAG INDEX
• Four channel mounted Sparrows	8
• Four Sparrows plus two 267 gallon nacelle mounted drop tanks	25
• Four Sparrows plus two tank racks	10
• Four Sparrows plus four Sidewinders on glove mounted special "Y" pylons	27
• Four Sparrows plus four Sidewinders plus two 267 gallon drop tanks	44
• Four Phoenix plus two 267 gallon nacelle mounted drop tanks	57
• Four channel mounted Phoenix	40
• Six Phoenix - Four channel mounted plus two glove mounted on MULTI-PURPOSE pylons	75
• Six Phoenix plus two 267 gallon nacelle mounted drop tanks	92
• Six Phoenix plus two tank racks	77
• Six Phoenix plus two Sidewinders mounted on outboard side of MULTI-PURPOSE pylons	92
• Six Phoenix plus two 267 gallon drop tanks plus two Sidewinders	109
• Six channel mounted MK-82 Snakeye	36
• Six MK-82 Snakeye plus two 267 gallon nacelle mounted drop tanks	53
• Fourteen channel mounted MK-82 Snakeye	52
• Fourteen MK-82 Snakeye plus two 267 gallon nacelle mounted drop tanks	69
• Two 267 gallon nacelle mounted drop tanks	17

REFERENCE CHARTS

(U) The reference charts (figure 11-2 and 11-3) are provided for ready use in determining standard units conversion and standard atmospheric conditions. These charts are self explanatory and cover the operating range of the aircraft.

AIRSPEED CONVERSION CHART

(U) This chart (figure 11-4, sheets 1 thru 3) provides a means of converting calibrated airspeed to true Mach number and true airspeed.

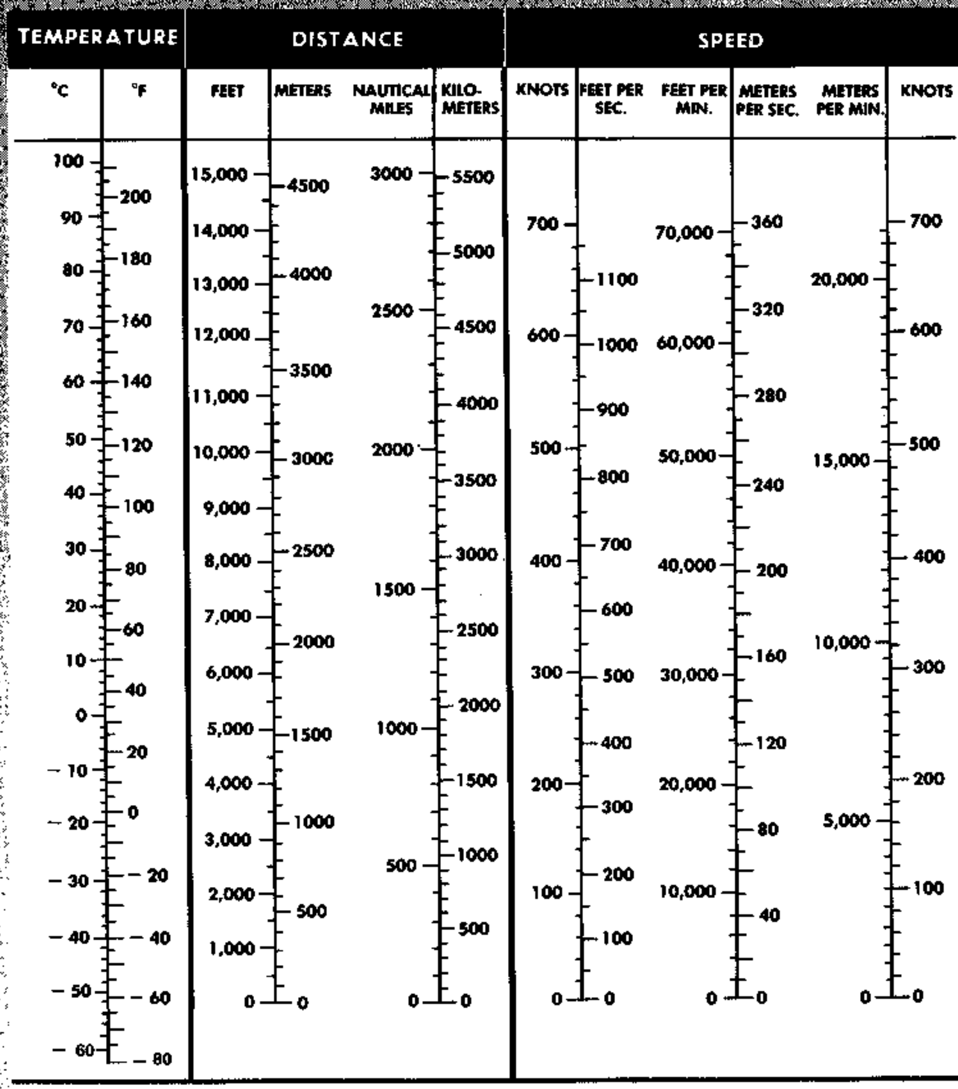
USE

(U) Enter the chart with calibrated airspeed and project vertically upward to the desired pressure altitude. From this point, project horizontally to the left for true Mach number and to the right, parallel to the constant airspeed guidelines, for true airspeed at ICAO standard day conditions. For temperature effects, from the intersection of the calibrated airspeed and pressure altitude, project horizontally to the right to the sea level pressure altitude guideline. At this point of intersection proceed vertically downward to the desired outside air temperature and then project horizontally to the right and read the corrected true airspeed.

(U) SAMPLE PROBLEM

A.	Calibrated airspeed	250 knots
B.	Pressure altitude	35,000 feet
C.	True Mach number	.74 M
D.	True airspeed (ICAO standard day)	427 knots
E.	Outside air temperature	-40°C
F.	True airspeed	432 knots

STANDARD UNITS CONVERSION CHART



NOTE

- TO OBTAIN US GALLONS MULTIPLY LITERS BY 0.264
- TO OBTAIN IMPERIAL GALLONS MULTIPLY LITERS BY 0.220
- TO OBTAIN INCHES OF MERCURY MULTIPLY MILLIBARS BY 0.0295
- TO OBTAIN POUNDS MULTIPLY KILOGRAMS BY 2.20

1-F050-004
058-0

Figure 11-2

STANDARD ATMOSPHERE TABLE

STANDARD SL CONDITIONS

TEMPERATURE 15°C (59°F)
PRESSURE 29.921 IN. HG 2116.216 LB/SQ FT
DENSITY .0023769 SLUGS/CU FT
SPEED OF SOUND 1116.89 FT/SEC 661.7 KTS

CONVERSION FACTORS

1 IN. HG 70.727 LB/SQ FT
1 IN. HG 0.49116 LB/SQ IN.
1 KNOT 1.688 FT/SEC
1 KNOT 1.151 MPH

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE		SPEED OF SOUND KNOTS	PRESS. IN. HG	PRESS. RATIO δ
			°C	°F			
0	1.000	1.0000	15.000	59.000	661.7	29.921	1.0000
1000	.9711	1.0148	13.019	55.434	659.5	28.856	.9644
2000	.9428	1.0299	11.038	51.868	657.2	27.821	.9298
3000	.9151	1.0454	9.056	48.302	654.9	26.817	.8962
4000	.8881	1.0611	7.076	44.735	652.6	25.842	.8637
5000	.8617	1.0773	5.094	41.169	650.3	24.896	.8320
6000	.8359	1.0938	3.113	37.603	648.7	23.978	.8014
7000	.8106	1.1107	1.132	34.037	645.6	23.088	.7716
8000	.7860	1.1279	-0.850	30.471	643.3	22,225	.7428
9000	.7620	1.1456	-2.831	26.905	640.9	21.388	.7148
10,000	.7385	1.1657	-4.812	23.338	638.6	20.577	.6877
11,000	.7155	1.1822	-6.793	19.772	636.2	19.791	.6614
12,000	.6932	1.2011	-8.774	16.206	633.9	19.029	.6360
13,000	.6713	1.2205	-10.756	12.640	631.5	18.292	.6113
14,000	.6500	1.2403	-12.737	9.074	629.0	17.577	.5875
15,000	.6292	1.2606	-14.718	5.508	626.0	16.886	.5643
16,000	.6090	1.2816	-16.699	1.941	624.2	16.216	.5420
17,000	.5892	1.3028	-18.680	-1.625	621.8	15.569	.5203
18,000	.5699	1.3246	-20.662	-5.191	619.4	14.942	.4994
19,000	.5511	1.3470	-22.643	-8.757	617.0	14.336	.4791
20,000	.5328	1.3700	-24.624	-12.323	614.6	13.750	.4595
21,000	.5150	1.3935	-26.605	-15.889	612.1	13.184	.4406
22,000	.4976	1.4176	-28.587	-19.456	609.6	12.636	.4223
23,000	.4806	1.4424	-30.568	-23.022	607.1	12.107	.4046
24,000	.4642	1.4678	-32.549	-26.588	604.6	11.597	.3876
25,000	.4481	1.4938	-34.530	-30.154	602.1	11.103	.3711
26,000	.4325	1.5206	-36.511	-33.720	599.6	10.627	.3552
27,000	.4173	1.5480	-38.492	-37.286	597.1	10.168	.3398
28,000	.4025	1.5762	-40.474	-40.852	594.6	9.725	.3250
29,000	.3881	1.6052	-42.455	-44.419	592.1	9.297	.3107
30,000	.3741	1.6349	-44.436	-47.985	589.5	8.885	.2970
31,000	.3605	1.6654	-46.417	-51.551	586.9	8.488	.2837
32,000	.3473	1.6968	-48.398	-55.117	584.4	8.106	.2709
33,000	.3345	1.7291	-50.379	-58.683	581.8	7.737	.2586
34,000	.3220	1.7623	-52.361	-62.249	579.2	7.382	.2467
35,000	.3099	1.7964	-54.243	-65.816	576.6	7.041	.2353
36,000	.2981	1.8315	-56.323	-69.382	574.0	6.712	.2243
36,089	.2971	1.8347	-56.500	-69.700	573.7	6.683	.2234
37,000	.2843	1.8753				6.397	.2138
38,000	.2710	1.9209				6.097	.2038
39,000	.2583	1.9677				5.811	.1942
40,000	.2462	2.0155				5.538	.1851

1-F050-004
059-1

Figure 11-3. (Sheet 1 of 2)

STANDARD ATMOSPHERE TABLE

STANDARD SL CONDITIONS

TEMPERATURE 15°C (59°F)
PRESSURE 29.921 IN. HG 2116.216 LB/SQ FT
DENSITY .0023769 SLUGS/CU FT
SPEED OF SOUND 1116.89 FT/SEC 661.7 KTS

CONVERSION FACTORS

1 IN. HG 70.727 LB/SQ FT
1 IN. HG 0.49116 LB/SQ IN.
1 KNOT 1.688 FT/SEC
1 KNOT 1.151 MPH

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESS. IN. HG	PRESS. RATIO δ
41,000	.2346	2.0645	-56.500	-69.700	573.7	5.278	.1764
42,000	.2236	2.1148				5.030	.1681
43,000	.2131	2.1662				4.794	.1602
44,000	.2031	2.2189				4.569	.1527
45,000	.1936	2.2728				4.355	.1455
46,000	.1845	2.3281				4.151	.1387
47,000	.1758	2.3848				3.956	.1322
48,000	.1676	2.4428				3.770	.1260
49,000	.1597	2.5022				3.593	.1201
50,000	.1522	2.5630				3.425	.1145
51,000	.1451	2.6254				3.264	.1091
52,000	.1383	2.6892				3.111	.1040
53,000	.1318	2.7546				2.965	.09909
54,000	.1256	2.8216				2.826	.09444
55,000	.1197	2.8903				2.693	.09001
56,000	.1141	2.9606				2.567	.08578
57,000	.1087	3.0326				2.446	.08176
58,000	.1036	3.1063				2.331	.07792
59,000	.09877	3.1819				2.222	.07426
60,000	.09414	3.2593				2.118	.07078
61,000	.08972	3.3386	-56.500	-69.700	573.7	2.018	.06746
62,000	.08551	3.4198				1.924	.06429
63,000	.08150	3.5029				1.833	.06127
64,000	.07767	3.5881				1.747	.05840
65,000	.07403	3.6754				1.665	.05566
66,000	.07055	3.7649				1.587	.05305
67,000	.06724	3.8564				1.513	.05056
68,000	.06409	3.9502				1.442	.04819
69,000	.06108	4.0463				1.374	.04592
70,000	.05821	4.1447				1.310	.04377
71,000	.05548	4.2456				1.248	.04171
72,000	.05288	4.3488				1.190	.03976
73,000	.05040	4.4545				1.134	.03789
74,000	.04803	4.5633				1.081	.03611
75,000	.04578	4.6738				1.030	.03442
76,000	.04363	4.7874				0.982	.03280
77,000	.04158	4.9039				0.935	.03126
78,000	.03963	5.0231				0.892	.02980
79,000	.03777	5.1454				0.850	.02840
80,000	.03600	5.2706				0.810	.02707

1-F050-004
059-2

Figure 11-3. (Sheet 2 of 2)

AIRSPEED CONVERSION

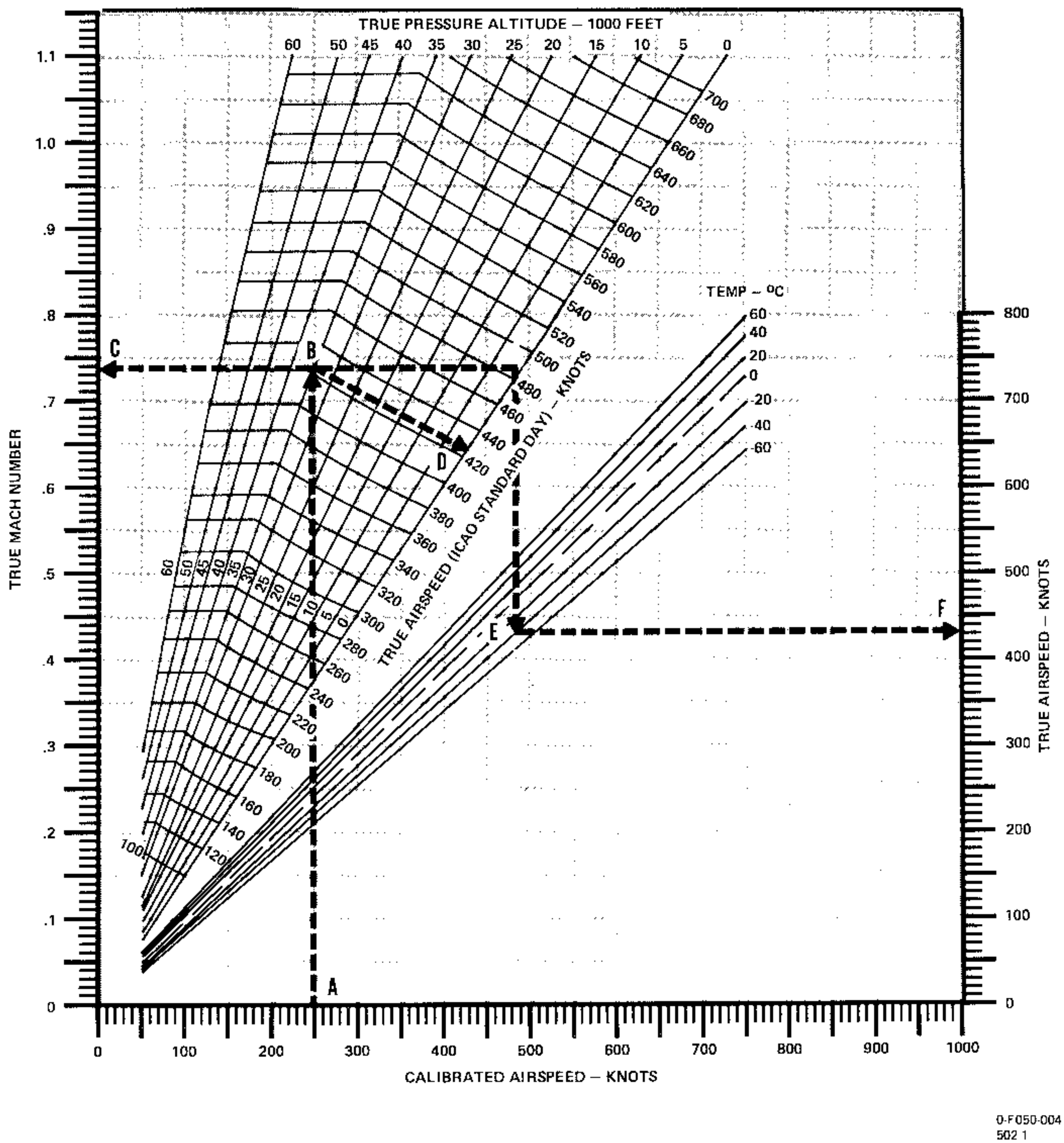


Figure 11-4. (Sheet 1 of 3)

AIRSPEED CONVERSION

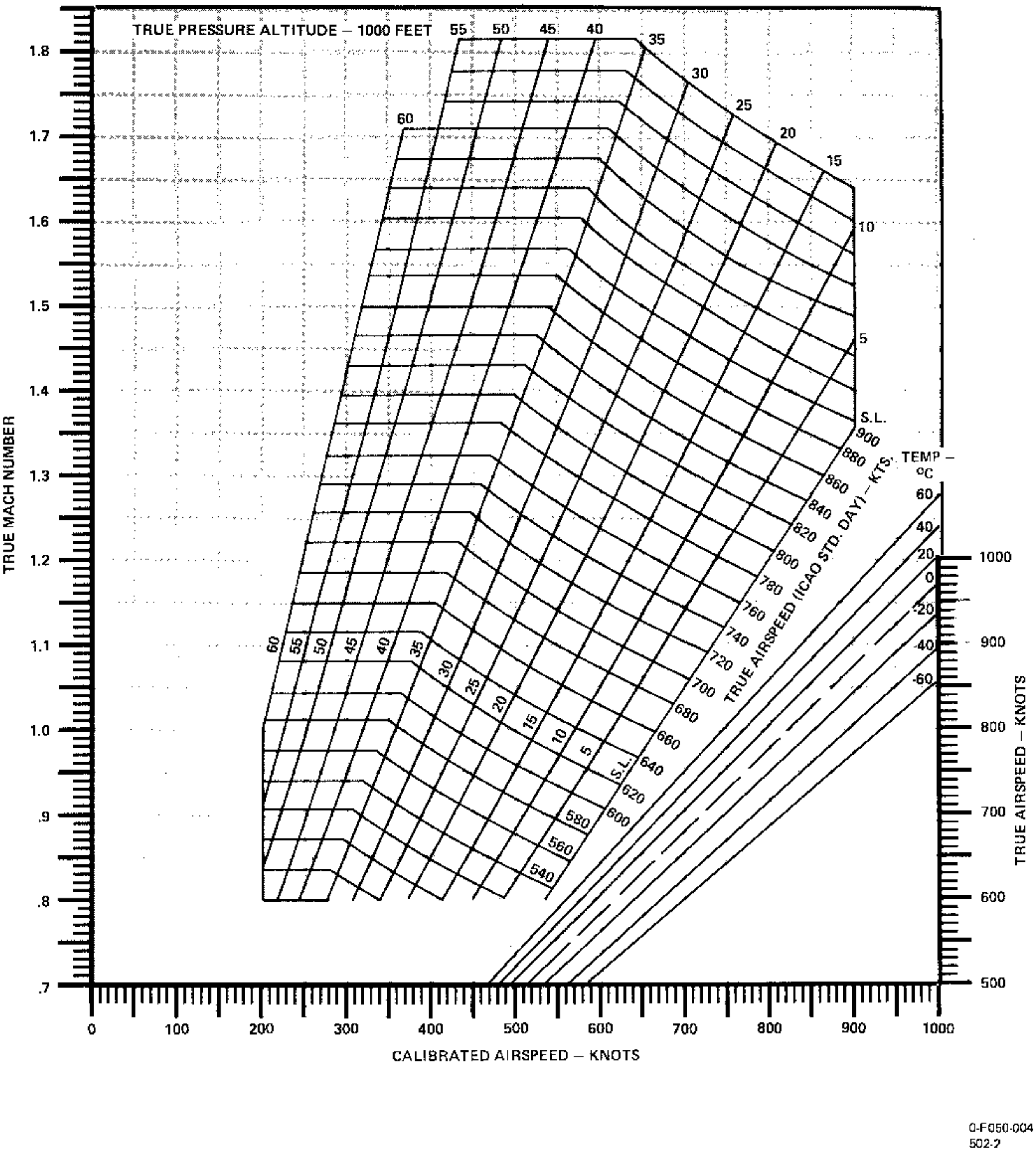
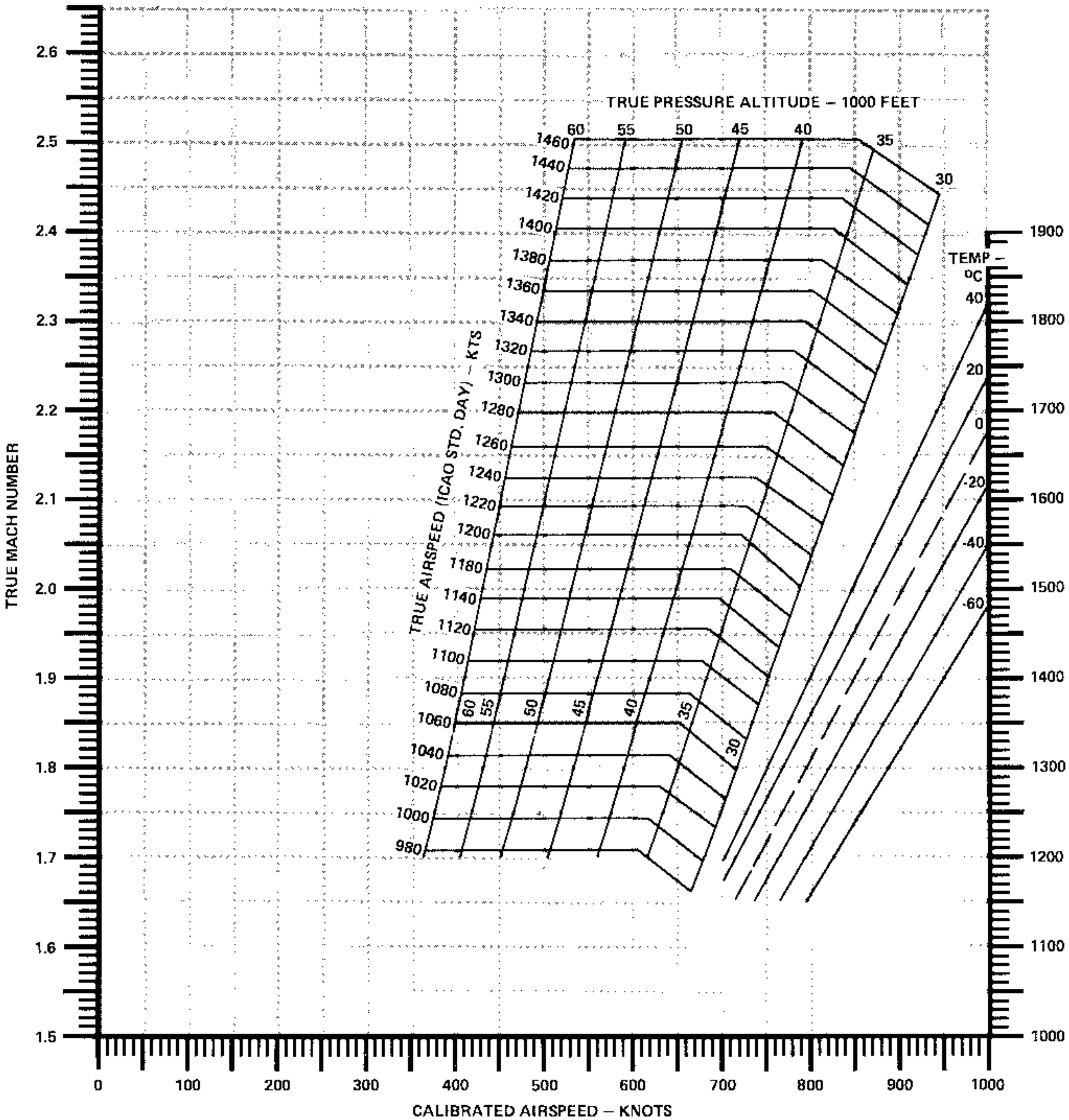


Figure 11-4. (Sheet 2 of 3)

AIRSPEED CONVERSION



0-F050-004
502-3

Figure 11-4.(Sheet 3 of 3)

TAKEOFF AND LANDING WIND COMPONENTS CHART

(U) A standard take-off and landing wind components chart (figure 11-5) is presented for computation of crosswind and headwind components. Crosswind directions are presented from 0 to 90 degrees in 10 degree increments and windspeeds from 0 to 60 knots in 1 knot increments.

USE

(U) Reduce the wind direction to a relative bearing by determining the difference between wind direction and runway heading. Enter the chart with a relative bearing of 20 degrees. Move along the relative bearing angle to intercept the windspeed arc of 35 knots. From this point, move vertically downward to read a crosswind component of 12 knots. To find the headwind component, move horizontally to the left to read a headwind of 33 knots.

(U) SAMPLE PROBLEM

A.	Windspeed	35 knots
B.	Wind direction	050 degrees
C.	Runway Heading	030 degrees
D.	Crosswind component	12 knots
E.	Headwind component	33 knots

WIND EFFECT ON TAKEOFF AND LANDING

(U) This chart (figure 11-6) provides the capability of adjusting the computed takeoff and landing distance for wind effects.

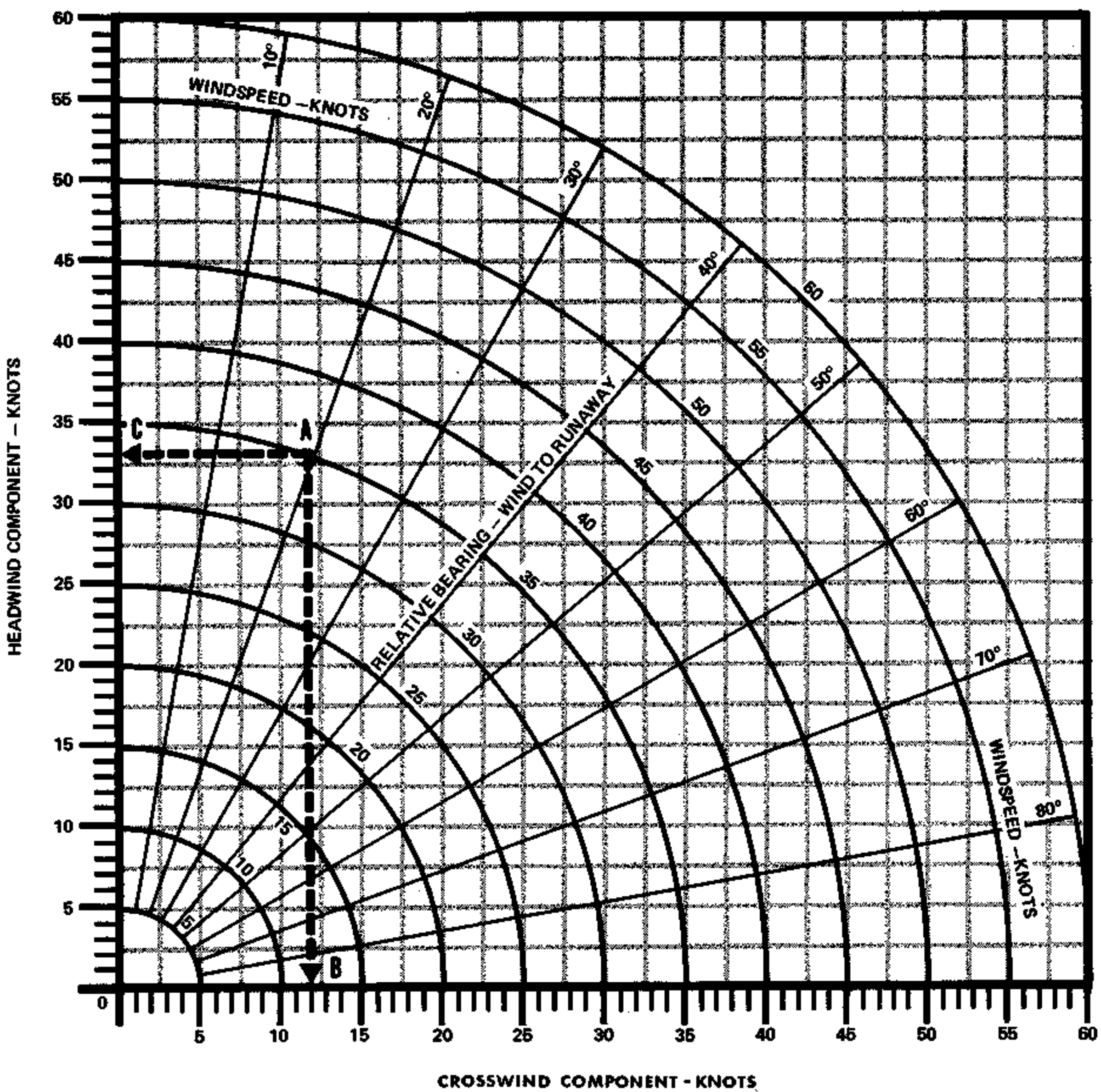
USE

(U) Enter the chart with the effective wind velocity (headwind or tailwind) and project horizontally to the right and intersect the previously computed aircraft takeoff speed. From this point, descend vertically to intersect previously computed takeoff ground run. At this point of intersection, project horizontally to the left to read distance adjusted for wind effect.

(U) SAMPLE PROBLEM

A.	Effective wind	15 knots
B.	Takeoff speed (previously computed)	100 KCAS
C.	Ground run distance (without wind)	3,000 feet
D.	Ground run distance (with wind)	2,200 feet

TAKE-OFF AND LANDING CROSSWIND CHART



0-F050-004
524-0

Figure 11-5.

WIND EFFECTS ON TAKEOFF AND LANDING

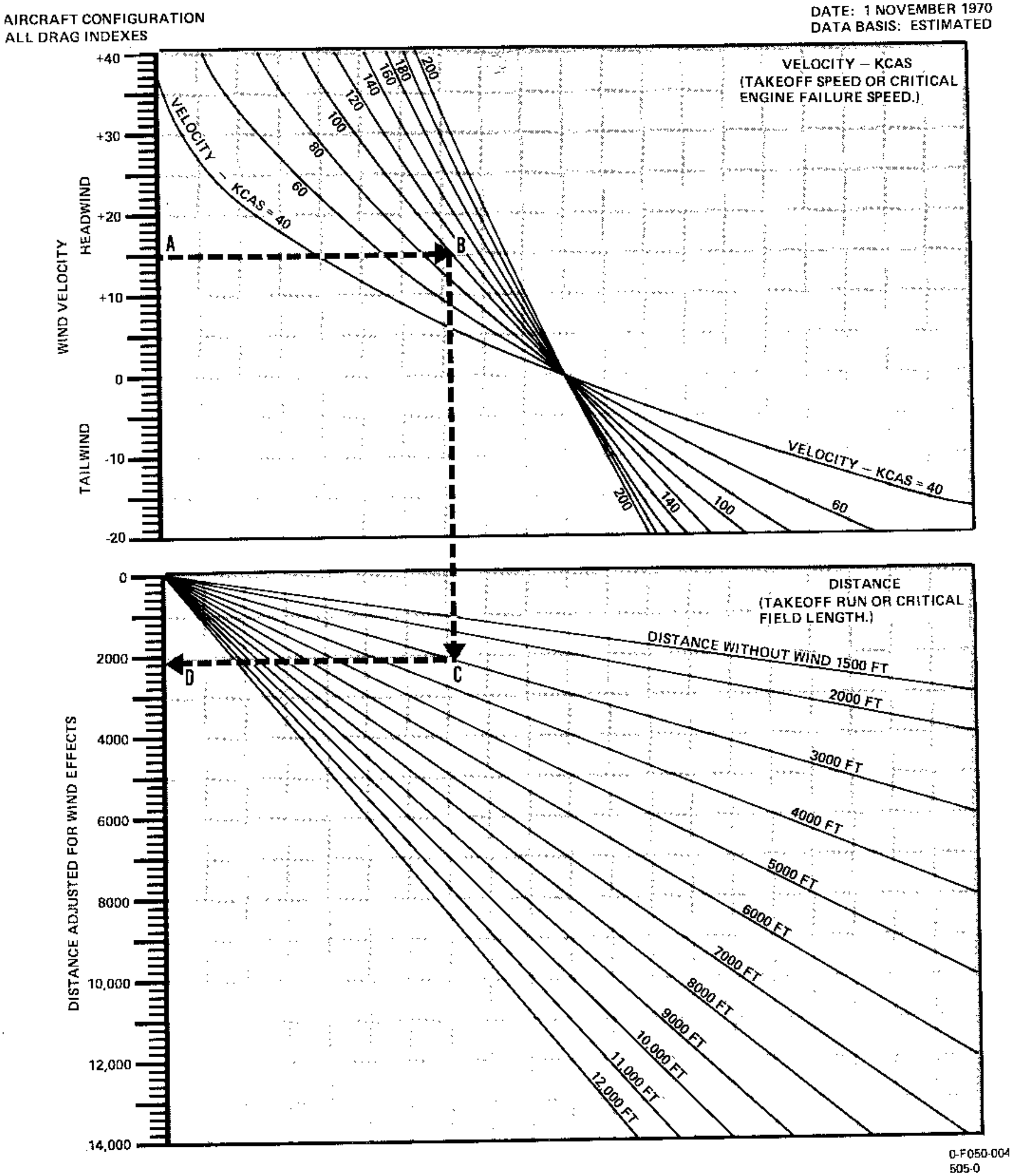


Figure 11-6.

part 2 Takeoff

TAKEOFF

(U) This part presents data necessary to takeoff planning for a wing sweep angle of 20 degrees and for military power thrust setting. The takeoff charts are presented for a flap setting of 35 degrees deflection. The performance data are computed for normal takeoff technique on a hard, dry runway, and represents two primary stores loading. For each chart an example of its use and a sample problem are shown as a guide in avoiding misinterpretation.

TAKEOFF SPEED AND DISTANCE

(U) The basic takeoff charts (figures 11-7, sheet 1 and 2) show speed, distance in ground roll and distance to clear an obstacle as affected by air temperature and gross weight at sea level field elevation.

USE

(U) Enter the chart at the applicable temperature and proceed vertically to intersect the sea level guide line. From this point, proceed horizontally to the right and intersect the desired takeoff gross weight line (note aircraft takeoff speed). Then descend vertically to read ground run distance. To find the total distance required to clear an obstacle, continue downward to the desired obstacle height and project horizontally to the left scale. Refer to the WIND EFFECTS ON TAKEOFF AND LANDING chart Part 1, to determine wind effects on these distances.

SAMPLE PROBLEM

(C) Configuration: Four (4) Sparrow (figure 11-7, sheet 1)

A.	Temperature	15°C
B.	Sea level guide line	intersect
C.	Gross weight	58,000 pounds
D.	Takeoff speed	121 KCAS

E. Ground run distance — 2,160 feet

F. Obstacle height — 50 feet

G. Total distance required to clear obstacle — 3,150 feet

(C) Configuration: Six (6) Phoenix Plus Two (2) Drop Tanks (figure 11-7, sheet 2)

A. Temperature — 15°C

B. Sea level guide line — intersect

C. Gross weight — 66,000 pounds

D. Takeoff speed — 130 KCAS

E. Ground run distance — 2970 pounds

F. Obstacle height — 50 feet

G. Total distance required to clear obstacle — 4,050 feet

TAKEOFF SPEED AND DISTANCE

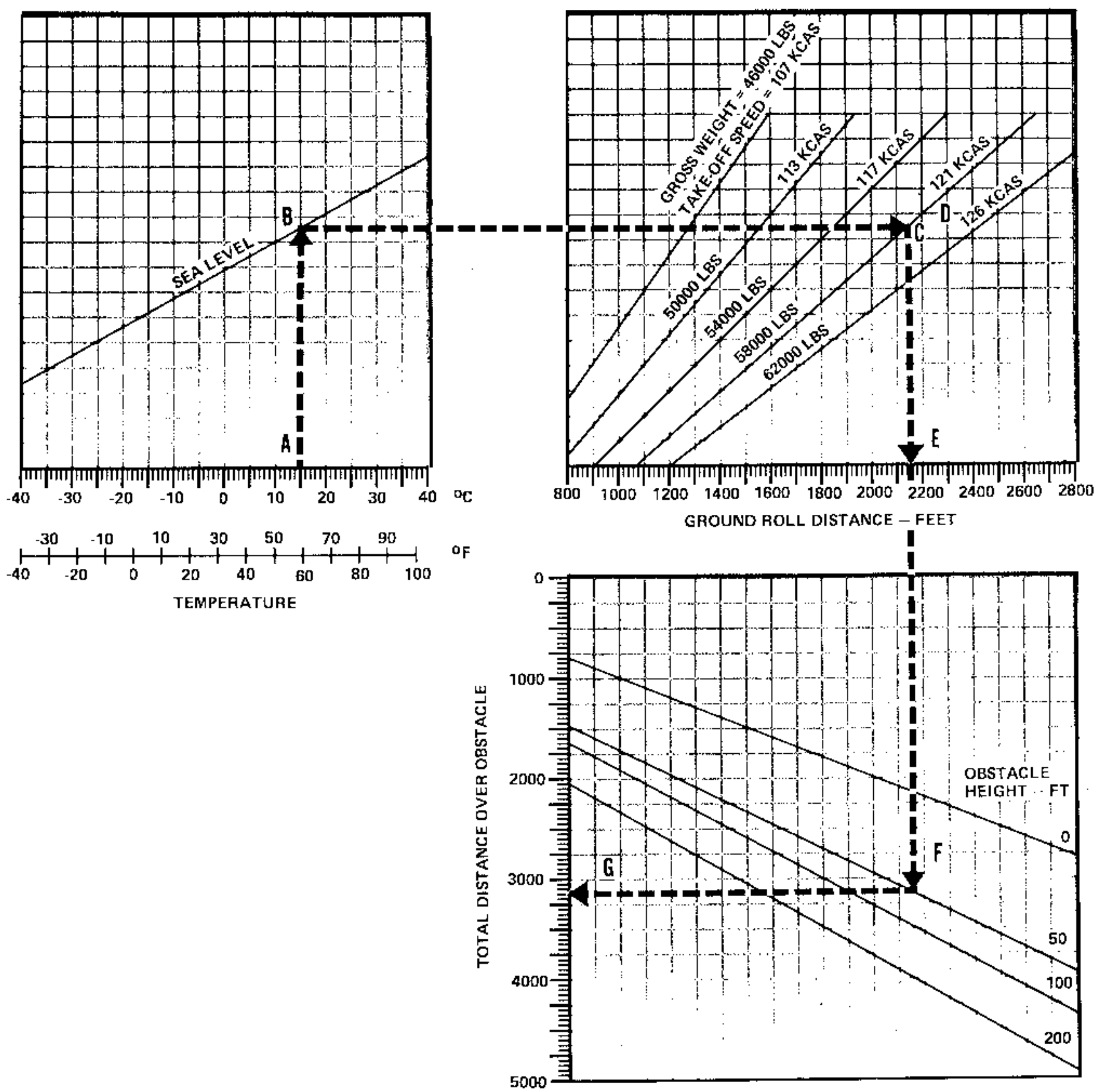
WING SWEEP $\Lambda_{LE} = 20°$

AIRCRAFT CONFIGURATION
FOUR (4) SPARROWS
35° FLAP
GEAR DOWN

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
MILITARY POWER
HARD DRY RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
504 1

Figure 11-7. (Sheet 1 of 2)

TAKEOFF SPEED AND DISTANCE

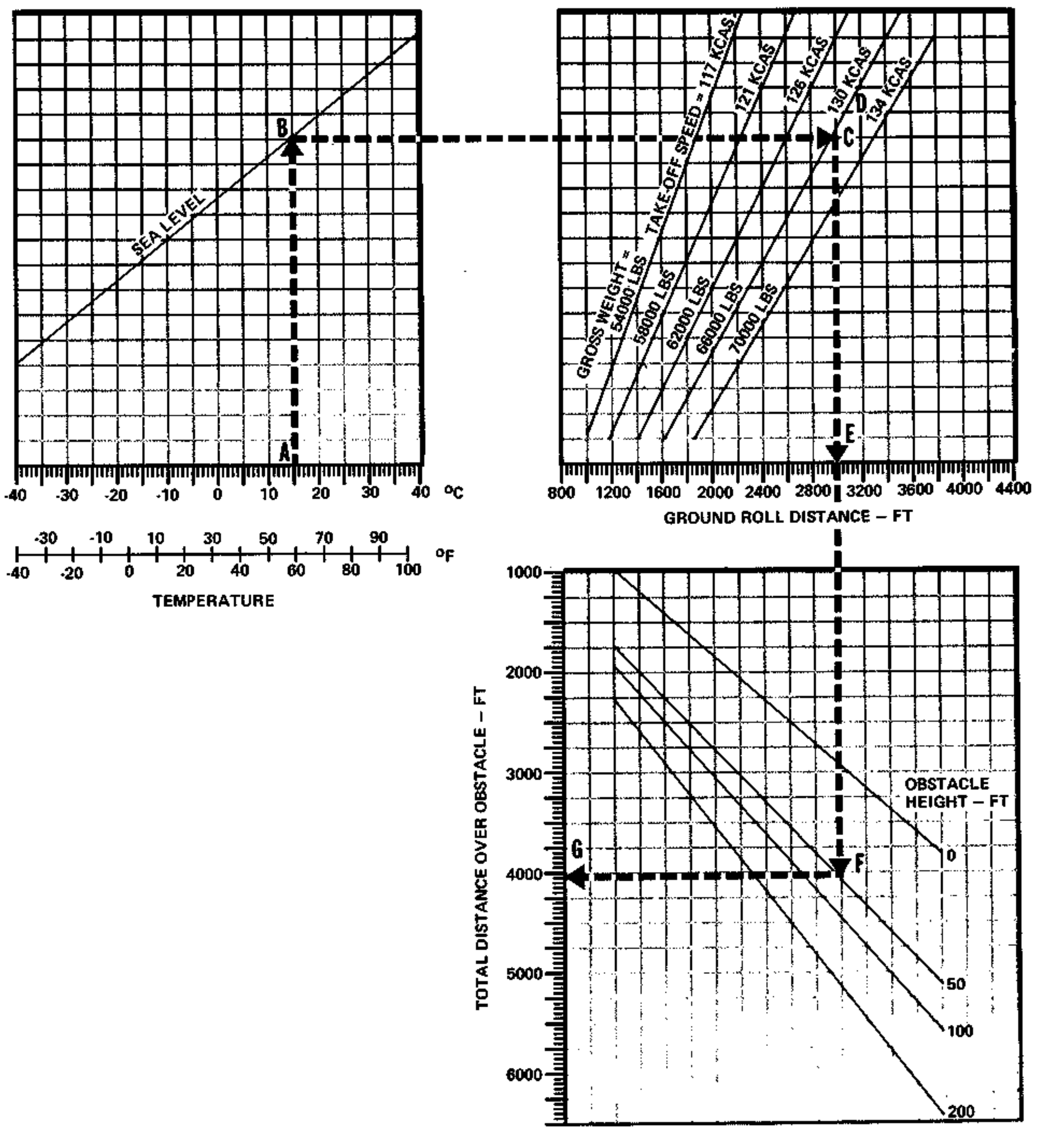
WING SWEEP $\Lambda_{LE} = 20°$

AIRCRAFT CONFIGURATION
SIX (6) PHOENIX PLUS TWO (2) DROP TANKS
35° FLAP
GEAR DOWN

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
MILITARY POWER
HARD DRY RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
504-2

Figure 11-7. (Sheet 2 of 2)

TAKEOFF REFUSAL SPEED

(U) The TAKEOFF REFUSAL SPEED charts (figures 11-8, sheets 1 and 2) present the maximum speed at which the aircraft can accelerate and then stop in the remaining available runway length. Stopping distance is based on using normal braking, with spoilers and speed brakes fully deflected (55 degrees and 60 degrees respectively).

USE

(U) Enter the chart at the aircraft gross weight and proceed horizontally to the intersection of the sea level guide line. Descend to intersect the temperature scale and proceed horizontally to intersect the total runway length. At this point of intersection go up vertically to read the maximum refusal speed.

SAMPLE PROBLEM

(C) Configuration: Four (4) Sparrow (figure 11-8, sheet 1)

A. Gross weight	58,000 pounds
B. Sea level guide line	intersect
C. Temperature	15°C
D. Runway length	8,000 feet
E. Refusal speed	178 KCAS

(C) Configuration: Six (6) Phoenix Plus Two (2) Drop Tanks (figure 11-8, sheet 2)

A. Gross weight	66,000 pounds
B. Sea level guide line	intersect
C. Temperature	15°C
D. Runway length	8,000 feet
E. Refusal speed	162 KCAS

REFUSAL SPEED

AIRCRAFT CONFIGURATION
FOUR (4) SPARROW CONFIGURATION
35° FLAP
GEAR DOWN

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
MILITARY POWER
HARD DRY RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

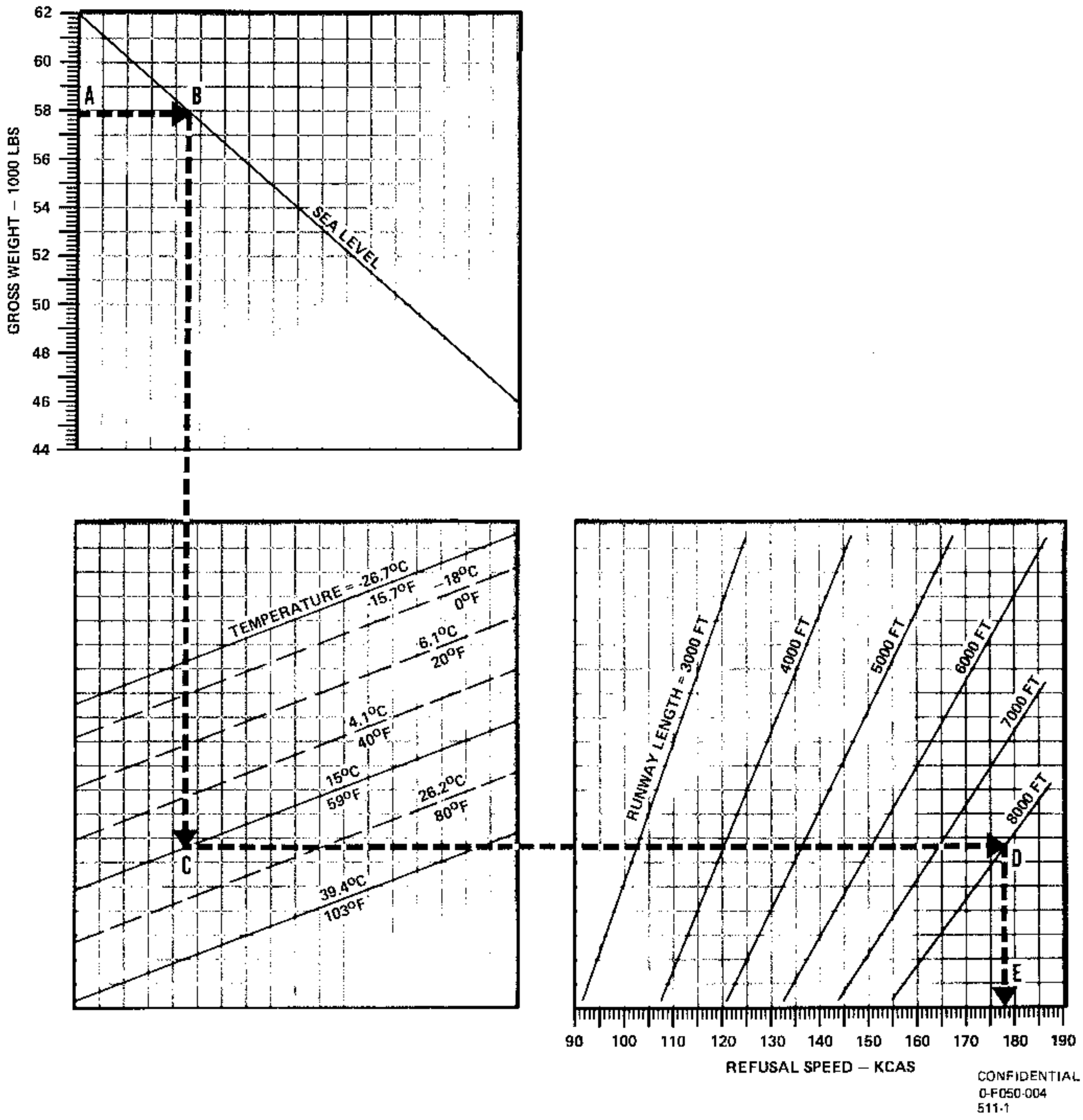


Figure 11-8. (Sheet 1 of 2)

REFUSAL SPEED

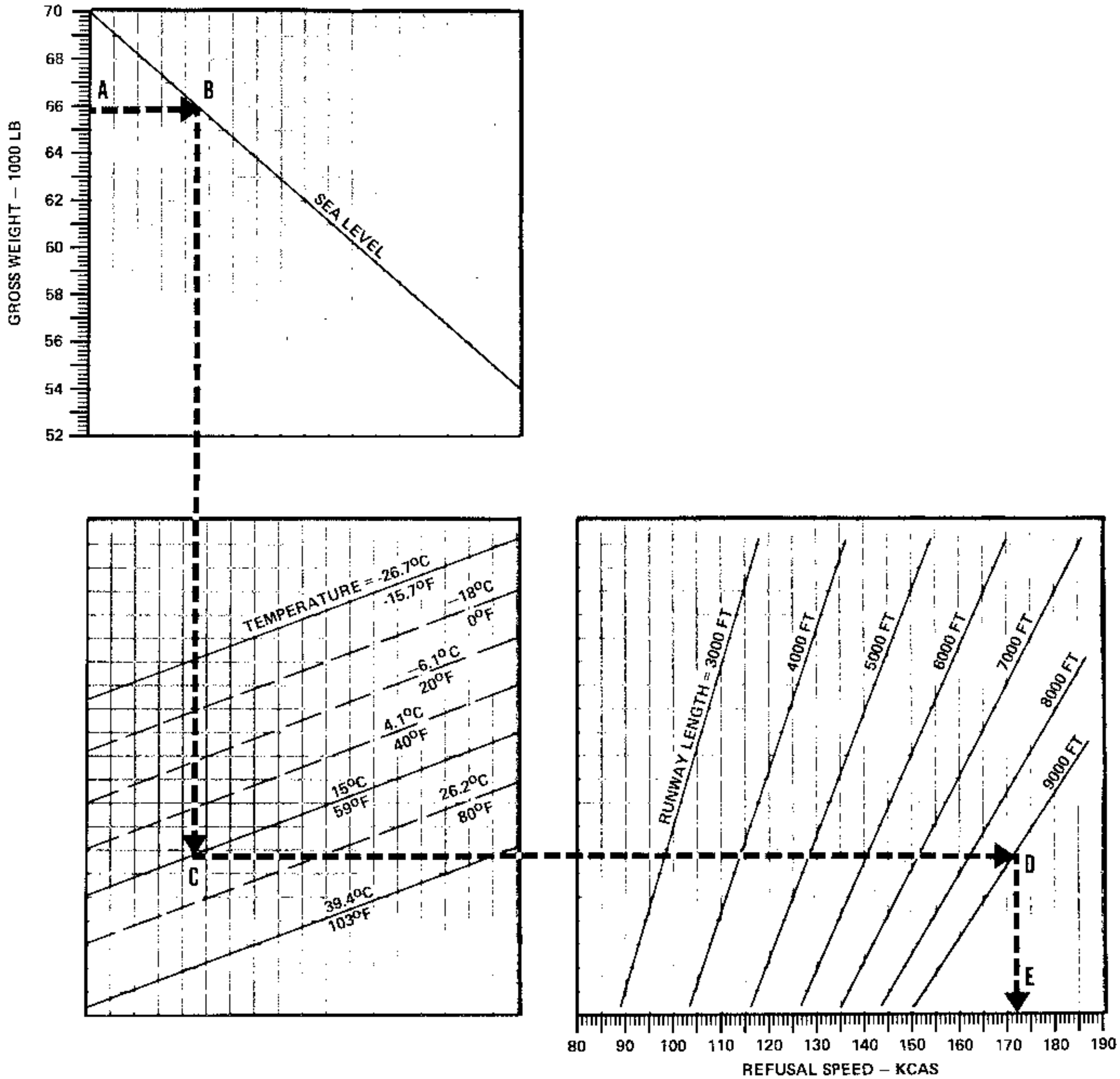
AIRCRAFT CONFIGURATION
SIX (6) PHOENIX PLUS TWO (2) DROP TANKS
35° FLAP
GEAR DOWN

MILITARY POWER

DATE: 1 NOVEMBER, 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
HARD DRY RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
D-F050-004
511-2

Figure 11-8. (Sheet 2 of 2)

part 3 Climb

CLIMB SCHEDULE

(U) Subsonic climb data (figures 11-9 and 11-10, sheets 1 thru 4) are provided for military power climb schedules. Separate charts are utilized to present speed, time, fuel and distance data for the entire drag index range. The charts may be used to obtain climb data from the flaps up condition after takeoff to any desired altitude or incrementally between altitudes.

USE

(U) Climb speed schedule is presented in both table and chart form for a range of drag index listing both calibrated airspeed and Mach number from sea level to 50,000 feet. These data are for all gross weights. From the applicable drag index level read the optimum speed and/or Mach number at the desired altitude.

(U) The method of presenting data on the time, fuel and distance charts is identical and the use of all three charts will be simultaneously undertaken in this example. Enter the charts with the initial climb gross weight and project horizontally to the right to intersect the desired altitude. Project vertically downward to intersect the computed drag index, then horizontally to the left to read time, fuel and distance.

SAMPLE PROBLEM

(C) Climb Speed Schedule (figure 11-10, sheet 1)

A.	Altitude	35,000 feet
B.	Drag index	0
C.	Calibrated airspeed	246 knots
D.	Mach number	.736

(C) Time to Climb (figure 11-10, sheet 2)

A. Gross weight	52,000 pounds
B. Altitude	35,000 feet
C. Drag index	0
D. Time required to climb from sea level to 35,000 feet	6.8 minutes

(C) Fuel to Climb (figure 11-10, sheet 3)

D. Fuel required to climb from sea level to 35,000 feet	1170 pounds

(C) Distance to Climb (figure 11-10, sheet 4)

D. Distance required to climb from sea level to 35,000 feet	48 nautical miles

CLIMB SPEED SCHEDULE TABLE

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

MILITARY POWER

REMARKS
ENGINE(S) : (2) TF 30-P-412
ISAO STANDARD DAY

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY : 6.18 LB/GAL

ALTITUDE FEET	DRAG INDEX											
	0		10		20		30		40		50	
	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH
S.L.	450	.685	443	.675	437	.663	428	.650	420	.640	412	.628
5000	416	.690	410	.680	403	.668	395	.656	390	.646	383	.635
10000	385	.697	377	.685	374	.675	373	.665	363	.655	356	.644
15000	358	.703	349	.694	348	.684	343	.675	337	.666	333	.655
20000	330	.710	327	.702	323	.694	318	.685	313	.677	307	.663
25000	304	.720	300	.711	295	.704	290	.695	289	.689	285	.680
30000	274	.728	270	.720	269	.715	265	.706	264	.701	260	.695
35000	248	.737	246	.733	244	.776	242	.720	240	.715	238	.710
40000	220	.737	219	.733	216	.725	215	.720	213	.715	211	.710
45000	197	.737	196	.733	195	.725	193	.720	192	.715	189	.710
50000	176	.737	175	.733	173	.725	172	.720	170	.715	169	.710

ALTITUDE FEET	DRAG INDEX											
	60		70		80		90		100		110	
	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH
S.L.	405	.615	395	.600	385	.585	377	.573	372	.562	363	.550
5000	377	.623	369	.610	360	.597	354	.586	347	.575	343	.565
10000	350	.634	343	.622	337	.610	330	.600	325	.590	320	.580
15000	328	.645	320	.630	318	.625	312	.616	308	.606	300	.597
20000	305	.659	300	.650	297	.641	291	.630	288	.625	283	.616
25000	284	.674	278	.665	275	.658	272	.652	268	.643	267	.635
30000	258	.689	256	.682	353	.675	250	.670	248	.663	245	.655
35000	237	.705	235	.699	233	.695	232	.690	228	.684	225	.676
40000	210	.705	208	.699	207	.695	205	.690	203	.684	200	.676
45000	188	.705	187	.699	185	.695	184	.690	183	.684	180	.676
50000	168	.705	167	.699	165	.695	164	.690	163	.684	160	.676

ALTITUDE FEET	DRAG INDEX									
	120		130		140		150		160	
	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH	KCAS	MACH
S.L.	355	.540	350	.530	339	.516	330	.501	320	.488
5000	337	.554	370	.546	320	.530	315	.520	305	.506
10000	316	.571	310	.560	300	.548	298	.539	290	.526
15000	297	.589	294	.579	285	.566	275	.555	274	.545
20000	277	.608	275	.599	270	.589	265	.580	262	.570
25000	262	.628	260	.626	255	.616	251	.604	248	.597
30000	242	.649	240	.643	236	.635	233	.628	230	.620
35000	224	.670	223	.665	221	.662	220	.659	219	.652
40000	198	.670	197	.665	195	.662	193	.659	192	.652
45000	178	.670	177	.665	175	.662	174	.659	172	.652
50000	159	.670	158	.665	156	.662	155	.659	154	.652

CONFIDENTIAL
0-F050-004
515-0

Figure 11-9.

CLIMB SPEED SCHEDULE

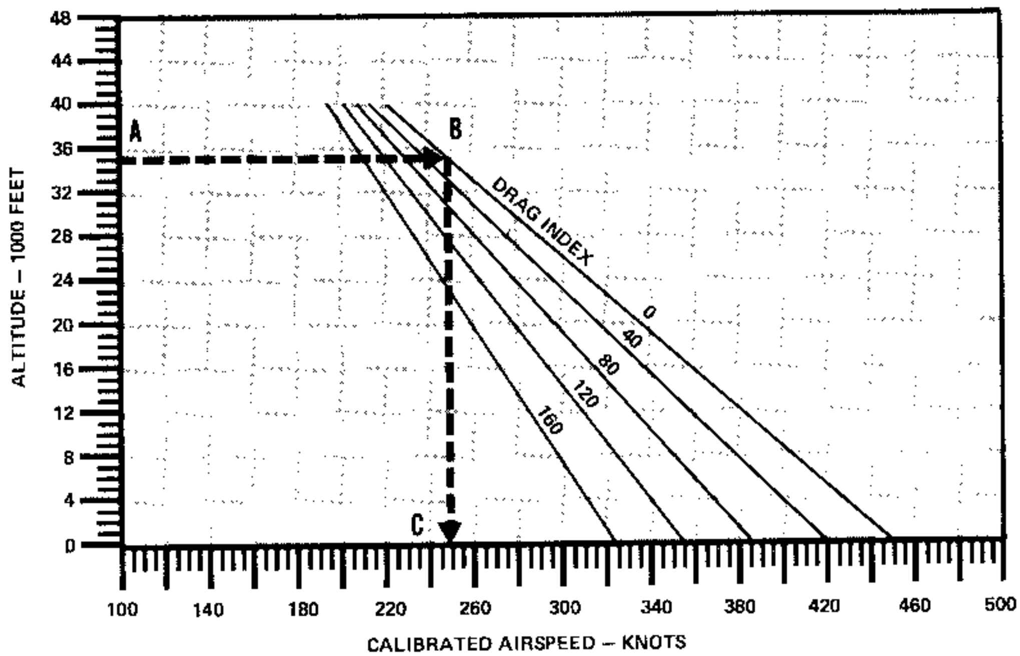
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

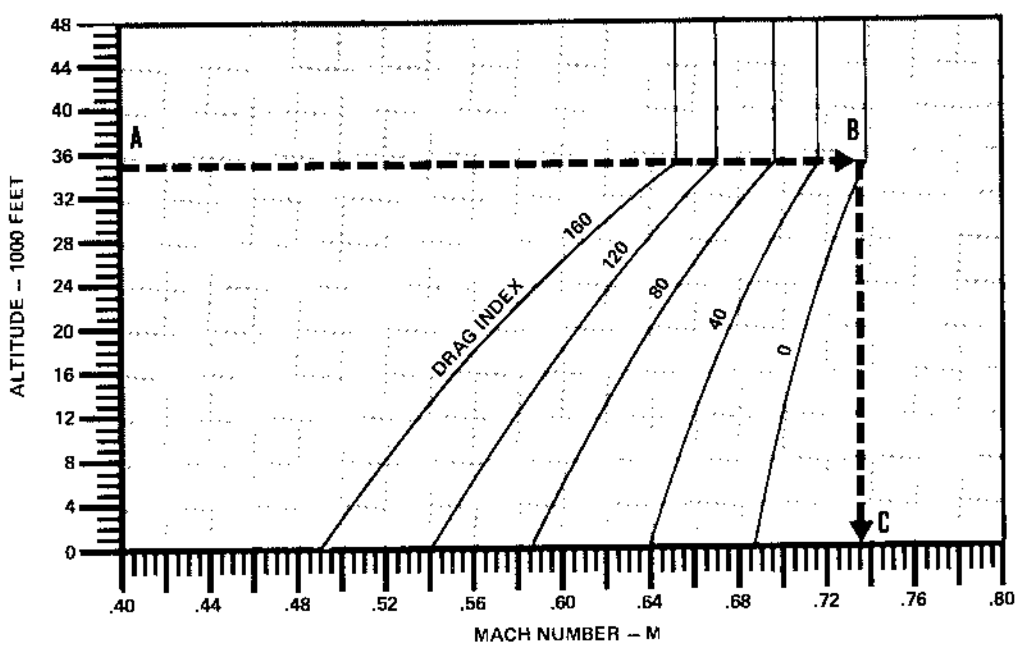
MILITARY POWER
ALL GROSS WEIGHTS

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL.





CONFIDENTIAL
0-F050-004
516-1

Figure 11-10. (Sheet 1 of 4)

CLIMB SPEED SCHEDULE

TIME TO CLIMB

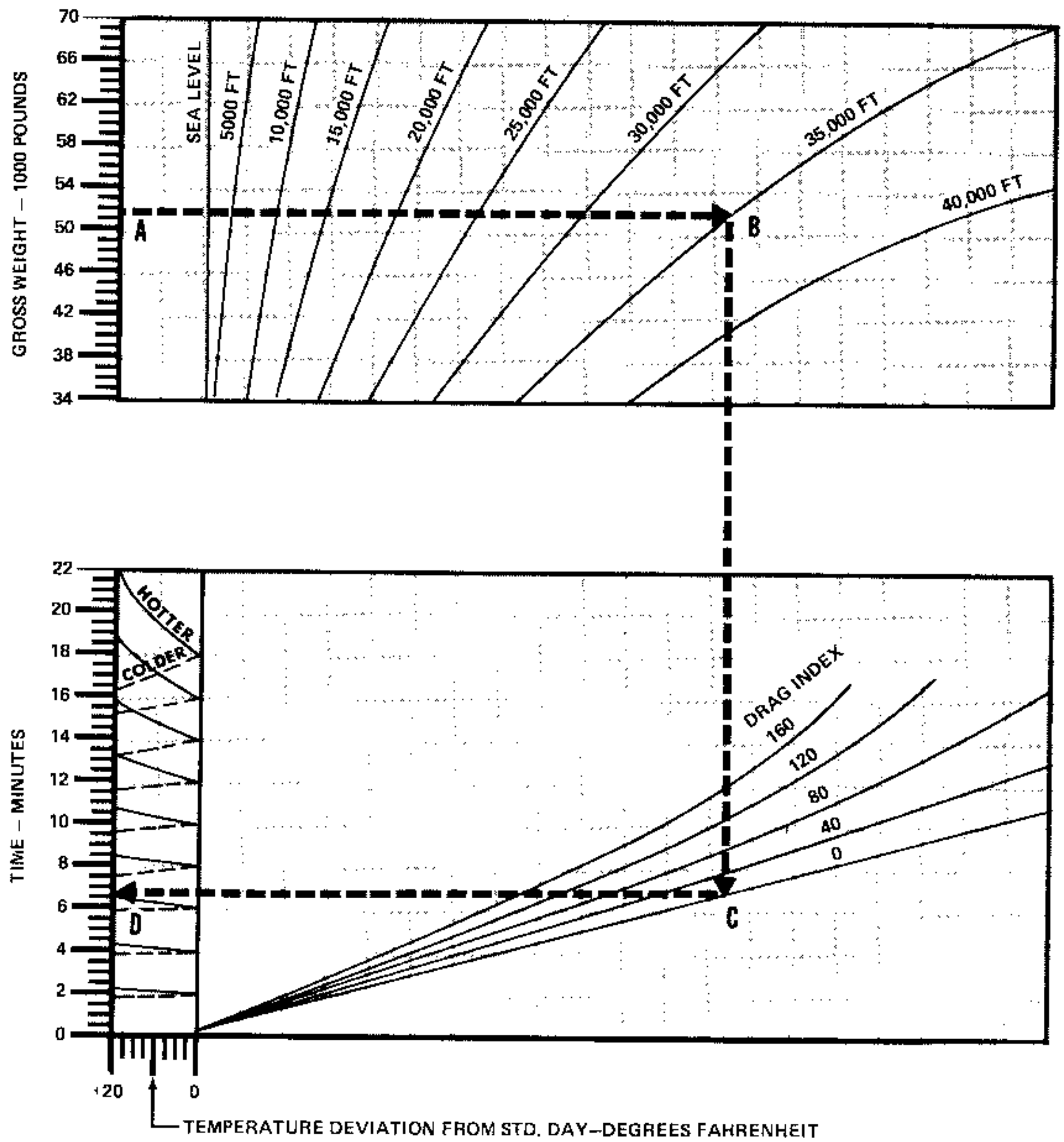
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

MILITARY POWER

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2)TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
516-2

Figure 11-10. (Sheet 2 of 4)

CLIMB SPEED SCHEDULE

FUEL REQUIRED

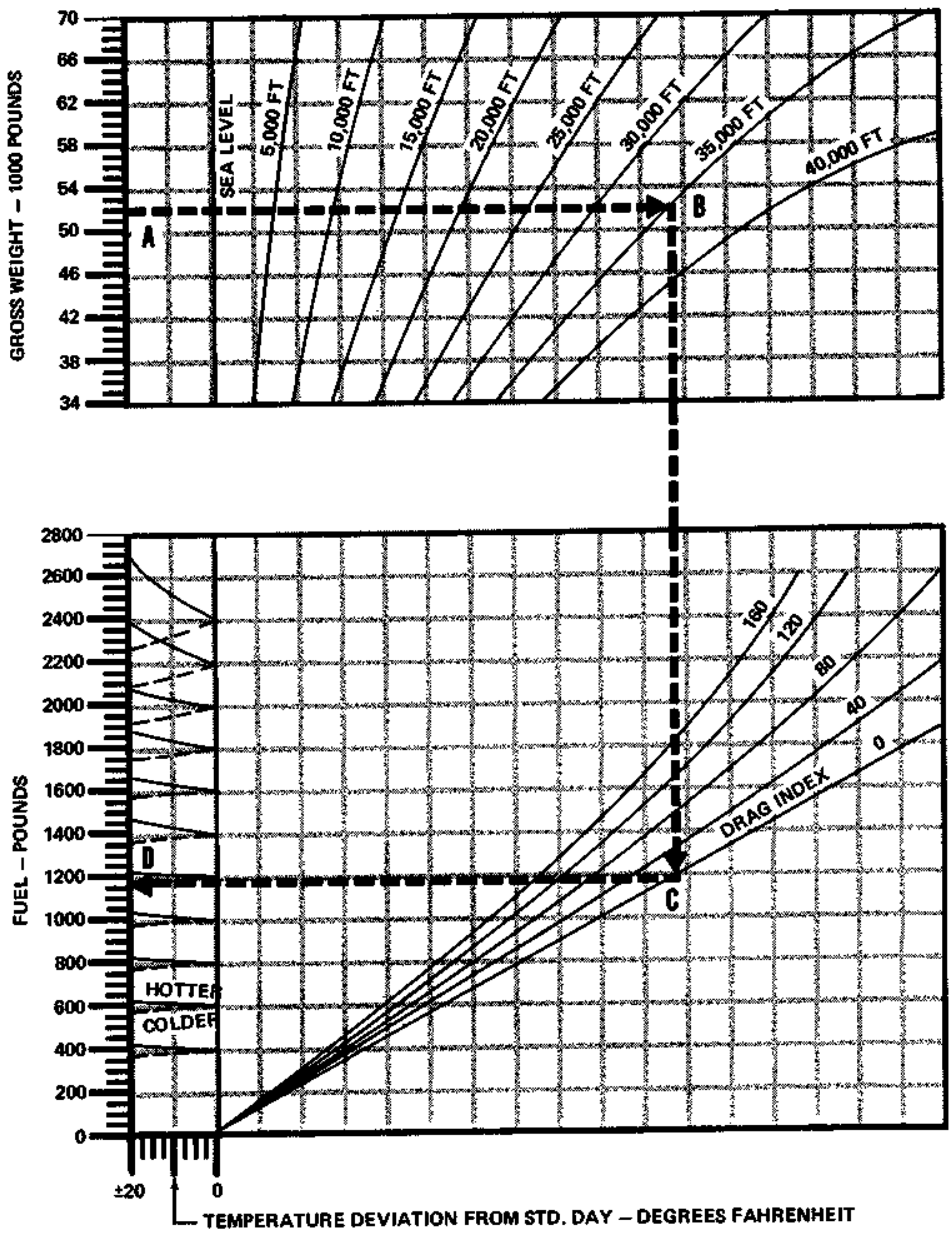
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

MILITARY POWER

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL.



CONFIDENTIAL
0-F050-004
516-3

Figure 11-10. (Sheet 3 of 4)

CLIMB SPEED SCHEDULE

DISTANCE REQUIRED

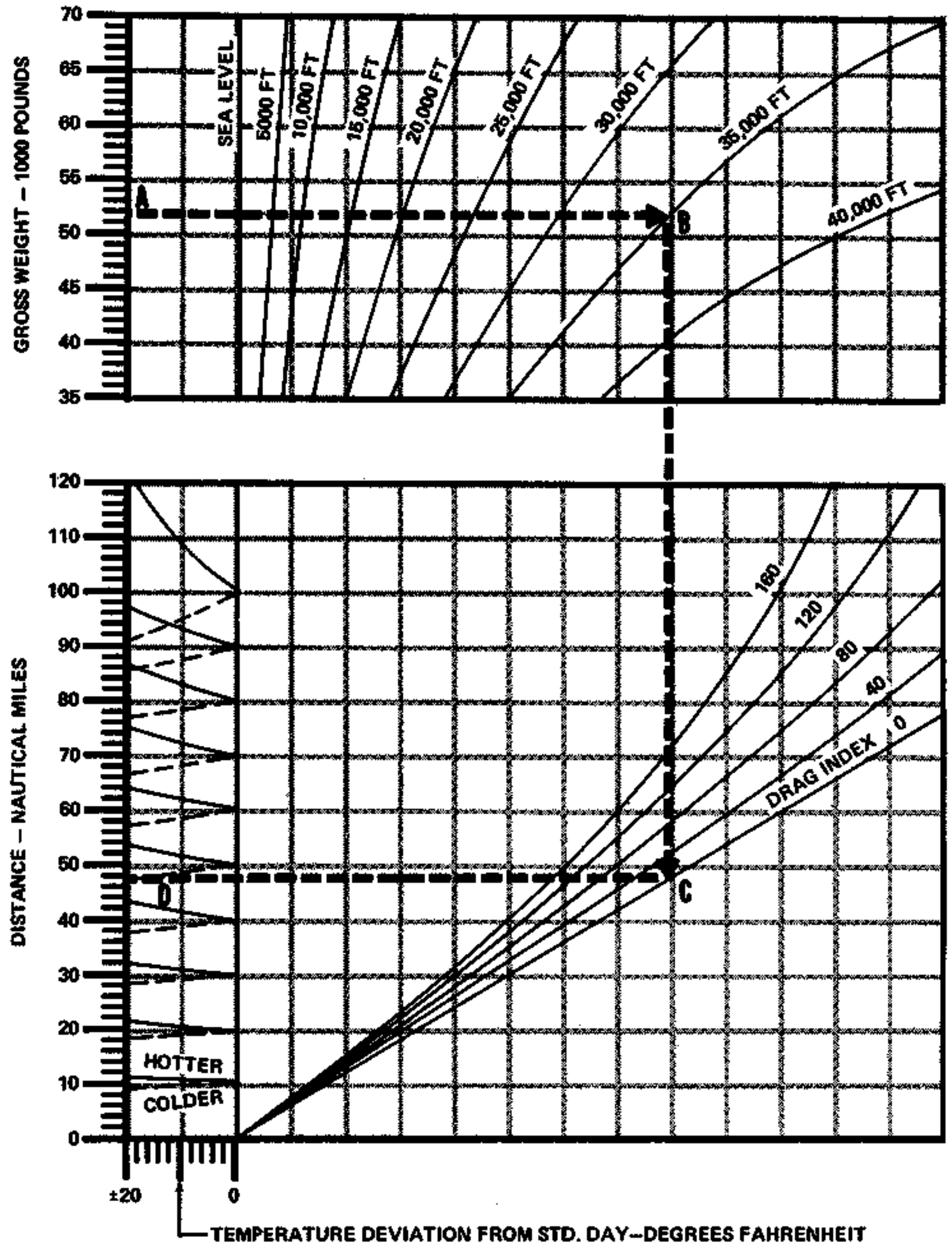
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

MILITARY POWER

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
516-4

Figure 11-10. (Sheet 4 of 4)

ALTITUDE CHARTS

OPTIMUM CRUISE ALTITUDE

(U) Optimum cruise altitude chart, figure 11-11, is shown as a function of gross weight and drag index. The optimum cruise altitude is defined as that altitude/Mach number combination which yields maximum cruise capability (nautical miles per pound of fuel).

USE

(U) Enter the chart at the aircraft gross weight and then proceed horizontally to the reflector line. From this point proceed vertically down to read pressure altitude.

(C) SAMPLE PROBLEM

A. Gross weight	48,000 pounds
B. Drag index	0
C. Optimum cruise altitude	38,500 feet

MAXIMUM ENDURANCE ALTITUDE

(U) Maximum endurance altitude chart, figure 11-11, is shown as a function of gross weight and drag index. The maximum endurance altitude is defined as that altitude which yields maximum time/minimum fuel flow.

USE

(U) Enter the chart at the aircraft gross weight and then proceed horizontally to the desired drag index. From this point proceed vertically down to read pressure altitude.

(C) SAMPLE PROBLEM

A. Gross weight	48,000 pounds
B. Drag index	0
C. Maximum endurance altitude	33,900 feet

CRUISE CEILING

(U) Cruise ceiling chart, figure 11-12, is shown as a function of gross weight and drag index. The cruise ceiling is defined as that altitude for 300 feet per minute rate of climb using normal rated power.

USE

(U) Enter the chart at the aircraft gross weight and then proceed horizontally to the desired drag index. From this point proceed vertically down to read pressure altitude.

(C) SAMPLE PROBLEM

A. Gross weight	48,000 pounds
B. Drag index	0
C. Cruise ceiling	43,700 feet

SERVICE CEILING

(U) Service ceiling chart, figure 11-12, is shown as a function of gross weight and drag index. The service ceiling is defined as that altitude for 100 feet/minute rate of climb using military rated power.

USE

(U) Enter the chart at the aircraft gross weight and then proceed horizontally to the desired drag index. From this point proceed vertically down to read pressure altitude.

(C) SAMPLE PROBLEM

A. Gross weight	48,000 pounds
B. Drag index	0
C. Cruise ceiling	44,000 feet

RATE OF CLIMB (Instantaneous)

(U) Rate of climb chart, figure 11-13, is presented for military thrust for all computed drag indexes. The chart provides the instantaneous rate of climb for any given altitude and gross weight condition.

USE

(U) Enter the chart with the aircraft gross weight and then proceed horizontally to the right to intersect the desired altitude. Reflect downward to intercept the desired drag index and then to the left to read rate of climb in feet per minute.

(C) SAMPLE PROBLEM

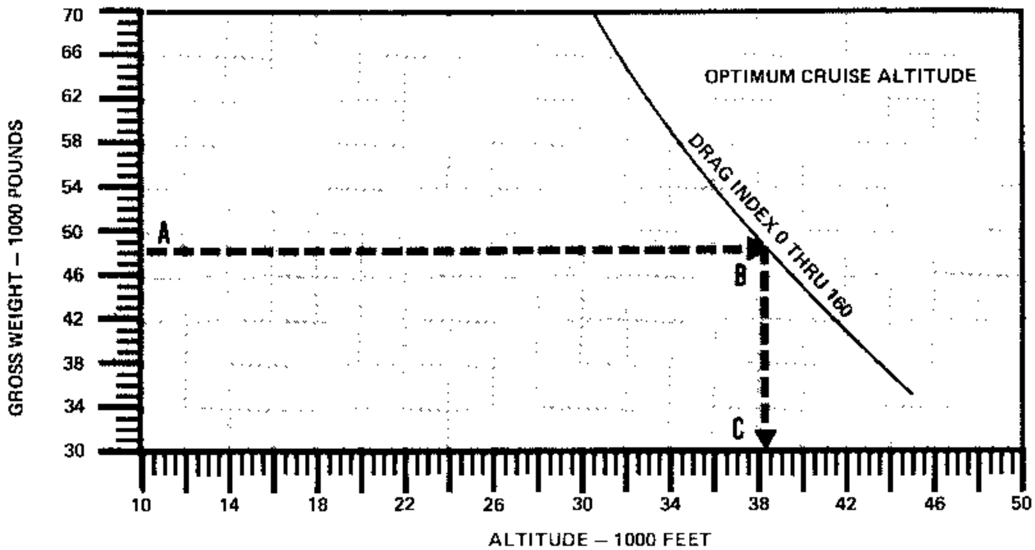
A. Gross weight	48,000 pounds
B. Altitude	35,000 feet
C. Drag index	0
D. Rate of climb	2,500 feet per minute

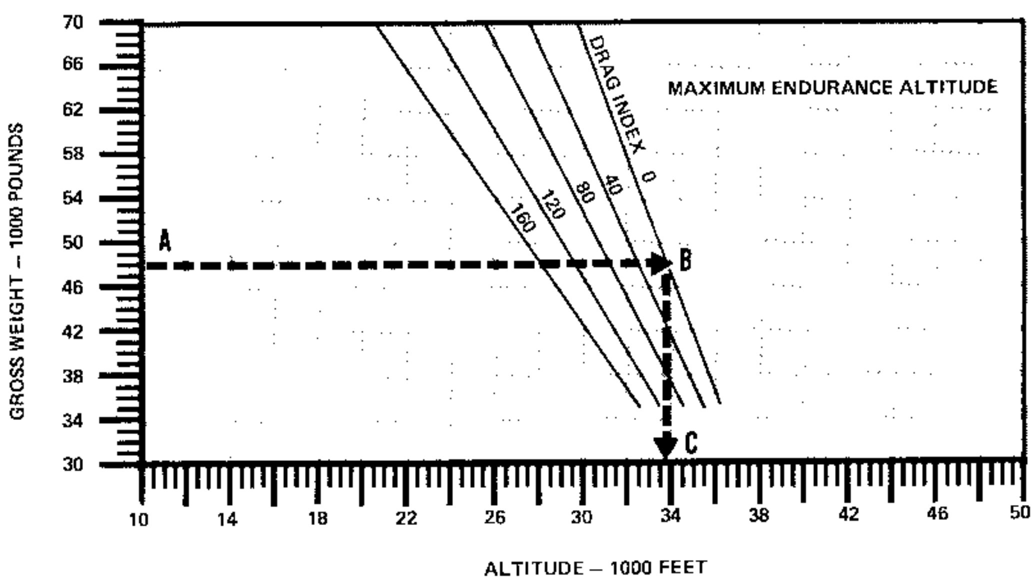
OPTIMUM CRUISE ALTITUDE AND MAXIMUM ENDURANCE ALTITUDE

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)
DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STARDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL





CONFIDENTIAL
D-F050-004
517-0

Figure 11-11.

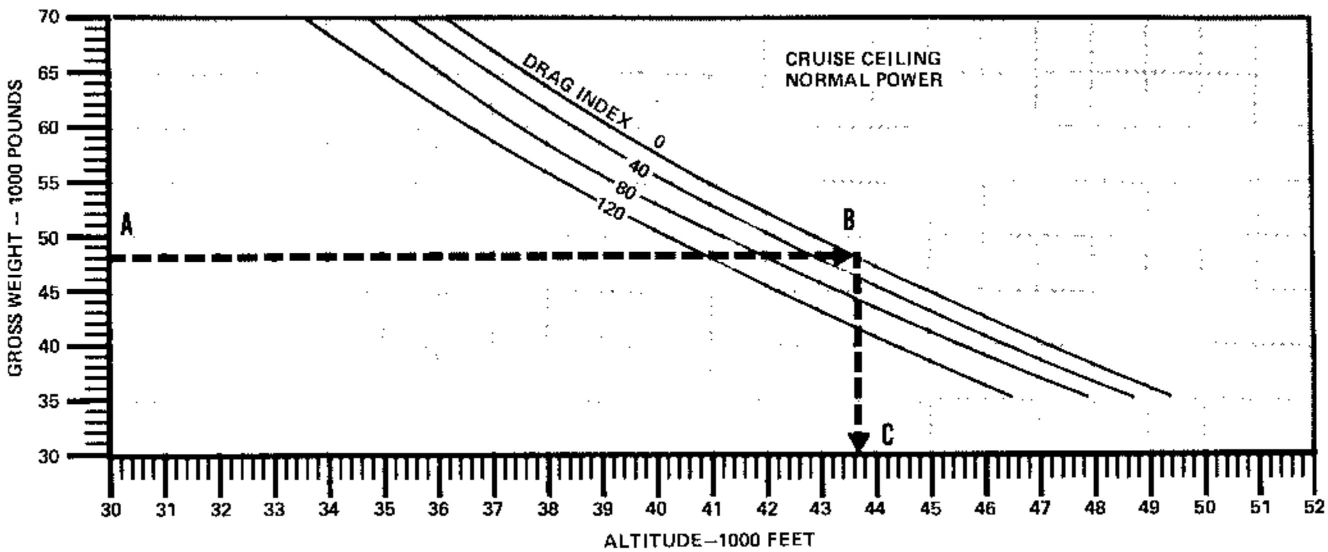
CRUISE CEILING AND SERVICE CEILING

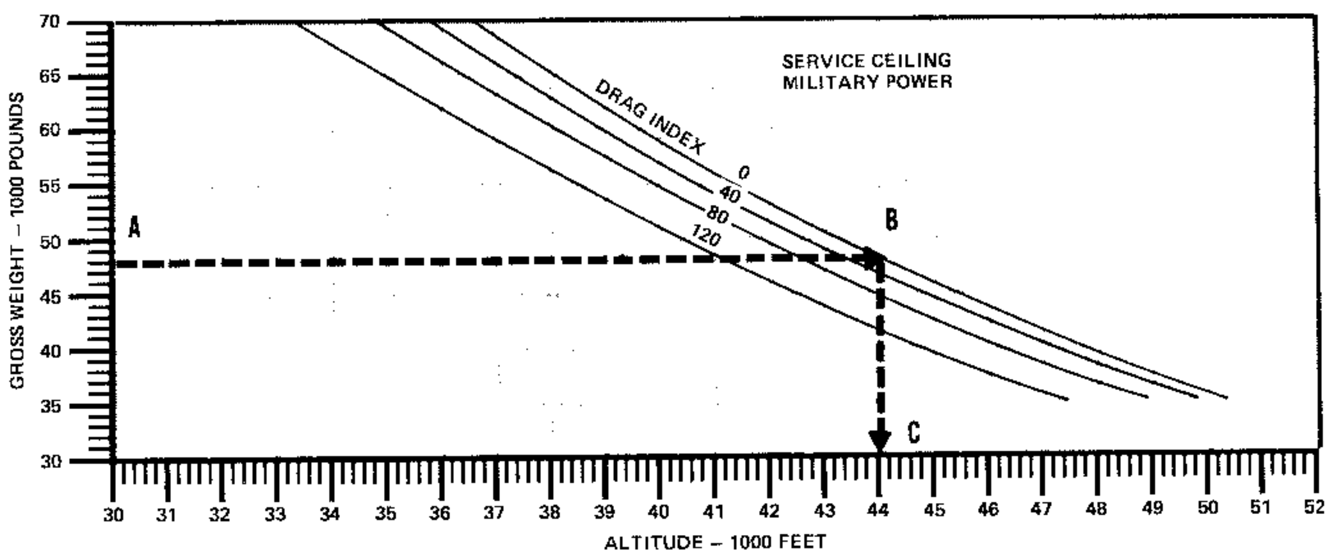
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 120)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL





CONFIDENTIAL
0-F050-004
518-0

Figure 11-12.

RATE OF CLIMB (instantaneous)

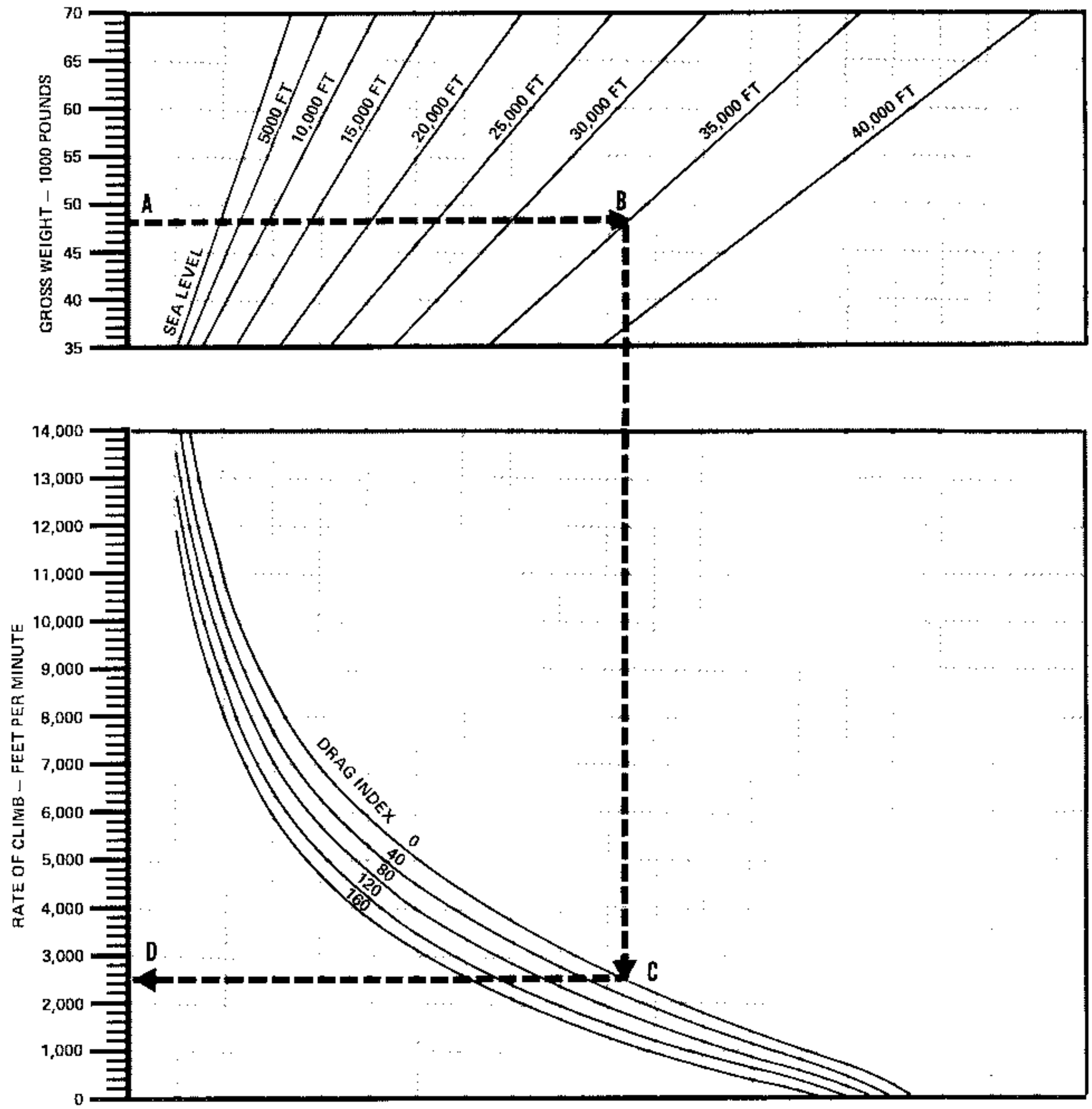
AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

MILITARY POWER

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
522-0

Figure 11-13.

part 4 Range

RANGE WIND CORRECTION

(U) Range wind correction chart, figure 11-14, provides a means of correcting specific or total range for existing wind effects. The presented range factors consider wind speeds of 25 to 150 knots (TAS) from any relative wind direction for aircraft speeds of 200 to 1,300 knots (TAS).

USE

(U) Determine the relative wind direction by subtracting the aircraft heading from the forecast wind direction. If the aircraft heading is greater than the forecast wind direction, add 360 degrees to the wind direction and then perform the subtraction. Enter the chart with relative wind direction and proceed vertically to the interpolated wind speed. From this point, project horizontally to intersect the airplane true airspeed and reflect downward to the lower scale and read the range factor. Multiply specific or total range by this range factor to find range corrected for wind effects.

(U) SAMPLE PROBLEM

Tailwind

A.	Relative wind direction	140 degrees
B.	Wind speed	50 knots
C.	Aircraft speed (TAS)	450 knots
D.	Range factor	1.082

RANGEWIND CORRECTION

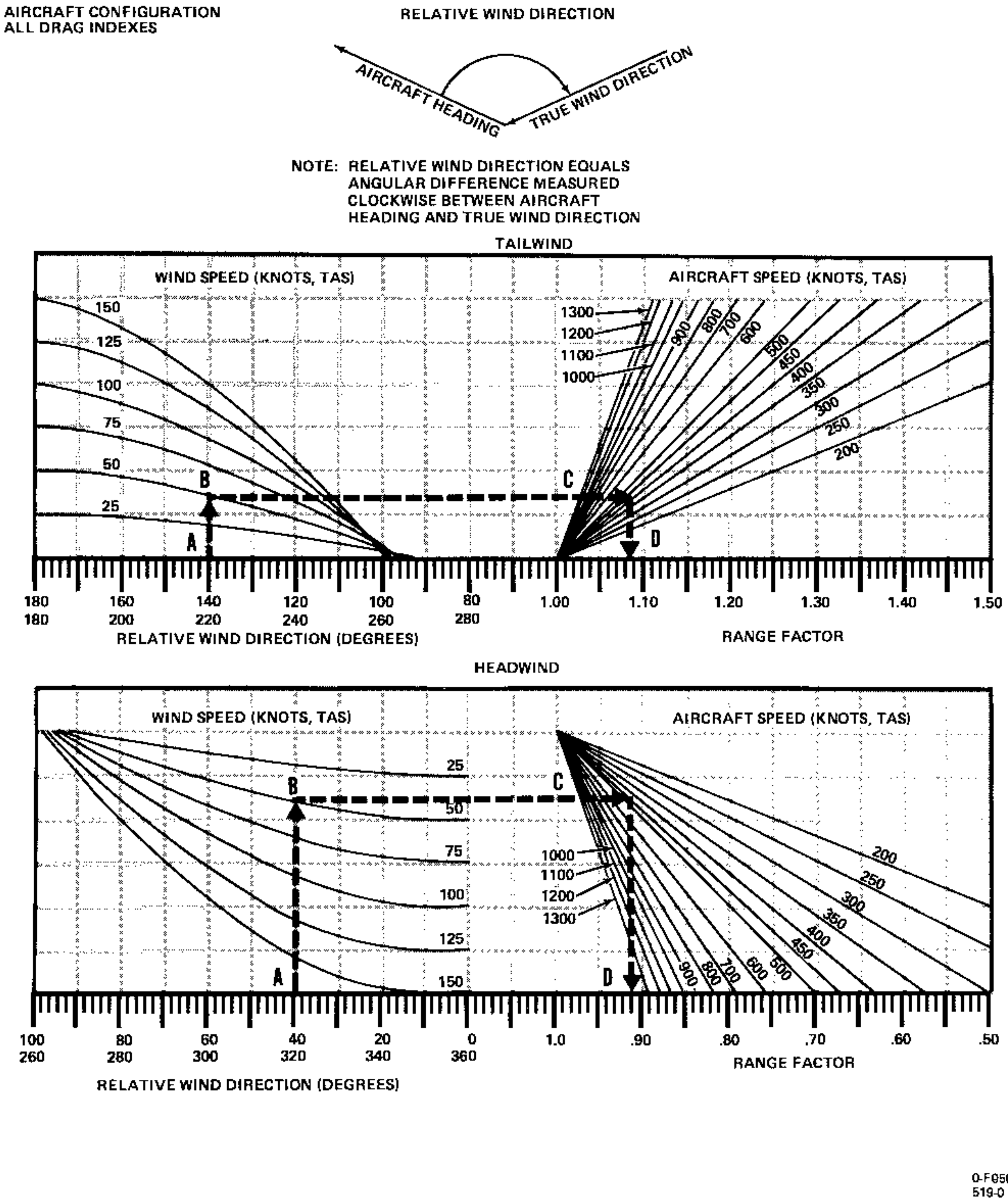


Figure 11-14

Maximum Range Cruise - Optimum Altitude

(U) These charts (figure 11-15, sheets 1 through 4) present the necessary planning data to set up optimum cruise schedules. The recommended procedure is to use an average gross weight for a given leg of the mission. One way to find the average gross weight is to divide the mission into weight segments. With this method readjust the cruise schedule each time a given amount of fuel is used. Subtract one-half of the fuel weight allotted for the first leg from the initial gross weight, the remainder is the average gross weight.

USE

(U) Enter sheet 1 with average gross weight and project horizontally to the right to intersect the reflector line, then project vertically downward to read the cruise altitude. Enter the left side of sheet 2 with the average gross weight and project horizontally to the right to intersect the computed drag index, then vertically down to obtain specific range (nautical miles per pound of fuel). Continue these projections to the right side of sheet 2 to obtain optimum cruise Mach number for the altitude obtained from sheet 1. Continue this projection vertically downward to the predicted outside air temperature and horizontally to the right to obtain the true airspeed. Trace back to the predicted outside temperature and project horizontally to the left to intersect the previously determined specific range. From this point project vertically downward to obtain the total fuel flow in pounds per hour. Enter sheet 3 with true airspeed and project horizontally to the predicted headwind or tailwind. Project vertically downward to the selected distance line and proceed horizontally to the left to obtain the time required to travel the selected distance. Enter sheet 4 with the required cruise time and project horizontally to intersect the previously determined fuel flow, then project vertically downward to obtain the fuel required.

SAMPLE PROBLEM

(C) Optimum Cruise Altitude (figure 11-15, sheet 1)

A.	Average gross weight	48,000 pounds
B.	Reflector line	intersect
C.	Optimum cruise altitude	38,500 feet

(C) Nautical Miles per Pound of Fuel, Mach Number, True Airspeed and Fuel Flow (figure 11-15, sheet 2)

A.	Average gross weight	48,000 pounds
B.	Drag index	0
C.	Specific range	.140 nautical miles per pound of fuel

D.	Drag index	0
E.	Mach number	.73
F.	Outside air temperature	-60°C
G.	True airspeed	415 knots
H.	Specific range	.140 nautical miles per pound of fuel
I.	Total fuel flow	2,965 pounds per hour

(C) TIME (figure 11-15, sheet 3)

A.	True airspeed	415 knots
B.	Headwind	0 knots
C.	Selected distance	600 nautical miles
D.	Time required	86.7 minutes

(C) FUEL REQUIRED (figure 11-15, sheet 4)

A.	Time required	86.7 minutes
B.	Total fuel flow	2,965 pounds per hour
C.	Fuel required	4,287 pounds

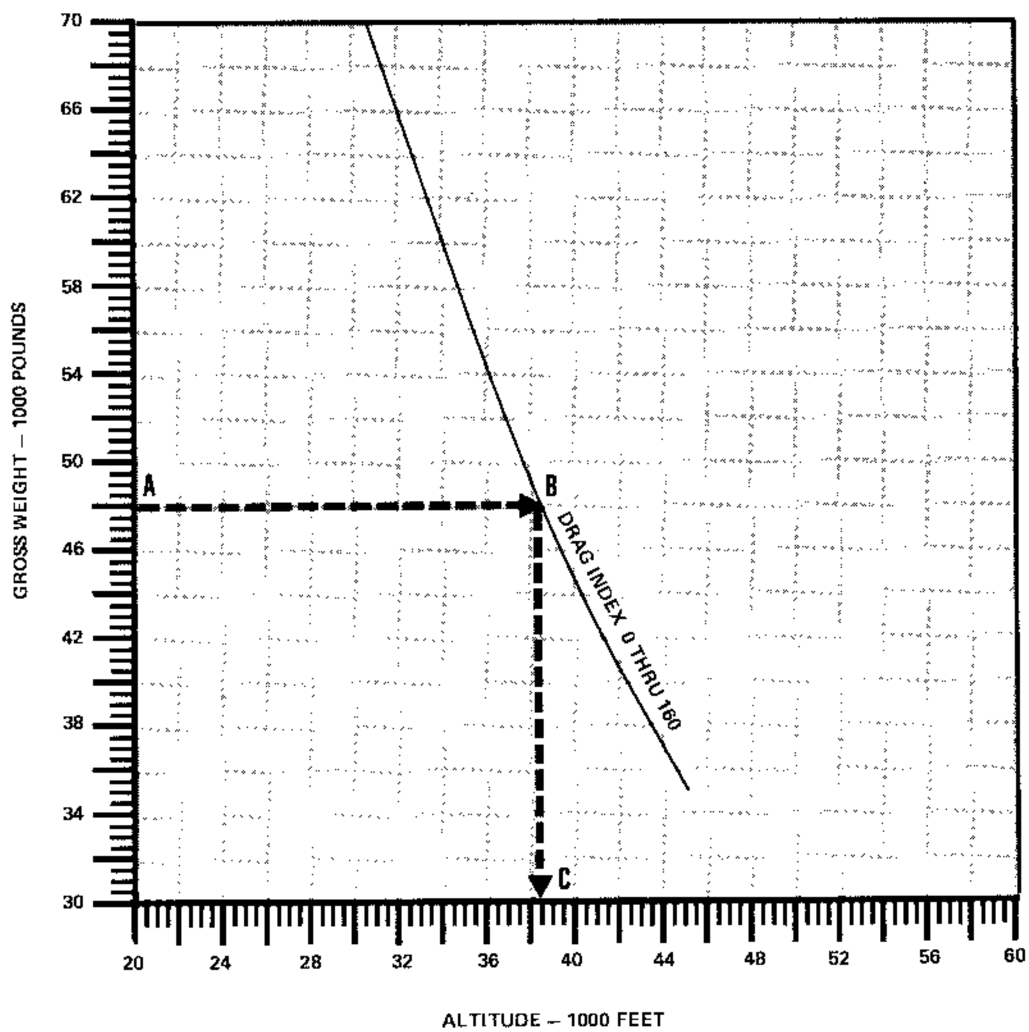
MAXIMUM RANGE CRUISE

OPTIMUM CRUISE ALTITUDE

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)
DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
520-1

Figure 11-15. (Sheet 1 of 4)

MAXIMUM RANGE CRUISE – OPTIMUM ALTITUDE

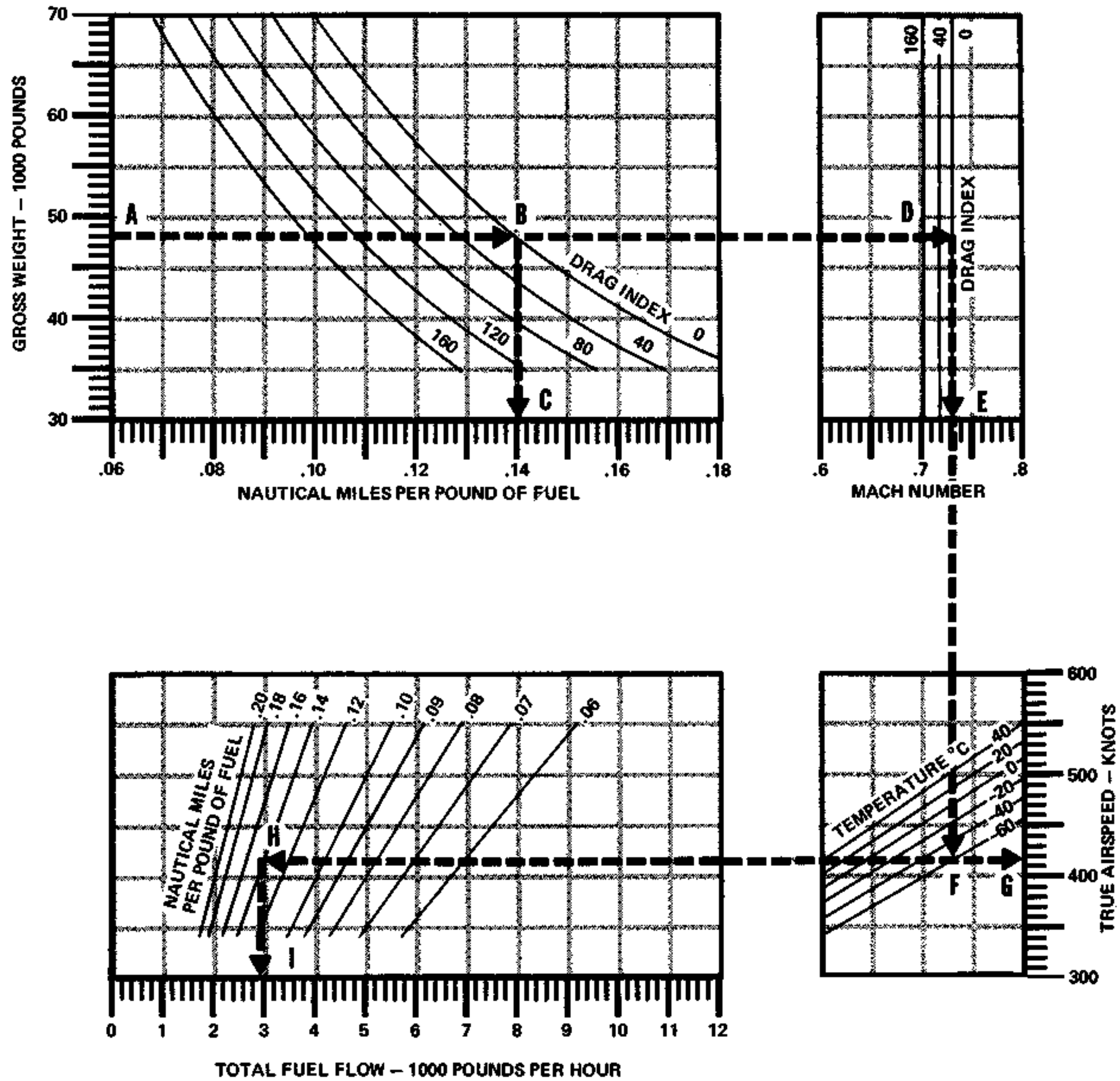
NAUTICAL MILES PER POUND OF FUEL, MACH NUMBER TRUE AIRSPEED AND FUEL FLOW

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
520-2

Figure 11-15. (Sheet 2 of 4)

MAXIMUM RANGE CRUISE – OPTIMUM ALTITUDE

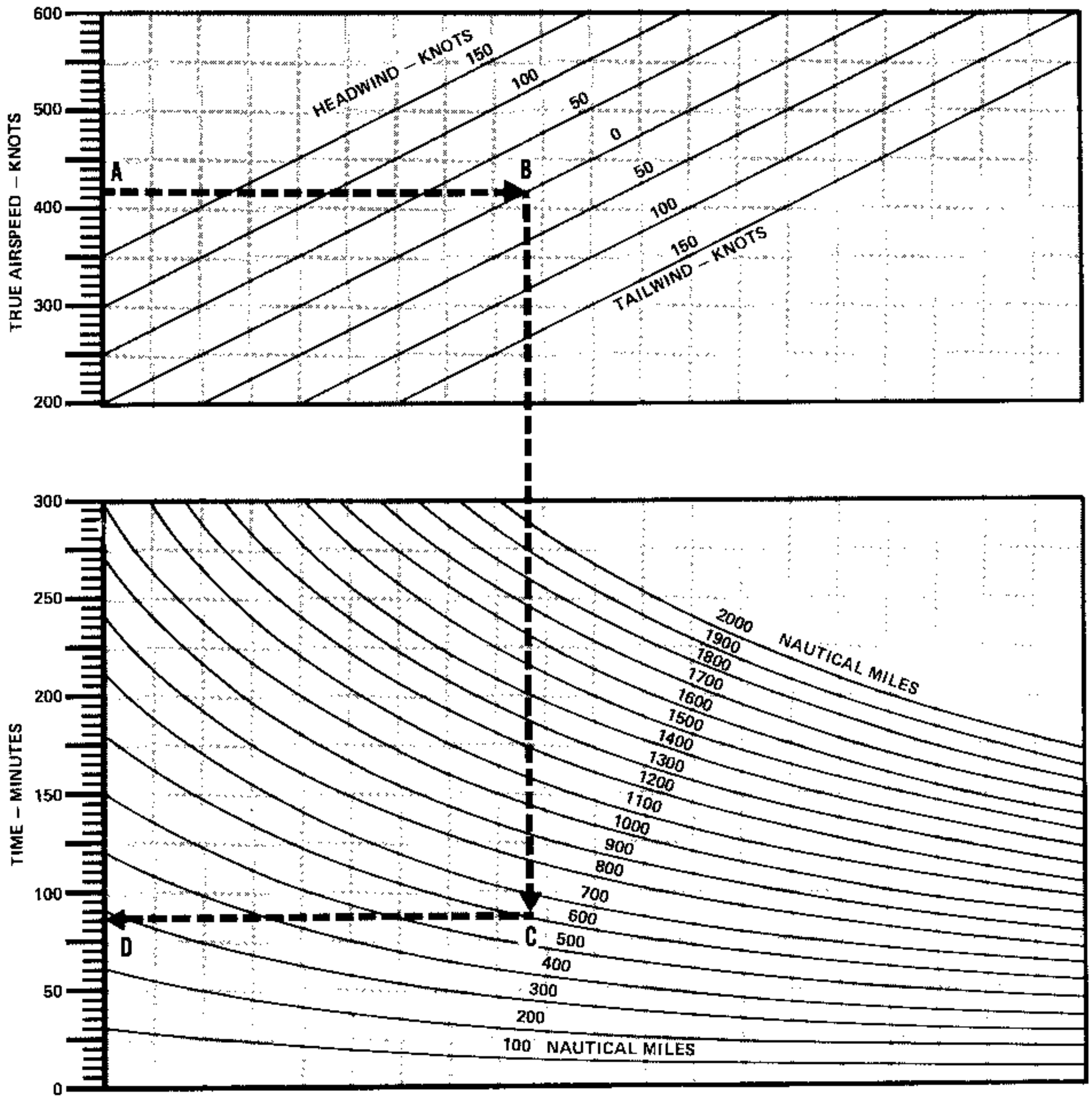
TIME REQUIRED

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
520-3

Figure 11-15. (Sheet 3 of 4)

MAXIMUM RANGE CRUISE – OPTIMUM ALTITUDE

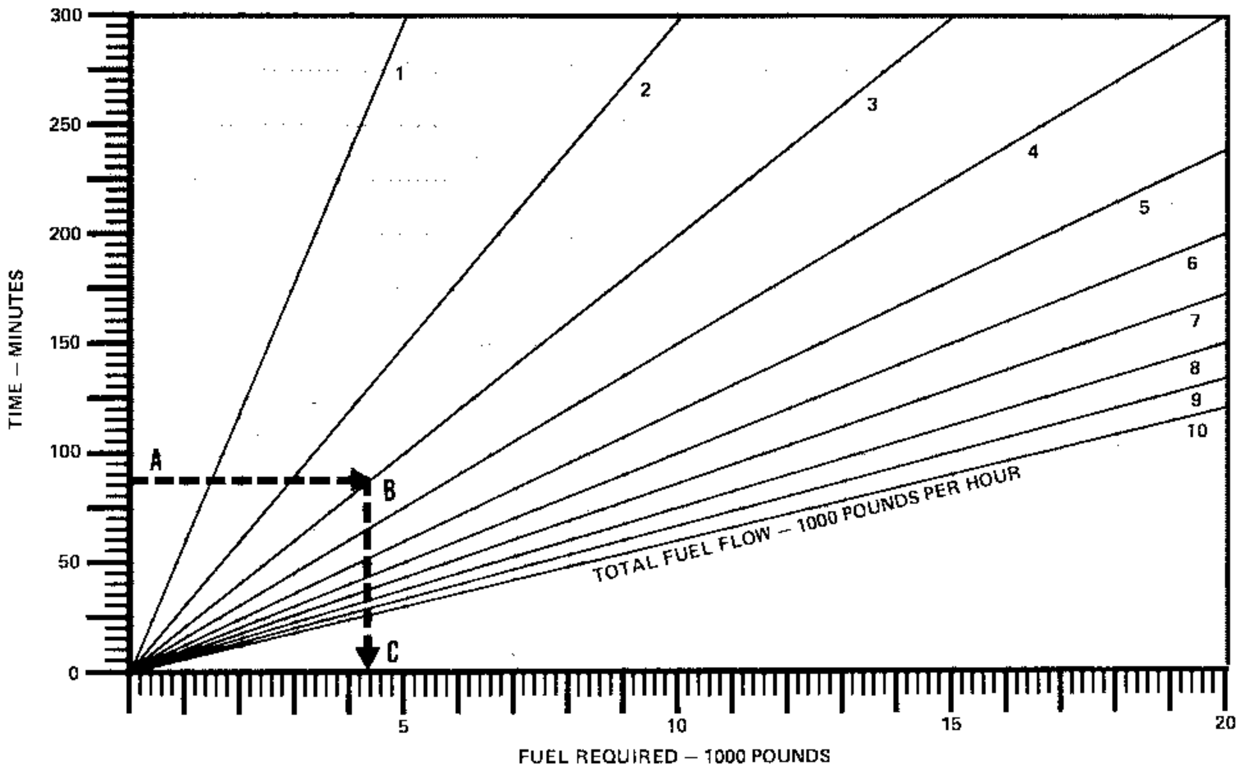
FUEL REQUIRED

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
520-4

Figure 11-15. (Sheet 4 of 4)

Maximum Range Cruise at Constant Altitude

(U) The chart (figure 11-16, sheets 1 to 4) present the necessary planning data to setup maximum cruise schedules at constant altitude. The recommended procedure is to use an average gross weight for a given leg of the mission. One way to find the average gross weight is to divide the mission into weight segments. With this method readjust the cruise schedule each time a given amount of fuel is used. Subtract one-half of the fuel allotted for the first leg from the initial gross weight, the remainder is the average gross weight for the leg.

USE

(U) Enter the left side of sheet 1 with the average gross weight. Project horizontally to the right to intersect the desired cruise altitude, vertically downward to the computed drag index, then horizontally to the left to obtain specific range (nautical miles per pound of fuel). Repeat these projections on the right side of sheet 1 to obtain optimum cruise Mach number for the desired altitude. Enter sheet 2 with the optimum cruise Mach number obtained from sheet 1. Project horizontally to the right to intersect the predicted outside air temperature, then vertically downward to obtain true airspeed. Continue vertically downward to the previously determined specific range, then horizontally to the left to obtain total fuel flow in pounds per hour. Enter sheet 3 with true airspeed and project horizontally to the predicted headwind or tailwind. Project vertically downward to the selected distance line and proceed horizontally to the left to obtain the time required to travel the selected distance. Enter sheet 4 with the required cruise time and project horizontally to intersect the previously determined fuel flow, then project vertically downward to obtain the fuel required.

SAMPLE PROBLEM

(C) Nautical Miles per Pound of Fuel and Mach Number (figure 11-16, sheet 1)

A.	Average gross weight	48,000 pounds
B.	Cruise altitude	35,000 feet
C.	Drag index	0
D.	Specific range	.1355 nautical miles per pound of fuel
E.	Cruise altitude	35,000 feet
F.	Drag index	0
G.	Mach number	0.67 M

(C) True Airspeed and Fuel Flow (figure 11-16, sheet 2)

A.	Mach number	0.67 M
B.	Outside air temperature	-60°C
C.	True airspeed	381 knots
D.	Specific range	.1355 nautical miles per pound of fuel
E.	Total fuel flow	2,812 pounds per hour

(C) Time (figure 11-16, sheet 3)

A.	True airspeed	381 knots
B.	Headwind	0 knots
C.	Selected distance	600 nautical miles
D.	Time required	94.5 minutes

(C) Fuel Required (figure 11-16, sheet 4)

A.	Time required	94.5 minutes
B.	Total fuel flow	2,812 pounds per hour
C.	Fuel required	4,428 pounds

MAXIMUM RANGE CRUISE AT CONSTANT ALTITUDE

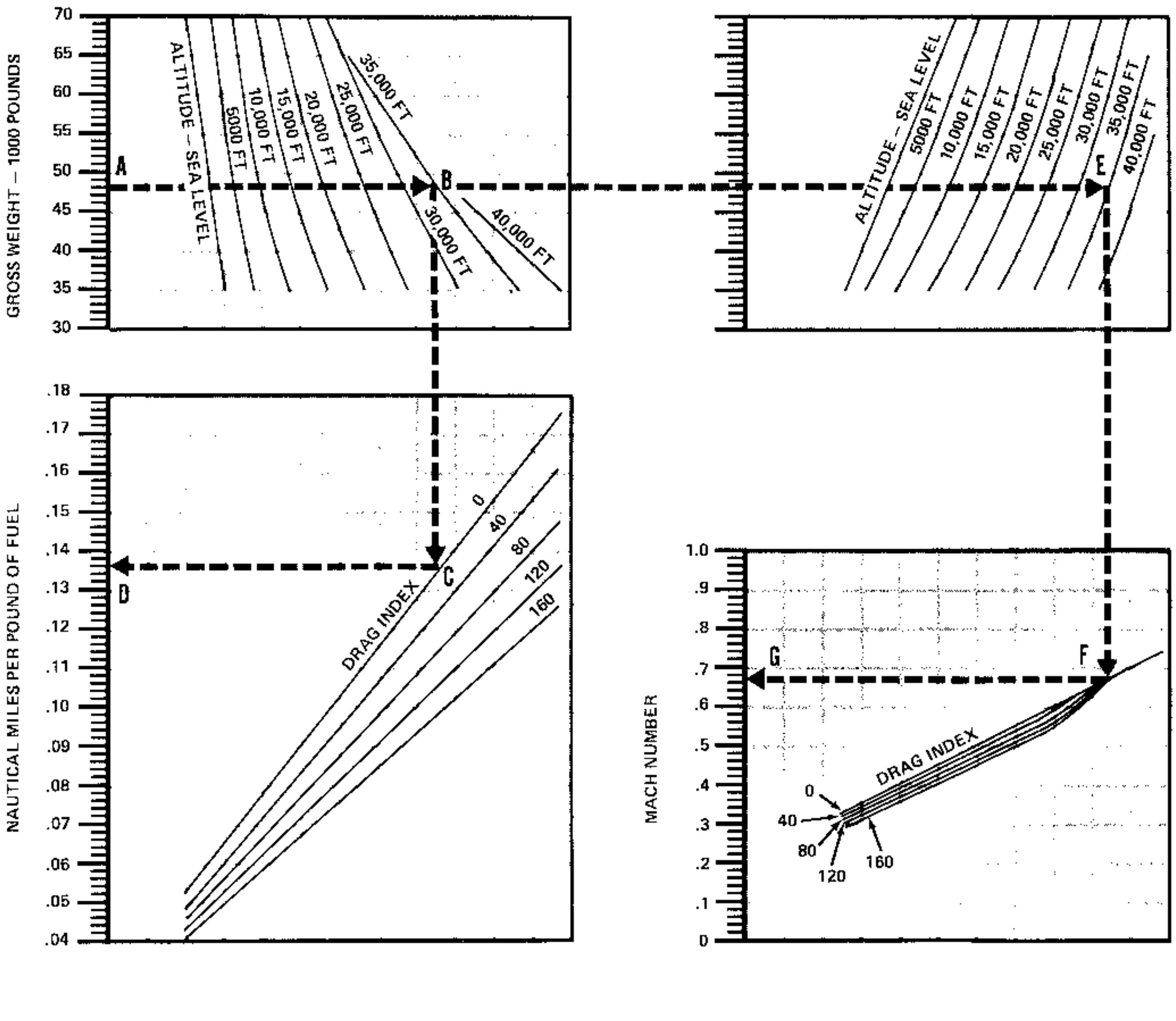
NAUTICAL MILES PER POUND OF FUEL AND MACH NUMBER

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050 004
523-1

Figure 11-16. (Sheet 1 of 4)

MAXIMUM RANGE CRUISE AT CONSTANT ALTITUDE

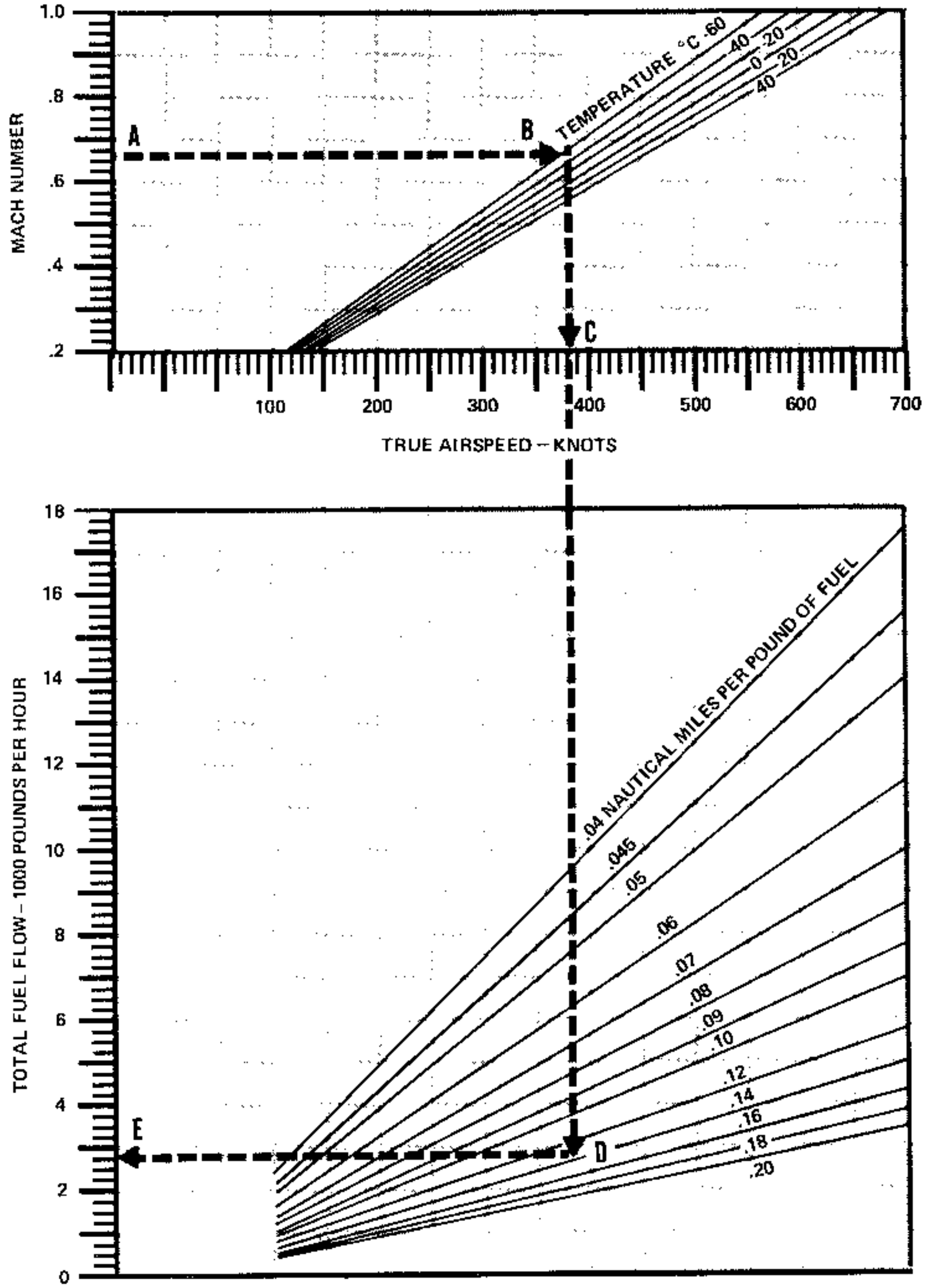
TAS AND FUEL FLOW

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDICES

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL.



CONFIDENTIAL
0-F050-004
523-2

Figure 11-16. (Sheet 2 of 4)

MAXIMUM RANGE CRUISE AT CONSTANT ALTITUDE

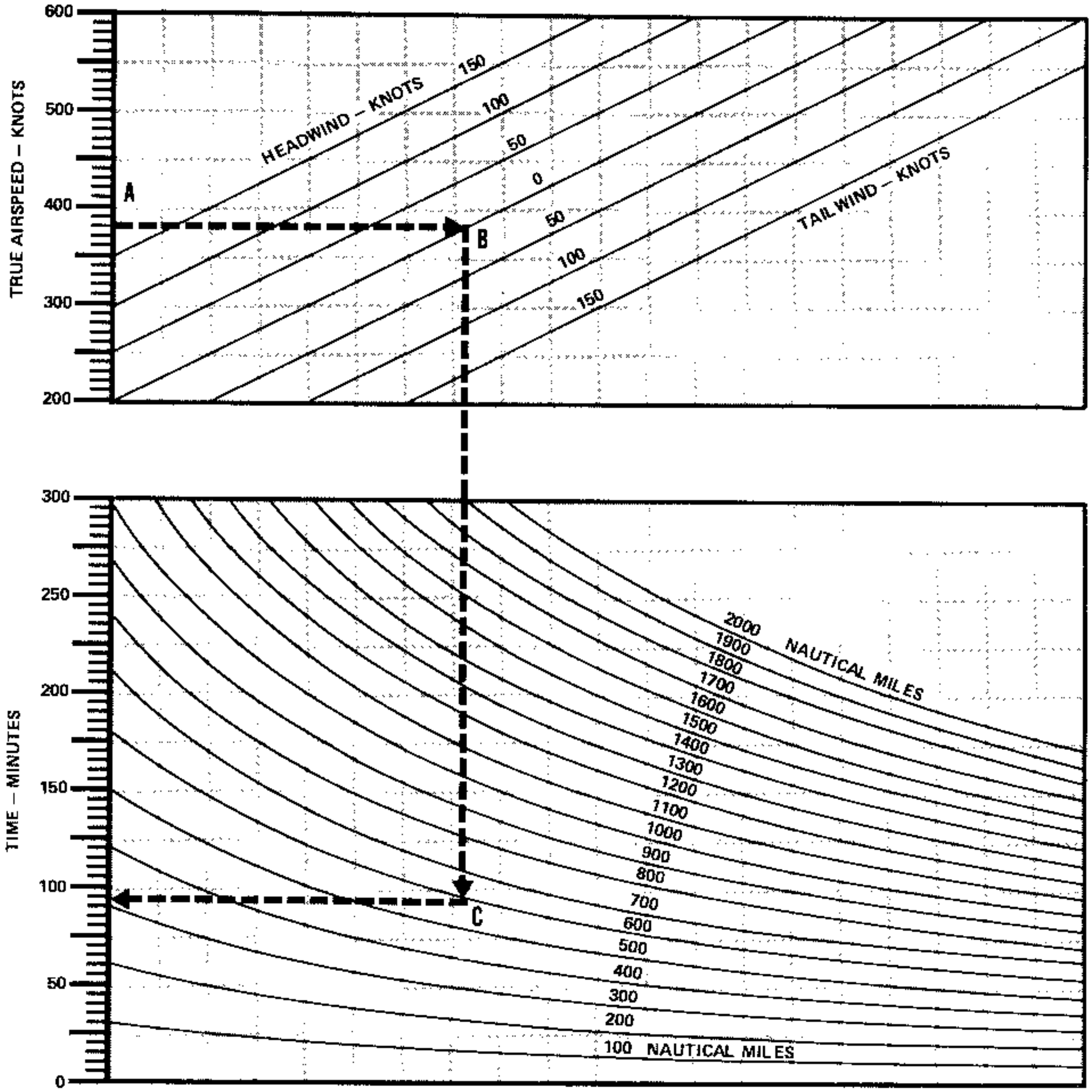
TIME REQUIRED

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
523-3

Figure 11-16. (Sheet 3 of 4)

MAXIMUM RANGE CRUISE AT CONSTANT ALTITUDE

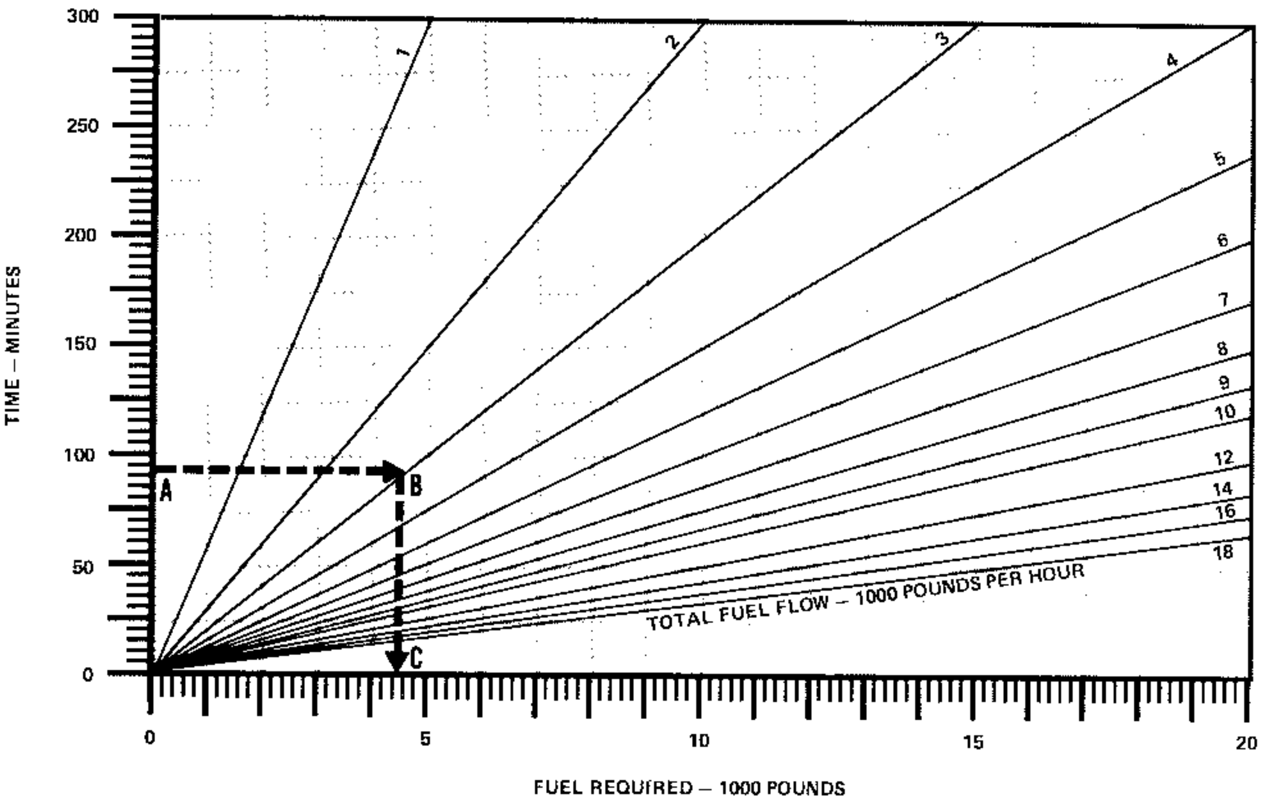
FUEL REQUIRED

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 NOVEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
523-4

Figure 11-16. (Sheet 4 of 4)

part 5 Endurance

MAXIMUM ENDURANCE ALTITUDE, MACH NUMBER AND FUEL FLOW

(U) The charts (figures 11-17, sheets 1 thru 3) present maximum endurance altitude, Mach number and fuel flow for all combinations of gross weight, altitude and drag index.

USE

(U) Enter the altitude chart with the average gross weight and proceed horizontally to the right and intersect the computed drag index. Reflect downward and read the maximum endurance altitude. Enter the Mach number chart with the average gross weight, and proceed horizontally to intersect the maximum endurance altitude. Then descend downward and intersect the computed drag index and horizontally to the left to read Mach number. To obtain calibrated airspeed proceed horizontally to the left from the previous intersection to the point of intersection with the pressure altitude. Then read the interpolated value of calibrated airspeed. Enter the fuel flow chart with the average gross weight and proceed horizontally to intersect the maximum endurance altitude. Reflect downward to the computed drag index, and then horizontally to the left to read total fuel flow.

SAMPLE PROBLEM

(C) Maximum Endurance Altitude (figure 11-17, sheet 1)

A.	Average gross weight	48,000 pounds
B.	Drag index	0
C.	Maximum endurance altitude	33,900 feet

(C) Mach Number and Airspeed (figure 11-17, sheet 2)

A.	Average gross weight	48,000 pounds
B.	Endurance altitude	33,000 feet

C. Drag index	0
D. Mach number	.58
E. Endurance altitude	33,900 feet
F. Calibrated airspeed	197 knots

(C) Fuel Flow (figure 11-17, sheet 3)

A. Average gross weight	48,000 pounds
B. Endurance altitude	33,900 feet
C. Drag index	0
D. Total fuel flow	2,820 pounds per hour

MAXIMUM ENDURANCE

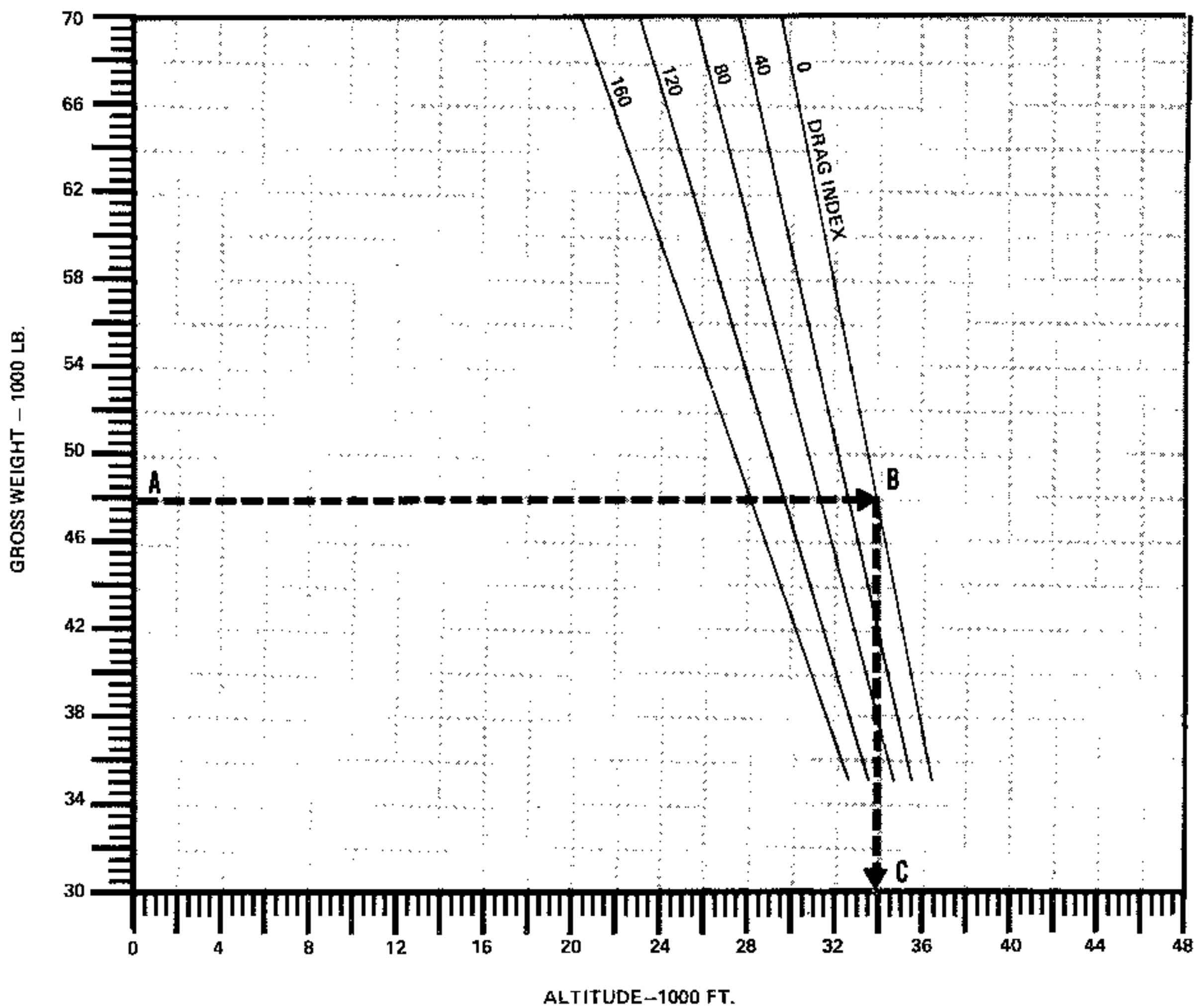
ALTITUDE

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
521-1

Figure 11-17. (Sheet 1 of 3)

MAXIMUM ENDURANCE

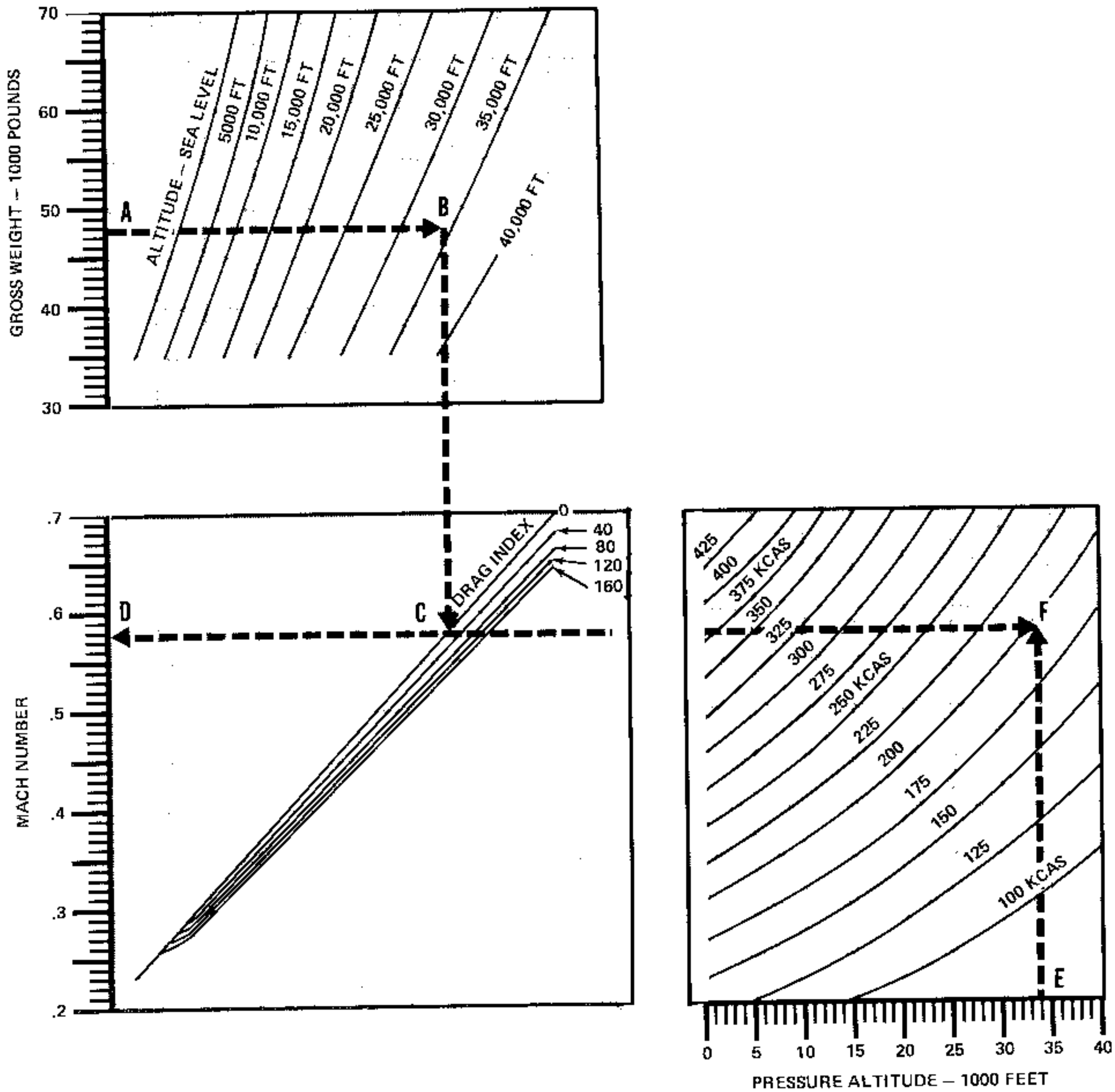
MACH NUMBER AND AIRSPEED

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
521-2

Figure 11-17. (Sheet 2 of 3)

MAXIMUM ENDURANCE

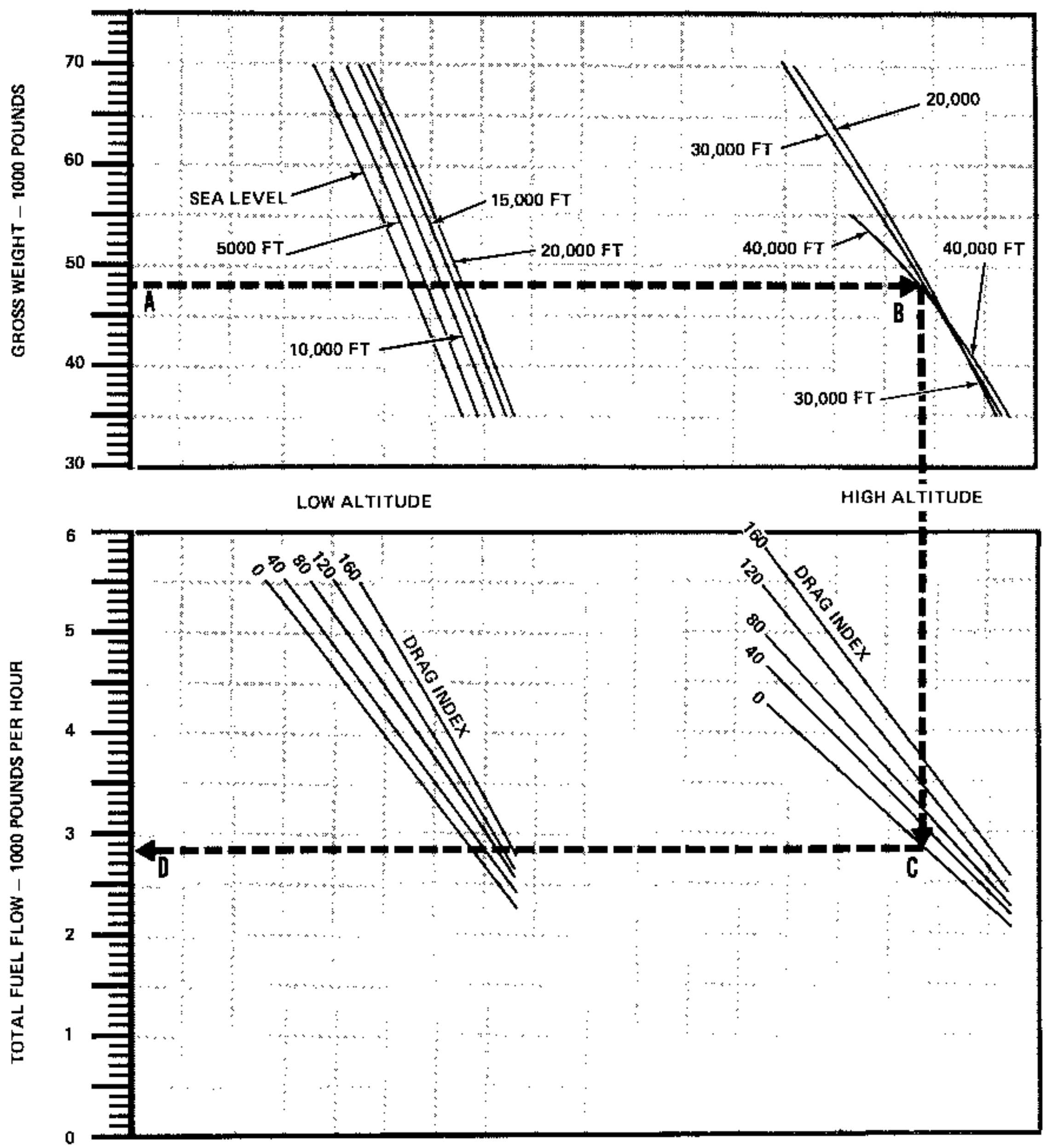
FUEL FLOW

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 160)

DATE: 1 SEPTEMBER 1971
DATA BASIS: ESTIMAGED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
521-3

Figure 11-17. (Sheet 3 of 3)

part 6 Inflight Refueling

Charts for this part will be supplied at a later date.

part 7 Descent

Charts for this part will be supplied at a later date.

This page intentionally left blank.

part 8 Landing

LANDING SPEED

(U) This chart (figure 11-18) contains recommended approach, touch down and stall speed curves for wing sweep angle of 20 degrees as a function of gross weight.

USE

(U) Enter the chart at the estimated gross weight and project vertically to intersect the various speed curves (i.e. stall, touch down, and approach speed). From the intersection of these curves, project horizontally to the left scale to read recommended speeds.

SAMPLE PROBLEMS

(C) A. Estimated landing gross weight	42,000 pounds
B. Speed reflector lines	intersect
C. Stall speed	98 KCAS
D. Touchdown speed	106 KCAS
E. Approach speed	118 KCAS

LANDING SPEEDS

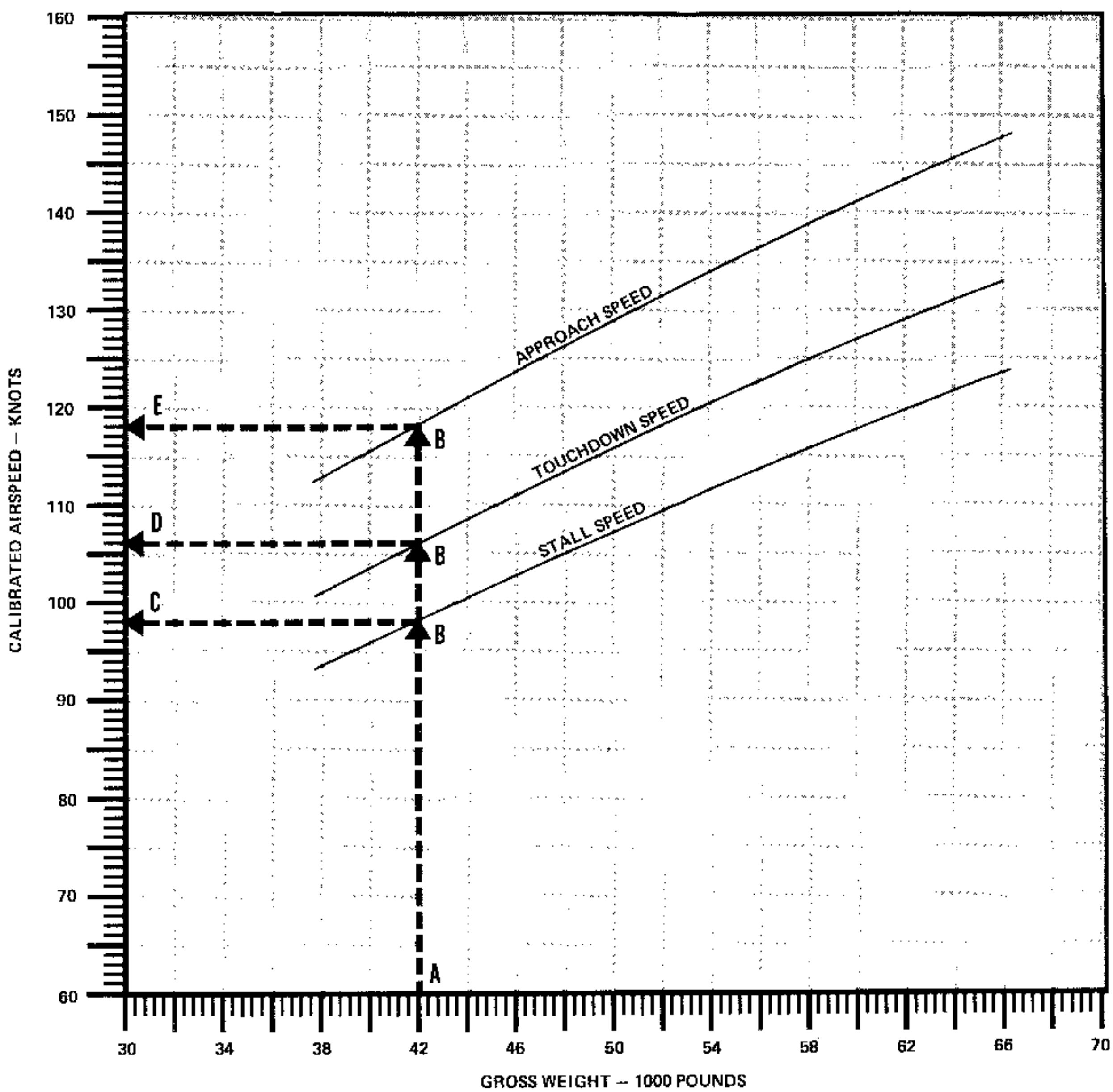
WING SWEEP $\Lambda_{LE} = 20°$

AIRCRAFT CONFIGURATION
ALL DRAG INDEXES
35° FLAP
GEAR DOWN

DATE: 1 NOVEMBER, 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD ATMOSPHERE

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
508-0

Figure 11-18.

LANDING DISTANCE

(U) Landing ground roll distance with deceleration devices for 20 degrees wing sweep is shown in figure 11-19. Corrections are made for ambient temperature and gross weight. Incremental distances are listed for effects of obstacle height (50 feet) and lack of deceleration devices. The deceleration devices included are spoilers and speed brakes with deflections respectively 55 degrees and 60 degrees.

USE

(U) Enter the chart with the temperature and project vertically upward to the sea level guide line. From this point, proceed horizontally to the right to the landing gross weight. Project vertically downward and read ground roll. If the landing is to be made over a 50 foot obstacle add 954 feet to the ground roll distance. For a landing without deceleration devices add 520 feet.

SAMPLE PROBLEM

(C) A. Temperature	15°C
B. Sea level guideline	intersect
C. Gross weight	42,000
D. Landing roll distance	2,300 feet

LANDING DISTANCE

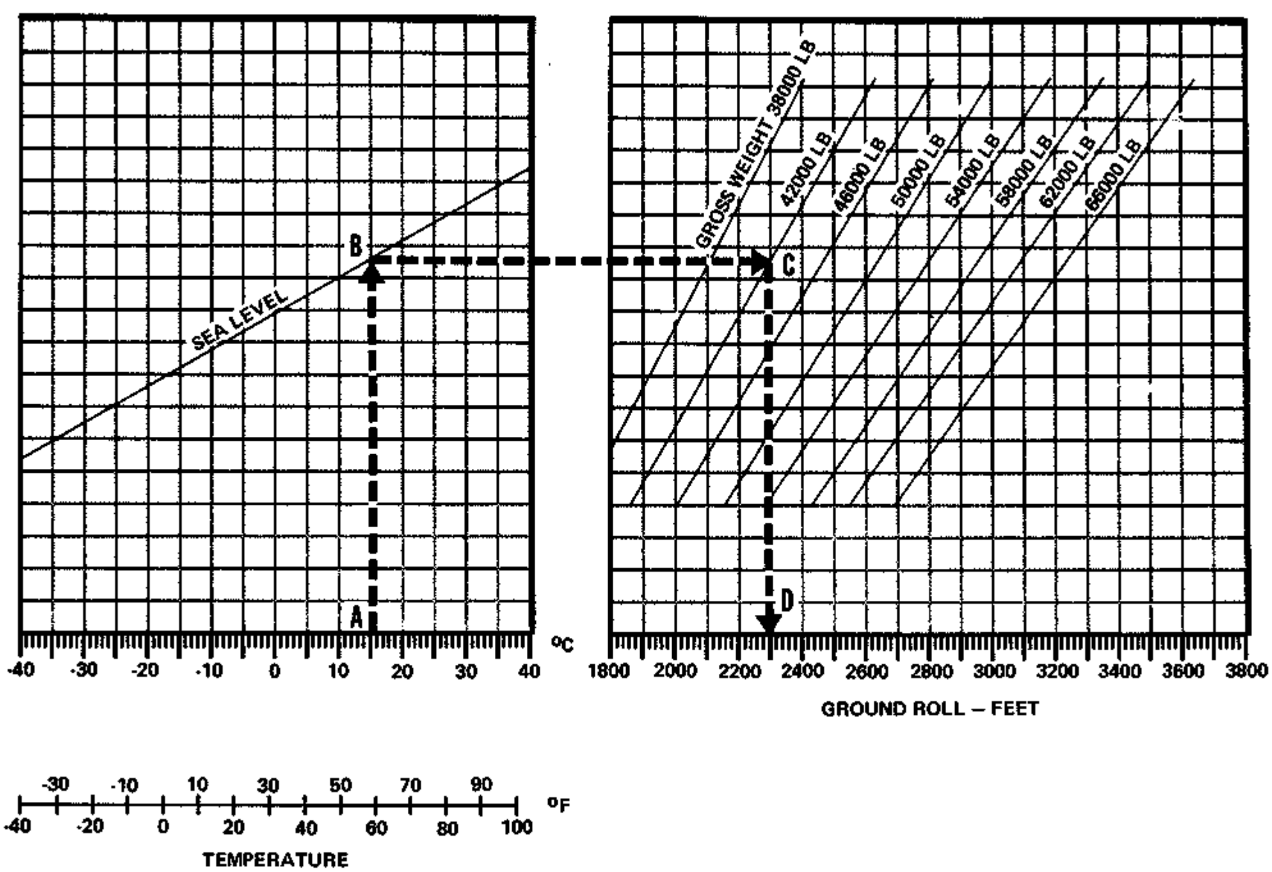
WING SWEEP $\Lambda_{LE} = 20°$

AIRCRAFT CONFIGURATION
ALL DRAG INDEXES
WITH DECELERATION DEVICES
35° FLAP
GEAR DOWN

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
507-0

Figure 11-19.

part 9 Combat Performance

Charts for this part will be supplied at a later date.

part 10 Emergency Operation

CRITICAL ENGINE FAILURE SPEED AND CRITICAL FIELD LENGTH

(U) These charts (figures 11-20, sheets 1 and 2) present the critical engine failure speed and corresponding critical fieldlength as a function of ambient temperature and gross weight for the two primary store configurations. The critical field length is presented as the takeoff distance to the obstacle from the start of the takeoff roll. This distance is always equal to or greater than the accelerate-stop distance. The takeoff performance presented for critical field length required can be achieved by applying takeoff power prior to brake release, releasing the brakes, and accelerating to the speed at which the engine fails (at which time a 1.5 second delay is assumed for pilot recognition). At this point the aircraft can be either:

- Accelerated to takeoff speed and climbed to the obstacle

or

- Stopped by immediately retarding the throttles, applying speed brakes, and extending spoilers.

USE

(U) Enter the chart at the applicable temperature and project vertically to intersect the sea level guideline. Then proceed horizontally to the right and intersect the desired gross weight curve. From the intersection of the gross weight curve, project vertically downward to ground roll distance. From this point, continue downward to the obstacle height guide line and project horizontally to the left to read the total distance over a 50 foot obstacle. To obtain critical engine failure speed continue horizontally to the left and at the appropriate gross weight project vertically downward to read the speed.

The critical engine failure speed (V_1) variation with temperature and gross weight is limited by the rotation speed (V_R); ie, $V_1 < V_R$. When $V_1 > V_R$, rotation speed and its corresponding distance over obstacle is used and a balanced field length no longer exists.

SAMPLE PROBLEM (figure 11-20, sheet 1)

(C) Configuration: Four (4) Sparrow

A. Temperature	15°C
B. Sea level guide line	intersect

C.	Gross weight	58,000 pounds
D.	Ground roll distance	2,800 feet
E.	Obstacle height guide line	intersect
F.	Total distance to clear 50 foot obstacle	4750 feet
G.	Gross weight	58,000 pounds
H.	Critical engine failure speed	121 KCAS

(C) Configuration: Six (6) Phoenix Plus Two (2) Drop Tanks (figure 11-20, sheet 2)

A.	Temperature	15°C
B.	Sea level guide line	intersect
C.	Gross weight	66,000 pounds
D.	Ground roll distance	3,875 feet
E.	Obstacle height guide line	intersect
F.	Total distance to clear 50 foot obstacle	7,700 feet
G.	Gross weight	66,000 pounds
H.	Critical engine failure speed	137 KCAS

CRITICAL ENGINE FAILURE SPEED AND CRITICAL FIELD LENGTH (single engine)

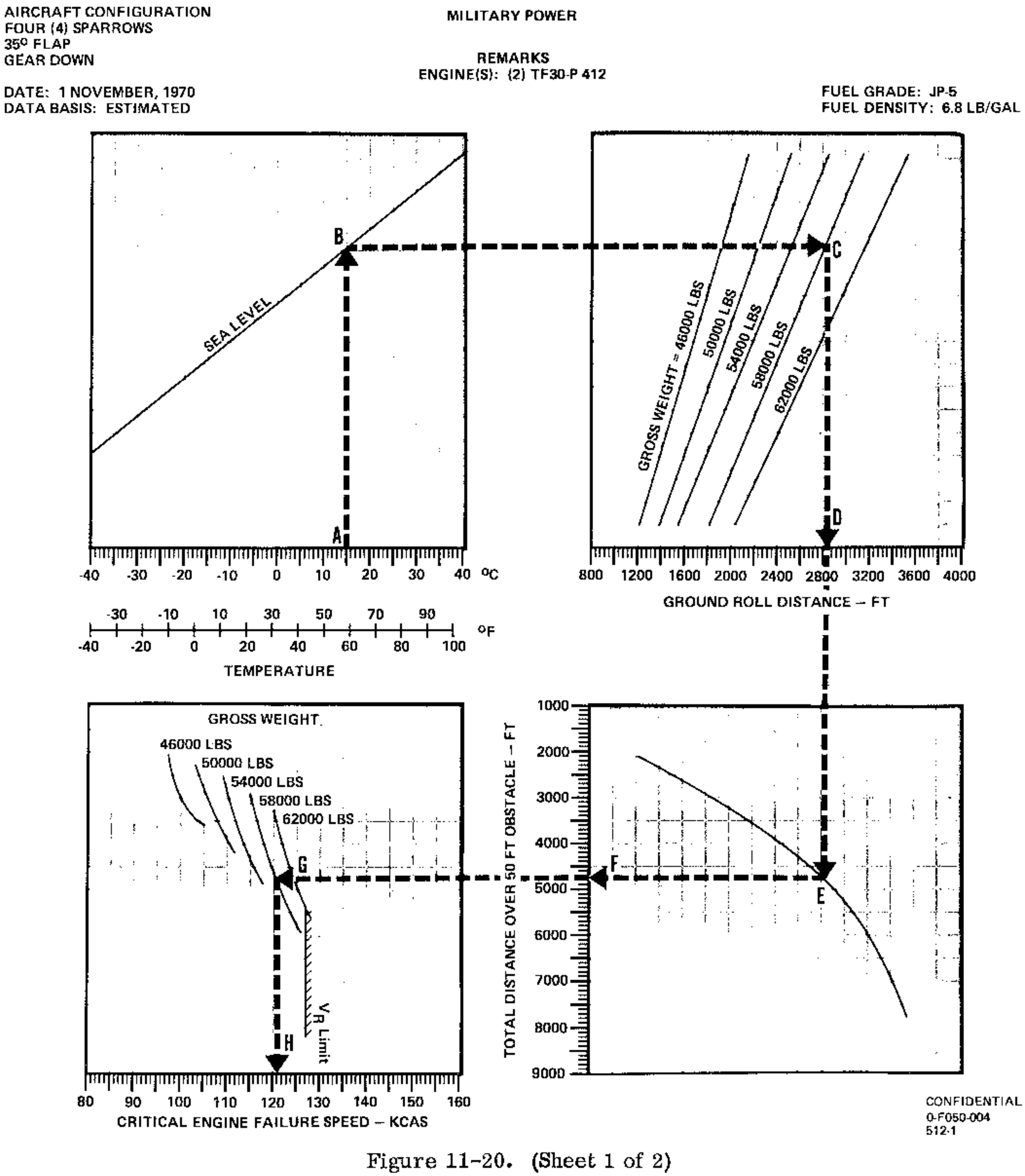


Figure 11-20. (Sheet 1 of 2)

CRITICAL ENGINE FAILURE SPEED AND CRITICAL FIELD LENGTH (single engine)

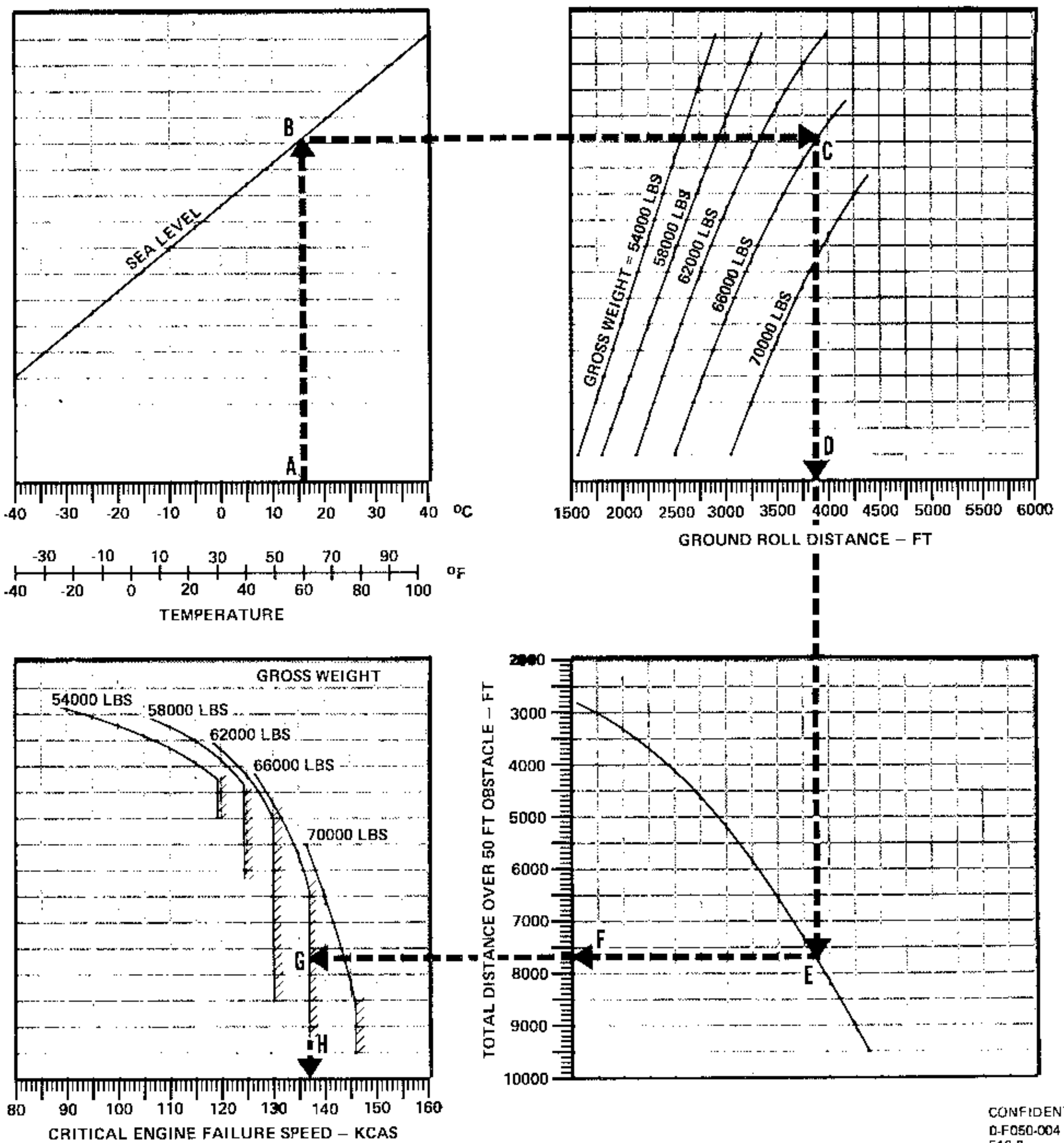
AIRCRAFT CONFIGURATION
SIX (6) PHOENIX + TWO (2) 267 GAL. DROP TANKS
35° FLAPS
GEAR DOWN

MILITARY POWER

REMARKS
ENGINE(S): (2) TF30-P-412

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
512-2

Figure 11-20. (Sheet 2 of 2)

CLIMB PERFORMANCE AFTER TAKEOFF (SINGLE ENGINE)

(U) This chart (figure 11-21) presents single engine climb performance after takeoff at sea level. Rate of climb and optimum climb speed are given as a function of gross weight and ambient temperature.

USE

(U) Enter the chart at the aircraft gross weight and proceed horizontally to intersect the reflector line. Descend vertically to intersect the temperature scale, and proceed horizontally to read the rate of climb. Re-enter the chart at the aircraft gross weight, and proceed horizontally to intersect the climb speed line. From this point, project vertically upward and read optimum climb speed.

(C) SAMPLE PROBLEM

A. Gross weight	58,000 pounds
B. Reflector line	intersect
C. Temperature	15°C
D. Rate of climb	400 feet per minute
E. Gross weight	58,000 pounds
F. Climb speed guide line	intersect
G. Optimum speed	140 KCAS

CLIMB PERFORMANCE AFTER TAKEOFF (single engine)

MILITARY POWER – SEA LEVEL

AIRCRAFT CONFIGURATION
FOUR (4) SPARROWS
35° FLAP, GEAR DOWN

DATE: 1 NOVEMBER, 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

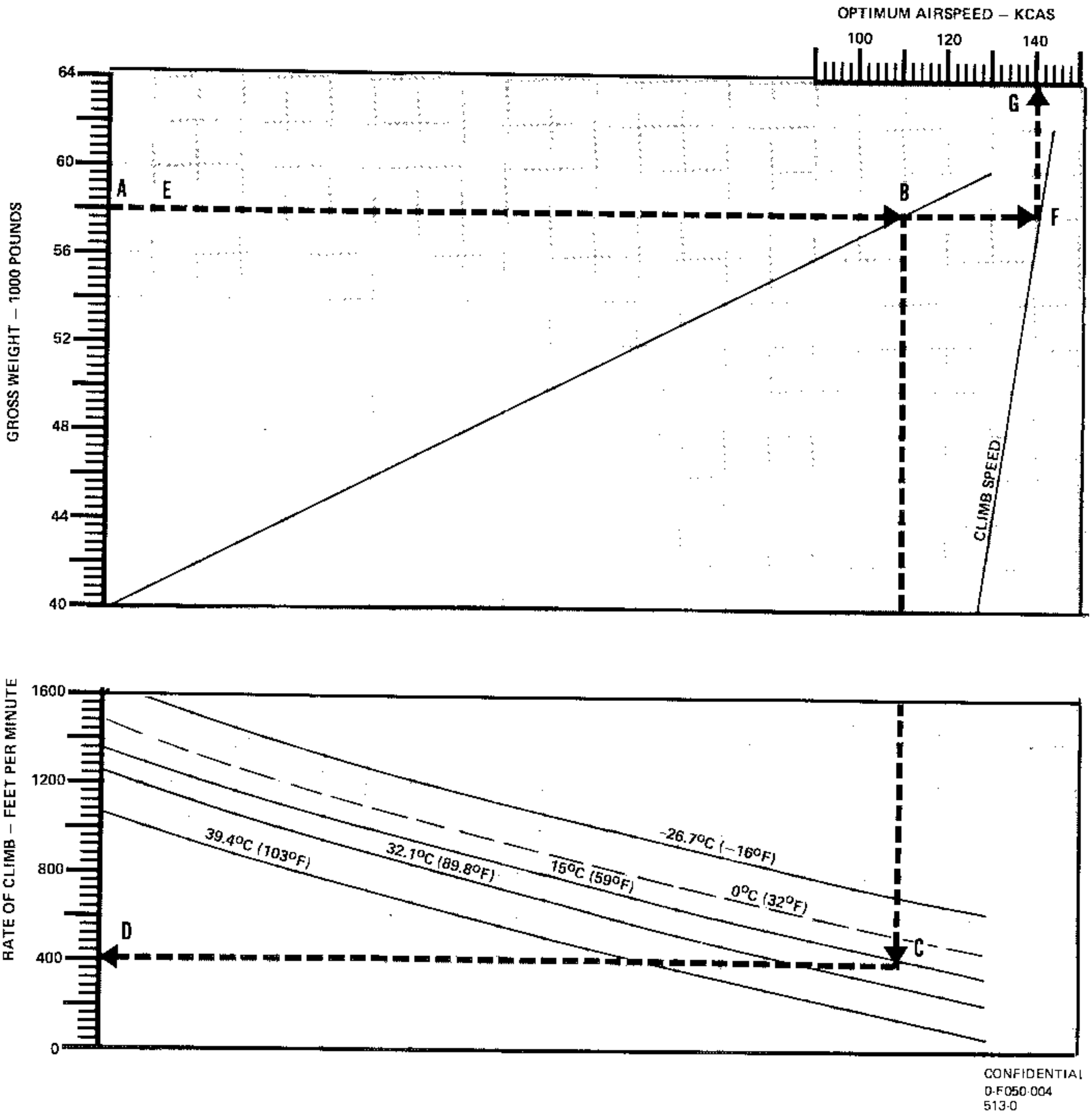


Figure 11-21.

MAXIMUM ENDURANCE (SINGLE ENGINE)

(U) These charts (figures 11-22 and 11-23) present optimum endurance altitude and maximum endurance specifics (pounds of fuel per hour and Mach number) for all combinations of gross weight, altitude and drag index.

USE

(U) Enter the maximum endurance altitude chart (figure 11-22) at the gross weight and proceed horizontally to the right and intersect the computed drag index. Reflect downward and read the optimum endurance altitude. Enter the endurance speed chart (figure 11-23) with the gross weight, and proceed horizontally to intersect the optimum endurance altitude. Then descend downward and intersect the computed drag index and horizontally to read true Mach number. A further plot to read CAS is also available. Enter the fuel flow chart (figure 11-24) with the gross weight proceed horizontally to intersect the optimum endurance altitude. Reflect downward to the computed drag index, and then horizontally to read total fuel flow.

SAMPLE PROBLEM

(C) ENDURANCE ALTITUDE (figure 11-22)

A.	Gross weight	50,000 pounds
B.	Drag index (clean)	0
C.	Optimum endurance altitude	17,400 feet

(C) MAXIMUM ENDURANCE SPEED (figure 11-23)

A.	Gross weight	50,000 pounds
B.	Optimum endurance altitude	17,400 feet
C.	Drag index (clean)	0
D.	True Mach number	.43 M_t
E.	Altitude	17,500 feet
F.	Airspeed	206 KCAS

(C) MAXIMUM ENDURANCE FUEL FLOW (figure 11-24)

A.	Gross weight	50,000 pounds
B.	Optimum endurance altitude	17,500 feet
C.	Drag index (clean)	0
D.	Endurance	2,850 pounds per hour

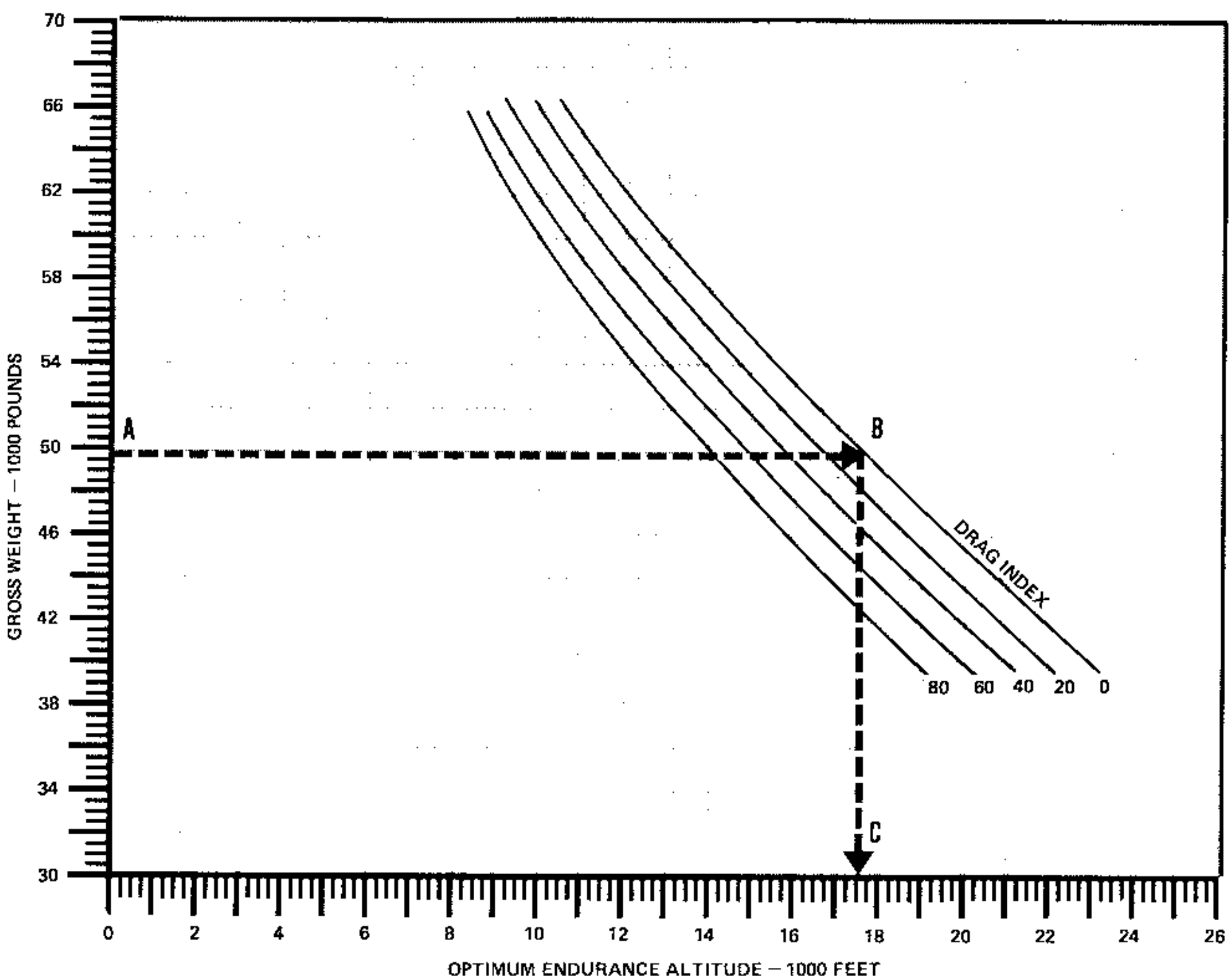
MAXIMUM ENDURANCE ALTITUDE (single engine)

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 80)

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD ATMOSPHERE
INOPERATIVE ENGINE WINDMILLING

DATE: 1 NOVEMBER, 1970
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 68 LB/GAL



CONFIDENTIAL
0-F050-004
506-0

Figure 11-22.

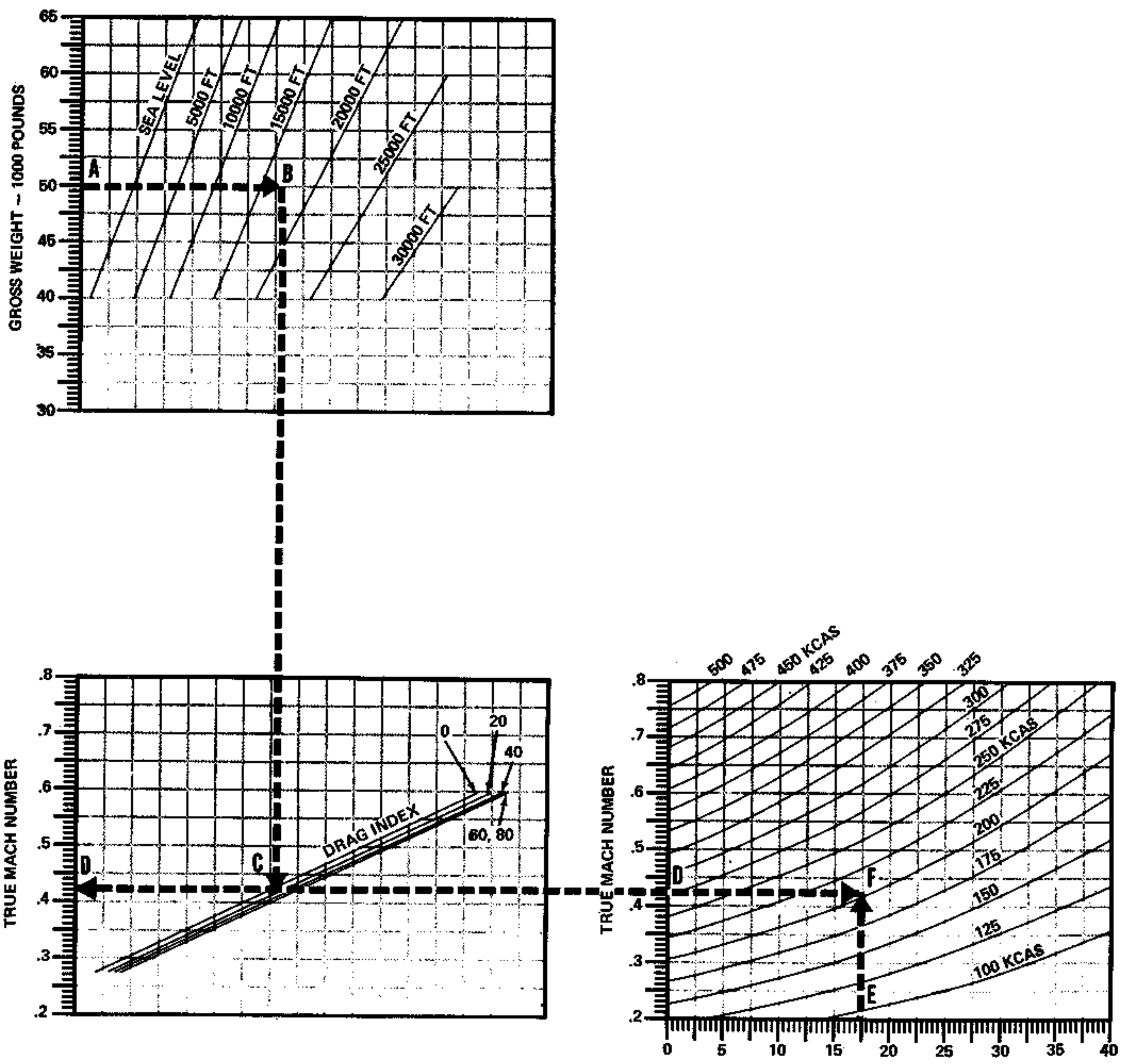
MAXIMUM ENDURANCE SPEED (single engine)

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 80)

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD ATMOSPHERE
INOPERATIVE ENGINE WINDMILLING

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050-004
503-0

Figure 11-23.

MAXIMUM ENDURANCE FUEL FLOW (single engine)

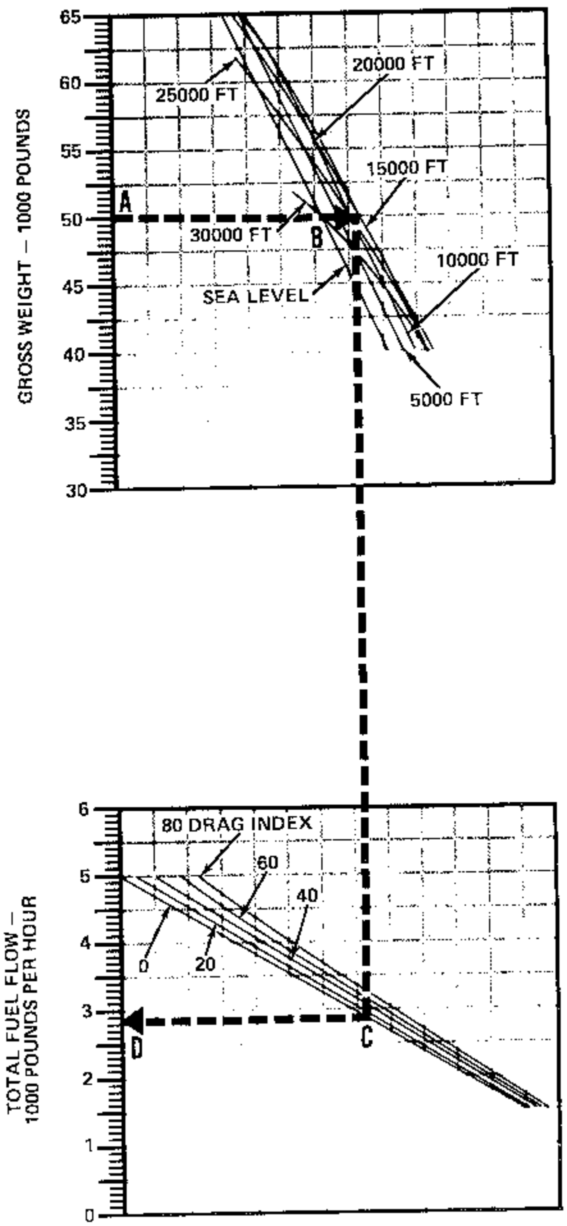
POUNDS OF FUEL PER HOUR

AIRCRAFT CONFIGURATION
INDIVIDUAL DRAG INDEXES
(0 TO 80)

DATE: 1 NOVEMBER 1970
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412
ICAO STANDARD ATMOSPHERE
INOPERATIVE ENGINE WINDMILLING

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



CONFIDENTIAL
0-F050 004
514-0

Figure 11-24.

AIRSTART ENVELOPE

(U) The airstart envelope (figure 11-25) is based on a combination of flight test results, engine test results, and analytical estimates. These data are for steady state flight conditions.

(U) In most circumstances, the automatic restart system will preclude the possibility of engine flameout. During the aircraft's research, development, test and evaluation program, most airstarts are the result of intentional engine shutdown for flight test airstart evaluation. This airstart envelope will be validated and modified by flight test results.

(U) When attempting an airstart, as little power as possible should be extracted from the engine until idle speed is reached. The integrated drive generator (IDG) should be kept off the line during the airstart. Data from flight tests have indicated that, up to 30,000 feet, the minimum speed for windmill starts is 300 to 315 KIAS. Above 30,000 feet a speed of 450 KIAS is required for airstarts. Flight test data indicates that a minimum airspeed of 250 to 280 KIAS is required for aircraft control power with two engines windmilling.

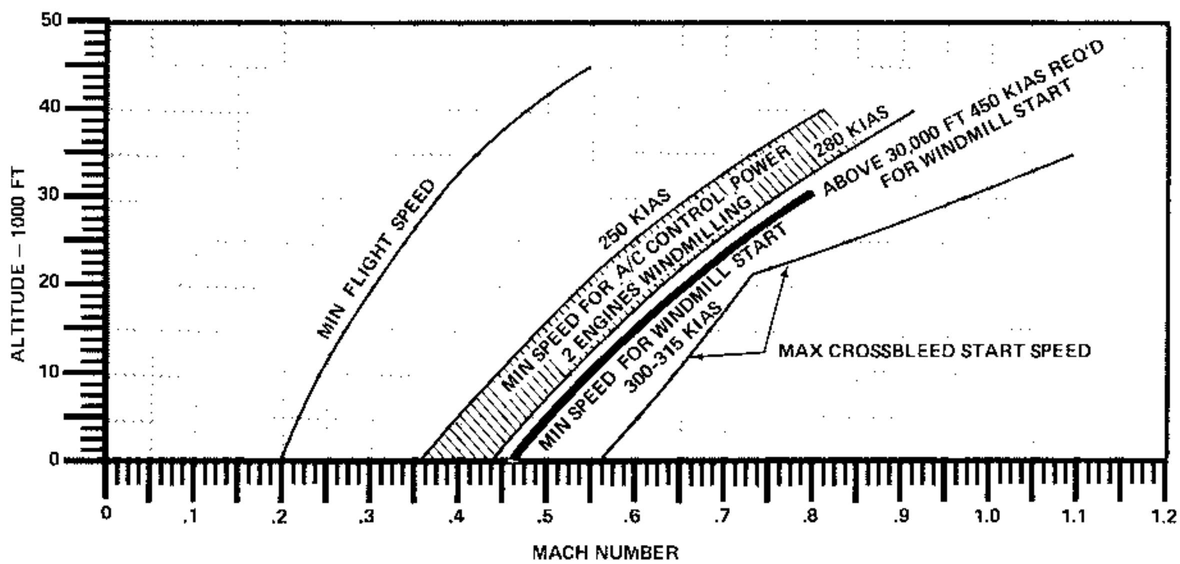
(U) Crossbleed airstarts should be attempted only when the windmilling engine indicates zero rpm (N_2). Maximum crossbleed starts speeds are based on starter limits of 800°F and 265 psig.

AIRSTART ENVELOPE

DATE: DECEMBER 1971
DATA BASIS: FLIGHT TEST AND ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-412

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



NOTE:
FOR ONE ENGINE SHUTDOWN – CROSSBLEED START SHOULD ONLY BE ATTEMPTED WHEN WINDMILLING ENGINE INDICATES ZERO RPM (N_2)

CONFIDENTIAL
0 F050-004
509-0

Figure 11-25.

This page intentionally left blank.

ALPHABETICAL INDEX

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)

ALPHABETICAL INDEX (Cont)