MINISTERO DELLA DIFESA
COSTARMAEREO
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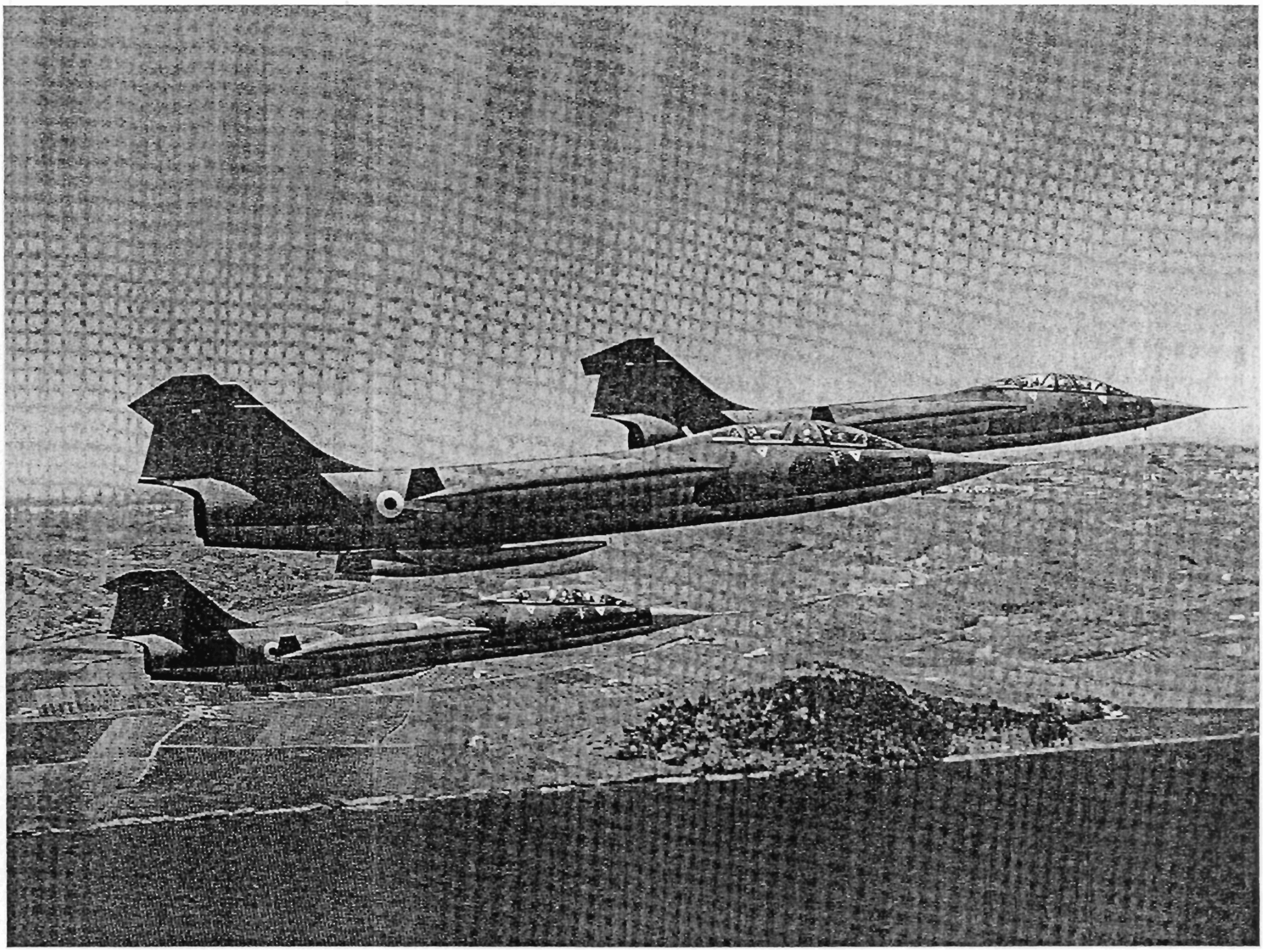
AER.1F–104(T)GM–1

FLIGHT MANUAL



TF–104G–M SERIES AIRCRAFT

ALENIA

(A0019)

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

1 DECEMBER 1996

LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the change is indicated by a vertical line on the outer margin of the page.
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Dates of issue for original and changed pages are:

Original .... 0 .... 1 December 1996

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iv blank	0	7-10 blank	0	A6-5 blank	0
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3-54 blank	0	A3-26 blank	0	Index-1 thru Index-7	0
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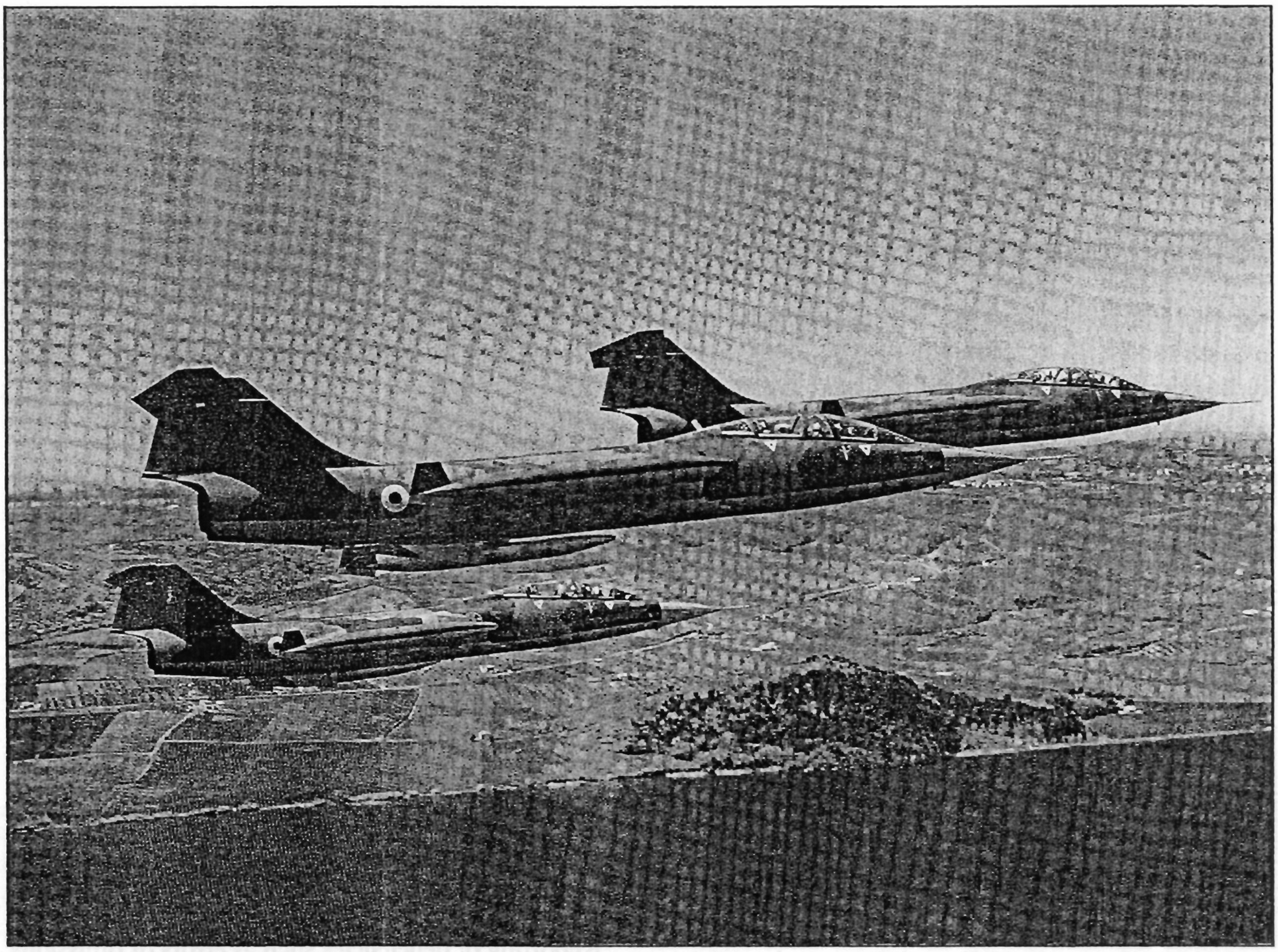
CURRENT FLIGHT CREW CHECKLIST

AER.1F-104(T)GM-1CL-2 Issue 1 December 1996

* Zero in this column indicates an original page

CAUTION

- This publication is valid only if it is composed of the above listed pages and duly amended.
- Copies of this publication may be obtained as follows:
  - FA activities, as directed by specification ILA-NL-9004-0001-00B00
  - COSTARMAEREO activities, as directed by specification AER.00-00-8
- Any deficiency and/or mistake in this publication shall be reported as directed by specification AER.00-00-4.



SCOPE

This manual contains the necessary information for safe and efficient operation of your aircraft. These instructions provide you with a general knowledge of the aircraft and its characteristics and specific normal and emergency procedures.
Your experience is recognized; therefore, basic flight principles are avoided. Instructions in this manual are prepared to be understandable by the least experienced crew that may be expected to operate the aircraft.
This manual provides the best possible operating instructions under most circumstances, but it is not a substitute for sound judgment. Multiple emergencies, adverse weather, terrain, etc. may require modification of the procedures.

PERMISSIBLE OPERATIONS

The flight manual takes a "positive approach" and normally states only what you may do. Unusual operations or configurations are prohibited unless specifically covered herein. Clearance from the using command shall be obtained before any questionable operation, which is not specifically permitted in this manual, is attempted.

HOW TO BE ASSURED OF HAVING LATEST DATA

Refer to the flight manual cover page and list of effective pages, the title block of each safety and operational supplements, and all status pages attached to formal safety and operational supplements. Clear up all discrepancies before flight.

ARRANGEMENT

The manual is divided into seven nearly independent sections to simplify its use as a reference manual. Also the Appendix - Performance Data is inserted as separate section.

SAFETY SUPPLEMENTS

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

OPERATIONAL SUPPLEMENTS

Information involving changes to operating procedures will be forwarded to you by operational supplements.
The procedure for handling operational supplements is the same as for safety supplements.

CHECKLISTS

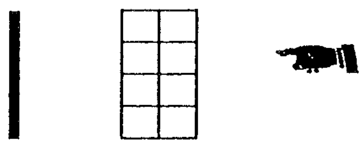
The flight manual contains itemized procedures with necessary amplifications. The checklist contains itemized procedures without the amplification. Primary line items in the flight manual and check list are identical. If a formal safety or operational supplement affects your checklist, the affected checklist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

HOW TO GET PERSONAL COPIES

Each flight crewmember is entitled to personal copies of the flight manual, safety supplements, operational supplements, and checklist. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your publication distribution officer - it is his job to fulfill your request.

CHANGE SYMBOL

The change symbol, is illustrated by a black line in the outer margin of the column or page, adjacent to the affected text, like the one printed next to this paragraph. The change symbol indicates text and tabular illustration changes made to the current issue.
Changes to illustrations (except tabular and plotted illustrations) are indicated by a pointing hand or by a shaded area box located at the upper right side of the illustration.
The box is divided into eight equal parts which represent eight proportional areas of the illustration. The shaded area of the box represents the area of the illustration which contains a change.



PUBLICATIONS DEFICIENCY REPORTING

Reporting of deficiencies such as errors or omissions in this manual shall be effected in accordance with AER.00-00-4.

WARNINGS, CAUTIONS, AND NOTES

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

WARNING

OPERATING PROCEDURES, TECHNIQUES, ETC., WHICH COULD RESULT IN PERSONAL INJURY OR LOSS OF LIFE IF NOT CAREFULLY FOLLOWED.

CAUTION

OPERATING PROCEDURES, TECHNIQUES, ETC., WHICH COULD RESULT IN DAMAGE TO EQUIPMENT IF NOT CAREFULLY FOLLOWED.

NOTE

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

USE OF WORDS SHALL, WILL, SHOULD AND MAY

The words shall, will, should and may have the following meanings in this manual:

Shall is used to express a provision that is mandatory.

Will is used to express a declaration of purpose. It is also used in cases where simple futurity is required.
Should and *may* are used to express a non-mandatory provision.

EFFECTIVITY

All text and graphics within this manual apply to the TF-104GM series aircraft.

YOUR RESPONSIBILITY – TO LET US KNOW

Every effort is made to keep the flight manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. We shall not correct an error unless we know of its existence. In this regard, it is essential that you do your part.
Comments, corrections, and questions regarding this manual or any phase of the flight manual program are welcomed. These should be forwarded as directed by national procedures and in line with national specifications.

IDENTIFICATION OF THE CONFIGURATION STATUS OF THIS MANUAL

This manual contains information regarding the predicted configuration of the TF-104GM two-seater series aircraft.

MODIFICATIONS NOT INCORPORATED IN THIS MANUAL

All modifications which are applicable to this manual, but whose information has not yet been introduced are listed below:

PTD No.	MINISTRY OF DEFENCE DOCUMENT (PTA)	DATE	TITLE
None			

FURTHER MODIFICATIONS INCORPORATED IN THIS MANUAL

Further modifications, not yet formally approved at the cut-off date of the Flight Manual current issue, but which for opportunity reasons have been incorporated in the Manual, are identified and temporarily listed below:

PTD No.	TITLE
None	

OPERATIONAL AND SAFETY SUPPLEMENTS INCORPORATED IN THIS MANUAL

All former Operational and Safety Supplements which have been incorporated in this manual are listed below:

NUMBER	DATE	TITLE
None		

LIST OF INCORPORATED PTA

This list contains only the modifications affecting the contents of this manual. Following embodiment of a modification in all effected aircraft, the corresponding number will not be deleted from the list, but the information regarding the pre-modification configuration will be deleted from the manual.

MINISTRY OF DEFENCE DOCUMENT		COMPANY DOCUMENT			
PRESCRIZIONE TECNICA APPLICATIVA (PTA)		PRESCRIZIONE TECNICA DITTA (PTD)			TITLE
No.	DATE	No.	DATE	CLASS	
None					

LIST OF ABBREVIATIONS

A

A	Ampere
A/A	Air to Air
A/B	Afterburner
ac/AC	Alternating Current
ADC	Air Data Computer
ADF	Automatic Direction Finder
ADJ	Adjustment
AGL	Above Ground Level
Ah	Ampere hour
AI	Attitude Indicator
ALN	Align
ALR	Alarm
AOA	Angle of Attack
APC	Automatic Pitch Control
ASAP	As Soon As Possible
ASCII	American Standard Code for Information Interchange
ATC	Air Traffic Control
AUX	Auxiliary
AWPT	Absolute Waypoint

B

BIT/AFI	Built In Test/Automatic Fault Isolation
BL	Butt Line
BLC	Boundary Layer Control

C

°C	Celsius Degrees
C/A	Coarse Acquisition
CAL	Calibrate
CCW	Counterclockwise
CDP	Compressor Discharge Pressure
CEP	Circular Error Probability
CG	Center of Gravity
CHG	Change
CIT	Compressor Inlet Temperature
CLR	Clear
CMPTR	Computer
COMP	Compass
CW	Clockwise

D

DC	Direct Current
deg	Degree(s)
DES	Destination
DME	Distance Measuring Equipment
DTM	Data Transfer Module
DWP	Destination Waypoint

E

EGT	Exhaust Gas Temperature
ELECT	Electrical
EMER	Emergency
ENT	Enter
ENG	Engine
EPE	Estimated Position Error
EPROM	Erasable Programmable Read Only Memory
EXT	External
EXT TRAN	External Transfer

F

°F	Fahrenheit Degrees
FF	Fixed Frequency
FOD	Foreign Object Damage
fpm	Feet per minute
FREQ	Frequency
ft/FT	Feet

G

gal/GAL	gallon(s)
GC	Gyro Compass
GC	Generator Contactor
GCA	Ground Controlled Approach
GCI	Ground Controlled Intercept
GCU	Generator Control Unit
GD	Guard Frequency
GEN	Generator
G/g	Gravity (Unit of)
GS	Ground Speed
GTK	Ground Track

H

HQ	Have Quick
HSI	Horizontal Situation Indicator
HYD	Hydraulic
HTR	Heater
Hz	Hertz

I

IAS	Indicated Airspeed
ICAO	International Civil Aviation Organization
IFF	Identification Friend or Foe
IGV	Inlet Guide Vanes
ILS	Instrument Landing System
IN CDU	Inertial Navigator Control Display Unit
IND	Indicator
INS	Inertial Navigation System
INU	Inertial Navigation Unit
IP	Initial Position

K

KEAS	Knots Equivalent Airspeed
KIAS	Knots Indicated Airspeed
Km/km	Kilometers
KVA/Kva	Kilovolt ampere

L

LAT	Latitude
lb/LB	Pound(s)
lb/hr	pounds per hour
LDG	Landing Gear
LE/L.E.	Leading Edge
LG	Landing Gear
LONG	Longitude
LST	List

M

M	Magnetic
MAC	Mean Aerodynamic Chord
MAX	Maximum
mb	millibar
MFC	Main Fuel Control
MHz	Mega Hertz
MSL	Mean Sea Level
MWOD	Multiple Word of Day

N

NASARR	North American Searching And Ranging Radar
NAVSTAR	NAVigation System using Time And Racing
NET	Network
NM	Nautical Miles
NXT	Next

O

OC	Overcurrent
OTF	On Top Fix(ing)
OV	Overvoltage

P

P code	Precision Code
PEC	Personal Equipment Connector
PNEU	Pneumatic
POS	Position
PP	Present Position
pph	Pounds per hour
PP1	Primary DC Bus
PP2	No. 1 Emergency DC Bus
PP3	No. 2 Emergency DC Bus
PP4	No. 1 Battery Bus
PP5	No. 2 Battery Bus
PPS	Precise Positioning Service
PSI/psi	Pounds per square inch
PRICE	Pressure Regulator Indicator, COntrols, Emergency
PTA	Prescrizione Tecnica Applicativa
PTD	Prescrizione Tecnica Ditta
PTT	Press To Transmit
PWR	Power

Q

QNH	Barometric Pressure at Sea Level
QT/QTY	Quantity

R

RAT	Ram Air Turbine
RCR	Runway Condition Reading
RCV	Receive
RDY	Ready
RDY NAV	Ready Navigation
RG	Rate Gyro
RPM	Revolution Per Minute
RWPT	Relative Waypoint

S

sec/Sec	Second(s)
SEL	Selector
SIF	Selective Identification Feature
SYNC	Synchronize
SLS	Side Lobe Suppression
SPS	Standard Positioning Service
STBY	Standby
STO	Stored
SW	Switch

T

T	Time
TACAN	TACtical Air Navigation
TAS	True Air Speed
TCN	TACAN
TE/T.E.	Trailing Edge
TE	Terminal Error
TOA	Time Of Arrival
TOD	Time Of Day
TRU	Transformer Rectifier Unit
TTG	Time To Go

U

UF	Under frequency
UHF	Ultra High Frequency
UTC	Universal Time Coordinated
UV	Undervoltage

V

V	Volt
VHF	Very High Frequency
VOL	Volume
Vs	Versus

W

WARN	Warning
WF	Wild Frequency
WOD	Word of Day
WPT	Waypoint

X

XMTR	Transmit
XP1	No. 1 Primary Wild Frequency AC Bus
XP2	No. 2 Primary Wild Frequency AC Bus
XP3	Secondary Wild Frequency AC Bus
XP4	Emergency AC Bus
XP5	Primary Fixed Frequency AC Bus
XP6	Instrument Fixed Frequency AC bus
XP7	Secondary Fixed Frequency AC Bus

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SECTION I

DESCRIPTION AND OPERATION

TABLE OF CONTENTS

THE AIRCRAFT

The two-place TF-104G-M aircraft is high-performance, all weather, day and night trainer powered by an axial-flow, turbojet engine with afterburner (refer to Figure FO-1). The aircraft, developed by the Lockheed Corporation, is designed for high subsonic cruise. Notable features of the aircraft include extremely thin flight surfaces, short straight wings with negative dihedral, irreversible-hydraulically-powered flight controls, a controllable horizontal stabilizer mounted at the top of the vertical stabilizer, engine inlet duct anti-icing, an anti-skid brake system, an automatic pitch control system. The wings have leading and trailing edge flaps, and a boundary layer control system which is used in conjunction with the trailing edge flaps to reduce landing speeds. An upward ejection system is used for emergency escape. A drag chute is installed to reduce the landing roll and an arresting hook is available for bringing the aircraft to an emergency stop. Internal fuel cells and external fuel tanks may be serviced through a single-point pressure refueling system.

DIMENSIONS

Overall dimensions of the aircraft are as follows:

Wing span (with tiptanks)	25 feet
Wing span (without tiptanks)	22 feet
Length (with boom)	58¼ feet
Height (to top of vertical stabilizer)	13½ feet
Tread	8¾ feet

Refer to Section II "Normal Procedures" for minimum turning radius and ground clearances.

GROSS WEIGHT

The approx gross weight of the fully loaded aircraft is 19732 pounds with JP-8 fuel (0.80 Kg/dm³).

NOTE

Refer to Appendix – Performance Data -- Part 1 Introduction for aircraft weights including tip and/or pylon fuel tanks.

COCKPIT GENERAL ARRANGEMENT

The front and rear cockpits (refer from Figure FO-4 to FO-7) are enclosed by two separate canopies hinged to the fuselage left side. The cockpits are equipped with two ejection seats in tandem and comprise the main instrument panels, the left and right consoles, the control stick grips and the engine throttles.

ENGINE

Refer to Figure FO-2 for engine layout. The aircraft powerplant is a J79-GE-11-B turbojet engine. Uninstalled engine sea level static thrust rating at Military Thrust (maximum thrust non-afterburning) is approximately 10000 pounds. Maximum thrust (full afterburning) under the same conditions is approximately 15800 pounds. During engine operation, the 17-stage axial flow compressor is driven by a 3-stage turbine. The turbine is impelled by combustion gases directed against the turbine buckets. Combustion gases after passing through the turbine section flow around the afterburner spraybars where additional fuel may be introduced to obtain a substantial gain in thrust. The inlet guide vanes and the first six stages of compressor stator vanes are variable to reduce the possibility of compressor stall. The variable stator system automatically positions the vanes to direct the air intake flow in relation to engine speed and compressor inlet temperature (CIT). The aft end of the engine is comprised of a variable area nozzle which increases exhaust gas velocity by decreasing the exhaust outlet area. Automatic nozzle area control is provided to obtain the desired thrust within the safe operating limits of the engine. Mechanical energy to power accessory components is obtained through engine-mounted gearboxes driven by a shaft splined to the compressor rotor. The engine systems, grouped for simplified discussion, are:

a. Starter and ignition systems

b. Engine fuel system

c. Afterburner fuel system

d. Variable stator system

e. Variable nozzle system

f. Lube oil system

Tachometer

The tachometer, mounted on the right side of the main instrument panel in each cockpit (refer to Figure 1-1), indicates engine speed in a percentage of the maximum rated speed of 7460 RPM.
The instrument is powered by a tachometer generator, which generates a frequency proportional to engine speed.

ENGINE STARTER AND IGNITION SYSTEMS

Engine Starter System

The engine starter requires compressed air from a ground turbine source, which is converted to mechanical energy by the starter to rotate the engine. The receptacle for connecting the pressurized air line is located in the right wheel weel. An electrical receptacle located adjacent to the air connector permits electrical connection from the cockpit start switches to the electrically controlled air valve on the ground starting unit by means of an auto-start cable. The cable connections should be made to assure pilot control of starting.

Start Switches. Two start switches are located on the left forward panel (refer to Figure 1-1) in each cockpit and marked 1 and 2. The switches have a START, a STOP START, and a center, neutral position. The switches are spring-loaded in the neutral position. By momentarily placing either switch in the START position, battery bus power is supplied to energize the ignition circuit and begin the 45-second ignition cycle.
During an ignition cycle, the start switches may be reset to initiate a new 45-second cycle. Placing a start switch in the STOP START position deenergizes the ignition circuits, but will not shut down the engine if combustion has started.

With the auto-start cable installed, the start switches are used to open and close the starting air control valve. Both switches are used simultaneously to energize the ignition systems to insure reliability during air starts.

Engine Motoring Switch. The engine motoring switch, located on the right console front cockpit only (refer to Figure 1-1), is spring-loaded in the OFF position. The switch is provided to purge the engine by allowing ground turbine air to motor the engine without the ignition system energized. The engine motoring switch receives power from the PP4 bus.

NOTE

> Observe the starter RPM and time limits detailed in Section II (Normal Procedures) and Section V (Operating Limitations).

Engine Ignition Systems

Dual ignition systems are provided for air start reliability. Each includes an individual battery and battery bus, a switch, and spark plug. When energized, the ignition circuit selected fires a spark plug for 45 seconds unless the start switch is moved to the STOP START position sooner. Ignition is propagated through combustion chamber cross-fire tubes. Activation of the circuit provides stand-by ignition for immediate engine re-light if flameout occurs.

Afterburner Ignition System

Afterburner ignition is controlled by a throttle-actuated ignition switch. The afterburner ignition unit receives power from the XP2 bus when the throttle is moved to any position in the afterburner range. A spark plug located within the pilot burner operates continuously during after-burning, assuring positive ignition of the pilot burner. The pilot burner may also be ignited by ground personnel. This is accomplished by actuating the test button on the afterbuner ignition switch located on the throttle shaft gearbox. Once the system has been actuated, the pilot burner remains lighted regardless of throttle position until the engine is shut down.

ENGINE FUEL SYSTEM

Refer to Figure FO-3 for engine fuel system. The engine fuel system pressurizes, meters, atomizes, and injects fuel into the compressor discharge airstream. The system is regulated by the engine fuel control unit as a function of throttle position, engine speed, compressor discharge pressure (CDP), and CIT. Fuel is supplied to the engine fuel pump for four booster pumps in the main tank. Pressurized fuel enters the engine fuel control unit which, in addition to metering engine fuel, supplies fuel as a hydraulic medium to position the variable stator vanes. From the fuel control unit, fuel is routed through the fuel-oil cooler, the pressurizing and drain valve, and is distributed by the fuel manifold to the 10 nozzles, where injection into the airstream occurs.

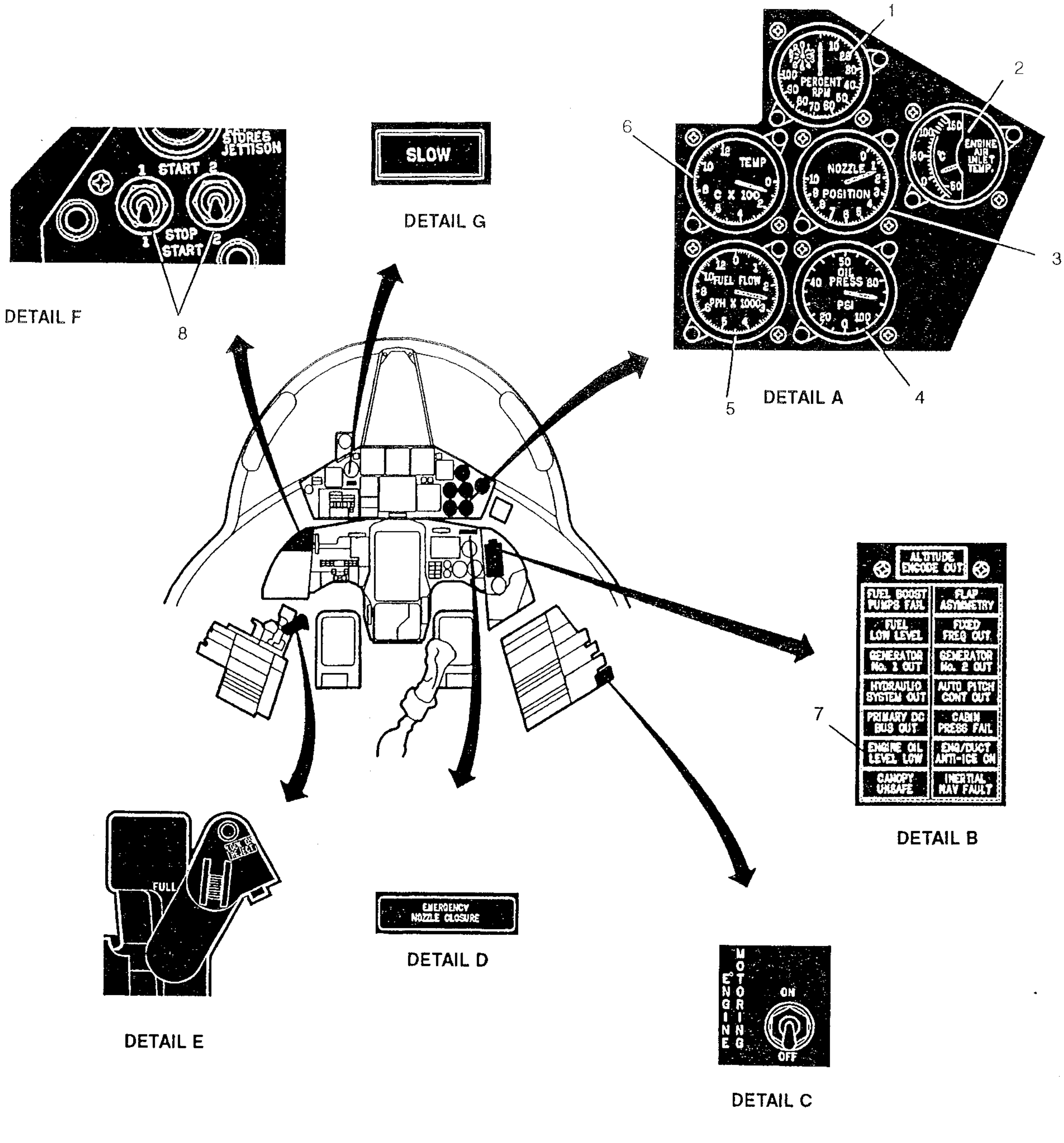
Engine Fuel Pump

The engine-driven fuel pump incorporates a centrifugal boost element and two positive displacement, gear-type elements to supply to engine high pressure fuel requirements. The centrifugal boost element supplements the fuel supply system boost pump pressure. Failure of the centrifugal boost element does not affect engine operation except during high fuel consumption operation in low altitude-high Mach number flight. Each gear-type element is capable of supplying sufficient fuel to the engine should one element fail.

Engine Fuel Control Unit

The engine fuel control unit meters the fuel flow rate as a function of throttle position, engine speed, CDP, and CIT. A metering valve in the fuel control unit is positioned in response to various internal operating signals and meters fuel to the engine as a function of the integrated signals. A by-pass valve ports excess fuel back to the fuel pump and a cutoff valve shuts off the fuel supply to the engine burners when the throttle is in the OFF position. In addition to regulating fuel flow, the fuel control unit produces coordinating signals to: (a) Schedule the variable stator vanes; (b) Prevent afterbuner operation until the correct engine speed of 93.4% exists; and (c) Limit the exhaust nozzle to cruise area when the engine is accelerating. The fuel control unit also supplies servo fuel at regulated pressure to the force amplifiers in the nozzle area control, and provides fuel for the afterburner pilot burner.

ENGINE INSTRUMENT CONTROLS AND INDICATORS (FRONT COCKPIT)



A – ENGINE INSTRUMENTS
1 TACHOMETER
2 COMPRESSOR INLET TEMPERATURE GAGE
3 EXHAUST NOZZLE POSITION INDICATOR
4 OIL PRESSURE GAGE
5 FUEL FLOW INDICATOR
6 EXHAUST GAS TEMPERATURE GAGE

B – WARNING LIGHTS PANEL
7 ENGINE OIL LEVEL LOW WARNING LIGHT

C – ENGINE MOTORING SWITCH

D – EMERGENCY NOZZLE CLOSURE HANDLE

E – ENGINE THROTTLE

F – LEFT FORWARD PANEL
8 START SWITCHES (1 AND 2)

G – SLOW WARNING LIGHT

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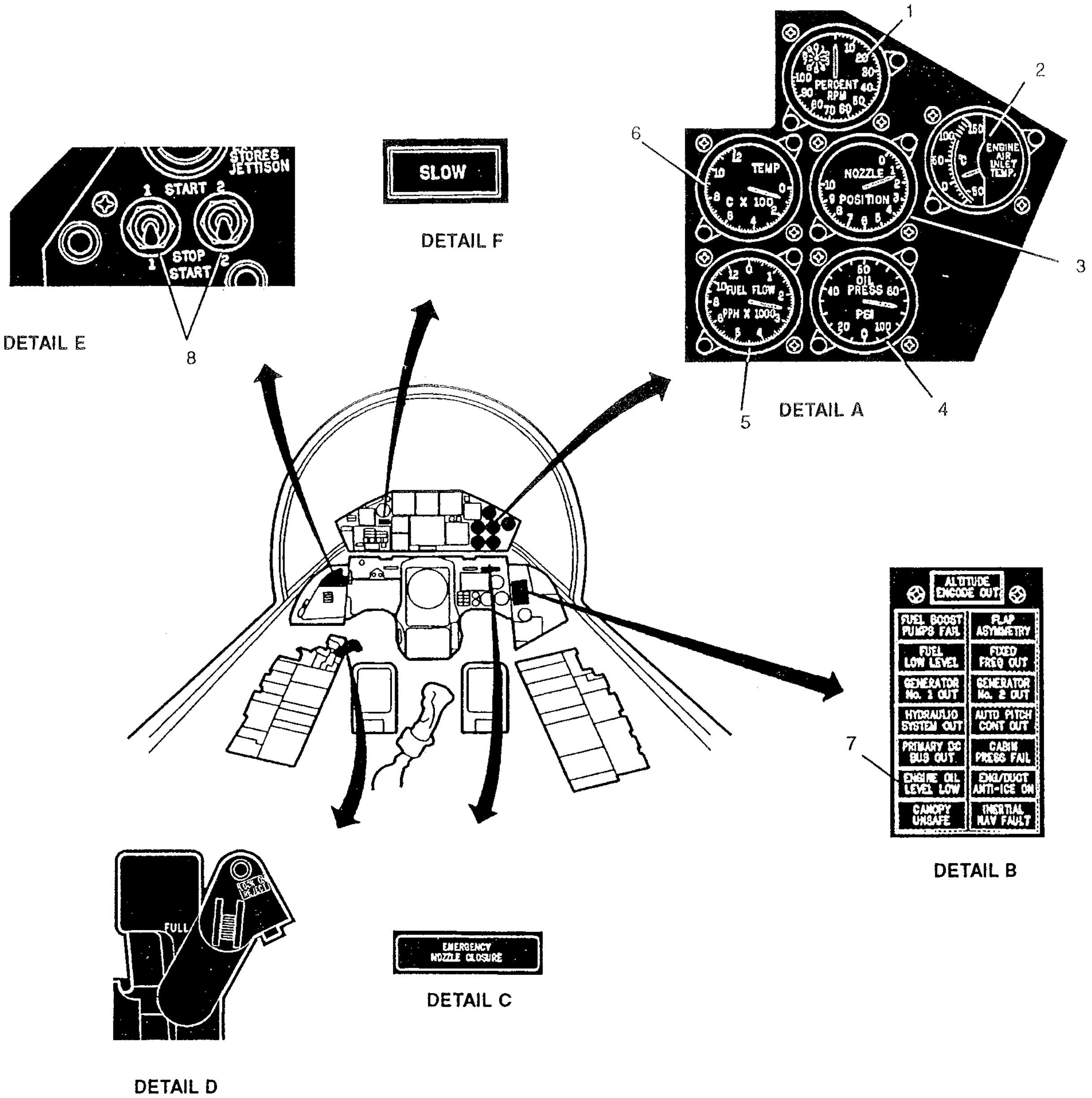
Figure 1-1 (Sheet 1 of 2)

ENGINE INSTRUMENT CONTROLS AND INDICATORS (REAR COCKPIT)



A – ENGINE INSTRUMENTS
1 TACHOMETER
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4 OIL PRESSURE GAGE
5 FUEL FLOW INDICATOR
6 EXHAUST GAS TEMPERATURE GAGE

B – WARNING LIGHTS PANEL
7 ENGINE OIL LEVEL LOW WARNING LIGHT

C – EMERGENCY NOZZLE CLOSURE HANDLE

D – ENGINE THROTTLE

E – LEFT FORWARD PANEL
8 START SWITCHES (1 AND 2)

F – SLOW WARNING LIGHT

FA0152

Figure 1-1 (Sheet 2 of 2)

Fuel Flow Transmitter

The fuel flow transmitter is located on the engine and consists of a movable vane held in place by a calibrated spring, and a transmitter unit. Fuel flow causes the vane position to vary. The vane is magnetically coupled to the transmitter unit, which converts the vane position to an electrical output signal to the cockpit fuel flow indicator.

Fuel Flow Indicator

The fuel flow indicator, located on the right side of each main instrument panel (refer to Figure 1-1), indicates the consumption rate in pounds per hour and is graduated from 0 to 12000 pounds. The indicator is powered by XP6 bus through the instrument auto-transformer on the electronic compartment circuit breaker panel. The instrument does not indicate afterburner fuel flow.

Fuel-Oil Cooler

The fuel-oil cooler reduces engine oil temperature by circulating the oil around tubes containing the cooling fuel flow. The assembly includes a fuel temperature control valve, and a fuel by-pass valve. The fuel temperature control valve senses oil outlet temperature and controls routing of the fuel. The fuel by-pass valve system is designed so that if the oil is cold or if the fuel becomes too hot, the oil will bypass the cooler; if the fuel flow becomes high, the fuel will bypass the cooler.

Pressurizing and Drain Valve

The pressurizing and drain valve maintains back pressure to the engine fuel control unit to provide and acceptable fuel pressure level for servo operation. The valve permits fuel to flow to the engine when the discharge pressure exceeds a preset value. The drain valve portion of the unit drains the engine fuel manifold when the engine is shut down.

Fuel Nozzles

The fuel nozzles atomize fuel into the combustion liners in a whirling, conical-shaped spray pattern. The nozzles are located in the compressor rear frame, one extending into the end of each of the 10 combustion liners.

Throttle

The engine is controlled by means of one throttle lever located on front and rear cockpit left consoles (refer to Figure 1-2). In addition to the throttle control lever (which provides a mounting point for the speedbrake switch and microphone pushbutton), the throttle quadrant is the installation panel for the wing flap control lever. A throttle-actuated switch for the landing gear warning signal circuit is also installed in the throttle quadrant. By establishing the fuel flow rate in the engine, the throttle controls engine RPM in non-afterburner positions, and thrust augmentation in afterburner operation. The throttle quadrant is marked OFF, IDLE, and FULL, and the throttle is spring-loaded inboard. Throttle advancement from the OFF position drops the lever into the IDLE position. Full travel from IDLE to the Military Thrust setting is obtained by a straight, forward motion. During ground starts, the throttle is set in the IDLE position. When combustion has started, throttle advancement increases engine speed until 100% RPM is reached (see Figure 1-7). Afterburning is initiated by moving the throttle outboard and forward into the afterburner slot. The throttle linkage provides adequate friction to prevent lever creepage and eliminates the need for a throttle friction control.

AFTERBURNER FUEL SYSTEM

The afterburner fuel system supplies and regulates fuel sprayed into the engine tailpipe for afterburner combustion. The ignited fuel-air mixture increases engine thrust by further heating and expanding the exhaust gases. To initiate afterburner operation, the throttle must be advanced into afterburner range and the engine speed must be sufficiently high to support afterburning (93.4% RPM). The afterburner fuel flow is through the afterburner on-off valve, afterburner fuel pump, afterburner fuel control unit, afterburner fuel-oil cooler, flow divider and selector valve, and spraybars. Afterburner ignition is initiated and maintained by a pilot burner.

Afterburner On-Off Valve

The afterburner on-off valve is an integral part of the afterburner fuel pump, and is located in the pump inlet. The valve allows fuel to enter the afterburner fuel pump when actuated by a high pressure fuel signal from the engine fuel control unit.

THROTTLE QUADRANT

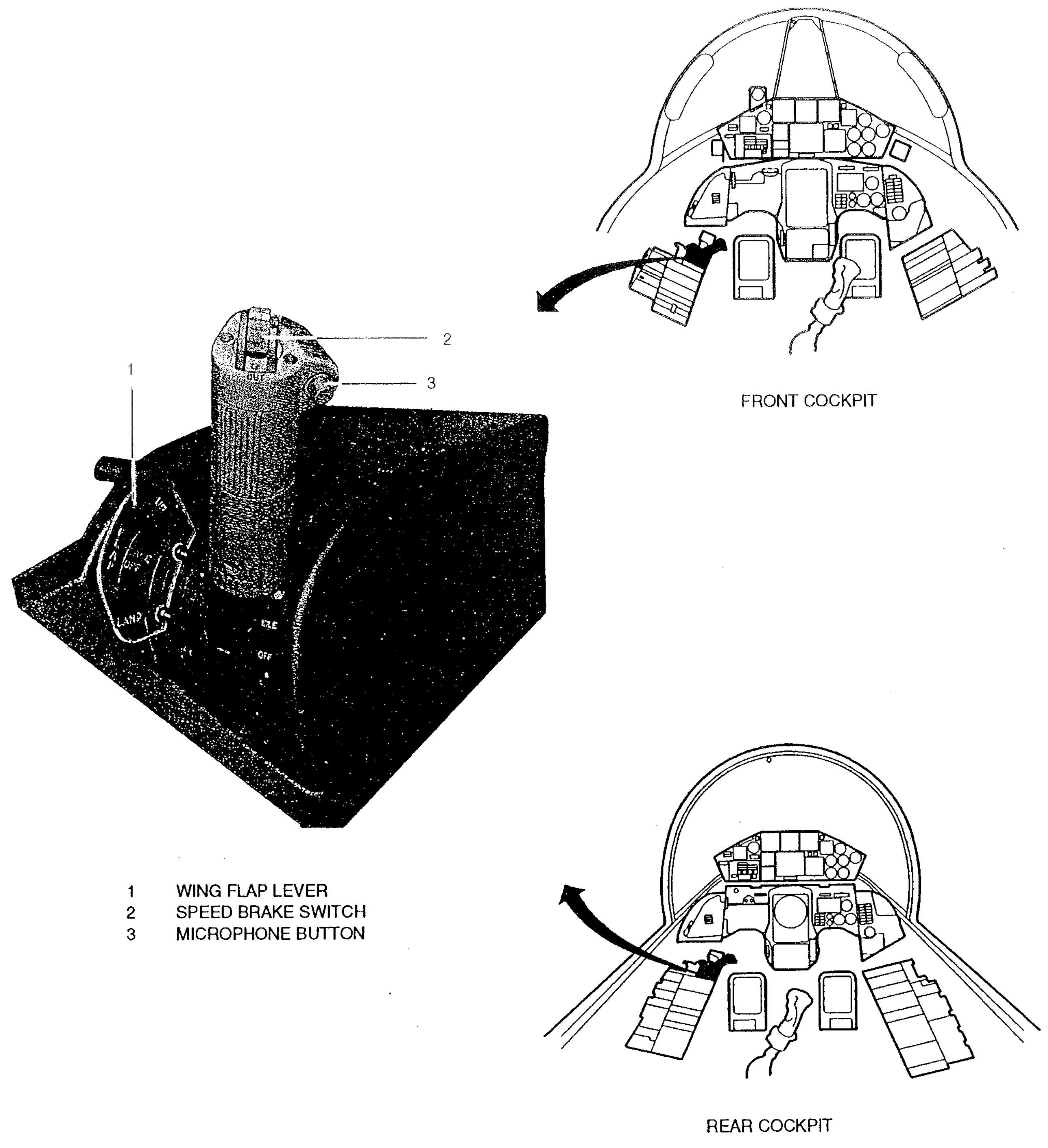


Figure 1-2

Afterburner Fuel Pump

The afterburner fuel pump is an engine-driven centrifugal pump. The pump rotates continuously, discharging fuel to the afterburner fuel system only when the afterburner on-off valve is open.

Afterburner Fuel Control

The afterburner fuel control is linked mechanically to the engine fuel control unit. Fuel entering the afterburner fuel control is metered in response to throttle position and changes in CDP.

Afterburner Fuel-Oil Cooler

The afterburner fuel-oil cooler reduces scavenge oil temperature, using afterburner fuel as the coolant. The cooler is similar in operation to the main fuel oil cooler.

Flow Divider and Selector Valve

During minimum afterburner operation, the flow divider and selector valve divides afterburner fuel between the primary sector and secondary sector of the spraybar manifold. As afterburner operation approaches maximum output, the flow divider and selector valve supplies fuel to the primary uniform and secondary uniform portions of the spraybar manifold.

Spraybar Manifold and Spraybars

The spraybar manifolds deliver afterburner fuel from the flow divider and selector valve to the tailpipe, where the fuel is sprayed into the exhaust gas stream (Figure 1-3).
The assembly consists of 4 manifolds and 20 dual spraybars. Each dual spraybar consists of two spraybar tubes mounted in tandem.
Fuel is supplied to the front spraybars from either the secondary sector or secondary uniform manifold.
Fuel is supplied to the rear spraybar tubes from either the primary sector or primary uniform manifold.
The four states of afterburner fuel flow (according to throttle position) are:

1. Primary sector
2. Secondary sector
3. Primary uniform
4. Secondary uniform

VARIABLE STATOR SYSTEM

The variable stator system is composed of the inlet guide vanes and the first six stages of the compressor. Its function is to prevent compressor stall by limiting the inlet air flow angle. The system uses fuel from the engine fuel control unit as the actuating medium. The component of the system are the variable stator actuators.

Variable Stator Actuators

The variable stator actuators are double-acting cylinders which position the inlet guide vanes and variable stator vanes.

COMPRESSOR INLET TEMPERATURE (CIT) WARNING SYSTEM

The aircraft is equipped with a system to give visual warning in the cockpit when compressor inlet temperature increases to a critical value. The system consists of two temperature-sensing detector, a warning light, and a warning flasher.

Compressor Inlet Temperature Gage

A temperature gage (refer to Figure 1-1), located on each upper instrument panel, indicates the CIT. The instrument is graduated from −70° to +150° C and receives power from the PP1 bus.
The compressor inlet temperature gage installed incorporates two temperature sensors in the right generator cooling duct. One sensor is connected to the front cockpit indicator and the other sensor is connected to the rear cockpit indicator. The temperature gage is red-lined at 100° C to compensate for system errors. The temperature of the air in the generator cooling duct reads approximately 10° C cooler than compressor inlet temperature. Also, the sensor has a lag error of approximately 10° C. This results in an overall error of approximately 20° C. For this reason the gage has been red-lined at 100° C to avoid overrunning the CIT limit of 120° C actual temperature during acceleration.
The system receives power from the XP3 bus. The FUEL QUANTITY and WARNING LIGHTS TEST switch located on the right side of the main instrument panels, when moved to the FUEL

AFTERBURNER FUEL MANIFOLD

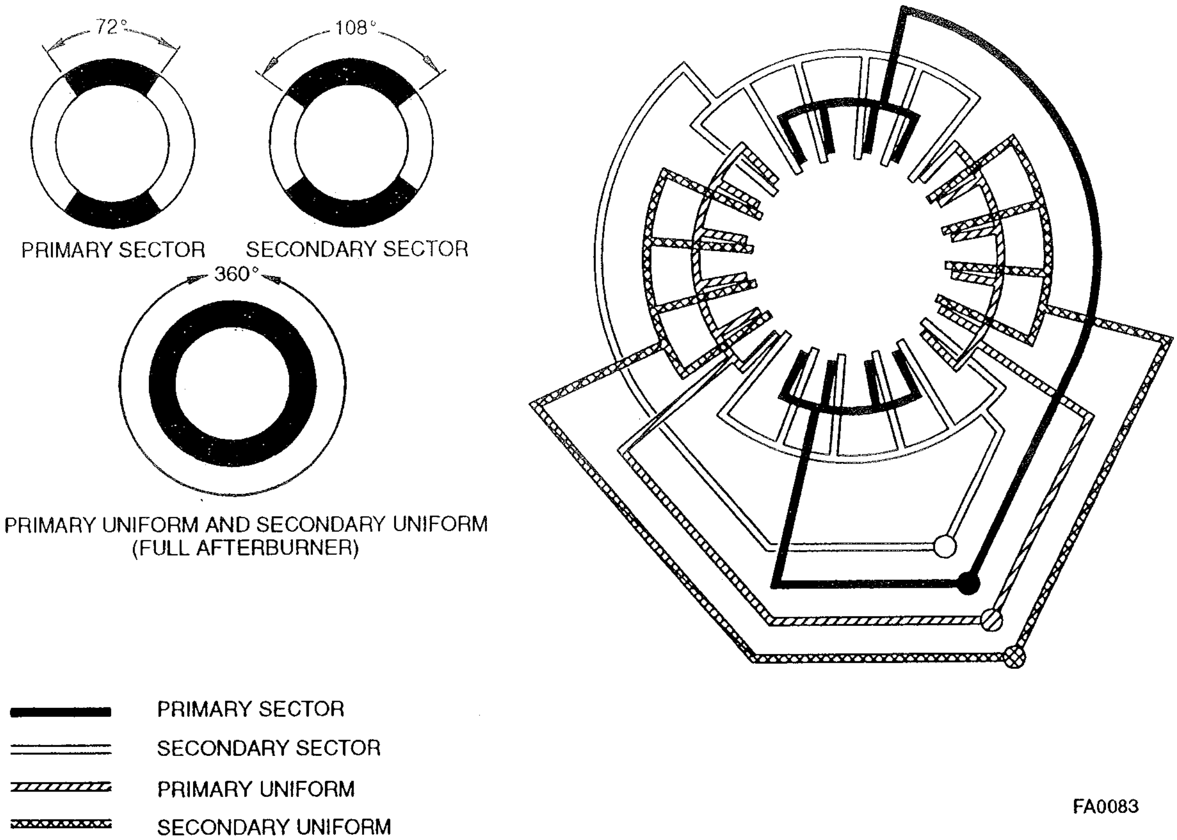


Figure 1-3

QUANTITY position will cause the gage needle to move below −70° C indicating that the system is functioning.

Compressor Inlet Temperature Warning Light Placard

A placard-type warning light, located on the upper left side of each main instrument panel (refer to Figure 1-1), illuminates when the CIT exceeds a predetermined limit. When energized, the flashing placard illuminates the word SLOW to alert the pilot to reduce aircraft speed to avoid excessive compressor inlet temperature. The light receives power from the PP2 bus. The warning light temperature sensing device is located in the left main ac generator blast tube. The light may or may not coincide with the indication on the gage, due to system lag. The air in the left generator cooling duct is approximately 10° C cooler than CIT; therefore, automatically the SLOW light will not illuminate if the limits of Mach 2 are observed during aircraft acceleration. If the aircraft is accelerated rapidly to Mach 2 without exceeding the 100° C or slow light limitations, continued operation at Mach 2 will cause a gradual increase in the indicated CIT with no increase in actual CIT due to the system lag.

Compressor and Variable Stator Operation

To understand the J79 engine it is important that the pilot understand the need for the variable inlet guide vanes and stator system. In order to optimize subsonic cruise performance in supersonic engines a high pressure ratio compressor is desirable.
High pressure ratio compressor can be designed to operate efficiently in only one speed range without incorporating some type of compensating device. This may be accomplished by several methods, most predominant of which are the dual compressor

system and the variable stator system as used in the J79 engine.
The J79 was designed for maximum operating efficiency at the higher RPM settings. With a fixed guide vane position, the higher the engine RPM the lower the effective angle of attack of the compressor blades, and conversely, as the RPM is reduced the effective angle of attack is increased to the point that blade stall will occur the same as an aircraft wing will stall when its critical angle of attack is exceeded. The variable IGV and stator system (referred to only as IGV) was designed in the J79 engine to allow stall-free operation throughout the entire speed range. As RPM is reduced from Military the IGV will start to close and track closed as a function of RPM at any given air inlet temperature.
At sea level standard day conditions the IGV will be full open above 94.5% RPM, and below 94.5% they will track as a function of RPM to the closed position, reaching the closed position at 67% RPM. The IGV track through a total of 35° of travel from open to closed. The closing of the IGV avoids the stall area by directing airflow against the compressor blades at below the critical angle as RPM is varied. In addition, total airflow through the compressor is reduced as RPM decreases and the IGV shift toward closed, relieving the load on the rear stages of the compressor, thereby avoiding compressor stall. The IGV control senses compressor inlet temperature (CIT) as well as as physical RPM of the engine. The IGV schedule follows a constant slope as a function of engine RPM; however, the slope is shifted as a result of CIT. At higher CIT values the IGV will start to close at a higher indicated RPM, and conversely will start to close at a lower RPM when the CIT is below standard.

Corrected RPM

When dealing with the J79 engine the term "corrected RPM" is often encountered. This term needs clarification to further understand the operation of the J79. The pumping characteristics of the compressor are affected by the temperature of the air entering the compressor, since temperature affects the density. An increase in CIT is effectively the same as a reduction in RPM as far as the compressor is concerned, and conversely, a decrease in CIT is effectively the same as an increase in RPM. To maintain a constant mass flow of air through the engine the corrected RPM must be varied inversely as a function of compressor inlet temperature.
As an example, 100% indicated RPM with a 15° C inlet temperature is 100% "corrected RPM". One-hundred percent indicated RPM with an inlet temperature at 95° C corrects to an effective or "corrected RPM" of 88.5% while 100% indicated RPM at −28° C is a corrected RPM of 108.6%.

Engine RPM Control Features

In addition to its normal governing functions, the engine fuel control until senses CIT, integrates it with the RPM signal, and a "corrected RPM" is computed mechanically within the unit. This data is used to vary the IGV angle and limits corrected engine RPM to a present maximum.

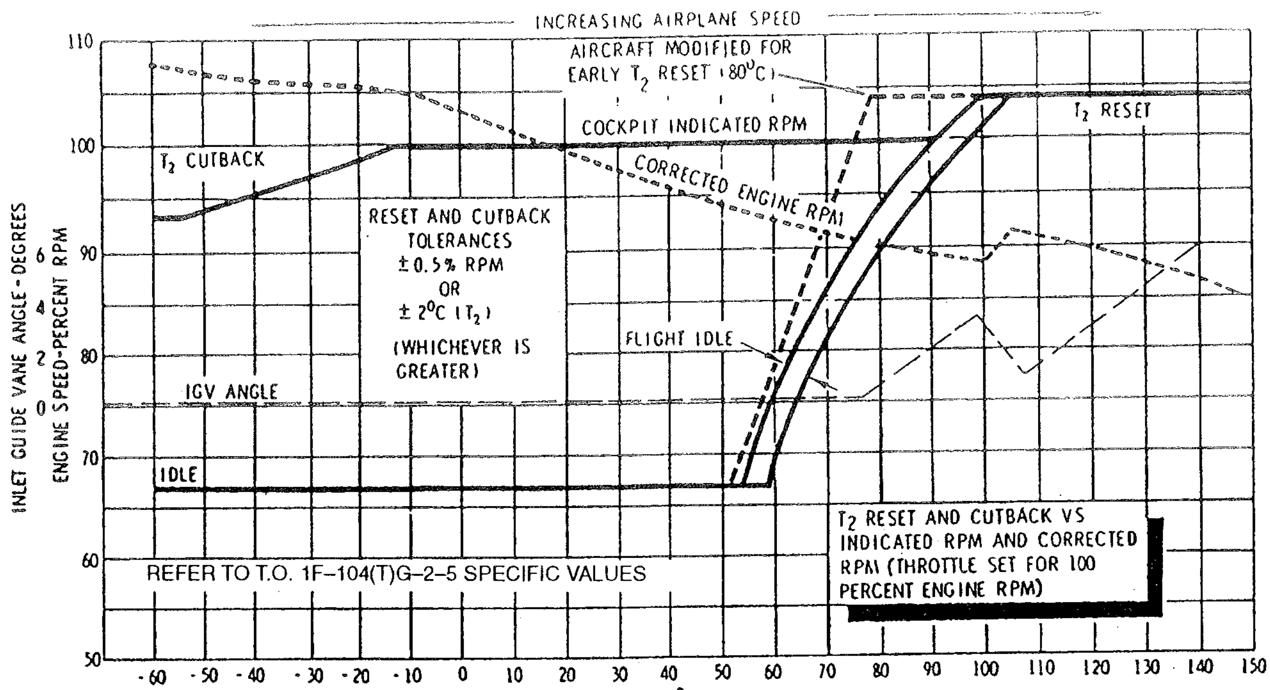
High Corrected RPM and RPM Cutback

With a set RPM of 100% physical RPM, the "corrected RPM" increases as CIT decreases. At approximately −12° C, 100% indicated RPM is equal to 105% corrected RPM. At this point the engine fuel control unit will limit fuel flow to the engine, thereby reducing physical RPM to limit "corrected RPM" to a maximum of 108.6%.
Reduction in RPM as the aircraft climbs into colder ambient air conditions is not an RPM "droop" as was prevalent with earlier jet engines, but rather is a scheduled reduction in physical RPM to maintain "corrected RPM" within limits (see Figure 1-4).

Low Corrected RPM and T2 Reset

As CIT is increased, the "corrected RPM" decreases and would decrease to the point where a low corrected RPM stall would be encountered if no compensating action were taken. The IGV schedule is designed to follow corrected engine RPM and will close to provide stall margin. Increasing physical engine RPM also increases corrected RPM for an increased stall margin and at the same time increases the maximum available thrust. This increase in RPM to a maximum of 103.5% (± 1.0) indicated is referred to as "T_2 reset". The flight idle fuel flow schedule is also raised as a function of CIT and flight idle RPM is the same as maximum RPM when in T_2 reset.
Therefore, a throttle "chop" below Military will not produce an immediate reduction in RPM. This feature was incorporated to prevent sudden reduction in RPM at high Mach numbers which would decrease engine air flow, thereby causing duct buzz (see Figure 1-4). Due to temperature lag and/or sensor location. T_2 reset may start as low as 70° C.

T2 RESET AND CUTBACK VS ENGINE SPEED



NOTE

THIS CHART IS FOR TRUE COMPRESSOR INLET TEMPERATURE (CIT). REFER TO TEXT FOR APPROXIMATE GAGE READINGS

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Figure 1-4

NOTE

T_2 reset will occur approximately 15° C lower for engines modified for early T_2 reset. On the ground, idle RPM may increase due to high ambient temperatures or close proximaty of intakes to other jet exhaust. Under this condition, idle may increase to as high as 85% RPM although thrust increase is not significant due to the open exhaust nozzle. Flight idle RPM will be the same as maximum RPM at 80° actual CIT and above. This condition may occur at low altitude and high speed due to the lower threshold of T_2 reset actuation.

CDP Limiter

At low altitude when the maximum airspeed region of the aircraft is approached, compressor inlet pressure is drastically increased. Compressor discharge pressure (CDP) approaches the maximum value permitted by the physical strength of the engine components. A CDP limiter is incorporated in the engine fuel control unit which senses CDP and reduces fuel flow to the engine, allowing RPM to decrease when the maximum CDP limit is attained.

CIT Sensor

Compressor inlet temperature (CIT) is sensed by capillary tube located in the lower left-hand side of the compressor inlet. This capillary unit transmits a signal to the engine fuel control unit. This signal reacts on internal portions of the fuel controller and modifies IGV and fuel flow schedules as a function of CIT.

NOTE

A COLD SHIFT may occur if the CIT sensor system produces an erroneous CIT signal (in the full cold direction) to the engine fuel control unit. When this condition occurs, as shown on Figure 1-4, the engine fuel control unit will schedule engine speed to the minimum T_2 cutback RPM of 94%.

ENGINE COMPRESSOR STALLS

Engine compressor stalls may be caused by various factors, such as engine fuel control malfunction, IGV misrigging. CIT sensor cold shift, afterburner surging, or inlet distortion.

Regardless of the cause, the result is that the compressor blades stall in much the same manner as the wings of an aircraft. During normal operation, the compressor blades generate axial airflow from the front of the compressor to the rear, at ever increasing pressure.

This pressurized air is delivered to the combustion chamber, where it is heated, and exhausted through the nozzle at a greater velocity than it had at the compressor inlet, thus producing thrust. A compressor stall occurs whenever this axial airflow is interrupted in its normal rearward travel and slows or stops at some stage of compression, thus stalling the airfoil-shaped compressor blades.

Some of the reasons for this airflow interruption are the following:

1. Foreign object damage which destroys the airfoil shape of the compressor blades, reducing their ability to pump air.
2. Corrosion on the compressor blades and stators which reduces their capability to pump air at everincreasing pressure, the same way frost destroys the ability of an aircraft wing to produce lift.
3. If the fuel "schedule" during a throttle burst is too high, pressures in the combustion chamber may increase to the point beyond which the compressor cannot pump air and the axial velocity of the air slows to the point where the blades stall.
4. If the inlet guide vanes are too wide open for a given engine condition, the front of the compressor will pump too much air and overflow the rear of the compressor, resulting in a pile-up of air which decreases the axial airflow velocity, and the compressor will stall.

Other factors, such as aircraft high G load, high angle-of-attack with its resulting compressor inlet pressure distortion, or operation at high Mach numbers and CIT outside the prescribed limits may also lead to compressor stall.

Primarily, there are two types of engine compressor stalls, both of which are relatively easy to recognize. the first type, most commonly associated with high Mach number flight, is characterized by loud banging and chugging sounds which are definitely noticeable in the cockpit. This type of stall may be eliminated by retarding the throttle below Military thrust. The second type of stall is normally associated with subsonic flight. It is not violent, and the only evidence of this stall is a loss of thrust and a mid rumble which may be felt by the pilot. While these symptoms are usually good indications, they are not always present; when the indications do exist, however, they should always be confirmed by the gages. The EGT gage should be the first instrument checked when a stall is suspected.

If the EGT gage reading is abnormally high, a stall probably exists. In a low-altitude stall, the RPM will also be unwinding or hung up at an intermediate value. Combined with the high EGT and unwinding or hungup RPM, will be a wide-open nozzle. The nozzle is not open due to a nozzle malfunction, but because it is attempting to reduce the overtemperature accompanying the stall; therefore, EGT, RPM, and nozzle position must always be considered to properly diagnose a stall (refer to engine AIRSTART/STALL CLEARING procedures in Section III).

Subsonic Stall

Subsonic stall normally begins with a chug or pop, followed by mild vibration. Thrust loss is immediate as evidence by rapid aircraft deceleration.

The engine gages will give the following positive indications of the stall:

1. Usually EGT will be in the 700° C to 800° C range. However, EGTs as low as 600° C may exist. If the stall persists, EGT may increase above 800° C.
2. RPM will decrease and hang up in the 70 to 85% range.

3. Nozzle will indicate 9 to 10 units as the nozzle goes to wide open in an attempt to lower EGT.

Engine response to throttle manipulation will not be normal. This simultaneously existence of high EGT, low RPM, and wide-open nozzle is conclusive proof that a stall exists. Compressor stall is easily distinguished from open nozzle failure, in which case the open nozzle is accompanied by low EGT and normal RPM response.

Engine Stall/Flameout — High Altitude Subsonic

Subsonic stall at high altitude may not be immediately recognized as it is usually accompanied by engine flameout. This stall will usually occur only if aircraft flight speed is decreased below minimum level-flight speeds at altitudes above 40000 feet, and engine transients, such as a throttle burst, afterburner light, termination or switchover, are made. If the aircraft is operated within the speed limits shown in Figure 6-6, this condition will not occur.

When operating near the minimum-speed line, use full rather than partial afterburner, since full uniform afterburning will not blow out anywhere in the steady-state maximum thrust envelope (Figure 6-6). If partial afterburner is used at extreme high altitudes and low speeds, the afterburner could blow out and the resulting engine transient could cause a stall. Since this stall is usually followed by a flameout it may be recognized by a slight bump or pop followed by silence and a sinking sensation. Engine speed will be unwinding rapidly and EGT will be low. If an airstart procedure is initiated immediately the engine may be starter before the RPM drops below 90%, thus avoiding the discomfort of possible loss of pressurization.

Supersonic Stall

Supersonic stall usually occurs only above Mach 1.8. Reasons for this type of stall include a deteriorated compressor, foreign object damage, late T_2 reset, and exceeding the CIT limit. Distortion of inlet flow, such as that caused by a gun firing, malfunction of the duct secondary air bypass flaps, or negative G may also reduce stall margin. The supersonic stall is often preceded by an intermittent muffled rumbling and aircraft yawing which coincides with the irregular rumbling. Engine gages will be normal at this time. The actual engine stall is marked by severe loud banging, accompanied by aircraft vibration and deceleration, EGT fluctuation between approximately 550° C to 700° C will occur concurrently with the banging.

Inlet Duct Stall

Inlet duct stall may occur at high Mach number and is mainly caused by side slip of the aircraft. The inlet duct stall may be recognized by a muffled rumbling. Engine instruments will be normal during the inlet duct stall. Continuation of the inlet duct stall may cause an engine stall.

RPM Hangup

If the RPM decreased below flight idle before an air start is accomplished, an RPM hangup may occur at approximately 70 to 80% following light-off, EGT will be moderate but rising abnormally (a slight high frequency vibration may also be felt). Hangup occurs because the engine minimum flow is slightly high for the high altitude and low airspeed existing at light-off. FOD or a closed exhaust nozzle will aggravate this condition. Therefore, air starts should be attempted only at lower power settings (refer to engine AIRSTART/STALL CLEARING procedure in Section III).

The RAT when extended may create a disturbance to inlet duct airflow reducing engine stall margin. Therefore, sudden attitude changes or sudden throttle movements should be avoided during this phase of flight (refer to the Flight With RAT Extended paragraph in Section III).

In the final analysis of compressor stalls the best protection is summarized as follows:

1. Knowledge of aircraft maneuvers and engine transients which may contribute to compressor stall, and knowledge of the areas of least stall margin in the aircraft envelope.
2. Knowledge of the engine symptoms which identify compressor stall.
3. Knowledge of correct stall-clearing procedures.

AERODYNAMIC VARIABLE-AREA EXHAUST NOZZLE

Two sets of flaps, operating together, form the variable exhaust nozzle. The primary exhaust nozzle flaps, hinged to the aft end of the tailpipe, control the convergent portion of the nozzle while the secondary exhaust nozzle flaps, hinged to a support ring, control the divergent portion of the nozzle. The two sets of flaps are linked together and maintain a scheduled area-and-spacing ratio which is infinitely variable between extremes. The flaps are moved automatically by four mechanically synchronized actuators. The exhaust gases leave the primary flaps at sonic velocity, and are accelerated to supersonic velocity by the controlled expansion of the gases. Control of this expansion is provided by the cushioning effect of the secondary airflow through the annular passage between the two sets of flaps.

Secondary Airflow Stream

The engine primary airstream is mainly used to support combustion. A secondary airflow stream (see Figure 1-5) is used for engine cooling and to augment the convergent effect of the exhaust nozzle. During ground operation, takeoff, and low speed flight conditions, the secondary airflow inlet is through eight sets of inward-opening doors, located aft of the firewall. With the landing gear up, secondary airflow is obtained through two bypass flaps on the leading edge of the engine.
The secondary airflow is routed externally around the engine, and in the tailpipe, passes through the space between the primary and secondary nozzle flaps, causing a mass of air to accumulate in the area. The secondary air mass prevents abrupt expansion of exhaust gases upon leaving the tailpipe, and also serves to cool and increase the velocity of the gases, thereby increasing thrust.

Emergency Pressure Relief Door

An emergency pressure relief door installed forward of the hydraulic access door on the fuselage underside is designed to separate from the fuselage when engine compartment pressure is 17 psi greater than the ambient pressure.

Nozzle Area Control System

The nozzle area control is an electrohydromechanical computer. The parameters affecting operation of the nozzle area control are throttle angle, nozzle position, electrical overtemperature signal from the temperature amplifier, and an "off-speed" signal from the engine fuel control unit. Regulated servo fuel is received from the engine fuel control unit to operate the force amplifiers in the nozzle area control. Throttle angle and exhaust gas temperature are the parameters used to "schedule" the correct nozzle area. The two parameters are combined within the nozzle area control unit.

During engine operation in the sub-Military thrust region, nozzle area is primarily a function of throttle angle. The nozzle indicator reads 8.5 to 9.5 at IDLE RPM and reads 1.0 to 3.5 as the throttle is advanced to MILITARY. However, during a rapid throttle burst, the engine fuel control unit generates an "off-speed" signal which is delivered to the nozzle area control unit. The signal overrides the mechanically "scheduled" nozzle position established by the throttle angle, and permits a rapid increase in RPM. During engine operation in the Military thrust and afterburner region it becomes necessary to modify the nozzle control, as established by throttle angle, to prevent exhaust gas temperatures from exceeding limits. Exhaust gas temperature is sensed by 12 dualloop thermocouples and the resulting signal is transmitted to a temperature amplifier. The amplifier, which receives its power from the engine-driven control alternator, compares the thermocouple signal with a present reference voltage representing the desired engine temperature. The difference voltage is amplified and transmitted to the nozzle area control unit which ultimately overrides the "mechanical schedule" and causes the nozzle flaps to move in the open direction.

Temperature Amplifier

The temperature amplifier provides an electrical signal to the nozzle area control unit. This signal overrides the "mechanical scheduled" control of the primary nozzle, preventing overtemperature operation of the engine. The amplifier receives power from the control alternator and an electrical signal, representing actual exhaust gas temperature, from the thermocouples. The signal generated by the thermocouple is compared with the signal representing desired exhaust gas temperature. The difference voltage is amplified and delivered to the torque motor of the nozzle area control.

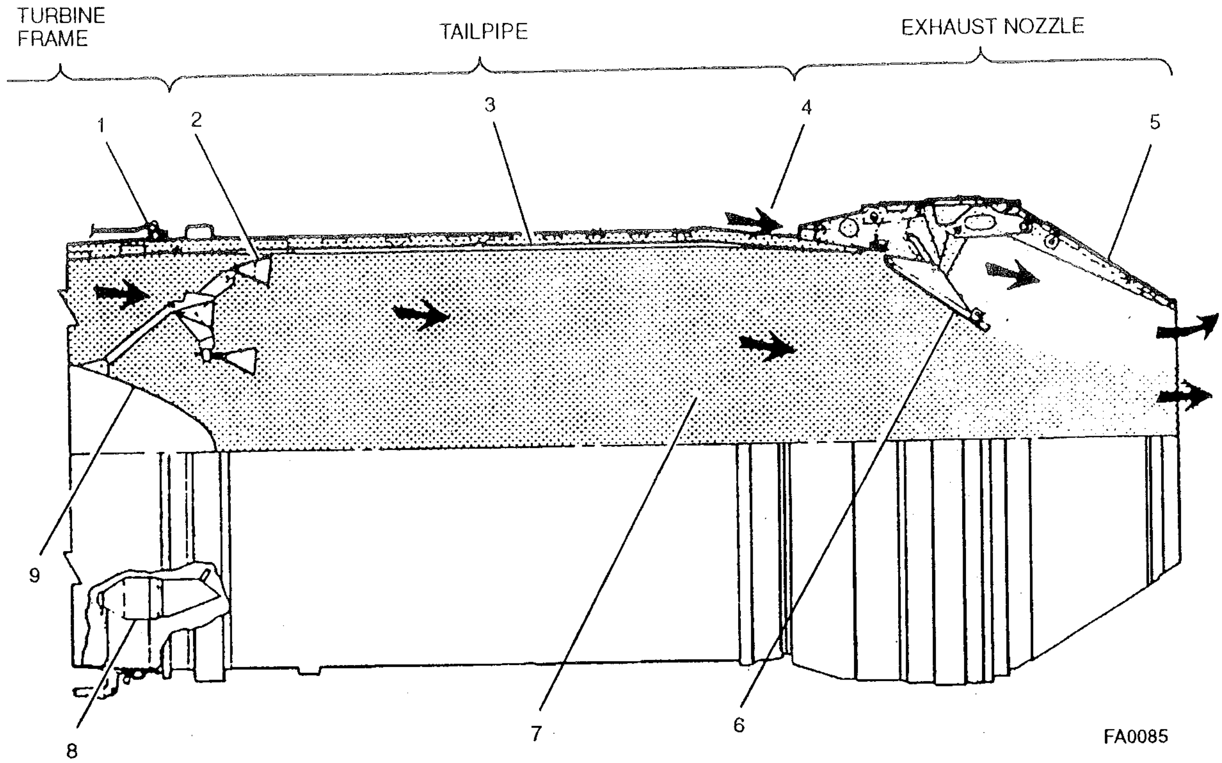
Nozzle Pump

The nozzle pump is a variable-pressure, variable-displacement pump driven by a single shaft. The amount and direction of flow are determined by a mechanical push-pull signal from the nozzle area control.

Exhaust Nozzle Flap Actuators

There are four nozzle flap actuators equally spaced on the tailpipe. The actuators are supplied with high-pressure engine oil from the nozzle pump. The

TAILPIPE AND EXHAUST NOZZLE AIRFLOW



1 9th STAGE MANIFOLD
2 FLAMEHOLDER
3 TAILPIPE LINER
4 SECONDARY AIR FLOW
5 SECONDARY NOZZLE FLAP
6 PRIMARY NOZZLE FLAP
7 COMBUSTION GAS MAIN FLOW
8 PILOT BURNER
9 REAR INNER CONE

Figure 1-5

nozzle flap actuators mechanically open and close the nozzles automatically through a series of cams and linkages.

EXHAUST NOZZLE EMERGENCY CLOSURE SYSTEM

The exhaust nozzle emergency closure system (Figure 1-6) incorporates lock assemblies, mounted on the No. 2 and 4 nozzle actuators, and a priority oil supply system. This system locks the nozzle to prevent it from exceeding the cruise position of approximately 3.0, thus preventing a full-open nozzle should the emergency nozzle closure system become inoperative as a result of an excessive oil loss.
The priority oil-supply system consists of a lube tank, an emergency hydraulic port, and a standpipe which extends within the tank to the 4-pint level. This system separates the engine lubrication system from the emergency nozzle closure system supply ports and provides a reserve supply of 4.0 pints of oil for engine lubrication.
During the normal engine operation, the primary nozzle hydraulic system regulates the exhaust nozzle exit area as a function of throttle angle and exhaust gas temperature. The emergency nozzle closure system is inoperative and the locks are in the unlocked position. In the event of a primary nozzle control system malfunction, the emergency nozzle lock system may be activated by pulling the emergency nozzle closure handle.

Emergency Nozzle Closure Handle

A T-handle labeled EMERGENCY NOZZLE CLOSURE, is installed on the lower right instrument panel in each cockpit (refer to Figure 1-1). This handle is connected by flexible cable to a transfer valve in the nozzle area control system. The transfer valve permits selecting either the normal or the emergency system for controlling the nozzle. When the emergency nozzle closure handle is in the NORMAL (forward detent) position, the transfer valve is in its normal position, allowing free flow of oil between the normal nozzle pump and the nozzle actuators; nozzle area control is then maintained by the normal control system.
Pulling the handle to the EMERGENCY (aft detent) position, mechanically positions the transfer valve to direct oil from the emergency nozzle pump to the nozzle actuators, closing the nozzle to the 1.0 to 3.0 position and actuating the nozzle locks. Some modulation of the nozzle opening will still occur as a result of changes in thrust setting (exhaust gas pressure variations).

NOTE

> The nozzle may not close under certain conditions of airspeed and altitude. If the nozzle does not close, zoom and reduce power to decrease airspeed/nozzle pressure.

ENGINE EXHAUST NOZZLE LOCKS

Nozzle locks are incorporated to prevent the nozzle from opening after the emergency nozzle closure system has been actuated. The locks are mechanical devices attached to two of the nozzle actuators and connected directly to the emergency nozzle closure handle.
When the handle is pulled, the locks assume an over center position which allows a retaining collar on the nozzle actuator to slide into the locks but not out of them until the handle is returned.
When the handle has been pulled and oil is available to the emergency nozzle pump, the nozzle will close to an area slightly smaller than the locked area. If oil to the emergency nozzle pump is depleted, the nozzle area will open slightly to the locked area. Under this condition, the nozzles cannot be unlocked from the cockpit and the handle cannot be pushed in.

Exhaust Nozzle Position Indicator

An indicator located on the right side of the instrument panel in each cockpit (refer to Figure 1-1) shows the exit area of the exhaust nozzle. Power for the instrument is derived from the XP5 bus. The instrument is placarded NOZZLE POSITION and is calibrated from 0 to 10.

Exhaust Gas Temperature Gage

An exhaust gas temperature gage, located on the right side of the main instrument panel, is calibrated from 0° to 1200° C. The unit is operated electrically by self-generating thermocouples and provides visual indications of exhaust gas temperature.

EXHAUST NOZZLE CONTROL SYSTEM

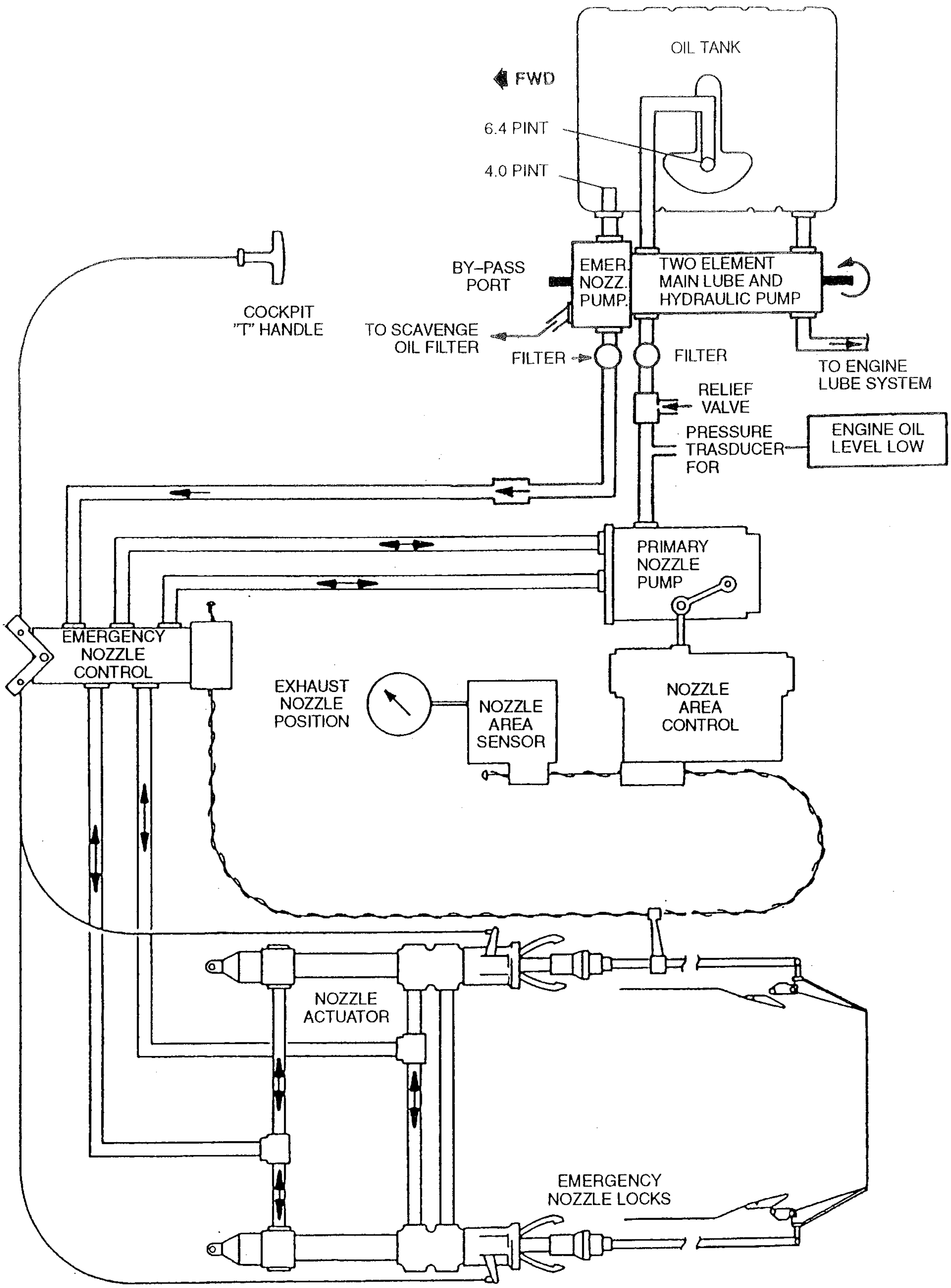
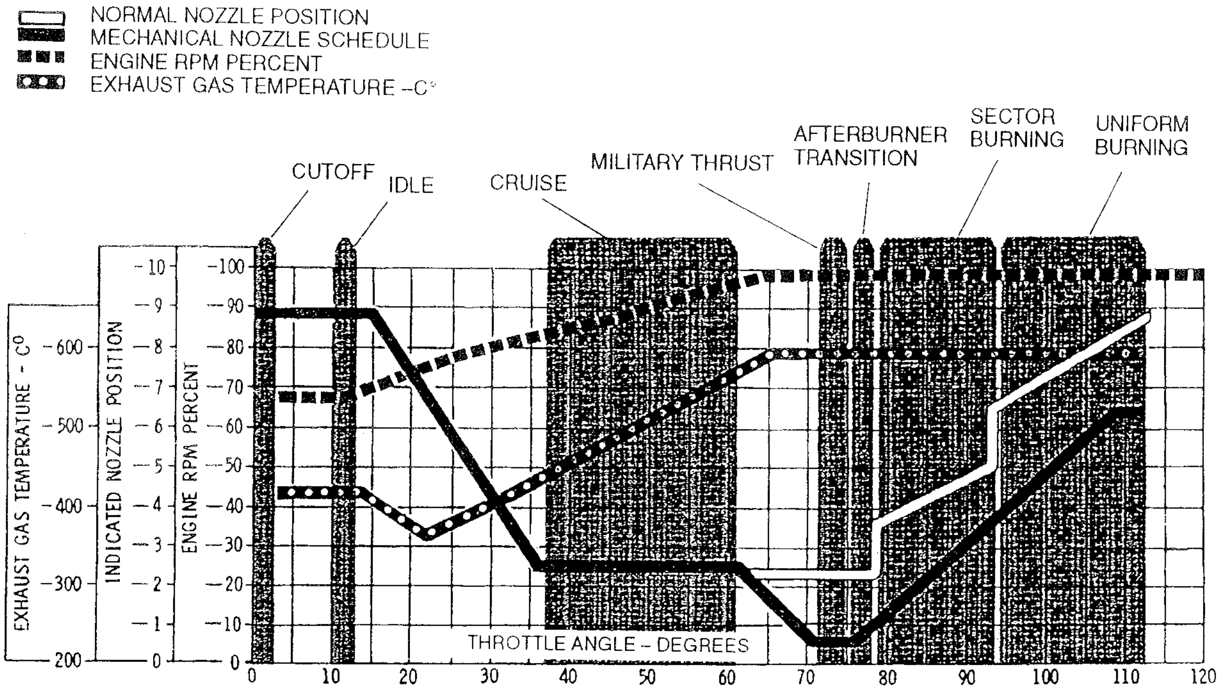


Figure 1-6

VARIATION OF ENGINE RPM, TEMPERATURE AND NOZZLE AREA WITH THROTTLE POSITION



NOTE

THE NOZZLE NORMALLY FOLLOWS A MECHANICAL SCHEDULE AS DEPICTED BY THE BLACK LINE UNTIL AN EGT OF 590° C IS REACHED. NOZZLE AREA IS THEN MODULATED BY THE TEMPERATURE AMPLIFIER TO MAINTAIN EGT RESULTING IN AN INDICATED NOZZLE POSITION VALUE GREATER THAN THE MECHANICAL SCHEDULE IN THE MILITARY AND AFTER-BURNER THRUST REGIONS AS SHOWN BY THE WHITE LINE. AT MILITARY THRUST, THE INDICATED NOZZLE POSITION MAY VARY FROM 1.0 TO 3.5 TO MAINTAIN EGT. AT MAXIMUM AFTERBURNING, THE INDICATED NOZZLE POSITION MAY VARY FROM 7.5 TO 9.5 TO MAINTAIN EGT. THE EGT IS THE MOST ACCURATE INDICATION THAT THE PROPER NOZZLE AREA IS BEING MAINTAINED. SHOULD THE NOZZLE FAIL TOWARD THE CLOSED POSITION, THE NOZZLE POSITION WILL NOT DECREASE BELOW THE MECHANICAL NOZZLE SCHEDULE LINE.

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Figure 1-7

OIL SUPPLY SYSTEM

The engine oil system (Figure 1-8) operates automatically and is a closed, dry sump, spray type system. It stores, circulates, and recovers oil used in lubricating the engine components and regulates the temperature and pressure of the oil. In addition, engine oil is used to actuate the variable exhaust nozzle system.

The oil system comprises three sections: oil supply section, scavenge section, and the pressurizing section. The oil supply section delivers filtered oil to the engine main bearing sumps and gear boxes.

The scavenge section recovers oil from the sumps, gear boxes, and from the variable nozzle system and then filters, cools, and returns the oil to the supply tank. The pressurizing section maintains the correct relationship between ambient air pressure and the air pressure within the sumps, gear boxes, and supply tank.

A positive displacement two-element main lube and hydraulic pump is provided. One element supplies pressurized oil to lubricate the engine bearings and gear boxes. The other element supplies oil to the exhaust nozzle system. The drive shaft driving the two-element main lube and hydraulic pump also powers the emergency nozzle pump. (See Exhaust Nozzle System in this Section). The oil system is serviced with 28 to 32 U.S. pints. Access to the oil quantity dipstick (calibrated in pints) is provided on the top surface of the fuselage directly over the wing.

ENGINE OIL PRESSURE

Abnormal engine oil pressure is frequently an early indication of engine malfunction. The important thing is to notice a change in the oil pressure reading

ENGINE OIL SUPPLY SYSTEM

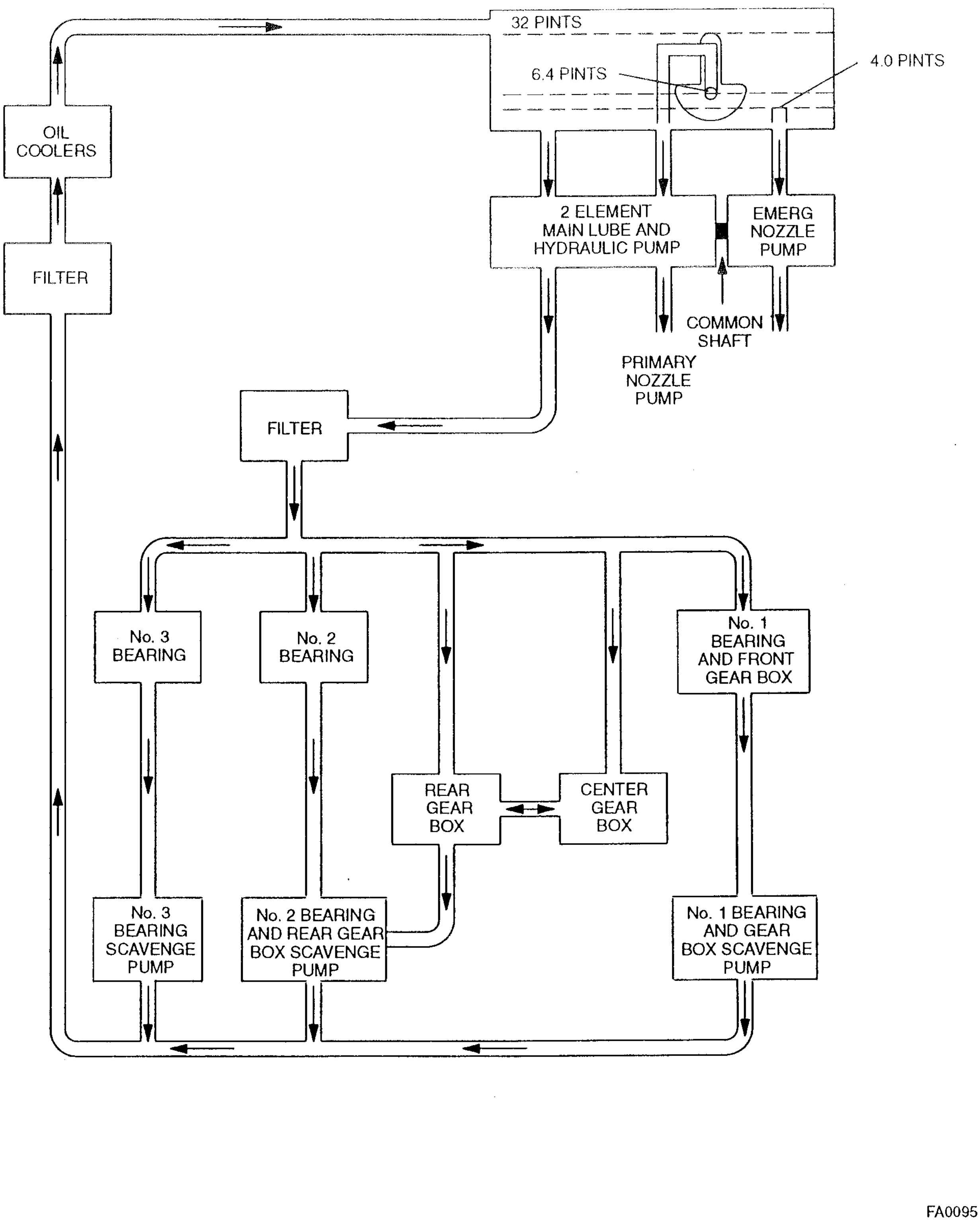


Figure 1-8

for a given RPM from the value that was considered normal prior to this time. The oil pressure gage is marked for each engine/airframe combination. The oil pressure gage markings are based on oil pressure at 100% engine RPM. During engine check at 100% RPM the indicated oil pressure should read the placard value of the oil pressure record card plus or minus 5 psi.

Oil Pressure Gage

An oil pressure gage is mounted on the right side of the upper instrument panel in each cockpit (refer to Figure 1-1). The gage registers oil pressure in pounds per square inch. The gage receives power from the XP5 bus through an autotransformer. Each oil pressure gage is marked in accordance with the oil pressure limit established for that particular engine/air-frame combination at 100% RPM.

Engine Oil Level Low Warning Light

An ENGINE OIL LEVEL LOW warning light on the warning lights panel (refer to Figure 1-1) illuminates via a transducer, located between the 2-element main lube and hydraulic pump and the primary nozzle pump. This transducer measures the oil quantity and gives an indication of the oil depletion to 6.4 pints or below. This light is powered by PP2 bus.
If the oil level in the tanks drops below the primary nozzle pump inlet line (appr. 6.4 pints remaining) loss of engine oil results, thereby illuminating the warning light.

Engine Oil Pressure Record Card

An engine oil pressure record card is located on the forward right side of the cockpit in front cockpit only. This card lists the normal engine oil pressure at 100% RPM for each engine-airframe combination.

FUEL SUPPLY SYSTEM

Refer to Figure FO-9 for fuel supply illustration. The aircraft internal fuel system consists of a forward main fuel tank, an aft main fuel tank composed of three interconnected cells, and the interconnected forward and aft auxiliary fuel cells. The forward main and aft main fuel tanks are bladder type non-selfsealing cells; the auxiliary tanks are metal and occupy the former gun bay compartment. The fuel system also consists of four submerged boost pumps mounted in the forward main fuel cell, check valves, drain valves, a main shutoff valve, a strainer and the necessary plumbing and electrical circuits. All necessary plumbing and electrical circuits are provided for the installation of tip and pylon tanks. All internal and external fuel tanks are serviced by single-point pressure refueling. The external pressure refueling switch panel is located adjacent to the refueling adapter in the left fuselage aft of the cockpit.
If pressure refueling ground equipment is not available, the internal fuel tanks may be gravity refueled through a filler well located in the aft main cell. The pylon tanks can be refueled through individual filler wells. Each tip tank may be refueled through two filler wells. A defueling disconnect is provided downstream from the strainer. Refer to Figure 1-9 and Figure 1-10 for fuel tank capacities. Refer also to Figure 1-12 for fuel grade properties and limits.

Forward Main Fuel Cell

All of the fuel that goes to the engines is fed from the forward main fuel cell. Fuel from the aft main fuel cell enters the forward main fuel cell through two flapper-type check valves. Fuel from the aft auxiliary fuel cell enters the forward main fuel cell through a flapper-type check valve by gravity flow. Fuel from the external tanks enters the aft main fuel cell through the refueling shutoff valve and then flows into the forward main fuel cell through the flapper valves.
A low-level warning switch quantity transmitter, four boost pumps, a dual fuel level pilot valve, two vent valves, and a fuel manifold are located inside the cell. Fuel used to cool the hydraulic fluid which drives the hydraulically driven fixed-frequency generator is returned to the forward main fuel tank after passing through the heat exchanger. Drain valves are located in the pump wells of the cell and are accessible from outside the aircraft.

Aft Main Fuel Cell

Three interconnected units make up the aft main fuel cell. These units are the aft center fuel cell and the aft right and left fuel cells.

FUEL QUANTITY DATA
JP-8, Jet A1 and Jet-A Fuel

Data Basis: Ground Test, Standard Day Conditions, Conversion Factor, 6.68 lb per US gal	Usable Fuel in Level Flight Attitude (1)		Fully Serviced in Static Attitude (2) (3)	
	lb	US Gal	lb	US Gal
Internal Fuel (4)	4603 ± 156	677 ± 23	4617 ± 156	670 ± 23
Tip Tanks (each)	1156	170	1190	175
Pylon Tanks (each)	1326	195	1353	199

TOTAL USABLE FUEL IN FLIGHT ATTITUDE

Internal Fuel Cells	4603 ± 156 lb	677 ± 23 US Gal
With Tip Tanks	6914 ± 156 lb	1017 ± 23 US Gal
With Pilon Tanks	7255 ± 156 lb	1067 ± 23 US Gal
With Tip and Pilon Tanks	9567 ± 156 lb	1407 ± 23 US Gal

NOTES:

(1) Level Flight Attitude — Top of Fuselage 3° Nose Up
(2) Static Attitude — Top of Fuselage 0°
(3) Fully serviced by gravity refueling or single point with gravity refueling top-off
(4) Due to structural tolerances, rubberized tank tolerances and temperature effect, internal fuel quantity may vary between 4447 and 4759 lb

Figure 1-9

FUEL QUANTITY DATA
JP-4 Fuel

Data Basis: Ground Test, Standard Day Conditions, Conversion Factor, 6.5 lb per US gal	Usable Fuel in Level Flight Attitude (1)		Fully Serviced in Static Attitude (2) (3)	
	lb	US Gal	lb	US Gal
Internal Fuel (4)	4400 ± 150	677 ± 23	4413 ± 150	679 ± 23
Tip Tanks (each)	1105	170	1137	175
Pylon Tanks (each)	1267	195	1293	199

TOTAL USABLE FUEL IN FLIGHT ATTITUDE

Internal Fuel Cells	4400 ± 150 lb	677 ± 23 US Gal
With Tip Tanks	6610 ± 150 lb	1017 ± 23 US Gal
With Pilon Tanks	6934 ± 150 lb	1067 ± 23 US Gal
With Tip and Pilon Tanks	9144 ± 150 lb	1407 ± 23 US Gal

NOTES:

(1) Level Flight Attitude — Top of Fuselage 3° Nose Up
(2) Static Attitude — Top of Fuselage 0°
(3) Fully serviced by gravity refueling or single point with gravity refueling top-off
(4) Due to structural tolerances, rubberized tank tolerances and temperature effect, internal fuel quantity may vary between 4250 and 4550 lb

Figure 1-10

NOTE

During fuel transfer and when the tip tanks empty, the internal fuel quantity may be as low as 4300 pounds in steady state level flight and possibly lower if maneuvering.

Air Pressure Shutoff Valves

Two air pressure shutoff valves are installed in the engine compartment on the left side. One valve shuts off the supply of engine compressor air to the tip tanks and the other shuts off the air to the pylon tanks. The valves are solenoid operated and controlled by a low-level float switch located in each external tank. The solenoids receive power from the PP2 bus. In case of electrical failure, the valves will fail to the open position.

Fuel Control Panels

The fuel control panels (Figure 1-11) contain the following controls and indicators.

Fuel Shut-Off Switch

The fuel shut-off switch in both cockpits (refer to Figure 1-11), is labeled ON and OFF and is guarded to the ON position. The switch may be used to actuate electrically to motordriven fuel shutoff valve, located just aft of the main fuel cell. The motor is connected through the fuel shutoff switch to the PP4 bus. The valve is used to shut off fuel to the engine in case of fire, crash landing, or ground maintenance.

NOTE

Engine flameout time may vary from 1 to 5 seconds depending on airspeed and power setting at the moment of fuel shut-off switch actuation.

External Tanks Refuel Selector Switch

The external tanks refuel selector switch, located on the left console in the front cockpit (refer to Figure 1-11), is labeled TIP, PYLON, and BOTH. It provides a method of selective refueling of the external tanks. The switch utilizes power from the PP2 bus.

Refuel Switch

The refuel switch, located on the fuel control panel in the front cockpit (refer to Figure 1-11) is labeled ON and OFF. It must be ON during pressure refueling of the external tanks. The switch utilizes power from the PP2 bus. This switch is located on the external refueling panel.

Fuel Shut-Off Open/Closed Valve Test Lights

Three green press-to-test lights are provided to check the position of the fuel shut-off switch. One test light is located in the electrical load center and labeled FUEL SHUT-OFF CLOSED VALVE TEST. This light, when pressed, will not illuminate when the fuel shut-off switch is in the open position. The other test light, located on the lower right instrument panels in both cockpits, is labeled FUEL SHUT-OFF OPEN VALVE TEST. It will illuminate green, when pressed, when the fuel shut-off switch is in the open position.

External Fuel Quantity Indicator Selector Switch

The external fuel quantity indicator selector switch, placarded EXT FUEL QTY IND SEL, is located on the lower right instrument panel in front cockpit only (refer to Figure 1-11). The two positions are TIP and PYLON. The switch is used to connect the external fuel quantity indicator to either the tip tanks or pylon tanks. The external fuel quantity indicator indicates the amount of fuel (in pounds) remaining in the respective tanks, depending on the position of the switch. This switch utilizes power from the PP2 bus.

NOTE

When war tiptanks (spare economy tanks not equipped with fuel quantity transmitters) are installed, tiptanks empty condition is indicated when a steady decrease is noted on the internal fuel quantity indicator. In addition, monitor fuel flow versus time interval for consumption of tiptank fuel.

Fuel Quantity System Test Switch

A test switch is located on the right forward panel in each cockpit (refer to Figure 1-11). When the aircraft electrical system is energized, operating the switch to the UP (FUEL QUANTITY) position will ground the system power supply from the XP7 bus. This will cause the fuel quantity gage indicating needle (both internal and external fuel quantity gage needles) to go toward zero and return to the original position when the switch is released if the system is functioning properly. This will not activate the low-level warning indication, as the systems are independent.

Internal Fuel Quantity Indicator

The internal fuel quantity indicating system indicates in pounds (0-5000) the fuel quantity remaining in all internal fuel cells.
The system consists of an indicator on the right side of the instrument panels (refer to Figure 1-11), and four fuel cell transmitters; two in the aft auxiliary cell, one in the forward main cell, and one in the aft main cell.
Operation of the indicators in both cockpits may be checked by the warning lights system and indicator test switch. The transmitter in the forward main cell is combined with a fuel density compensating unit. The system is powered by the XP5 bus.

External Fuel Quantity Indicator

The external fuel quantity indicating system indicates in pounds the fuel remaining in the tip or pylon tanks, as selected. The system consists of a dual indicator on the right side of the lower instrument panel (refer to Figure 1-11), and ten fuel cell transmitters, two in each tip tank and three in each pylon tank. The indicator needles show the fuel remaining in the left (L) and right (R) tip or pylon tanks, as selected. The system is powered by the XP7 bus.

Fuel Low-Level Warning Light

A fuel low-level warning system is installed to indicate to the pilot that the fuel level in the forward main fuel cell has decreased to a predetermined level. The system includes a float-actuated switch, installed in the forward main fuel cell, and a light on the warning panels (refer to Figure 1-11). When the fuel level falls to approximately 1275 ± 250 pounds in level flight, the switch closes the circuit and energizes the FUEL LOW LEVEL warning light.

NOTE

- Changes in aircraft attitude or acceleration may cause the FUEL LOW LEVEL warning light to illuminate when the fuel level is close to the warning actuation level. Illumination of the light should be considered as a caution indication only. As soon as possible check the fuel quantity gage indication during steady-state straight-and-level flight.
- A 10-second time-delay is installed in this circuit and prevents operation of the low-level warning light after the opening of the low-level float switch unless it remains open for 10 seconds. This prevents sloshing fuel from activating the low-level warning light.
- The fuel low-level warning light placard may be energized erroneously during sustained climbs or descents because of trapped fuel.

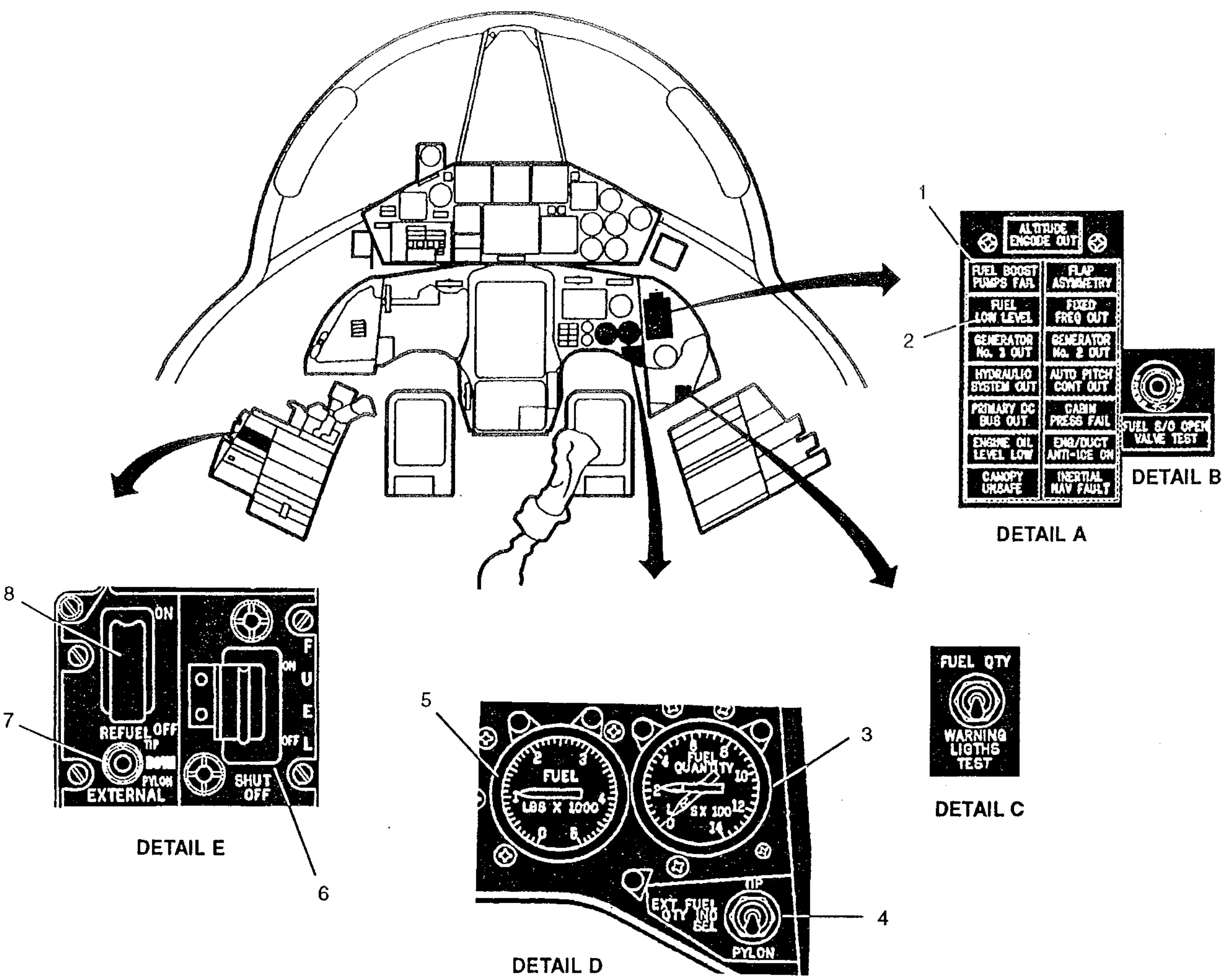
Fuel Boost Pump Failure Warning Light

A pressure switch, installed downstream from the aircraft boost pumps, monitors the fuel pressure in the lines to the engine. The switch is normally closed but opens when the fuel boost pumps are in operation. If fuel pressure at the switch falls below approximately 12 psi, the switch will close and the FUEL BOOST PUMPS FAIL warning light in each cockpit (refer to Figure 1-11), will illuminate on the warning light panel. Failure of all four boost pumps is required to illuminate the warning light.

PRESSURE REFUELING SYSTEM

The pressure refueling system makes it possible to service all internal fuel cells and external fuel tanks on the ground by single-point refueling.

FUEL CONTROL PANELS AND REFUELING SWITCH PANELS (FRONT COCKPIT)



A – WARNING LIGHTS PANEL
1 FUEL BOOST PUMPS FAIL WARNING LIGHT
2 FUEL LOW LEVEL WARNING LIGHT

B – FUEL SHUT–OFF VALVE OPEN TEST LIGHT

C – FUEL QUANTITY SWITCH

D – FUEL QUANTITY INDICATIONS
3 EXTERNAL FUEL QUANTITY INDICATOR
4 EXTERNAL FUEL QUANTITY INDICATOR SELECTOR SWITCH
5 INTERNAL FUEL QUANTITY INDICATOR

E – FUEL CONTROL PANEL
6 FUEL SHUT OFF SWITCH
7 EXTERNAL TANK REFUEL SELECTOR SWITCH
8 REFUEL SWITCH

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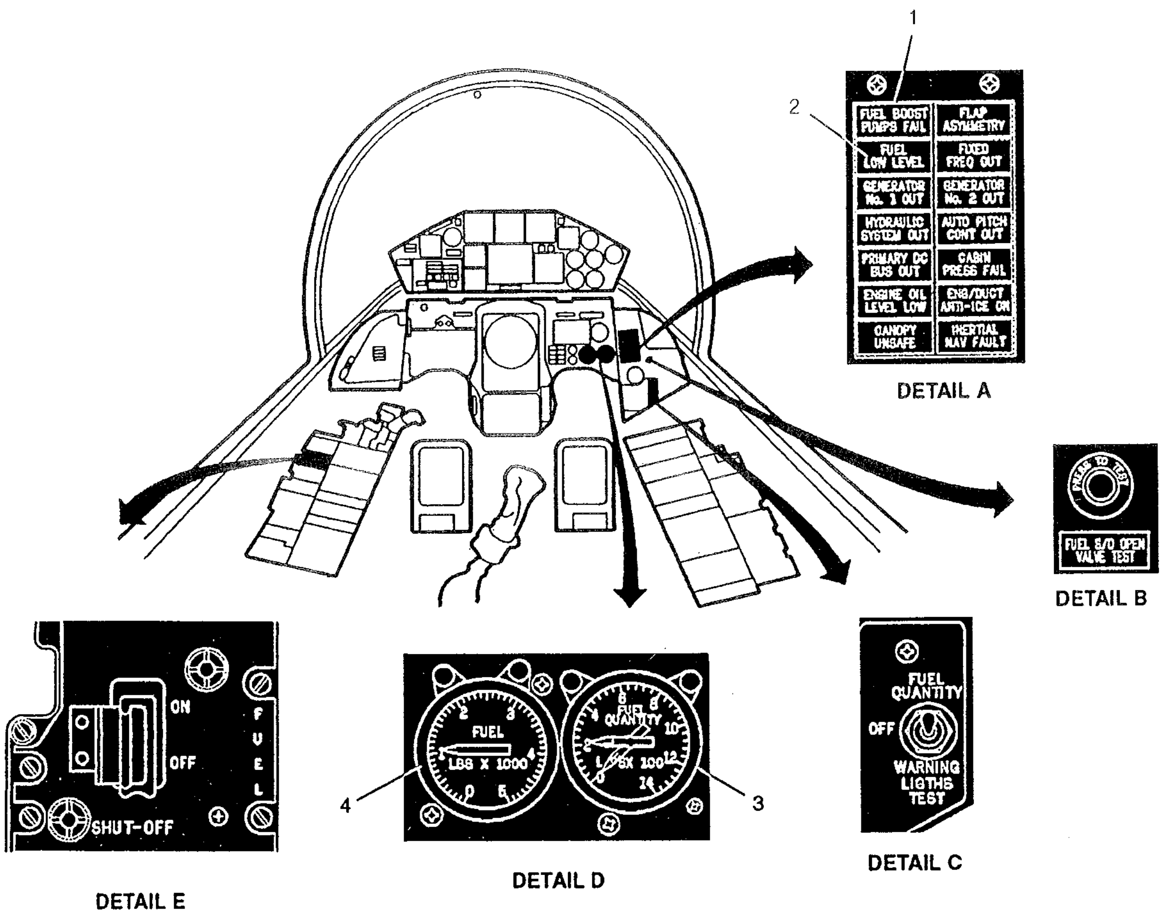
Figure 1-11 (Sheet 1 of 2)

FUEL CONTROL PANELS AND REFUELING SWITCH PANELS (REAR COCKPIT)



A – **WARNING LIGHTS PANEL**
1 FUEL BOOST PUMPS FAIL WARNING LIGHT
2 FUEL LOW LEVEL WARNING LIGHT

B – **FUEL SHUT–OFF VALVE OPEN TEST LIGHT**

C – **FUEL QUANTITY SWITCH**

D – **FUEL QUANTITY INDICATIONS**
3 EXTERNAL FUEL QUANTITY INDICATOR
4 INTERNAL FUEL QUANTITY INDICATOR

E – **FUEL SHUT OFF SWITCH**

Figure 1-11 (Sheet 2 of 2)

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FUEL GRADE PROPERTIES AND LIMITS

USE	FUEL TYPE	GRADE	NATO SYMBOL	U.S. MILITARY SPECIFICATION/ COMMERCIAL	SPECIFIC GRAVITY	FREEZE POINT		LIMITS
						°F	°C	
Primary Fuel	Kerosene	JP-8 Jet A-1	F-34 F-35	MIL-T-83133 ASTMD 1655	.840-.775 .840-.775	−58 −53	−50 −47	1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6
Alternate Fuel	Kerosene Wide Cut Gasoline	Jet-A JP-4	None F-40	ASTMD 1655 MIL-T-5624	.840-.755 .802-.751	−40 −72	−40 −58	1, 2, 3, 4, 5, 6 6
Emergency Fuel	Aviation Gasoline (Avgas)	80/87 100/130 115/145	F-12 F-18 F-22	MIL-G-5572 MIL-G-5572 MIL-G-5572	.706 .706 .706	−76 −76 −76	−60 −60 −60	1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6

LIMITS

1. Whenever the use of alternate fuel is necessary, the specific gravity setting on the main fuel control and afterburner fuel control shall be adjusted to correspond with a mid-range or average value of the specific gravity of the fuel selected. The specific gravity or average value of authorized fuels can be obtained from the above table. The procedures for changing the specific gravity setting on the fuel controls are pusblished in the aircraft power plant manuals. Whenever the specific gravity adjustments are changed from their standard preset point, an entry will be made in DD Form 781A. This entry can only be cleared when the aircraft is reserviced with the primary fuel and the specific gravity adjustments have been reset to the standard setting.
2. There is no operating time limit with alternate fuels. Use of emergency fuel is restricted to a maximum of 6 hours operation.
3. Airstarts initiated as soon as possible will assure best possible condition for restart.
4. Engine ground and aerial start times will increase when using JP-8, Jet A-1 and Jet-A fuels. Refer to Section III "Emergency Procedures" Figure 3-2 for engine estimated air start envelope.
5. Engine throttle transients, A/B light-off capability and thrust are not degraded by use of JP-8, Jet A-1 or Jet-A. A slight improvement in stall margin will result with the use of JP-8, Jet A-1 or Jet-A.
6. If there is any indication of improper fuel handling procedures, a fuel sample should be taken in a glass container and observed to fogginess, prescence of water, or rust. The primary fuels JP-8 and fuels identified by NATO symbols F-34, F-35 and F-40 contain an icing inhibitor.
7. a. Whenever the use of aviation gasoline is required, the aircraft will be restricted to a one-time flight not to exceed six hours duration. Specific gravity adjustments to the main and afterburner fuel controls are not required. Neigher is the addition of lubricating oil additives required. When using AVGAS there is no restriction on afterburner operation, but the aircraft ceiling is limited to 35000 ft and aircraft velocity shall not exceed subsonic speed at any altitude. In addition to these limitations, certain engine parameters may be degraded under some atmospheric conditions:

   (1) Longer time to start and accelerate with possible missed starts or start stalls.
   (2) Maximum engine rpm and EGT may not be attained.
   (3) Slow acceleration throughout the operating range.
   (4) Reduced engine thrust.
   (5) Reduced aircraft range.

   b. If aircraft exceeds 6 hours of operation of AVGAS, drain aircraft fuel system completely and refuel with primary fuel. Inspect turbine exhaust nozzle area and perform ground run check. If no defects or engine malfunctions are found, release aircraft for flight.

CAUTION

- AVOID FLYING AT ALTITUDES WHERE INDICATED OAT IS BELOW THE FREEZE POINT OF THE FUEL. PRIOR TO USING EMERGENCY COMMERCIAL FUEL, OBTAIN FREEZE POINT FROM VENDOR OR AIRLINE SUPPLYING THE FUEL; THEN FOLLOW THE LIMIT.
- ABOVE 22000 FEET THE TIME FOR ENGINE RESTART MAY BE 15 SECONDS LONGER WHEN USING JP-8/F-34 INSTEAD OF JP-4/F-40.

Figure 1-12

Single-Point Refueling

The internal fuel cells and external fuel tanks are normally filled by using the single-point refueling system. The internal fuel cells may be filled in about 3 minutes, the internal fuel cells and external fuel tanks in about 5 minutes. The single-point refueling receptacle is located on the left side of the fuselage, forward of the intake ducts. Cell-mounted, dual fuel level control valves and refueling shutoff valves automatically shut off fuel to the internal fuel cells as they become full. Precheck test switches are provided on the external refueling switch panel to ensure proper operation of these valves. A refueling valve is provided for each set of external fuel tanks. Float switches, located in each tank, close the respective refueling valve as soon as either tip tank or pylon tank becomes full.

Refueling Switch Panel

A refueling switch panel (Figure 1-79) located forward of and below the single-point refueling receptacle, contains only two switches: precheck test switch and fuel control power switch.

Precheck Test Switch

The master precheck switch is labelled PRI and SEC and is used to test the primary and secondary operation of the refueling shutoff valves. This switch is spring-loaded to the center OFF position. During the first few seconds of refuelling the master precheck switch should be placed first to the PRI, and then to the SEC position to check that the dual fuel level control valves close.
Satisfactory valve operation is indicated by the shutoff of fuel flow causing gradual stiffening of the refueling hose after the switch is moved to each position. A more positive indication of fuel shutoff, however, may be obtained by checking the counter on the ground refueling equipment. There is approximately a 10 second delay between the time the switch is operated and the shutoff of fuel. If fuel flow continues, pressure refueling must be stopped immediately to prevent possible fuel cell rupture or airframe damage; enter note of this fact in aircraft forms. Individual valve precheck switches are also provided for isolating faulty or leaky valves in the ammunition, auxiliary, and main cells. If necessary, the aircraft may be refueled by the gravity refueling method.

Battery Switch

The fuel control power switch is labeled BATTERY and NORMAL and is used for ground refueling. When on BATTERY position the switch is powered by PP5 bus (the Ext tank refuel circuit breaker in the junction box may be pulled), and when on NORMAL position the switch is powered by the PP2 bus (the Ext tank refuel transfer circuit breaker in the front cockpit may be pulled).

GRAVITY REFUELING

When single-point refueling cannot be employed, the aircraft may be refueled in the conventional manner. Aircraft incorporate only one filler well. Individual filler wells are provided for refueling the pylon tanks. Two filler wells are provided for each tip tank. Both must be used to fully refuel both compartments of each tip tank.

EXTERNAL STORES JETTISON SYSTEM

This system allows the emergency and selective jettison of the external stores (refer to Figure 1-13). The emergency jettison is available to the pilot by means of the:

- Emergency Stores Cutout Switch
- Emergency External Stores Jettison Button (EXT STORES JETTISON)

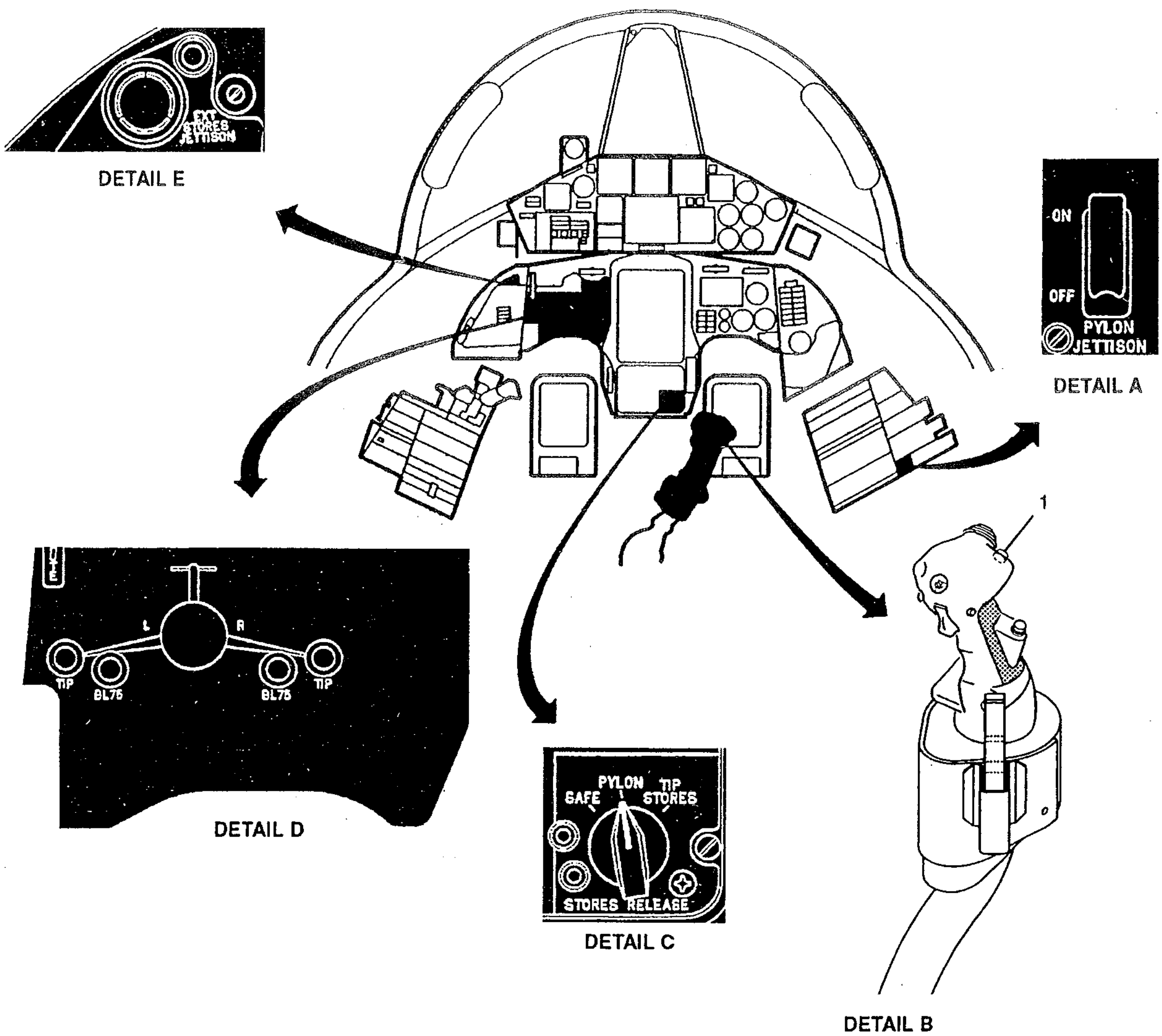
The selective jettison is available to the pilot by means of the:

- Emergency Stores Cutout Switch
- External Stores Selector Buttons and Position Indicators
- External Stores Release Selector Switch
- Droppable Stores Release Button
- Pylon Jettison Switch

Emergency Stores Cutout Switch

The emergency stores cutout switch is a two-position guarded toggle switch placarded OFF (down) and ON (up), located on the rear cockpit

EXTERNAL STORES JETTISON
(FRONT COCKPIT)



A – PYLON JETTISON SWITCH

B – CONTROL STICK GRIP
1 – DROPPABLE STORES RELEASE BUTTON

C – EXTERNAL STORES RELEASE SELECTOR SWITCH

D – EXTERNAL STORES SELECTOR BUTTONS AND POSITION INDICATORS

E – EMERGENCY EXTERNAL STORES JETTISON BUTTON

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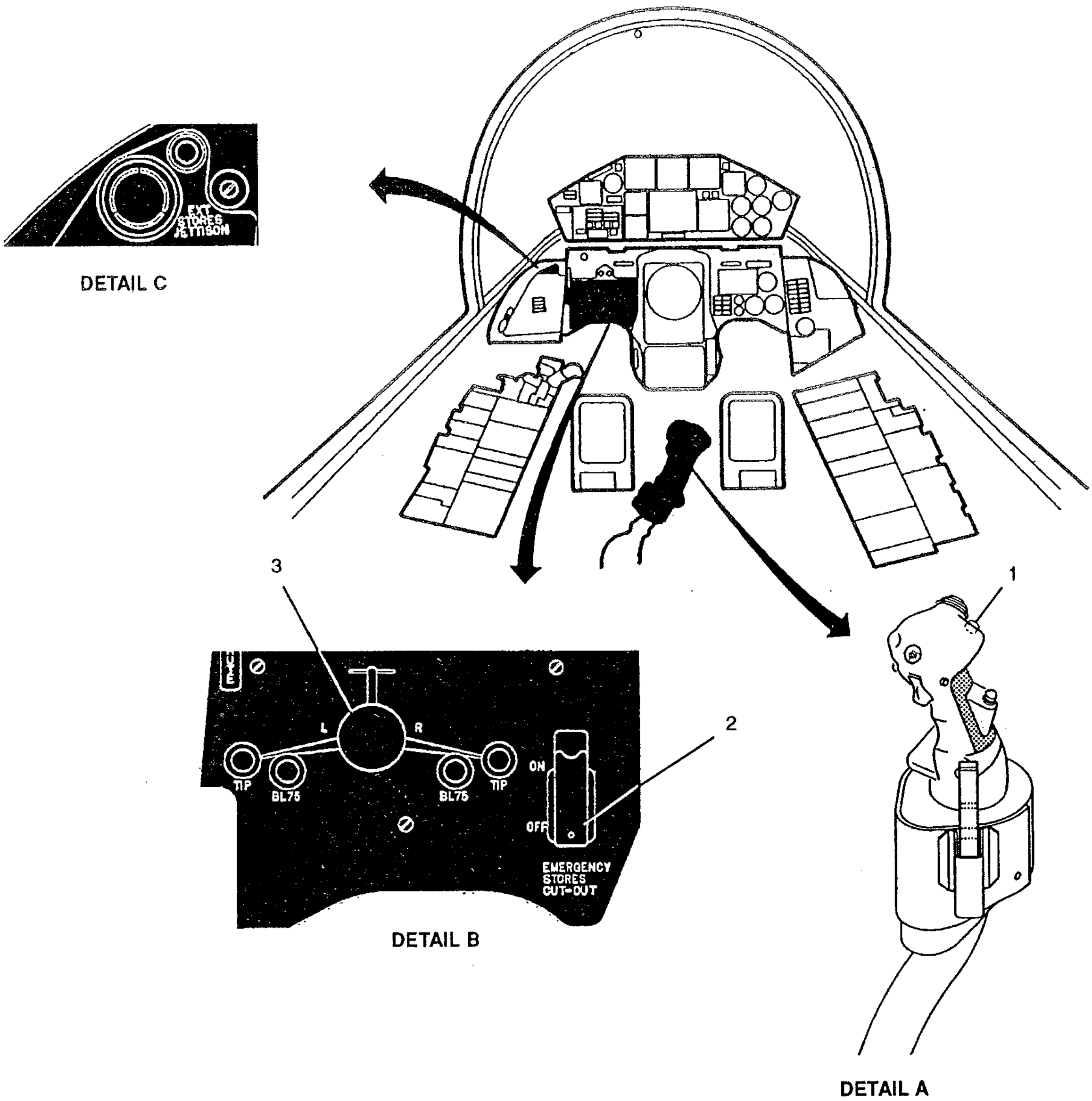
Figure 1-13 (Sheet 1 of 2)

EXTERNAL STORES JETTISON (REAR COCKPIT)



A – CONTROL STICK GRIP
1 DROPPABLE STORES RELEASE BUTTON

B – EXTERNAL STORES PANEL
2 EMERGENCY STORES CUTOUT SWITCH
3 EXTERNAL STORES POSITION INDICATOR

C – EMERGENCY EXTERNAL STORES JETTISON BUTTON

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Figure 1-13 (Sheet 2 of 2)

armament control panel (Figure 1-13). The normal position is ON to enable the stores selection by means of the external stores release selector switch (front cockpit only) and the stores jettison via the droppable stores release button (both control stick grips). This switch, when in the OFF position, prevents the selective jettison of the external stores but,. does not prevent emergency jettison of all stores. The switch is powered by the PP4 bus.

Emergency External Stores Jettison Button

The emergency external stores jettison button is located on the left main panel of both front (Figure 1-13) and rear cockpit.
The jettison circuit is connected to the PP5 bus. By pushing the button, all stores except the pylon racks will be jettisoned, independently of the position of the other switches and selectors.

External Stores Selector Buttons and Position Indicators

A combination of external stores selector buttons and position indicators are located at the external stores attachments points of the decaled rear-end view of the aircraft on both front and rear cockpit armament control panel (Figure 1-13).
There are two for the wing-tip stores and two for the pylon stores. These buttons provide individual selection for jettison of the particular store at its location. When a button is pushed in, an holding coil is energized to hold it and a green light within the button is illuminated to indicate that the external store represented by that button is selected. The button will pop out and the indicator light will extinguish when the store is released or when the external stores selector switch on the control transfer panel is set to SAFE position.
If there is not external store on the attachment point represented by the button, the holding coil will not be energized and the button will not remain in when depressed nor will the indicator light illuminate. For droppable stores the buttons are connected in series with the external stores release selector switch on the control transfer panel.

External Stores Release Selector Switch

The external stores release selector switch (Figure 1-13) is provided for the selection of the droppable stores and is located in the front cockpit control transfer panel and is powered by PP4 bus.

The position clockwise and functions are as follows:

SAFE	Disables the external stores jettison circuit.
PYLON	Select BL 75 pylon tanks for selective jettison. It is connected in series with the external stores selector buttons and the droppable store release button.
TIP STORES	Select wing-tip tanks for selective jettison. It is connected in series with the external stores selector buttons and the droppable store release button.

Droppable Stores Release Button

The droppable stores release button (Figure 1-13) is located on both control-stick grips and controls the selective jettison sequence of the fuel tanks as selected through the external stores selector buttons and the stores release selector switch.

Pylon Jettison Switch

A guarded ON-OFF pylon jettison switch is installed on the right console of the front cockpit (Figure 1-13). Use of the switch provides the only means of jettisoning the BL 75 pylon racks when desired. Power for the system is supplied by the PP5 bus.

NOTE

The BL 75 pylons shall not be jettisoned unless the stores have been jettisoned.

Emergency Jettison

By pushing the emergency external stores jettison button, located on the left main panel of both front and rear cockpit (refer to Figure 1-13), all stores, except the pylon racks, will be jettisoned.

NOTE

The emergency stores cutout switch, also when in the OFF position, does not prevent the emergency jettison of the external stores.

Selective Jettison

Selective jettison of external stores is controlled by the droppable stores release button on the control stick grip (refer to Figure 1-13).
To select the stores it is necessary first, to operate the stores release selector switch as desired on the store release panel and then, to push the appropriate external stores selector button on the armament control panel.
The selection may be accomplished from the front cockpit only. However, the position indicators of the rear cockpit will show the position of the selector buttons.
Both BL 75 pylon stores will be jettisoned if both selector buttons are pushed. Both BL 75 pylon stores will be jettisoned if both selector buttons are pushed. In any case, if the selective jettison of a single store on the BL 75 pylon is necessary, this is possible by pushing only one selector button. An attempt to jettison a single tip store by pushing only one selector button, will cause the jettison of both tip stores due to the external stores automatic drop system.
BL 75 pylon racks may be jettisoned by means of the pylon jettison switch.

NOTE

- The rear cockpit emergency stores cutout switch, when in the OFF position, prevents the selective jettison of the external stores commanded in the front cockpit.

- The BL 75 pylons shall not be jettisoned unless the stores have been jettisoned.

External Stores Automatic Drop System

A tip store automatic drop system is provided. If a tip tank should become disengaged accidentally, the system will automatically jettison the corresponding tank on the opposite wing tip, provided the PP2 bus is energized.
To prevent the automatic drop system from operating on the ground when one tank is intentionally removed, safety pins are provided for the tip tanks to disarm the system. The pins are inserted under each wing near the tip tanks.

NOTE

The automatic drop capability is not provided for the BL 75 pylon tanks.

ELECTRICAL POWER SUPPLY SYSTEM

The aircraft electrical components operate on alternating current (AC), direct current (DC), or both. AC Wild Frequency (WF) power for normal operation is provided by two engine-driven generators. For emergency operation a Fixed Frequency (FF) ram air turbine-driven generator supplies power to emergency loads. Fixed-Frequency AC is supplied by one hydraulically driven generator. Autotransformer is provided for those components requiring 26 V Fixed-Frequency AC power (XP6 Instrument bus bar).
The DC power requirements are supplied by one transformer-rectifier 120 A for normal operation and one transformer-rectifier 20 A for emergency operation.

AC ELECTRIC POWER SUPPLY

The AC electrical power (refer to Figure FO-10 and FO-11) is derived from two engine-driven wild frequency generators GEN.1 and GEN.2, one hydraulically driven fixed-frequency generator GEN.3, and an emergency ram air turbine-driven generator.

Wild Frequency Engine-Driven Generators

Two 20 KVA engine-driven generators, supplying 115/200 V, 3-phase, 320 to 522 Hz AC power, constitute the main electrical power source. Each generator is controlled by it's own Generator Control Unit (GCU1 an GCU2).
Normally, the No. 1 generator energizes the XP1 No. 1 primary and XP3 secondary AC busses and the No. 2 generator energizes the XP2 No. 2 primary AC bus and XP4 emergency AC bus. During high altitude engine air starts, the No. 1 wild frequency generator will power the No. 2 fuel boost pump.
If an overvoltage, undervoltage, underfrequency or overcurrent short circuit condition exists for either generator, that generator is automatically disconnected from its respective bus and both warning

light panels illuminate to indicate which generator is inoperative.

CAUTION

IF BOTH GENERATORS FAIL, THERE WILL BE NO WARNING LIGHT INDICATION ON BOTH PANELS IN FRONT AND REAR COCKPITS UNTIL THE RAM-AIR TURBINE IS EXTENDED.

The bus transfer system provides six possible modes of operation. The XP2 No. 2 primary AC bus also directs power to the 120 A transformer-rectifier unit TRU 1 which transforms the AC to 28 V DC to energize the PP1 primary DC, PP2 and PP3 No. 1 and No. 2 emergency DC busses.
The XP4 emergency AC bus directs power to the 20 A transformer-rectifier that provides 28 V DC to energize the PP4 and PP5 (No. 1 and No. 2 battery busses) for charge the batteries.

Wild Frequency Generator Control Unit (1 and 2)

The GCU provides control of the generator field (including voltage regulation and current limiting), generator line contactor and generator transfer contactor.
The field control function acts to maintain the system voltage within prescribed limits, and to limit the maximum phase current. It also controls the shaft power and generator heating for symmetrical and asymmetrical faults.
The generator contactor (GC1/2-G) control function, connects and disconnects the generator output from the load bus by means of the logic circuit. The generator contactor (GC1/2-T) control function, allows the load bus to be transferred to other power generating sources, whenever its own source is in fault condition or not available.
The GCU provides protection from out of limit of voltage, frequency and current to prevent damage to the bus bars loads. It ensures protection for undervoltage (UV), overvoltage (OV), underfrequency (UF) and overcurrent (OC) phenomena. The GCU provides also the short circuit and anti-cycling protection to prevent generator cycling in case of failure. Each GCU provides warning indication if a generator failure occurs.

NOTE

The electrical power supply system is equipped with GCUs which drop the two 20 KVA generators off the busses when engine RPM drops below approximately 67%. Under this condition, all electrically operated equipment, except the No. 2 boost pump and the battery busses, will be inoperative. The No. 2 boost pump will continue to operate at lower engine RPM (down to approximately 40%). This feature ensures sufficient boost pump pressure for high altitude air starts.

Generator Switches. In order to control the 20 KVA generators, two switches are provided on both cockpits (refer to Figure 1-14). On the rear cockpit the switches are configured in manned that the pilot is able to give/take the control of the generators to/from front cockpit. Therefore the rear cockpit generator switches have three function positions OFF/RESET – ON – FWD. The functions OFF/RESET and ON are used to control the generator from the rear cockpit, while the function FWD is used to transfer the control of the generator to the forward generator switches. On the front cockpit the generator switches have two function positions OFF/RESET – ON that are operative only when the correspondent rear cockpit switch is selected on FWD position.

Generator-Out Warning Lights. Two generator warning lights are provided to each cockpit (refer to Figure 1-14), the lights are identified as GENERATOR NO. 1 OUT and GENERATOR NO. 2 OUT. Each light is powered by the PP2 No. 1 emergency DC bus and will illuminate whenever the respective generator is not generating or a protection occurs.

Current Transformers

Each generator's single phase is provided with a current transformer capable of transmitting signals directly proportional to the entity of current delivered. These signals are processed by the GCUs for overcurrent protection.

Hydraulically-Driven Fixed-Frequency Generator (GEN 3). A 2.5 KVA hydraulically driven generator provides 115/200 V, 3-phase fixed-frequency (400 Hz) AC power to the XP5 and XP7 primary and secondary fixed frequency busses whenever the No. 2 hydraulic system is functioning. These busses energize those equipments requiring fixed-frequency AC power (refer to Figure FO-10 and FO-11). If both main generators fail, the secondary fixed frequency relay opens, deenergizing the XP7 secondary fixed frequency bus. If the hydraulically driven generator fails, as indicated on the warning light panels, the hydraulic generator power relay closes in transfer position, connecting variable frequency power from the XP4 emergency AC bus to XP5 and XP7 fixed frequency busses. Since the radar NASARR requires power at fixed frequency only, on this fail condition the NASARR will be automatically disconnect from the XP7 bus bar.

NOTE

The No. 2 hydraulic system pressure will be adequate to operate the hydraulically driven generator at engine windmill speeds as low as 20% RPM. Thus, the fuel flow, oil pressure, nozzle position indicator and hydraulic pressure indicators will be operative to provide the pilot with the capability of monitoring the engine relight.

Fixed-Frequency Reset Button. This button is installed on the right forward panel in the front cockpit only (refer to Figure 1-14) and has two functions:

On Ground:

- after engine start, pressing for almost 5 seconds, it permits the starting of hydraulically driven generator GEN 3. The FIX FREQ OUT warning light extinguishes.

In Flight or On Ground:

- if a GEN 3 fails (FIXED FREQ OUT warning light lit in both cockpits) is possible to reset the electrical fixed-frequency power system, pressing it for almost 5 seconds

The button is powered through the PP2 No. 1 emergency DC bus.

Fixed-Frequency-Out Warning Lights. These lights, located on the warning light panel on both cockpits (refer to Figure 1-14), are energized by the PP2 No. 1 emergency DC bus. The lights will illuminate whenever the hydraulically driven generator GEN 3 is out. In this case the XP4 emergency AC bus is energizing the XP5, XP6 and XP7 fixed-frequency AC busses.

Emergency AC Power Supply

The aircraft is equipped with an extendable ram air turbine (RAT) which drives an emergency hydraulic pump and a 4.5 KVA generator that supplies 115/200 V, 3-phase (400 Hz), AC power for emergency operation. Once extended, the ram air turbine cannot be retracted in flight. If both No. 1 and No. 2 engine driven generators fail, the ram air turbine-driven generator (when extended) will energize the XP4 emergency AC bus. In turn, if the hydraulically-driven generator is not operating, the RAT energizes the XP5 primary fixed-frequency AC bus, in addition to both PP2, PP3 emergency DC busses and both PP4, PP5 battery busses, through the 20 A transformer-rectifier.

NOTE

The XP4 emergency AC bus will not operate the XP5 primary fixed-frequency AC bus if the hydraulically driven generator is operating.

Ram Air Turbine (RAT) Extension Handles. Emergency AC power is made available by extending the RAT into the aircraft slipstream. This may be done by pulling the yellow RAT extension handles (refer to Figure 1-14), located on the right side of both instrument panels. The handles require a firm pull of about 4 inches to the stop to extend the RAT.

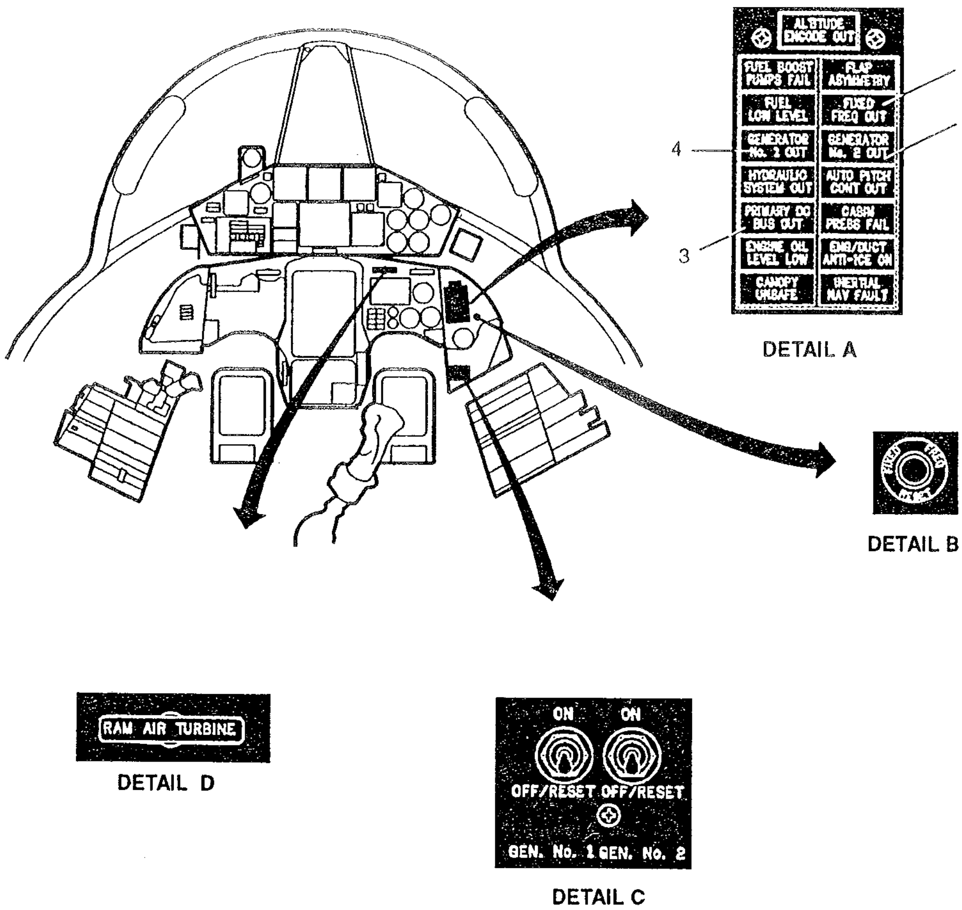
NOTE

There is no means of retracting the RAT in flight.

DC ELECTRIC POWER SUPPLY

The direct current power (refer to Figure FO-10 and FO-11) requirements of the aircraft are normally supplied from the XP2 No. 2 primary AC bus through a 120 A transformer-rectifier unit TRU1. This transforms 115/200 V AC to 28 V DC, which powered the PP1 primary DC bus. PP1 supplies power to the equipment as per Figure FO-10 and FO-11. The PP2 and PP3 No. 1 and No. 2 emergency DC busses, are also connected to PP1 pri-

ELECTRICAL POWER SUPPLY SYSTEM CONTROL AND INDICATORS (FRONT COCKPIT)



A – **WARNING LIGHTS PANEL**
1 FIXED FREQUENCY OUT WARNING LIGHT
2 GENERATOR NO. 2 OUT WARNING LIGHT
3 PRIMARY DC BUS OUT WARNING LIGHT
4 GENERATOR No. 1 OUT WARNING LIGHT

B – **FIXED FREQUENCY RESET BUTTON**

C – **GENERATOR SWITCH**

D – **RAM AIR TURBINE EXTENSION HANDLE**

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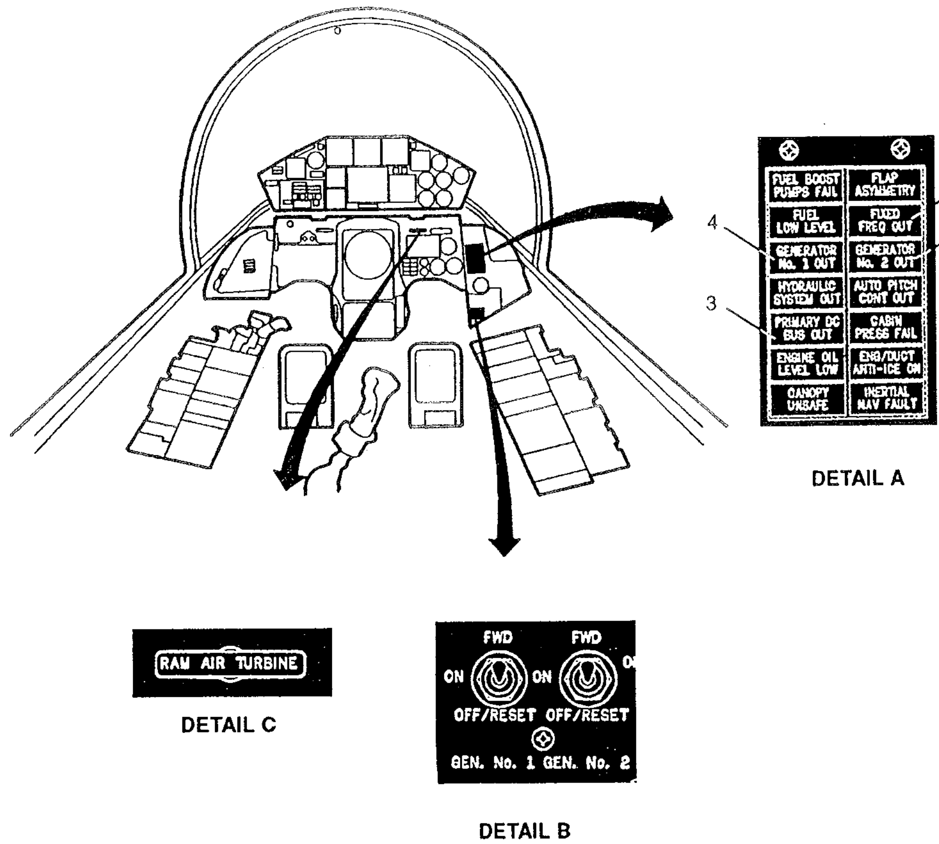
Figure 1-14 (Sheet 1 of 2)

ELECTRICAL POWER SUPPLY SYSTEM CONTROL AND INDICATORS (REAR COCKPIT)



A – **WARNING LIGHTS PANEL**
1 FIXED FREQUENCY OUT WARNING LIGHT
2 GENERATOR NO. 2 OUT WARNING LIGHT
3 PRIMARY DC BUS OUT WARNING LIGHT
4 GENERATOR No. 1 OUT WARNING LIGHT

B – **GENERATOR SWITCH**

C – **RAM AIR TURBINE EXTENSION HANDLE**

FA0120

Figure 1-14 (Sheet 2 of 2)

mary DC bus during normal operation. The PP2 and PP3 No. 1 and No. 2 emergency DC busses, supply power to units which are considered essential for safe operation of the aircraft. Due to this requirement, an alternate source of power to these busses is provided in the event that power from the PP1 primary DC bus is not available. Under this condition, the PP2 and PP3 No. 1 and No. 2 emergency DC busses, will be connected automatically to the 20 A transformer-rectifier unit TRU 2 which is connected to the XP4 emergency AC bus.

CAUTION

NO WARNING LIGHT INDICATION IS PROVIDED IF PP2 NO. 1 EMERGENCY DC BUS IS NOT ENERGIZED.

NOTE

When the ram-air turbine-driven AC generator is operative following electrically emergency condition, it is important that the load on the emergency AC bus be minimized when using the aircraft leading and trailing edge flaps since they are powered directly from the XP4 emergency AC bus. To reduce loads and ensure maximum flap effectiveness, the PP2 No. 1 emergency DC bus is automatically disconnected from the 20 A transformer-rectifier unit TRU 2 as long as the flaps are in operation, and those units which are powered from this bus, including the UHF command radio, will be inoperative during the period of flap operation. It should be noted that the Emergency Radio, powered by PP4 No. 1 Battery bus, remains operating.

BATTERIES

Two 3.6 Ah Nickel Cadmium Batteries provide power to the PP4 and PP5 No. 1 and 2 battery busses for the following main functions:

- Engine relight
- Emergency UHF radio
- External emergency store release
- Arrestor hook

The batteries and battery busses are fed by the 20 A transformer-rectifier unit TRU 2. The batteries output is prevented from discharging to the PP2 and PP3 by blocking rectifiers. There is no battery control switch in the cockpit.

Primary DC Bus-Out Warning Light

The PRIMARY DC BUS OUT warning lights are located on the warning light panels of both cockpits (refer to Figure 1-14), and are energized by the PP2 No. 1 emergency DC bus. The lights will illuminate whenever power to the PP1 primary DC bus is not available.
Refer to Figure FO-10 and FO-11 for units which will be inoperative when the primary DC bus-out warning lights are lit.

CIRCUIT BREAKERS

The circuit breaker panels (refer to Figure FO-8) on the left and right consoles of both cockpits contain push-to-reset, pullout-type breakers for certain AC and DC circuits. All the distribution circuits in the electrical system are protected by various types of circuit breakers.
Circuit breaker panels which are not accessible during flight but which should be inspected before, are located in the electronic compartment behind the cockpit and in the electrical load center on the right side of the fuselage.

CAUTION

CIRCUIT BREAKERS SHOULD NOT BE PULLED OR RESET WITHOUT A THOROUGH UNDERSTANDING OF ALL THE CONSEQUENCES. PULLING CIRCUIT BREAKERS MAY ELIMINATE FROM THE SYSTEM SOME RELATED WARNING SYSTEM, INTERLOCKING CIRCUIT, OR CANCELING SIGNAL, WHICH COULD RESULT IN AN UNDESIRABLE REACTION.

EXTERNAL POWER SUPPLY

The aircraft is equipped with a receptacle (refer to Figure 1-79) for connecting an external AC power source to the electrical system. This receptacle is located on the lower right side of the fuselage and is accessible through a door above the hydraulic panel. Each main aircraft's generator has priority on the external power. The external power is controlled and monitored by means of the GCU No. 1. The EXT PWR RESET button shall be pressed to power the aircraft: the EXT PWR OUT indication light shall extinguish.

HYDRAULIC POWER SUPPLY SYSTEMS

Refer to FO-12 and FO-13 for illustration of hydraulic systems. Two hydraulic systems (No. 1 and No. 2) and an emergency system provide power to the various hydraulically operated units in the aircraft. The No. 1 and No. 2 systems function simultaneously during all normal operations and supply fluid at 3000 psi pressure to their respective hydraulically operated units. The No. 1 and No. 2 systems are provided with separate reservoirs, differing only in size and location; the No. 2 reservoir has the larger capacity. Each system includes an engine-driven pump, a cylindrical accumulator, a pressure transmitter, a pressure switch, and filters. In case of failure of either No. 1 and No. 2 system, the remaining system will maintain fluid pressure for flight control but at a reduced rate. If both No. 1 and No. 2 systems fail, and sufficient fluid is available in the No. 1 system, the emergency ram-air turbine-driven pump will furnish enough fluid pressure through the No. 1 system for operating flight controls at a reduced rate.

NOTE

When hydraulic systems are pressurized, hydraulic system interflow will occur across the return bends of the valve spools in the servo control valves due to normally higher return pressures in the No. 2 system, resulting in flow of fluid from No. 2 system into No. 1 system. However, if No. 2 system is lost flow of fluid will be from No. 1 system into No. 2 system and will result in loss of fluid of the No. 1 hydraulic reservoir.

Accumulators

The cylindrical accumulators are charged with nitrogen at approximately 1000 psi and are provided with an air valve and an air pressure gage. The accumulators store a supply of high-pressure fluid and also act as surge chambers.
The accumulators and pressure gages for both No. 1 and No. 2 systems are accessible upon opening a large engine access door on the underside of the fuselage below the engine.
Graduations on the gage dial are in increments of 100, from 0 to 5000 psi. The pressure gage shows the initial nitrogen charge (1000 psi) in the accumulators only when hydraulic pressure is zero.

Hydraulic Panel

Most of the hydraulic units are mounted directly on a hydraulic panel on the inside face on the engine access door. Upon opening the door the various units are exposed for servicing, for testing, and for checking quantity indicators.

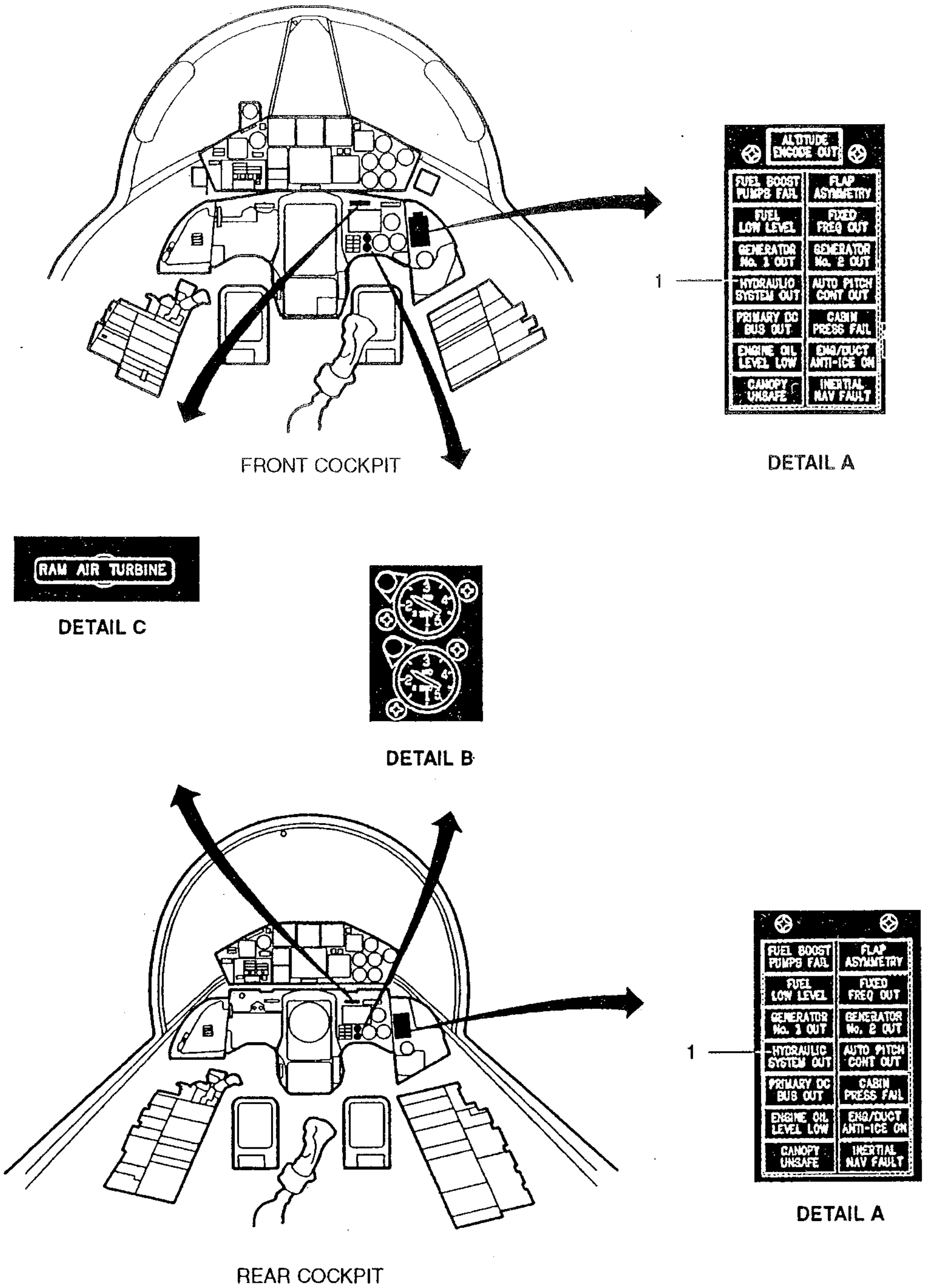
Ground Test Selector Valve

The manually controlled ground test selector valve on the hydraulic panel is the only link between the No. 1 and No. 2 systems. A three-position lever extends from the top of the valve and a mechanical linkage from this lever to a fixed bracket inside the fuselage causes the lever to be operated to the No. 2 position and locked in place when the engine access door is closed.

NO. 1 HYDRAULIC POWER SUPPLY SYSTEM

The No. 1 hydraulic system supplies fluid (under regulated pressure) to the flight controls exclusively. Power is supplied to the stabilizer aft cylinder, the five inboard cylinders for each aileron, the bottom cylinder of each rudder actuator, the yaw damping control valve, and the automatic pitch actuator. The system includes a reservoir, an engine-driven pump, a cylindrical accumulator, a pressure transmitter, a pressure switch, a pressure-regulating flow-control valve, and a filter.
Fluid is supplied to the pump from a piston-type reservoir. Fluid from the pump is supplied under 3000 psi pressure directly to the fight control components. The pressure regulating flow-control valve

HYDRAULIC SYSTEMS CONTROLS AND INDICATORS



A – WARNING LIGHTS PANEL
1 HYDRAULIC SYSTEM OUT WARNING LIGHT

B – HYDRAULIC SYSTEMS PRESSURE GAGES

C – RAM AIR TURBINE EXTENSION HANDLE

FA0091

Figure 1-15

is connected to the pressure line. This valve contains a relief valve which relieves excessive system pressures to the return line.
A pressure switch in the No. 1 system energizes the warning light on the warning light panel when the pressure drops below 1250 psi and sends a signal to the APC system.

NO. 2 HYDRAULIC POWER SUPPLY SYSTEM

The No. 2 hydraulic system supplies fluid under regulated pressure to the flight controls, pitch and roll damper control valves, hydraulically driven generators, landing gear, antiskid brake system, nosewheel steering, engine air bypass flaps, and speed brakes.
The system includes a reservoir, an engine-driven pump, a cylindrical accumulator, a pressure transmitter, a pressure switch, a relief valve, a priority valve, and four filters. Fluid is supplied to the pump from a piston type reservoir. Fluid from the pump is supplied under 3000 psi pressure, through a filter directly to the accumulator for each aileron, the stabilizer forward cylinder, and the top cylinder of each rudder actuator.
A line connected to the pressure line immediately downstream from the filter feeds through a restrictor valve to the pressure switch and pressure transmitter. Another pressure line connected to the outlet port of the filter is routed to the pressure relief valve and the priority valve. The pressure relief valve will relieve system pressure to the return line if excessive pressure occurs in the system. The priority valve opens to full flow at 2600 psi. A pressure line from the priority valve outlet port carries fluid to the utility hydraulic system which includes the engine air bypass flaps, landing gear system, anti-skid brake system, nosewheel steering system, and speed brake selector valve. The priority valve reseats to zero flow when system pressure drops to 2175 psi, thus retaining for flight controls and the hydraulically driven generator all system pressure below this range. A pressure switch in the No. 2 system energizes the warning light on the warning light panel when the pressure drops below 1250 psi.

Hydraulic Systems Pressure Gages

The hydraulic systems pressure gages located on the right side of the lower instrument panel, provide a visual indication of the pressure available in the hydraulic systems. The gages receive power from the XP6 bus. The gage dials are calibrated from 0 to 4000 psi in increments of 500.

Hydraulic System Out Warning Light

The HYDRAULIC SYSTEM OUT warning light, located on the warning light panel, illuminates when pressure in either the No. 1 or No. 2 hydraulic system decreases to approximately 1250 psi. The hydraulic pressure indicating system may be used to determine which system is out.

WARNING

THE HYDRAULIC SYSTEM OUT WARNING LIGHT WILL NOT INDICATE A SECOND FAILURE. THE REMAINING GAGE MUST BE MONITORED TO DETERMINE IF SUBSEQUENT FAILURE OCCURS.

EMERGENCY HYDRAULIC POWER SUPPLY SYSTEM

The emergency hydraulic system comprises a pump supplied with fluid from the No. 1 system reservoir. It delivers hydraulic pressure to the No. 1 system through the pressure regulating flow-control valve. The pump is a constant-volume gear type, powered by the RAT.
The pressure regulating flow-control valve diverts emergency pump fluid to return until the RAT (and pump) has reached operating speed. Thus, a hydraulic load cannot be imposed on the turbine before it has reached a speed high enough to handle the demand. With the turbine and pump operating at the proper speed, fluid is then fed upon demand to the No. 1 system.

NOTE

- In addition to furnishing emergency hydraulic power, the RAT will furnish emergency electrical power if necessary.
- Once extended, the RAT cannot be retracted in flight.

Ram Air Turbine Handle

A yellow handle, located on the right side of the lower instrument panel, may be used to extend the RAT, which powers the emergency hydraulic pump. The handle is labeled RAM AIR TURBINE and requires a firm pull of about 4 inches to the stop to extend the turbine. With the RAT extended into the airstream, the emergency hydraulic pump will supply pressure, through the No. 1 system, for operation (at about one-sixth the normal rate) of the various hydraulic units normally operated by the No. 1 system.

FLIGHT CONTROL SYSTEM

Flight controls consist of conventional cable and pushrod systems, mechanical and electrohydraulic servosystems, electrical trim systems, electrical control systems, and hydraulic control systems. The primary flight control surfaces include the ailerons, a pivoted one-piece controllable horizontal stabilizer, and a rudder.

FULL POWER IRREVERSIBLE CONTROL SYSTEM

The ailerons, horizontal stabilizer, and rudder depend upon a complete hydraulic power control system for operation. Movement of the controls in any direction, even to the slightest degree, immediately activates a servomechanism which immediately responds and directs hydraulic pressure to the control surface control cylinders to move the control surfaces in the required direction.
As soon as the control surfaces begin to move, a follow-up linkage begins to cancel the original control signal to stop the control surfaces at the required deflection.
When the required deflection of the control surfaces is reached, they are hydraulically locked in that position by the actuating cylinders and cannot be moved by the action of external aerodynamic forces.

Artificial Feel System

The use of a full power, irreversible control system for actuating the flight controls prevents air loads and resulting feel from reaching the cockpit controls. Therefore, an artificial feel system is installed to provide a sense of control feel under all flight conditions. Normal control forces are simulated by a system of cams and centering springs. This system applies loads to the controls in proportion to the degree of control deflection and proportionally to the number of G's in the case of the stabilizer control system.

CONTROL STICK

The control stick is mechanically connected (by means of control cables and pushrods) to hydraulic control valves at the ailerons and to horizontal stabilizer control valves.
Movement of the stick positions these control valves so that power from the flight control hydraulic system is directed to the control surface actuators to move the control surfaces. A follow-up system automatically closes off the flow of hydraulic fluid to the actuators when the desired control surface deflection is reached.
The control-stick grip (Figure 1-17) incorporates the primary aileron and horizontal stabilizer trim switch, and the nosewheel steering /microphone button, trigger switch (inoperative), droppable stores release button, radar action reject button and APC emergency disconnect switch.

RUDDER PEDALS

Primary control for the rudder consists of conventional rudder pedals, mechanically connected to a hydraulic control valve at the rudder hydraulic actuator. Movement of the rudder pedals positions the valve so that power from the flight control hydraulic systems is directed to the control surface actuators to move the rudder. A follow-up system automatically closes off the flow of hydraulic fluid to the actuators when the desired rudder deflection is obtained.
The rudder pedals may be adjusted by the rudder pedal adjustment handle, labeled PEDAL ADJ and located to the left of the center control panel. The wheel brakes are applied conventionally by toe action on the rudder pedals (refer to Wheel Brake System paragraph). Rudder pedal movement also controls the nosewheel steering (refer to Nosewheel Steering System paragraph).

AILERON AND RUDDER TRAVEL LIMITING SYSTEM

Aileron and rudder travel is limited automatically when the gear is UP, and unlimited when the gear is DOWN. With gear DOWN, aileron travel is approximately $\pm 20°$ and with gear UP, aileron travel is approximately $\pm 10°$. An additional 5° of aileron travel may be obtained with gear UP by using aileron trim.

With gear DOWN, rudder travel is 20° (±2°) to either side of neutral and with the gear UP, rudder travel is limited to 6° either side of neutral. An additional 4° of rudder travel is available with gear UP if rudder trim is used. Aileron and rudder limiting circuit is powered by PP2.

If electrical failure occurs, rudder and aileron travel will not be limited. Pulling the RUD AIL LIMIT CONT circuit breaker on the left console in the rear cockpit will remove the rudder and aileron travel limiters. The same effect will be obtained in the front cockpit left console by setting the AIL RUD LIMITER cover guarded switch in UNLTD position. This switch normally is set in LTD position (cover guard down); (refer to Section VI "Flight Characteristics" for additional information on this system).

Aileron and rudder travel is automatically unlimited when gear is DOWN or when gear is UP and a flap asymmetry condition exists as detected by the flap asymmetry detection system. When the aileron and rudder travel is unlimited, the AIL AND RUD UNLIMITED warning light illuminates on the main instrument panel in both cockpits.

WARNING

THE USE OF FULL UNLIMITED AILERON OR RUDDER TRAVEL IN MANEUVERS AT SPEEDS ABOVE 300 KIAS CAN RESULT IN STRUCTURAL DAMAGE AND POSSIBLE LOSS OF THE AIRCRAFT.

Aileron and Rudder Unlimited Warning Light

The AIL AND RUD UNLIMITED warning light on the left side of the instrument panel (refer to Figure FO-4 and FO-6) will illuminate when the aileron and rudder travel is unlimited. This condition exists when gear is down or when the gear is up and the flap asymmetry has been detected by the flap asymmetry detection system.

STABILITY AUGMENTATION SYSTEM

Aircraft dynamic response and handling characteristics are greatly improved through the use of a three-axis stability augmentation system. Electrical power is supplied from XP5 bus. The system measures the rate-of-change of aircraft attitude and generates an electrically amplified signal. This signal moves a system of valves which in turn direct hydraulic pressure to the actuating cylinders to move the rudder, stabilizer, or ailerons to a position relative to the amount of correction necessary. This operation does not move the normal surface control linkage or has any effect upon cockpit controls.

The stability augmentation system also includes a "washout" circuit which allows the pilot to execute maneuvers without interference by the stability augmentation devices. The stability augmentation system causes the control surfaces to be deflected to correct for rapid small disturbances. The washout circuit cancels these signals in favor of pilot-initiated signals. In order to decrease the possibility of excessive pitch changes with resultant high negative-G forces, there is no washout circuit incorporated in the pitch axis.

Roll, Pitch, and Yaw Damper Switches

Three guarded switches (refer to Figure 1-16), placarded STABILITY CONT, ROLL, PITCH and YAW are located on the left console in each cockpit. These switches are guarded in the ON position but may be used to disconnect the stability augmentation control system in any one or all three axes, whenever required, by operating the switches to the OFF position. Any one or any two of the systems may be disconnected without adversely affecting stability augmentation control of the remaining system. (Refer to Section VI "Flight Characteristics" for additional information on this system).

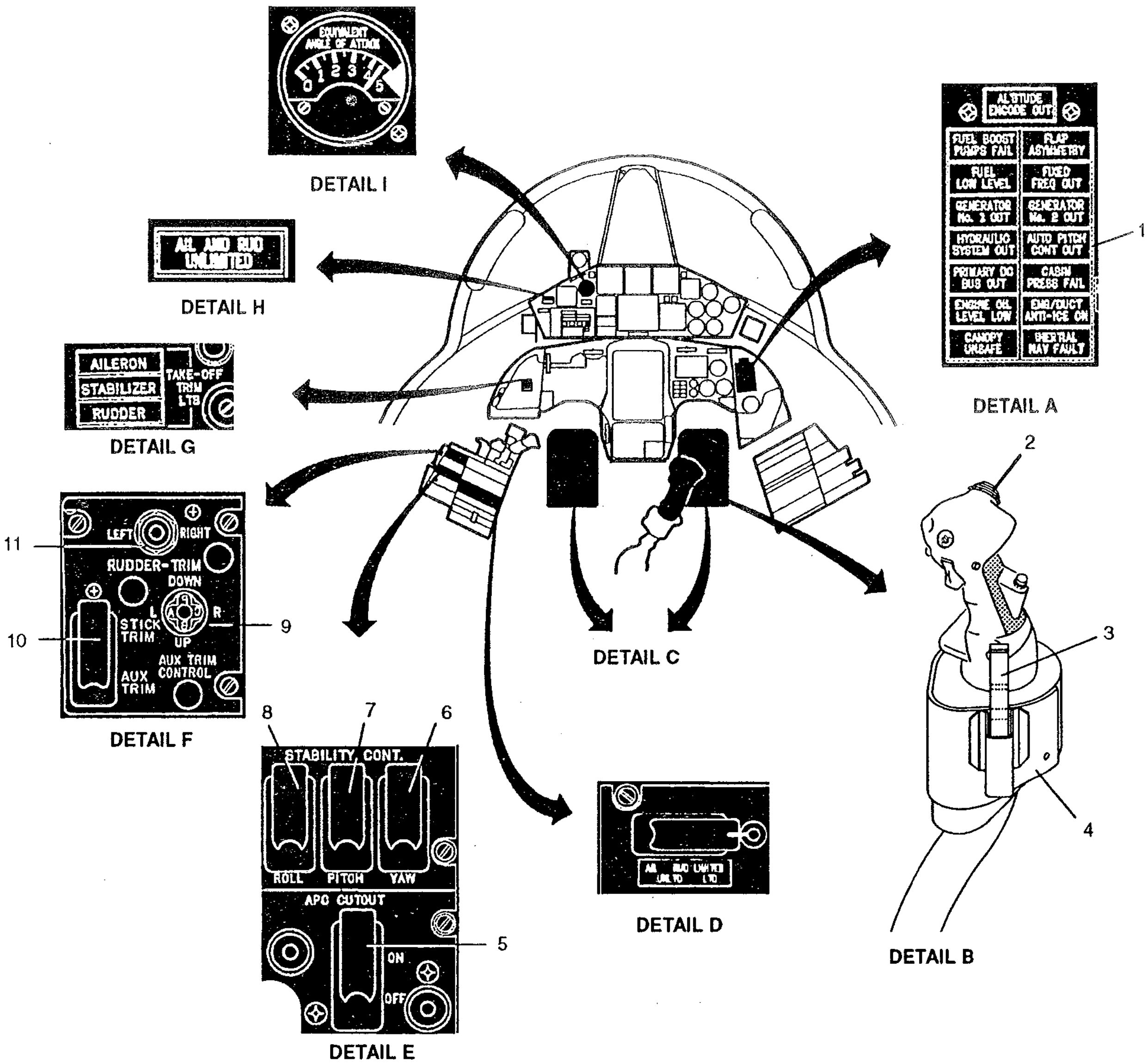
NOTE

The front and rear cockpit switches must both be ON to obtain system operation.

AUTOMATIC PITCH CONTROL SYSTEM

In the automatic pitch control system (APC) an automatic pitch actuator provides artificial stall warnings and prevents inadvertent stall by moving the control stick approximately ½ inch forward of neutral when either the pitch velocity or the angle of attack, or a combination of both, reaches a critical value.

FLIGHT SYSTEM CONTROLS AND INDICATORS (FRONT COCKPIT)



A – **WARNING LIGHTS PANEL**
1 AUTO PITCH CONT OUT WARNING LIGHT

B – **CONTROL STICK GRIP**
2 AILERON AND HORIZONTAL STABILIZER TRIM SWITCH
3 APC DISENGAGE SWITCH
4 STICK SHAKER ASSEMBLY

C – **RUDDER PEDALS**

D – **AIR RUD LIMITER SWITCH**

E – **STABILITY CONTROL PANEL**
5 APC CUT OUT SWITCH
6 YAW DAMPER SWITCH
7 PITCH DAMPER SWITCH
8 ROLL DAMPER SWITCH

F – **TRIM CONTROL PANEL**
9 AUXILIARY TRIM CONTROL PANEL
10 AUXILIARY TRIM SELECTOR SWITCH
11 RUDDER TRIM SWITCH

G – **TAKEOFF TRIM INDICATOR LIGHTS**

H – **AIL AND RUD UNLIMITED WARNING LIGHT**

I – **APC SYSTEM INDICATOR**

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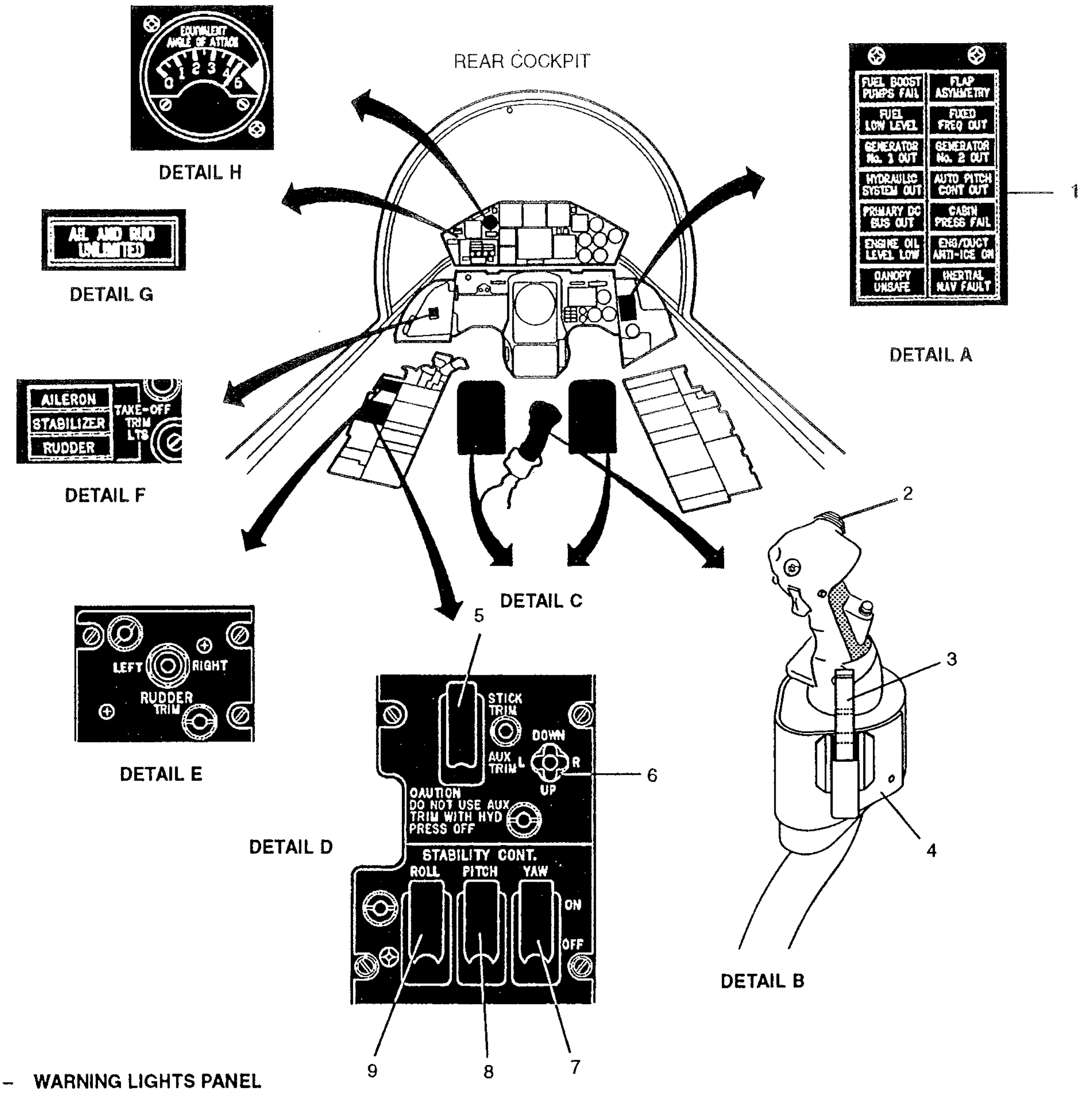
Figure 1-16 (Sheet 1 of 2)

FLIGHT SYSTEM CONTROLS AND INDICATORS (REAR COCKPIT)



A – WARNING LIGHTS PANEL
1 AUTO PITCH CONT OUT WARNING LIGHT

B – CONTROL STICK GRIP
2 AILERON AND HORIZONTAL STABILIZER TRIM SWITCH
3 APC DISENGAGE SWITCH
4 STICK SHAKER ASSEMBLY

C – RUDDER PEDALS

D – STABILITY AND TRIM CONTROL PANELS
5 AUXILIARY TRIM SELECTOR SWITCH
6 AUXILIARY TRIM CONTROL SWITCH
7 YAW DAMPER SWITCH
8 PITCH DAMPER SWITCH
9 ROLL DAMPER SWITCH

E – RUDDER TRIM SWITCH

F – TAKEOFF TRIM INDICATOR LIGHTS

G – AIL AND RUD UNLIMITED WARNING LIGHT

H – APC SYSTEM INDICATOR

FA0118

Figure 1-16 (Sheet 2 of 2)

Signals to do this come from a rate gyro and a vane-actuated, angle of attack detecting system. These signals are amplified and when a critical value is reached, they trigger a solenoid valve to operate a small hydraulic cylinder, which causes the stabilizer to assume an aircraft nosedown deflection and the control stick to move forward. This is the only control surface deflection that is transmitted back through the control system to the control stick. Thus, the pilot is immediately made aware of an approaching stall attitude of the aircraft. A force of 40 to 50 pounds, in addition to normal control forces, applied on the stick, can override the APC system.

NOTE

When the stick is more than ½ inch forward of neutral, the kicker will still operate; however, the kicker action will not be felt and will have no effect on the stabilizer.

Electrical power for the APC system is supplied by the XP5 bus. Hydraulic power is supplied from the No. 1 hydraulic system (refer to Section VI "Flight Characteristics" for additional information on this system).

NOTE

- Momentary shaker operation may be experienced during engine startup when the fixed-frequency generator becomes operative at approximately 20% RPM. This indicates normal operation and does not require corrective action.
- A failure of, or the deactivation of, the pitch damper will be felt as a sudden trim change. An intermittent failure or the reactivation of the pitch damper will result in a small rapid stick movement.
- With the APC system deactivated, the pilot cannot predict the onset of pitch-up during rapid maneuvering flight. Pitch-up may be hazardous at any altitude and at low altitudes it may be impossible to recover from pitch-up in time to prevent an accident. Also, the dynamic motion of the aircraft during pitch-up may make it difficult to reach the seat ejection ring. Therefore, unless the system is malfunctioning, flying with the APC system deactivated is not recommended.
- LOCAL OPERATION OPTIONAL If local operating procedures dictate flying with the APC system deactivated, avoid rapid maneuvers during pull-outs or turns which induce high pitch rates. Stay out of the stick shaker boundary as there is no way of knowing how far the boundary has been penetrated until pitch-up occurs. If the stick shaker boundary is penetrated inadvertently, immediately relieve the G-load and increase power if necessary.

Effect of Flap Lever and Gear Position on APC Operation

The stick shaker is operative in all gear and flap positions. The kicker is inoperative either with gear down or flap lever in LAND position.

NOTE

When flaps are lowered to TAKEOFF position, the angle at which the angle-of-attack sensing vane energizes the stick shaker or the kicker is automatically increased, thereby permitting aircraft maneuvering to a higher angle of attack than with flaps UP.

CONTROL STICK GRIP

1 AILERON AND HORIZONTAL STABILIZER TRIM SWITCH
2 DROPPABLE STORES RELEASE BUTTON
3 TRIGGER SWITCH (INOPERATIVE)
4 RADAR ACTION REJECT BUTTON
5 NOSEWHEEL STEERING/MICROPHONE BUTTON
6 APC EMERGENCY DISCONNECT SWITCH
7 STICK SHAKER ASSEMBLY

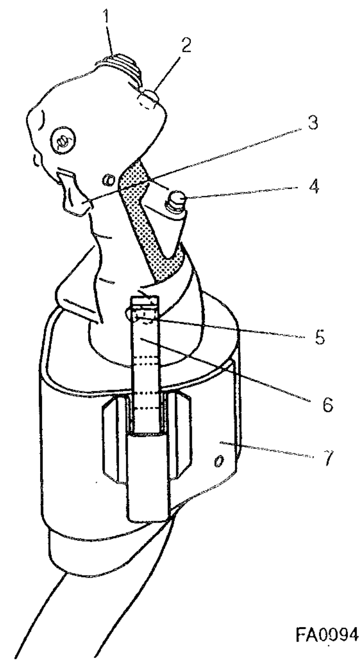


Figure 1-17

WARNING

THE ANGLE AT WHICH THE ANGLE-OF-ATTACK SENSING VANES ENERGIZE THE SHAKER OR KICKER IS DEPENDENT UPON THE WING FLAP LEVER SETTING AND NOT THE ACTUAL POSITION OF THE FLAPS. IN NORMAL OPERATING CONDITION AFTER WING FLAP LEVER AS BEEN SELECTED FROM UP TO TAKEOFF (APC BOUNDARY VARIATION) AND BEFORE SAFE TAKEOFF INDICATION IS ACHIEVED A TIME INTERVAL OF 7/8 SECONDS EXISTS. THEREFORE INCREASING THE LOAD FACTOR (AOA) DURING THIS TRANSIENT CAN CAUSE A PITCH-UP WITH NO PREVIOUS SHAKER OR KICKER OPERATION.

NOTE

Normally a single APC system malfunction will not result in a kick if the landing gear is down or wing flap lever is in LAND position.

Stick Shaker Stall Warning

A control stick shaker stall warning has been incorporated in the APC system. The shaker will operate when pitching velocity or angle-of-attack, or a combination of both, reach a value which is slightly less than that which the APC system requires for kicker actuation.
When energized, electrical power from the XP4 bus activates an eccentric motor on the control stick which agitates it in a rotary motion. This shaking is a warning that a stall condition is imminent. Stick shaking will commence before the automatic pitch kicker is actuated. (Refer to Section VI "Flight Characteristics" for additional information on this system).

If an inadvertent stick shaker and kicker are encountered, after overriding and padding off the kicker, the APC cutout switch should be actuated prior to pulling the Stick Shaker Circuit Breaker.

NOTE

- When the flaps are lowered to TAKEOFF or LAND position the angle at which the angle-of-attack sensing vane energizes the stick-shaker motor is automatically increased, thereby permitting operation at a higher angle of attack than with flaps UP.
- No switch is provided to deactivate the stick shaker; however, a circuit breaker (refer to Figure FO-8) on the left console can be used to deactivate the shaker if necessary.

WARNING

DEACTIVATING THE STICK SHAKER BY PULLING THE STICK SHAKER CIRCUIT BREAKER ILLUMINATES THE AUTO-PITCH CONT OUT WARNING LIGHT; HOWEVER, THE APC KICKER REMAINS OPERATIVE. IF IT IS NECESSARY TO PULL THE STICK SHAKER CIRCUIT BREAKER IN FLIGHT, OBSERVE THE EMERGENCY PROCEDURES UNDER STICK SHAKER FAILURE IN SECTION III SINCE THERE WILL BE NO FURTHER VISUAL INDICATION OF AN APC KICKER FAILURE.

CAUTION

IF THE STICK SHAKER MOTOR IS OPERATED CONTINUOUSLY FOR LONGER THAN SIX MINUTES, DAMAGE MAY RESULT FROM OVERHEATING.

APC Cutout Switch

An APC kicker cutout switch is located on the left console of forward cockpit and is guarded in the ON position. The OFF position may be used to deenergize the APC kicker if necessary. Power for the switch is received from the XP5 bus. When the switch is in the OFF position, the AUTO-PITCH CONT OUT warning light illuminates in both cockpits. An APC emergency disconnect switch is installed on the control stick approximately below the grip.
For emergency disconnect of the APC kicker, activate the emergency APC disconnect switch and hold. If permanent deactivation of the APC kicker is desired, the emergency switch should be held in the activated position until the APC cutout switch is turned to the OFF position.

NOTE

The APC system cutout switch does not deactivate the stick shaker.

Automatic Pitch Control System Indicator

An automatic pitch control (APC) system indicator is located on the upper left side of the main instrument panel. The indicator (which is actuated by the combination of the APC rate gyro and the right vane) may be used throughout the flight to ascertain that the APC system is operating, as well as to inform the pilot of the aircraft's relation to the stall condition. The indicator dial is graduated from 0 to 5 with a red area over 5, the point at which the APC kicker becomes effective.

NOTE

The indication is still available even if the APC cutout switch or the APC emergency disconnect switch have been operated.

Automatic Pitch Control Out Warning Light

The AUTO-PITCH CONT OUT warning light on both warning panels will illuminate if the APC system malfunctions or the No. 1 hydraulic system fails.

This light is provided to warn the pilot that more attention will be required to prevent a stall by avoiding higher accelerations and low airspeed.

WARNING

WHEN THE AUTO-PITCH CONTROL OUT WARNING LIGHT IS ILLUMINATED, THE PILOT MUST EXERCISE CARE TO PREVENT STALLS BY AVOIDING HIGH ANGLES-OF-ATTACK AND LOW AIRSPEEDS.

NOTE

- The AUTO-PITCH CONT OUT warning lights and CAUTION lights may flicker during ground checks or change of wing flap lever position. This condition is normal provided the lights do not stay illuminated.

- When the aircraft power supply is switched from ground power 28-volt dc source to the aircraft 28-volt dc source, the AUTO-PITCH CONT OUT lights may flash on then off, without turning on the master caution light. This condition is normal and no corrective action is necessary.

TRIM CONTROL SYSTEM

Aileron, Stabilizer, and Rudder

The aileron trim actuators are mechanically connected to the trim motor by flexible driveshaft and provide electrical trim of the control surfaces by movement of the servovalve assembly input linkage arms. The stabilizer and rudder trim actuators are directly connected to the servovalve assembly input linkage arms.

Movement of the input linkage arms causes deflection of the control surfaces to a trimmed position but does not move cockpit controls. The trim motors are powered by PP2 bus.

The aileron actuator contains cam-actuated, up-and-down limit switches. The stabilizer and rudder trim actuators do not have limit switches, but utilize nonjamming mechanical stops to limit the travel.

CAUTION

- DO NOT USE PRIMARY OR AUXILIARY TRIM CONTROL WITHOUT HYDRAULIC PRESSURE AS THIS MAY DAMAGE THE TRIM MOTOR.

- CONTINUED OPERATION, IN SAME DIRECTION AFTER REACHING EXTREME LIMIT OF TRIM TRAVEL, SHOULD BE AVOIDED TO PREVENT POSSIBLE OVERLOADING OR BURN-OUT OF TRIM MOTOR.

Control Stick Aileron and Stabilizer Trim Switches

Normally, roll and pitch trim control is provided by a spring-loaded, thumb-actuated switch located on top of the control-stick grip (Figure 1-17). Movement of the switch to the left causes a left aileron up, right aileron down operation of the trim motor and trim actuators. Switch movement to the right causes a reverse operation. Forward movement of the switch causes a stabilizer leading edge up (aircraft nose down) operation of the trim motor and actuator. Aft movement causes reverse operation. The switch is powered by PP2 bus.

Auxiliary Trim Selector Switch

A two-position, guarded trim selector switch (Figure 1-17) is located on the left console in each cockpit and is labeled STICK TRIM and AUX. TRIM. If failure of the control stick trim switch occurs, the selector switch allows use of the auxiliary trim switch for control of the stabilizer and aileron trim circuits. The switch is powered by PP2 bus.

Auxiliary Trim Control Switch

A spring-loaded toggle switch (Figure 1-17) is located on the left console. This switch produces the same effects as the control stick trim switch, provided the auxiliary trim selector switch is in the AUX TRIM position. The switch is labeled AUX TRIM CONTROL and functions as an auxiliary, or standby switch, used if the control-stick trim switch fails. The switch receives power from PP2 bus.

NOTE

The front and rear cockpit trim switches may be used independently; however, primary trim control is in the rear cockpit and will override any trim control operation in the front cockpit.

Rudder Trim Switch

Directional trim control is provided by a three-position toggle switch (Figure 1-17) located on the left console in each cockpit. The switch is spring-loaded to the center position from the LEFT and RIGHT positions and is used to control PP2 bus powered trim motor to position the rudder in a trimmed position.

Stabilizer, Aileron, and Rudder Takeoff Trim Indicator Lights

Three trim indicator lights, located on the left forward panel, are provided to indicate takeoff trim position of the flight control surfaces. When illuminated, the lights read AILERON, STABILIZER and RUDDER. With the aircraft on the ground, the trim light will illuminate whenever the trim motors are run through the take-off trim position by operating the trim switch, and will extinguish when the switch is released.

CAUTION

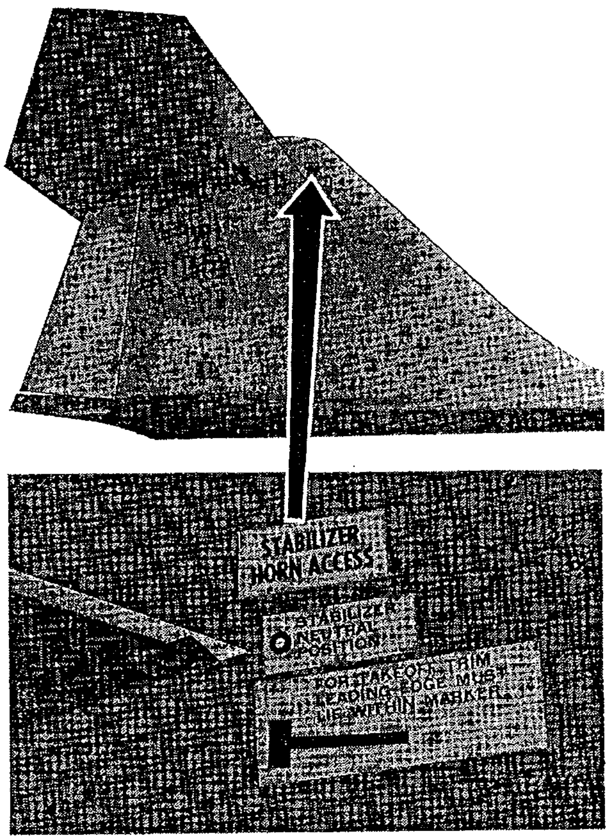
DO NOT USE PRIMARY OR AUXILIARY TRIM CONTROL WITHOUT HYDRAULIC PRESSURE AS THIS MAY DAMAGE THE TRIM MOTORS.

The AILERON trim light will operate both on the ground and airborne. The STABILIZER trim light will remain illuminated (on the ground only) when the stabilizer takeoff trim is set, and the trim switch is released. When airborne, operation of the stabilizer and rudder trim lights are precluded by the actuation of the air-ground safety switch. The three lights may be tested by operating the warning light system test switch. Power for the lights is provided by the PP2 bus.

Stabilizer Trim Marker

A black "T" (Figure 1-18) painted on right side of the vertical stabilizer is used as a takeoff trim index. When trim is set for takeoff, the leading edge of the horizontal stabilizer should be aligned with the index. The pilot should obtain ground confirmation that the stabilizer is within the correct tolerance.

STABILIZER TRIM MARKER



FA0096

Figure 1-18

WING FLAP SYSTEM

The wing flap system comprises a set of trailing edge flaps and a set of leading edge flaps. The trailing edge flaps are attached to the aft beam of each wing panel, between the wing fillets and the inboard end of the ailerons and are hinged at the forward lower edge. The leading edge flaps form the leading edge of each wing, between the fuselage and the wing-tip fairings; they are hinged at the aft lower edge.
Both sets of flaps are actuated when the flap lever is operated. Each set is electrically connected by a control circuit, and mechanically connected by flexible driveshafts: the trailing edge and leading edge flaps are only electrically interconnected by the control circuit. Each set of flaps is operated by two ac motor powered actuators. One motor is capable of operating a set of flaps, at a reduced rate, if the other is inoperative.
However, when only one actuator is functional, it is possible to overload the actuator motor causing the actuator to disengage from the flap gear train. The flaps will stop in an intermediate extended position, and a continuous barber pole indication will appear on the flap position indicator. The actuator motor will continue to run under this condition unless the flap lever is returned to the prior position (thus reversing the motor). See Flap Failure procedures in Section III. A leading edge flap lock is incorporated in the flap system to lock the flaps up. Each leading edge flap is provided with a locking assembly and lock actuator, the left flap lock and actuator being driven by the PP3 bus powered motor, and the right flap lock actuator by the left lock actuator through an interconnecting flexible driveshaft.
A boundary layer control system is automatically operated when the wing flaps are in the landing configuration. Whenever the RAT is extended for electrical power, and the wing flap lever is moved to the TAKEOFF position, a wing flap sequencing system operates automatically to prevent electrical power from going to the leading edge flaps until the trailing edge flaps have moved to TAKEOFF position. This sequence reduces the electrical demands on the RAT-driven generator under emergency conditions. The control circuit for the flaps receives power from the PP3 bus and the actuating power is from the emergency ac bus. During sequenced flap travel, the PP2 bus is deenergized.

Trailing Edge Asymmetry Detection System

The trailing edge wing flap asymmetry detection system consists of a cam-actuated switch assembly mounted on a plate which is driven by the right-hand flap. The left-hand flap drives a disc cam. During symmetrical flap travel, the switch roller rides in the cam detent. In the event of asymmetrical flap travel in excess of 3-½° ±1°, the switch roller rides out of the detent causing the switch to open the trailing edge flap control circuit, stopping further flap travel. Thus, trailing edge flaps will not exceed approximately 3-½° ±1° of asymmetry; therefore, a safe landing can be accomplished.
If trailing edge flap travel has been stopped by the asymmetry detector, thus preventing further operation of the trailing edge flaps, the landing must be accomplished in whatever trailing edge flap configuration existed at the time asymmetry occurred. Detection of an asymmetrical condition of the trailing edge flaps will cause illumination of the FLAP ASYMMETRY warning lights in both cockpits.
The rudder and aileron limiters will disengage at the time of asymmetry condition and the AIL AND RUD UNLIMITED warning lights illuminate in both cockpits. If flaps return to a symmetrical condition, the flaps will move to the selected position, the limiters will engage, the FLAP ASYMMETRY and the AIL AND RUD UNLIMITED warning lights will extinguish in both cockpits.

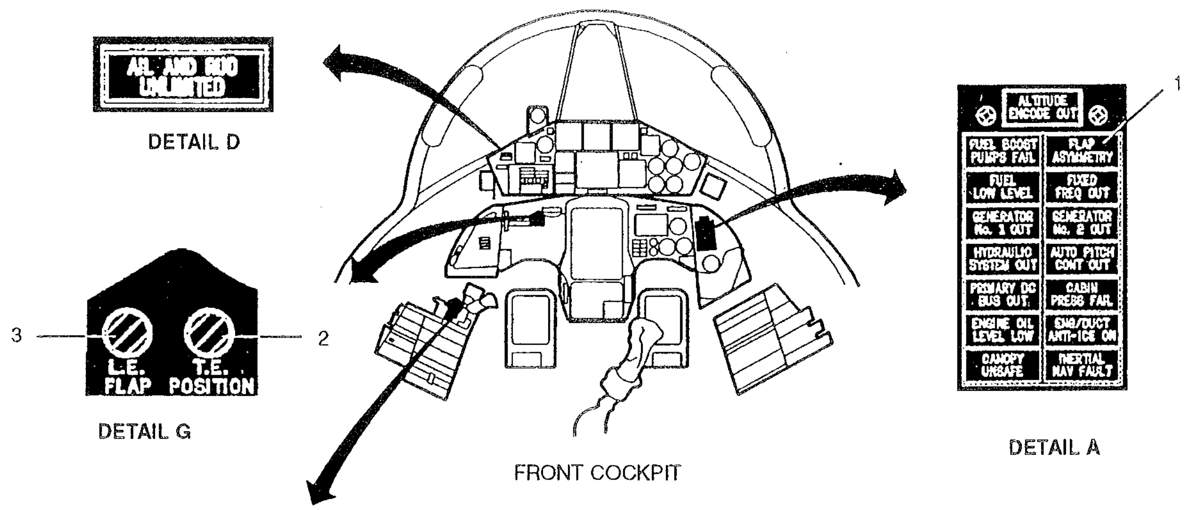
NOTE

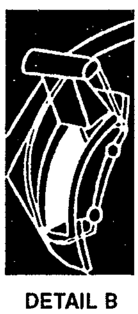
> The trailing edge flap asymmetry detection system will not detect asymmetry nor affect operation of the leading edge flaps.

WING FLAP LEVER

The wing flap lever is located immediately to the left of the throttle. The lever position are UP, TAKEOFF, and LAND. A spring-loaded guard prevents inadvertent pulling of the lever from the UP position. Both leading and trailing edge flaps are controlled by the flap lever. Selection of the TAKEOFF position will extend the leading edge flaps and trailing edge flaps 15° from faired.
In the LAND position, the leading edge flaps extend 30° from faired and the trailing edge flaps extend 45° from faired. In the UP position both sets of flaps will retract to the UP (faired) position however, the position indicator will not indicate UP until the leading edge flaps are fully retracted and locked.
When moving the flap lever from the LAND to the UP position, the lever will latch at the TAKEOFF position. In order to release the latch, the lever must

WING SYSTEM CONTROLS AND INDICATORS





A – **WARNING LIGHTS PANEL**
1 FLAP ASYMMETRY WARNING LIGHT

B – **WING FLAPS LEVER**

C – **WING FLAPS POSITION INDICATORS**
2 TRAILING EDGE FLAPS
3 LEADING EDGE FLAPS

D – **AIL AND RUD UNLIMITED WARNING LIGHT**

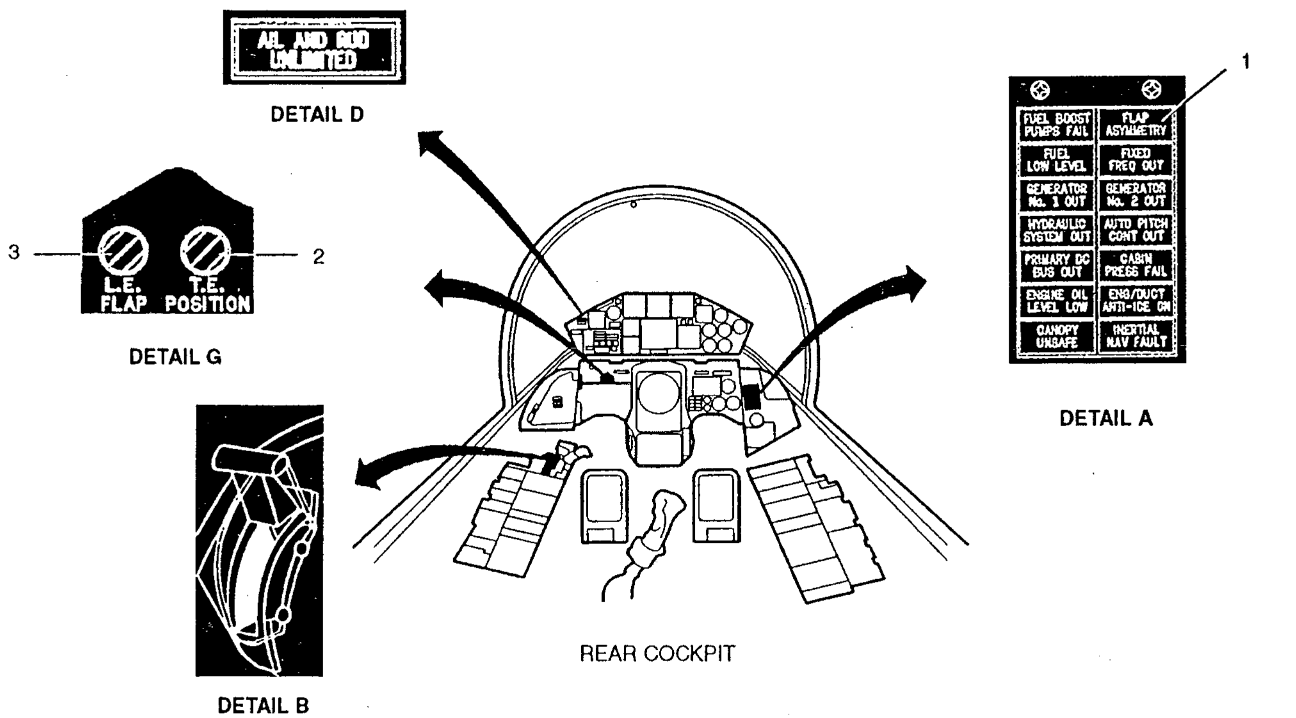


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Figure 1-19

be pulled back (toward LAND) approximately ¼ inch. The lever may be moved forward to the UP position. The front and rear cockpit wing flap levers are interconnected and receive power from the PP3 bus.

Wing Flap Position Indicators

Position indicators for the trailing and leading edge flaps are located on the left side of the lower instrument panel in each cockpit. The right indicator is for the trailing edge flaps and the left indicator is for the leading edge flaps.
Two windows are provided and are labeled FLAP POSITION, LE, and TE. Flap position indications for leading and trailing edge flaps are given in their respective windows. UP, T.O., or LAND rotates into view in each window to correspond with flap deflection.
Crosshatched indications appear in the window when the flaps are between placarded positions, in any position other than that selected, or when the electrical system is not energized. The indicators are powered by PP2 bus.

NOTE

The leading edge wing flap position indicator will not indicate UP until both leading edge flaps are up and both leading edge flap latches actuate the series-wired switches installed in the latches.

Trailing Edge FLAP ASYMMETRY Warning Light

The trailing edge FLAP ASYMMETRY warning light, on both warning panels, illuminates immediately upon actuation of the asymmetry detection system thus warning the pilots of the asymmetrical flap condition. The lights are powered by the PP2 bus.

BOUNDARY LAYER CONTROL SYSTEM

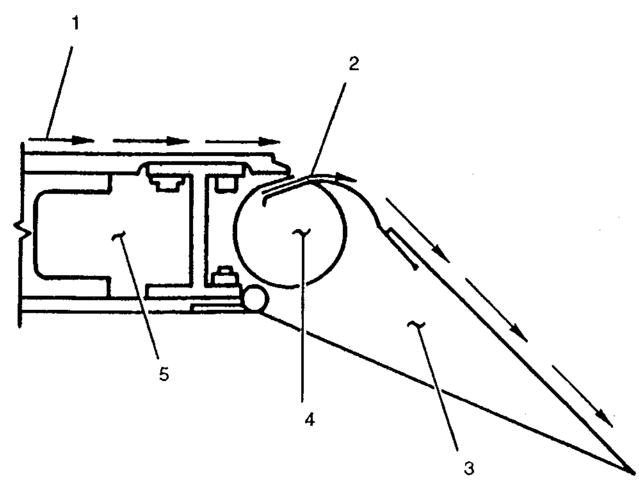
Air is bled from the last compressor stage of the engine and ducted to the boundary layer control manifold, which is located above the trailing edge flap hinge line (see Figure 1-20).

The boundary layer control manifold has a series of nozzles which direct this high-pressure, high-temperature air over the upper surface of the flap when the LAND position is used. The high velocity created by this jet of air causes it to adhere to the curved fairing and bend around and pass over the upper surface of the flap.
This curving jet causes the adjacent layer of air to adhere and bend through the flap-deflection angle, thus preventing airflow separation. This results in a reduced landing speed. The system operation is completely automatic.

Boundary Layer Control Valve

Since boundary layer control is used only with a 45° flap setting there is no airflow for flap angles of 15° or less. This is accomplished by a valve which is mechanically driven by the flap actuator; thus, the position of the valve always corresponds to that of the flaps. The valve butterfly remains closed from 0° to 15°, the valve moves until it is full open at 45°.

BOUNDARY LAYER CONTROL DUCT SECTION



1 AIR FLOW
2 BOUNDARY LAYER CONTROL NOZZLE
3 TRAILING EDGE FLAP
4 BOUNDARY LAYER CONTROL DUCT
5 AFT WING SECTION

FA0098

Figure 1-20

SPEED BRAKE SYSTEM

The speed brakes consists of two flaps, one on the left and one on the right side of the fuselage, just aft of the trailing edge of each wing. Total projected area of the speed brakes is approximately 8.25 square feet. The flaps move both outward and aft as they are extended by hydraulic cylinders. Maximum outward deflection is approximately 52° from the retracted (faired) position.
The speed brakes are electrically controlled by power from the PP2 bus and hydraulically actuated by the No. 2 hydraulic system through the priority valve. The priority valve will close and prevent speed brake operation if the No. 2 hydraulic system pressure drops to below 2175 psi. The speed brakes close automatically in the event of the electrical power loss, because of a solenoid-operated valve in the system that fails to the closed position, allowing normal hydraulic pressure or windmilling engine hydraulic pressure to close the speed brakes. No position indication is provided for the speed brakes.

Speed Brake Switch

The speed brake switch is a thumb-actuated, slide-type switch located on top of each throttle lever; it is powered by the PP2 bus. The three positions are IN, NEUTRAL and OUT. Incremental positioning of the speed brakes may be made by moving the switch back to the NEUTRAL position, which hydraulically locks the speed brakes in any intermediate position between OUT and IN. The switch in the rear cockpit is spring-loaded to the neutral position. In the other positions it overrides the forward cockpit.

LANDING GEAR SYSTEM

The aircraft is equipped with two main landing gear and a nose landing gear. Normally, the landing gear system is hydraulically operated and electrically controlled by the PP2 bus. Normal extension time of the landing gear is $\leq$6 seconds; normal retraction time of the landing gear is $\leq$7 seconds.
The gear may be recycled, if necessary, before reaching either the full up or full down position. A manual release system is provided for emergency extension of the gear, which also requires $\leq$6 seconds. There is no means of retracting the landing gear in flight after an emergency extension.

MAIN LANDING GEAR

The main landing gear retracts forward and inward. A linkage causes the wheels to rotate 90° during retraction so that they fit into the wheel wells. Each main gear when retracted is enclosed by a forward and aft door. The forward door is hydraulically operated, while the aft door is mechanically linked to the gear and travels up and down with it. The doors are locked in the closed position by four latches on the fuselage structure. The latches also serve as main-gear uplocks because, in the event of hydraulic pressure loss, the doors support the gear in the closed position. The forward door is held open by hydraulic pressure and air loads while the gear is being extended.
During normal operation, after the gear is extended, the forward door is returned to within 4 inches of the fully closed position and held there by a mechanical detent. The main gear is locked in the down position by the drag strut cylinder assembly. Barrier engagement fingers are located on the forward doors to retain the barrier cable during low-speed engagements. When the gear is extended by the manual release the forward doors remain in the open position. Ground safety lockpins are provided for manual installation in the downlock linkage at the forward end of the drag strut cylinder (Figure 1-22) of each gear.

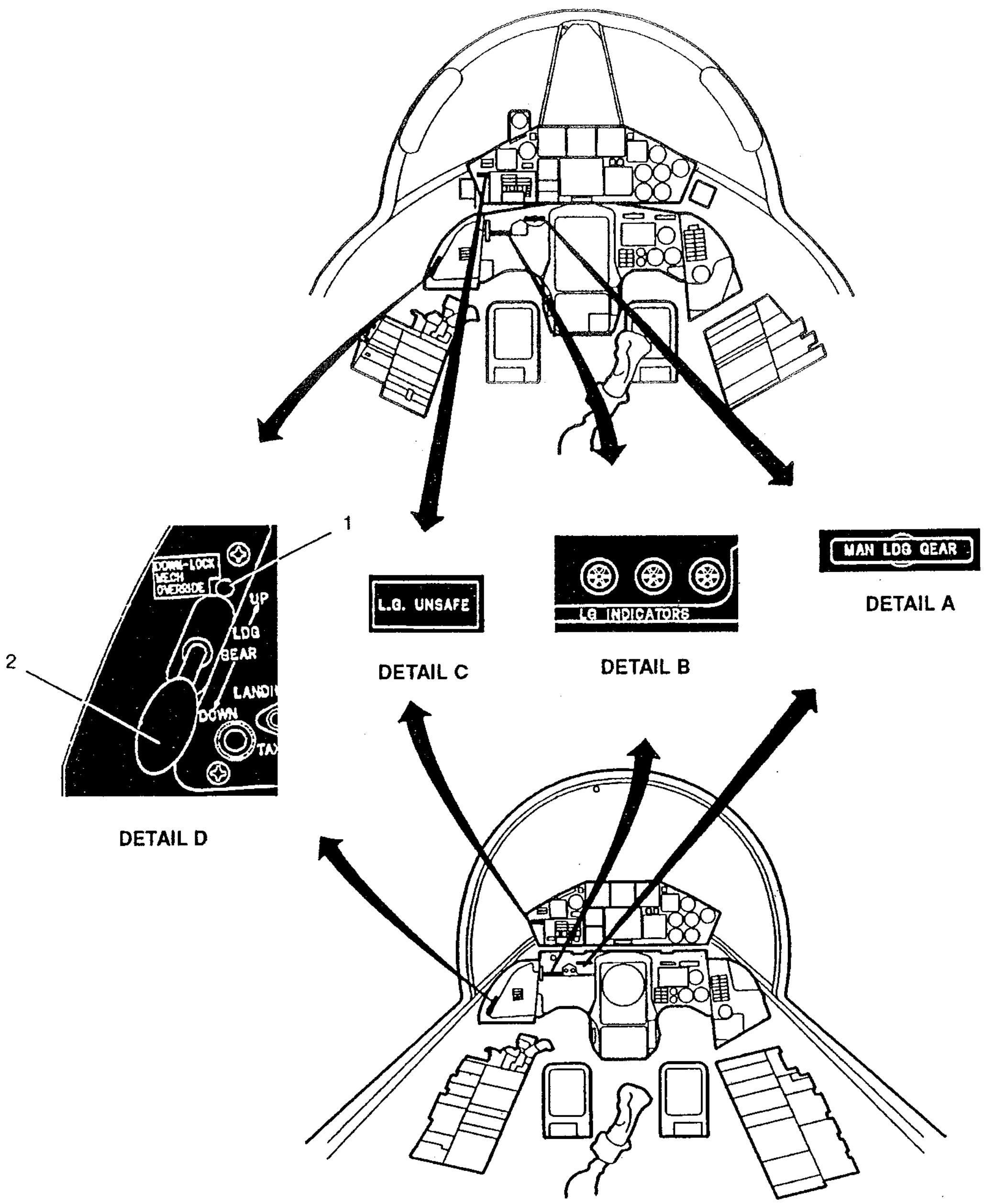
NOSE LANDING GEAR

The nose gear retracts aft on aircraft into a wheel well aft of the cockpit. The nose gear incorporates a conventional air-oil shock strut. When the nose gear is retracted, it is enclosed by two doors that are mechanically operated through contact with the nose gear strut. When the gear is extended, a downlock mechanism serves to lock the knee joint in the extended position.
An uplock cylinder is mounted on the drag strut support beam and is linked to an uplock hook mounted on the upper drag strut pin. The uplock engages a lug on the nosewheel fork to lock the gear up. A ground safety clip is provided for manual installation on the downlock stop cartridge (Figure 1-22). The nose gear is steerable by the rudder pedals (refer to Nosewheel Steering System paragraph).

LANDING GEAR LEVER

The landing gear lever is located on the left forward panel. The lever has two positions, UP and DOWN. The lever electrically controls the landing

LANDING SYSTEM CONTROLS AND INDICATORS



A – MANUAL LANDING GEAR RELEASE HANDLE

B – LANDING GEAR POSITION INDICATOR LIGHTS

C – L.G.UNSAFE WARNING LIGHT

D – LEFT FORWARD PANEL
1 LANDING GEAR LEVER OVERRIDE BUTTON
2 LANDING GEAR LEVER

FA0092

Figure 1-21

LANDING GEAR GROUND SAFETY PINS/CLIP

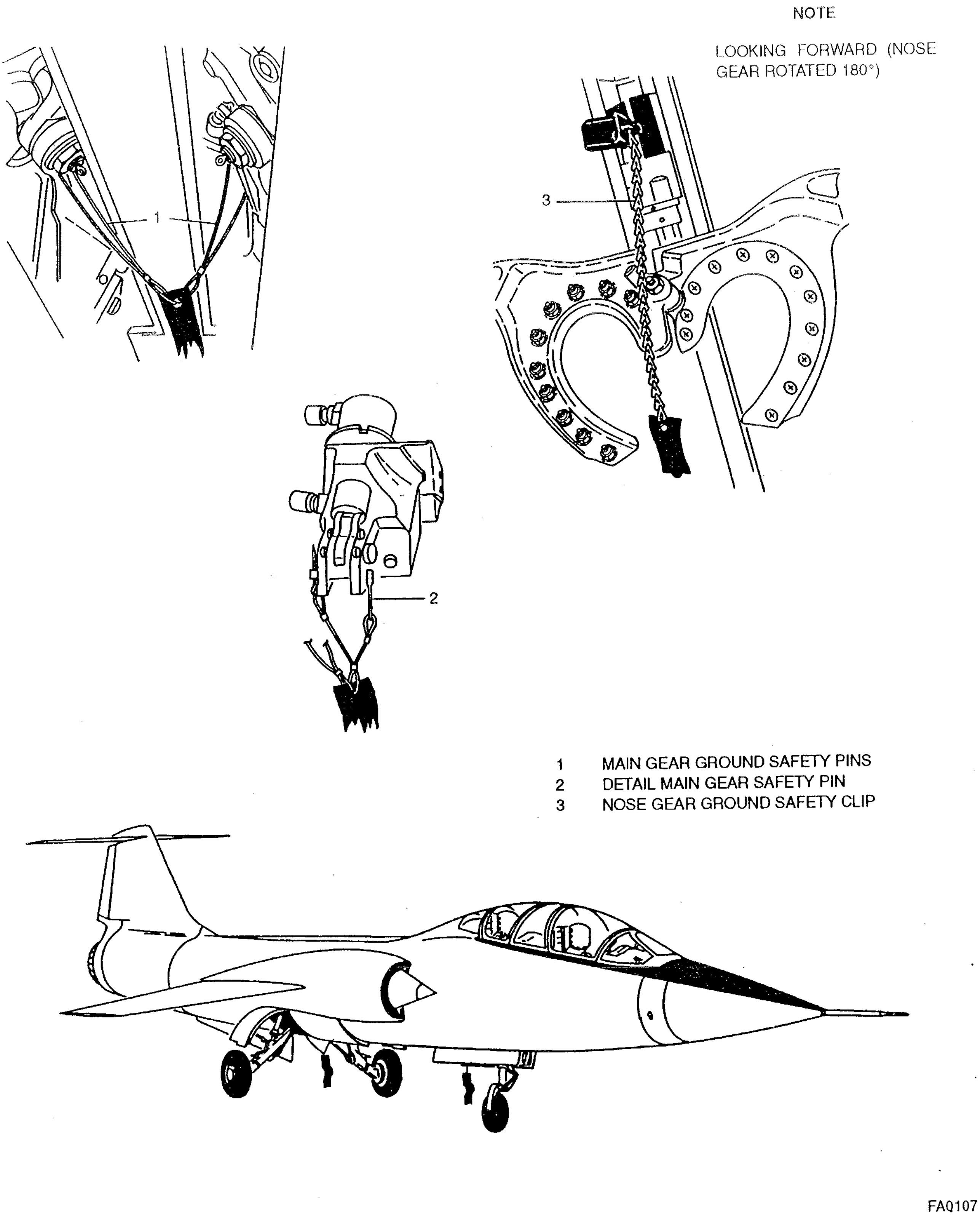


Figure 1-22

gear and landing gear door hydraulic selector valve. When the lever is moved to the UP position, selector valves which are electrically sequenced to direct hydraulic pressure to open the main gear forward doors, retract the nose and main gear (and aft doors), and then reclose the main gear forward doors. The nose gear doors are mechanically connected to the nose gear strut and open and close with nose gear extension and retraction. When the landing gear lever is moved to the DOWN position the selector valves direct hydraulic pressure to lower the nose gear, open the main gear forward doors, and lower the main landing gear (which opens the aft main gear doors). When the gear reaches the down-and-locked position, hydraulic pressure is automatically selected to close the main forward doors to the mechanical detent position. The front and rear cockpit landing gear levers are interconnected.

Landing Gear Lever Uplock Button

A landing gear lever uplock mechanism is provided to prevent the gear lever from being lowered inadvertently. A button, which extends upwards from the top of the landing gear lever is used to release the lever uplock mechanism.

Landing Gear Lever Override Button

A landing gear lever override button is provided just above the landing gear lever. The button may be used in an emergency to override the lever downlock if it becomes necessary to raise the gear when the weight of the aircraft is on the landing gear. When the aircraft is on the ground with the gear down and locked, a solenoid-operated locking mechanism locks the landing gear lever in the DOWN position. This locking mechanism is provided with a mechanical downlock bypass, operated by the push-button. When the weight of the aircraft is off the nose gear PP2 bus power is directed to the control lock solenoid which permits the lever to be moved to the UP position.

Manual Landing Gear Release Handle

A yellow handle, located in the top left corner on the lower instrument panel in each cockpit, is labelled MAN LDG GEAR; its purpose is to release the main and nose landing gear door uplocks and to open the dump valves, which allows the gear to lower by gravity and air-load forces. The nose gear is extended against the airstream bymeans of a bungee. The gear is then locked down by spring-loaded downlocks.

Approximately a 10-inch pull (up to 90 lb force) to the stop is required to release the gear. Pulling the manual release handle with the landing gear lever in the DOWN position will require a force exceeding 90 pounds. The landing gear cannot be retracted in flight after being lowered by means of the manual landing gear release handle.

If the manual landing gear release handle is used to lower the landing gear, the gear cannot be retracted until the system valves are repositioned manually prior to the next flight.

CAUTION

GEAR EXTENSION BY MEANS OF THE MANUAL LANDING GEAR RELEASE HANDLE WILL RENDER THE ANTISKID AND POWER BRAKES INOPERATIVE. NOSEWHEEL STEERING WILL OPERATE IF NORMAL HYDRAULIC SYSTEM PRESSURE IS AVAILABLE.

Landing Gear Position Indicator Lights

Three green lights, labeled LG INDICATORS, are installed on the left side of the lower instrument panel in each cockpit. When the lights are illuminated they indicate that the respective landing gear is down and locked.

As each gear reaches the down-and-locked position power from the PP2 bus is directed through the warning light circuit to illuminate the indicator light. The lights go out any time the gear is not down and locked, except when they are energized by the warning light test switch.

Landing Gear Lever Unsafe Warning Light

A red warning light is installed in the transparent knob of the landing gear lever. This light provides the pilot with a visual signal whenever the landing gear is not in the position selected. The light receives power from the PP2 bus.

NOTE

- Once a landing gear is retracted and the warning light is extinguished, the landing gear circuit is made insensitive to sequencing switch malfunctions and will remain in the UP position until the landing gear handle is placed in the DOWN position.

- If the warning light flickers once the gear is up and locked, this indicates improper tolerances in the door microswitches; however, the gear and doors will remain up and locked. If this happens, the mission may be continued but the incident should be written up in the aircraft forms.

- If the warning light comes on steadily, this indicates a possible malfunction of the uplock relays. In this condition, the gear could be back under control of the sequencing switches. High positive or negative g maneuvres should be avoided, and the mission aborted. The warning light will remain illuminated until the forward main gear doors close during retraction.

Landing Gear Unsafe Warning Light

A landing gear unsafe warning light is located on the instrument panel in both cockpits. The words L.G. UNSAFE are illuminated whenever the landing gear is not in the position selected. This provides an additional visual warning signal of landing gear malfunction.

Landing Gear Warning Signal

A landing gear warning signal, relative to engine speed and air data computer sensing (altitude and airspeed) is transmitted to the pilot's earphones through the interphone system when the landing gear is not down and locked. If the gear is up and the throttle is retarded below 95 to 97% RPM, the aircraft is below 10000 ± 1000 feet, and the airspeed is less than 220 ± 10 knots IAS, the warning signal will be generated. Also anytime the gear is unsafe, i.e., unlocked, and the throttle is retarded below 100% RPM, the warning signal will be transmitted regardless of airspeed or altitude. In addition, the landing gear unsafe warning lights will illuminate anytime the warning signal is generated. The signal system receives power from PP2 bus.

NOSEWHEEL STEERING SYSTEM

The steering system provides power steering for the nosewheel when the aircraft is on the ground. The nosewheel is steerable 25° either side of center. The aircraft is steered by an hydraulically powered steerdamper unit controlled through a cable system by the rudder pedals No. 2 hydraulic system pressure is routed to the steering system through a solenoid shutoff valve, a filter, and a pressure-reducing valve that reduces system pressure from 3000 to 2500 psi. The solenoid shutoff valve is controlled by a switch on the control-stick grip. The system is irreversible in fact forces on the nosewheel cannot be transmitted back to the rudder pedals. Upon retraction, the nosewheel automatically centers.

NOTE

The ground-air safety switch and relay are connected to the shutoff valve circuit in a way which ensures that the steering system is inoperative unless the weight of the aircraft is on the main landing gear. The nosewheel steering is governed by the nosegear scissor switch.

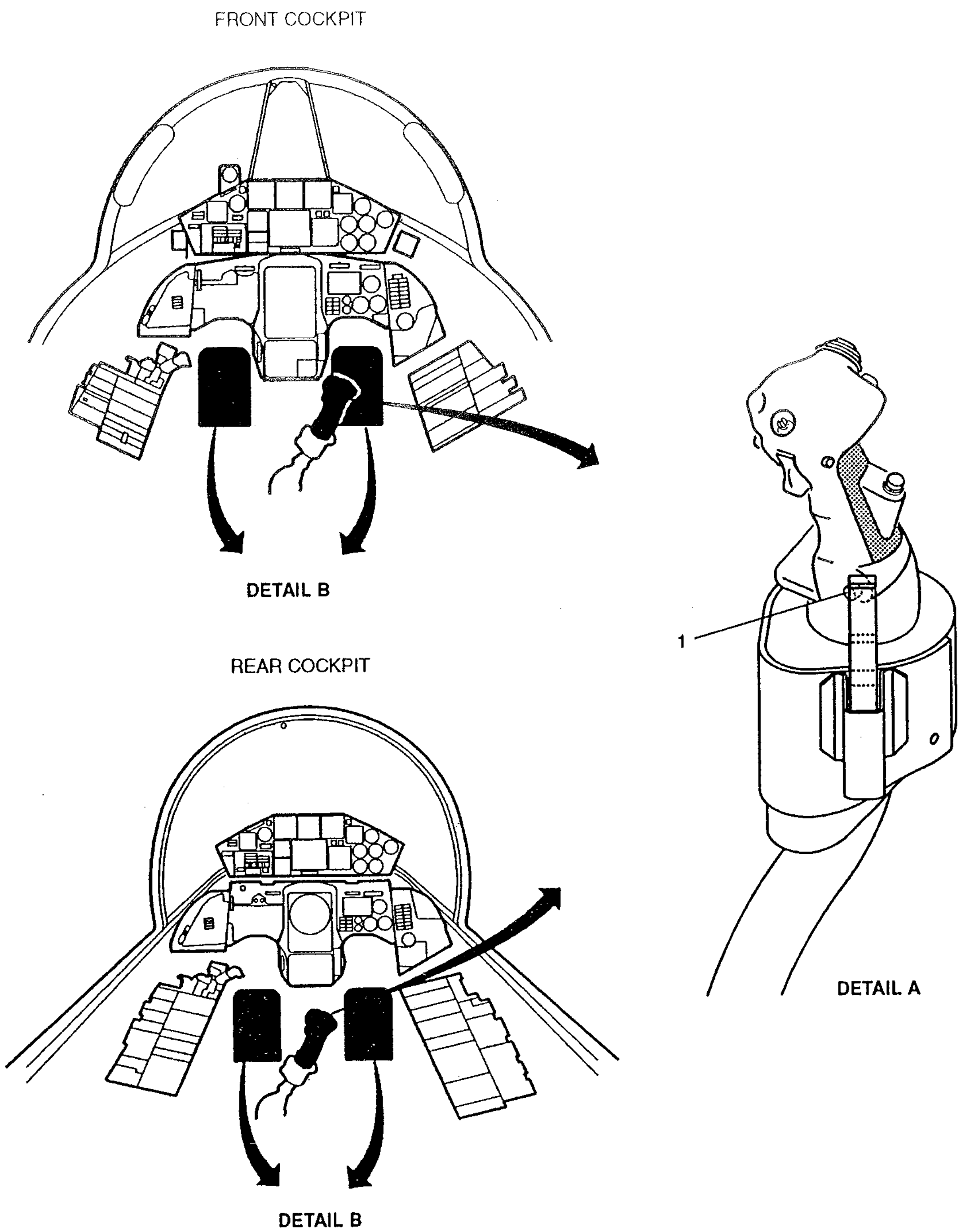
STEER-DAMPER UNIT

The steer-damper unit transforms hydraulic pressure into steering force when the unit is pressurized and when nosewheel steering is engaged. When unpressurized, the unit absorbs shock loads and dampens nosewheel shimmy. When pressure is applied, an internal clutch engages the unit with the rudder cables through a control pulley.

Nosewheel Steering/Microphone Button

A pushbutton (Figure 1-23) mounted on the control-stick grip engages the nosewheel steering system. The nosewheel steering button must be

NOSEWHEEL SYSTEM CONTROLS AND INDICATORS



A – CONTROL STICK GRIP
1 NOSE WHEEL STEERING/MICROPHONE BUTTON

B – RUDDER PEDALS

FA0093

Figure 1-23

pressed and held for nosewheel steering actuation. When the system is activated through the nosewheel steering button, PP1 bus power is directed to a shutoff valve which directs No. 2 hydraulic system pressure to the nosewheel steering unit. A clutch is then engaged hydraulically to link the rudder controls with the steering unit when rudder pedals and nosewheel are aligned.

NOTE

- Nosewheel steering should not be engaged utilizing the nosewheel steering button on the control stick unless the nosewheel and rudder pedals are aligned. If the nosewheel steering button is actuated when the pedals are deflected, clutch friction within the steering system may cause an undesired turn as the pedals are moved to align with the nosewheel. If the nosewheel is deflected during a turn and nosewheel steering is inadvertently disengaged (release of button), the nosewheel will align to neutral. To continue the turn, the rudder must then be neutralized, the nosewheel steering button pressed again and the pedal deflected in the direction of desired turn.

- The nosewheel steering button will energize nosewheel steering only if PP1 bus power is available and the weight of the aircraft is on the main landing gear and/or nose landing gear.

- The nosewheel steering button(s) may be activated during flight to mute radio interference from an outside source during interphone communication between the front and rear cockpits. With the weight off the main gear, the nosewheel steering button functions as a microphone button for AN/ARC 150 radio and Emer UHF radio, and will active the I/P pulse of the IFF/SIF if the I/P switch is in MIC position.

WHEEL BRAKE SYSTEM

Each main gear incorporates a hydraulic brake assembly. The brakes are of the self-adjusting, segmented rotor type. They are actuated by pressure supplied by the hydraulic power system or the standby brake system.

Normally the hydraulic power system with antiskid control is used. The standby brake system is automatically available as an emergency backup to the power system.

POWER BRAKE/ANTISKID SYSTEM

Normally, the brakes are operated by a hydraulic power system which incorporates an antiskid control. No. 2 system hydraulic pressure at 3000 psi is delivered to the system through the landing gear down line. Hydraulic pressure is directed to the brake assemblies through a solenoid shutoff valve and power brake valves. The power brake valves are controlled conventionally by toe pressure on the rudder pedals. The valves meter and control hydraulic pressure up to 1000 psi maximum to the right or left wheel brake through three-way, solenoid operated shutoff valves.

Antiskid action is accomplished by means of a dc generator, located in each main gear axle. The output voltage of each generator is proportional to wheel speed.

This voltage controls electrical current to the three-way, solenoid-operated shutoff valve which in turn controls brake pressure. This valve is electrically energized from the primary dc bus and dumps brake pressure (independently for each brake) when an incipient skid causes a sudden drop in generator output. Upon recovery of wheel speed, the valve is deenergized, allowing pressure to be reapplied to the brake. A pressure switch is located in the landing gear hydraulic down line which closes when the gear is extended on the normal system. This, in series with the antiskid switch, energizes the antiskid system.

The electrical system includes a "touchdown circuit" which prevents application of brake pressure until after touchdown. A fail-safe circuit is also included which automatically shuts off the power brake system (and antiskid control) in the event of a malfunction which would prevent the application of brake pressure for longer than 3 seconds. The braking operation will then automatically return to the standby system.

WHEEL BRAKE SYSTEM CONTROLS AND INDICATORS

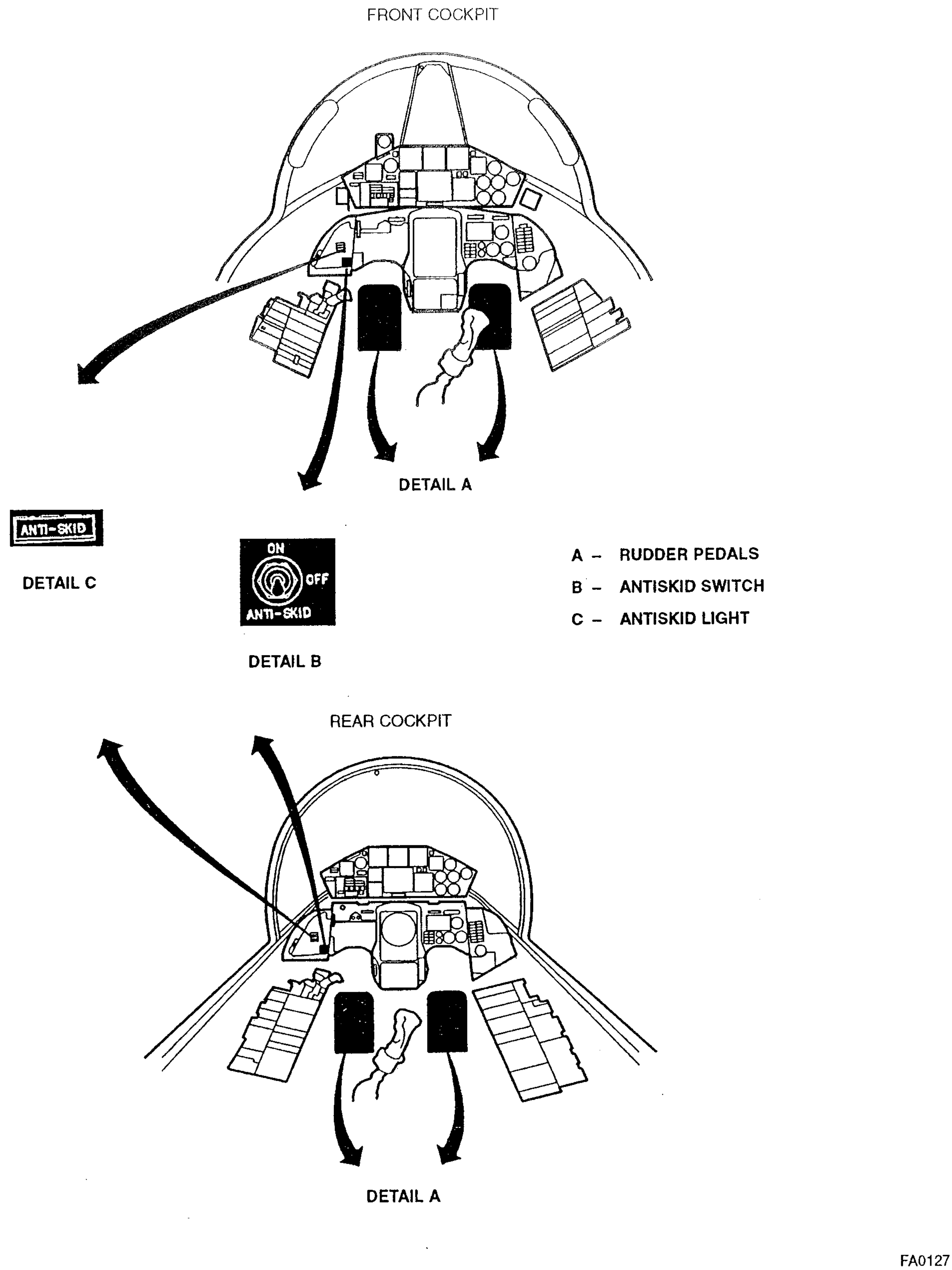


Figure 1-24

NOTE

- The power brake system is not available separately from the anti-skid system except at speeds below 10 knots.

- At speeds below 10 knots, the output of the dc generators is too low for antiskid operation. The brakes at these low speeds operate as straight power brakes: hence, with maximum braking the wheels will lock and skid when the speed drops below 10 knots.

- During the time the speed brakes move from open to closed, or closed to open, nosewheel steering and/or power brakes may be momentarily inoperative; also, antiskid brakes, if on, may automatically revert to manual. Upon completion of the movement of the speed brakes, nosewheel steering becomes available; however, it may be necessary to move the rudder pedals to reengage the steering. The brakes will revert automatically to power/antiskid.

STANDBY BRAKE SYSTEM

The standby brake system operates conventionally by toe pressure applied on the rudder pedals connected to a master brake cylinder which is integral with the power brake valve. The system is supplied fluid from the hydraulic return lines. A small reservoir is incorporated in the return system for fluid supply when the return system (aircraft reservoir) is not pressurized. The system will be available under any of the following conditions: (1) antiskid switch OFF, (2) operation of fail-safe circuit in the antiskid system (3) loss of electrical power, (4) loss of hydraulic system pressure, and (5) after manual gear extension.

Antiskid Switch

The antiskid switch is located on the left forward panel in both cockpits (refer to Figure 1-24). With the landing gear extended on the normal system and the switch in the ON position, electrical power from the PP1 bus is available for operation of the antiskid system. Both antiskid switches shall be on to energize the antiskid circuit.

Antiskid Light

An amber light marked ANTI-SKID is located on the left forward panel in each cockpit (refer to Figure 1-24). The light illuminates with the antiskid switch ON, the aircraft airborne, and the landing gear extended on the normal system. The antiskid light indicates to the pilot the following:

a. When illuminated in the air-the system is ready for use and that brake pressure cannot be applied until after touchdown.

b. Light off after touchdown-brake pressure may be applied through the antiskid system.

c. Light on after touchdown-the antiskid system is not functioning properly and the fail-safe circuit has shut off the power brake system. Brake pressure will be applied through the standby brake system.

WARNING

IF THE GROUND-AIR SAFETY SWITCH DOES NOT CLOSE AFTER TOUCHDOWN, THE ANTI-SKID LIGHT WILL REMAIN ON AND THE SWITCH WILL HAVE TO BE TURNED OFF TO SELECT STANDBY BRAKES. UNDER THIS CONDITION, NOSEWHEEL STEERING WILL BE INOPERATIVE.

DRAG CHUTE SYSTEM

A drag chute is provided to reduce landing distances. The chute, packed in a deployment bag, is stowed in a compartment located in the lower part of the aft fuselage. Deployment is mechanically controlled from the cockpit.

DRAG CHUTE SYSTEM CONTROLS AND INDICATORS

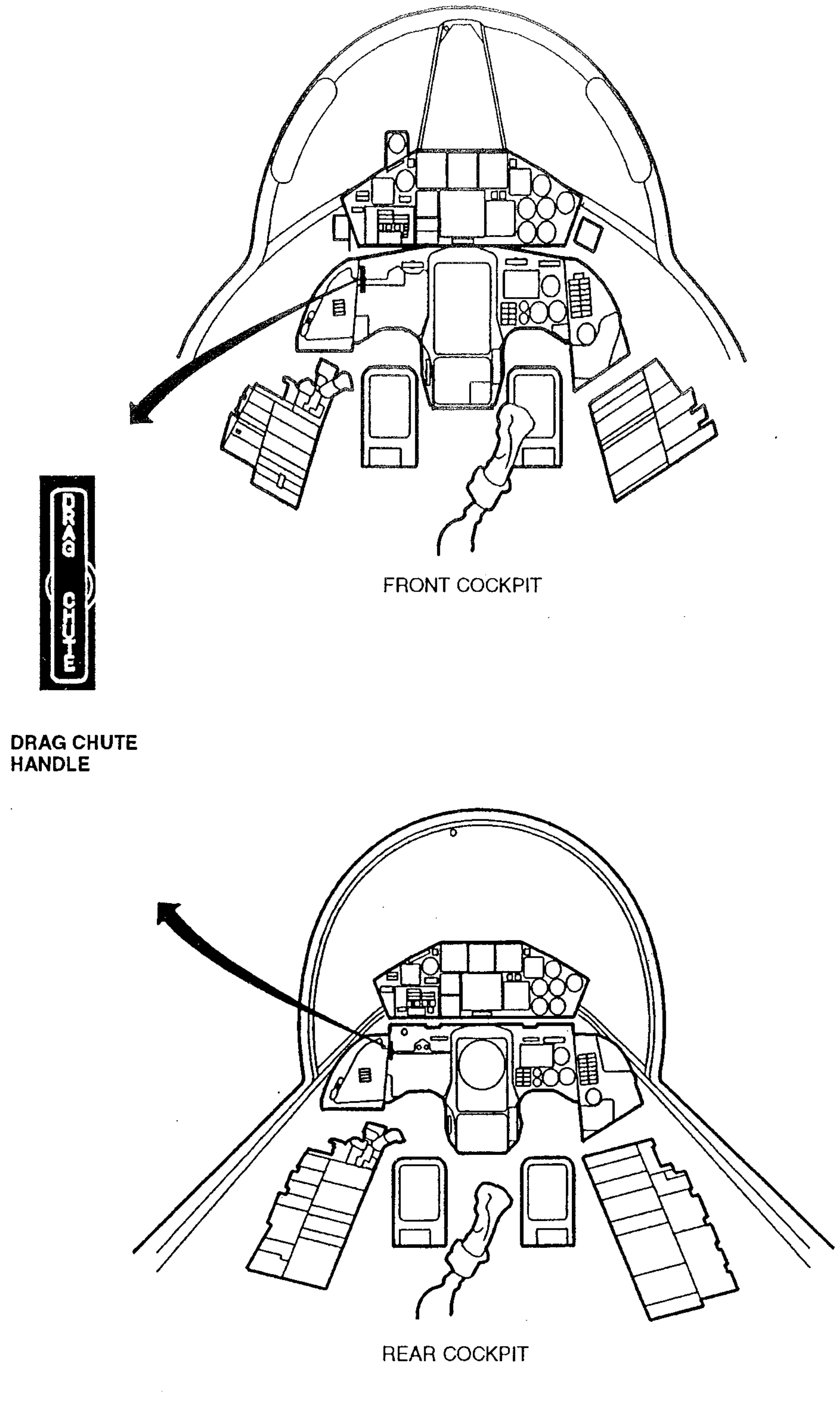


Figure 1-25

CAUTION

- DUE TO THE POINT OF ATTACHMENT OF THE DRAG SHUTE, IT CAUSES AN AIRCRAFT NOSE-DOWN PITCHING MOMENT WHEN DEPLOYED; THEREFORE, IT SHOULD NOT BE DEPLOYED UNTIL THE NOSEWHEEL IS ON THE GROUND AND SPEED IS BELOW THE DESIGN LIMIT OF THE CHUTE.
- IF THE DRAG CHUTE IS DEPLOYED INADVERTENTLY AT AIRSPEEDS IN EXCESS OF DESIGN LIMITS, EITHER A LINK WILL FAIL RELEASING THE DRAG CHUTE OR THE DRAG CHUTE CANOPY WILL DISINTEGRATE. THIS SITUATION MAY BE DETECTED BY A SUDDEN DECELERATION FOLLOWED BY AN IMMEDIATE ACCELERATION. IN EITHER CASE, FOLLOWING SUCH AN OCCURRENCE THE DRAG CHUTE SHOULD BE JETTISONED, THUS DUMPING THE TRAILING CHUTE REMNANTS IN THE EVENT THE LINK DID NOT FAIL. DRAG CHUTE DEPLOYMENT LIMIT IS 200 KIAS FOR THE IRVING VPCR CHUTE.

NOTE

Landing distance performance is provided in Appendix for the 18-foot ribbon-type drag chute.

Drag Chute Handle

The drag chute handle is located at the left of the lower instrument panel in each cockpit (refer to Figure 1-25). When pulled straight aft (about 2 inches) and held firmly against the stop (without turning the handle) the spring-loaded drag chute door opens and a pilot chute is deployed; the pilot chute in turn deploys the drag chute. The drag chute handle should be held until the opening shock of the chute is encountered.

The drag chute can then be jettisoned at any time by turning the handle 90° clockwise and pulling to the next stop (about 4 inches).

The handle is under spring tension during the final pull. When the handle is released it will retract to the first stop.

NOTE

Turning the drag chute handle, after deployment of the drag chute, should only be performed when ready to jettison to prevent inadvertent or premature jettison of the drag chute.

ARRESTING HOOK

The arresting hook installation is an emergency system designed to engage a barrier cable, reduce the landing roll, and bring the aircraft to an emergency stop. The hook installation consists of the hook assembly, piston, two springs and linkage, the self-contained hydropneumatic snubber assembly, the solenoid-operated latch assembly, arresting hook indicator light, and the hook release button.

In the stowed position the hook lies beneath the drag chute compartment; this location requires the hook to drop to an intermediate position when the drag chute is deployed. The hook will extend from either the intermediate or stowed position.

The hook is mounted slightly to the right of fuselage centerline and partially submerged in the fuselage to reduce aerodynamic drag and possible interference with the TACAN antenna patterns.

WARNING

STAY CLEAR OF ARRESTING HOOK WHEN IT IS IN A STOWED OR INTERMEDIATE POSITION AS INADVERTENT RELEASE COULD CAUSE SERIOUS INJURY.

ARRESTING HOOK CONTROLS AND INDICATORS

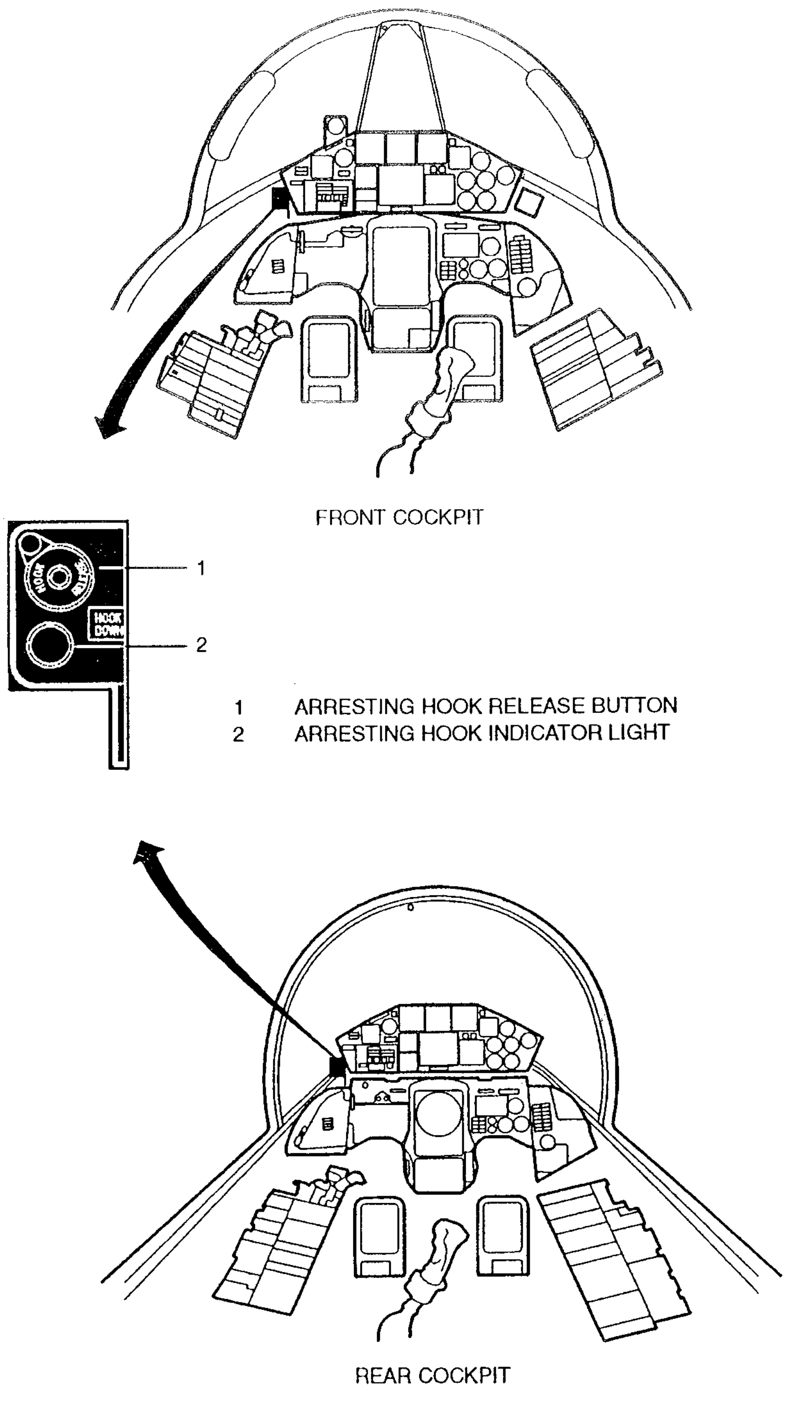
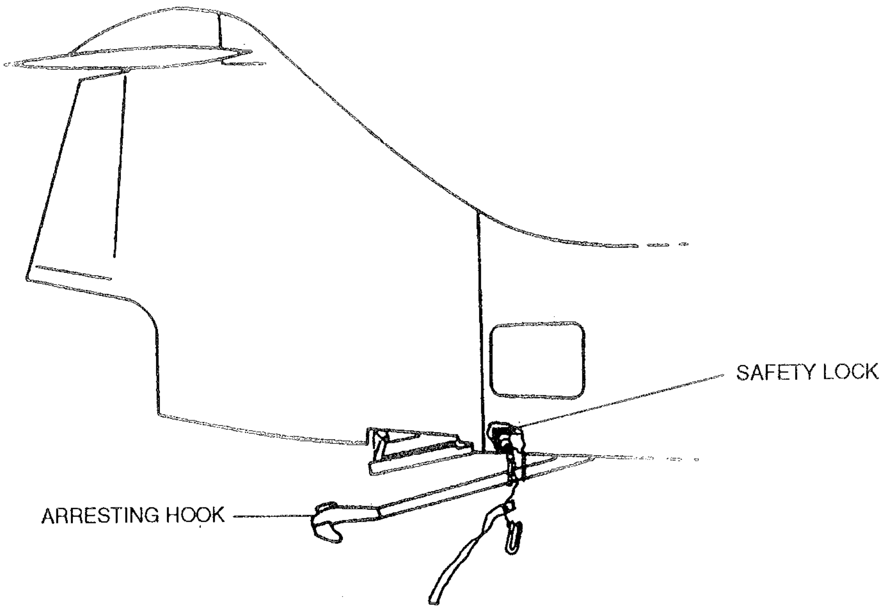
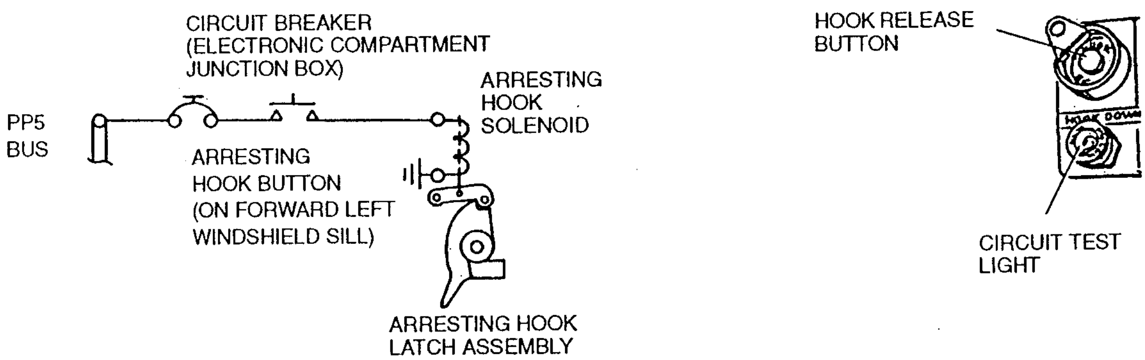


Figure 1-26

ARRESTING HOOK





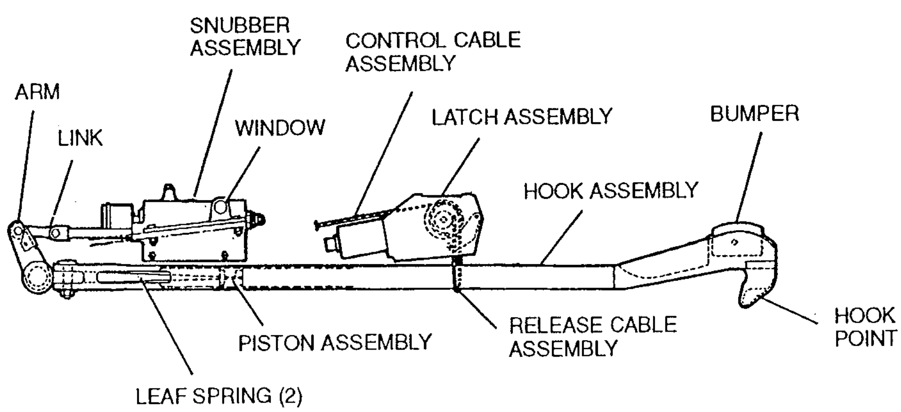


Figure 1-27

NOTE

- The arresting hook is not retractable from within the cockpit either in flight or on the ground. The hook must be retracted and latched manually by the ground crew.

- A safety lock assembly is provided to assure positive uplock of the arresting hook. This lock should be installed when the aircraft is on the ground.

Arresting Hook Release Button

The arresting hook is extended by use of the HOOK RELEASE BUTTON located on the left windshield sill in both cockpits (refer to Figure 1-26).
When the button is pressed, the solenoid-operated latch assembly releases the hook and the hook is forced down to runway. The hook release button receives power from the PP5 bus.

Arresting Hook Indicator Light

A HOOK DOWN indicator light (refer to Figure 1-26), located below the hook release switch, illuminates anytime the hook extends below the drag chute deployment (intermediate) position. The light is powered from the PP5 bus. The press-to-test function is also a circuit test function.

INSTRUMENTS

For information regarding instruments that are an integral part of particular system, refer to applicable paragraphs in this Section.

PITOT PRESSURE AND STATIC SYSTEM

The pitot pressure and static system operate the airspeed indicator, altimeter, and vertical velocity indicator. The system is also connected to the air data computer and CPU-46A altitude computer.

The pitot-static head is mounted on a boom extending forward from the nose radome.
A heating element in the head is controlled by the pitot-pitch temp probe switch on the right console in both cockpits (refer to Anti-icing System in this Section for additional information).

Airspeed and Mach Number Indicator

The airspeed and Mach number indicators (refer to Figure 1-28) consists of a pitot-static-operated indicated airspeed mechanism which drives a pointer to indicate air speed on a fixed dial.
The indicator also contains a static-pressure-operated altitude mechanism which drives a moving scale to indicate Mach number. The gearing between the moving scale and the altitude mechanism is such that Mach number is indicated by the pointer on the moving scale at any combination of indicated airspeed and altitude within range of the instrument.
The indicator is designed to operate over a range of 80 to 850 knots indicated airspeed at altitudes up to 80000 feet. The Mach number range is from 0.5 to 2.5. In addition, there is a maximum allowable pointer for a limiting equivalent airspeed of 600 to 800 knots.

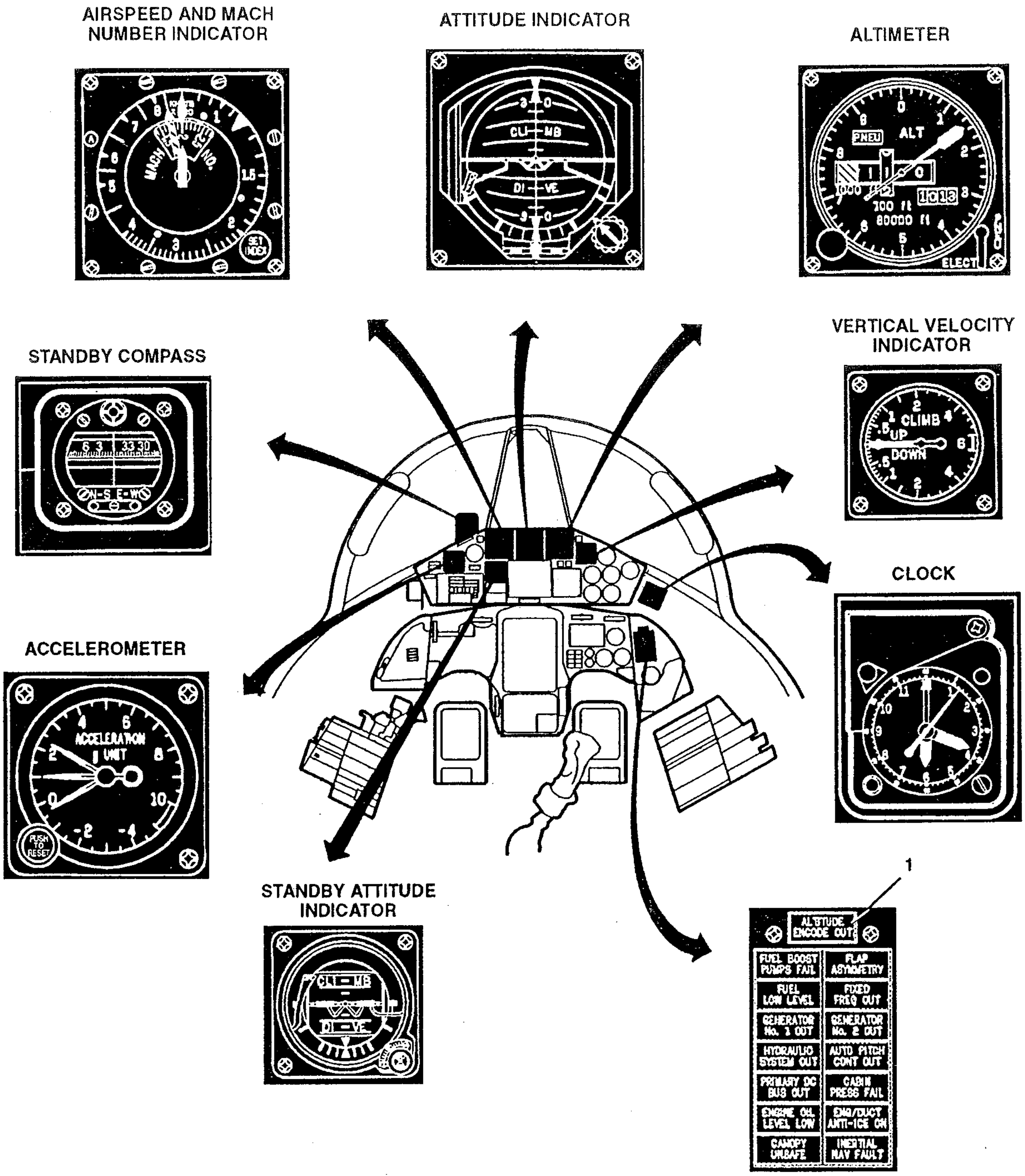
Maximum Speed Pointer

The maximum allowable speed pointer is set to reflect an equivalent airspeed of 750 KEAS. The equivalent airspeed of 750 KEAS is indicated in terms of IAS which will vary with altitude. For example, equivalent airspeed of 750 KEAS would be indicated as 750 KIAS at sea level and 799 KIAS at 20000 feet.

WARNING

THE MAXIMUM ALLOWABLE AIRSPEED IS LESS THAN 750 KEAS FOR SOME EXTERNAL STORE CONFIGURATION. OBSERVE THE AIRSPEED LIMITS IN SECTION V.

INSTRUMENTS
(FRONT COCKPIT)



A – WARNING LIGHTS PANEL
1 ALTITUDE ENCODE OUT WARNING LIGHT

DETAIL A

FA0123

Figure 1-28 (Sheet 1 of 2)

INSTRUMENTS (REAR COCKPIT)

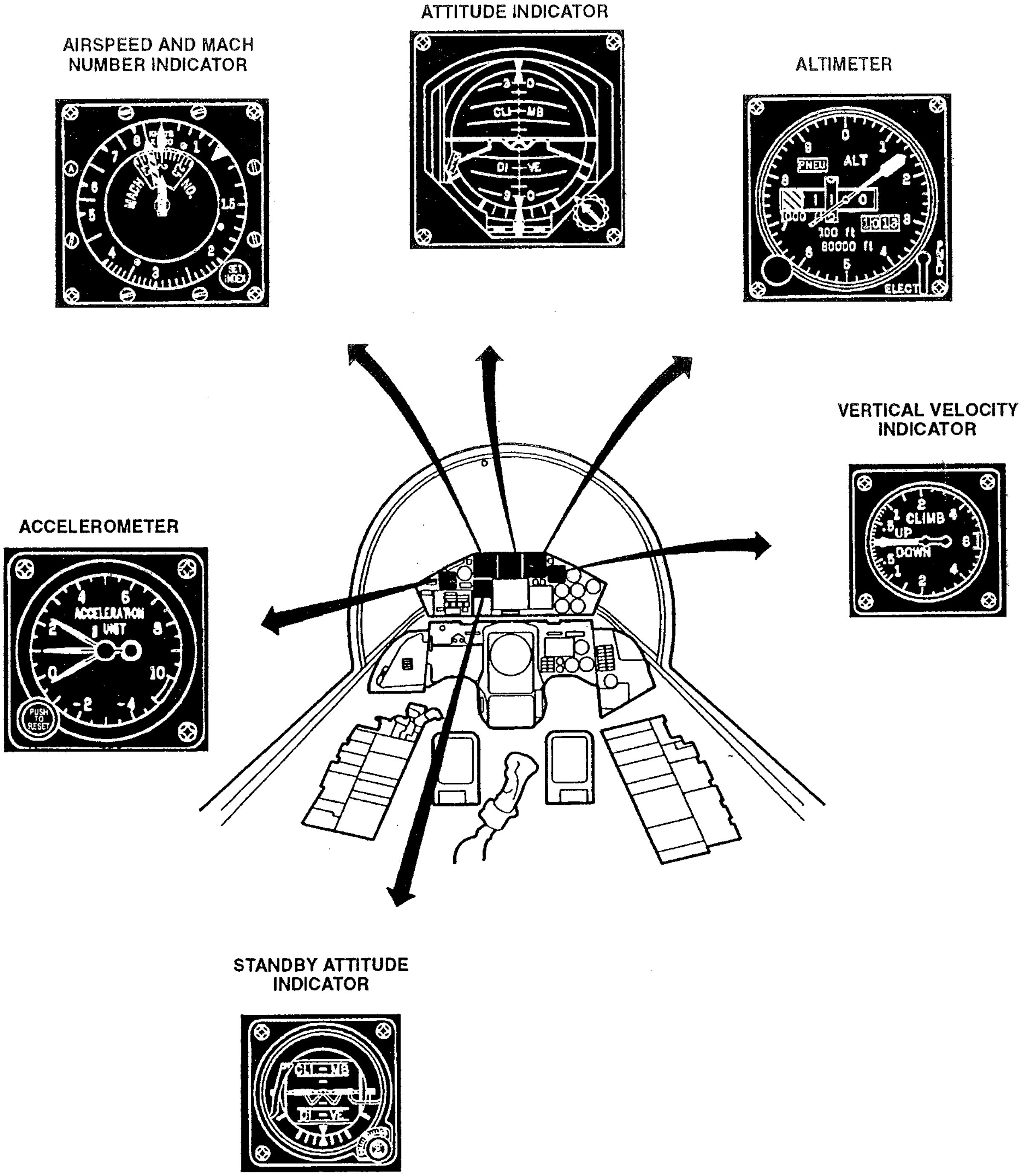


Figure 1-28 (Sheet 2 of 2)

Airspeed Marker

An airspeed marker is provided to assist the pilot in setting a speed reference. The setting of the marker is controlled by a knob at the lower right.

ALTIMETER

The AAU-37A servo-pneumatic altimeter, located on the main instrument panel (refer to Figure 1-28), has two modes of operation: servo mode (ELECT) and pneumatic mode (PNEU).
In the servo mode of operation the altimeter is controlled by signal inputs from the Altitude Computer (CPU 46/A). Direct readout of the altitude is accomplished by the numbers on the three drum counter indicating from −1000 to 80000 feet and by a single pointer which simultaneously indicates 1000 feet per revolution both in servo and in pneumatic modes.
In the pneumatic mode the altimeter is mechanically driven by a pressure transducer integral to the altimeter. An internal vibrator reduces mechanical friction of gear trains and linkages of assembly. The instrument is automatically converted from servo to pneumatic mode in case of electrical and/or servo system failure.
An ELECT/PNEU function selector allows manual selection from servo to pneumatic mode. No indication is provided on the window when the servo-pneumatic altimeter is operating in servo mode; the appearance of the PNEU flag, yellow backgrounded, will indicate the pneumatic mode of operation (manually or automatically selected) or a power interruption or a failure in the altimeter or in the altitude computer with possible resulting of loss of automatic altitude reporting (IFF mode C).

NOTE

Following the CPU-46A altitude computer failure, the servo/pneumatic altimeter switches to baro mode (PNEU).

A barometric scale control, located on the lower left corner of the instrument, allows barometric settings from 950 to 1048 millibars which are displayed on the barometric window; clockwise rotation increases barometric settings.
The left counter of the drum is covered by a black and white striped flag at altitudes below 10000 feet and by a red and white striped flag at altitudes below 0 feet.
The instrument is supplied by XP5 bus and the vibrator is supplied by PP2 bus.

Servo Mode of Operation

In the servo mode of operation corrected pressure altitude (position error correction) synchro signals are sent from the altitude computer to the receiver-transmitter of the IFF system and to a servomechanism in the altimeter.
These signals are computed only for a barometric pressure of 1013 mb. To correct the altimeter indicated altitude for other than 1013 mb, set the correct altimeter setting in the altimeter barometric scale.
When the IFF system is interrogated for altitude reporting (Mode C), the receiver-transmitter will automatically report the aircraft altitude to the nearest 100 feet for a barometric pressure of 1013 mb, regardless of the setting in the altimeter barometric scale.

Pneumatic Mode of Operation

In the pneumatic mode of operation, the altimeter receives static air pressure directly from the pitot-static system and operates in exactly the same manner as a standard pressure altimeter. Altimeter installation error corrections shall be used to correct the aircraft altitude.
Mode C reporting is available regardless of the selected mode of operation. Mode C altitude reporting will not be available if the system has automatically reverted to PNEU due to an altitude computer failure.

Altitude Computer (CPU-46A). The altitude computer, located in the left electrical compartment, senses pitot pressure and static pressure from the pitot-static system and known error parameters of the pressure system. It performs mechanical analog computation to provide analog outputs and one digital output.
The altitude computer consists of two pressure sensors, a computing mechanism and an electronic package. The analog output drives the servoed altimeter. The digital output comprises Mode C pressure-altitude information which is directed to the transponder for transmission in reply to a valid Mode C interrogation. The altitude computer is powered by XP5 bus.

Altitude Encode Out Warning Light. The ALTITUDE ENCODE OUT warning light, located on the right forward panel in the front cockpit only (refer to Figure 1-28), is installed to warn the pilot that the Altitude Computer fails to operate or the system is not energized. The light is energized by the PP5 bus.

Standby Compass

A standby magnetic compass (refer to Figure 1-28) is provided for back-up navigation if the electrical system fails and may be also used for heading crosscheck with the HSI.
The compass is located on the left side above the instrument panel glareshield in the front cockpit only and is hinged to fold forward. The light within the compass case is controlled by the instrument light rheostat.

Accelerometer

A three-pointer accelerometer (refer to Figure 1-28), located on the instrument panel in each cockpit, indicates positive and negative G loads. In addition to the indicating pointer there are two recording pointers (one for positive-G loads and one for negative-G loads) which follow the indicating pointer to its maximum attained travel.
The recording pointers remain at the maximum travel positions reached by the indicating pointer, thus providing a record of maximum-G loads encountered.
To return the recording pointers to the normal 1 G position, press the knob on the lower left corner of the instrument.

ATTITUDE INDICATOR

The attitude indicator (refer to Figure 1-28) located on the main instrument panel in both cockpit, is powered by XP7 bus and provides the pilot with a pictorial presentation of aircraft attitude.
The installation consists of an attitude sphere background which is supported and rotates about two servo-driven gimbal axes, a bank index and scale, a miniature aircraft, a self-contained slip/skid indicator, a rate of turn indicator and an OFF warning flag indication.
The sphere shall be of two colours, one depicting earth, the other sky, and markings with pitch angles. The pitch angle shall be read by means of pitch marking against the miniature aircraft that is located in front of the attitude sphere at zero pitch pattern. A pitch trim setting shall be allowed by means of a knob. Roll angle shall be read by means of the bank index, which rotates with the roll gimbal, against a fixed bank scale around the periphery of the display dial.
The spheroid rotation freedom is unlimited about the roll axis and ±90° of indication about the pitch axis.
The accuracy of pitch and roll indications with respect to input synchro-transmitter are within 1° for positions up to 30° in roll and pitch and 2° for any other position. The follow up rate is not less than 300° per second in roll and 180° per second in pitch.
Aircraft slip is indicated by a floating ball inside a tubular case located in the lower portion of the instrument with a range of ±5°.
Rate of turn is provided by a gyro transmitter and is indicated by the rate of turn pointer and travels with a linear movement over the rate of turn scale. The rate of turn indication is a "four minutes" turn type.

WARNING

THE "FOUR MINUTES TURN" INDICATION IS RELIABLE DURING CONSTANT ALTITUDE TURN AT ANY AIRCRAFT SPEED WITHIN ±40° BANK ATTITUDE.

OFF Flag

The attitude malfunction detection circuit shall be designed to accept an attitude data validity signal from external source.
The OFF warning flag indicates that the attitude data is invalid.

WARNING

THE ATTITUDE INDICATOR OFF FLAG MAY NOT APPEAR WITH A SLIGHT REDUCTION IN AC POWER, OR FAILURE OF OTHER COMPONENTS WITHIN THE SYSTEM. FAILURE OF CERTAIN COMPONENTS MAY RESULT IN ERRONEOUS OR COMPLETE LOSS OF PITCH AND BANK PRESENTATIONS WITHOUT A VISIBLE OFF FLAG.

Pitch Trim Knob

A pitch trim knob is provided for adjusting the position of the spheroid. The knob is mounted on the lower right-hand side of the indicator display.
It has approximately 1/2 turn of freedom in the clockwise (CW) direction and 1/4 turn of freedom in the counterclockwise (CCW) direction from the zero pitch trim position.
When the knob is rotated to the stop in the CW direction, the spheroid shall rotate to deflect the horizon line upward to indicate a dive of about 15°.
When the knob is rotated to the stop in the CCW direction, the spheroid shall rotate to deflect the horizon line downward to indicate a climb of about 8°.

STANDBY ATTITUDE INDICATOR

A 2 inch standby attitude indicator (refer to Figure 1-28) located on the main instrument panel is installed to provide back-up aircraft attitude indications in case of failure of the attitude indicator.
The indicator electrically powered by the XP5 bus incorporates a vertical gyroscope mounted on a pitch gimbal which in turn is mounted on a roll gimbal. The vertical gyroscope is erected to gravity with erection reducing during accelerations resulting from turns and fore and aft velocity changes. A moving drum, mechanically driven by the pitch gimbal and mounted on the roll gimbal, registers aircraft displacement in the pitch and roll axes with respect to a miniature aircraft. The miniature aircraft is mounted on an adjustable braket and located at the centre of the indicator cover glass. A moveable roll index is fixed to the roll gimbal and moves over a scale fixed to the case.

The attitude presentation is 360° in roll, 87.5° in climb and 82.5° in dive. Rotation of a pitch trim knob located on the bezel adjust the position of the miniature aircraft to 5° in dive and 5° in climb. Pulling the knob cages the gyro with reference to the case and the indicator displays 0° in pitch and roll. This knob, when pulled, should be immediately released. Rotating the knob clockwise while is fully extended, locks the gyro in the caged position. An OFF warning flag comes into view when the gyro is caged or if the power supply is interrupted.

Vertical Velocity Indicator

The vertical velocity indicator (refer to Figure 1-28) is mounted on both main instrument panel. The indicator registers the rate of climb or descent in feet per minute (±6000) and is operated by the static air system.

WARNING LIGHT SYSTEM

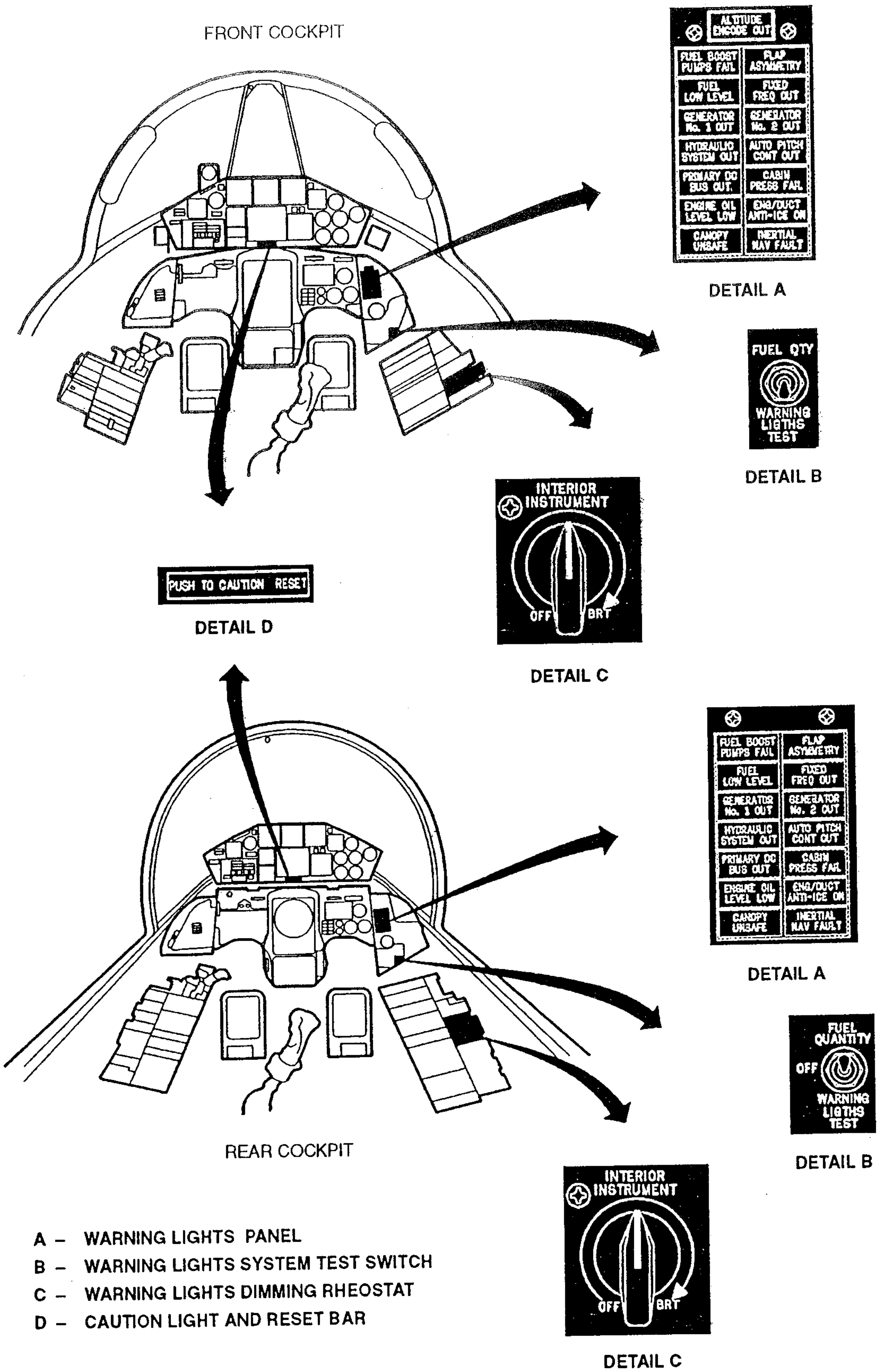
WARNING PANEL LIGHTS SYSTEM

The warning panel lights system (refer to Figure 1-29) gives the pilots visual indication of failure of certain critical equipment and gives an indication of failure or unsafe conditions in critical areas of the aircraft. The system consists of a warning light panel in each cockpit, a CAUTION light and reset bar, and the associated equipment to automatically operate amber placard-type lights on the warning panel and CAUTION bar.
The warning panel contains placard-type warning lights, each having its own operating circuit to indicate a particular condition in the aircraft. If a failure occurs in one of the systems, the warning light for that particular system remains on until the failure is corrected. The warning panel lights system (except for the fire warning lights) is powered by the PP2 bus (refer to the particular system associated with each warning light in this section).

NOTE

With both generators off, all warning lights are inoperative except for the FIRE warning lights and tail hook light.

WARNING LIGHT CONTROLS AND INDICATORS



A – WARNING LIGHTS PANEL
B – WARNING LIGHTS SYSTEM TEST SWITCH
C – WARNING LIGHTS DIMMING RHEOSTAT
D – CAUTION LIGHT AND RESET BAR

FA0131

Figure 1-29

Caution Light

The CAUTION light illuminates when any of the warning panel lights are energized. A rest bar on which the CAUTION light is mounted, permits the pilot to push and deenergize the CAUTION light even though a malfunction continues and the individual warning light stays on. This permits the CAUTION light to indicate a second malfunction if one should occur while the first malfunction is uncorrected.

Warning Lights Panels

NOTE

> The ALTITUDE ENCODE OUT warning light is located on the front cockpit warning light panel only.

The following placard-type warning lights are located on the warning lights panels:

ALTITUDE ENCODE OUT	
FUEL BOOST PUMPS FAIL	FLAP ASYMMETRY
FUEL LOW LEVEL	FIXED FREQ OUT
GENERATOR NO. 1 OUT	GENERATOR NO. 2 OUT
HYDRAULIC SYSTEM OUT	AUTO PITCH CONT OUT
PRIMARY DC BUS OUT	CABIN PRESS FAIL
ENGINE OIL LEVEL LOW	ENG/DUCT ANTI-ICE ON
CANOPY UNSAFE	INERTIAL NAV FAULT

Warning Lights Dimming System

The warning lights dimming circuit provides a means for reducing the brilliance of all warning lights with a single rheostat. The warning light dimming relay coil is connected to the No. 1 emergency dc bus through the WARN LTS. circuit breaker and the instrument lights rheostat. When the rheostat is in the OFF position, as it is for daylight flying, full bus voltage is directed to the warning lights and they burn at maximum brilliance when energized. When the switch is moved from OFF, the warning light dimming relay is energized and bus voltage is directed to the warning lights through a dimming resistor and the lights operate at reduced brilliance. Once the aircraft electrical system has been deenergized the warning light dimming relay automatically returns the warning lights to full brilliance, regardless of the position of the rheostat. To re-dim the warning lights the rheostat must be returned to the OFF position and again moved out of the OFF position. The fire warning lights, canopy unsafe light, engine air inlet temperature warning SLOW light, landing gear unsafe and landing gear indicator lights, lights on the warning light panel, trim lights, radar lights, antiskid light, external stores light, terrain clearance light, and the CAUTION light are dimmed by use of this rheostat.

NOTE

> If the instrument lights rheostat is inadvertently moved out of the OFF position the warning lights and landing gear position lights may not be visible during daylight operation.

Warning Light Test System

The warning light test circuit provides a means for checking warning light filaments simultaneously by a single switch. The warning light test relay coils are tied to the PP2 bus through a circuit breaker and test switch. The fire warning lights are tested with power from the PP5 bus.

Warning Light System Test Switch. A warning light system test switch is located on the right panel in each cockpit (the switch is also used to check the fuel quantity indicating system). When the test switch is moved to the WARNING LIGHTS TEST (down) position the FIRE warning, CANOPY UNSAFE, engine air inlet temperature warning, SLOW landing gear unsafe, landing gear indicator, takeoff trim indicator, antiskid, external stores, terrain clearance, radar, caution, and the warning panel lights are energized.

ENGINE FIRE DETECTOR WARNING SYSTEM

The aircraft is equipped with a system to give visual warning in the cockpit of an overtemperature con-

ENGINE FIRE WARNING LIGHT

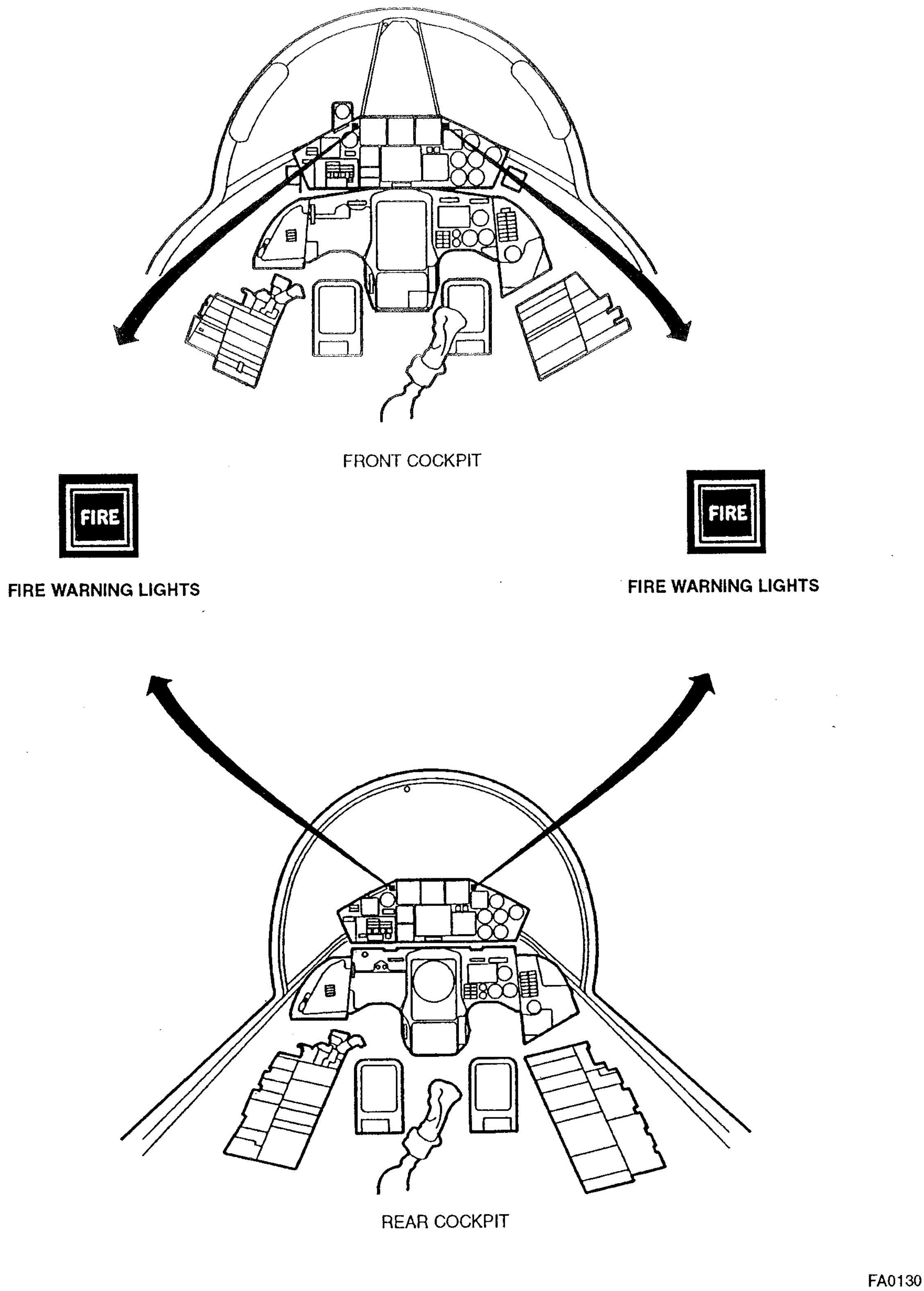


Figure 1-30

dition in the engine compartment or tail section. The system consists of 11 temperature-sensing detectors, in the engine compartment, four detectors in the tail section, and two fire warning lights.
There are two fire warning lights in each cockpit. If any sensors are activated, both lights will illuminate.

Fire Warning Lights

Two FIRE warning lights are located on the main instrument panel (refer to Figure 1-30). The lights are powered by the PP5 bus.
The word FIRE will be illuminated by these lights if any of the overtemperature detectors close their contacts. The detectors in the engine compartment close at 235° C and those in the tail section at 343° C.
Because of the secondary air flow used with this engine installation it is impossible to install a secondary firewall between the hot and cold ends of the engine. Due to the high compression ratio of the engine, the aft end of the compressor section is as hot as the combustion chamber on many other engines.
A secondary firewall would not effectively separate that portion of the engine containing fuel and oil system components from a high temperature region; therefore, no overheat warning lights have been provided.

SURVIVAL PACK

The survival pack (Figure 1-31) serves the dual purpose of seat cushion and container for dinghy and survival equipment. The seat cushion is designed and shaped to give maximum support and comfort to the pilot.
Survival equipment is stowed in a carrier and housed in the forward part (horns) of the glass fibre case giving rigid support to the thighs on ejection. The dinghy included in the pack is a SS. Mk. 5 single seat dinghy and embodies a separately inflatable canopy and floor which together provide protection against the elements.
It is left the decision of the pilot to release the dinghy pack from the harness following ejection and thereby inflating the dinghy automatically. Pulling the survival pack release knob (Figure 1-31) on either side of the pack releases the survival gear from the pilot and inflates the life raft.
If the survival pack was not released before landing, wind will separate the pilot automatically from parachute harness and canopy when he opens the quick release box.
Although in this case the dinghy will be inflated automatically and the survival equipment will be retained, the survival gear and life raft remains attached to the life vest by an approximately 15-foot long lowering line.

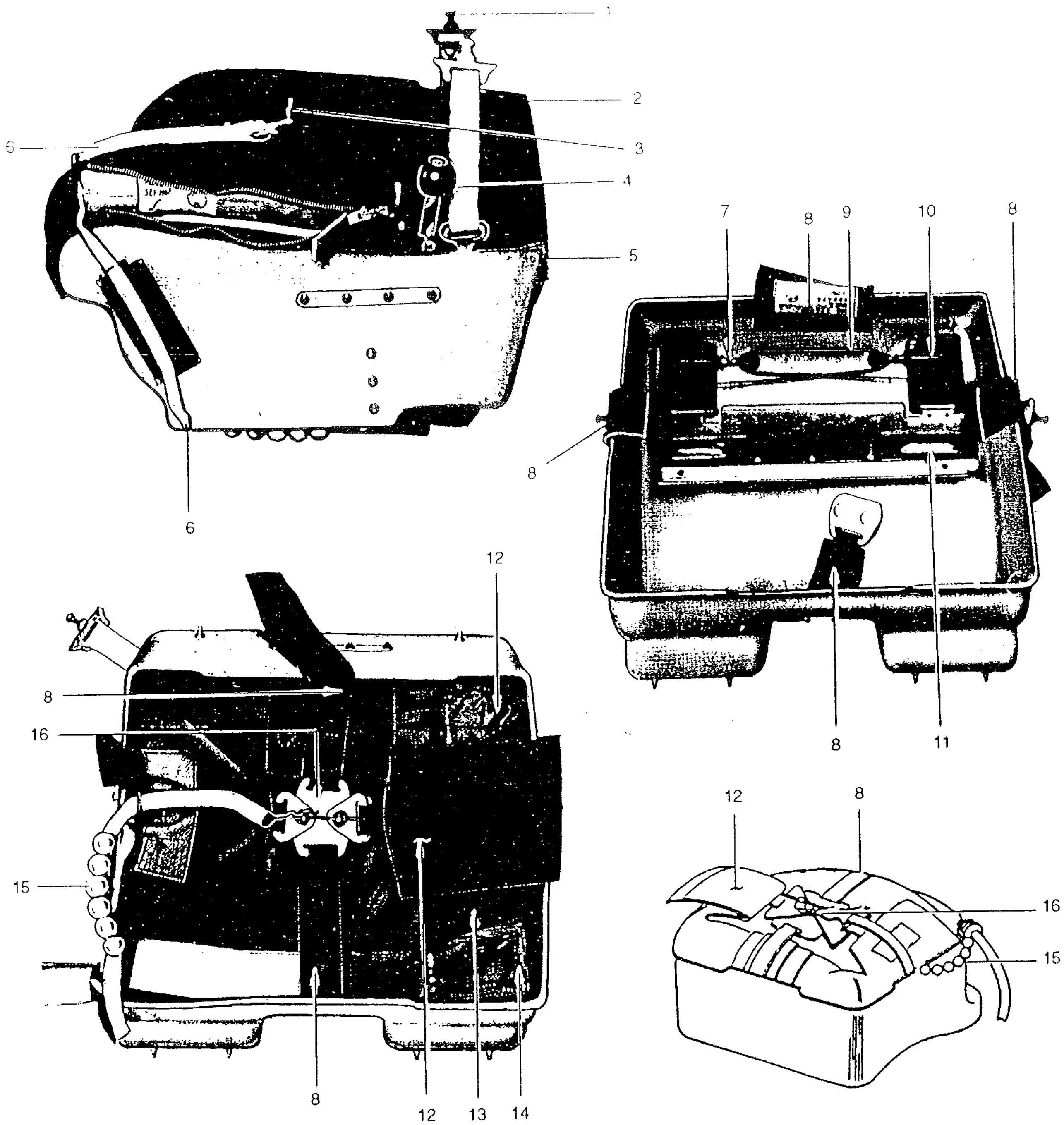
NOTE

> Should the survival pack not be released by pulling the survival pack release knob on either side of the pack before landing, the dinghy may be inflated after landing by pulling sharply the additional manual release handle (Figure 1-31) on the bottom of the survival pack after the survival pack has been released from the parachute harness.

CANOPY

The jettisonable canopy consists of a two pieces (front and rear seat) of transparent plastic secured within a frame which is hinged to the left cockpit sill. Normal operation of the canopy is completely manual. Cartridge-type charges are provided for jettisoning the canopy during an emergency.
When jettisoned, the canopy is released from both sides and is raised about 2 inches above the canopy sills by the canopy unlatching thruster. The canopy unlatching thruster in turn fires the canopy ejector thrusters on the forward sills to ensure upward rotation of the canopy.
From this point the canopy is automatically hinged at the upper rear, allowing it to rotate upward and backward.
The canopy is automatically jettisoned during the pilot escape ejection sequence by actuating either firing handle.

SURVIVAL PACK



1	SURVIVAL PACK ATTACHMENT STRAP	10	LOCK FOR SURVIVAL PACH ATTACHMEN STAP
2	SEAT CUSHION	11	COVER FOR LOCKING MECHANISM
3	LIFE VEST CONNECTION	12	COVER FLAP
4	SURVIVAL PACK RELEASE KNOB	13	CARRIER FOR SURVIVAL EQUIPMENT
5	CONTAINER	14	SERVICING BOOKLET
6	DINGHY LOWERING LINE	15	MANUAL RELEASE HANDLE
7	LOCKING BOLT FOR SURVIVAL PACK	16	SURVIVAL PACK RELEASE ASSEMBLY
8	RESTRAINING STRAP		
9	RELEASE CABLE FOR SURVIVAL PACK		

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Figure 1-31

WARNING

CANOPY LOSS EXPERIENCE HAS SHOWN THAT EVEN IF THE CANOPY IS IN THE OPEN POSITION AND THE CANOPY THRUSTERS ARE NOT IN A POSITION TO MAKE THE THRUSTER PAD CONTACT, IMMEDIATELY JETTISONING THE CANOPY WILL RELEASE THE CANOPY HINGE ON THE LEFT CANOPY RAIL AND GREATLY LESSEN ENGINE FOREIGN OBJECT AND PILOT DISTRACTION.

CANOPY SEAL

An inflatable rubber seal is installed in the edge of the canopy frame and seats against the mating surfaces of the canopy sill and windshield to provide sealing for cockpit pressurization. The seal pressurization switch is actuated by the center canopy latch when the canopy is down and locked. The switches operate a valve which allows engine compressor air to inflate the seal. Seal pressure will be dumped when the canopy is in the unlocked position, or when seat ejection is initiated. Electrical power is supplied to the switches from the PP2 bus.

Canopy Full Open Lock Release

The canopy is released from the full-open position by depressing a small canopy lock release (Figure 1-32) attached to a handle mounted on the right canopy frame. This allows the canopy to be lowered until it comes to rest on two lifter cams which protrude through the right canopy sill and hold the canopy approximately 2 inches from the sill. This places the canopy in position to be locked closed by use of the internal locking lever.

Canopy Internal Locking Lever

A lever (Figure 1-32) located below the canopy sill on the right forward of the cockpit, is used to lock or unlock the canopy. After the canopy has been lowered so that it rests on the lifter cams if may be fully locked by moving the locking lever to the fully locked position. A very positive overcenter feel will be noticed as the lever is moved aft. As the lever is moved aft, the lifter cams retract and the canopy lowers by gravity to the sill where three hooks engage three canopy brackets. These hooks may be observed by the pilot.

CAUTION

- THE CANOPY OPENING AND CLOSING OPERATION SHOULD WORK SMOOTHLY AND EFFORTLESSLY. IF THE CANOPY IS SLAMMED SHUT OR OPEN, THE SYSTEM MAY BE DAMAGED. IF ANY FORCING IS NECESSARY TO FACILITATE HOOK ENGAGEMENT, THE CANOPY IS EITHER OUT OF RIG OR IMPROPERLY FITTED, AND CORRECTIVE ACTION MUST BE TAKEN BEFORE FLIGHT.
- TO PREVENT DAMAGE TO THE CANOPY, A TAXI SPEED OF 50 KNOTS MUST NOT BE EXCEEDED WITH THE CANOPY IN ANY POSITION OTHER THAN FULLY CLOSED AND LOCKED.

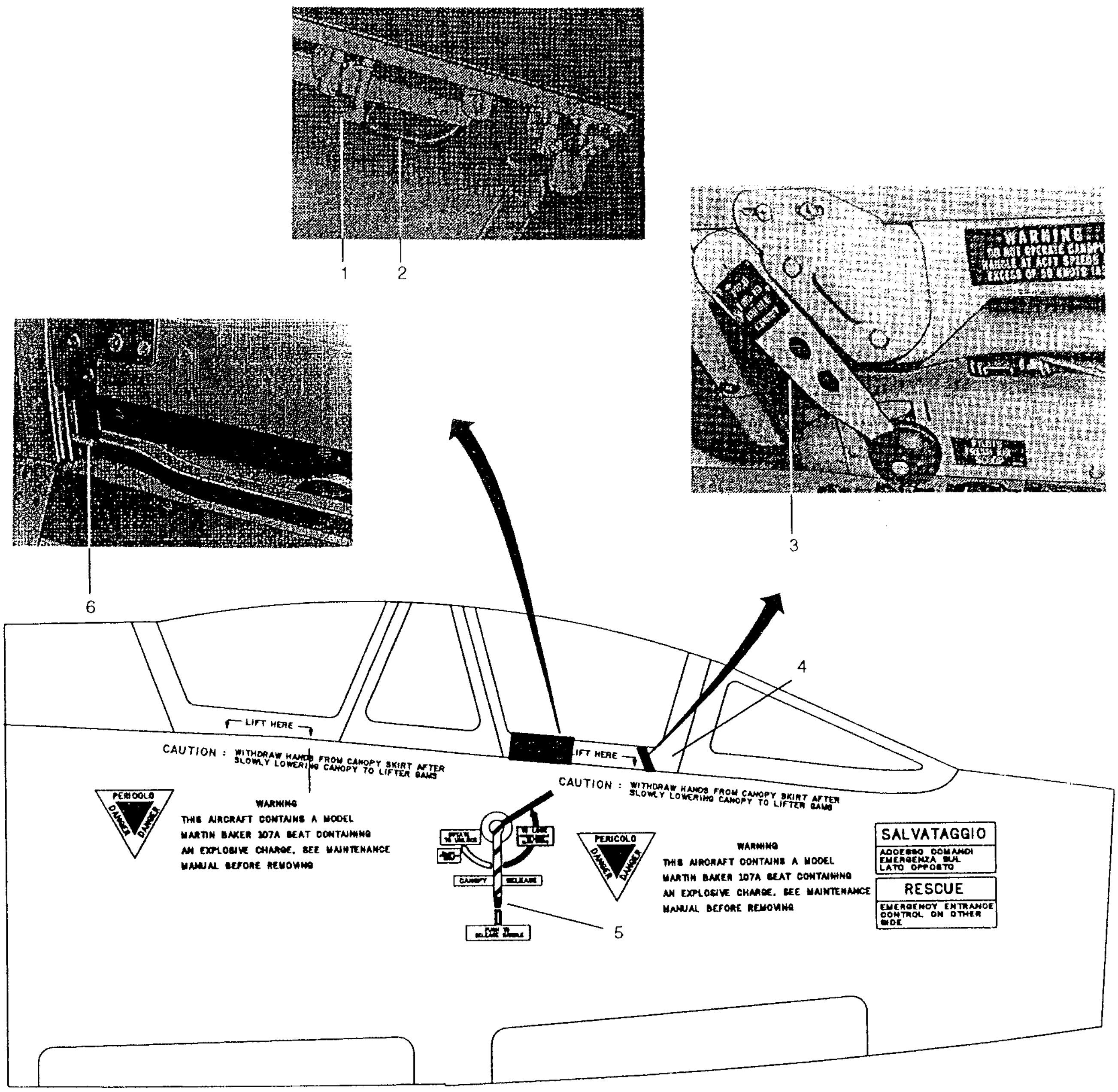
WARNING

DO NOT MANUALLY UNLOCK THE CANOPY IN FLIGHT. DUE TO CANOPY HINGE ATTACHMENT ON THE LEFT CANOPY RAIL, CANOPY SEPARATION IS UNPREDICTABLE AND MAY RESULT IN PILOT INJURY AND/OR ENGINE FOREIGN-OBJECT-DAMAGE.

Canopy External Locking Lever

An external flush-mounted yellow lever (Figure 1-32) provides external control of the canopy identical to the internal locking lever in the cockpit. The external locking lever is placarded CANOPY RELEASE ands is located on the right

CANOPY CONTROLS



NOTE

ELECTRONIC COMPARTMENT COVER LOCKING PIN IS A SAFETY DEVICE WHICH MAKES IT IMPOSSIBLE TO CLOSE AND LOCK CANOPY UNLESS ELECTRONIC COMPARTMENT COVER HAS BEEN PREVIOUSLY CLOSED AND LOCKED (6). IT ALSO PROVIDES ASSURANCE THAT ELECTRONIC COMPARTMENT COVER CANNOT BECOME UNLOCKED AS LONG AS CANOPY IS DOWN AND LOCKED. PIN ATTACHES TO LOCKING LINKAGE ON ELECTRONIC COMPARTMENT COVER AND RETRACTS FLUSH WITH FRAME WHEN COVER IS IN LOCKED POSITION.

1 LOCK RELEASE
2 CANOPY HANDLE
3 INTERNAL LOCK LEVER
4 CANOPY EJECTOR THRUSTER (WITH CANOPY OPEN)
5 EXTERNAL LOCKING LEVER
6 ELECTRONIC COMPARTMENT COVER LOCKING PIN

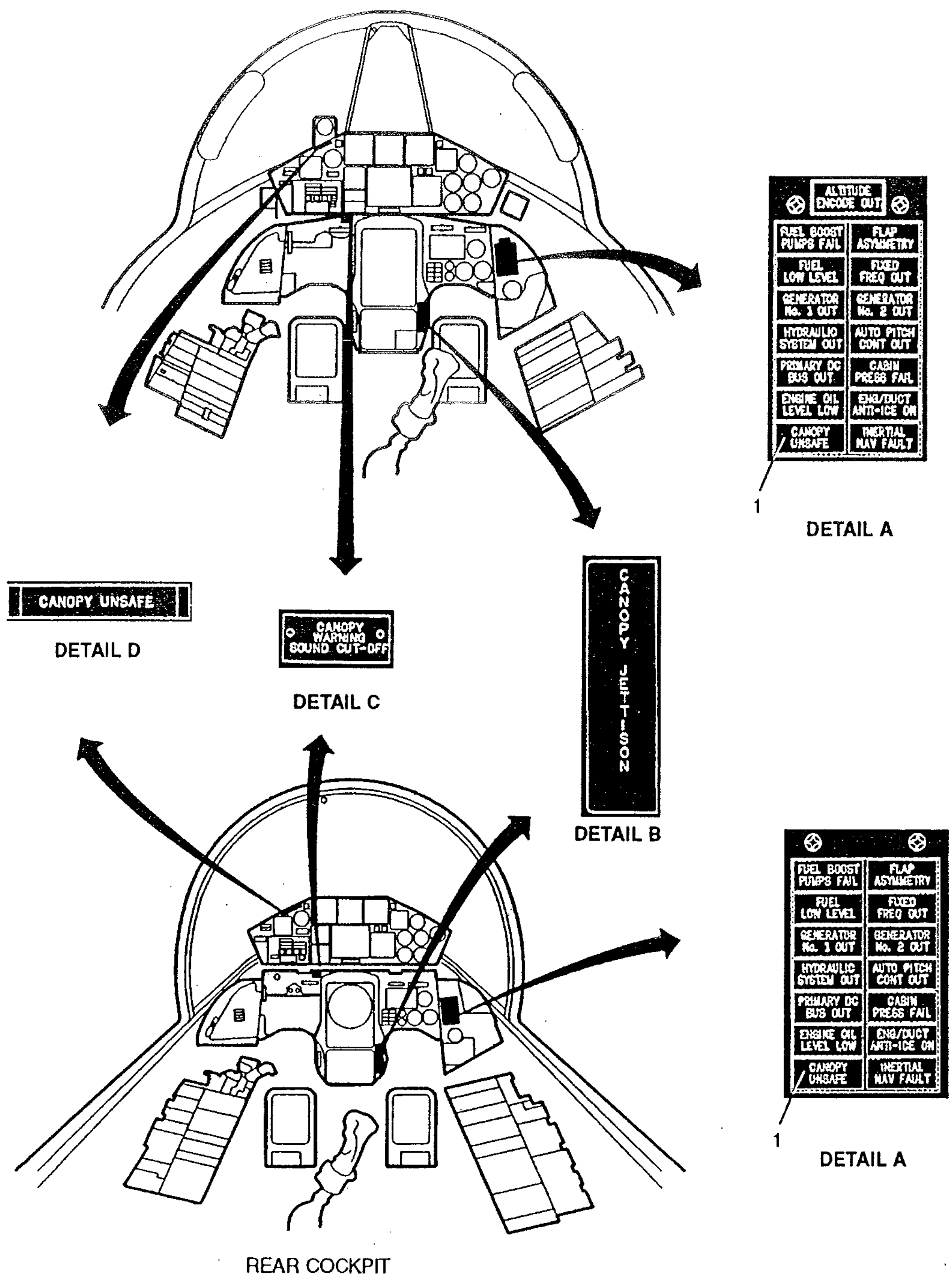
NOTE

THE CANOPY CONTROLS OF THE REAR SEAT ARE THE SAME

FA0113

Figure 1-32

CANOPY CONTROLS AND INDICATORS



A – WARNING LIGHTS PANEL
1 COCKPIT UNSAFE WARNING LIGHT

B – CANOPY INTERNAL JETTISON HANDLE

C – CANOPY WARNING SOUND OUT–OFF SWITCH

D – CANOPY UNSAFE WARNING LIGHT

FA0135

Figure 1-33

side of the fuselage below the windshield. The handle may be extended for use by pushing on the release at the lower end of the handle.

Canopy Internal Jettison Handle

A yellow canopy jettison handle, located on the lower right instrument panel in both cockpits, may be used by the pilot or pilots to jettison the canopy independently of the automatic canopy-seat ejection system. The canopy may be jettisoned by the canopy emergency jettison handle with the ejection seat safety pin installed. A flagged canopy safety pin is provided so that the canopy initiator located to the right side of the seat may be safetied during ground operation.

NOTE

The internal canopy jettison handle in the front cockpit, when pulled will jettison instantaneously only the front canopy. It will not jettison the rear canopy. The internal canopy jettison handle in the rear cockpit, when pulled, will jettison the rear canopy only, 3 seconds after actuation. It will not jettison the front canopy.

Canopy External Jettison Handle

The canopy external jettison handle (Figure 3-1), located on the left side of the fuselage, permits ground rescue personnel to jettison the canopy from the aircraft for emergency entrance. The handle cover is labeled EMERGENCY CANOPY JETTISON ACCESS DOOR. Operating this handle jettisons the canopy, using the same linkage used by the canopy jettison handle to fire the canopy.

WARNING

THE FORWARD CANOPY WILL FIRE FIRST AND THE REAR CANOPY WILL FIRE APPROXIMATELY 3 SECONDS LATER.

Canopy Unsafe Warning Light

If the canopy is not properly locked, microswitches in the canopy locking mechanism and right canopy rail illuminate the CANOPY UNSAFE warning light located on the warning lights panel. An additional CANOPY UNSAFE warning light is installed on the instrument panel glare shield above the accelerometer. A microswitch in the front cockpit throttle quadrant completes the circuit to illuminate the flashing red CANOPY UNSAFE light. The light will flash when the throttle is advanced to a position corresponding to 95% RPM and the canopy(s) is not properly locked. Power for the warning light is derived from the PP2 bus through the existing warning circuit.

Canopy Breaking Tool

The canopy breaking tool, located on the left sill in each cockpit, may be used to break the canopy if other methods of opening fail. No set pattern of blows is necessary, normally three to four blows with the canopy breaking tool will open an adequate escape hole. Grasp the canopy breaking tool with both hands, curved edge toward your face, and put your whole body into an arm-swinging thrust. (With the curved edge away from your face, the tool will glance off the curved canopy and may inflict a head injury.) Aim the tool so as to strike perpendicular to the canopy surface. Always use the point of the tool as blade alignment will determine the direction of the cracks. Reversing the tool, to hammer with butt, products ragged and unpredictable cracking.

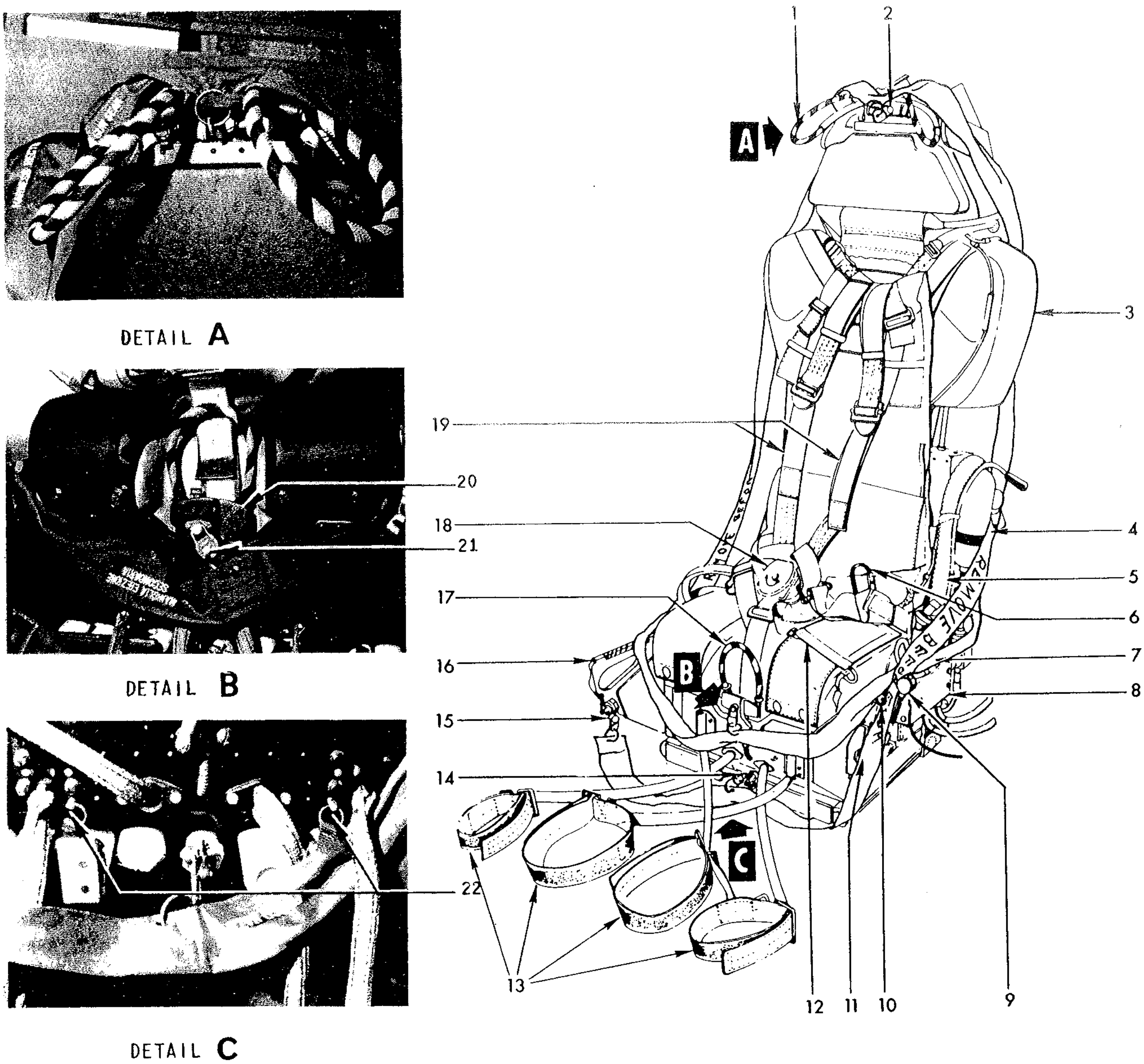
EJECTION SEAT

The Martin Baker rocket assisted ejection seat (Figure 1-34 and Figure 1-35) is designed to provide safe escape under zero speed/zero altitude conditions. It is fully automatic in operation throughout the ejection sequence (Figure 1-36).

WARNING

THE MINIMUM EJECTION SPEED AT ZERO ALTITUDE IS 60 KIAS TO INSURE CANOPY SEPARATION.

EJECTION SEAT

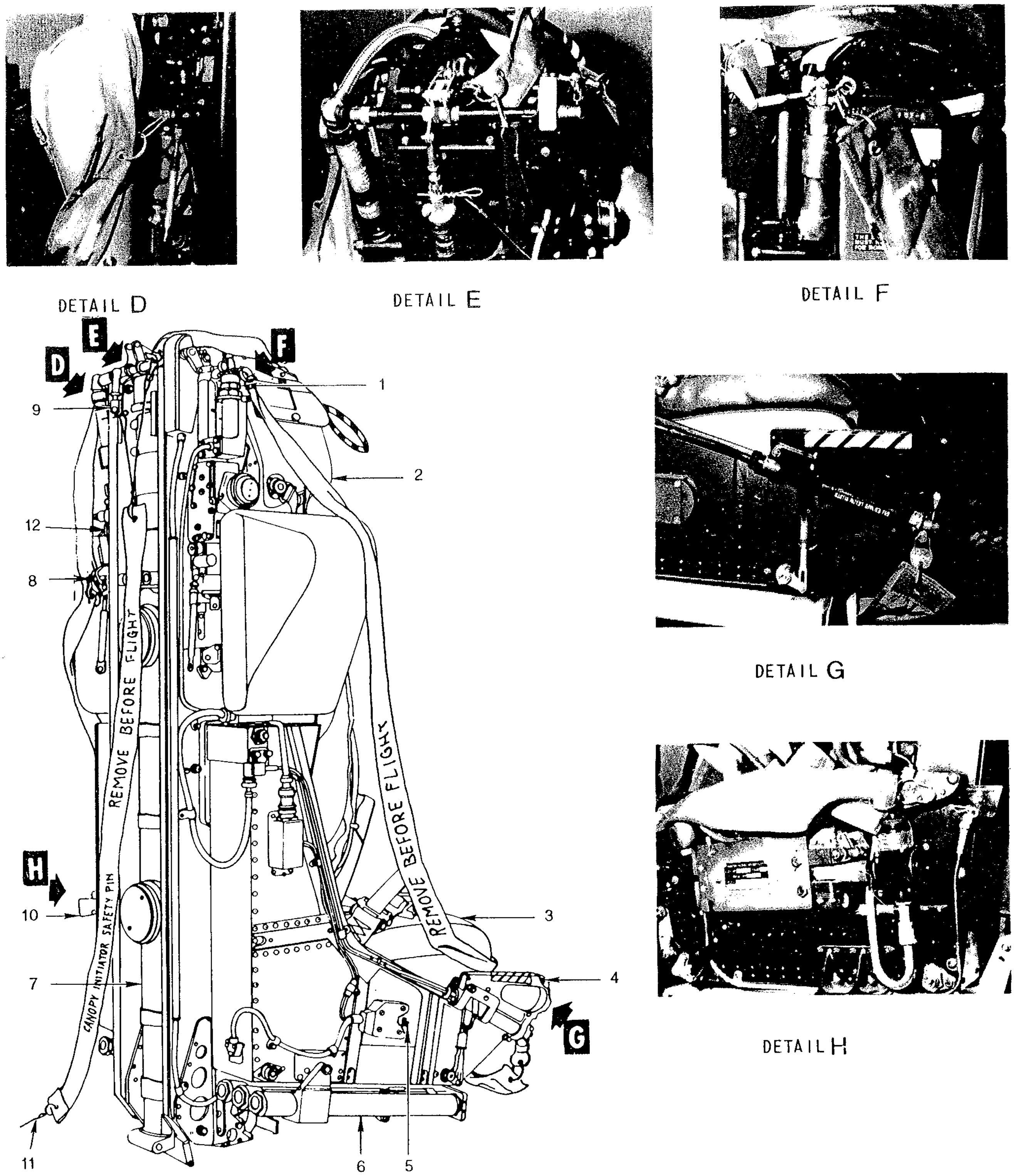


1	PRIMARY FIRING HANDLE	12	DINGHY LOWERING LINE
2	SAFETY PIN	13	LEG RESTRAINING GARTERS AND STRAPS
3	PARACHUTE PACK	14	SAFETY PIN (ROCKET PACK INITIATOR)
4	EMERGENCY OXYGEN BOTTLE	15	SAFETY PIN (GUILLOTINE FIRING UNIT)
5	OXYGEN HOSE	16	MANUAL OVERRIDE HANDLE
6	O–RING (PARACHUTE)	17	SECONDARY FIRING HANDLE
7	PEC–PILOT PORTION	18	QUICK RELEASE BOX
8	PEC–SEAT PORTION	19	COMBINED HARNESS
9	MANUAL RELEASE KNOB (GREEN APPLE)	20	SWIVEL GUARD
10	GO–FORWARD LEVER	21	SAFETY PIN
11	LEG LINE RELEASE LEVER	22	RELEASE RINGS

FA0112

Figure 1-34

EJECTION SEAT



1 SAFETY PIN (POWER RETRACTION)
2 DROGUE PARACHUTE CONTAINER
3 SURVIVAL PACK
4 RELEASE BUTTON
5 SEAT ADJUSTMENT ACTUATOR SWITCH
6 ROCKET PACK
7 MAIN GUN
8 SAFETY PIN (DROGUE GUN)
9 SAFETY PIN OF CANOPY JETTISON INITIATOR
10 OXYGEN RELEASE LEVER AND SAFETY PIN
11 SAFETY PIN OF CANOPY EMERGENCY JETTISON INITIATOR (CANOPY RH SIDE)
12 SAFETY PIN (ROCKET PACK REMOTE CONTROL)

Figure 1-35

FA0114

SEAT EJECTION SEQUENCE

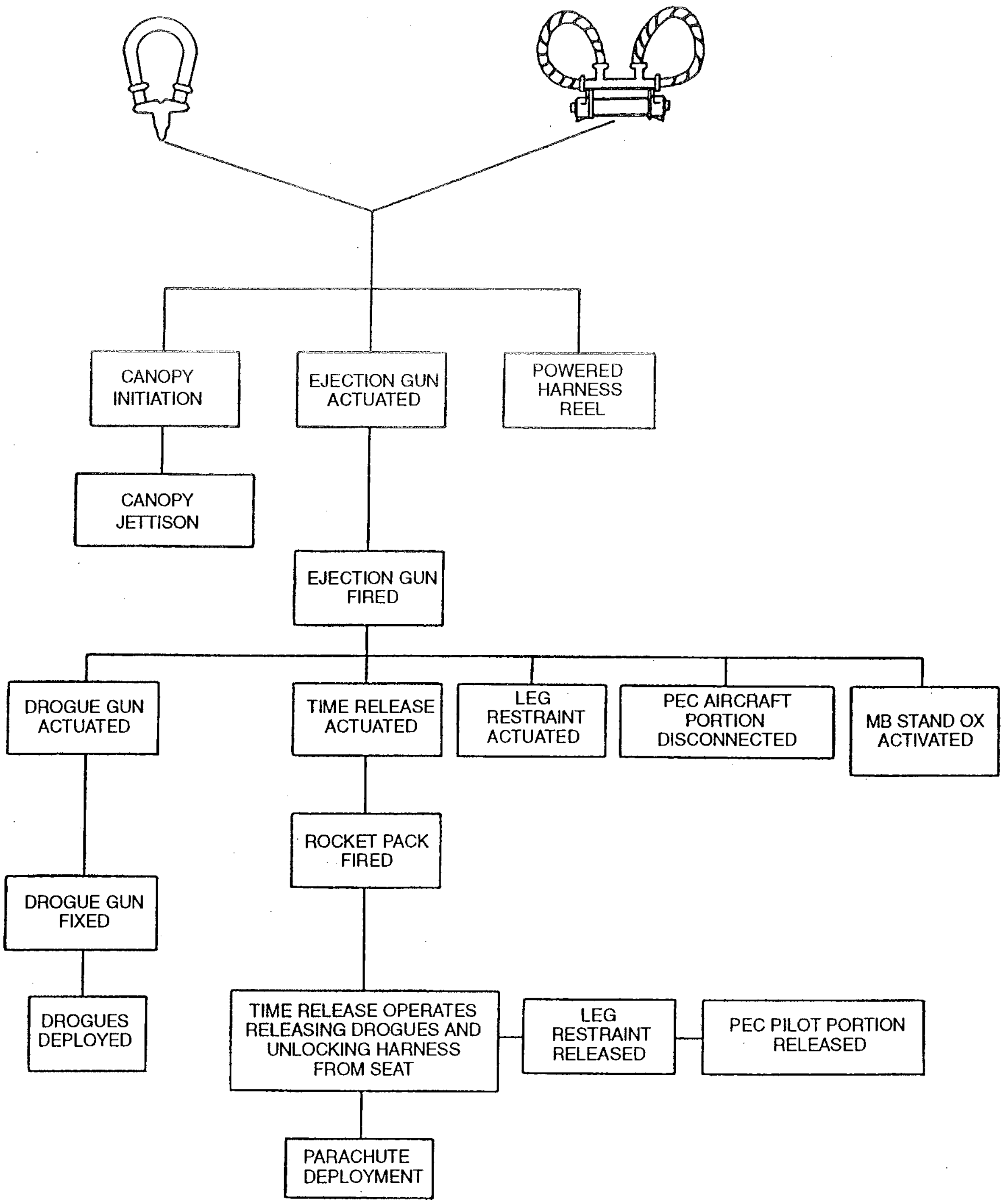


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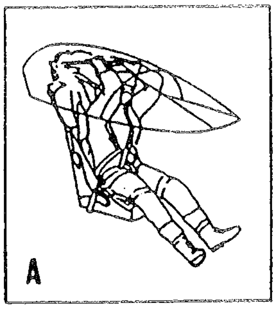
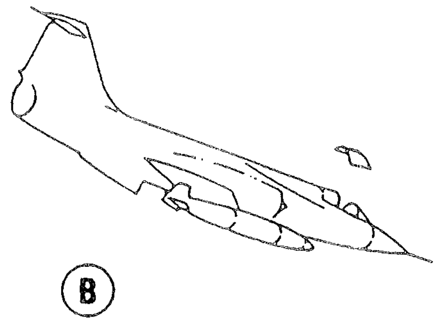
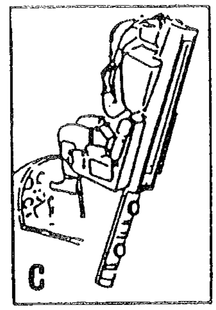
Figure 1-36 (Sheet 1 of 2)

SEAT EJECTION SEQUENCE



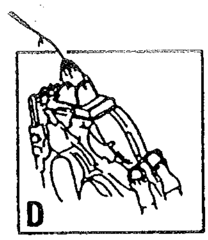




A – PULLING EITHER FIRING HANDLE, THE EJECTION SEQUENCE BEGINS,

B – WHILE THE CANOPY IS JETTISONED AND THE EJECTION GUN IS ACTUATED THE HARNESS POWER RETRACTION UNIT OPERATES AND THE PILOT IS BROUGHT TO THE CORRECT EJECTION POSTURE.

C – AFTER FIRING OF EJECTION GUN, AS THE SEAT ASCENDS, THE DROGUE GUN, THE TIME RELEASE UNIT AND THE EMERGENCY OXYGEN SYSTEM ARE OPERATED AND THE AIRCRAFT PORTION OF THE P.E.C. IS DISCONNECTED. AT THE SAME TIME, THE LEG RESTRAINT CORDS TIGHTEN TO DRAW BACK AND RESTRAIN THE PILOT'S LEGS TO THE FRONT OF THE SEAT PAN. WHEN THE SEAT LEAVES THE AIRCRAFT, THE ROCKET PACK IS FIRED TO SUPLLEMENT THE UPWARD THRUST OF THE EJECTION GUN.

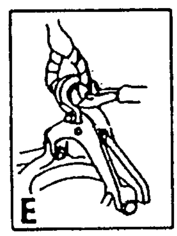


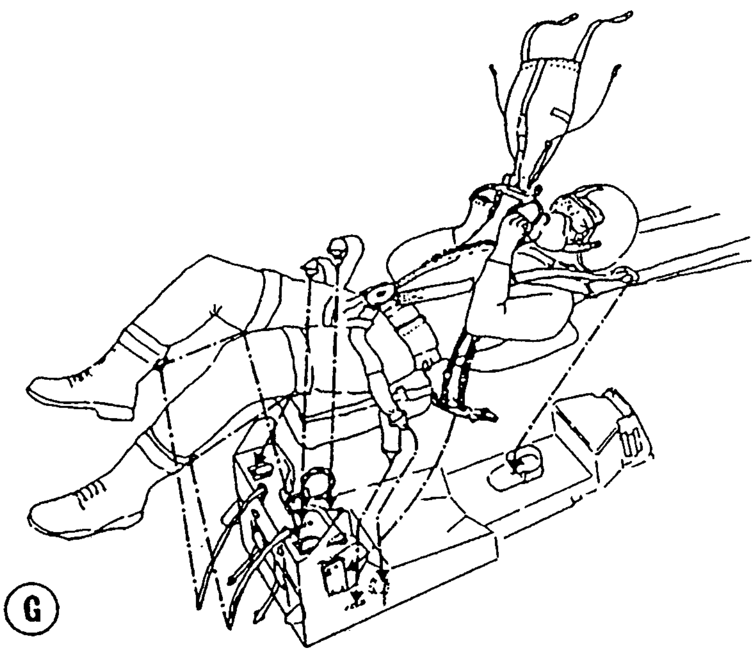
D – 3/4 SEC. AFTER EJECTION, THE DELAY MECHANISM OPERATES AND THE DROGUE GUN IS FIRED DEPLOYING THE DROGUES.

E – THE DROGUES, WHEN FULLY DEVELOPED, STABILIZE AND RETARD THE SEAT AS LONG AS THE CONDITIONS OF HEGHT AND SPEED ARE SUCH THAT THE BAROSTAT DOES NOT ALLOW THE TIME–RELEASE UNIT TO OPERATE.

F – THE TIME–RELEASE UNIT OPERATES, ALLOWING OPENING OF SCISSOR SHACKLE.

G – THE HARNESS LOCKS, THE LEG CORDS AND THE MAIN PORTION OF THE P.E.C ARE RELEASE. AT THE SAME TIME, THE DROGUES DEPLOY THE PARACHUTE AND THE PILOT IS LIFTED OUT OF THE SEAT.





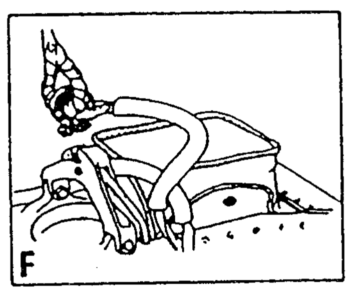


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Figure 1-36 (Sheet 2 of 2)

EJECTION GUN AND GUIDE RAILS

The telescopic ejection gun provides the initial power for ejection of the seat and pilot from the aircraft, the guide rails mounted on opposite sides of the outer tube serving as guides to the seat. The ejection gun comprises three tubes, namely outer tube, intermediate tube and inner tube. The intermediate and inner tube are telescoped inside the outer tube. Power is obtained from one primary and two auxiliary cartridges.
The ejection gun thrust is imparted against a top cross beam. The top latch piston assembly is located on the left side of this cross beam. The piston secures the seat to the ejection gun and prevents the seat from riding up the rails during negative G flight. The plunger on the internal portion of the top latch piston is forced aside as the ejection gun is fired allowing the inner tube to thrust against the top cross member and lift the seat.

ROCKET PACK

The thrust of the ejection gun augmented by the rocket pack, consists of a series of tubes arranged in two banks and connected to a common combustion chamber fitted with discharge nozzles. Situated in the front bank is the breech unit fired by a static line attached to the base of the ejection gun.

DROGUE GUN AND DROGUE SYSTEM

The drogue gun, incorporating a time delay mechanism, fires a metal piston which in turn extracts the small controller drogue and the main stabilizer drogue from the drogue container.
The drogues, when deployed, tilt the seat into a horizontal attitude thus insuring deceleration approximately in line with the seat axis. Between the controller and the main drogue is a nylon line which allows the controller drogue to be fired clear of the seat wake.

TIME RELEASE MECHANISM

The ejection seat is fitted with a time release mechanism the function of which is to release the harness locks, thereby unfastening the pilot from the seat, and to unlock the scissor shackle by which the drogues are connected to the seat, thereby enabling the drogues to stream the main parachute.
For high altitude ejections a barostatic control prevents operation of the time release mechanism until the seat and pilot have descended to an altitude of approximately 10000 ft, stabilized by the drogue chute system. This prevents prolonged exposure to low temperature and rarefied air, and enables the pilot to descend quickly to a more tolerable altitude, while strapped into the seat, stabilized and controlled by the drogues and supplied automatically with emergency oxygen before the main parachute is deployed.

MAIN PARACHUTE AND COMBINED HARNESS

The main parachute (Figure 1-37) is stowed in a rigid pack behind the pilots shoulders. The parachute harness and seat harness are combined and connected to a single quick release box.
The harness has a three point attachment to the seat: two locks in the seat pan securing the lap harness and the negative "G"-strap, and a third lock on the central cross member which secures the shoulder harness with roller brackets through the looped straps.

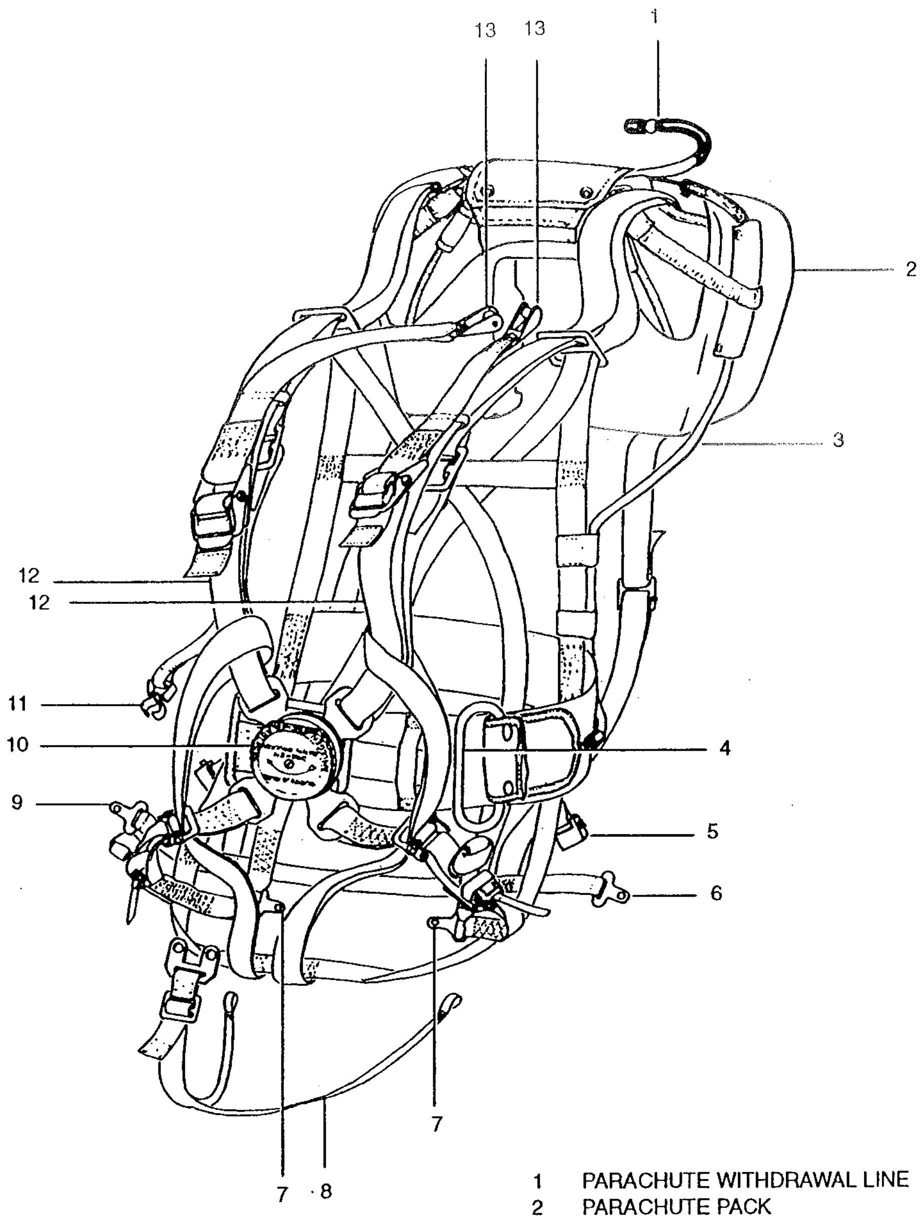
AUTOMATIC LEG RESTRAINT

The leg restraining system is fitted to the ejection seat to draw back and restrain the pilots legs close to the seat pan during ejection. The system consists of leg restraining cords, snubbing units, leg cord locks and leg restraint garters. The lower end of the leg restraining cords contain a fitting by which they are attached to the aircraft structure, each fitting containing a shear rivet designed to fail under a load of approximately 400 pounds. From the structure each cord passes over a roller, through a snubbing unit and out through the front of the seat pan where it terminates in a metal end fitting, which after passing through "D" rings on the garters is plugged into a lock on the front of the seat pan. Provision is made on each snubbing unit to allow the pilot to adjust the leg restraining cords individually to give comfortable leg movement. The leg line release lever (Figure 1-34) is located on the left hand side of the seat pan.

POWER RETRACTION UNIT

The function of the cartridge operated power retraction unit is to ensure that, regardless of the pilots position when ejection is initiated, he will be positioned and retained in the correct ejection posture before the seat commences to move and "G" forces are applied. During flight the pilot can, by operating the go-forward lever (Figure 1-34) on the left side of the seat pan, lean forward and backward at will

MAIN PARACHUTE AND COMBINED HARNESS



1 PARACHUTE WITHDRAWAL LINE
2 PARACHUTE PACK
3 RIPCORD CABLE
4 D–RING
5 SURVIVAL PACK QUICK RELEASE FITTING
6 STICKER STRAP LUG
7 LOWER HARNESS LUG
8 NEGATIVE G–STRAP
9 STICKER STRAP LUG
10 QUICK RELEASE BOX
11 SAFETY CLIP
12 COMBINED HARNESS
13 ROLLER BRACKETS

FA0128

Figure 1-37

but any sudden forward movement by the pilot will be prevented by an inertia operated clutch mechanism.
Operation of the go-forward lever to the full rear position from which it will automatically move to the center position raises an inertia pawl clear of the ratched teeth on the port webbing drum permitting free forward and backward movement of the pilot. Should the seat then be subjected to any severe deceleration or acceleration forces (e.g., crash landing or ejection) then the "G" forces will rotate the pawl downwards and thus lock the webbing reels.
Upon initiation of the ejection seat, the cartridge operated power retraction mechanism retracts the shoulder harness so that the pilot is forced into a suitable position for ejection prior to seat movement.

MANUAL OVERRIDE HANDLE AND GUILLOTINE SYSTEM

The manual override handle (Figure 1-34) is mounted on the right side of the seat pan and is locked by a push button catch mechanism. The operation of the handle opens in a single action the three harness locks, the PEC-pilot portion, the leg line locks, and initiates the guillotine firing unit by which the parachute withdrawal line is cut thus enabling the pilot to leave the seat manually with the personal parachute.
During automatic separation the withdrawal line is pulled out of the guillotine body by the drogue parachutes.

PERSONAL EQUIPMENT CONNECTOR (PEC)

The personal equipment connector (Figure 1-38) couples the personal leads of the pilot to the appropriate aircraft supplies in a single action thus ensuring simultaneous connection.
During ejection all personal leads except emergency oxygen are disconnected from the aircraft supply and sealed, emergency oxygen being provided via a separate PEC-inlet to the pilots oxygen hose. Five supplies are incorporated in the PEC:

a. Tele/mic (and exhalation valve heating when partial pressure helmet is used)

b. Anti-G suit supply

c. Suit air ventilation

d. Main oxygen supply

e. Emergency oxygen supply

The personal equipment connector is mounted on the left hand side of the seat pan. A green apple is located on the front part of the PEC. When pulled by the pilot, emergency oxygen will be supplied during flight.
The emergency oxygen system mounted on the seat automatically provides oxygen supply from ejection until seat/man separation.

SEAT ADJUSTMENT MECHANISM

The height of the seat pan may be adjusted by an electrical actuator. The seat adjustment actuator switch is located on the right hand side of the seat pan (Figure 1-35).
Power for seat adjustment of the seat is derived from the XP2 bus.

CAUTION

DO NOT OPERATE THE SEAT ADJUSTMENT MECHANISM FOR MORE THAN 30 SEC WITHIN 10 MINUTES OF TIME.

FIRING CONTROLS

The ejection seat is provided with two firing controls (Figure 1-35), the upper firing handle located at the front of the drogue container, and the lower firing handle located at the front edge of the seat pan. The lower firing handle may be used if due to injuries, acceleration forces or other circumstances the upper firing handle cannot be reached.

SAFETY PINS

There are 10 safety devices (9 safety pins and 1 swivel guard). The pilot has to rotate the swivel guard of the lower firing handle to "down" position for flight (Figure 1-35).
Maintenance personnel have to remove the following safety pins:

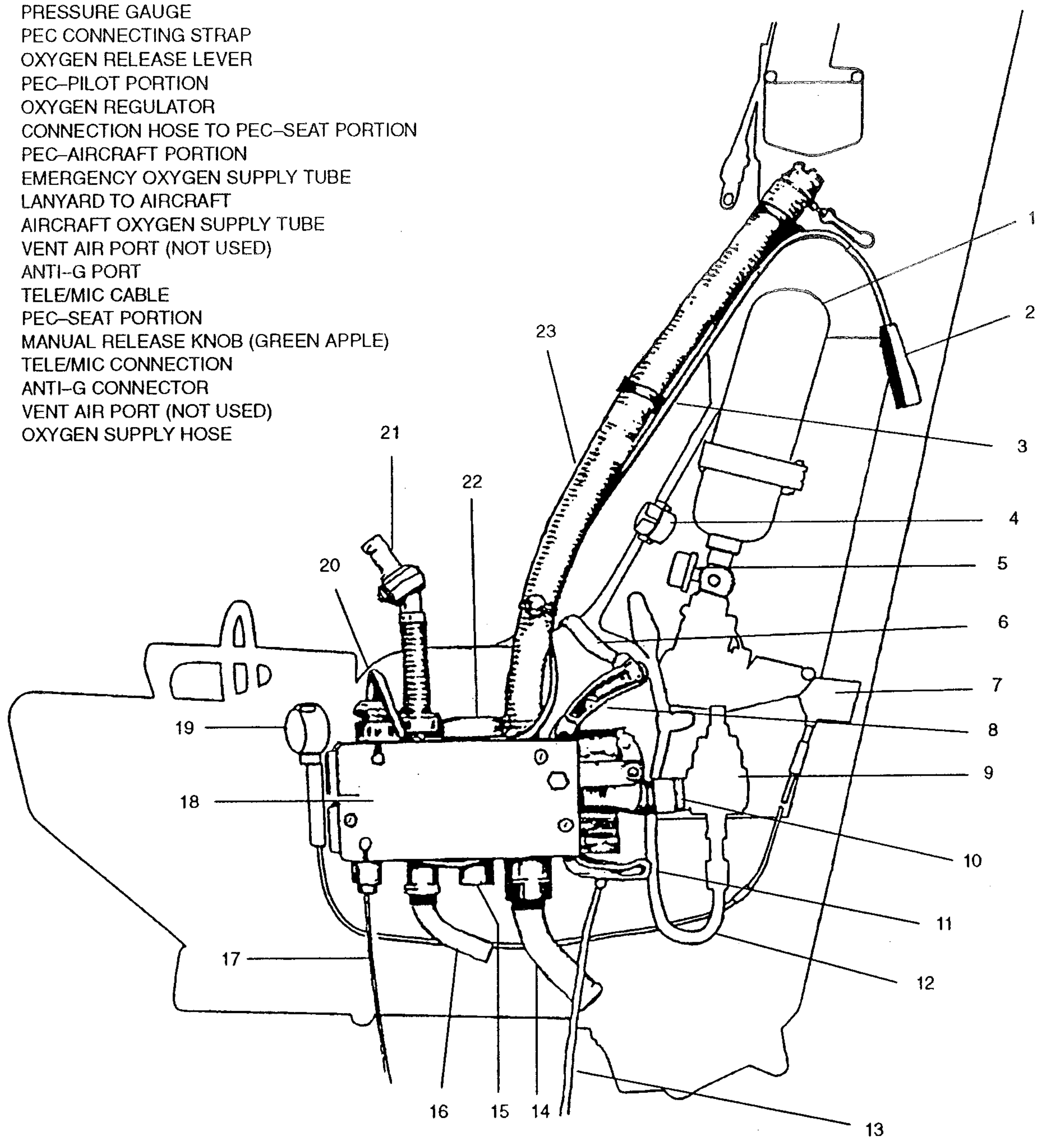
- Safety pin of upper firing handle (after completion of Pilot/Crew Chief Check) (Figure 1-35, Detail A)

- Drogue gun (Figure 1-34, Detail E)

- Main gun (Figure 1-34, Detail B)

PERSONAL EQUIPMENT CONNECTOR (PEC) AND STANDARD EMERGENCY OXYGEN SUPPLY SYSTEM (MB-STAND OX)

1 EMERGENCY OXYGEN BOTTLE
2 TELE/MIC PLUG
3 OXYGEN SUPPLY HOSE
4 DUST COVER FOR OXYGEN SUPPLY HOSE
5 PRESSURE GAUGE
6 PEC CONNECTING STRAP
7 OXYGEN RELEASE LEVER
8 PEC–PILOT PORTION
9 OXYGEN REGULATOR
10 CONNECTION HOSE TO PEC–SEAT PORTION
11 PEC–AIRCRAFT PORTION
12 EMERGENCY OXYGEN SUPPLY TUBE
13 LANYARD TO AIRCRAFT
14 AIRCRAFT OXYGEN SUPPLY TUBE
15 VENT AIR PORT (NOT USED)
16 ANTI–G PORT
17 TELE/MIC CABLE
18 PEC–SEAT PORTION
19 MANUAL RELEASE KNOB (GREEN APPLE)
20 TELE/MIC CONNECTION
21 ANTI–G CONNECTOR
22 VENT AIR PORT (NOT USED)
23 OXYGEN SUPPLY HOSE



FA0129

Figure 1-38

- Canopy jettison initiator (Figure 1-34, Detail F)
- Power retraction (Figure 1-34, Detail C)
- Guillotine (Figure 1-34, Detail D)
- Rocket pack (Figure 1-34, Detail B)
- Lower firing handle (Figure 1-35, Detail B)
- Oxygen release lever (Figure 1-34, Detail A)

The following six safety pins will be stowed in a case on the left side of the cockpit (at the throttle control housing) during flight:

- Upper firing handle
- Lower firing handle
- Rocket pack
- Guillotine
- Main gun
- Power retraction

The safety pins of drogue gun, canopy jettison initiator, initiator breech, and oxygen release lever; which are attached to the red streamer (Figure 1-35), will be stowed in the map case of the cockpit during flight.

EMERGENCY OXYGEN SUPPLY SYSTEM

The system consists of an emergency oxygen bottle, pressure gage, pressure reducer valve, oxygen regulator and a valving mechanism (Figure 1-38).

AIR-CONDITIONING AND PRESSURIZATION SYSTEM

COCKPIT AIR CONDITIONING

Heated, compressed air for cockpit air conditioning and pressurization is obtained by bleeding air from the 17th (last) stage of the engine compressor (Figure 1-39).

After passing through a primary heat exchanger, a small part of the air is directed to the fuel tank pressurization system. The main flow of a air passes through a shutoff valve, after which a portion goes to the rain-remover duct, canopy and electronics compartment, anti-G suit, radar pressurization.

The remainder then passes through or around a refrigeration unit, depending on the position of the bypass valves. The compressor air which goes through the bypass valves is directed to an air mixing chamber where it mixes with the air which has gone through the refrigeration unit.

This mixture is directed through a water separator and enters the cockpit through body outlets and foot warmers. The temperature of the air entering the cockpit depends upon the position of the bypass valves.

For maximum heating, the bypass valves will be fully open and most of the air entering the cockpit will bypass the refrigeration unit.

For maximum cooling, the bypass valves will be completely closed and all the air entering the cockpit will pass through the refrigeration unit, which includes a secondary heat exchanger, a water boiler, and a cooling turbine. The water boiler operates in such a manner that if the inlet temperature is above water-boiling temperature, the water will boil and the air will be cooled through an evaporation process.

The temperature of the air entering the cockpit may be varied by the pilot's cockpit heat rheostat in the front cockpit only, which controls the position of the bypass valves. In normal operation, air temperature is maintained between 4° C and 38° C automatically by means of an electronic temperature control system which senses cockpit temperature, compares this temperature with the cockpit temperature selector demand, and sends electrical signals to the bypass valves to change their positions as necessary.

The pilot may also control the temperature manually by means of the cockpit temperature mode selector switch. When this switch is in any position except automatic (AUTO), the thermostat control is cut out of the system, except that the maximum duct air temperature is limited.

Manual control signals then bypass the electronic controller and position the temperature control valves electromechanically, depending on the position to which the pilot operates the control switch.

The system is powered through XP2 and PP2 busses.

AIR CONDITIONING AND PRESSURIZATION SYSTEM

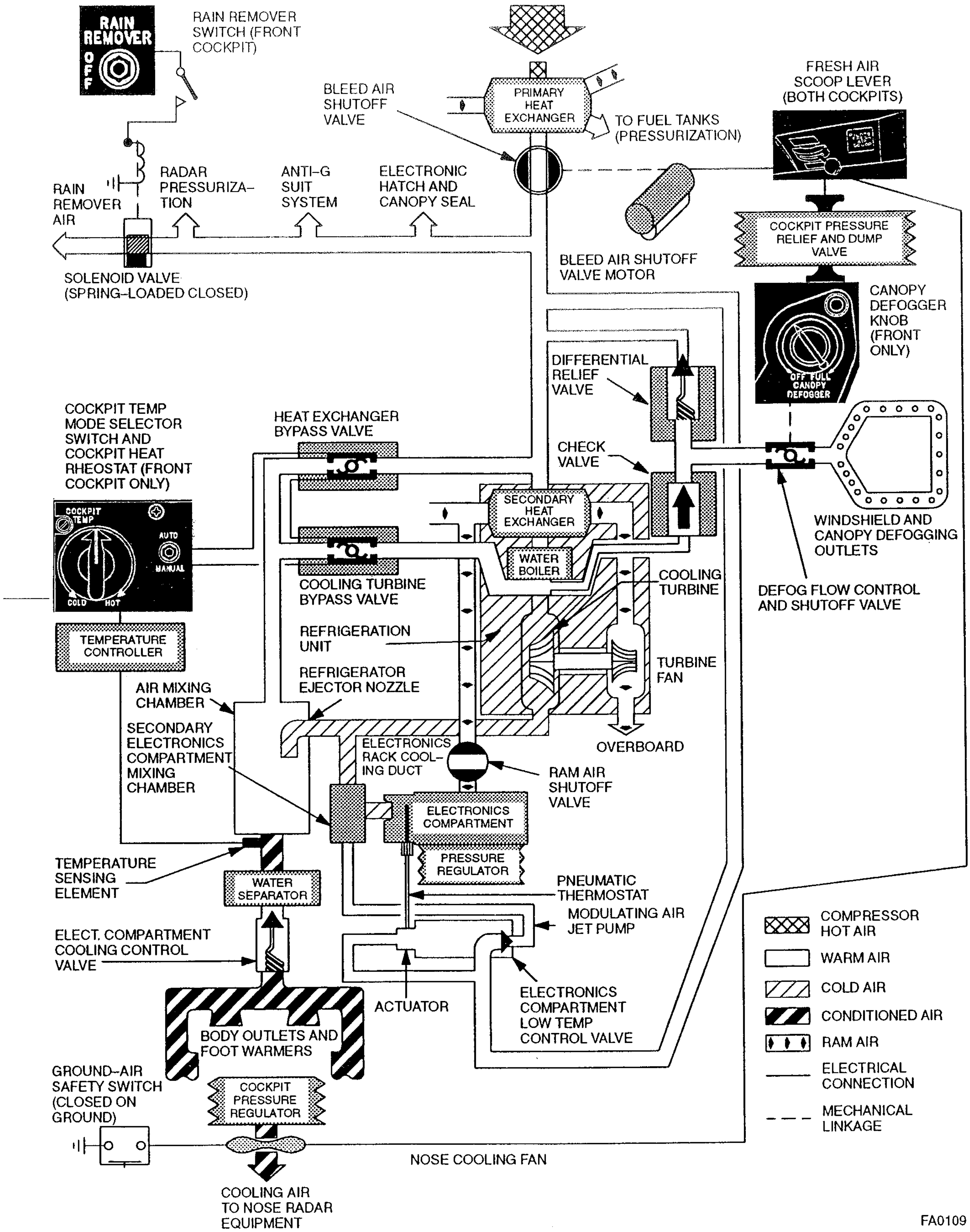


Figure 1-39

COCKPIT PRESSURIZATION

Cockpit pressurization is maintained at the proper level by an automatic cockpit pressure regulator located in the left forward area.
Below 5000 feet there is only slight pressure in the cockpit caused by forcing cabin exhaust air through the radar package.
Between 5000 and 18350 feet, the cockpit altitude will be constant, while the differential pressure will vary from 0 to 5.0 psi (see Figure 1-40).
Above 18350 feet, cabin pressure is maintained at 5.0 psi differential, regardless of aircraft altitude. Exhaust air from the cockpit pressure regulator is ducted through the radar equipment forward of the cockpit for cooling purposes.
The cockpit pressure regulator unit incorporates a cooling fan which forces cockpit air into the radar compartment when the aircraft is on the ground and the aircraft electrical system is energized. This fan is also actuated, when the aircraft is airborne, whenever the fresh-air scoop is opened. If the pressure regulator malfunctions, excessive cabin pressure will be relieved through the cockpit pressure relief and dump valve.
If the cockpit air becomes contam inated, the pilot may open the fresh-air scoop to alleviate this condition. Opening the fresh-air scoop allows outside ram air to enter the cockpit, shuts off the flow of compressor air into the cockpit, and releases cabin air overboard through the cockpit pressure dump valve.
With engine operating, pressurization may be checked on the ground with the canopy locked by pulling the LANDING GEAR CONT circuit breaker, located on the left console in the front cockpit only and noting a slight pressure rise on the cabin altitude indicator.

Cockpit Temp Mode Selector Switch and Cockpit Heat Rheostat

The cockpit heat rheostat located in front cockpit only (Figure 1-41) is used in conjunction with the cockpit temperature mode selector switch (Figure 1-41) (both located on the heating control panel) to control cockpit temperature. These switches are powered from the XP2 bus.
Under normal conditions the selector switch is used in the AUTO (automatic) mode with the heat rheostat set at any point between COLD and HOT. Cockpit temperature will then be maintained automatically through the temperature control unit. With the switch in MANUAL, the bypass valves are positioned in response to the cockpit heat rheostat. It requires approximately 10 seconds for an excursion of the valves from full HOT position to full COLD position in the manual mode.
Operation of the system in the manual mode causes the low temperature limiter to drop out of the system. Operation in manual full cold, or nearly full cold, at altitudes where humidity is significant may cause freezing in the water separator. If this happens, airflow to the cabin will be reduced, and total cooling may be less than it would be if duct temperatures were held above freezing.

COCKPIT PRESSURIZATION SCHEDULE

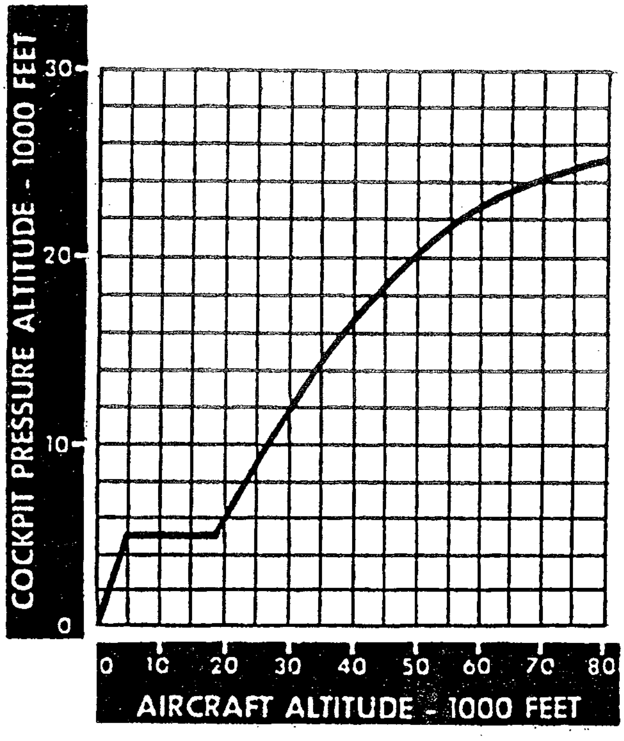
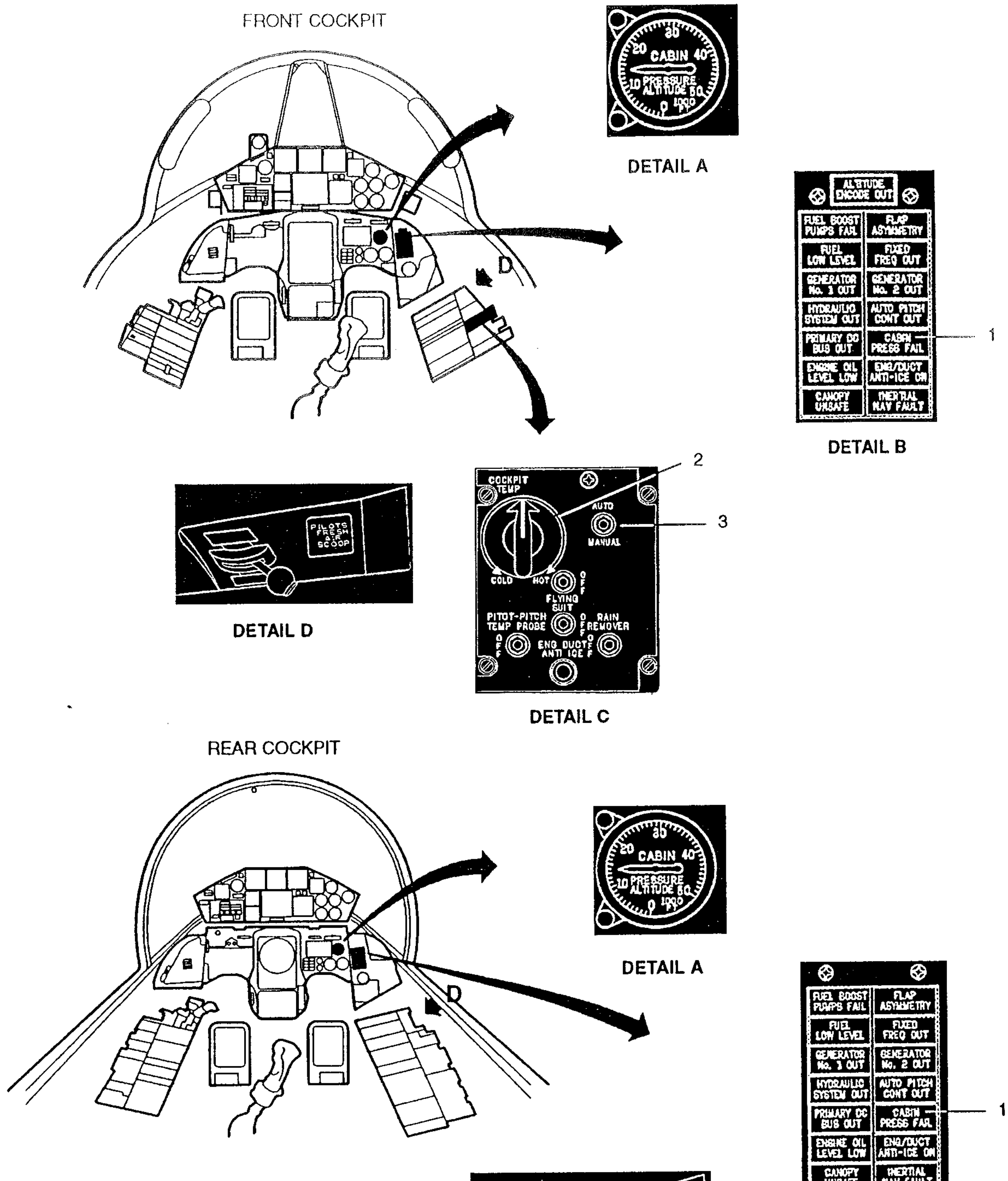
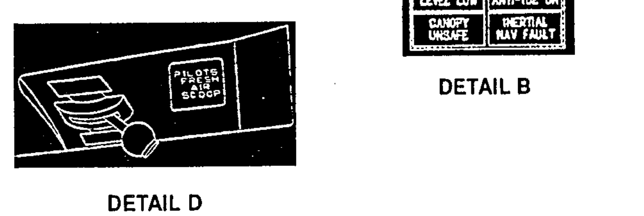


Figure 1-40

HEATING CONTROL PANELS





A – CABIN ALTIMETER

B – WARNING LIGHTS PANEL
1 CABIN PRESS FAIL WARNING LIGHT

C – HEATING CONTROL PANEL
2 COCKPIT HEAT RHEOSTAT
3 COCKPIT TEMPERATURE MODE SELECTOR SWITCH

D – FRESH AIR SCOOP LEVER

FA0138

Figure 1-41

CAUTION

DO NOT USE THE FULL COLD POSITION AT IDLE POWER SETTINGS OR DURING GROUND OPERATIONS. THE COMBINATION OF LOW FLOW, HIGH AMBIENT TEMPERATURES, AND HIGH HUMIDITY MAY CAUSE FREEZING TO OCCUR IN THE WATER SEPARATOR.

NOTE

It should be recognized that, while duct temperature response to revised selector setting is immediate, cockpit air temperature cannot stabilize in less than approximately 2 minutes due to the change in the temperature level of the metal and nonmetal masses in the cockpit area.

Fresh Air Scoop Lever

The fresh air scoop lever is located outboard of the right console in both cockpits. Depressing a button on the handle of the lever allows forward movement which opens the scoop.
Aft movement of the lever closes the fresh-air scoop but does not require depressing the button; however, the button must be depressed before the lever may be placed in the CLOSED position (last aft detent).
Initial movement of the lever out of the last aft detent actuates a switch which causes PP2 bus power to close the bleed air shutoff valve, to open the cockpit relief and dump valve, and to energize the nose cooling fan.

NOTE

Closing the bleed air shutoff valves shuts off bleed air to the air-conditioning package, rain-remover system, and the gun purge system as well as shutting off pressurization to the radar.

Forward movement of the lever out of the last aft detent into the next detent does not open the scoop; however, further forward movement of the lever will open the scoop.

CAUTION

- DO NOT ATTEMPT TO OPERATE THE RADAR WITH THE FRESH-AIR SCOOP CLOSED AND THE HANDLE OUT OF THE LAST DETENT. IN THIS CONDITION THERE IS A NEGATIVE PRESSURE IN THE COCKPIT AND THE NOSE COOLING FAN WILL NOT DEVELOP SUFFICIENT PRESSURE TO FORCE COOLING AIR THROUGH THE RADAR NOSE PACKAGE.

- DURING GROUND OPERATION WITH THE ENGINE OPERATING AND THE FRESH-AIR SCOOP OPENED, THE SUPPLY OF COOLING AIR TO THE ELECTRONIC COMPARTMENT IS SHUT OFF. THIS MAY CAUSE THE ELECTRONIC EQUIPMENT TO OVERHEAT SINCE SUFFICIENT RAM AIR IS NOT AVAILABLE. THEREFORE, VERIFY THAT THE FRESH AIR SCOOP HANDLE IS DEFINITELY IN THE LAST AFT DETENT TO ENSURE THAT AIR IS AVAILABLE FOR COOLING THE ELECTRONIC COMPARTMENT AND FOR COCKPIT PRESSURIZATION.

ELECTRONICS COMPARTMENT COOLING

The electronics compartment is cooled by cold air from the cooling turbine. Minimum temperature of air entering the electronics rack is kept at approximately 27° C. Under most ambient conditions moisture will then be prevented from condensing onto the surfaces of the electronics equipment.

At low engine RPM and electronics compartment cooling control valve, located downstream of the water separator, moves toward the closed position which decreases the airflow through body and foot warmer outlets and vents a greater proportion of cooling air to the electronics compartment. In the event bleed air pressure is lost (flameout, engine seizure, etc.) or shut off by the pilot by opening the fresh-air scoop, a cooling air emergency shutoff valve automatically opens, diverting outside cooling air to the electronics compartment.

Cabin Altimeter

The cabin altimeter located on the right side of the lower instrument panel, is vented to the inside of the cockpit only.
This instrument gives an accurate indication of the cockpit pressure altitude.

Cabin Pressure Fail Warning Light

A warning light, placarded CABIN PRESS FAIL, is located on both warning light panels to illuminate when cabin pressure equals a pressure altitude of 40000 (±2000) feet.
A pressure chamber, located in the cockpit forward of the instrument panel, is the sensing device which activates the warning light circuit. The warning light is powered by the PP2 bus.

NORMAL OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

a. Fresh air scoop lever – CLOSED

b. Cockpit temp mode selector switch – AUTO

c. Cockpit heat rheostat – Positioned as desired (12 o'clock position recommended initially)

NOTE

- If the aircraft is operated in excessively humid conditions, moisture may condense in the air distribution ducts after engine shutdown. This may be evaporated if the air conditioning system is operated in manual HOT for a brief period after engine start. In order to minimize fog entering the cockpit, the cockpit heat rheostat should be set at as high a temperature in AUTO mode as practical. The defogger system should be turned up as high as needed to keep the transparent surfaces clear. Under no circumstances should the mode selector switch be set to manual full COLD as this will result in freezing the water separator bag and forcing the bypass valve open allowing all moisture to pass through the water separator.

- During aircraft operation in humid conditions the air flow in the air conditioning and pressurization system may set up vibrations, which may be initially interpreted as engine problems. The problem may be solved by either positioning cockpit heat rheostat to HOT or opening RAM AIR SCOOP (pending altitude).

EMERGENCY OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

Slight temperature cycling or pressure surging may occur when selecting full hot air. This is caused by the maximum temperature (thermal) switch, which produces a rapid reduction in airflow by closing the hot air bypass valve on the air-conditioning package when duct temperature reaches 93° C (±5° C). When duct temperature drops sufficiently, the thermal switch again opens, allowing hot air to flow. This cycle may repeat five or six times. Cycling is most likely to occur at high airspeed when duct temperature may exceed 205° C. If cockpit temperature is not maintained at the desired level automatically, do the following:

a. Cockpit temp mode selector switch – MANUAL

b. Cockpit heat rheostat – Move TOWARD HOT OR COLD until temperature approaches desired level

NOTE

- If airflow surge occurs, leave cockpit temp mode selector switch in MANUAL and turn heat rheostat toward COLD.
- A decrease in cockpit airflow while in a manual cold mode may indicate freezing of the water separator. Move selector to HOT until airflow again increases. If cabin temperature is too high, move selector back to COLD and select a warmer setting than previously selected, and observe for evidence of water separator refreezing.
- If cockpit temperature is excessive and cannot be decreased automatically or manually, open the fresh-air scoop.

CAUTION

OPENING THE FRESH-AIR SCOOP WILL DUMP COCKPIT PRESSURE AND SHUT OFF BLEED AIR SUPPLY TO THE AIR-CONDITIONING UNIT.

WARNING

MANUAL CONTROL IS PROVIDED AS A BACK UP FEATURE ONLY AND SHOULD NOT BE USED EXCEPT IN CASE OF AUTOMATIC CONTROL FAILURE. THERE IS A POSSIBILITY OF FOGGING THE COCKPIT ON TAKEOFF IF MANUAL CONTROL IS USED.

DEFOGGER AND RAIN-REMOVER SYSTEMS

DEFOGGER SYSTEM

The defogger system consists of a number of small air jets directed parallel to the canopy and windshield surfaces. These jets entrap cockpit air and cause it to flow over the inside surface, thereby raising the surface temperature. As long as the surface temperature is above the cockpit dewpoint, no fog or frost will be formed. Air for the defogger system is normally routed from downstream of the water boiler on the air-conditioning unit, through a check valve and defog flow control and shutoff valve, to the defogger outlets located along the inside base of the windshield and electronics hatch transparent surface and in the forward frame of the canopy. This airflow, in itself, is not sufficient to meet all the requirements of the system.
To supplement the flow, a bleed line is provided which directs air from a point just downstream of the hot air shutoff valve to a differential relief and check valve. When pressure in the normal flow line drops because of large demands on the system (defog flow-control valve open), the differential relief valve will open and furnish the additional air necessary for effective defogging under all conditions.

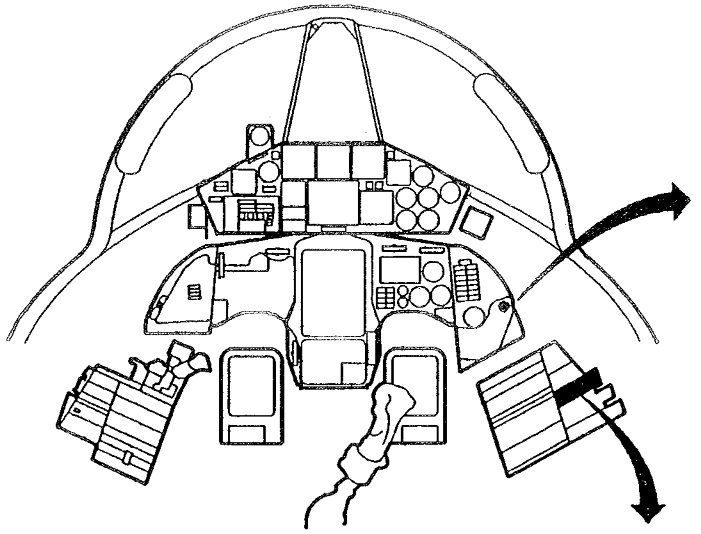
Canopy Defogger Knob

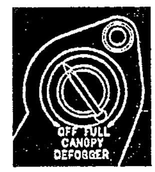
The amount of air directed to the windshield and canopy defog outlets is determined by the position of the canopy defogger knob located on right forward panel in the front cockpit only (refer to Figure 1-42).
Clockwise movement of the knob increases the amount of defogging air to the outlets by actuating the pneumatically operated shutoff valve. With the knob in the FULL (clockwise) position, the valve is open.

NOTE

- The amount of control knob dead band motion required to initiate flow will vary considerably with altitude changes, decreasing as the altitude increases.

HEATING AND RAIN REMOVER CONTROLS AND INDICATORS (FRONT COCKPIT ONLY)

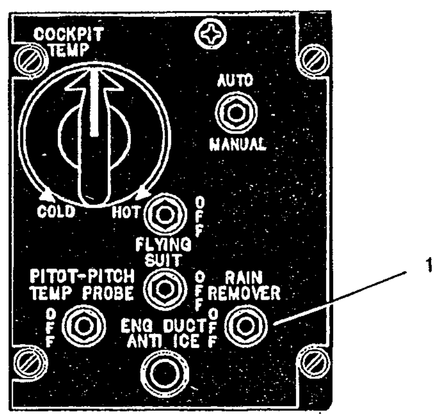




DETAIL A

A – CANOPY DEFOGGER KNOB

B – HEATING CONTROL PANEL
1 RAIN REMOVER SWITCH



DETAIL B

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Figure 1-42

- The windshield and canopy defogging system should be operated throughout the flight at the highest flow possible consistent with pilot comfort so that sufficiently high temperature is maintained to preheat the canopy and windshield areas. It is necessary to preheat because there is not sufficient time during rapid descents to heat these areas to temperatures which prevent the formation of frost or fog.

RAIN-REMOVER SYSTEM

The rain-remover system receives compressor air from the same line that furnishes pressure for the canopy seal, radar and the anti-G suit (Figure 1-42). The air then passes through a pilot-controlled shutoff valve and is routed to a nozzle in a popup head at the outside base of the left windshield panel. This high-velocity hot air flows over the panel to remove rain and to prevent windshield icing.

WARNING

RAIN-REMOVER AIR IS DUCTED THROUGH THE LEFT SIDE OF THE COCKPIT. IF A LEAK SHOULD DEVELOP IN THIS DUCT LINE, AIR AT VERY HIGH TEMPERATURE WILL ENTER THE COCKPIT, IN WHICH CASE THE FRESH-AIR SCOOP SHOULD BE OPENED TO SHUT OFF ALL COMPRESSOR AIR TO THE DUCT LINE AND DIRECT COLD RAM AIR INTO THE COCKPIT.

Rain-Remover Switch

The unguarded rain-remover switch (refer to Figure 1-42) located on the heating control panel on the right console in the front cockpit only, controls the flow of compressor air to the rain-remover outlets. Moving the switch forward from OFF to the RAIN REMOVER position closes a primary dc bus circuit to the rain-remover shutoff valve, opening the valve and allowing hot pressurized air to pass through the valve to the outlets.

CAUTION

- DO NOT TURN THE RAIN-REMOVER SYSTEM ON ABOVE ITS LIMIT AIRSPEED BECAUSE THE RAIN-REMOVER NOZZLE MAY BE DAMAGED OR THE WINDSHIELD MAY CRACK. MAXIMUM AIRSPEED IS 295 KIAS.
- THE RAIN REMOVER SHOULD NOT BE USED FOR TAKEOFF. THE COMBINATION OF HIGH TEMPERATURE AIRFLOW FROM THE RAIN REMOVER AND LOW AIRFLOW OVER THE CANOPY MAY DAMAGE THE WINDSHIELD. THESE TEMPERATURES ARE MORE CRITICAL AT HIGH RPM. HOWEVER, TO PRECLUDE THE POSSIBILITY OF DAMAGE, THE RAIN REMOVER SHOULD NOT BE OPERATED FOR LONGER THAN 30 SECONDS WITHOUT WINDSTREAM EFFECT.

Normal Operation of Defogger and Rain-Remover System

If any portion of the windshield or canopy becomes obscured by moisture, operate the following controls:

a. Canopy defogger knob – FULL

b. Cockpit temp mode selector switch – AUTO

c. Cockpit heat rheostat – HOT

d. Rain-remover switch – RAIN REMOVER (If precipitation obscures forward visibility).

NOTE

Canopy defogging air should be used at the highest possible temperature consistent with pilot comfort. This will minimize the possibility of widshield and canopy fogging caused by extreme temperature differentials accompanying an engine failure or a rapid descent from altitude.

Emergency Operation of Defogger and Rain-Remover Systems

If the windshield cannot be cleared by normal procedures and it is necessary to land without delay, do the following:

a. Canopy defogger knob – CHECK (FULL position)

b. Cockpit temp mode selector switch – MANUAL

c. Cockpit heat rheostat – HOT

d. Rain-remover switch – RAIN REMOVER

e. Engine RPM – MAXIMUM, if fuel and time permit

The above procedure will direct compressor air to the windshield outlets at its maximum available temperature and pressure.

NOTE

If excessive fog, vapor, or visible moisture of any kind enters the cockpit, restricting visibility on takeoff, open the fresh-air scoop.

ANTI-ICING SYSTEMS

ENGINE/DUCT ANTI-ICING SYSTEMS

The engine duct anti-icing system is designed to prevent formation of ice in the engine air inlets and on the cones and scoops of the inlet ducts. Ice accumulating in the engine air inlet reduces airflow through the engine and causes a loss of power accompanied by a possible increase in nozzle area and a decrease in fuel flow. For anti-icing of the engine air inlet, hot 17th-stage compressor air flows through a port in the compressor rear frame to the inlet of the solenoid-operated anti-ice valve.

The anti-ice valve, powered by the XP2 bus, regulates pressure and airflow to the four struts at 2, 3, 9 and 10-o'clock positions of the engine front frame. Air is passed through the struts into a manifold in the hub of the frame, from which it passes into the 20 hollow inlet guide vanes and the remaining 4 front frame struts.

The anti-icing air is then discharged into the primary airstream through holes in the outer ends of these four struts and the trailing edges of the inlet guide vanes. Anti-icing of the cones and scoops of the inlet ducts is accomplished by electrical current from the XP3 bus.

Electricity flowing through a power relay to the scoop and cone heaters warms these areas to a temperature high enough to eliminate or prevent ice formation. Overheat protection is provided automatically by thermostats located in each cone skin panel. Scoop and cone heaters are inoperative on the ground as they are deenergized by the air-ground safety switch. The operation of both electrical and compressor air system is controlled by the engine duct anti-ice switch. Circuit breakers for the system are provided on the right console in rear cockpit only.

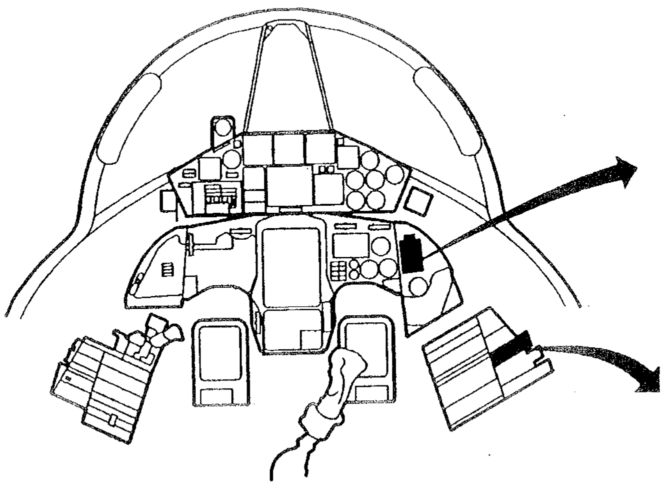
Engine/Duct Anti-Ice Switch

The engine/duct anti-ice switch is located on the heating control panel on the right console in the front cockpit only (Figure 1-43). This switch controls the flow of compressor air to the engine inlet guide vanes by actuating the solenoid-operated engine anti-ice valve and the cone and scoop heaters. The switch is powered by the PP1 bus.

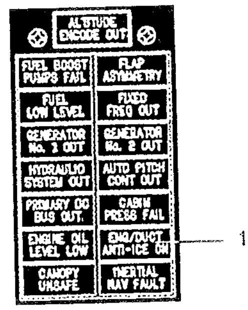
Engine/Duct Anti-Ice On Warning Light

A light on both warning light panels is provided as a visual indication that the engine duct anti-icing system is operating. A differential pressure-operated switch which responds instantly to changes in pressure within the anti-icing system causes the ENG/DUCT ANTI-ICE ON light illuminate when the anti-ice switch is operated.

ANTI-ICING CONTROLS AND INDICATORS



FRONT COCKPIT

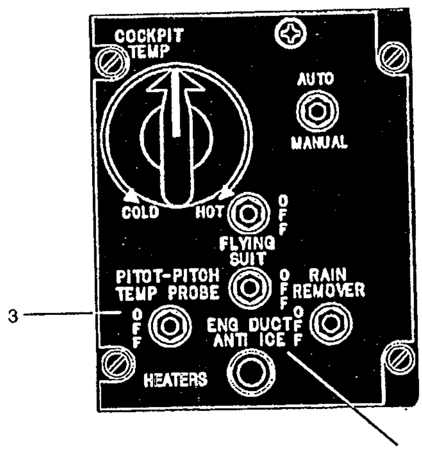


DETAIL A

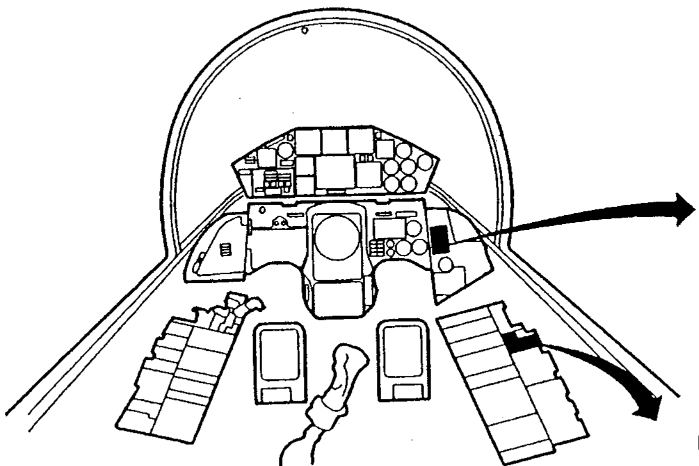
A – WARNING LIGHTS PANEL
1 ENG/DUCT ANTI-ICE ON WARNING LIGHT

B – HEATING CONTROL PANEL
2 ENGINE DUCT ANTI-ICE SWITCH
3 PITOT-PITCH TEMP PROBE HEATERS SWITCH

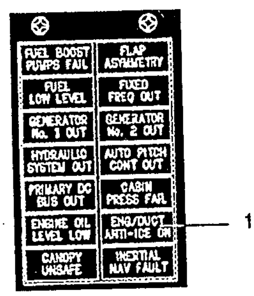
C – PITCH SENSOR PITOT HEAT SWITCH



DETAIL B



REAR COCKPIT



DETAIL A

DETAIL C

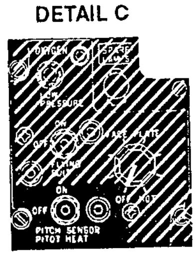


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Figure 1-43

NOTE

- Due to the extreme sensitivity of the warning light pressure sensing switch, the warning light may illuminate momentarily during rapid throttle bursts with the anti-ice switch in the OFF position. This is not considered a malfunction if steady state indication is correct.
- The placard normally extinguishes immediately after the engine/duct anti-icing switch is placed in the OFF position.

ENGINE/DUCT ANTI-ICING SYSTEM OPERATION

Icing will occur on the inlet ducts and engine compressor front frame at subsonic speeds only. Ram air temperature rise at supersonic speeds is sufficient to prevent icing. If the engine is operated above 82% RPM, the anti-icing air temperature is sufficient to prevent rapid ice buildup on the engine front frame and inlet guide vanes. The engine may safely ingest inlet duct ice at engine speeds less than 88% RPM. At higher engine speeds, the inlet guide vanes may be damaged. Engine operation is still possible with limited inlet guide vane damage. The requirement for engine anti-icing is a direct function of indicated compressor inlet temperature (CIT).

NOTE

Operation of the anti-icing system at a CIT above 10° C degrades the service life expectancy of the magnesium front frame. If weather conditions indicate a need for anti-icing, the system should be actuated only at a CIT indication of 10° C or below.

Ground Check Prior to Flight Under Icing Conditions

a. Engine – IDLE RPM

b. Engine/duct anti-ice switch – ON

ENG/DUCT ANTI-ICE ON warning light should illuminate within 5 seconds. If light does not illuminate, advance throttle to 80% RPM.

c. Eng/duct anti-ice switch – OFF (after placard illuminates)

ENG/DUCT ANTI-ICE ON warning light must extinguish within 5 seconds.

CAUTION

ABORT FLIGHT IF WARNING PLACARD DOES NOT ILLUMINATE OR REMAINS ILLUMINATED MORE THAN 5 SECONDS AFTER ENGINE/DUCT ANTI-ICE SWITCH IS PLACED IN OFF POSITION. MAKE NOTATION OF MALFUNCTION IN AIRCRAFT LOG.

Inflight Procedure

a. If flight through icing conditions is anticipated, activate the anti-icing system when at subsonic speed and when the indicated CIT is 10° C or below. Do not exceed a maximum speed of 350 KIAS or Mach 1.0, whichever is lower, with the anti-ice valve open.

b. After flying in moderate to heavy icing for 2 minutes or more, reduce thrust (where practical) to 88% to minimize inlet duct ice ingestion damage to the engine.

c. Should it be necessary to fly in known icing conditions at low altitude and low thrust settings (80 to 86% RPM), the engine power should be increased to 100% RPM every 5 minutes to ensure that adequate anti-icing air circulation is available at the engine compressor front frame. This thrust increase should be maintained for approximately 30 seconds.

ELECTRICALLY HEATED WINDSHIELD

The left windshield panel is electrically heated. Glass temperature is controlled by a thermal switch which is powered by the PP2 bus (no manual switch is installed).
This switch will provide electrical power to the heating element whenever the glass temperature falls below 35° C (±3° C). When the glass is heated to 40° C (±3° C) the electrical power will be disconnected automatically.

The windshield heating unit obtains power from the emergency ac bus. A circuit breaker located on the left consolle in the front cockpit only, labeled WINDSHIELD DEFOG, (Figure 1-42) may be pulled to deactivate the system in case of a malfunction.

CAUTION

DO NOT PLACE OBJECTS SUCH AS CHECKLISTS OR CLIPBOARDS ON LEFT QUARTER PANEL GLARE SHIELD OR THE THERMAL SENSOR MAY BE DISLOCATED RESULTING IN DAMAGE TO THE ELECTRICALLY HEATED WINDSHIELD.

PITOT-PITCH AND TEMPERATURE PROBE HEATERS

The automatic pitch control and stick-shaker pitch sensor vanes, the pitot-static head, and the free air temperature probe are heated electrically by power from the XP2 bus.
Heating elements within the pitch sensor vanes, free air temperature probe and the pitot-head receive this power whenever the heating circuit is closed. This is done by moving the switch (Figure 1-43) on the heating control panel from OFF to PITOT-PITCH TEMP. PROBE position.
This also actuates the free air temp probe heater through a double-throw switch. Heat should be applied any time instrument flying conditions are encountered in order to prevent the formation of ice on these units.
A circuit breaker, labeled PITOT HEAT (refer to Figure FO-8), is provided on the right console in the rear cockpit only. An on/off switch is provided on the right console in both cockpits.

CAUTION

DO NOT OPERATE THE HEATERS FOR MORE THAN FOUR MINUTES CONTINUOUSLY WHILE THE AIRCRAFT IS ON THE GROUND.

COMMUNICATIONS AND NAVIGATION EQUIPMENT

Refer to Figure 1-44 for a summary table.

ANTENNA SYSTEM

Antennas are installed on the aircraft to provide the sensing and radiating elements for communications, navigation, and identification systems.
Antennas installed on the aircraft are of the flush-mounted type.

UHF RADIO AN/ARC-150

The equipment permits the selection of a frequency channel every 0.025 MHz in the range of 225.000 to 399.975 MHz, providing 7000 possible frequency channels. Any 19 of these 7000 possible channels may be preset to facilitate immediate use. The receiver and transmitter are automatically tuned after a channel or frequency change.
In addition to the main receiver a separate fixed tuned crystal controlled UHF receiver with a frequency range of 238.000 to 248.000 MHz is installed to provide a constant alert emergency channel.
This emergency guard channel frequency is normally set at 243.000 MHz and is pretuned prior to installation.
The transmitter is tonemodulated at an audio frequency of 1020 Hz for emergency or direction-finding purposes. The AN/ARC-150 (HQ) radio set may operate with the HAVE QUICK jam-resistant technique. Slow-frequency hopping is used and both transmitter and receiver use a time-varying frequency pattern which ranges throughout the UHF band. The UHF antennas receive and radiate UHF signals through a frequency range of 225.000 to 399.975 MHz.
Two flush-mounted antennas are provided: one, the bottom antenna, is located on the lower cockpit access hatch, while the top antenna is mounted on the ammunition compartment cover.
The pilot, by the antenna selector switch located on left console in both cockpits, manually select either antenna. The UHF command radio is powered by the PP2 DC bus.

COMMUNICATION, IDENTIFICATION AND NAVIGATION EQUIPMENT TABLE

TYPE	DESIGNATION	FUNCTION	RANGE	CONTROL LOCATION
UHF COMMAND	AN/ARC-150	TWO-WAY COMMUNICATION	LINE OF SIGHT	LEFT CONSOLE/ INSTRUMENT PANEL/ CONTROL STICK GRIP/ ENGINE THROTTLE
EMERGENCY UHF	SIT-301			
INTERPHONE	AN/AIC-18	BOTH PILOTS/GROUND CREW INTERCOMMUNICATION	NOT APPLICABLE	LOWER INSTRUMENT PANEL
IFF	SIT-421T	AIRCRAFT IDENTIFICATION AND NOMINATIVE	LINE OF SIGHT	RIGHT CONSOLE/ CONTROL STICK GRIP/ ENGINE THROTTLE (FRONT COCKPIT ONLY)
DIRECTION INDICATOR	C-2G COMPASS	PROVIDE MAGNETIC HEADING	0° - 360°	RIGHT CONSOLE (FRONT COCKPIT ONLY)
NAVIGATION	TACAN	DISTANCE MEASURING AND STATION BEARING	LINE OF SIGHT 390 NMI (max)	RIGHT CONSOLE/ MAIN INSTRUMENT PANEL
NAVIGATION	HSI	INDICATES BEARING DISTANCE TO DESTINATION AND HEADING	RANGE 0 ÷ 399 NM BEARING 0° ÷ 360° HEADING 0° ÷ 360°	INSTRUMENT PANEL
NAVIGATION	INERTIAL NAVIGATOR	PROVIDES ATTITUDE, PRESENT POSITION, STEERING PARAMETERS TO HSI, AI, CDU AND OTHER EQUIPMENT	ALL ATTITUDE WORLD WIDE NAVIGATION	RIGHT CONSOLE/ MAIN INSTRUMENT PANEL
NAVIGATION	AI	INDICATES PITCH AND ROLL, RATE OF TURN, SIDE SLIP	PITCH ± 90° ROLL ± 180°	MAIN INSTRUMENT PANEL
NAVIGATION	ADAPTER	INTERFACES IN WITH OTHER EQUIPMENT	N/A	N/A
NAVIGATION	IN/CDU	PROVIDES LOADING OF MISSION DATA, DISPLAYS NAV DATA AND STEERING PARAMETERS	N/A	MAIN INSTRUMENT PANEL

Figure 1-44

UHF AN/ARC-150 (HQ) Control Panel

The AN/ARC-150 (HQ) panels (refer to Figure 1-45) are located on the left console. These control panels are used in conjunction with the AN/ARC 150 (HQ) channel/frequency indicators located on main instrument panels.

Manual Frequency Selector Knobs. Five manual frequency selector knobs are provided across the panel to set up any desired operating frequency not preset on the channel selector.
From left to right the first knob selects the proper number for hundreds of MHz, the second knob selects tens of MHz, the third knob selects units of MHz, the fourth knob selects tenths of MHz and the fifth knob varies the frequency by 0.025 MHz steps. These numbers appear in a window above their respective knob so that any frequency in the covered portion of the UHF band may be selected manually. In addition the first knob allows selection of the A and T function of have quick mode.

For the HQ facilities the following selections are available:

Manual Frequency Selector Knobs	Selections
Hundreds of MHz	T-TOD updating A-HQ operation
Tens of MHz	First NET digit selection
Units of MHz	Second NET digit selection
Tenths of MHz	Third NET digit selection
Units of MHz	00, 25, 50 and 75 HQ mode of operation

NOTE

- "A" selection enables the HQ active mode of operation.
- The "T" selection is a momentary selection. After "T" selection the hundreds of MHz shows "2", "3" or "A".
- "A" + BOTH selections enables the guard channel reception.
- "A" + MAIN selections disables the guard channel.

Channel Selector. The channel selector at the top of the panel selects any of 19 preset channels for transmission and reception when the mode switch is in the PRE position. The selected channel number appears in a window to the right of the selector. The 20th channel is reserved for the HAVE QUICK mode. In HQ the CHAN 20 shall not be used for frequency storing.

Memory Button. When depressed the button sets a selected channel to the frequency value which had been set up using the frequency selector knobs, providing the function switch is set to the PRE position. In HQ, the button is used to memorize the WOD (CH 20) as well as the training frequencies (CH 15 to 19).

Tone Button. When depressed the button excites a tone oscillator which feeds the tone (1020 Hz) into the modulation, thereby providing continuous tone transmission on the selected channel frequency for emergency or direction finding operation.
In HQ, when pressed momentarily, is used for the TOD transmission. When pressed for a longer time a tone is generated after the TOD.

Mode Switch. The mode switch determines the method of frequency selection. When the switch is in the MAN position, operation is permitted on the frequency selected by the manual frequency selector knobs. When the switch is in the PRE position use of the channel selector is permitted for operation of any one of the 19 preset frequencies.
In addition the mode switch allows selection of the 243.000 MHz emergency guard frequency (GD). In the HQ the GD selection overrides the "A" selection.

Volume Control Knob

This knob, labeled VOL, permits the audio signal adjustment.

Squelch Switch. This switch permits the activation of the squelch circuit. When the noise threshold exceeds the calibration value, the squelch provides for the elimination of noise signal. When set to OFF the squelch circuit is deactivated.

Function Switch. Operating the function switch to one of the four position set up the functions of the equipment as follows:

OFF Deenergizes the equipment.

MAIN — Turns the radio on. The transmitter and main receiver operate on the same frequency. The equipment is switched from the receiver to the transmitter by depressing the microphone button. When the button is released the equipment returns to a receiving condition.

BOTH — The main receiver, transmitter and guard receiver are operative. The main receiver and transmitter operate on the frequency determinated by the frequency control component. The guard receiver operates on a frequency preset on the ground. The BOTH position permits simultaneous monitoring of the main and guard receivers.

NOTE

The BOTH position provides a monitor (receive) function only and does not permit transmission on the guard frequency. Transmission on the guard frequency is selected by positioning the mode selection to GUARD or by setting up the guard frequency on the manual frequency selector knobs.

ADF — This position is inoperative.

Channel/Frequency Indicator

The channel/frequency indicator (refer to Figure 1-45) are located on the left side of the upper instrument panels. This indicator has the following controls and indicators:

Channel/Frequency Display. A LED display which shows the channel/frequency which has been selected through the channel selector or through the manual frequency selector knobs, both located on the UHF radio control panel. When the Active HQ mode is selected, the first digit shows "A".

TEST/FREQ Switch. A three position TEST, neutral and FREQ switch.
When set to TEST (momentary position) a display check is carried out (888.888 displayed).
When set to FREQ (momentary position) the indicator shows the frequency (MAN) or the channel (PRE) selected on the UHF control panel.

Dimming Knob. This knob labeled DIM permits the channel/frequency display brightness control and adjustment.

Antenna Selector Switch

An antenna selector switch is located on the cockpit left console circuit breaker panel and provides a means for manually controlling the antenna selector. Two position TOP and BOTTOM, enable the pilot to select a particular antenna when desired or necessary for proper reception.

Microphone Buttons

Two microphone buttons (refer to Figure 1-45) are available for transmitting. One is located on the throttle and is always usable while the other is located on the stick.

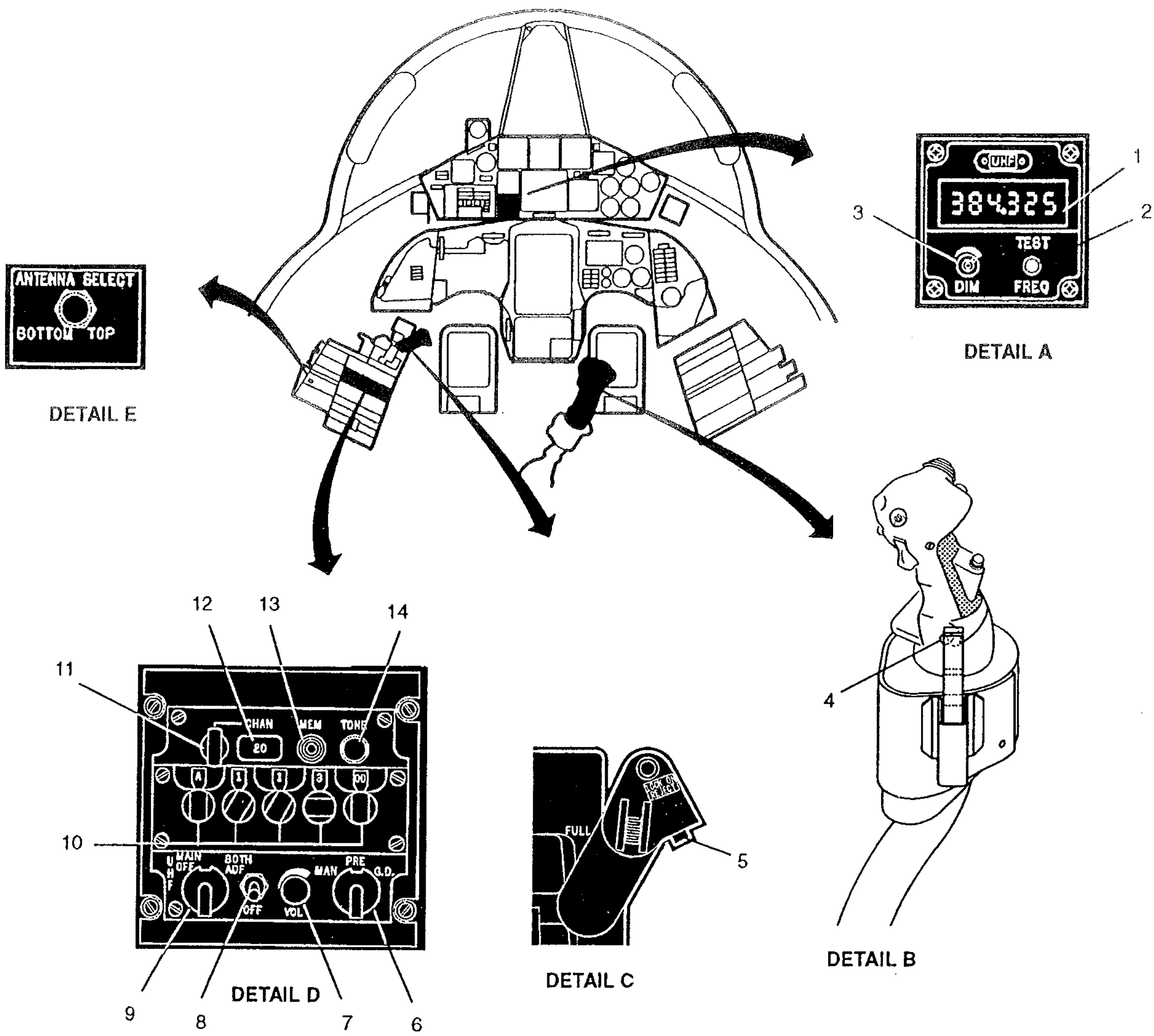
NOTE

The nosewheel steering/microphone button (located on the stick) may be used for communication in flight only. During on-ground procedures the button is used for the nosewheel steering control.

UHF AN/ARC-150 (HQ) Operation

1. Channel frequencies: check
2. Function switch: as required (MAIN or BOTH)
3. Mode selector switch: PRE or MAN
4. For pre-operation select a preset channel by rotating the channel selector knob so that the desired channel number appears in the window
5. Before transmitting message, check for operation and warmup of the transmitter, using either of the microphone buttons while listening for a tone
6. Adjust volume control, rotating (as desired) the VOL control knob
7. If it is desired to transmit and receive on a frequency not previously preset on the channel selector, place the mode switch in the MAN

UHF AN/ARC-150 (HQ) CONTROL PANEL (FRONT COCKPIT)



A – CHANNEL FREQUENCY INDICATOR
1 CHANNEL/FREQUENCY DISPLAY
2 FREQUENCY SWITCH
3 DIMMING KNOB

B – CONTROL STICK GRIP
4 NOSEWHEEL STEERING/MICROPHONE BUTTON

C – ENGINE THROTTLE
5 MICROPHONE BUTTON

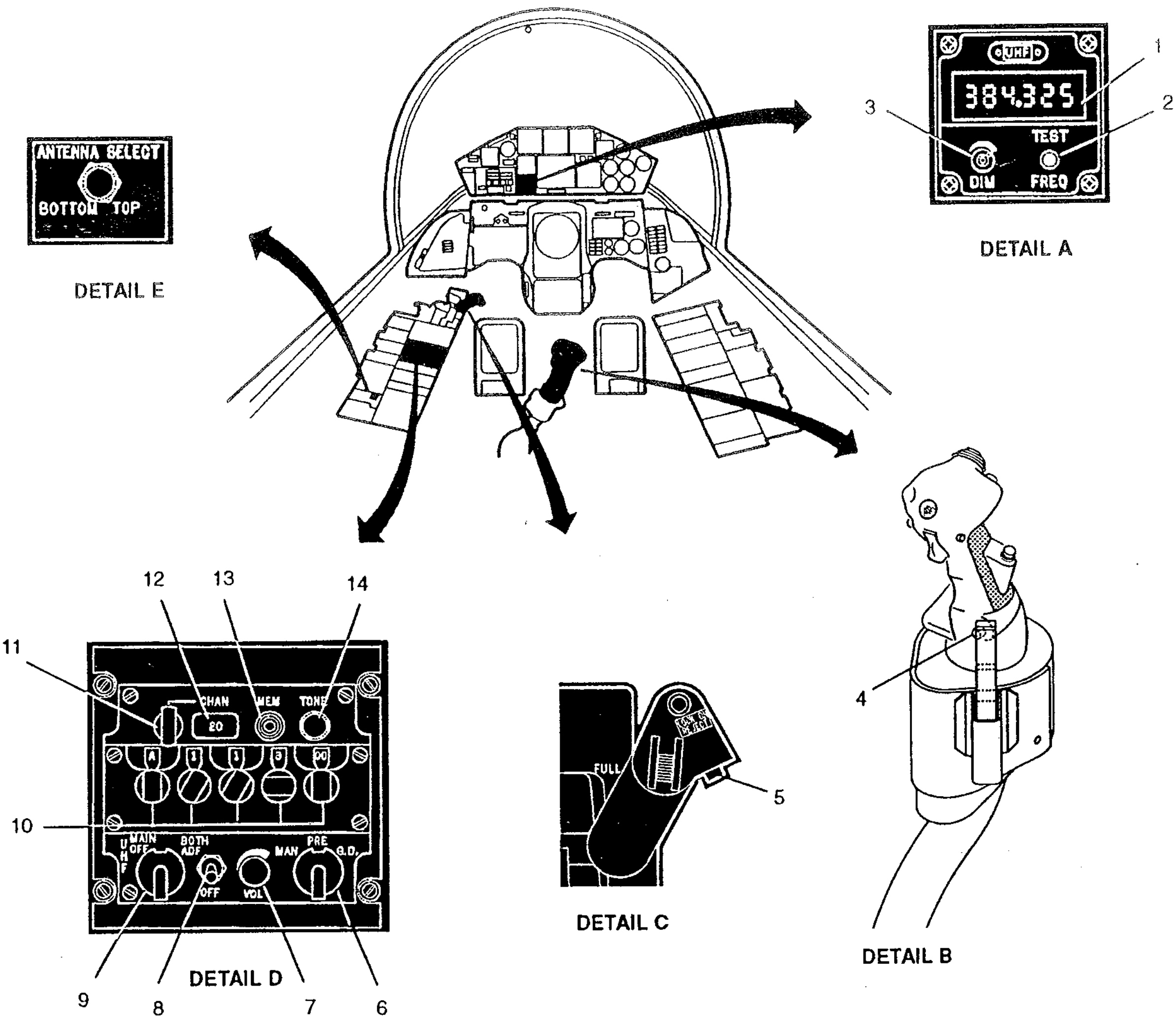
D – UHF CONTROL PANEL
6 MODE SWITCH
7 VOLUME CONTROL KNOB
8 SQUELCH SWITCH
9 FUNCTION SWITCH
10 MANUAL FREQUENCY SELECTOR KNOB AND DISPLAY
11 CHANNEL SELECTOR AND DISPLAY
12 WINDOW
13 MEMORY BUTTON
14 TONE BUTTON

E – ANTENNA SELECTOR SWITCH

FA0136

Figure 1-45 (Sheet 1 of 2)

UHF AN/ARC-150 (HQ) CONTROL PANEL (REAR COCKPIT)



A – CHANNEL FREQUENCY INDICATOR
1 CHANNEL/FREQUENCY DISPLAY
2 FREQUENCY SWITCH
3 DIMMING KNOB

B – CONTROL STICK GRIP
4 NOSEWHEEL STEERING/MICROPHONE BUTTON

C – ENGINE THROTTLE
5 MICROPHONE BUTTON

D – UHF CONTROL PANEL
6 MODE SWITCH
7 VOLUME CONTROL KNOB
8 SQUELCH SWITCH
9 FUNCTION SWITCH
10 MANUAL FREQUENCY SELECTOR KNOB AND DISPLAY
11 CHANNEL SELECTOR AND DISPLAY
12 WINDOW
13 MEMORY BUTTON
14 TONE BUTTON

E – ANTENNA SELECTOR SWITCH

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Figure 1-45 (Sheet 2 of 2)

position and setup the new frequency selector knobs

8. Turn the function switch OFF to deenergize the set

HAVE QUICK (HQ) FACILITY

The Have Quick (HQ) facility provides the aircraft on effective air-to-air and air-to-ground jam resistance in the UHF voice communications which allows the pilot to carry out its mission in a jammed environment.
The HQ programme consists of a simoultaneous frequency change (frequency hoping) in the transmitter and receiver, following a pseudo-casual frequency sequence in a way that users are syntonized, instant for instant, on the same frequency. This is done provided that they have a "common reference", both during transmission and during reception. This synchronization of all radio equipment may be available provided they have a common time frame of reference defined as Time Of Day (TOD). The sequence of the frequency hopping is determined by a key defined as Word Of Day (WOD) that changes daily in order to avoid the possibility of a cross reference of the disturber with the communications.
The system is able to store up to six WOD defined as Multiple WOD (MWOD), each consisting of six elements. The frequency hopping modality is established by the number of network (NET). The NET number is defined as the starting point of the common frequency hopping.
The HQ may be operated in the Active mode ("A") as well as in the Training ("T") mode.

UHF HAVE QUICK SYSTEM OPERATION

NOTE

CHAN 20 is used for active ("A") HQ operation only.

In order to operate in HQ is necessary to have WOD and TOD loaded. If WOD and TOD are not loaded, a continuous high tone (3125 Hz) is heard in headset. To initializate the system proceed as follows:

ON-GROUND OPERATION

System Initialization

1. Function switch – Set to MAIN
2. Mode switch – Set to PRE
3. Channel selector – Set 20
4. Manual frequency selector knobs – Set 220.000
5. Memory button – Press, check audio tone

WOD Loading

1. Mode switch – Set to PRE
2. Channel selector – Set 20
3. Manual frequency selector knobs – Set 220.025
4. Memory button – Press
5. Channel/frequency indicator – Set momentarily to FREQ, check 220.025 displayed, then release
6. Mode switch – Set to MAN
7. Manual frequency selector knobs – Set first WOD segment
8. Tone button – Press
9. The remaining 5 segments are loaded following the above steps from 6. to 8. down to CHAN 15

TOD Loading

10. Channel selector – Set 14
11. Manual frequency selector knobs – Set 3AB.000 where:
    A – Tens of day
    B – Units of day
12. Tone button – Press

MWOD Loading

In order to load the remaining five WOD, repeat steps from 7. to 12.

NOTE

Pilot shall be aware of the following:
- at each WOD, a different operative date shall be associated
- storing of the first segment shall be carried out in CHAN 20.

Final Loading

13. Mode switch – Set to PRE
14. Channel selector – Set 20
15. Manual frequency selector knobs – Set 220.000
16. Memory button – Press

WOD/MWOD Erasing

1. Channel selector – Set 20
2. Mode switch – Set to PRE
3. Manual frequency selector knobs – Set to 220.050
4. Memory button – Press
5. Mode selector switch – Set to MAN
6. Tone button – Press

HQ Training Loading

1. Channel selector – Set 20
2. Mode switch – Set to PRE
3. Manual frequency selector knobs – Set to 220.075
4. Memory button – Press
5. Mode switch – Set to MAN
6. Manual frequency selector knobs – Set desired frequency
7. Tone button – Press
8. The remaining 15 training frequencies loading may be carried out following the above steps 6. and 7. down to CHAN 5

PRE-FLIGHT OPERATION

CAUTION

IF THE RADIO IS SWITCHED OFF FOR MORE THAN 5 SECONDS, "TOD" AND "WOD" INFORMATION IS LOST.

WOD Recall

1. Mode switch – Set to PRE
2. Channel selector – Set to 20
3. Manual frequency selector knobs – Set 220.025
4. Memory button – Press
5. Channel selector – Set 1
6. Mode switch – Set to MAN
7. Manual frequency selector knobs – Set 3AB.000 where:
   A – Tens of day
   B – Units of day
8. Tone button – Press
9. Mode switch – Set to PRE
10. Channel selector – Set 20
11. Manual frequency selector knobs – Set 220.000
12. Memory button – Press

WOD Check

1. Channel selector – Set 20
2. Mode switch – Set to MAN
3. Manual frequency selector knobs – Set 3AB.000 where:
   A – Tens of day
   B – Units of day
4. Channel selector – Set 19 then 20. Check audio tone in headset

NOTE

The WOD is not available if no audio tone is heard in headset.

TOD Acquisition

NOTE

"T" selection, on the UHF control panel, causes the "A" indication to be displayed on the channel/frequency indicator.

1. Manual frequency selector knobs – Set as required
2. Manual frequency selector knob – Set "T" for at least 2 seconds
3. Call an HQ operative station and check audio tone (1667 Hz) within 1 minute

Auto Synchronization

1. Manual frequency selector knob – Set and hold "T" and simultaneously press tone button. Check audio tone

NOTE

To avoid TOD transmission, release tone button before the manual frequency selector knob.

TOD TRANSMISSION

1. Mode switch – As desired
2. Manual frequency selector knobs – Set as required
3. Tone button – Press for at least 2 seconds then release. Check audio tone (1667 Hz)

Active HQ Activation

NOTE

Active mode is available provided that WOD and TOD have been previously loaded.

1. Mode switch – Set to MAN
2. Antenna selector switch – TOP or BOTTOM
3. Manual frequency selector knobs – Set in sequence NET (3 digit), HQ mode (2 digit) then "A"
4. Channel/frequency indicator. Check A + NET + HQ displayed

NOTE

Following a wrong NET setting, an intermittent 3125 Hz audio tone is listened.

INTERPHONE SYSTEM

The AN/AIC-18 interphone system is an integral part of the AN/ARC-150 UHF radio. The interphone system receives its power from the PP2 bus. If the PP2 bus fails, the emergency interphone power will be supplied by the PP4 bus; however, the left main gear door uplock switch will deenergize the emergency interphone power circuit when the landing gear is extended, the emergency power circuit will be deenergized by the air/ground safety switch when the aircraft lands.

The interphone provides amplification of all transmitted and received audio signals and communications between pilot and ground crew during on ground procedures.

The interphone is connected to the aircraft through a jack located on the external power receptacle. In order to transmit to the ground crew, the pilot set the INTERCOM MIC switch to the ON position in the front or rear control transfer panel (refer to Figure 1-48). The landing gear warning, the TACAN and AN/ARC-150 UHF command radio are connected to the interphone system.

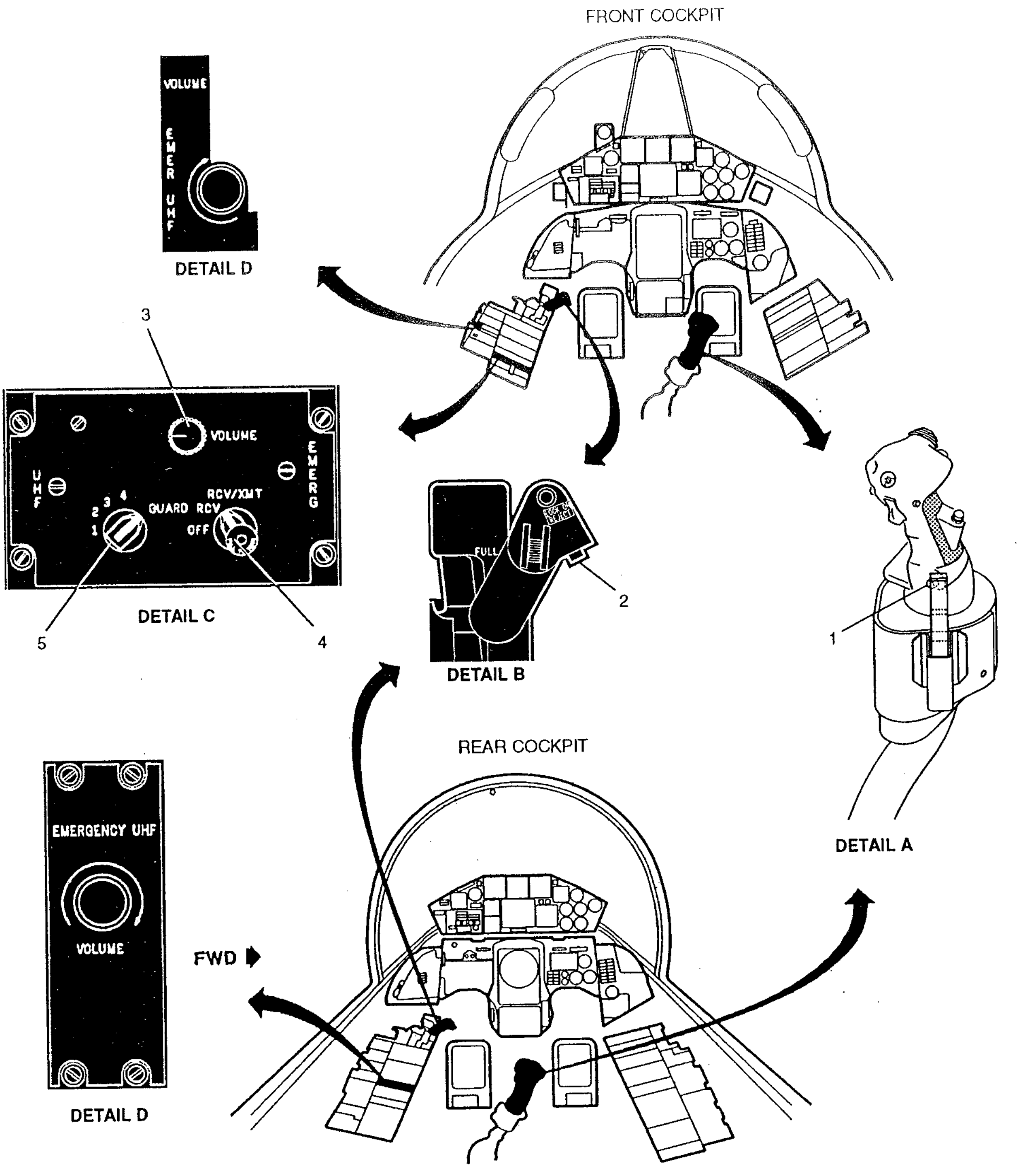
Hot-microphone operation is available whenever the PP2 bus is energized (external power ON or engine running).

The INTERCOM MIC volume selector switch and the auxiliary volume control knob are provided on the control transfer panel (Figure 1-48) for selecting and adjusting interphone audio level.

EMERGENCY UHF RADIO

The emergency UHF equipment SIT-301 (refer to Figure 1-46) provides two-way, short range voice transmission in the event of AN/ARC-150 UHF failure.

EMERGENCY UHF SIT-301



A – **CONTROL STICK GRIP**
1 NOSEWHEEL STEERING/MICROPHONE BUTTON

B – **ENGINE THROTTLE**
2 MICROPHONE BUTTON

C – **EMERGENCY UHF CONTROL PANEL**
3 VOLUME CONTROL KNOB
4 RECEIVER/TRANSMIT SELECTOR SWITCH
5 CHANNEL SELECTOR SWITCH

D – **EMERGENCY UHF VOLUME CONTROL KNOB**

FA0150

Figure 1-46

The integrative emergency solid-state UHF equipment (receiver, transmitter, power supply, and control panel) is contained in a one-package unit which is installed in the front cockpit left console only.
The emergency UHF receiver-transmitter operates in the 241.000 to 245.000 MHz range, providing 5 preset channels on a 4 MHz segment. This equipment normally remains in the receiver condition (receiver/transmit rotary switch in the RCV position) and it is used only if the main UHF or its power source becomes inoperative.
Both emergency UHF and main UHF communications receiver outputs are connected to the headset. The microphone is connected to the emergency UHF equipment only when the emergency UHF control panel receiver/transmit rotary switch is in the RCV/XMT position.
Transmission is not possible on the normal UHF transmitter when the receiver/transmit rotary switch is in the RCV/XMT position. Under this condition the status of normal UHF is receive only. The emergency UHF radio is powered by the PP4 battery bus.

Emergency UHF Radio Control Panel

The emergency UHF control panel (refer to Figure 1-46) located on the left console, consists of a receive or transmit selector switch, a channel selector switch, and a volume control knob.

Channel Selector Switch. This five-position switch selects the UHF emergency preset channel of operation. Frequencies from 241.000 to 245.000 MHz can be preset in channels from 1 to 4 as desired. The GUARD position selects a crystal-controlled guard frequency of 243.000 MHz.

RCV/XMT Switch. The PULL TO TURN receiver/transmit switch transfers the microphone switching circuitry between the main UHF (receive position) and emergency UHF (receive and transmit position).

Volume Control Knob. The volume control knob provides a mean of adjusting emergency UHF audio level.

Microphone Buttons

Two microphone buttons (refer to Figure 1-46) are available for transmitting. One is located on the throttle and is always usable while the other is located on the stick.

NOTE

The nosewheel steering/microphone button (located on the stick) may be used for communications in flight only. Durin on-ground procedures the button is used for nosewheel steering control.

IFF SYSTEM

The SIT 421T IFF system consists, primarily, of a control panel, a transponder unit and one antenna. The IFF control panel is located on the right console in the front cockpit only while the IFF transponder is located in the forward electronic compartment. The upper antenna is installed on the panel of the refrigeration compartment. The IFF system provides automatic identification of the aircraft when challenged by other airborne or ground radar installation.
The modes of operation have the following significance: Mode 1 – security identity, Mode 2 – personal identity, Mode 3/A – air traffic identity, Mode C – altitude reporting. The IFF system transmits coded replies to correctly coded interrogations. The system utilizes power from PP2 bus.
The IFF transponder is a diversity transponder operating according to the MARK XA-SIF system in Modes 1, 2, 3/A and C.
The transponder receives interrogations on a carrier frequency of 1030 MHz via two antennas; the signals are decoded and revealed, and if interrogation is correct, reply transmission is enabled via the antenna which has received the strongest signal. Reply signals are transmitted on a carrier frequency of 1090 MHz. Normal operation of the transponder is constantly monitored by the BIT circuits; incorrect performance causes a NO GO indication lamp to be displayed.

IFF Control Panel

NOTE

The IFF crypto-computer (which enables the IFF Mode 4 operation) is not installed. For this reason the following IFF MODE 4 control panel controls and indicators and relative warning light are not operative:
- STATUS KIT lamp
- CODE selector
- TEST-ON-OUT switch
- AUDIO-LIGHT-OUT switch
- REPLY lamp
- IFF MODE 4 warning light

and omitted in this description.

The IFF control panel (refer to Figure 1-47) is located on the right console in the front cockpit only and has the following controls and indicators:

TEST-GO Lamp. When illuminated it indicates correct operation of the IFF system in self-test of Mode 1, or 2, or 3/A or C when the respective switches are positioned to TEST position.

TEST/MON-NO GO Lamp. When illuminated it indicates incorrect operation of the IFF system during self-test of Mode 1, or 2, or 3/A or C when the respective switches are positioned to TEST position. It also indicates that the IFF system does not respond properly to the received interrogations.

ANT Switch. This three position toggle switch permits the following antenna selection:

TOP Only the upper antenna is used.

DIV The system automatically selects the antenna to be used for transmission; reception is through both antennas.

BOT Only the lower antenna is used.

MASTER Selector. This four position rotary selector has the following positions:

OFF The system is de-energized.

STBY Places receiver-transmitter in warm-up (standby condition). Selector should remain in STBY a minimum of 1 minute for standard temperature conditions and 5 minutes under extreme ranges of temperature.

NORM The system is in operation.

EMER Transmits emergency reply signals to Mode 1, 2 or 3/A interrogations regardless of mode control setting.

NOTE

- The EMER position enables the functions of all operating modes regardless of the position of the M-1, M-2, M-3A or M-C switches.
- The EMER function enables Mode C regardless of the position of the MASTER selector. The transmission of an emergency reply requires that an interrogation signal be received prior to enabling transmission.
- The emergency functions are automatically enabled, through a pull connector on the ejection seat, when the pilot ejects.

RAD TEST-OUT Switch. The RAD TEST-OUT switch is a two toggle switch with the following positions:

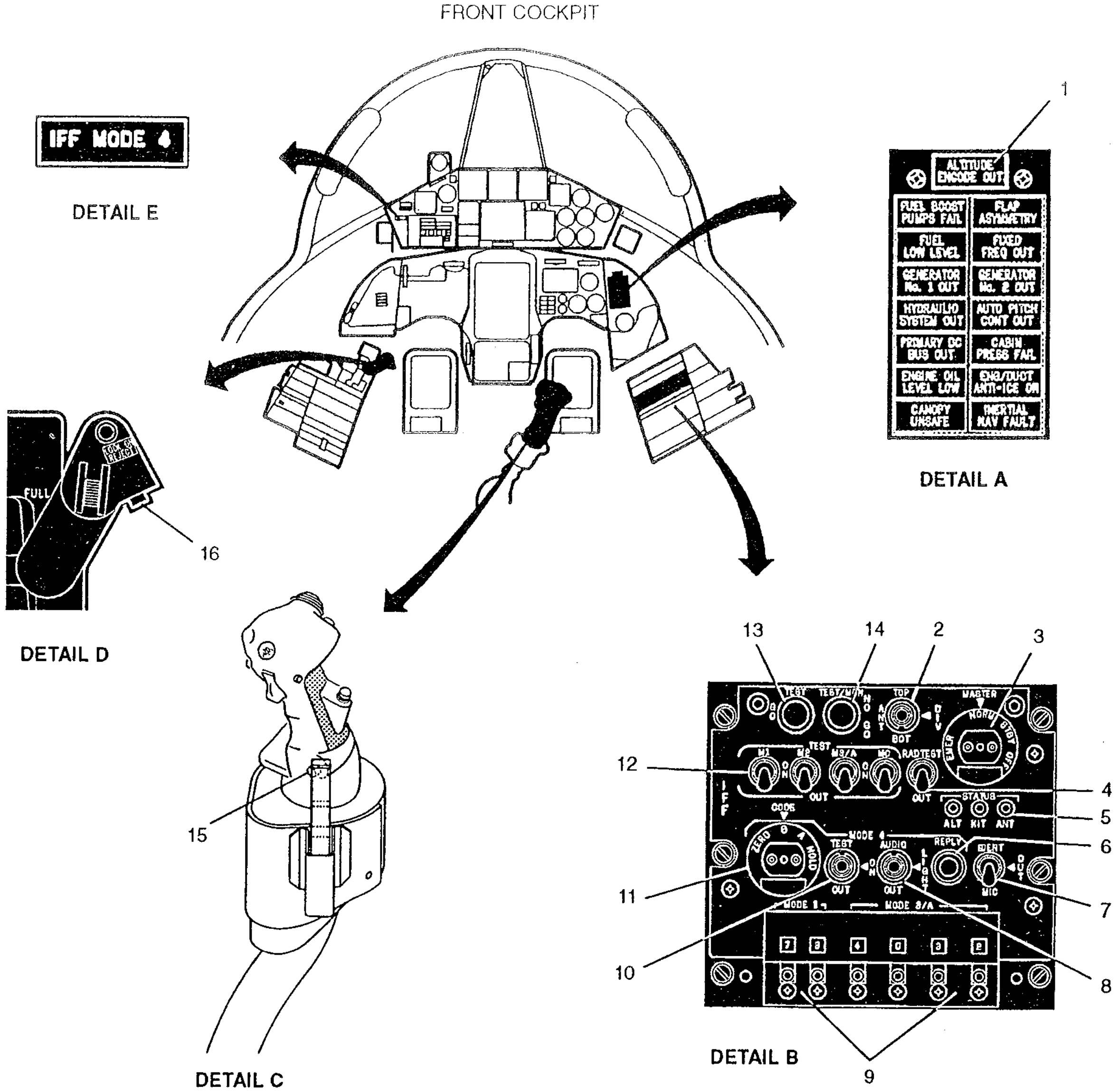
RAD TEST Enables an appropriately equipped transponder to reply to TEST mode interrogations from test equipment. In RAD TEST position, the switch is spring-loaded to return to the OUT position.

OUT Deenergizes RAD TEST position. The OUT position should be selected when activating a switch to TEST position.

ALT, KIT, ANT Lamps. These lamps are illuminated when M1, M2, M3/A, M4 or M-C mode switches are in TEST position. The ALT lamp illuminates when Altitude Computer (CPU 46/A) data are not received by the IFF system.
The KIT lamp is not operative.
The ANT lamp illuminates indicating a malfunction of one of the two antennas or that the standing wave ratio is higher than 3:1.

REPLY Lamp. The REPLY lamp is not operative.

IFF CONTROL PANEL (FRONT COCKPIT ONLY)



A – WARNING LIGHTS PANEL
1 ALTITUDE ENCODE OUT WARNING LIGHT

B – IFF CONTROL PANEL
2 ANTENNA SELECTOR SWITCH
3 MASTER SELECTOR KNOB
4 RADIATION TEST OUT SWITCH
5 STATUS ALT KIT ANT LAMP (KIT LAMP INOPERATIVE)
6 MODE 4 REPLY LAMP (INOPERATIVE)
7 IDENTIFICATION SWITCH
8 MODE 4 MONITOR SWITCH (INOPERATIVE)
9 MODE 3/A, 1 AND 2 CODE SELECTORS
10 MODE 4 TEST SWITCH (INOPERATIVE)
11 MODE 4 FUNCTIONAL KNOB (INOPERATIVE)
12 MODE SWITCHES
13 TEST LAMP
14 FAILURE/TEST LAMP

C – CONTROL STICK GRIP
15 NOSEWHEEL STEERING/MICHROPHONE BUTTON

D – ENGINE THROTTLE
16 MICROPHONE BUTTON

E – IFF MODE 4 WARNING LIGHT (INOPERATIVE)

FA0146

Figure 1-47

IDENT-OUT-MIC Switch. This three position switch has the following positions:

IDENT	When momentarily actuated (spring-loaded to return), enables a special reply for approximately 20 seconds.
OUT	Prevents transmission of identification position replies.
MIC	Enables identification of position replies to be transmitted for approximately 20 seconds every time the PTT pushbutton, located on the engine throttle or on the control stick grip, is pressed.

MODE 3/A Code Selectors. It includes four aligned pushbuttons permitting selection of the reply code for Mode 3/A (digits from 0 to 7).

MODE 1 Code Selectors. It includes two pushbuttons selecting the reply code for Mode 1. The left one selects a digit from 0 to 7, the next one a digit from 0 to 3.

Mode Switch. Four three-position toggle switches, labeled M-1, M-2, M-3/A and M-C, permit the following operation:

OUT	The corresponding mode is disabled.
ON	Predisposition replies when the IFF transponder is interrogated in the corresponding mode.

NOTE

Mode C is enabled with M-3/A switch to ON only.

TEST	Momentary position. A self-test is carried out (TEST GO light on), provided the MASTER selector is in NORM or EMER position.

IFF SYSTEM OPERATION

Operation of the system is obtained by moving the master selector knob from OFF to any position. The STBY position is used to warm up (the system is ready in 5 to 10 seconds) and maintain the system ready for operation.
The system is ready to respond to the interrogations only when the master selector knob is either in the NORM or EMER position. In case of ejection, the system permits transmission of the emergency signals regardless of the master selector knob position. The transponder normally operates with the master selector knob at NORM.

IFF MODES SYSTEM OPERATION

The IFF system may operate in the following modes:

- MODE 1
- MODE 2
- MODE 3/A
- MODE C

The M-1 mode switch, when set to ON, enables Mode 1 (security identity) operation of the transponder, by activating the Selective Identification Feature (SIF) decoder to recognize Mode 1 interrogations and enable Mode 1 replies. The settings of the Mode 1 code selector, on the IFF control panel, select the coders of the A and B pulse combinations for reply to valid Mode 1 interrogations. The Mode 1 code selector of the tens digit designates the group A pulses and the units digit designates the group B pulses. The coding capability is 32 code combinations. The mode and validity of the interrogation signals are determined by the SIF decoder. If the interrogation signal is valid, the SIF decoder enables the commutator circuit and the A and B coders to reply. The commutator circuit determines the timing and sequence of all pulses (initial frame pulse, code pulses selected via coders A and B, and final frame pulse) routed to the modulator drive for transmission.

The M-2 mode switch, when set to ON, enables Mode 2 (personal identity) operation of the transponder; in this condition the SIF decoder identifies Mode 2 interrogations and enables Mode 2 replies. Mode 2 code shall not be changed during flight: selection is carried out on the ground only. The Mode 2 code selectors, select combination of A, B, C and D group pulses in reply to valid Mode 2 interrogations. The mode and validity of the interrogation signals are determined by the SIF decoder. If the interrogation signal is valid, the SIF decoder enables the commutator circuit and the A, B, C and D coders to reply. The commutator circuit determines the timing and sequence of all pulses (initial frame pulse, code pulses selected via coders A, B, C and D, and final frame pulse) routed to the modulator driver for transmission.

The M-3/A mode switch, when set to ON, enables Mode 3/A (traffic identity) operation of the transponder; in this condition the SIF decoder identifies Mode 3/A interrogations and enables Mode 3/A replies. The settings of the Mode 3/A code selectors select combination of A, B, C and D pulse combinations for reply to valid Mode 3/A interrogations. The Mode 3/A code selector of the thousands digit designates group A pulses, the hundreds digit designates group B pulses, the tens digit designates group C pulses, and the units digit designates group D pulses. The mode and validity of the interrogation signals are determined by the SIF decoder. If the interrogation signal is valid, the SIF decoder enables the commutator circuit and the A, B, C and D coders to reply. The commutator circuit determines the timing and sequence of all pulses (initial frame pulse, code pulses selected via coders A, B, C and D, and final frame pulse) routed to the modulator driver for transmission.

The M-C mode switch, when set to ON, enables Mode C operation of the transponder.

NOTE

To permit Mode C operation, the Mode 3/A shall be set to ON.

In this condition the SIF decoder identifies Mode C (altitude reporting) interrogations and enables Mode 3/A replies. The digital signals reporting pressure altitude are routed from the altitude computer (CPU-46A) to coders A, B, C and D. When the IFF system is interrogated for altitude reporting (Mode C), the IFF transmitter-receiver will automatically report the aircraft altitude to the nearest 100 feet for a barometric pressure of 1013 mb., regardless of the altimeter barometric setting.

NOTE

- The IFF Mode "C" is available regardless of the ELECT or PNEU servo/pneumatic altimeter setting, provided the altitude computer (CPU-46A) is operative.
- Following altitude computer (CPU-46A) failure, the IFF Mode "C" is lost and the servo-pneumatic altimeter switches to baro mode (PNEU). As a consequence the ALTITUDE ENCODE OUT (front cockpit only) warning light illuminates.

The digital signals select the pulse combinations for reply to valid Mode C interrogations. The mode and validity of the interrogation signals are determined by the SIF decoder. If the interrogation signal is valid, the SIF decoder enables the commutator circuit and the A, B, C and D coders to reply. The commutator circuit determines the timing and sequence of all pulses (initial frame pulse, code pulses selected via coders A, B, C and D, and final frame pulse) routed to the modulator driver for transmission.

System Starting

To start the system proceed as follows:

1. MASTER selector – STBY for 1 minute under standard temperature conditions (5 minutes under extreme temperature conditions), then to NORM
2. Mode code selectors – As required
3. Mode switches – On
4. IDENT/OUT/MIC switch – OUT
5. ANT switch – DIV

Identification-of-Position Operation

When the IDENT-OUT-MIC switch is energized, the system transmits position identifying signals to all interrogating stations on Modes 1, 2 and 3/A. Transmission of the identification-of-position signal occurs in these modes even if the mode switches are in the OUT position. The two types of identification-of-position are as follows:

1. Momentarily hold the IDENT-OUT-MIC switch in the IDENT position, then release. This action causes the identification-of-position signal to be transmitted for a period of 20 seconds to all interrogation stations on Mode 1, 2 and 3/A. Repeat as required
2. Set the IDENT-OUT-MIC switch to the MIC position. Identification-of-position signals are transmitted by pressing one of two PTT pushbuttons located on the control stick grip or on the engine throttle. When the need for further identification signals has ended, return the IDENT-OUT-MIC switch to the OUT position

Emergency Operation

During an emergency or distress condition, with the master selector knob set to EMER, the IFF system transmits automatically in the mode corrisponding to the interrogation signal received. This occurs independently of the selected mode and the response code selected manually on the panel. The code signals are transmitted as follows:

- For 1, 2 and 3/A Modes, the military emergency reply consists of 4 pulse trains: the first train contains the selected mode code if the reply is Mode 1 or 2; code 7700 if the reply is Mode 3/A. The remaining 3 pulse trains merely consist of a pair of "frame" pulses spaced from the first reply framing pulse.

- The emergency selection enables Mode C regardless of control settings, but does not affect the reply code format of these modes (the transmitted codes are the same used during normal operation).

These emergency codes are transmitted as long as the master selector knob is set to EMER position. The transmission of an emergency reply requires that an interrogation signal be received. For emergency operation, proceed as follows:

A. Pull and rotate the master selector knob to the EMER position.

B. Leave the master selector knob at EMER for the duration of the emergency.

C. When the emergency has ended, return the master selector knob to the NORM position.

NOTE

The emergency functions are automatically enabled, through a pull connector on the ejection seat, when the pilot ejects, regardless of the master selector knob position.

IFF System Self-Test

Self-test is possible for Modes 1, 2, 3/A and C. With the MASTER selector placed in NORM position, self-test is commenced by turning one of the Mode switches to TEST. A signal generated for the selected mode of operation arrives at the set and is processed as a normal interrogation signal. Normal operation of the mode being tested is indicated by the illumination of the TEST GO lamp.

AIR DATA COMPUTER

The air data computer is located in the electronics compartment. The system utilizes power from the PP1 and XP7 busses. The transducers accept input information in the form of angle of attack, pitot pressure, static pressure, and total temperature. These inputs are then converted to electrical signal outputs of air density ratio, angle of attack, impact pressure, Mach number, pressure altitude, total pressure, and true airspeed.
They are then automatically supplied to the using equipment (when operating) as follows:

OUTPUT	USING EQUIPMENT
Pressure altitude Angle of attack	RADAR
Pressure altitude	Inertial Navigator
Impact pressure Pressure altitude	Landing Gear Warning

NOTE

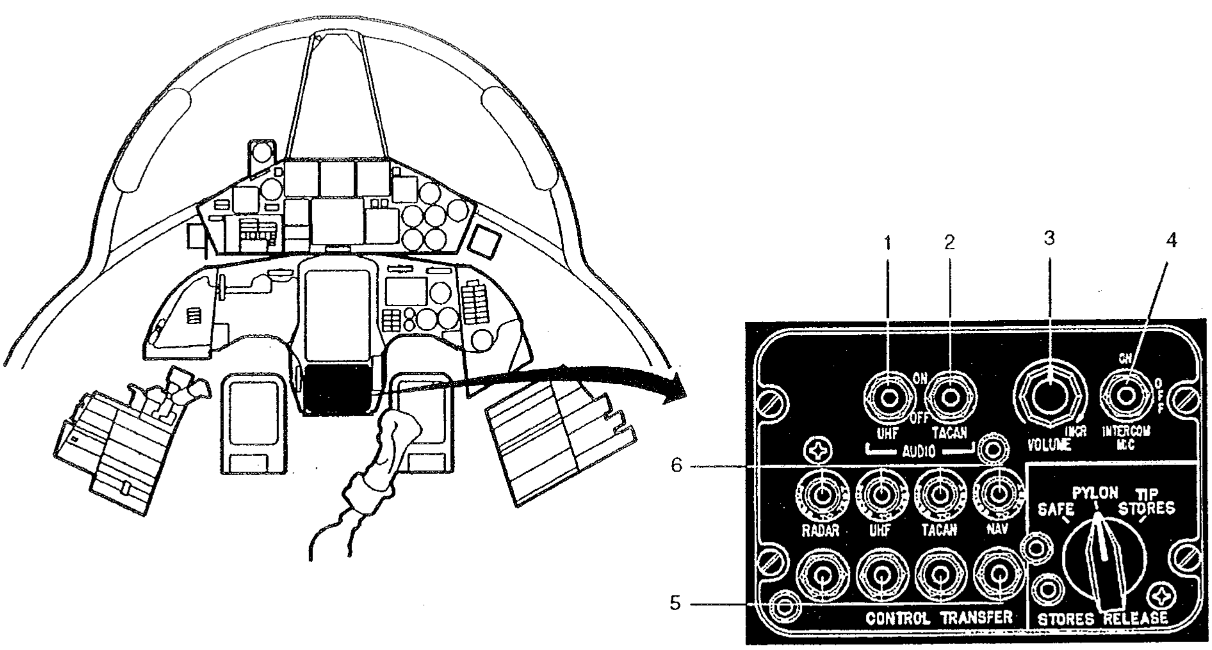
The air data computer is entirely automatic, containing no indicators or pilot-operated controls. Consequently, failure can only be detected by malfunction of any or all of the above using equipment.

CONTROL TRANSFER PANEL

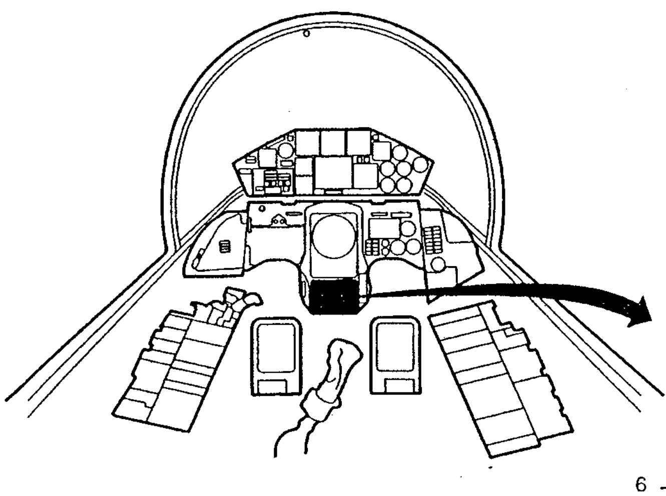
The control transfer panel is located on the lower instrument panel below the radar indicator in each cockpit (see Figure 1-48). Each panel contains four transfer switches for transfer control of the RADAR, UHF, TACAN, and NAV (IN/CDU, HSI, IN/SEL, TCN/SEL selectors); and four corresponding indicator lights.
Each panel also contains an auxiliary volume control for the UHF, TACAN, and intercom mic, and

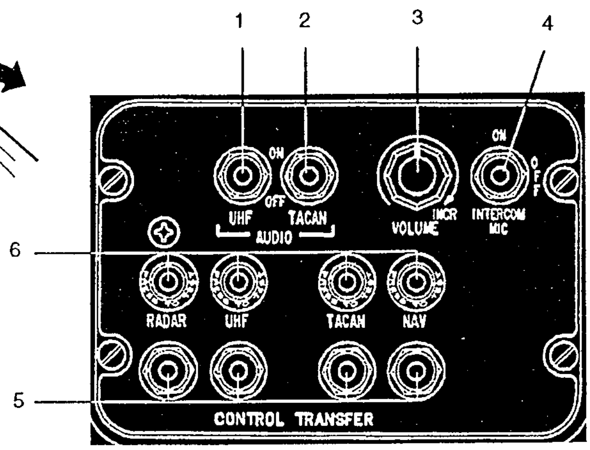
CONTROL TRANSFER PANEL

FRONT COCKPIT



REAR COCKPIT





1 UHF VOLUME SELECTOR SWITCH
2 TACAN VOLUME SELECTOR SWITCH
3 AUXILIARY VOLUME CONTROL KNOB
4 INTERCOM MIC VOLUME SELECTOR SWITCH
5 CONTROL TRANSFER INDICATOR LIGHTS
6 CONTROL TRANSFER SWITCHES

FA0166

Figure 1-48

three corresponding ON-OFF volume selector switches. In addition to the control transfer switches, the front cockpit control panel contains the stores release selector switch.

Control Transfer Switches and Indicator Lights

The four control transfer switches are located at the bottom of the control transfer panel. They are momentary type switches, spring loaded to the OFF (center) position. Beneath each switch is the individual placarded: (from left to right) RADAR, UHF, TACAN, and NAV.
Indicator lights are located immediately above each transfer switch. To transfer control of the RADAR, UHF, TACAN, or NAV between cockpits hold the selected switch up until the corresponding indicator light illuminates. Thus, control may be taken from one cockpit by the other. However, it cannot be returned. The transfer switches and lights receive power from the PP1 bus. The equipment functions as follows:

RADAR. Refer to Radar System paragraph in this Section.

UHF. The operator in the cockpit that does not have control may transmit and receive but cannot change frequencies.

TACAN. The operator in the cockpit that does not have control may receive and observe bearing, distance, drift, TO/FROM and course error information on the HSI indicator, but cannot change channels.

NAV. A moding function shall be provided to share the control of some equipment/functionalities. This shall be done by means of NAV selector (momentary switch) available in both the front and the rear cockpits. The NAV selector, via the switching box to IN, serves to acquire the control of the IN/CDU, IN and TACAN steering mode selection by the IN/SEL and TCN/SEL selector pushbuttons, plus the course knob of the HSI. The equipment located in the cockpit currently selected by means of NAV selector will be named "master", the other "slave". The slave IN/CDU shall work as simple display showing all the selection made on the master IN/CDU, and all its pushbuttons shall be disabled. In any moment, it shall show the same information displayed by the master one. The slave HSI shall present a commanded course indication oriented as the desired course set on the master one.

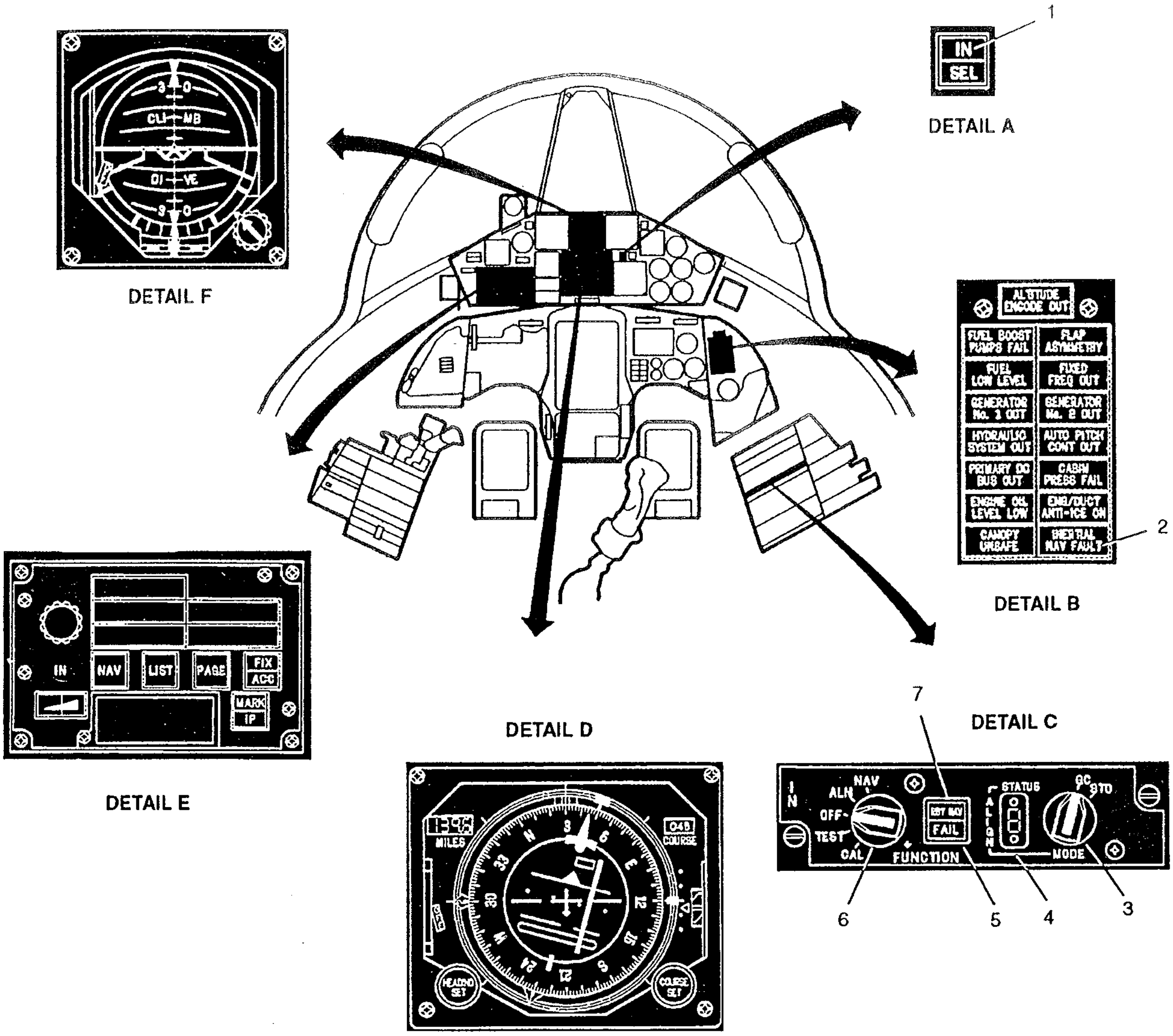
Auxiliary Volume Control and Individual Volume Selector Switches

The auxiliary volume control knob and individual volume selector switches are located along the top of the control transfer panel. The switches have 2 positions, ON and OFF, and are placarded UHF, TACAN and INTERCOM MIC. They provide selected volume control for the UHF, TACAN or intercom.
The volume control knob adjusts the audio level of the selected equipment and receives power from the UHF command radio.
When the INTERCOM MIC switch is ON in either cockpit, communication between cockpits is available. It is not necessary for both INTERCOM MIC switches to be on. With the INTERCOM MIC switch in the ON position, the pilot may communicate with the ground crew through a jack located on the external power receptacle.

INERTIAL NAVIGATION SYSTEM (INS)

The LN39A2 Inertial Navigation System consists of an inertial navigation unit (INU) and a control panel labeled IN in the forward cockpit only. The INU is a self contained, fully automatic, gyro stabilized platform with built-in test equipment facilities. The IN shall provide generation of present position, attitude angles, velocity components and magnetic heading parameters. In addition to these typical functions, it shall provide processing functions and interfacing capabilities for other equipments.
The system shall support analog interface with two Horizontal Situation Indicators (HSI), two Attitude Indicators (AI), Air Data Computer (ADC), Adapter and two Warning Panels; moreover it shall communicate with two CDUs by means of two EIA RS-422 interface.
The pilot in the front cockpit shall control the INU equipment by means of a dedicated Control Panel (refer to Figure 1-49).
The INU receives primary AC power from the XP5 bus, in case of failure of the fixed frequency generator the IN is switched on the wild frequency generator and powered from XP4 bus; concerning the DC power the IN receive supply from PP1 bus and, as back-up, supply from the PP4 battery bus.
The system Function Electrical Interface is shown in Figure 1-50. An avionic system architecture prospect is shown in Figure 1-51.

INERTIAL NAVIGATION SYSTEM CONTROLS AND INDICATORS (FRONT COCKPIT)



A – NAVIGATION STEERING MODE SELECTOR PUSHBUTTON
1 IN INDICATOR

B – WARNING LIGHTS PANEL
2 INERTIAL NAV FAULT WARNING LIGHT

C – INERTIAL NAVIGATOR CONTROL PANEL
3 MODE SELECTOR KNOB
4 ALIGNMENT STATUS INDICATOR
5 FAIL LAMP
6 FUNCTION SELECTOR KNOB
7 READY NAVIGATION LAMP

D – HORIZONTAL SITUATION INDICATOR

E – IN/CDU

F – ATTITUDE INDICATOR

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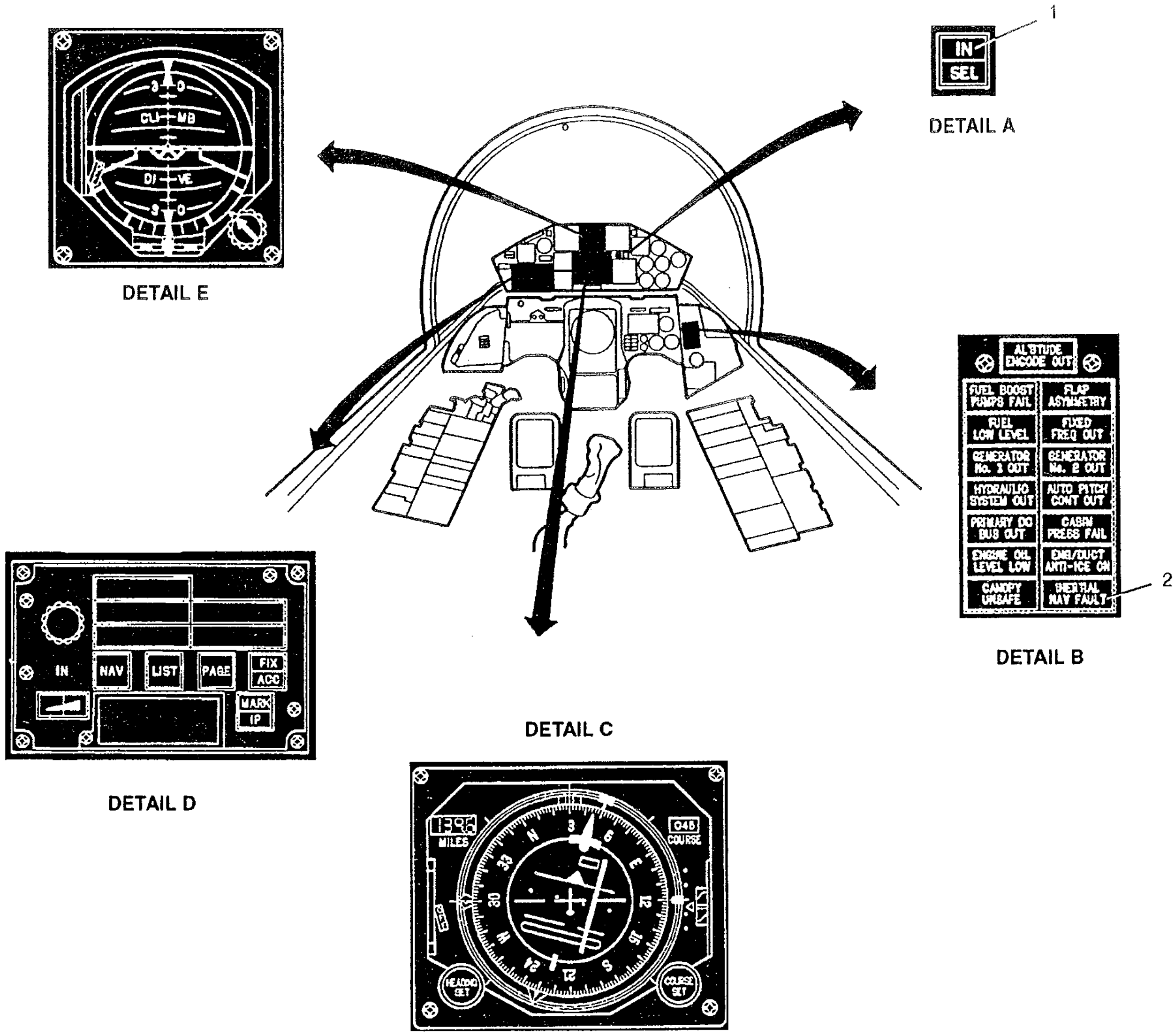
Figure 1-49 (Sheet 1 of 2)

INERTIAL NAVIGATION SYSTEM CONTROLS AND INDICATORS (REAR COCKPIT)



A – NAVIGATION STEERING MODE SELECTOR PUSHBUTTON
1 IN INDICATOR

B – WARNING LIGHTS PANEL
2 INERTIAL NAV FAULT WARNING LIGHT

C – HORIZONTAL SITUATION INDICATOR

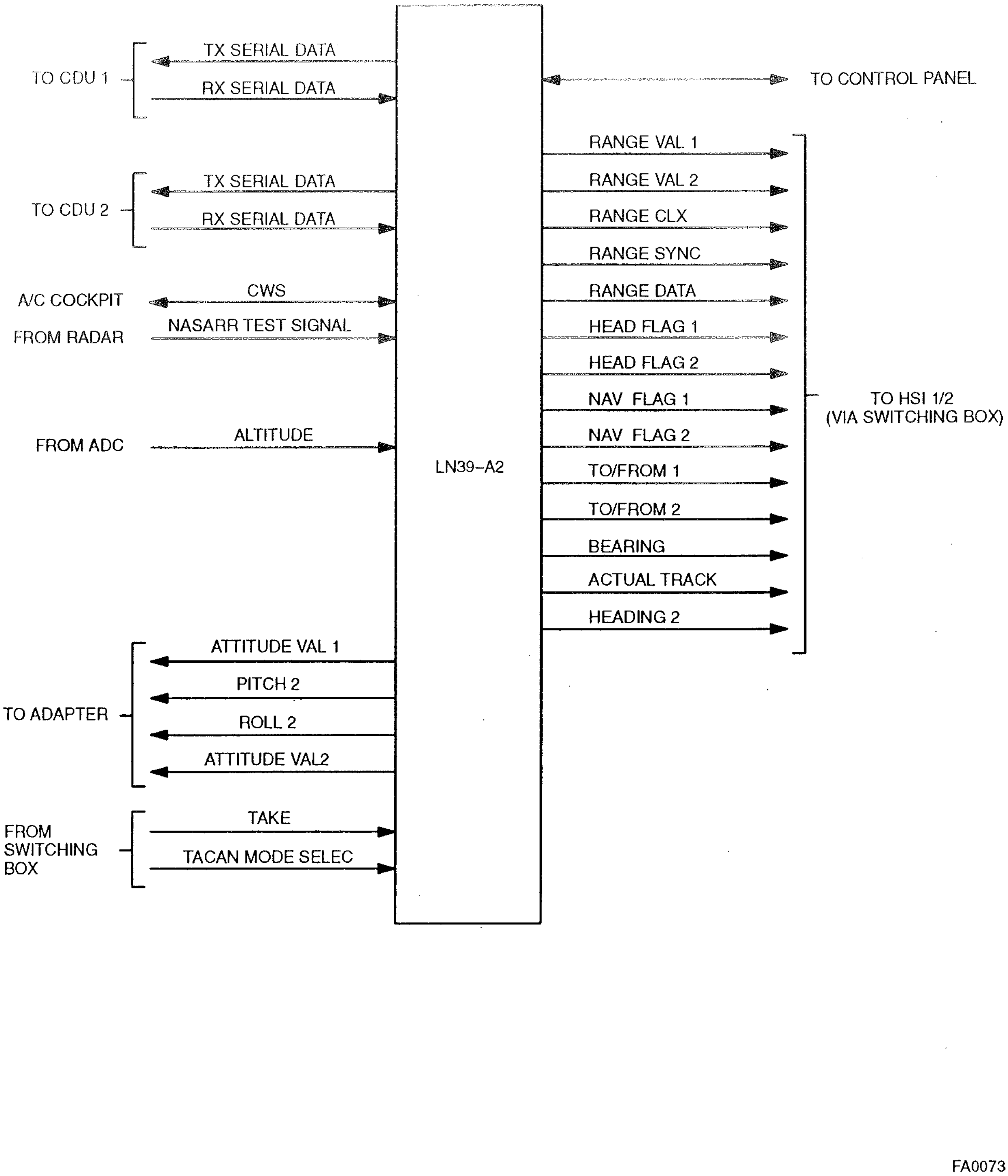
D – IN/CDU

E – ATTITUDE INDICATOR

FA0149

Figure 1-49 (Sheet 2 of 2)

INS — FUNCTIONAL ELECTRICAL INTERFACE



CONNECTIONS TO HSI

Signal marked as 1 and 2 are generated by the INS separately from the equipment in the front and rear cockpit. Bearing, Actual Track and Heading drive in parallel (via switching box) the front and rear HSI.

Figure 1-50

AVIONIC SYSTEM ARCHITECTURE

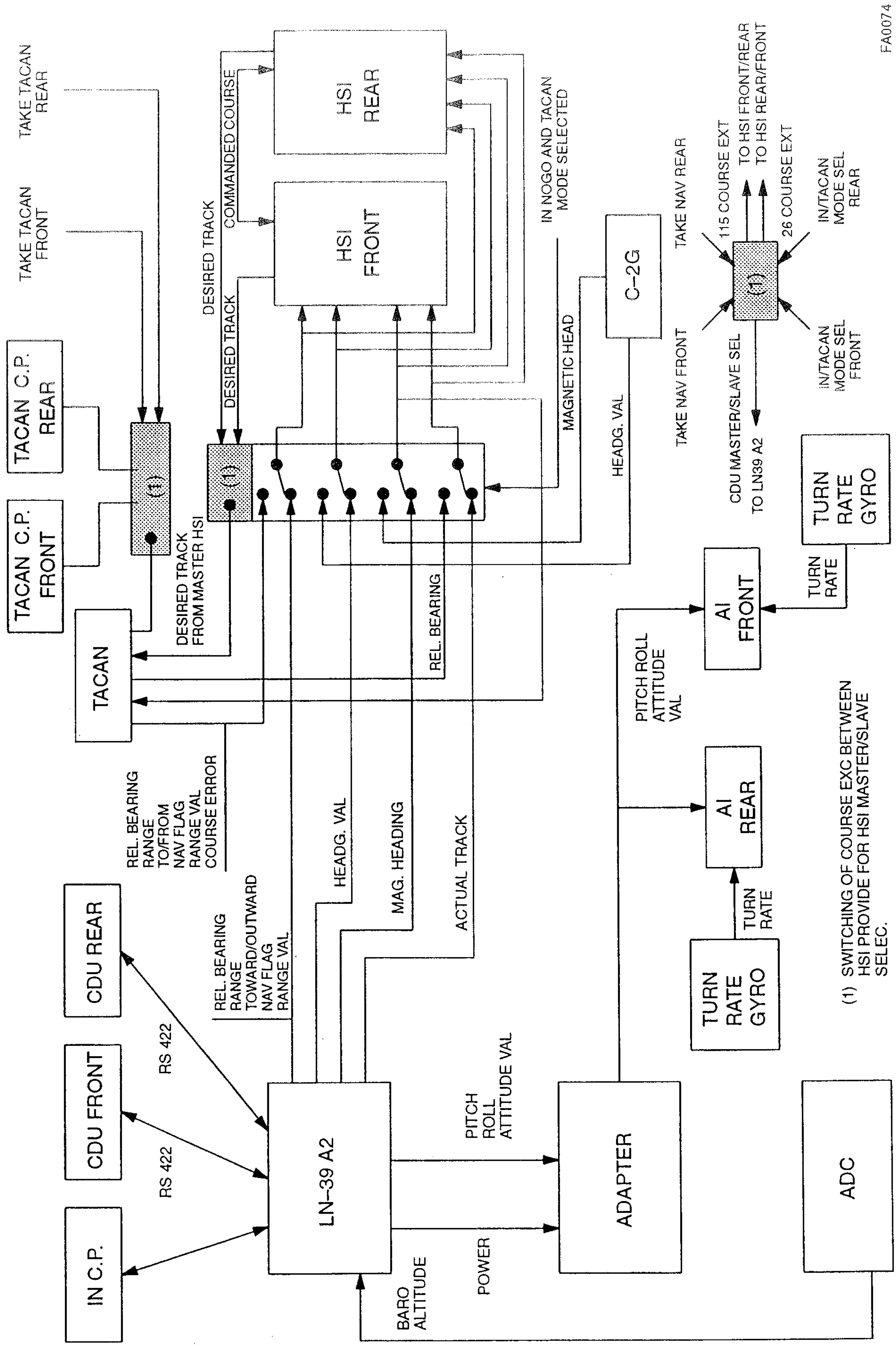


Figure 1-51

INS CONTROLS AND INDICATORS

The INS controls and indicators are illustrated in Figure 1-49.

INS Control Panel

The INS control panel, labelled IN and located on the right console, has the following controls and indicators:

Function Selector Knob. This knob labeled FUNCTION has five positions:

OFF — The system is turned off

NOTE

> The INS switching off causes the deletion of all station points (including the mark point, if any).

TEST — Permits the IN self-test

ALN — Provides the IN alignment in the GC or STO mode

NAV — The IN is operative for navigation

CAL — Permits the IN calibration by ground personnel only

Mode Selector Knob. This knob labeled MODE has two positions:

GC — Selects the gyrocompassing alignment mode

STO — Selects the alignment mode which employs the stored heading

Ready Navigation Lamp. This lamp, labeled RDY NAV, illuminates steadly to indicate that the IN has got an alignment status sufficient to sustain the NAV mode.

NOTE

> The flashing RDY NAV indication means the achievement of maximum alignment status for each alignment selected mode.

Fail Lamp. This amber lamp, labeled FAIL, illuminates if an INS failure occurs.

Alignment Status Indicator. This indicator, labeled ALIGN/STATUS, is a one digit display that indicates approximately the radial position error in nautical miles, for each hour of flight. If an INS failure occurs the alignment status indicator extinguishes.

NOTE

> During the alignment procedure, for all modes, the INU will compute the present alignment status and make it available on the alignment status indicator, while fixing it when the NAV mode is selected.

Warning Light Panel

The INERTIAL NAV FAULT warning lamp incorporated in the warning lights panel up whenever a failure is detected in the INS.

Navigation Steering Mode Selector Pushbutton

The navigation steering mode selector pushbutton, located on the main instrument panels is a dual captions press-to-select pushbutton. The IN caption is always illuminated. Pressing the pushbutton the SEL caption illuminates indicating the selection of the INS as steering mode source.

NOTE

> - The IN caption is automatically lit after electrical power on.
> - Refer to Navigation Sub-System paragraph for further information.

Horizontal Situation Indicator (HSI)

The HSI will show steering parameters and navigation information.

Attitude Indicator

The attitude indicator shows pitch and roll information.

IN/CDU

The IN/CDU shows all loaded mission data, navigation data, as well as fixing error and mark points.

INS MODE OF OPERATION

ALIGN Mode — "ALN"

In the ALIGN mode, system operation shall be initiated by selecting the ALIGN method with the CP ALIGN MODE switch and then switching the CP FUNCTION switch from "OFF" to "ALN" position. At the switch-on (selector on ALN) the system shall download the mission from the DTM previously inserted by the pilot in the socket on the front IN/CDU, that includes information of latitude, longitude, ICAO code, station height above the sea level and TACAN channel, to the navigation subsystem, validating, by means of the single waypoint checksums and by means of the checksum of the whole database, the integrity of the data. The following Align modes shall be available:

Gyrocompassing — "GC"

During gyrocompassing alignment, the equipment shall perform a self-alignment and levelling procedure requiring only input of initial position (IP).
Aircraft initial position is acquired and entered when the function selector knob is set to ALN. During the first 2 minutes period, the initial position may be updated and used by IN without restart of the IN alignment procedure. A full gyrocompassing alignment or a rapid gyrocompassing alignment shall be achieved depending upon time at which navigate function is selected.
When rapid gyrocompassing alignment is completed, the INU shall provide a steady "RDY NAV" indication and the ALIGN STATUS indicator shall display "3": at this time the "NAV" position may be selected if a fast alignment is desired.
Otherwise the "NAV" position should be selected after the "RDY NAV" lamp flashing is initiated, indicating that a full gyrocompass alignment is completed (STATUS "1"). During alignment, the INU shall compute and make available for display an alignment status indication as defined in Figure 1-52.

Stored Heading — "STO"

Stored Heading alignment mode shall be a fast alignment mode having as pre-requisites the following:

- the aircraft is stopped and a full Gyrocompassing alignment is performed before IN shut down
- the aircraft is not moved prior to next alignment.

During Stored Heading alignment the Equipment shall perform a levelling procedure and shall align to the Last Stored Heading utilizing the Last Stored present position for initialization. In case of input of initial position it shall be ignored. When Stored Heading alignment is completed the INU shall provide a flashing "RDY NAV" light indication to CP. As this time the NAV mode shall be selected.
During Stored Heading alignment mode the INU shall compute and make available for display on CP, an alignment status indication as defined in Figure 1-52.

NAVIGATE Mode — "NAV"

NAV mode is the flight mode of operation and shall be selected after a satisfactory alignment is reached and the "RDY NAV" lamp is lit or flashing. When NAV is selected the IN shall enter the navigation mode and the "RDY NAV" extinguishes.

NOTE

In the event of an internal computer failure or the selection of NAV mode prior to the system capability of sustaining NAV mode, the IN shall revert to a first order levelling mechanization in order to avoid platform damage during flight.

CALIBRATE Mode — "CAL"

A CAL mode shall provide automatic calibration of three axis gyro bias drift. The CAL mode shall require no more than 90 minutes to complete, and shall include provision for updating the affected

ALIGN STATUS PREDICTED NAVIGATION ACCURACY DEFINITION

ALIGN STATUS	DEFINITION	MODE	RDY NAV
8	IN is performing levelling procedure If NAV is selected the IN fails	GC or STO	--
7 ÷ 3	IN is performing gyrocompassing alignment or stored alignment If NAV is selected the navigation is performed with degraded performance The navigation accuracy is predicted to be equivalent to the Align Status number displayed, expressed in nm/hr (C.E.P.)	GC RAPID or STO	STEADY "3" FLASH "3"
3 ÷ 1	The navigation accuracy is predicted to be equivalent to the Align Status number displayed, expressed in nm/hr (C.E.P.)	GC FULL	FLASH "1"

Note 1: Align status codes shall be used for operator information only and do not imply system performance requirement

Note 2: Align status 7 shall be achieved in 7.5 minutes max. after turn-on at T = −40 °C

Figure 1-52

calibration constants stored in the INU. The CAL mode does not require external inputs other than aircraft present position for operation. It shall operate in the ground alignment environment, without the need for any support equipment. At the end of calibration the "RDY NAV" flashing indication is displayed.

TEST Mode – "TEST"

To perform correctly the INS self-test the aircraft shall not be moved until the test is completed. The INS self-test is initiated by setting the function selector knob to TEST position. During the self-test procedure the alignment status indicator will provide a numeric indication varying from 8 to 1. When the number 1 is displayed it means that the self-test is valid and completed. If a INS failure is detected during the self-test procedure, the FAIL lamp comes on while, the alignment status indicator, maintains a fixed number on the display.

INS OPERATION

To operate the system proceed as follows:

A. Set the mode selector knob to the desired alignment mode (GC or STO)

B. Set the function selector knob from OFF to ALN. When the ALN function is selected the INS platform alignment is initiated.

NOTE

The INS may be switched off during alignment procedure (or in other term before setting the function selector knob to NAV) by setting the function selector knob to OFF without causing damage to the system. After the alignment status indicator digit extinguishes, the function selector knob may be set again to ALN position.

C. Set the function selector knob from ALN to NAV position after a satisfactory alignment is completed. The INS navigate function is initiated and the "RDY NAV" caption extinguish.

For IN alignment and emergency procedures refer to Sections II and III respectively.

TACTICAL AIR NAVIGATION (TACAN) SYSTEM

The tactical air navigation system consists of a receiver-transmitter unit installed in the after cockpit lower hatch, two control panels installed in the cockpit, and an antenna. The equipment receives power from the PP1 and XP6 busses. The system operates from 962 MHz to 1213 MHz.
126 two-way operating channels, spaced 1 megacycle apart, are available on X and Y mode.
The equipment operates to provide continuous distance and bearing information from any received surface beacon within a line-of-sight distance from the aircraft of up to 390 NM nautical miles or less depending on aircraft altitude and station location. The pilot, knowing his approximate location, may select a nearby beacon and navigate by it. Visual indication of bearing to the station is provided by the pointer of the HSI system. Range to the station is also displayed on the HSI. Following a correct TACAN station acquisition, the station identification audio signal is sent to the pilot's headset.
The equipment is provided with a self-test capability automatically activated at the switch-on or manually selected by pressing the TEST button: the test lamp flashes once. During the first two seconds of the test the HSI displayes are not valid, then bearing displays 180° for six seconds. At the end of the test, normal HSI/TACAN conditions are resumed.

TACAN Control Panel

The TACAN control panel (Figure 1-53) is located on the right console in both cockpits. The panel contains the following controls.

Channel Window. A four digit display shows the selected operating channel and mode of operation.

Function Selector Switch. The function selector is a five-position rotary type knob, placarded clockwise as follows:

OFF	The system is turned off.
REC	The system provides indication of the bearing to the selected station on HSI.
T/R	The system provides bearing and slant range distance to the selected station on HSI.
A/A REC	The system provides bearing to the cooperating aircraft on HSI.
A/A T/R	The system provides bearing and distance to the cooperating aircraft on HSI.

Volume Control Knob. Ground beacon identification audio level may be adjusted by rotating the knob placarded VOL to the right or left as desired.

Channel/Mode Selector Knobs. The channel selector on the left selects hundred and tens (digit 0-120). The selector on the right contains two concentric knobs. The inner knob selects units (digit 0-9), the outer knob selects X/Y modes.

TEST Button. When pressed it test the TACAN.

TEST Lamp. It flashes once automatically after switch-on or when the TEST button is pressed, indicating that the TACAN system self test has been initiated. It also illuminates if a TACAN malfunction is detected during self test or normal operation.

Navigation Steering Mode Selector Pushbutton

The navigation steering mode selector pushbutton, located on the main instrument panel on both cockpits, is a dual captions press-to-select pushbutton. The TCN caption is always illuminated. Pressing the pushbutton the SEL caption illuminates indicating the selection of the TACAN as steering mode source.

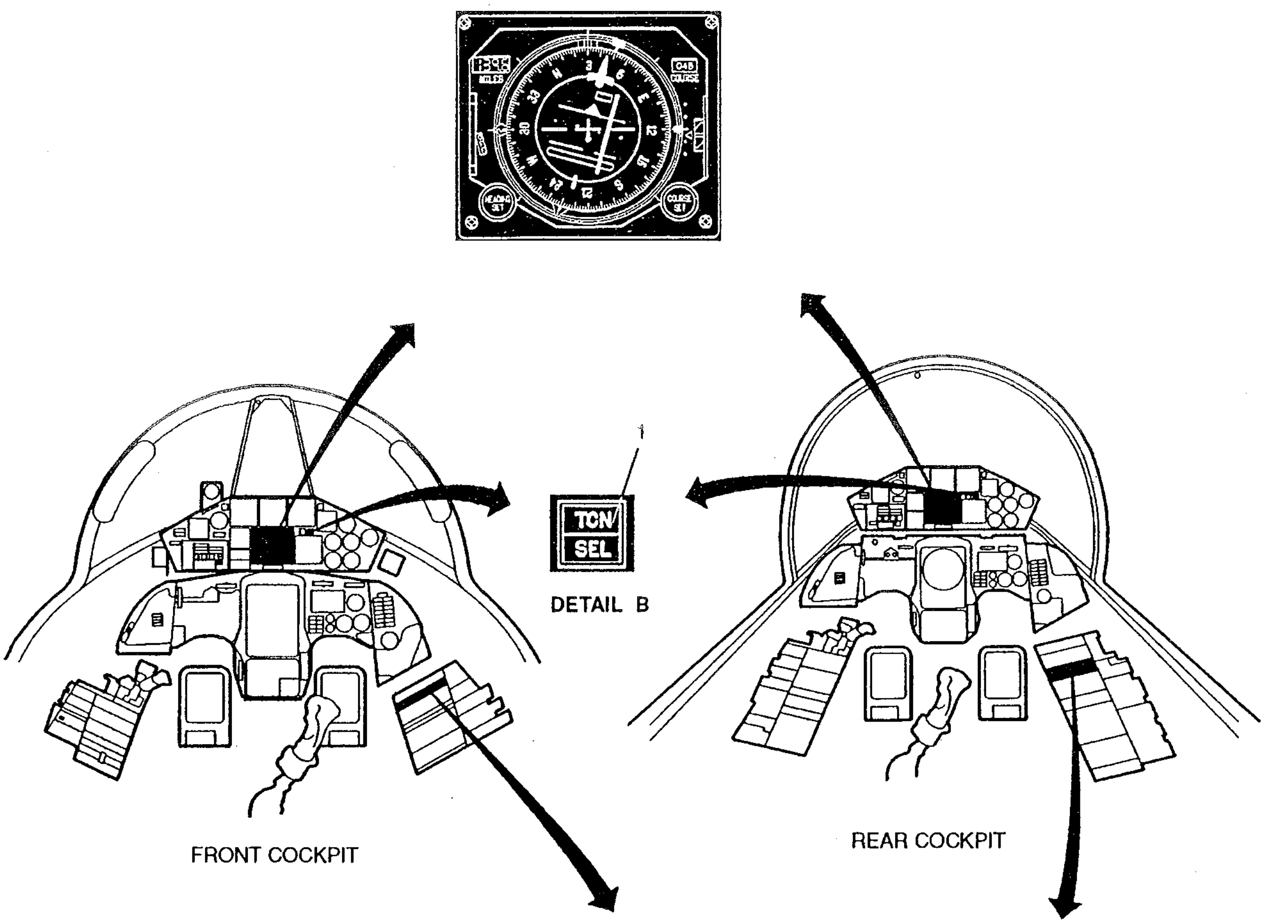
NOTE

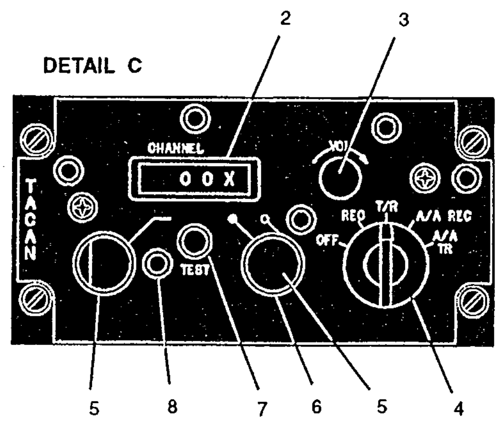
- The TCN caption is automatically lit after electrical power on.
- Refer to Navigation Sub-System paragraph for further information.

Horizontal Situation Indicator (HSI)

The HSI will show bearing and range information to the selected TACAN station.

TACAN CONTROL PANEL





A – HORIZONTAL SITUATION INDICATOR (HSI)

B – NAVIGATION STEERING MODE SELECTOR PUSHBUTTON
1 TCN INDICATOR

C – TACAN CONTROL PANEL
2 CHANNEL WINDOW
3 VOLUME CONTROL KNOB
4 FUNCTION SELECTOR SWITCH
5 CHANNEL SELECTOR KNOBS
6 MODE SELECTOR KNOB
7 TEST LAMP
8 TEST BUTTON

FA0147

Figure 1-53

TACAN Operation

To operate the TACAN equipment, proceed as follows:

1. Channel/Mode selector knobs — Select desired channel and mode of operation
2. Function selector switch — As desired
3. Volume control knob — As desired
4. Verify station identification
5. TCN navigation steering mode selector push-button — Press, check SEL caption lit
6. Set desired course on HSI
7. Monitor bearing TO/FROM and course Deviation on HSI
8. Observe distance and bearing to the station as indicated by the HSI

NOTE

In case of IN fail, perform a C2-G synchronize operation.

TACAN Station Approach from a Desired Radial

In conjunction with the HSI, the TACAN equipment allows the approach of the selected TACAN station from a desired radial. The course pointer and the digital course readout on the HSI indicate the desired course manually set using the COURSE SET knob. The COURSE SET knob on HSI can be operated from the master cockpit only. The HSI in the slave cockpit operates as a repeater. The TACAN calculates the track angular error, as a function of the desired course setting and the selected TACAN station bearing, to drive the HSI course deviation bar (5° per dot). Rightward deflection of the HSI course deviation bar requires a rightward correction maneuver of the aircraft while, leftward deflection requires an analogous leftward maneuver. The "TO/FROM" arrows of the HSI are also driven from the TACAN, and are referred to an axis perpendicular to the set desired course which passes through the selected TACAN station and delimits two zones: "TO" is the zone containing the aircraft, the other is "FROM".

C-2G DIRECTIONAL GYRO SYSTEM

The C-2G directional gyro system provides stabilized magnetic heading to the HSI when the TCN navigation mode is selected following an INS failure. The system consists basically of a gyro-amplifier, a compass control panel in the front cockpit only, a power converter, and a flux valve.
The system receives power from the XP7 bus and from the XP6 bus.

C-2G Compass Control Panel

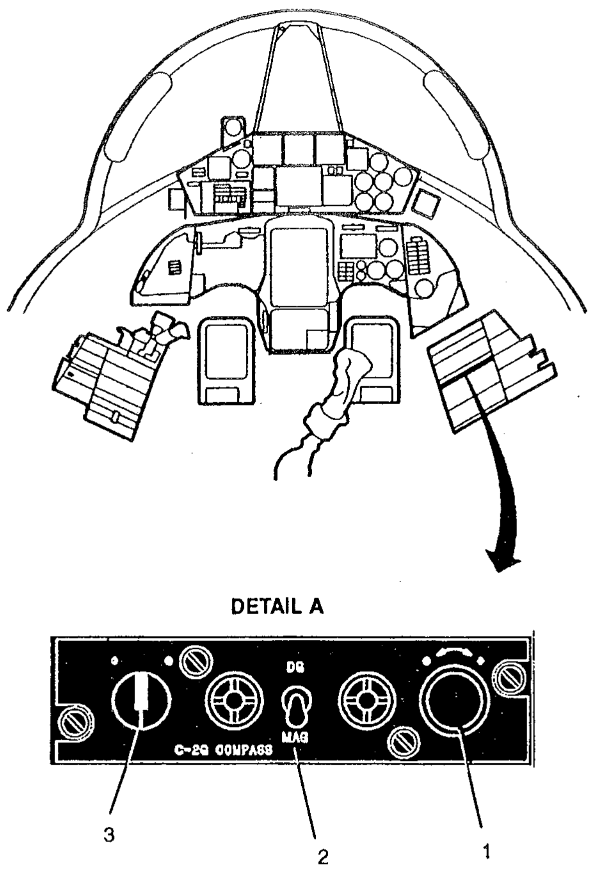
The C-2G compass control panel (refer to Figure 1-54) is located on the right console in the front cockpit only. The panel contains the following controls.

DG-MAG Selector Switch. The DG/MAG selector switch (refer to Figure 1-54) is provided to select the mode of operation of the directional gyro system. When this switch is in the MAG position, the system operates as a normal slaved gyro magnetic compass, and the directional indicator responds accordingly. Setting the DG/MAG selector switch to the DG position causes the compass to operate as a directional gyro.
This mode of operation is designed for use in the polar regions where the earth's magnetic field is such that the slaved gyro magnetic compass is very unreliable.

Annunciator. The annunciator (refer to Figure 1-54) indicates in the MAG mode of operation whether or not the heading output and the sensed direction of the flux valve are the same. When they are the same, the white bar on the annunciator dial will be centered, indicating system synchronization.
If they are not the same, the discrepancy is evidenced by the white bar moving toward the dot (•) or cross (+) markings on the panel, depending on the direction of this misalignment. This fluctuation is normal and indicates that the compass system is operating properly. The annunciator is inoperative in the DG mode of operation.

Synchronizing Knob. The synchronizing knob (refer to Figure 1-54) is used to manually synchronize the system. In the MAG mode of operation the knob is rotated in the direction of the arrow on the panel, depending on the direction of misalignment indicated by the annunciator.

C-2G COMPASS CONTROL PANEL (FRONT COCKPIT ONLY)



A – C-2G CONTROL PANEL
1 SYNCHRONIZING KNOB
2 DG–MAG SELECTOR SWITCH
3 ANNUNCIATOR

FA0145

Figure 1-54

If the white bar is toward the dot, the knob shall be rotated counterclockwise in the direction of the arrow, toward the dot marking on the panel beside the knob.
In the DG mode of operation the synchronizing knob may be used to arbitrarily rotate the heading output. The design of the C-2G synchronizing circuit and control makes it extremely difficult to remove errors incurred when leveling off after a turn, by fast synchronization. If extreme care is not used, it is possible to get an instantaneous null which could be false.
This false null, if it does occur, will be for only a short duration; therefore, after nulling the SYNC needle, a recheck should be accomplished after the aircraft is straight and level.

CAUTION

- IT IS RECOMMENDED THAT IN CASE OF "INS" FAILURE THE PILOT AT ALL TIMES USES THE C-2G COMPASS AS A REFERENCE AND CROSS-CHECK READINGS WITH STANDBY COMPASS.
- IF THE SYNCHRONIZING KNOB IS ROTATED OPPOSITE TO THE DIRECTION INDICATED, THE SYSTEM MAY SYNCHRONIZE ON A FALSE NULL 180° FROM TRUE HEADING. THIS NULL IS UNSTABLE.

NOTE

During acceleration, the flux gate pendulum will swing in the direction of the force applied; therefore, the C-2G compass should be synchronized and checked only after straight and level unaccelerated flight has been attained.

INERTIAL NAVIGATION CONTROL DISPLAY UNIT (IN/CDU)

The two IN/CDUs operate as an input/output terminal and are connected to the INU by means of two RS422 interface.

NOTE

Following an INS failure, the IN/CDUs are lost.

As input terminal they provide a means for the mission data base loading using a Data Transfer Module (DTM) in the front cockpit only, which contains an electrical erasable programmable memory (EPROM) in which all mission data are stored. The mission data base consists of a maximum of 58 generic and/or radio station waypoints identified from 01 to 58. Points having latitude, longitude and height data are defined as generic waypoints. Points having latitude, longitude, height and TACAN channel/mode data are defined as radio waypoints.

CAUTION

AS LONG AS THE IN/CDU EQUIPMENT IS POWERED THE DTM SHALL NOT BE REMOVED FROM ITS HOUSING.

NOTE

- The DTM shall be inserted before INS switch-on (ALN selection). If the INS is already switched-on, the insertion of the DTM is ignored.
- The "DTM FAIL" indication is displayed if the validation of the DTM fails.

- If before power on the DTM is not inserted, the "NO DTM" indication is displayed at the system switch-on.
- All stored waypoints may be identified with an ICAO or an appropriate code.
- The mission data shall be loaded on the DTM on ground only.

The DTM has to be inserted in a proper socket located in the front side of the IN/CDU in the front cockpit only. The socket of rear IN/CDU will be protected by a proper cover. The IN/CDU in the master cockpit allows selection and acceptance of the IP value for INS alignment procedure.
The IN/CDU shall be designed to be capable of being configured as master or slave equipment in both cockpits (front and rear). One of the two IN/CDU shall be configured as slave and the other as master depending on the current selection made by Pilots on the NAV selector located on the control transfer panel.
The slaved IN/CDU shall have all its keyboard disabled and shall simply act as repeater of the information displayed on the master one. The IN/CDU provides a means for:

- listing of all mission data
- manual recall and presentation of all navigation data
- selection of a destination waypoint (fly-to function)
- visual on-top fixing (OTF) navigation updating
- visual mark point data acquisition with OTF technique

The IN/CDU is electrically powered by the INS.

IN/CDU CONTROLS AND INDICATORS

The IN/CDU control panel labeled IN (refer to Figure 1-55) located on the left lower part of the main instrument panel in both cockpits, has the following controls and indicators.

Hard Keys. Five hard keys provide, when pressed, the following selections:

NAV	The navigation format (also shown as default) is displayed
LIST	Stored waypoints data are displayed
PAGE	Navigation and station point data are displayed in subsequent pages
FIX/ACC	When pressed the first time enables the system for on-top fixing activation. When pressed a second time enables the fix error to be accepted in the system
MARK/IP	When pressed during the GC INS alignment procedure enables the system to accept the IP data. When pressed during flight enables the mark point data acquisition function

Brightness Control Switch. It permits to control the brightness of the IN/CDU alphanumerical display.

NOTE

> The IN/CDU lighting control is carried out by means of the INTERIOR INSTRUMENT switch, located on the right console.

Rotary Switch. A rotary switch permits when operating the selection of any stored waypoint to be displayed on the alphanumerical display

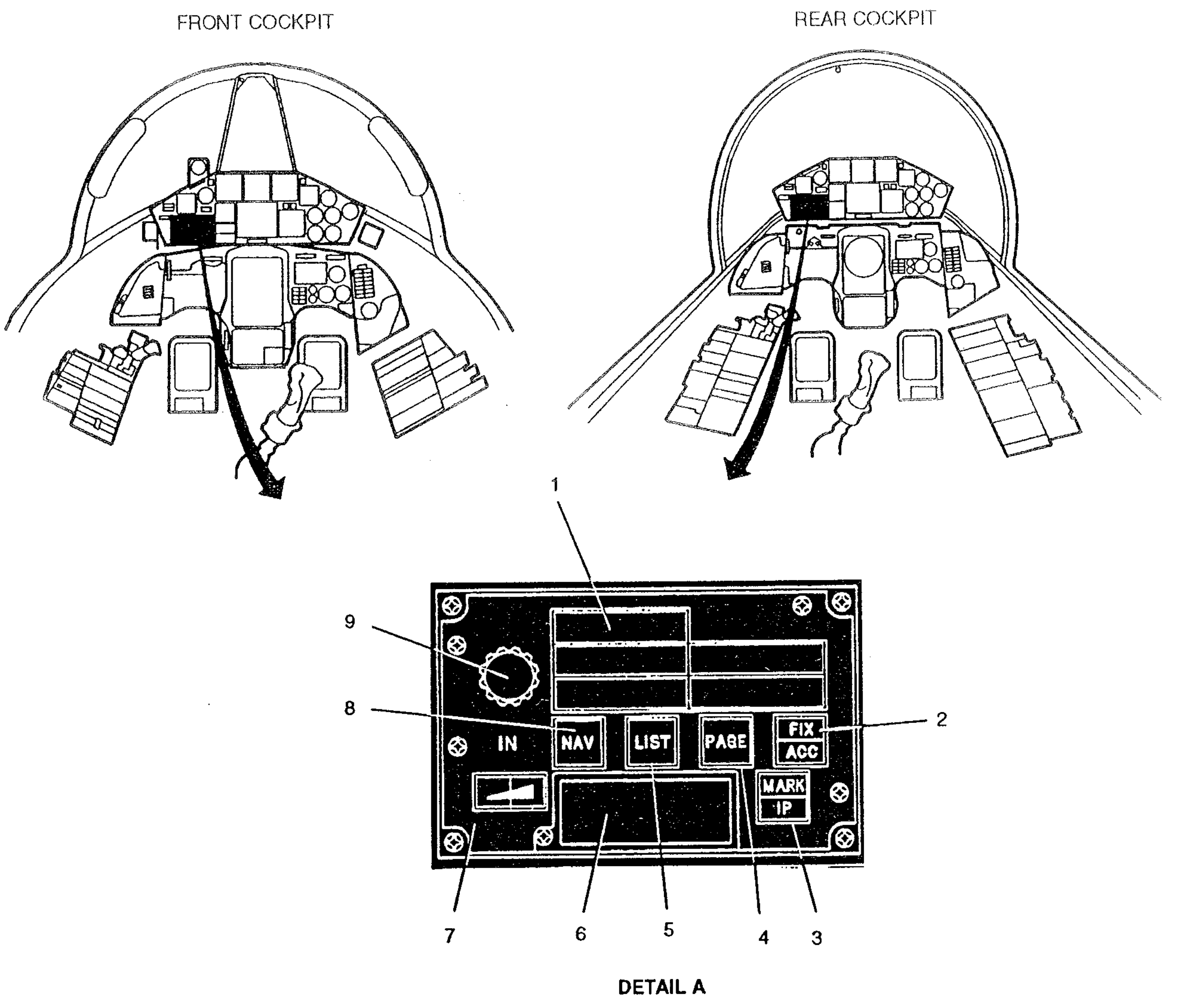
Socket. A socket, located in the lower part of the IN/CDU, permits the DTM insertion/extraction

Alphanumerical Display. It consists of three rows in which all stored waypoint data are shown. On the first row up to eight data may be shown while on the other two up to sixteen data may be shown.

IN/CDU SYSTEM OPERATION

After power on, a five seconds automatic test is carried out. A successfully test is indicated by the "CDU OK" indication while an unsuccessful test is indicated by the "CDU FAIL" indication on both IN/CDU.
After the successful test completion the mission data base stored in the DTM is transmitted, via RS422, to the INS. Then these data are validated by the

IN/CDU CONTROLS AND INDICATORS



A – IN /CDU
1 ALPHANUMERICAL DISPLAY
2 FIX/ACC HARD KEY
3 MARK/IP HARD KEY
4 PAGE HARD KEY
5 LIST HARD KEY
6 SOCKET
7 BRIGHTNESS CONTROL SWITCH
8 NAV HARD KEY
9 ROTARY SWITCH

FA0144

Figure 1-55

INS itself and should a failure be detected the "DTM FAIL" caption appears on both IN/CDU.

IN/CDU MODE OF OPERATION

The IN/CDU may be used on ground for initial position selection and acceptance (for INS/GC alignment mode only) and during flight (or on the ground with the INS set to NAV) for the following:

- IN steering mode data presentation
- station point display
- fly-to function
- on-top fixing updating
- mark points data acquisition

IN/CDU Mode of Operation during Alignment Procedures

With the INS set to ALN/GC, it is possible to select the IP data (for INS/GC alignment purpose) different from the last stored INS/PP provided that the DTM (in which the station point are loaded) has been inserted.
This operation may be made by both IN/CDU (front or rear) acquiring the control of cockpit by the pilot on the NAV selector located on the control transfer panels.

Being the last stored INS/PP shown as default, the following format is displayed:

```
IP  STO
IP   N 3 6°34.50
STO  E02 3°18.21
```

If the DTM has been inserted, only the MARK/IP and rotary switch (for IP selection and station point presentation) are operative on the master cockpit.

If the DTM is not inserted none of the IN/CDU controls are available and on the alphanumerical display the following format is shown:

```
NO  DTM
IP   N 3 6°34.50
STO  E02 3°18.21
```

By rotating the rotary switch on the master IN/CDU the pilot may select the station point on which the INS will be aligned that, in other terms, shall be the initial position.

NOTE

> The INS continues to align using the stored INS/PP until the MARK/IP hard key actuation.

The IN/CDUs alphanumerical display shows the following:

```
IP  STO
12   N 3 7°54.59
ABCD E02 9°18.73
```

After the IP data are shown, the MARK/IP hard key shall be pressed: the INS will start the alignment procedure using the selected data and the IN/CDU shows the following:

```
IP  12
12   N 3 7°54.59
ABCD E02 9°18.73
```

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The NAV, PAGE, LIST and FIX/ACC hard keys are disabled in both IN/CDU (front and rear) while, by means of the rotary switch in master IN/CDU, it is possible to show the loaded station points. In the slave IN/CDU also MARK/IP pushbutton and rotary switch are disabled.

Pilot has to check, on the IN control panel located on the right console, the alignment status indicator which is in accordance with the desired GC alignment mode (fast or full).
If a STO INS alignment is carried out the IN/CDU shows the default format: in this condition only the rotary switch in master IN/CDU is operative, which permits the station points presentation. The NAV, LIST, PAGE, FIX/ACC and MARK/IP hard keys are disabled, on both IN/CDUs.

IN/CDU Navigation Moding

Navigation Format

After INS alignment, with the IN FUNCTION selector knob set to NAV, the IN/CDUs may show the navigation information relevant to the stored station points. IN/CDUs data presentation and moding is not function of the selected steering mode. Navigation information is an INS outputs and referred to the selected fly-to waypoint. Navigation data calculations are based on PP and Ground Speed (GS) derived from INS for both IN and TACAN steering mode.

NOTE

> IN/CDU displayed data are consistent with the HSI indications only when the IN steering mode is selected; in TACAN steering mode the HSI shows navigation information derived from the TACAN system itself while the information shown on the IN/CDU are INS outputs.

Refer to Figure FO-14 for the IN/CDU navigation moding block diagram. The navigation format is shown as default (after the IN FUNCTION selector knob set to NAV) and is referred to the page one.

NOTE

> The DWP is the station point "00" or in other terms the point on which the INS alignment was carried out.

The page one of the navigation format may be also shown if, while in station point listing or in fixing error format, the NAV hard key is pressed.
If the FIX/ACC hard key has been pressed while in navigation format, pressing a second time the FIX/ACC or if the 30 seconds period has elapsed without pressing the FIX/ACC, the previously selected navigation page is displayed on the IN/CDUs. If the FIX/ACC hard key has been pressed while in navigation format, pressing the FIX/ACC or if the 30 seconds period has elapsed without pressing the FIX/ACC and following the rotary switch actuation for a fly-to function, the first navigation page (page one) is displayed on the IN/CDU. The page one of the navigation format may be also be displayed when the mark point format is displayed if the NAV hard key is pressed or if the rotary switch is actuated.

NOTE

> If the MARK/IP hard key has been pressed while in navigation or in LIST format, when the 10 seconds period has elapsed without any action on the IN/CDU, the previously selected navigation or list page is displayed on the IN/CDUs.

The "NAV" format is formed by three pages, each of one cyclically selected by pressing the PAGE hard key. Fly-to function may be carried out by acting on the rotary switch and following steering information shown on the HSI and on the IN/CDUs. The fly-to activation causes the first page of the intended fly-to waypoint to be automatically displayed. If the LIST hard key is pressed the first page of the station point listing is shown. In the navigation format both the OTF and mark point data acquisition facilities are available. NAV hard key actuation shall have no effect when in "NAV" format. Refer to Figure 1-56 for a typical navigation format display.

Station Point Listing Format

The station point listing format is shown when, in "NAV" format, the LIST hard key is pressed. As default the first of the two pages is shown.
The station point list format may be also shown when, in fixing error format, the LIST hard key is pressed: the first page is shown. If the FIX/ACC hard key has been pressed while in station point listing format, pressing again the FIX/ACC or if the 30 seconds period has elapsed without pressing the FIX/ACC, the previously selected station point listing page is displayed on the IN/CDU. The station point list format may be also shown when, in mark point data acquisition format, the LIST hard key is pressed: the first page is shown.

IN/CDU NAVIGATION FORMAT DISPLAY (TYPICAL)

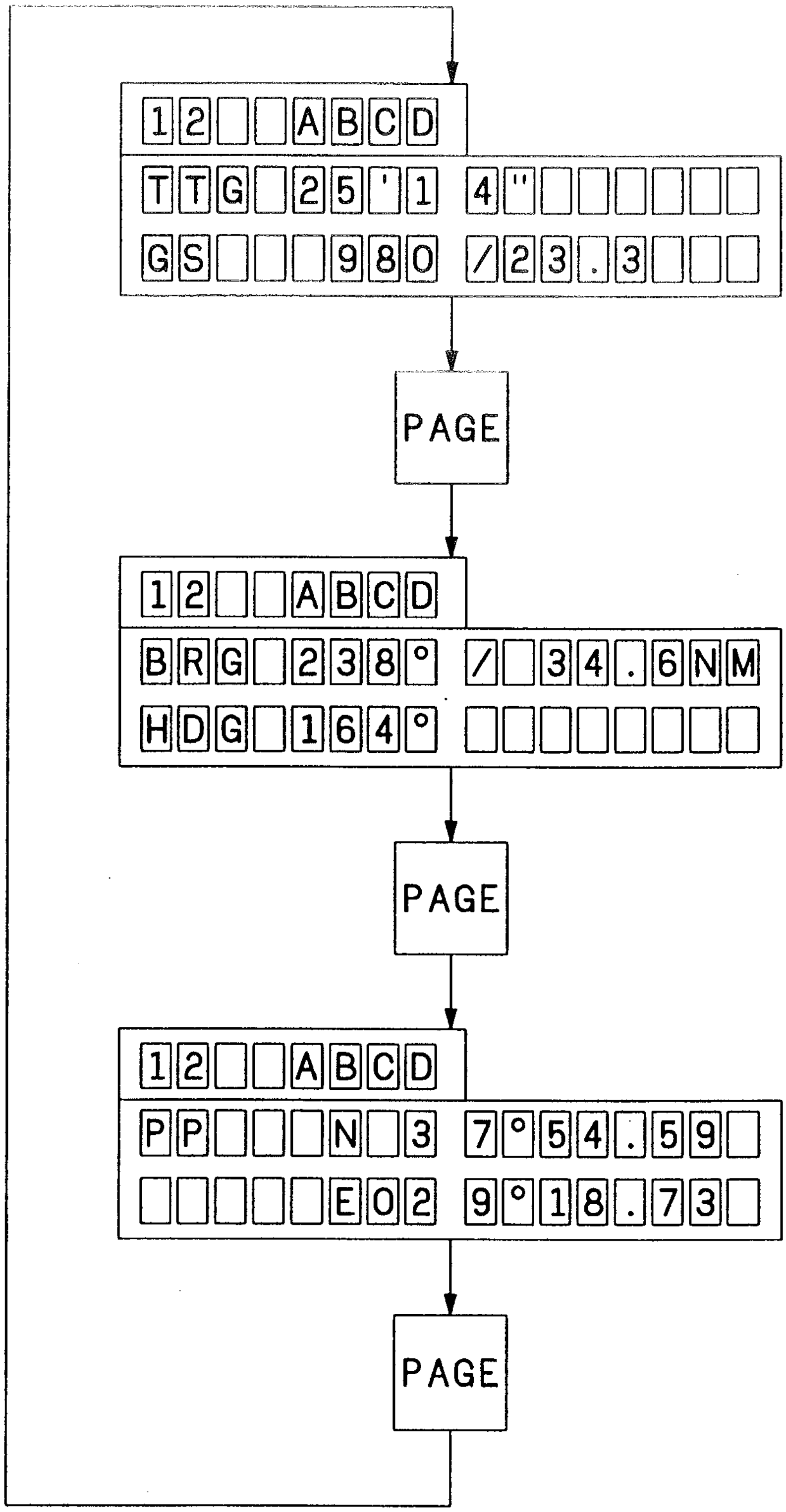

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Figure 1-56

NOTE

If the MARK/IP hard key has been pressed while in station point listing format, when the 10 seconds period has elapsed without any action on the IN/CDU, the the previously selected station point listing page is displayed on the IN/CDUs.

The "LIST" format is formed by two pages, each of one cyclically selected by pressing the PAGE hard key. Rotating the rotary switch the information relevant to the other stored waypoints is shown. The waypoint station identification number with its identification (ICAO) code will be flashing. If the NAV hard key is pressed the first page of the navigation format is shown. In the station point listing format both the OTF and mark point data acquisition facilities are available.
LIST hard key actuation shall have no effect when in "LIST" format. Refer to Figure 1-57 for a typical station point listing format display.

On-Top Fixing (OTF) Format

The on-top fixing format is shown when the FIX/ACC hard key is pressed while in navigation or station point listing or mark point data acquisition format. The OTF format is available in the IN or TACAN steering mode and has to be carried out only for the "selected to" waypoint only.
The OTF format is formed by a single page. Pilot shall be aware of the following:

- if fixing error has to be accepted the FIX/ACC hard key shall be pressed a second time within 30 seconds. After this action the IN/CDUs show the previously selected format
- if no action are carried out within 30 seconds, the IN/CDUs show the previously selected format and no system updating shall be carried out
- if NAV or LIST hard key is pressed the first page (page one) relevant to the selected waypoint of the relevant format shall be displayed and no fixing updating is carried out
- the MARK/IP and PAGE hard keys are disabled
- it is possible to act on the rotary switch to perform the fly-to function: the selected waypoint is shown in the first row of the IN/CDUs. This action does not imply the fixing error which is shown on the second and third rows of the IN/CDUs. After the FIX/ACC actuation or after the 30 seconds have elapsed the first page (page one) of the navigation format is resumed.

During OTF procedure the FIX ERROR readout is displayed on the second row of the alphanumerical display while on the third the fixing error (in the form of bearing and range) is displayed.
Range error up to 99.99 NM may be shown; if the error exceedes this value the "> 100 NM" readout is displayed. Refer to Figure 1-58 for a typical fixing error readout.

Mark Point Data Acquisition Format

The mark point data acquisition format is shown when the MARK/IP hard key is pressed while in navigation or station point listing format. The mark point data acquisition format is available for all steering modes. The mark point data acquisition format is formed by a single page. Up to two mark points may be acquired: further acquisition delete the previously stored mark point. After 10 seconds from MARK/IP hard key actuation the previously format shown is resumed. Pilot shall be aware of the following:

- if NAV hard key is pressed within 10 seconds the first page (page one) of the navigation format is shown
- if the rotary switch is operated within 10 seconds the fly-to function is activated: after 10 seconds from MARK/IP actuation the first page (page one) of the navigation format is shown
- if LIST hard key is pressed within 10 seconds the first page (page one) of the station point listing is shown
- if FIX/ACC hard key is pressed the fixing error format is shown
- if the MARK/IP hard key is pressed a new mark point acquisition selection is carried out
- the PAGE hard key is disabled

The "MARK" readout is shown on the second row while the station identification number (59 or 60) is shown on the third row of the alphanumerical display. Refer to Figure 1-59 for a typical mark point data acquisition readout.

INICDU STATION POINT LISTING FORMAT DISPLAY (TYPICAL)

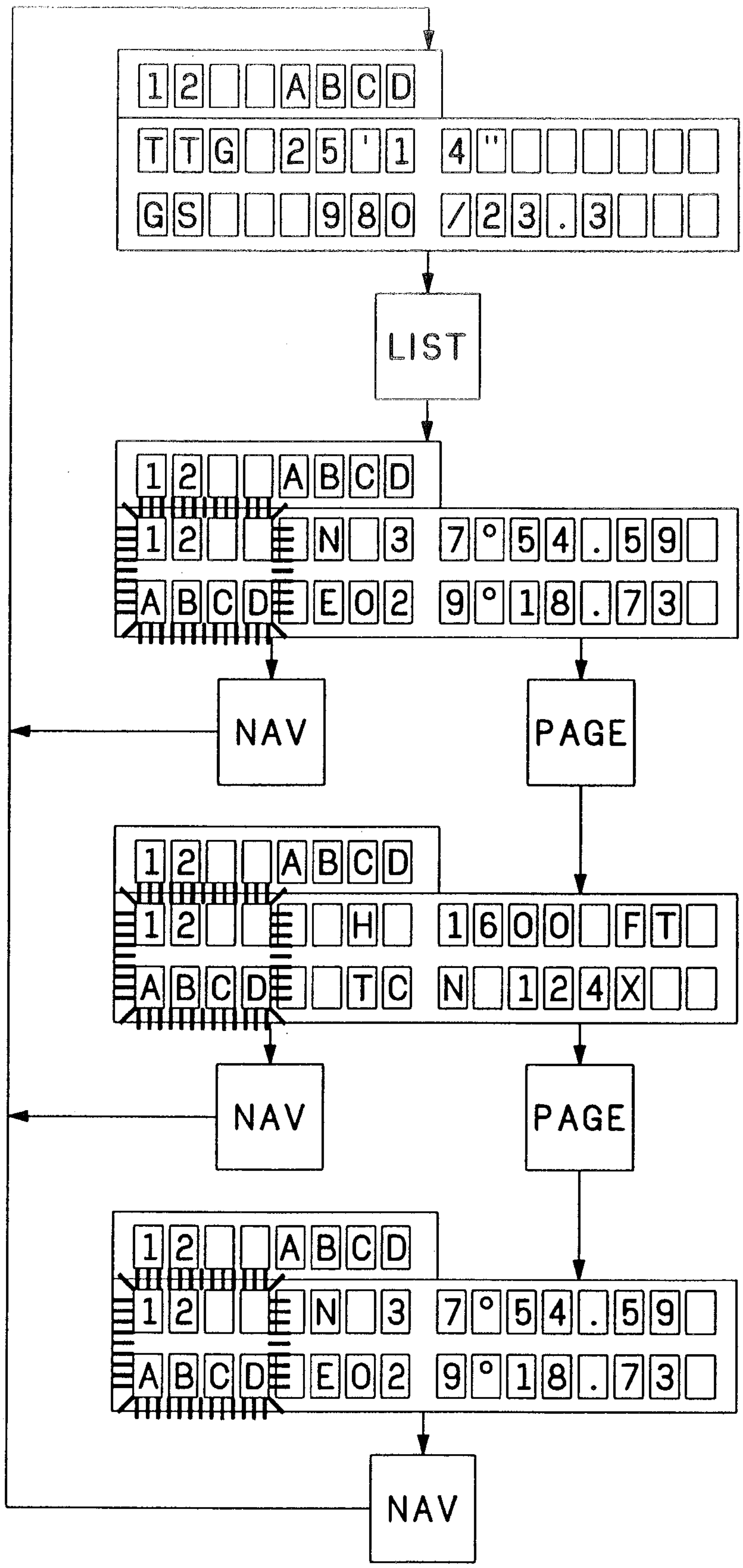

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Figure 1-57

IN/CDU ON-TOP FIXING FORMAT DISPLAY (TYPICAL)

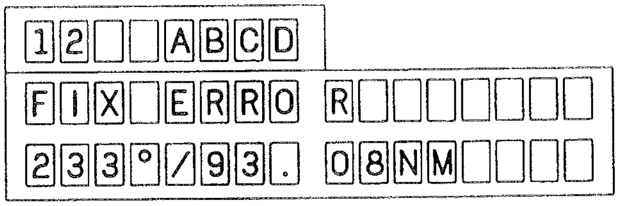


Figure 1-58

IN/CDU MARK POINT DATA ACQUISITION FORMAT DISPLAY (TYPICAL)

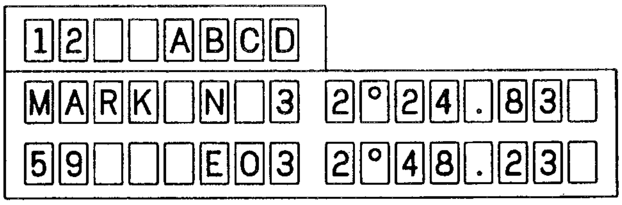


Figure 1-59

HORIZONTAL SITUATION INDICATOR (HSI)

The Horizontal Situation Indicator located on the main instrument panel on both cockpits, provide a horizontal view of the aircraft with respect to the navigation situation. The HSI operate according to the following steering modes of operation:

- IN
- TACAN

These modes shall be selected by pressing the relevant navigation steering mode selector pushbutton. The aircraft symbol in the center of the HSI represents the aircraft superimposed on a compass card which rotates so that the aircraft heading is indicated on the upper fixed lubber line. The system is powered by the XP7 bus.
Loss of power to the HSI will cause the OFF failure flag to appear on the left side of instrument. When IN mode is selected, the HSI shall show steering indications toward the destination waypoint currently selected by the pilot on the IN/CDU.
The INS will not compute course error. In these modes, the TO/FROM indication shall be derived by INS (and supplied to the HSI) as a TOWARD/OUTWARD indication (TOWARD: range decreasing condition; OUTWARD: range increasing condition).

CAUTION

COURSE ERROR IS NOT COMPUTED AND DISPLAYED WHEN FLYING IN "IN" STEERING MODE.

The desired course indication (and therefore the axis of the TO/FROM arrows) will be slaved to the course knob and will rotate together with the compass card.
When TACAN mode is selected, the HSI shall show steering indications toward the radio station currently selected on the TACAN control panel.
Magnetic heading and actual track shall be supplied by INS. When TACAN mode is selected and INS is in "NO GO" status, the actual track indication shall not be available, and will be hidden below the

bearing indication supplied by TACAN, while magnetic heading shall be provided by C2-G.
The TACAN, receiving the desired course information from master HSI, shall use it to compute and supply to the HSIs the TO/FROM and course error parameters in TACAN steering mode. The desired course indication will be slaved to the course knob.

HORIZONTAL SITUATION INDICATOR CONTROLS AND INDICATORS

The horizontal situation indicator displays and controls are illustrated in Figure 1-60. Refer also to Figure 1-61 which shows the HSI display indications according to each navigation steering mode.

NAVIGATION SUB-SYSTEM

The navigation sub-system is designed to provide navigation parameters computation and displaying, by means of the following equipment:

- Inertial Navigation System (INS)
- TACAN
- C-2G Directional Gyro System
- Inertial Navigation Control Display Unit (IN/CDUs)
- Horizontal Situation Indicator (HSIs)
- Attitude Indicator (AIs)
- Adapter
- Rate Gyro (RGs)

The INS is the primary self-contained source of position, velocity, attitude and magnetic heading data. It also provides central processing functions and interface capabilities for HSIs (via switching box), AIs and ADC. A dedicated warning light (INERTIAL NAV FAULT), located on the warning lights panel, illuminates when an INS failure is detected. Via two RS-422 lines, the INS is interfaced with the Control Display Unit (IN/CDU) located on the main instrument panel on both cockpits.
The TACAN is a receiver/transmitter radio equipment whose function is to provide aircraft slant range and bearing information with respect to a ground station or to another airborne TACAN equipment.

The C-2G is a gyrostabilized magnetic compass to be used as reversionary magnetic heading sensor to the HSI following a INS failure with TACAN steering mode selected.
The AIs provide head down attitude information as well as turn rate indication generated by a rate gyros. The gyro transmitter provides electrical signals directly proportional to the aircraft yaw rate. The parameter, in terms of magnitude and polarity, is displayed on the attitude indicator.
The HSI displays head down navigation/steering information when flying either in IN or TACAN steering mode.

NOTE

- Magnetic heading datum is always provided by the INS regardless the selected steering mode.
- Following an INS failure, the C-2G shall supply the magnetic heading datum when TACAN steering mode is selected.

NAVIGATION SYSTEM FUNCTIONS

Two navigation steering modes are available:

- IN mode
- TACAN mode

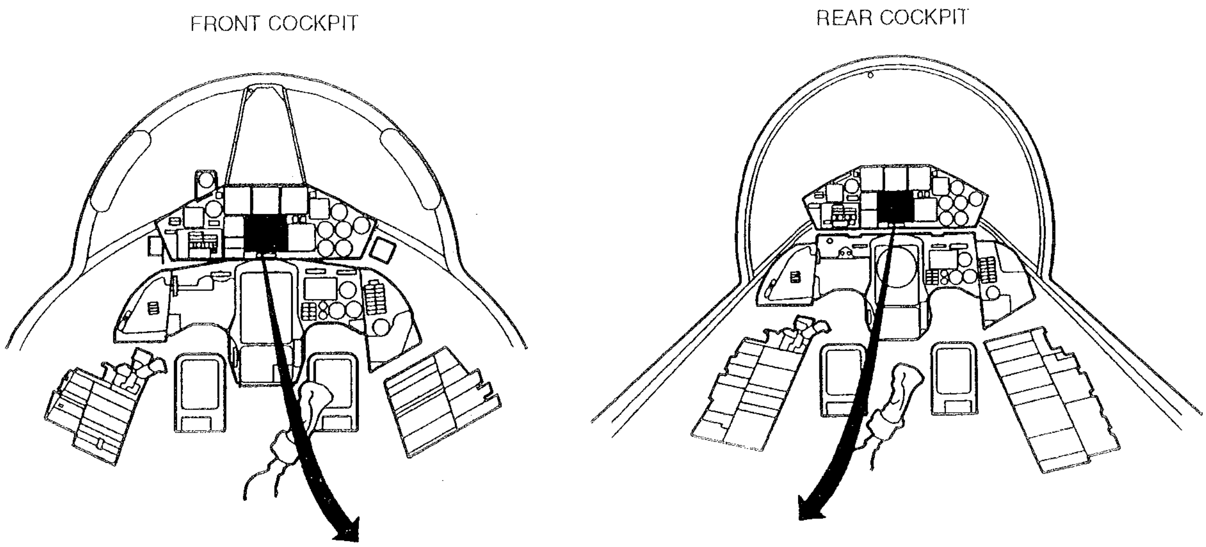
These steering mode are manually selectable by means of two dedicated navigation steering mode selector pushbuttons located on the main instrument panel in both cockpits.

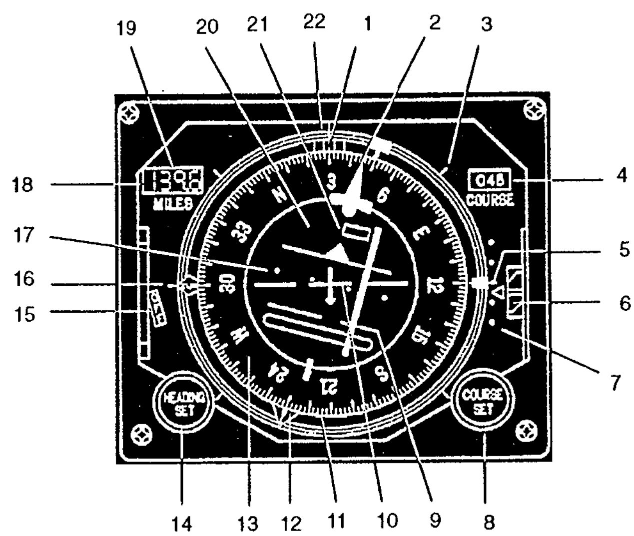
NOTE

The navigation steering mode selector pushbuttons lighting control is carried out by means of the INTERIOR INSTRUMENTS switch located on the right console. When the INTERIOR INSTRUMENTS switch is set to OFF, the navigation steering mode selector pushbuttons are lit at the maximum level.

The pilot, after having inserted the DTM in the IN/CDU in the front cockpit only, shall select the INS ALN function on the IN control panel thus allowing the data loading of the Station Points from

HORIZONTAL SITUATION INDICATOR (HSI)





1	UPPER FIXED LUBBER LINE	12	POINTER No. 1
2	COURSE POINTER	13	COMPASS CARD
3	REFERENCE MARK	14	HEADING SET KNOB
4	DIGITAL COURSE READOUT	15	DISPLAY FAILURES FLAG
5	GLIDESLOPE DEVIATION SCALE	16	POINTER No. 2
6	GLIDESLOPE FLAG	17	LATERAL DEVIATION SCALE
7	GLIDESLOPE DEVIATION SCALE	18	NUMERICAL RANGE DISPLAY
8	COURSE SET KNOB	19	RANGE FAILURE FLAG (OUT OF VIEW)
9	LATERAL DEVIATION BAR	20	TO/FROM ARROW
10	AIRCRAFT SYMBOL	21	NAV FLAG
11	LOWER FIXED LUBBER LINE	22	HEADING INDEX

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Figure 1-60

HSI DISPLAY INDICATION

INDICATION	IN STEERING MODE	TACAN STEERING MODE
Course Pointer	Present but inoperative	Selected TACAN radial
Course Display	Present but inoperative	Selected TACAN radial
Compass Card and Lubber Line	Magnetic Heading	Magnetic Heading
Pentagonal Pointer (Pointer No. 2)	DWP Bearing	TACAN Station Bearing (1)
Triangular Pointer (Pointer No. 1)	Actual Track	Actual Track (2)
Distance Indicator	Horizontal Range to selected DWP	Slant range to selected TACAN station selected DWP
Heading Index	Present but inoperative	Present but inoperative
TO/FROM Arrow	Toward/Outward	TO/FROM
Glideslope Deviation Bar	Inoperative and centered	Inoperative and centered
Glideslope Flag	Inoperative	Inoperative
Course Deviation	Inoperative and centered	Track angular error (5° per dot)
Power OFF Flag	– HSI failure – INS Heading failure	– HSI failure – Heading failure (3)
NAV Flag	– HSI failure – INS steering failure	– HSI failure – TACAN steering failure

(1) During the TACAN searching phase the pointer rotates clockwise until the TACAN station has been tracked

(2) In case of IN fail the actual track indication is hidden by the relative bearing indication provided by the TACAN

(3) Following an INS failure, the magnetic heading is derived from the C-2G. In this condition the OFF flag is not shown. Following a C-2G failure the OFF flag will be shown

Figure 1-61

the DTM to the INS memory. One navigation steering mode selector pushbutton selects the IN steering mode while the other one selects the TACAN steering mode.

Alignment Phase

The navigation sub-system shall be aligned before takeoff. Two alignment modes are available: "CG" or "STO". The initialization phase starts at the INS switch-on (ALN selection) and lasts until the INS is manually set to NAV. At the INS switch-on, the system downloads the mission data from the DTM, previously inserted on its receptacle located on the IN/CDU. In case of mission data loading error detection, the DTM FAIL caption appears on the IN/CDU readouts. Should the DTM not be inserted, the "NO DTM" caption is displayed.

CAUTION

BOTH FAILS OF DTM DOWNLOADING AND DTM NOT INSERTED CAUSE ABSENCE OF MISSION DATA BASE IN THE SYSTEM WHICH ENTERS IN AN IMPROPER MODE OF OPERATION. THE NAVIGATION SYSTEM SHALL NOT BE ABLE TO PROVIDE STEERING INFORMATION TOWARD THE WAYPOINTS EXCEPT FOR THE WAYPOINT USED FOR INS ALIGNMENT (LAST STORED PRESENT POSITION).
NEVERTHELESS THE INS MAY BE SET TO NAV AND THE "IN" STEERING MODE IS AVAILABLE FOR SELECTION.

The INS begins its alignment procedure using the last stored present position. During initialization, the pilot shall select, using the rotary switch on the IN/CDU, the initialization point (IP). After the IP data are shown on the readouts, the MARK/IP pushbutton shall be pressed and the INS will align on the selected IP. During initialization phase (which depends on the selected alignment mode) no navigation functions are available as well as no IN/CDU pushbutton actuations except the MARK/IP pushbutton (IP function only) and the rotary switch (for IP selection).

During the initialization/navigation INS transition, the data used for initialization (IP) are stored in the first location, identified with "00". During navigation, this waypoint may be used as any other destination waypoint. As soon as the initialization phase is terminated, the INS shall be set to NAV.

NOTE

On ground, after alignment is completed, it will be possible to select the first desired destination WPT being on HSI shown the steering information relevant to the IP.

IN Steering Mode

The IN steering mode is the primary autonomous navigation mode and is based on the INS as the sole navigation source. If an OTF operation has been carried out, the PP will be updated accordingly.
The IN steering mode uses the stored coordinates of the selected destination waypoint in order to calculate steering parameters toward the waypoint. Destination waypoint is selected by using the rotary switch located on the IN/CDU and shown on the alphanumerical display.
Rotating the switch all the programmed and stored waypoint are selectable one by one, and the "current to" waypoint is recognized by the navigation system as the one toward which compute all the steering parameters: no further actions are required for selection and acceptance.

NOTE

- The selected waypoint becomes the new destination waypoint only after 1 second from the stop of the rotary switch.
- Waypoint identifier and label appear immediately on the first readout row while, the other waypoint information (TTG, GS and range), appear on the second and third rows after some instance later.

Steering information are displayed on both the HSI and IN/CDU, related to the same destination waypoint and referred to the magnetic north (refer to Figure 1-62).

IN STEERING MODE — DATA GENERATION

HSI		IN/CDU	
Parameter	Source	Parameter	Source
Magnetic Heading	INS	Magnetic Heading	INS
Horizontal Range to DWP	INS	Horizontal Range to DWP	INS
Bearing to DWP	INS	Bearing to DWP	INS
Actual Track	INS		
Toward/Outward Indication	INS		
		Time to DWP	INS
		Ground Speed	INS
		A/C Present Position	INS

Figure 1-62

TACAN Steering Mode

The TACAN steering mode uses the TACAN equipment as the basic navigation sensor. The TACAN is used in the receive and transmit/receive functions. In the receive function the TACAN provides the avionic system with the bearing to a selected radio station.
In the transmit/receiver function the TACAN provides the avionic system with slant range and bearing information relative to the selected radio station. Range and bearing information is used to generate steering data as relative bearing, course error, to/from and slant range to destination.
By operating the TACAN, the pilot shall select the desired TACAN/DME channel; successively the selection of the TACAN channel/mode shall cause the displaying of the corresponding navigation data on the HSI only. In case of selection of a TACAN channel, the pilot shall be able to select the Desired Course value towards the station by acting on the HSI Course Knob. In case of DME channel selection the Slant Range only shall be displayed.

NOTE

Pilot shall be aware that if DME channel is selected, being the bearing not available, the HSI NAV flag will be in view; course knob not operative.

Magnetic heading parameter is computed by the INS. The steering indications are displayed on the HSI while on the IN/CDU the steering indications are INS outputs and relevant to the destination waypoint selected on the IN/CDU itself (refer to Figure 1-63).

NOTE

Pilot shall be aware that HSI and IN/CDU steering information is different.

FLIGHT AND STEERING PARAMETERS DISPLAYING

The following equipment display the flight and steering parameters:

- HSIs
- IN/CDUs
- AIs

The HSI shows flight and steering information according to the selected steering mode.
When flying in the IN steering mode, the HSI shows steering information toward the destination waypoint selected on the IN/CDU. When flying in the TACAN steering mode, the HSI shows steering information toward the radio station selected on the TACAN control panel. Magnetic heading and actual track are INS outputs.

TACAN STEERING MODE — DATA GENERATION

HSI		IN/CDU	
Parameter	Source	Parameter	Source
Magnetic Heading	INS (*)	Magnetic Heading	INS
Slant Range to Selected TACAN Station	TACAN	Horizontal Range to DWP (**)	INS
TACAN Station Bearing	TACAN	DWP Bearing (**)	INS
Track Angular Error	TACAN		
Actual Track (***)	INS		
TO/FROM Indication	TACAN		
		Time to DWP (**)	INS
		Ground Speed	INS
		A/C Present Position	INS

(*) Following an INS failure the magnetic heading is derived from the C-2G
(**) Data base mission DWP selected on IN/CDU
(***) Following an INS failure the actual track pointer is hidden by the bearing pointer

Figure 1-63

The attitude indicator shows pitch and bank angles as long as the INS is available. Turn rate will be provided by the dedicated gyro equipment.
The IN/CDU displays various navigation data sets. The data shown are always INS outputs. Steering indications are always referred to the destination waypoint selected on the IN/CDU regardless the selected navigation steering mode.

NAVIGATION SYSTEM FUNCTIONALITY UNDER FAILURE CONDITIONS

Following an INS failure (INERTIAL NAV FAULT warning light lit on the warning lights panel), the steering information is not available on IN/CDUs and HSIs in IN steering mode. Moreover the attitude indications are not displayed on the AIs. TACAN selection permits navigation toward the selected radio station with steering indication shown on the HSI. The magnetic heading is provided by the C-2G.

NOTE

On HSI, the actual track indication is hidden by the relative bearing indication provided by the TACAN.

Following a TACAN failure, IN steering mode of operation may be selected, with steering information shown on the HSI and IN/CDU display.

INFLIGHT OPERATION

The navigation sub-system permits, during flight, the following:

- Fly To function
- On Top Fixing (OTF) procedure
- Mark point data acquisition

Fly To Function

This function permits the pilot to reach, from the aircraft's PP, any point whose position is stored in the system. The navigation system is capable to supply steering parameters to fly the aircraft to that point by means of the HSI. Selection of the fly-to waypoint is carried out on the IN/CDU by using the rotary switch.

On Top Fixing Procedure

The aircraft's PP updating may be performed by the on top fixing (OTF) procedure. This updating shall be carried out by flying over the selected destination waypoint, and is based on the assumption that this point has been visually identified. The point on which the OTF procedure has to be carried out has to be one of those inserted in the mission data base. The OTF facility is available and may be automatically or manually rejected.
The function shall be activated pressing the FIX/ACC hard key on the IN/CDU, whichever mode is currently active on the IN/CDU. The OTF function is activated whenever is desirable to update the current PP (affected by the platform error) with the known destination WPT coordinates selected on the IN/CDU. At the first press of the FIX/ACC hard key, the IN/CDU shall receive the computed fixing error via RS422 interface and shall display it in polar coordinates with:

Angular error:

- Range: 0 – 359 deg
- Resolution: 1 deg

Range error:

- Range: 0 – 99.99 NM
- Resolution: 0.01 NM

NOTE

> In case the computed range error is greater than 99.99 NM, the IN/CDU shall display the range error as "> 100 NM".

If, starting from the moment of the fixing error displaying, 30 seconds elapse without any other pilot action on any pushbutton of the IN/CDU, the fixing error format shall disappear and the IN/CDU shall revert to the format and the page displayed before the OTF activation.
At the end of flight, with the aircraft in steady state condition, the IN final error check may be performed using the OTF technique.
The pilot selects (if not already done), on the IN/CDU, the relevant waypoint which shall be coincident with the OTF point. After having pressed the FIX/ACC pushbutton, the fixing error is displayed (refer also to Figure 1-58) for a typical fixing error readout). Pilot shall be aware of the following: the destination waypoint identified as "00" shall not be used for this purpose because on ground, with a ground speed < 10 knots, the relevant fixing data are not reliable. This does not affect the utilization of waypoint "00" (as any other waypoint) on ground and/or in flight provided that the ground speed is > 10 knots.

MARK POINT DATA ACQUISITION

The mark point data acquisition provides a means to acquire the position (in terms of latitude and longitude) of an unknown point. This procedure shall be carried out by flying over the interested point and is based on the assumption that this point has been visually identified.
Up two mark points may be acquired within the system and they shall be identified by the identification numbers 59 and 60. The position of the point is stored in the navigation system. After its/their acquisition, they may be selectable on the IN/CDU as any other waypoint by using the rotary switch. After INS switch off the mark points data are lost.
Pressing the first time the MARK/IP hard key the IN/CDU shall command to the INS, via RS422, to store the actual present position of the aircraft in the first memory location. Pressing the second time the MARK/IP hard key the same operation is done for the second point on the second location.
Subsequent presses of the MARK/IP hard key shall cyclically memorize the PP of the aircraft in the first/second location, thus deleting the previously acquired coordinates.

RADAR SYSTEM

The system comprises the radar nose package (containing the antenna), the radar indicator, the clearance plane and antenna tilt indicators, and the radar control panel.

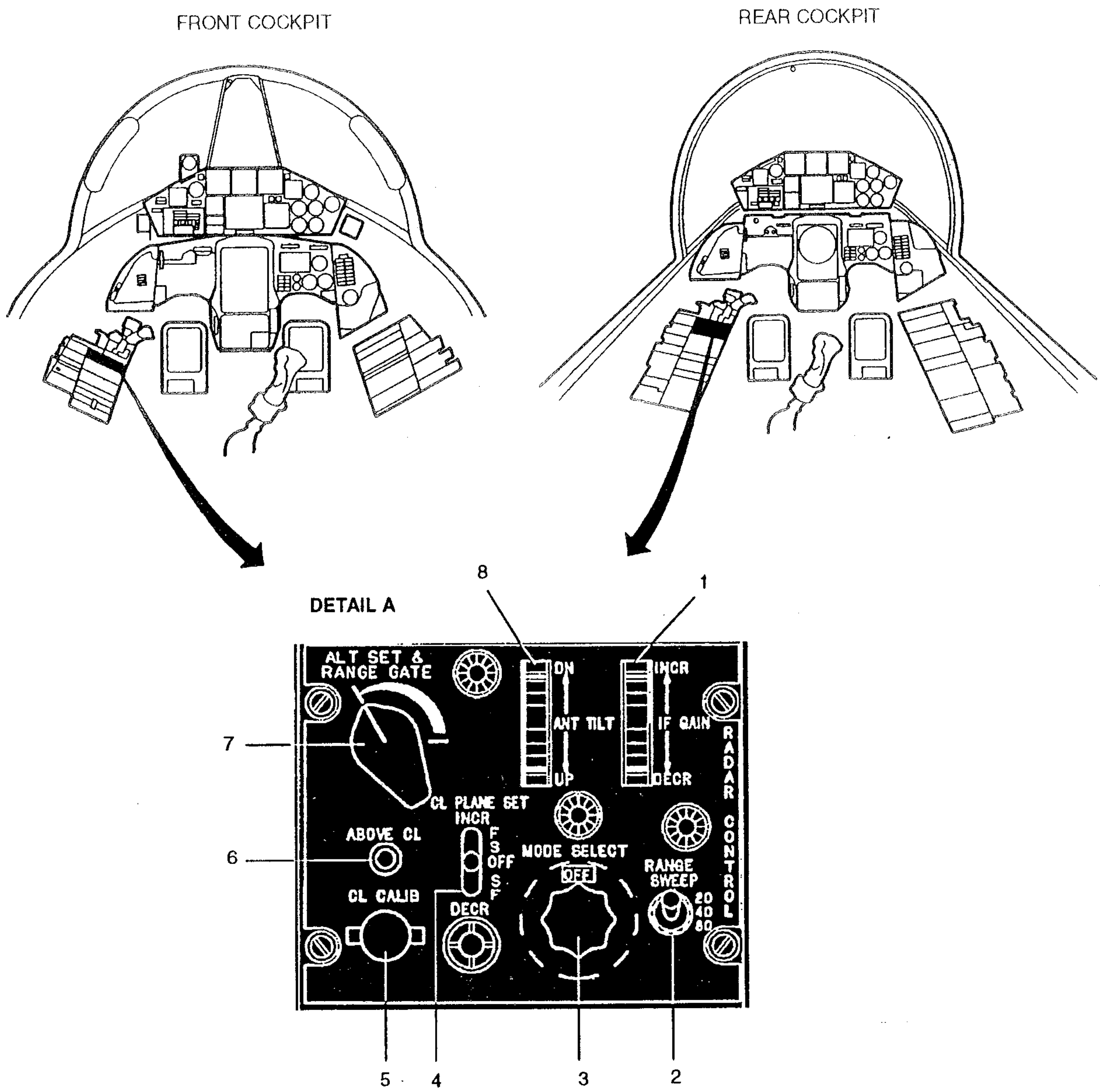
Radar Control Panel

The radar control panel (Figure 1-64) is located on the left console in both cockpits, aft of the throttle quadrant. The panel has the following controls and indicators.

ALT SET & RANGE GATE Knob

The altitude set and range gate knob (Figure 1-64) has three positions in the air-to-air mode: a detent

RADAR CONTROL PANEL



A – RADAR CONTROL PANEL
1 IF GAIN THUMBWHEEL
2 RANGE SWEEP SWITCH
3 MODE SELECT SWITCH
4 CLEARANCE SET PLANE SWITCH
5 CLEARANCE CALIBRATE KNOB
6 ABOVE CLEARANCE BUTTON
7 ALTITUDE SET AND RANGE GATE
8 ANTENNA TILT THUMBWHEEL

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Figure 1-64

for visual acquisition, out-of-detent clockwise for controlling range gate position, and out-of-detent counterclockwise (against a spring) for resume search. In the GMP and GMS modes, manual positioning of the knob clockwise provides an altitude delay voltage proportional to height of the aircraft above mean sea level. (Out-of-detent counterclockwise permits measurement of aircraft altitude above terrain). In the A/A mode, lockon is broken if rotated to the full out-of-detent counterclockwise position.

ABOVE CL Button

The momentary type ABOVE CL (above-clearance) pushbuttons (Figure 1-64) is held down to mechanize the clearance plane above the aircraft in the CM and TA modes. When released, the clearance plane returns to below the aircraft.

CAUTION

IN BOTH CM AND TA MODES, THE ACCURACY OF THE CLEARANCE PLANE ABOVE THE AIRCRAFT IS DEGRADED; CONSEQUENTLY THE ABOVE CLEARANCE PLANE SHOULD BE CONSIDERED INACCURATE AND ONLY USED DURING AN EMERGENCY.

CL CALIB Knob

The CL CALIB (clearance calibrate) knob (Figure 1-64) is used to align the antenna to calibrate the clearance plane.

CL SET PLANE Switch

The CL SET PLANE (clearance set plane) switch (Figure 1-64) has five positions, (INCR) F, S, OFF, (DECR) S and F (F for fast and S for slow). It is spring loaded to the OFF position and used to set the clearance plane to a desired clearance in the CM and TA modes.

ANT TILT Thumbwheel

The ANT TILT (antenna tilt) thumbwheel (Figure 1-64) positions the antenna scan pattern in elevation during ground map and air-to-air search. Rotating thumbwheel aft raises the antenna scan. Rotating the thumbwheel forward lowers the antenna scan.

IF GAIN Thumbwheel

The IF gain thumbwheel (Figure 1-64) adjusts receiver gain during ground map (GMP and GMS) and air-to-air (A/A) operations. Rotating the thumbwheel forward increases receiver gain. Rotating the wheel aft decreases the receiver gain.

MODE SELECT Switch

The mode selector switch (Figure 1-64) is an eight-position rotary switch used for initial power application to the radar system as well as the selection of the modes of operation. On modified aircraft to rotate the switch counterclockwise from SBY to OFF it is necessary to first lift the switch. The positions of the switch clockwise and functions are as follows:

Position	Function
OFF	Deenergizes the radar
SBY	Provides power to the set for warmup
A/A	Selects air-to-air mode
GMP	Selects ground map, pencil beam mode
GMS	Selects ground map, spoiled-beam mode
CM	Selects contour map mode
TA	Select terrain-avoidance mode
A/G	Not used

Mode Indicator Lights

The mode indicator lights located to the right of the radar indicator, are placarded GM SPOIL, GM PENCIL, AIR-TO-AIR STANDBY, CONTOUR MAP, TERRAIN AVOID, and AIR TO GRD. Each light, when lighted, indicates the radar system is operating in the corresponding mode.

RANGE SWEEP Switch

The range sweep switch (Figure 1-64) has three positions, marked 20, 40 and 80 nautical miles. This switch is used to select the desired indicator range display. The 80-nautical-mile range is available only in the ground map modes.

Radar Indicator

The radar indicator (Figure 1-65) is located in the center of the lower instrument panel. The indicator contains the following controls and indicators.

Radar Screen

The radar screen (Figure 1-65) displays radar information to the pilot. No radar information is displayed in the standby or air-to-ground modes.

ERASE INTENSITY Dial

The erase intensity dial (Figure 1-65) increases the intensity of the indicator presentations when rotated clockwise. When rotated fully counterclockwise to the ERASE position, it erases the indicator display.

CAUTION

TO AVOID DAMAGE TO THE SCOPE, THE VIDEO PEDESTAL DIAL SHOULD BE TURNED FULLY COUNTERCLOCKWISE BEFORE ERASING THE DISPLAY.

CURSOR Dial

The cursor dial (Figure 1-65) is concentric within the ERASE INTENSITY dial. It controls the brilliance of the horizon line and the azimuth cursor. Rotating the dial clockwise increases brilliance. Rotating cursor counterclockwise will first cause the azimuth cursor to disappear (when displayed) and then the horizon lines.

XMTR TUN Switch

The transmitter tune switch (Figure 1-65) is a three-position switch used to increase or decrease transmitter frequency. The positions are INCR, OFF, and DECR. The switch is spring loaded to the OFF position.

Day and Night Visors

Day and night visors provided for use with the radar scope are readily attachable to the indicator face. The day visor aids in preventing direct sunlight from striking the display, minimizes reflections, and is rotatable for optimum positioning with respect to the sun. The night visor incorporates a red polaroid filter to reduce glame emanating from the display. A knob on the visor allows the pilot to adjust the filter-level to the desired brightness.

LT DIM Dial

The light dimming dial (Figure 1-65) controls the brilliance of the bezel lighting. Rotating the dial clockwise increases brilliance.

Range Lights

The range lights (Figure 1-65) indicate in nautical miles which range (20, 40, or 80) is presented on the display.

HORIZ CTR Dial

The horizon line center dial (Figure 1-65) positions the artificial horizon line for pitch indications. Rotating the dial clockwise elevates the artificial horizon. The manual setting supplements the movement caused by the vertical reference.

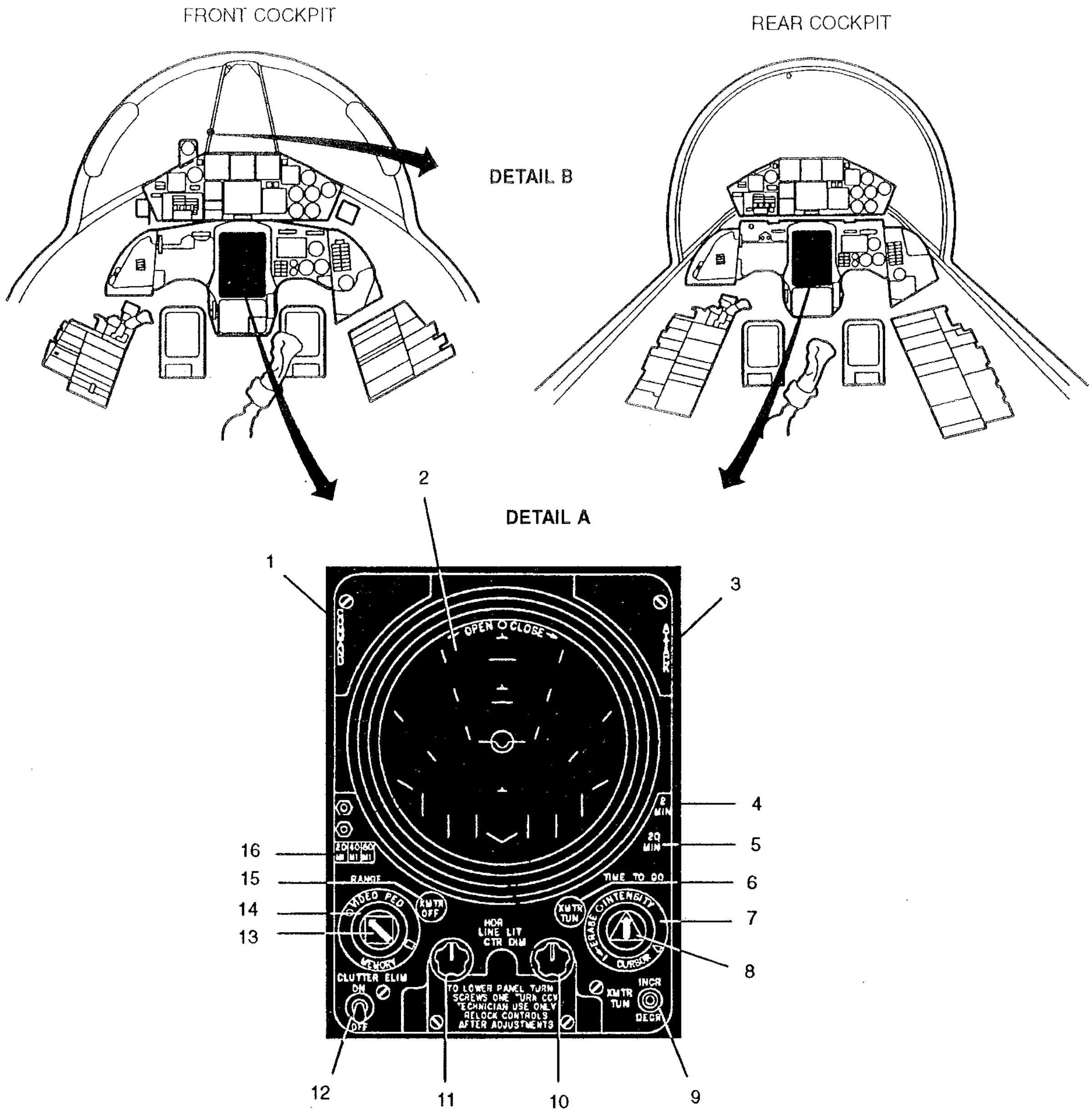
CLUTTER ELIM Switch

The clutter elimination switch (Figure 1-65) reduces ground clutter on the radar screen when placed in the ON position. It functions in the air-to-air search mode only.

MEMORY Dial

The memory dial (Figure 1-65) is concentric within the VIDEO PED dial. It controls persistence of the indicator display. Rotating the dial clockwise increases display persistence.

RADAR INDICATOR



A –	**RADAR INDICATOR**	10	LIGHT DIM DIAL
1	COMMAND LIGHT	11	HORIZON LINE CENTER DIAL
2	RADAR SCREEN	12	CLUTTER ELIMINATION SWITCH
3	ATTACK LIGHT	13	MEMORY DIAL
4	TIME–TO–GO 2 – MINUTE LIGHT	14	VIDEO PEDESTIAL DIAL
5	TIME–TO–GO 20 – MINUTE LIGHT	15	TRANSMITTER OFF LIGHT
6	TRANSMITTER TUNE LIGHT	16	RANGE LIGHTS (20–40–80)
7	ERASE INTENSITY DIAL		
8	CURSOR DIAL	**B –**	**RADAR LOCK–ON LIGHT**
9	TRANSMITTER TUNE SWITCH		

FA0154

Figure 1-65

VIDEO PED Dial

The video pedestal dial (Figure 1-65) controls the amount of signal threshold plus background noise of the range sweep and targets on the indicator. Rotating the dial clockwise increases signal threshold illumination.

XMTR TUN Light

The transmitter tune light (Figure 1-65) illuminates when transmitter frequency is being changed and extinguishes when the desired operating point is reached and the pilot releases the tune switch. It will also extinguish when the limit of tuning range in one direction or the other is reached.

XMTR OFF Light

The transmitter off light (Figure 1-65) illuminates when the system is not transmitting because of a malfunction, when in test standby, and when the transmitter and waveguide are not pressurized.

CAUTION

IF THE TRANSMITTER LIGHT BEGINS TO FLASH, TURN THE SET TO STANDBY. RECYCLE THE SET TO A WORKING MODE AND IF THE LIGHT CONTINUES TO FLASH TURN THE SET OFF OR STANDBY AND HAVE THE TROUBLE INVESTIGATED.

Radar Indicator Bezel

The radar indicator bezel (Figure 1-66) is mounted over the face of the radar indicator screen. Reference marks inscribed on the bezel aid in interpreting the radar displays. Figure 1-66 illustrates the location of the following bezel markings.

PPI Sweep Azimuth Position

References the azimuth position of the PPI sweep, and therefore azimuth position of the antenna, at ±45°.

PPI Sweep Starting Point (Apex)

Indicates the point at which the PPI sweep should start. Also provides a reference for adjusting the ALT SET knob during ground map mode and RANGE GATE knob during air-to-air mode.

PPI Sweep ± 30° Azimuth

References ± 30° azimuth for position of the PPI sweep.

PPI Seep ± 15° Azimuth

References ± 15° azimuth for position of the PPI sweep.

Range Sweep Ranges

Indicates ranges represented by the range sweep as follows:

a. In a 20-nautical mile range, the indexes represent 5, 15, and 20 nautical miles.

b. In a 40-nautical mile range, the indexes represent 10, 30, and 40 nautical miles.

c. In an 80-nautical mile range, the indexes represent 20, 60 and 80 nautical miles.

The center of the bezel represents the second increment of range in all the ranges (10, 20, and 40 nautical miles in 20, 40, and 80 nautical mile-ranges, respectively).

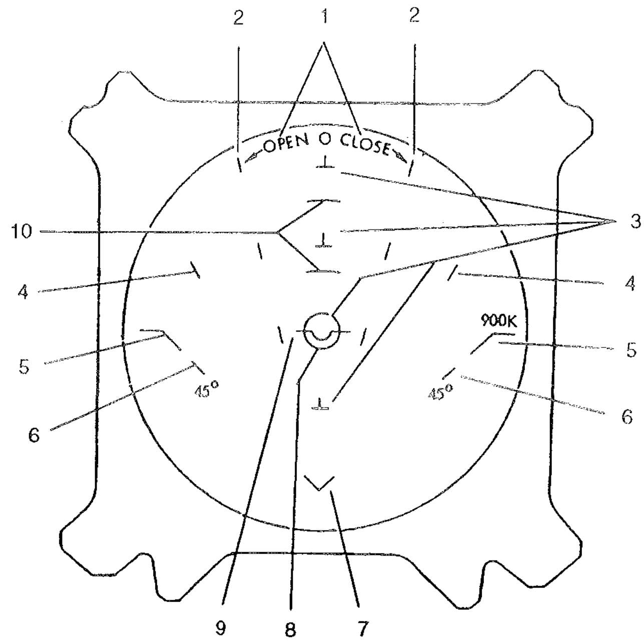
Miniature Aircraft

Represents the wings of the aircraft to show the pilot which direction to roll, using the artificial horizon as an indicator.

Radar Lockon Indicator Light

The radar lockon indicator light located in the front cockpit only, illuminates when the radar has locked on a target.

RADAR INDICATOR BEZEL
(TYPICAL)



1 RANGE RATE GAP
2 PP1 SWEEP ± 15° AZIMUTH
3 RANGE SWEEP RANGES
4 PP1 SWEEP ± 30° AZIMUTH
5 RANGE RATE GAP REFERENCE POINT
6 PP1 SWEEP AZIMUTH POSITION
7 PP1 SWEEP STARTING POINT
8 STEERING REFERENCE CIRCLE
9 MINIATURE AIRPLANE
10 TIME REFERENCE

FA0163

Figure 1-66

Clearance Plane and Antenna Tilt Indicator Panel

The clearance plane and antenna tilt indicator panel (Figure 1-67) is located on the lower right portion of the instrument panel.
It indicates clearance in feet between the clearance plane and the aircraft during CM and TA modes and the elevation tilt angle of the antenna in all modes of operation.
The panel contains the following indicators and light.

Counter

Indicates, in feet, the amount of clearance selected by the pilot in the CM and TA modes.

ABOVE Light

Illuminates when an above clearance plane has been selected.

BELOW Light

Illuminates when a below clearance plane has been selected.

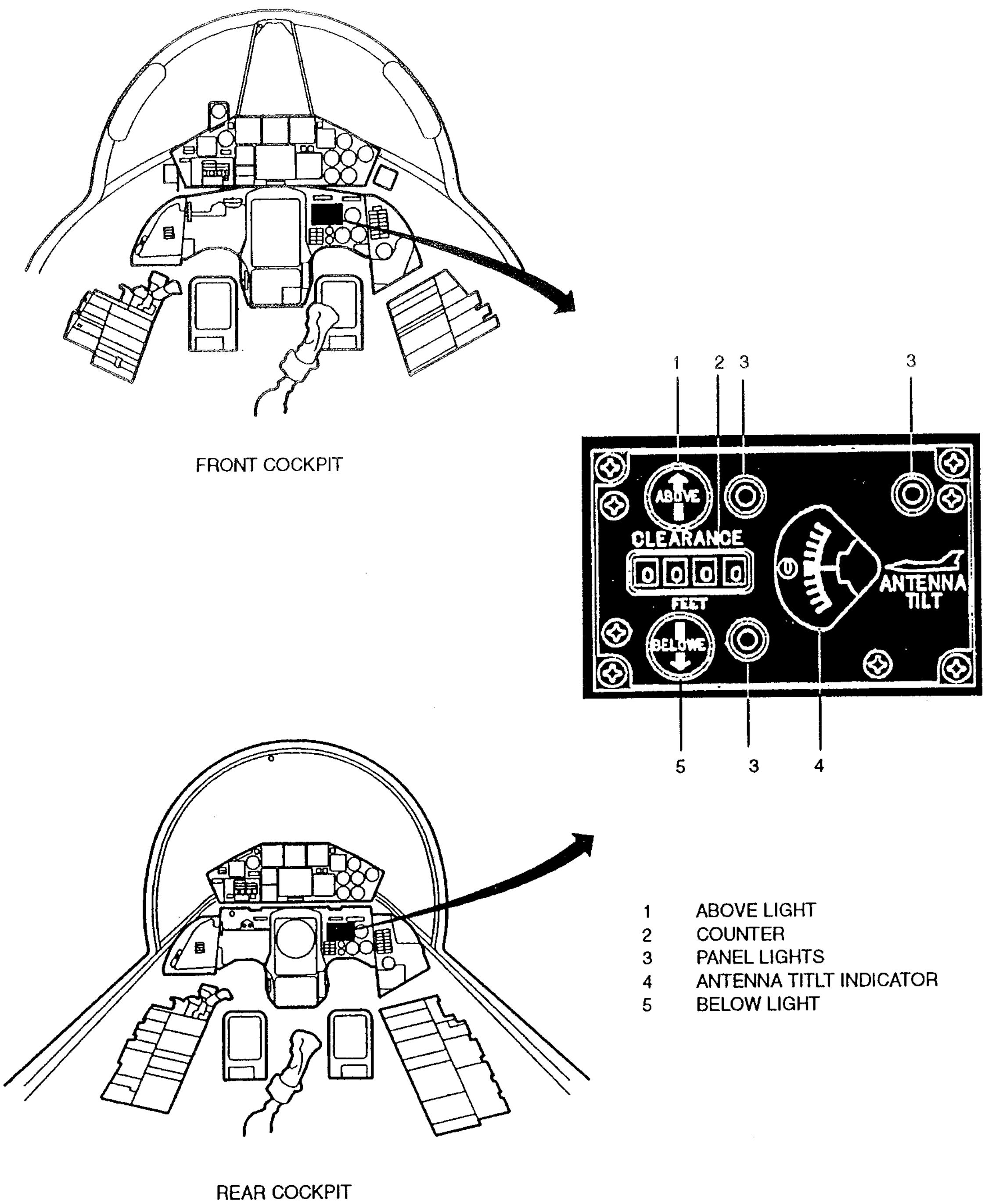
Antenna-Tilt Indicator

Indicates antenna elevation tilt angle in degrees.

MANUAL ANTENNA TILT PROCEDURE

In the normal air-to-air search mode, the radar antenna is stabilized parallel to the horizontal. The antenna is stabilized in such a manner that should the aircraft rotate through any pitch angle, the antenna will tilt through an equivalent angle and thus remain aligned to the horizon regardless of the aircraft's pitch attitude with zero manual antenna tilt. The radar beam sweeps in a two-bar scan pattern 90° in azimuth and 10° in elevation.
In the event that radar contact is to be made with a target, it may become necessary to manually tilt the antenna. If the target slant range and altitude differential are known, the tilt angle required (Δ tilt angle) may be obtained from Figure 1-68).

CLEARANCE PLANE AND ANTENNA TILT INDICATOR



FA0164

Figure 1-67

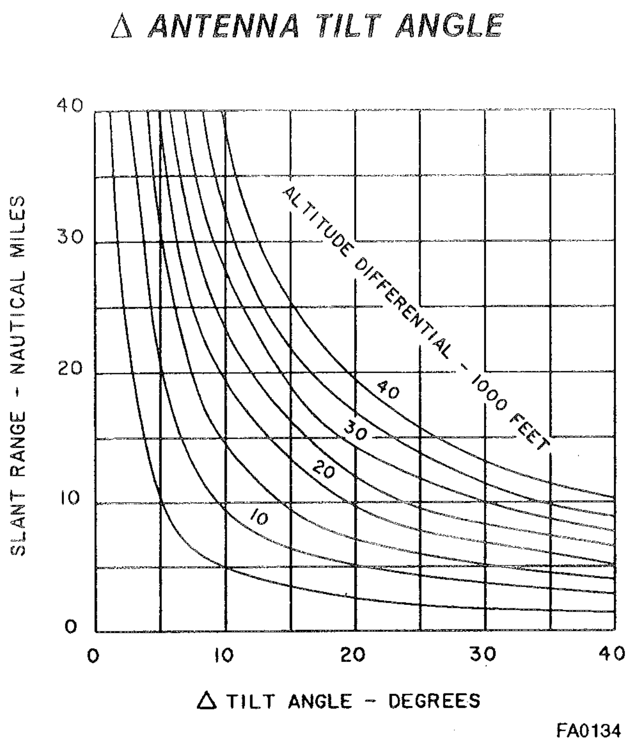


Figure 1-68

RADAR MODES OF OPERATION

The radar modes of operation are presented in the sequence in which they appear by clockwise rotation of the MODE SELECT switch on the radar control panel. They are as follows:

STANDBY MODE

In the standby mode the system operates the same as in the air-to-air mode (provided the approximately 3-minute warmup period is over) except that the transmitter is off; therefore, no target display is presented on the radar indicator except for data link (if installed) presentations.

AIR-TO-AIR MODE

The air-to-air mode is subdivided into the three submodes, or phases: search, acquisition, and track. During search, the target is located (detected) on the radar indicator. In blind acquisition, the antenna is operated in a ±5° sector scan to make the transition from ±45° search to track (lockon achieved), and during track the target is tracked. In visual acquisition, the antenna servos to the ADL when the range gate control is in detent.

If the target is within the antenna beam width and within 500 to 4000 yards range the radar automatically locks onto the first target encountered after action reject button release.

Search

In air-to-air search, the antenna beam sweeps ahead of the aircraft 90° in azimuth and 10° in elevation. The antenna is roll-corrected and pitch-stabilized and may be manually tilted up or down between +20° and −38°. Pitch corrections will permit antenna movement within the elevation limits of +25° and −57°.

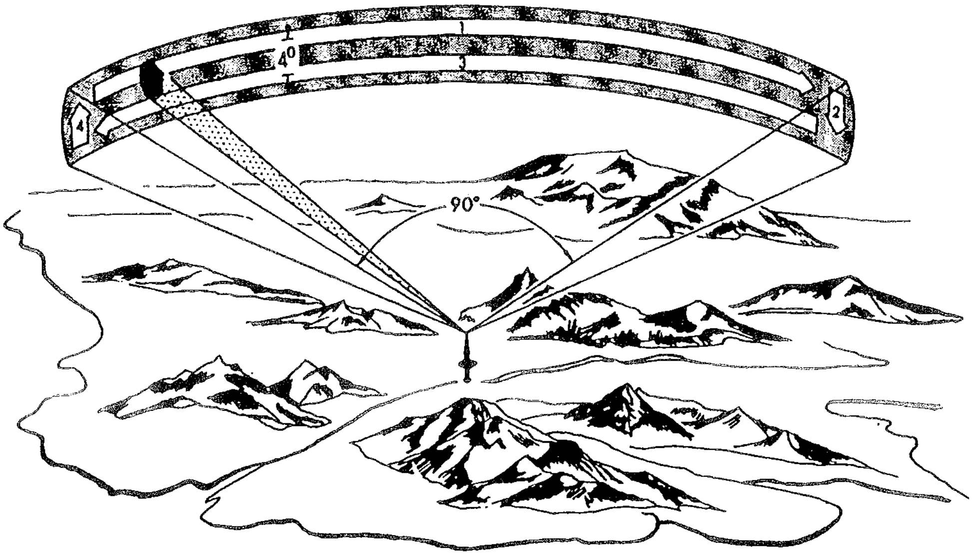
Target information is presented on the radar indicator in a PPI 90° sector-scan display in coordinates of range and relative bearing.

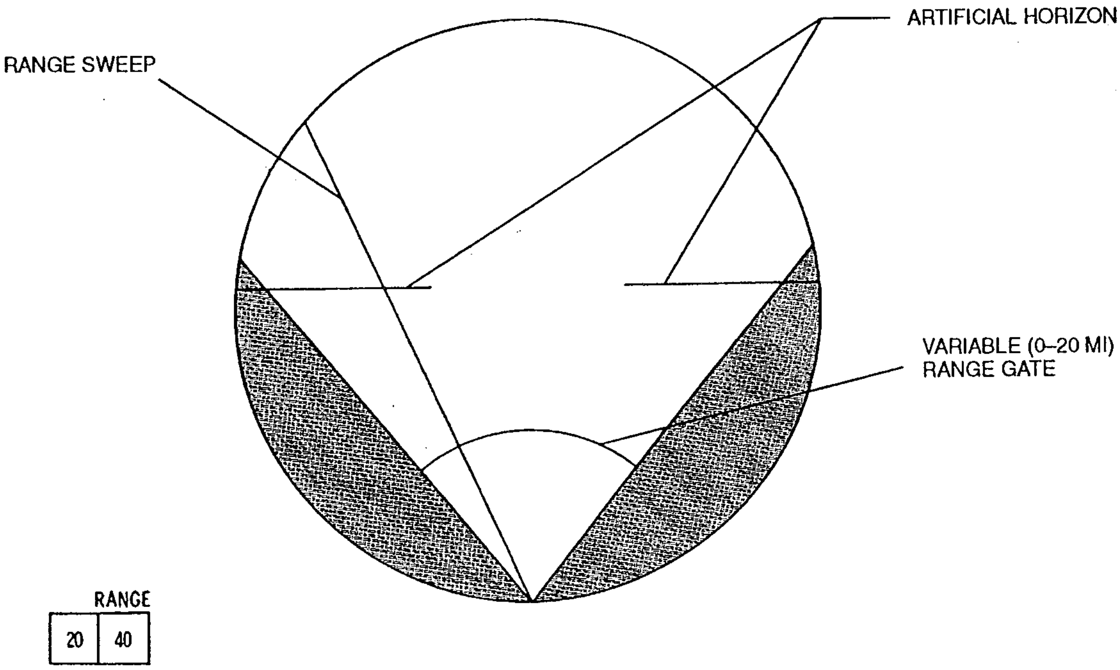
An artificial horizon line and a range gate cursor are provided. Range scale selections of 20 and 40 nautical miles are available (refer to Figure 1-69).

NOTE

Full IF gain should be used whenever possible to provide maximum lockon performance and target detection range. At low altitudes, however, ground clutter usually prevents the use of full IF gain. The clutter eliminator may be used with full IF gain or IF gain may be reduced. A combination of clutter eliminator and IF gain setting (or IF gain only, depending on the situation) should produce the best display. Normally, at close range use of the clutter eliminator should not be necessary, and the target may be detected using reduced IF gain only. The clutter elimination switch must be used with discretion, since the target may also be eliminated if within a dense clutter display. An alternate method of eliminating ground clutter (when feasible) is to remain below the target altitude so the radar antenna scans upward. The clutter elimination switch should be placed in the OFF position after lockon to assure proper display of the intensified range gate.

AIR-TO-AIR SEARCH
ANTENNA PATTERN AND RADAR SCOPE DISPLAY





FA0165

Figure 1-69

Target Acquisition and Track

WARNING

DUE TO ABSENCE OF AIMING COMPUTERS, TO AVOID DANGEROUS SITUATIONS AFTER TARGET ACQUISITION, IF LOCKON OCCURS, THE BREAK-AWAY ("X" CROSS) ON RADAR INDICATOR IS DISPLAYED IMMEDIATELY. NO WAILING TONE IS GENERATED (REFER TO FIGURE 1-70).

PROCEDURES

Before Takeoff

The before takeoff checks cover initial system activation an preliminary control settings common to all operations except as indicated.

a. Radar cooling – Air conditioning system operating (engine running, fresh air scoop fully closed, radar blower on).

WARNING

THERE IS SUFFICIENT TRANSMITTED POWER TO CAUSE A HAZARD TO PERSONNEL AND EQUIPMENT. THEREFORE, RADAR OPERATION ON THE GROUND IS LIMITED TO 15 MINUTES IN SBY OR 3 MINUTES IN ANY OPERATING MODE. LONGER PERIODS OF OPERATION MAY RESULT IN DAMAGE TO THE RADAR EQUIPMENT DUE TO INADEQUATE COOLING.

NOTE

During preflight test, rotating angle of attack vane to the extreme up or down position may cause radar antenna to return to center, or fall to the extreme down position.

b. MODE SELECT switch – SBY; allow approximately 3 minutes warmup. This step applies electrical power to the radar system through time-delay circuits for approximately a 3 minute warmup period.

c. IF GAIN thumbwheel – AS DESIRED

d. ANT TILT thumbwheel – FULL UP (aft)

e. VIDEO PED dial – ROTATE FULLY COUNTERCLOCKWISE

f. MEMORY dial – 12 O'CLOCK OR OFF (full counterclockwise)

g. ERASE INTENSITY dial – ROTATE FULLY COUNTERCLOCKWISE to erase display, then re-position fully clockwise

Normally, this control should remain in a fully clockwise position at all times, except if the display is too bright during night operations, rotate the dial counterclockwise slowly to reduce display brilliance.

h. MODE SELECT switch – GMP or A/A

NOTE

If the mode select switch is positioned to any mode prior to the system being completely warmed up, no damage will be done: however, no display will appear on the radar indicator.

i. VIDEO PED dial – ROTATE CLOCKWISE SLOWLY until a range sweep appears. Continue to rotate to the fully increase position and note whether or not the indicator blooms. If full pedestal control is available, the indicator will show a bright return over the entire sweep. Rotate the VIDEO PED dial counterclockwise slowly until a return from the sweep is noted half way down the side of the indicator. This point is just below the internal noise level of the set.

TRACK

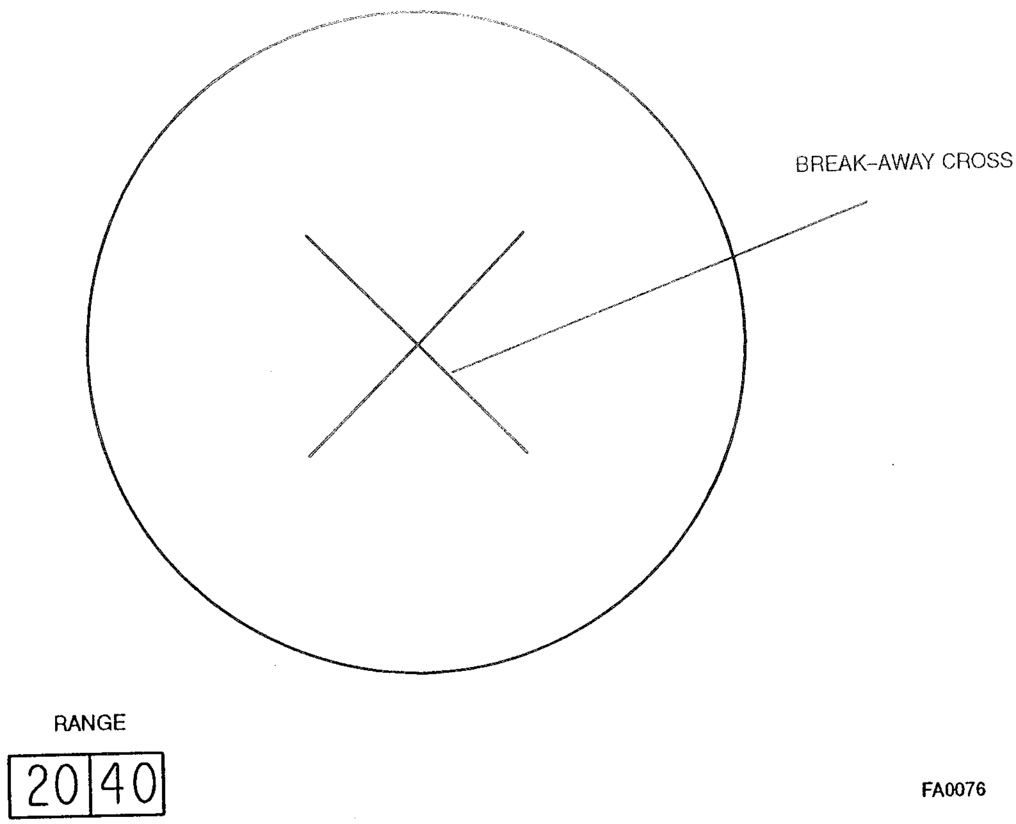


Figure 1-70

NOTE

For air-to-air operation, the VIDEO PED dial may be set slightly higher so that random specks of noise appear. This is the maximum detection setting for the radar. In order to paint noise on the indicator, increase the IF gain.

j. Cursor dial – ADJUST

k. ANT TILT thumbwheel – ADJUST TO DESIRED POSITION

l. IF GAIN thumbwheel – ADJUST TO DESIRED POSITION (full increase for A/A)

m. Panel lights brilliance – ADJUST (night)

n. LIGHT-DIM dial – ADJUST RADAR INDICATOR BEZEL BRILLIANCE (night)

o. HORIZ CTR dial – POSITION OUT OF VISUAL HORIZON LINE to the desired position

p. MEMORY dial – ADJUST DESIRED PERSISTANCE of radar indicator.

q. CLUTTER ELIM switch – OFF unless necessary to reduce excessive ground clutter.

r. XMTR TUN switch – INCREASE OR DECREASE as desired

NOTE

- Transmitter frequency adjustment is an aid in preventing jamming by ECM equipment or other X-band radars. Monitor XMTR TUN light when adjusting transmitter frequency.
- The transmitter tuning motor may or may not operate at idle RPM. If motor does not tune, increase engine RPM.

s. XMTR OFF light — CHECK; if illuminated, rotate mode select light to SBY then back to A/A. If the light remains on, the radar is inoperative.

NOTE

During cold weather te transmitter-off light may illuminate at idle RPM, indicating that the radar equipment pressure has not reached 15 psi. With the cockpit temperature mode selector switch in AUTO the turbine and heat exchanger bypass valves are open, resulting in a lowering of pressure to the radar equipment. By temporarily switching to manual COLD mode, thereby closing the turbine bypass valve, sufficient pressure is provided to extinguish the transmitter-off light in matter of seconds. After the light is extinguished (radar equipment pressurized) reselect the AUTO mode for flight.

t. MODE SELECT switch — SBY, all other radar control panel or radar indicator controls NORMAL or OFF

GROUND MAPPING

Three mapping modes are used for navigation to a target area and also as a tactical aid to finding the target. They are ground map, countour map, and terrain avoidance.

Ground Map Pencil and Ground Map Spoiled Modes

The function of both the ground map pencil (CMP) and ground map spoil (GMS) operating modes is to provide a radar picture of the terrain ahead of the aircraft. This is accomplished by programming the radar antenna scan 45° either side of the aircraft centerline. Ranges of 20, 40, and 80 nautical miles are available in both modes.

The purpose of the mapping modes is to continuously assist the pilot with his navigation regardless of weather conditions or time of day. However, dead reckoning (time and distance) remains the primary means of navigation. Any attempt to substitute radar for dead reckoning, especially at the lower altitudes, will some day cause confusion and eventually get the pilot lost.

The antenna, as in the air-to-air mode, is manually adjustable from 20° above to 38° below the armament datum line and is adjusted to provide uniform scope presentation or to position the radar on areas of interest. The antenna sweeps horizontally in a one-bar scan which is pitch stabilized and roll corrected. Pitch stabilization is necessary to hold the radar presentation constant as the aircraft changes its pitch attitude. Roll correction is effective up to about 15° of bank at which time the radar picture will begin to narrow. The antenna cannot be fully roll stabilized because there is no roll gimbal. Therefore, to compensate for roll, the azimuth and pitch functions are slowly exchanged to the degree necessary to sustain the presentation.

The main difference between the pencil and spoil submodes is the shape of the transmitted radar beam. The radiated pencil beam, measured at the one-half power points, is 3.6° in azimuth and 6° in elevation. This beam is common to all of the F-15 operating modes except spoil. The radiated spoil beam, while retaining the 3.6° azimuth pattern, is spread in elevation to provide a 55° beam. This is accomplished by extending a retractable plate or spoiler on the upper face of the antenna deflecting a large portion of the transmitted RF energy downward.

Before the returned radar signal is presented on the face of the radar scope, it passes through an electronic circuit known as "mono-pulse resolution improvement" or, more simply, as MRI. The purpose of this circuit, which is used only with ground mapping modes, is to reduce the effective beam width from 3.6° to only 2.2°. The elevation of the beam remains unchanged, being a function of the mapping sub-mode selected. With an effective beam width of only 2.2°, any return which appears on the radar scope will be more sharply defined. This feature is desirable from a navigation viewpoint in that

closely grouped targets which normally tend to merge with one another now stand apart making identification easier and much more positive.
Three items of information which are presented on the radar scope in addition to the terrain presentation are:

a. Horizon lines

b. Azimuth cursor

c. Altitude hole

The horizon lines provide an artificial horizon for pitch and bank reference. They should be aligned with the center of the radar scope prior to flight using the "Horizon Line Center" knob on the radar indicator.
The azimuth cursor is a vertical line extending from the apex to the top center of the radar scope. This line represents an extension of the aircraft longitudinal axis and may be removed from the display by counter-clockwise rotation of the cursor knob, which is located within the intensity erase dial. Continued counter-clockwise rotation of the cursor dial after the azimuth cursor has disappeared will then remove the horizon lines.

GROUND MAP (PENCIL OR SPOILED-BEAM) OPERATION

Two ground map modes are available, ground map pencil (GMP), and ground map spoiled (GMS). In either mode a map of the terrain in front of the aircraft is displayed on the radar indicator (Figure 1-71). The narrow pencil beam is normally used at lower altitudes. Proceed as follows:

1. MODE SELECTOR switch – GMP or GMS as desired

2. Range sweep switch – 80-nautical miles position

3. ALT SET & RANGE GATE knob – ADJUST so that the ground targets begin to appear at the apex of the radar screen (see operation of Altitude Set & Range Gate)

4. IF GAIN and ANT TILT dials – ADJUST to obtain the most desirable presentation

5. Range sweep switch – RESET to 20 or 40, as desired

6. Azimuth cursor – OBSERVE as a reference for aircraft heading or indication of drift

Techniques of Using Antenna Tilt, GM Modes

When using GMS at all altitudes (or GMP at 2000 feet AGL and below) only two or three small antenna tilt changes will be necessary when tracking targets. Select the desired range scale and increase the IF Gain to a relatively high position. Now this high gain setting will probably obscure the desired target, but this will be identified.
With gain up high, tilt the antenna slowly until a full scope is seen. By using the tilt meter, a good estimate may be set at once. Then the final tilt should be set by viewing the radar indicator. Let's look at several examples:

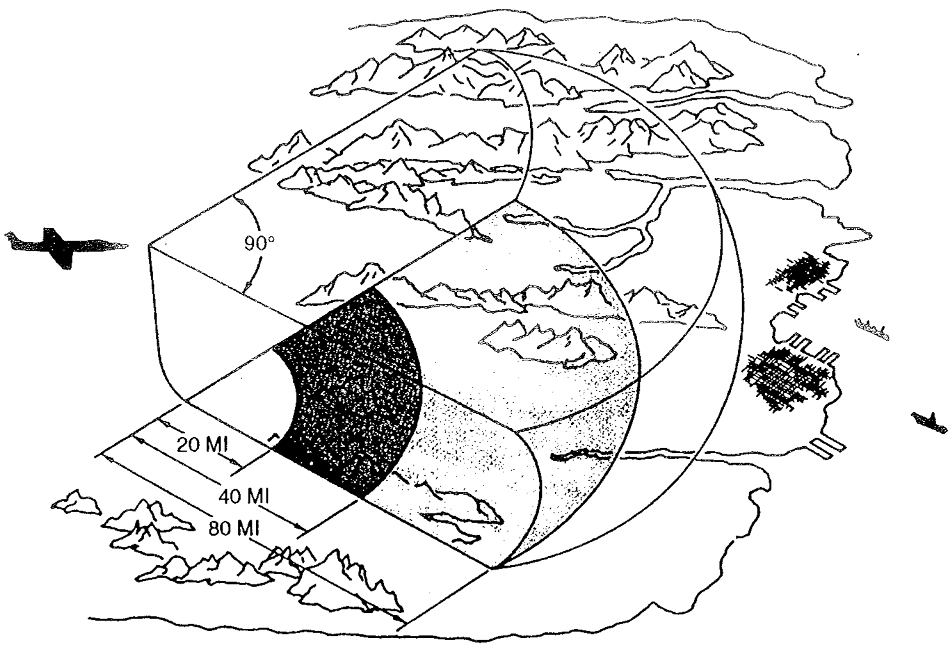
NOTE

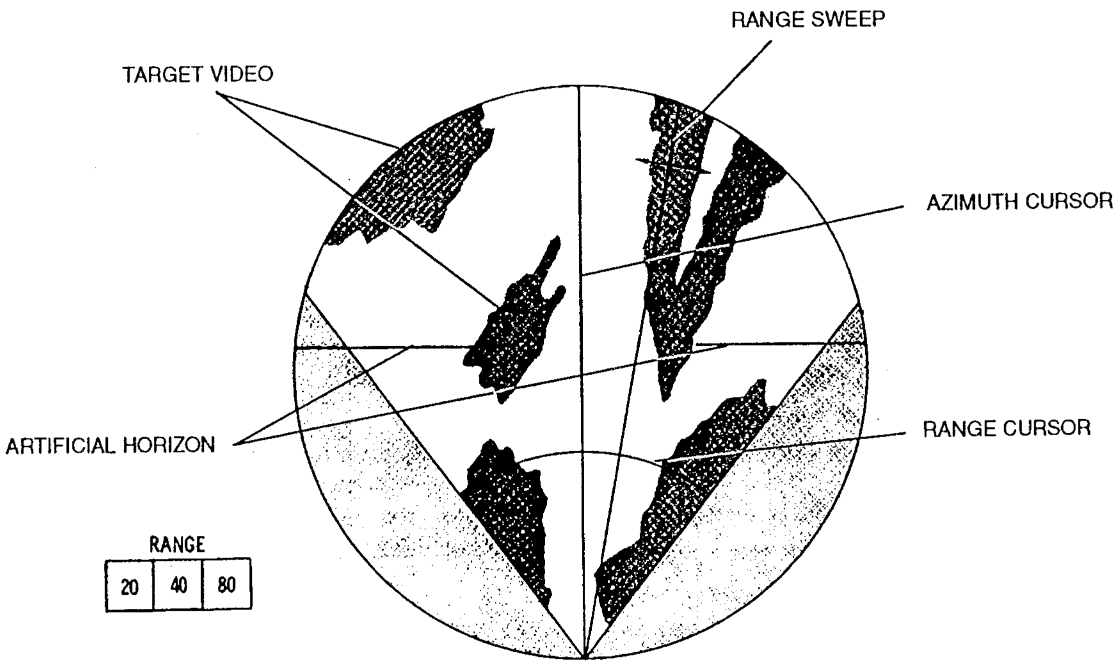
Full scope not available at high altitude in GMP.

Mode	Scale	Alt	Approx Tilt
GMS	80	30M AGL	5° dn
GMS	40	30M AGL	10-12° dn
GMS	20	30M AGL	15-17° dn
GMS	40 & 20	1-5M AGL	1-2° dn
GMP	40 & 20	1-2M AGL	0-1° dn
GMP	40 & 20	500′ AGL	0-1° up

The above figures are starting points only, as just mentioned. For final adjustments or specific results, the radar indicator should be used.
Once a good presentation is obtained, reduce the IF Gain and identify the desired target. For tracking targets at high altitude, tilt changes will be required when a new scale is used. For tracking targets to the apex of the scope, only gain changes are required. The most common error is to tilt the antenna excessively down in an attempt to follow the target with the main beam.
This causes a high antenna gain, narrow beam pattern, and required a drastically reduced IF Gain. The pilot must move each control perfectly or a loss of target will result. Not only that, but in order to use the radar for the next checkpoint, a new antenna tilt setting is required.
Once a good antenna setting has been attained, track the target by use if IF Gain only. As the target nears the aircraft (and passes out of main beam), the gain will usually have to be increased.
With this technique, only very small antenna changes are necessary and the transition to full scope operation may be accomplished simply by increasing the gain.

GROUND MAP
ANTENNA PATTERN AND RADAR SCOPE DISPLAY





GROUND MAP RADAR INDICATOR DISPLAY

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Figure 1-71

This technique is especially advantageous during combat operations when the radar is used for short periods at pre-selected points.
When the radar is switched to an operational mode, the pilot does not have time to adjust the tilt and gain each time.

Operation of the Altitude Set and Range Gate Knob in GM Modes

The altitude set and range gate knob has one function in ground mapping.

Ground Map

a. On the ground map mode of operation, the Alt, Set and Range Gate Control controls the point at which video is displayed on the scope. This control is the sweep delay in the radar. When flying over terrain which is higher than sea level, follow the procedures in Paragraph c.

b. With the sweep delay removed, the first target return on the indicator will be the ground under the aircraft. This will also give you approximate altitude above the ground. Also, in this position, all range information is slant range.

c. In order to read true ground range on the radar indicator, proceed as follows: GMS-20MI Scale, high IF Gain and rotate the Alt. Set knob (or wheel on modified aircraft) full clockwise. In this position, the sweep delay is gone and a blank spot on the bottom of the scope will be seen. This is called the "alt, hole". With this present, slowly rotate the Alt. Set knob counter-clockwise until the first return is moved to the apex. Now the indicator reads true ground (nautical) miles. Unless absolute accuracy is desired the Alt. Set knob may be left in detent.

The physical property of the ground which is to be interrogated determines how to set the controls and what the pilot may expect from the display. If the area consists solely of terrain having uniform reflectant characteristics throughout the entire sweep, the indicator will display a uniform intensity throughout the sweep.
However, most areas have features whose configuration or composition will exhibit varying reflection/absorption/dispersion ratio to RF energy. The signal returns received will either "paint" or will not appear at all, depending on the gain applied to the receivers. In other words, a given return will either "paint", (having the same intensity throughout), or will not be seen at all.
Weak received signals may be made to paint by increasing the IF Gain. However, the very weak returns will paint with the same intensity as the strongest returns. In the mountainous areas, there are extremes of these signals.
There are three general classes of targets, which call for three corresponding nominal IF Gain control settings.

High Gain. High gain is used when the pilot wishes to observe a "no show" area such as a river, mountain shadow (radar), coastline, dry lake, etc. High gain tends to "cover over", the scope and only general characteristics will be displayed.

Intermediate Gain. Intermediate gain is used when the pilot wants to observe the orientation of "no show" features to the "show" features, such as towns or cities along highways, orientation of mountain ranges or ridges to surrounding terrain. This technique is useful when a specific area or target must be identified by its orientation to other known returns. (In this case, returns are pointing to a specific target or area which is not clearly displayed.) The orientation of the target to the pointers is established by radar prediction during the flight planning phase.

Low Gain. Low gain is used when the pilot wants to identify and isolate a target to the maximum possible degree. Features or targets on which low gain will produce a "show" have high reflective characteristics. Example of this type are large hangars, marshalling yard on the edge of a town, a concentration of tall buildings (steel and concrete), a large bridge, etc.

General GMS Techniques

As previously stated, GMS is used most effectively above 3000 feet AGL. Let us assume a level off at 30000 feet with the controls adjusted as on chart. With this setting, the video returns on the scope should be fairly uniform.
If they are not, slowly rotate the antenna down until a uniform video return is obtained. Once the scope is "full", rotate the IF Gain to the desired position. Slowly decrease the IF Gain until a good picture of the terrain is presented. To optimize the display, slowly adjust the intensity and memory. (It may be necessary to adjust the antenna slightly.)

If selection of a target is required, proceed as above. When definite identification of the selected target has been accomplished, continue to decrease the IF Gain until target "breaks out" of the ground returns and clutter. From this point, the selected target may be tracked down the scope.
Since the RF energy is optimized to 40 miles in the spoiled mode, it will be to your advantage, when at altitude, to use the pencil beam when 80 mile check point identification is desired.
When flying at intermediate and low altitude (above 3000 ft AGL), the same technique may be used.

Ground Map Pencil

The inherent advantage of the pencil beam is realized through clarity of the returns and ability to "break out" targets, even at high altitudes, and distant ranges.
In the Ground Map Pencil mode, the area covered by the transmitted energy is a function of altitude, antenna elevation angle and beam-width height of the transmitted energy. Since the transmitted-beam height remains constant at 6.2°, the primary variables affecting the ground area covered are altitude and the antenna-tilt angle.
In the GMP mode, consider for an example, that the antenna angle is 0°. At an altitude of 3000 feet AGL, the bottom of the beam will contact the ground at about 11 nautical miles.
At an altitude of 7200 feet (AGL), the bottom of the beam will contact the ground at about 22 nautical miles. From this, we may establish a rule of thumb for the use of Pencil and Spoil submodes of the GM mode.
When using GMP at higher altitudes (15000 feet and above) you will notice the narrow beam pattern on the indicator. The gain must be readjusted for optimum display if GMS were being used. It will be necessary to adjust the antenna tilt every few miles in order to track, or map a specific area. The IF Gain must be reduced as range to target is reduced; this is caused by increased antenna gain. GMS is best at intermediate and high altitudes, and close ranges.
At low altitude (below 3000 ft AGL), GMP is inherently superior to GMS. The relative flat dispersion of RF energy causes a uniform display at low altitude. The high concentration of energy may require a low IF Gain setting. Very little movement of the antenna will be necessary. Once a satisfactory gain setting has been achieved, the display is best optimized through use of intensity and memory.

Effect of Bank Angle GMS/P, CM/TA

Since there is no roll gimbal, the antenna is not truly roll-stabilized. However, roll compensation is computed and permits bank angles up to approximately 15° without an appreciable error affecting the relationship of the target presentation.
Sector narrowing, in ground map modes, is caused by an effective physical reduction in the azimuth sweep limits, whereas the effects on contour map and terrain avoidance when a bank angle is introduced are caused by a rotation of the transmitted beam (fixed antenna).

Effect of Beam Pattern

The pencil beam pattern consists of:

a. The main beam

b. Several secondary beams above, below, and to the sides of the main beam.

The secondary beams are referred to as side lobes. These are due to antenna configuration and radome characteristics. The side lobes have identical characteristics of the main beam except the transmitted power is much lower than that of the main beam, and in CM/TA the phase relationship is reversed. The only side lobe which is of importance is the one appearing immediately below the main beam.
This side lobe is approximately 12° down from the main beam and its phase relationship is inverse to that of the main beam. Therefore, in CM and TA, if the system sees the return from this side lobe, a false height of the target will usually be computed, and a warning signal will be displayed.
This signal from the side lobe will be displayed at any time when the object in the lobe has high reflective characteristics such as mountains, hangars, buildings of concrete and steel, etc. The range at which this side lobe warning will be displayed is a function of altitude.
In contour map and terrain avoidance, the side lobe warning also becomes an inverse function of the clearance plane setting, that is, a reduction of the clearance plane setting will increase the range of the side lobe return.
In most cases, this depends on altitude above the terrain, that is, when in excess of 2000 feet above level terrain, the clearance plane set at 1000 feet or less, the side lobe return may show a warning at a range outside of the fail safe cursor. The pilot must be aware of this characteristics.
If the pilot lowers and raises the clearance plane with no apparent, or very little, movement of the suspected side lobes, it may be assumed these are

side lobe returns. Always check the map and desired course to help determine the position where side lobes may be expected.

CONTOUR MAP MODE

The contour map mode provides a display of the terrain contour in the forward quadrant. In contrast to ground mapping (which simply differentiates between surface features), contour mapping differentiates between altitudes of topographic projections above a preset clearance plane. The clearance plane is parallel to true horizontal at all times, at an elevation that is manually adjustable from 0 to ±6000 feet relative to the altitude of the airplane. Only those surface features that have sufficient height to project into the clearance plane are visible on the contour mapping display. These features appear on the display in a manner similar to the way they would appear on a conventional topographic map of the area.

Contour Map Operation

1. MODE SELECT switch – CM
2. Range sweep switch – 20 NM POSITION

NOTE

The 40 nautical mile range scale may be used but the presentation is not reliable.

3. CL PLANE SER switch – HOLD AT INCR OR DECR as required to set desired clearance
4. Observe clearance plane indicator for the clearance as set in feet
5. Observe radar display for contour of terrain at intersection with the clearance plane
6. ABOVE CL button – The contour displayed will be that of terrain at the preset clearance but above instead of below the aircraft. This may be desirable when flying in a valley with mountains on either side. As the clearance is being set, the antenna tilt pointer will move in synchronization with antenna movement. When the ABOVE CL button is depressed the pointer will move to an up indication corresponding to the down position previously indicated.

WARNING

IN BOTH THE CM AND TA MODES, ACCURACY OF THE CLEARANCE PLANE IN THE ABOVE-CLEARANCE AND THE 40-MILE RANGE FUNCTIONS ARE DEGRADED, CONSEQUENTLY THE ABOVE-CLEARANCE PLANE FUNCTION SHOULD BE CONSIDERED INACCURATE AND ONLY USED DURING AN EMERGENCY.

Characteristics and Limitations of the Clearance Plane in Contour Map and Terrain Avoidance

The clearance plane establishes a reference plane below the aircraft for determining unsafe obstruction along the flight path.

a. In contour map the clearance plane remains parallel to the true horizontal regardless of aircraft pitch attitude.

b. In terrain avoidance the clearance plane remains parallel to the aircraft flight path.

c. The distance below the aircraft at which the clearance plane occurs is established by pilot selection, using the clearance plane set control on the radar control panel. Clearance plane altitude is read on the clearance plane and antenna tilt indicator panel.

The system displays a return depending on the computed elevation of the object in relation to the clearance plane. Only objects protruding into the clearance plane will "paint".
The radar, through various circuits, will compute whether or not an object is protruding into the clearance plane and thereby paint a return on the indicator, or whether the signal is negative (target below the clearance plane, and therefore not painting). Some very hard targets, such as a large hangar, will produce a warning, or "show", due to side lobe effect, even though that particular target may be below the clearance plane setting.

CONTOUR MAP FUNCTION

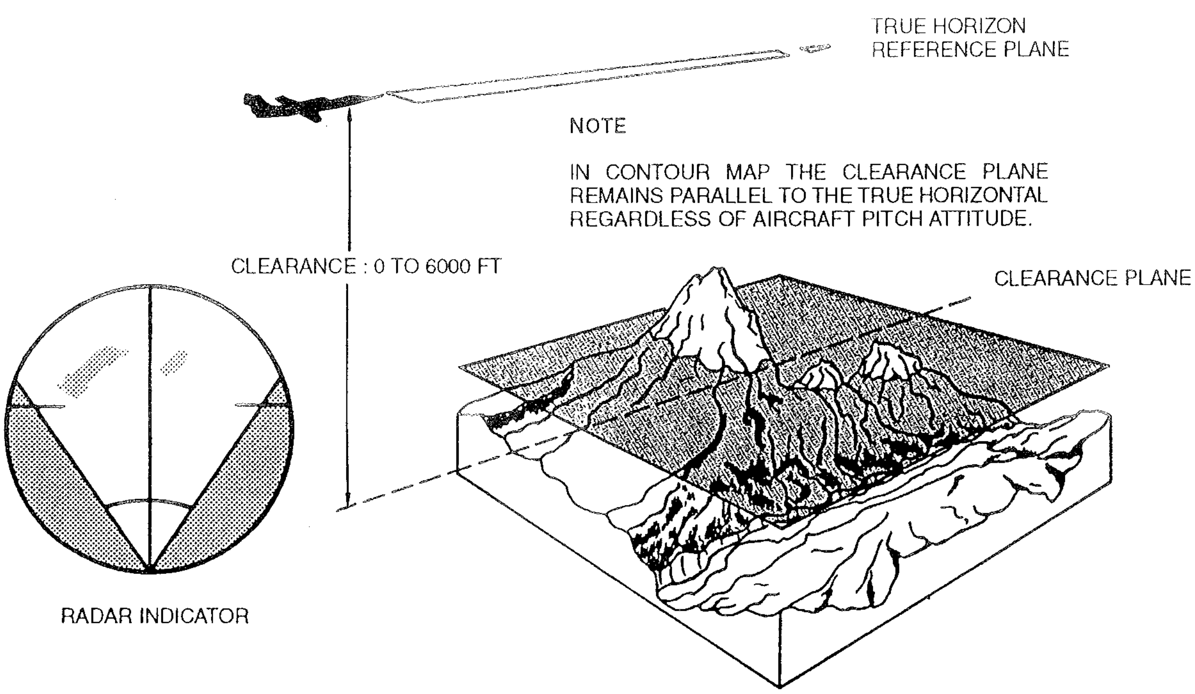


Figure 1-72

Effective Antenna Alignment

The antenna is bore-sighted physically and electronically during the ground calibration procedures so that the centerline is projected 25 milliradians depressed from the 10 nmi line on the clearance plane. In contour map, the output of the inertial navigator stable platform is used to maintain the antenna sweep parallel to local horizontal and is roll corrected. The clearance plane set switch keeps the antenna "lookdown" angle at the proper declination to maintain the proper crossover point with the horizontal clearance plane.

In the terrain avoidance mode, roll correction is maintained but angle of attack information is substituted for pitch stabilization so that the clearance plane is no longer stabilized to the true horizontal but will follow and be parallel to the flight path. Plane reference is taken from the right guide vane and fed through the ADC.

WARNING

THE MISALIGNMENT OF ANTENNA TILT (CLEARANCE CALIBRATION KNOB) WHEN AIRBORNE WILL YIELD A TILTED CLEARANCE PLANE WHICH WILL, IN EFFECT, JEOPARDIZE THE PILOT'S SAFETY WHILE USING CONTOUR MAP OR TERRAIN-AVOIDANCE.

Generally ground calibration of the CM/TA plane will be flight checked on each flight if possible. If sufficient flight time is available to air calibrate a misaligned set it would be accomplished and then noted in the A/C forms.

TERRAIN AVOIDANCE MODE

In the terrain avoidance mode, as in contour mapping, only those objects projecting into a preset clearance plane are displayed on the radar indicator. The clearance plane elevation is adjustable from 0 to $\pm$6000 feet with respect to the airplane. There is one significant difference from the contour map mode, however: in the terrain avoidance mode, the orientation of the clearance plane is maintained parallel to the aircraft flight path, rather than to true horizontal as in contour mapping.
Indication of obstacles in the flight path will appear on the radar indicator in sufficient time for avoidance of collision. When an obstacle appears on the indicator, the pilot climbs until the obstacle disappears. A fail-safe cursor is also displayed in this mode to indicate satisfactory performance; however, the fail-safe cursor does not appear on the 40 mile range. Target information displayed inside the cursor is unreliable. (See Figure 1-73 for reliable target range boundaries as a function of clearance plane in contour map and terrain-avoidance modes of operation).

NOTE

Hard targets may show outside the fail-safe cursor with low CL plane setting. These are caused by the side lobes.

Terrain Avoidance Operation

The terrain avoidance (TA) mode is similar to the contour map mode, except that in contour map the clearance plane is always parallel to local horizontal, whereas in terrain-avoidance mode the clearance plane is parallel to the aircraft flight path (Figure 1-74).
The procedure is similar to that for contour map except that the mode select switch is turned to TA. There are two differences in the indicator display: no azimuth cursor is displayed and a fail-warning cursor is presented. Absence of the fail-warning cursor means the terrain avoidance mode is unreliable. Targets that are outside the boundaries of the curves illustrated on figure may or may not be displayed. If they are displayed, they must be considered unreliable.
The range sweep beyond 20 nautical miles is unreliable. The RANGE VERSUS CLEARANCE PLANE curve extending from zero nautical mile range to 7 nautical miles corresponds to the fail-warning cursor range displayed in the terrain-avoidance mode.
However, a fail-warning cursor is not displayed in contour-map mode and there is no indication of "outside the upper limit of beam" in either contour-map or terrain-avoidance mode.
In the CM/TA mode clutter signals may appear in the form of light noise specks or false targets. This may occur over certain types of terrain with clearance plane settings above 500 feet. The probability of getting the clutter or false targets will increase as the clearance plane setting increases.
These clutter specks or false targets may be distinguished from normal terrain targets as follows:

a. The clutter specks or false targets will appear suddenly (generally in the fail-safe cursor) and will be at a lesser range that the real target.

b. These clutter specks do not follow in range with the normal terrain targets, nor change in intensity as range to presented target decreases.

c. By lowering the clearance plane to 500 feet the real targets will increase in size and intensity, while the clutter specks should disappear.

Clearance Plane Accuracy Check

An accurate check of the clearance plane must be accomplished before either the CM or TA mode is used. Two requirements are necessary to make this check:

a. An accurate terrain elevation point.

b. An accurate altimeter setting for the check area.

If the mission is low level, it is better to make the check as soon as possible after takeoff. The point used to check the clearance plane must be a sufficient distance from the takeoff point to allow for turn level off at the desired altitude, usually between 30 and 40 nmi.
If the mission is high level after takeoff, a forecast altimeter setting and a check point for the CM/TA plane check is necessary in the area of letdown.
The procedures for making the check, either after takeoff or in the letdown area, are the same and are as follows:

a. Assume terrain elevation of 1700 ft MSL for the check-point.

ANTENNA ELEVATION COVERAGE CHART

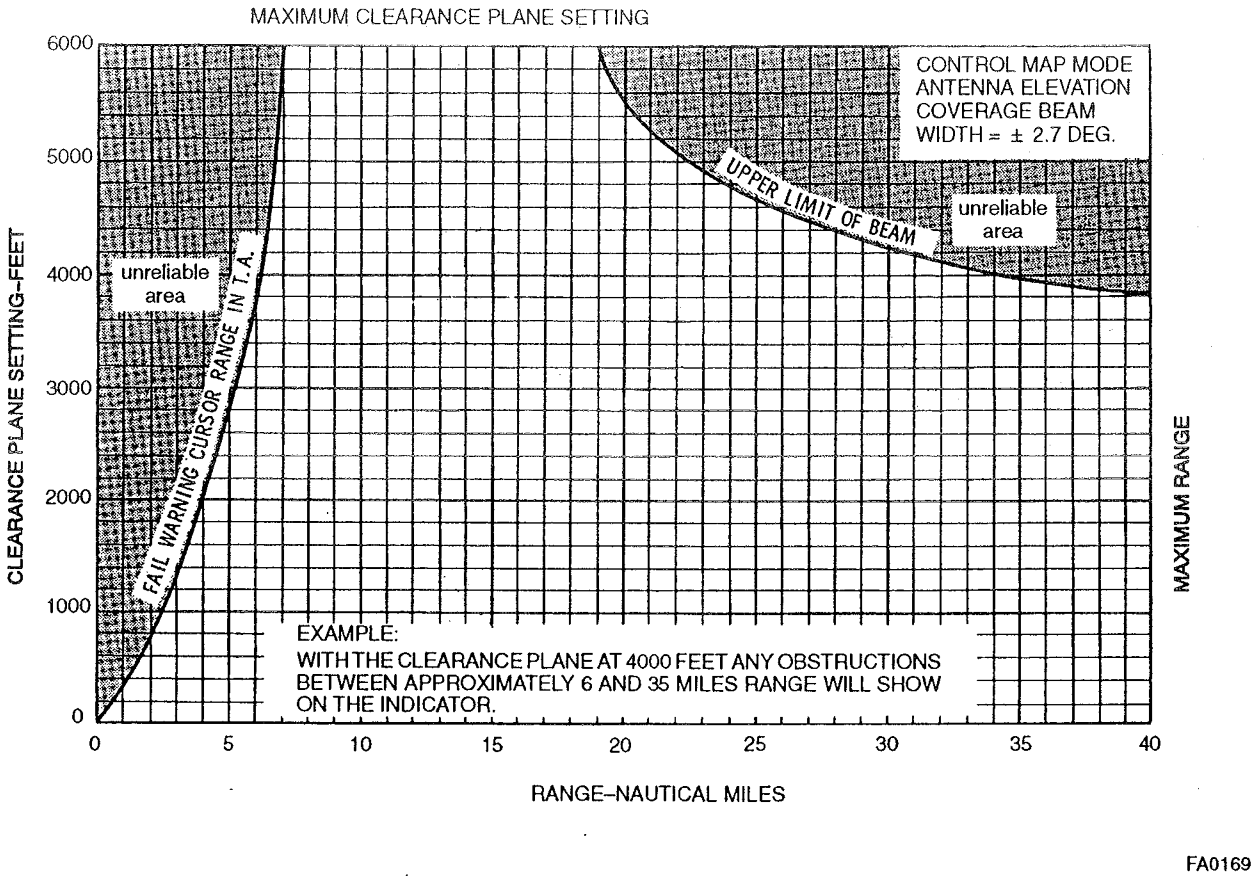


Figure 1-73

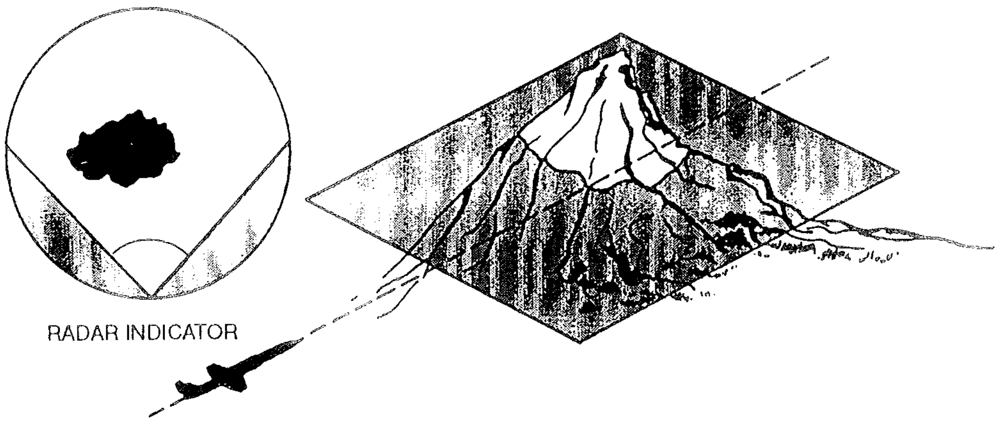
b. Assume that an altitude of 500 ft above, or 2200 ft MSL is established. Select contour map mode and lower the clearance plane to 1000 ft. Identify the peak and adjust the clearance plane up until a small return is painted on the indicator. Now, look at the readout on the Antenna Tilt and Clearance Plane Indicator. If the clearance plane is perfect, the readout will be 500 ft.

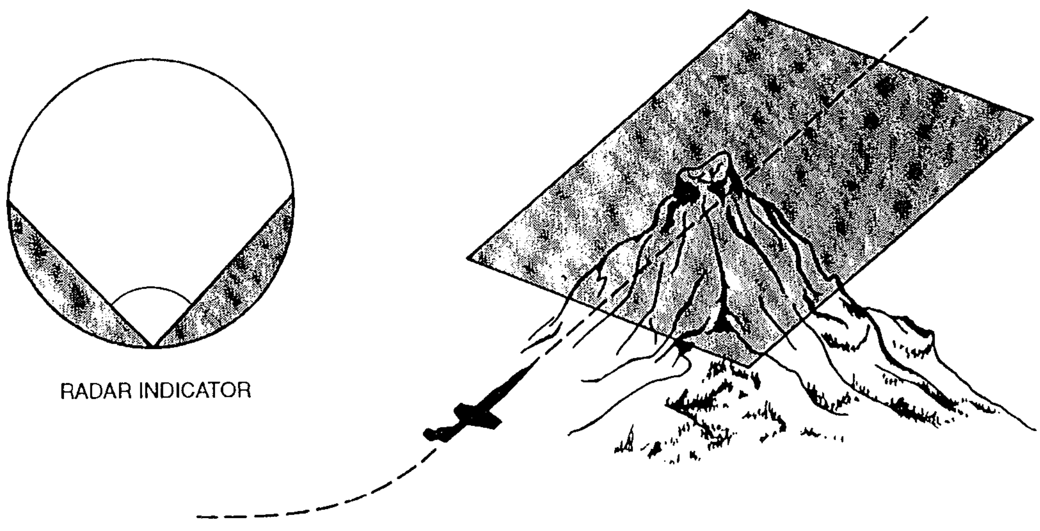
c. If the readout is not 500 ft, which will probably be the case, note the difference. For instance, 300 ft readout, actual altitude 500 ft. Since the clearance plane has a tolerance of ±125 ft, and calibration readout is normally equal to ±200 ft, the two errors mentioned above are not cumulative. A well-calibrated clearance plane will read out ±200.

d. Track the selected target from 20 nmi down to 5 nmi. At 15, 10, and 5, perform checks of terrain avoidance. Here again, there is an allowance of ±125 ft of readout difference between CM and TA. To check the allowable tolerance, the clearance plane may have to be raised or lowered slightly.

e. If the target return remains approximately the same size on the indicator, all the way to the 5 nmi point, the clearance plane is level. The clearance plane itself is allowed a certain amount of curve. This sine/cosine curve may vary the readout slightly. The normal clearance plane tolerance, ±125 feet, should not be exceeded when checking between contour map and terrain avoidance.

TERRAIN AVOIDANCE FUNCTION





FA0170

Figure 1-74

Tilted Clearance Plane

WARNING

ANY TIME THE CLEARANCE PLANE IS TILTED, IT ENDANGERS THE LIFE OF THE PILOT. BY FOLLOWING THE SAME PROCEDURES GIVEN ABOVE, LARGE VARIANCES IN READOUT WILL INDICATE TO THE PILOT WHETHER THE PLANE IS TILTED.

For example, at 20 nmi the readout is 000, and a large return pattern is still seen on the indicator. As range to target decreases, the size of the return pattern decreases. As the target comes to within 5 nmi, the clearance plane must be lowered to 500 ft in order to maintain target contact. This indicates a clearance plane error that is in excess of 500 ft. This is dangerous.

NOTE

Clearance plane accuracy in 40 mile scale is degraded.

Check at Co-Altitude

The co-altitude procedures is not recommended for performing the clearance plane check. There is no satisfactory method of checking the errors and clearance plane tolerances.

Level Flight

Level flight is a condition which is chosen by the pilot. He establishes the flight altitude based on MERA's. If the flight is over level terrain, the display will indicate level terrain from minimum range to maximum range selected (20 or 40 nmi). Height of present position is determined by lowering the clearance plane setting and observing terrain display from two miles minimum to five miles.

Once established at Minimum Enroute Radar Altitude over level terrain, the clearance plane should be raised slightly to give a threshold or "just paint" condition on the indicator.

If the terrain is rolling and mountainous, MERA may be established from the lowest terrain indicated along the route (and that height held until higher terrain is indicated on the radar).

The higher terrain should not be approached closer than 10 miles if it is on the course line. From this, an action distance is calculated. The action distance is a function of speed and height above the ground (in no case less than 5 miles in the F-104). When the action distance is reached, a climb over or a turn to go around the obstacle must be executed. If the terrain presents a hard reflective surface to the side lobe, "false" warnings may appear on the scope at a range where the side lobe contacts the surface. The strength and range of this characteristics will vary with individual aircraft, radar system, and clearance plane settings.

Climbs and Level Off

In terrain avoidance, a climb over any obstacle whose height is above the course altitude should also be initiated from the action distance. In mountainous areas, set the clearance plane at a minimum of 700 feet.

When an obstruction is indicated by the radar, check the map to determine the number of feet that must be climbed. When the return reaches the action distance (usually 10 nmi) select TA and a minimum of 700 feet clearance plane setting. Change the pitch attitude slowly and initiate a climb. When the target disappears in the TA mode, hold the established climb attitude on the attitude indicator. Select the CM mode and continue the climb until the target disappears. As soon as the return fades (it should fade at approximately 5 nmi), level off.

The clearance plane may be lowered slightly in order to establish check point, return identification, or a slight descent may be started to re-establish target contact. Check the altimeter - it should read the number of feet set in the clearance plane, above the obstacle. Start timing to determine obstacle passage. When the obstacle passes below the aircraft, establish a descent, if required.

Letdown and Level Off

Select ground map and interrogate the area for a navigational check point and its general features, if necessary (pre-plan this). Select TA mode and push-over until the first returns are at approximately 7-8 nmi on the scope. Continue descent, allowing the returns to approach 6 nmi.

Hold your vertical speed and Mach number constant during the descent. When the terrain (check point if possible) approaches 6 nmi and letdown angle is 5-6°, select CM mode. Do not allow returns to come closer than 5 nmi during the descent. Adjust aircraft pitch to control the range of the returns. Level off in CM mode at the desired clearance plane setting.
Use MERA technique after determining the characteristics of the terrain, and establish the height above ground, the height of your future position, and the action distance (between 5 and 10 miles) until stabilized in level flight at the desired speed.

Terrain Following

This technique may be used in the F-104G only if the grade of the terrain is shallow and extensive. Flight path clearance may be established in terrain avoidance so long as the terrain characteristics remain fairly constant over a considerable distance.
Frequent use of contour map and the 40 mile scale is required to determine any terrain features which may be beyond the capabilities of the established "action" distance. This will prevent any sudden surprises for the pilot.
When following the terrain in terrain avoidance, pull the nose up until the display comes into the fail safe cursor in order to establish a flight path (and clearance plane) that is parallel to the terrain. When the display reaches the fail safe cursor, the flight attitude should be held.

NOTE

The 40 mile scale is not reliable and should be used only as an indication that an obstacle of undeterminable height is in front of the aircraft.

If the grade of the terrain drops off, which requires a slight descent, the display will indicate this by an absence of painting at the top of the indicator. However, a rise in the terrain may not be noted if the rise of the grade is less than the attenuation slope of the upper half of the radar beam. For this reason, slight changes in flight path (up) are required to obtain a threshold or "just paint" indication of the terrain beyond 5 nmi (using the bottom half of the beam). If the rise of the grade is greater than the attenuation slope of the upper half of the beam "shadows" will appear.

Fail Safe Warning and Emergency Action

The TA fail safe cursor position indicates two things:

a. Minimum range of usable information (a function of clearance plane setting).

b. Radar system malfunction (when absent or broken).

When flying contour map, the fail safe cursor should be checked periodically by switching to TA. If the cursor does not appear, or appears "broken", do not use CM/TA modes, since a malfunction has occurred. Further, the range of the cursor should be established when in the terrain avoidance mode, or when changing clearance plane settings in CM mode, switch to TA to establish cursor range.
If, for any reason, the pilot suspects or sees a malfunction indicated by the absent or broken fail safe cursor, the MEA should be immediately adhered to.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

Landing and Taxi Lights

A landing light is installed on each main gear aft door. The light will be in position for use any time the landing gear is extended, these lights are used also as taxi lights. The landing lights receive power from XP2 bus and both lights are controlled by a switch on the left forward panel in both cockpits. The switch has three position, LANDING LIGHTS, OFF and TAXI LIGHTS. The lights automatically shut off when the nose gear is retracted, and either position energizes both taxi and landing lights.

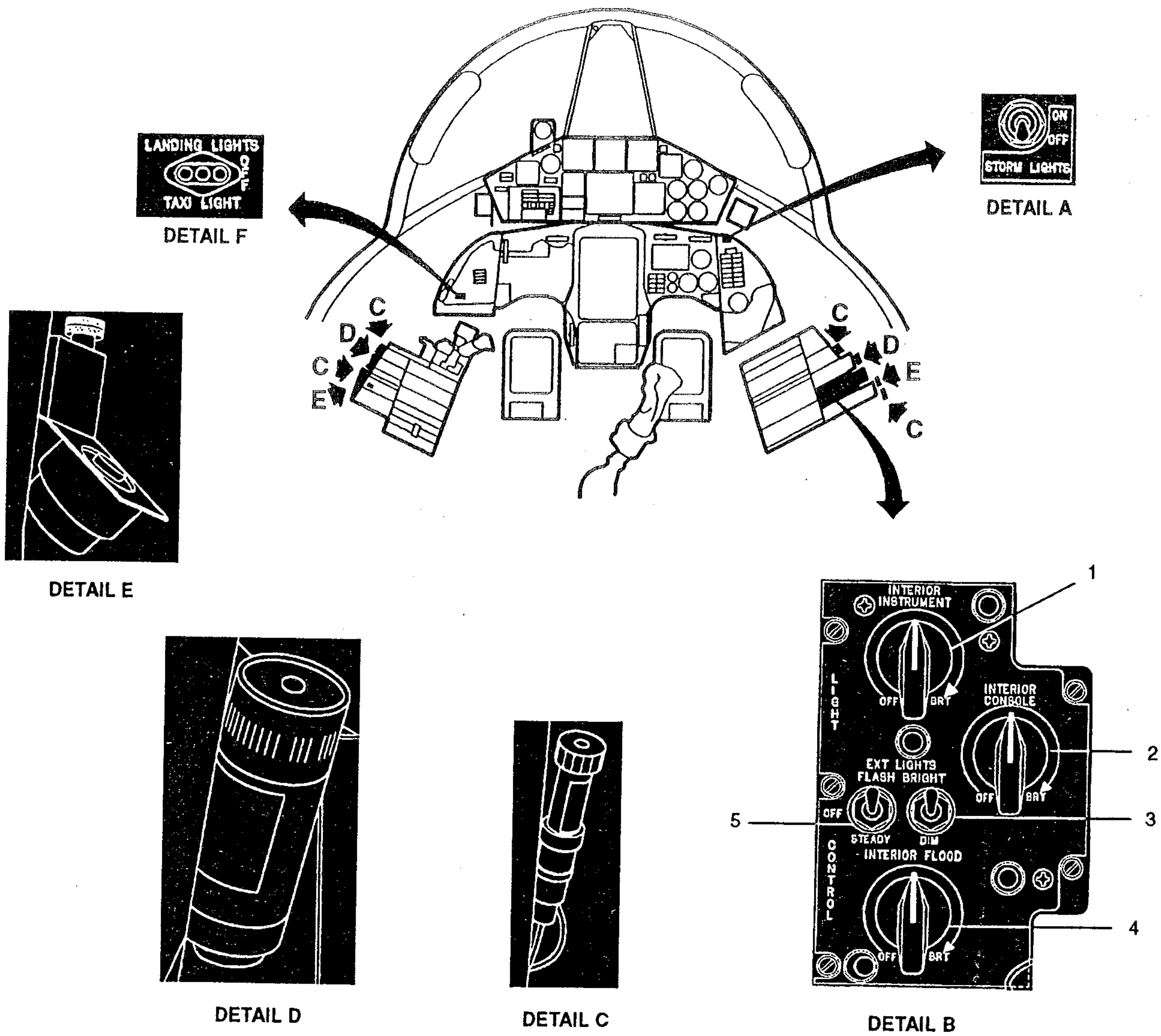
NOTE

Damage to the landing lights may be caused by a prolonged use on the ground.

Navigation Lights and Switches

The navigation lights include two yellow upper tail lights two white lower tail lights, a green and a red

LIGHTING CONTROL PANELS (FRONT COCKPIT)



A – STORM LIGHTS SWITCH

B – LIGHTING SYSTEM CONTROL PANEL
1 INTERIOR INSTRUMENT SWITCH
2 INTERIOR CONSOLE SWITCH
3 NAVIGATION LIGHTS DIMMING SWITCH
4 FLOODLIGHTS SWITCH
5 NAVIGATION LIGHTS SELECTOR SWITCH

C – FLOOD LIGHTS

D – SPOT LIGHTS

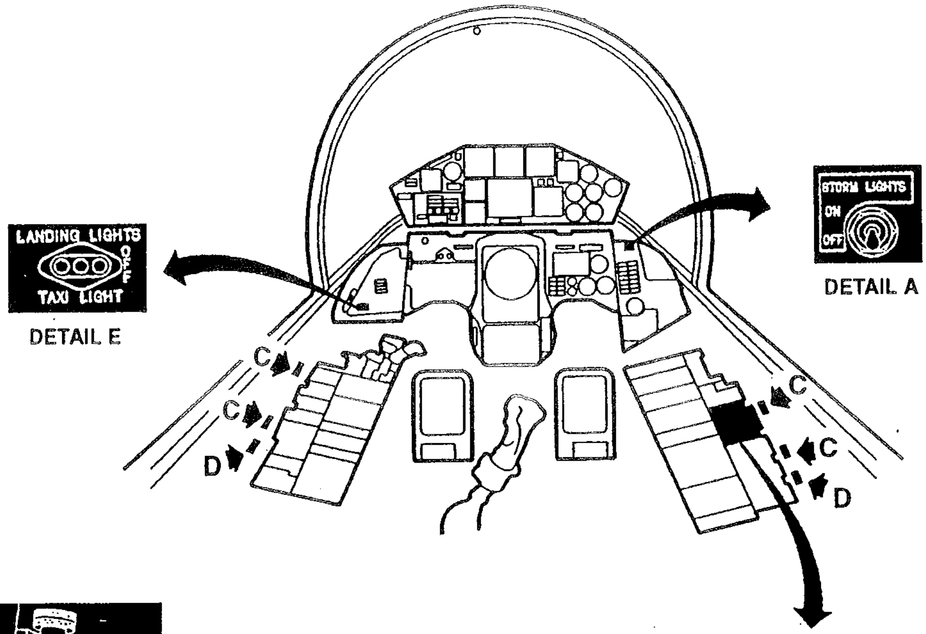
E – THUNDER STORM LIGHTS

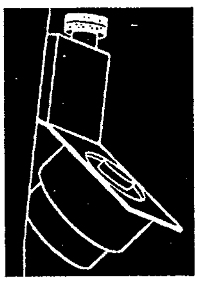
F – LANDING AND TAXI LIGHTS SWITCH

FA0132

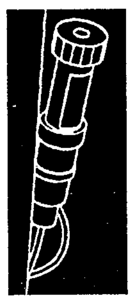
Figure 1-75 (Sheet 1 of 2)

LIGHTING CONTROL PANELS (REAR COCKPIT)

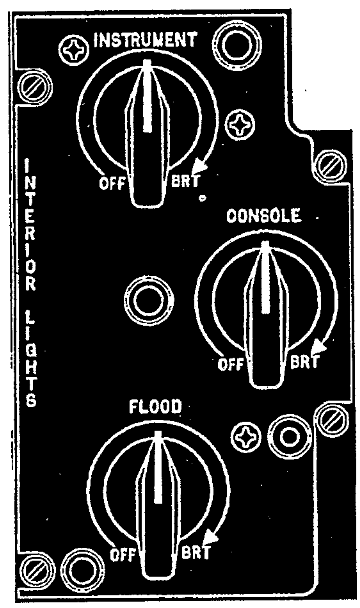




DETAIL D



DETAIL C



DETAIL B

A – STORM LIGHTS SWITCH

B – INTERIOR LIGHTS PANEL
1 INSTRUMENT SWITCH
2 CONSOLE SWITCH
3 FLOODLIGHTS SWITCH

C – FLOOD LIGHTS

D – THUNDER STORM LIGHTS

E – LANDING AND TAXI LIGHTS SWITCH

FA0133

Figure 1-75 (Sheet 2 of 2)

fuselage lights and white top and bottom fuselage lights. The formation lights are installed on the tip tanks. The lights are controlled by a selector switch and a dimming switch located on the right console lighting control panel in the front cockpit only (Figure 1-75).
The lights are energized from the PP1 bus when the selector switch is moved from OFF to STEADY or FLASH. Intensity of the navigation lights depends upon the position (BRIGHT or DIM) of the dimming switch. With the selector switch in the STEADY position, all of the navigation lights are energized continuously.
In the FLASH position, the top and bottom fuselage lights still burn steadily while the remaining navigation lights are energized intermittently through a flasher unit at a rate of 40 flashes per minute.

INTERIOR LIGHTING

The interior lighting system comprises instrument lights, console panel lights, console floodlights, thunderstorm lights, spotlights, and associated wiring, circuit breakers, and controls.

Instruments and Console Panel Lights

The HSI indicators incorporate edge lighting and integral illumination. The remaining individual instruments are illuminated with post or shielded light fixtures located as required adjacent to each indicator. The major left and right console control panels are indirectly illuminated by edge-lighted plastic plates. The instrument lights are controlled by a rheostat-type switch placarded INTERIOR INSTRUMENT in the front cockpit and INSTRUMENT in the rear cockpit. The console panel lights are controlled by a rheostat-type switch placarded INTERIOR CONSOLE in the front cockpit and CONSOLE in the rear cockpit. These switches are located on the lighting control panels (Figure 1-75). They turn the lights on and provide brightness control. These lights receive power from the XP5 bus.

NOTE

The interior instrument rheostat-type switch also controls the warning lights brilliancy (refer to Warning Lights Dimming System).

Floodlights

Four floodlights are installed to provide overall lighting for each of the consoles and the instrument panel. Two lights are located on each side of the cockpit, above the aft and forward portion of the consoles. They are controlled by a rheostat-type switch placarded INTERIOR FLOOD in the front cockpit and FLOOD in the rear cockpit, located on the lighting control panels (Figure 1-75), similar to the instruments and console panel light switches.
Power is provided by the XP2 bus through the CKPT. FLOOD LTS. circuit breaker located on the right console in the rear cockpit.

Thunderstorm Lights

High intensity (thunderstorm) lights are installed in the cockpit to illuminate the upper and lower main instrument panel. These lights are normally used during thunderstorm conditions or when normal illumination is inadequate. Thunderstorm lights are controlled by the two position (ON/OFF) STORM LIGHTS switch, located on the right side of the main instrument panels. They are electrically powered by the PP2 bus through the CKPT SPOT LTS circuit breaker located on the right console in the rear cockpit.

Spotlights

Two spotlights are used for emergency illumination. Both are located on the end of the left and right consoles. The right spotlight may be also placed in a proper fitting located in the right side of the cockpit. The light intensity may be controlled by a rheostat located in the rear part of each spotlight. They are electrically powered by the PP2 bus through the CKPT SPOT LTS circuit breaker located on the right console in the rear cockpit.

OXYGEN SYSTEM

A liquid oxygen system (Figure 1-76) is used to provide the normal oxygen supply requirements. The liquid oxygen is converted to a gaseous state in a converter-container tank which has a capacity of 10 liter (2.6 U.S. gal).
The oxygen is made suitable for breathing after passing through a heat exchanger which keeps the

OXYGEN SUPPLY SYSTEM (STAND OX)

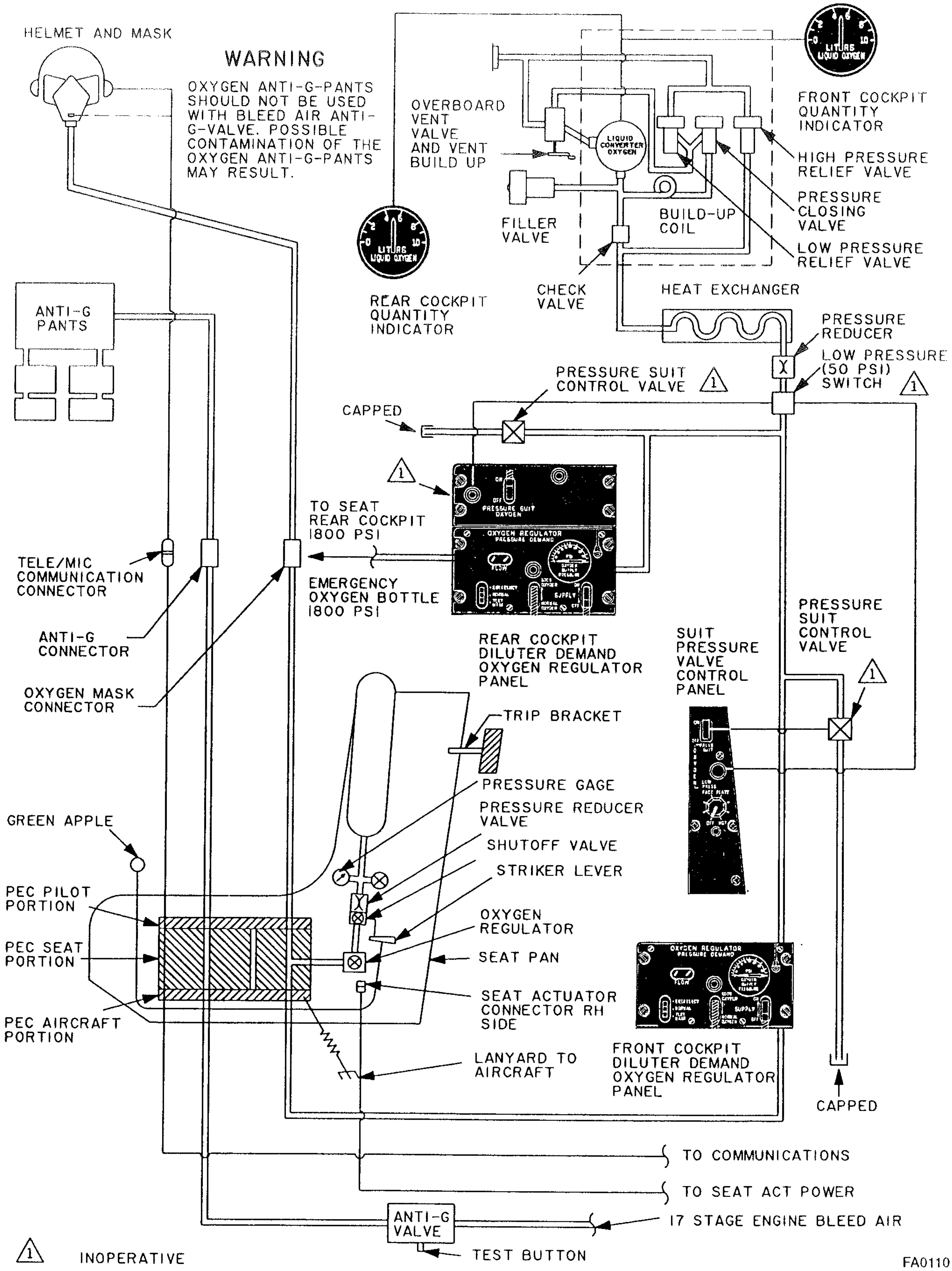


Figure 1-76

oxygen within a few degrees of cockpit ambient temperature. Oxygen is delivered at a pressure of approximately 70 psi.

DILUTER-DEMAND OXYGEN REGULATOR PANEL

The aircraft incorporates a combination pressure-breathing, diluter-demand oxygen regulator (refer to Figure 1-77) on the right console in each cockpit. The panel has three levers, the supply lever, the diluter lever and the emergency lever.
When the diluter lever is at NORMAL and the supply lever at ON, the regulator mixes air with oxygen in varying amounts, according to altitude and makes available a quantity of the mixture each time the pilot inhales. Above 28000 feet, the regulator supplies oxygen to the pilot at continuous positive pressure. The delivery pressure automatically changes with cockpit altitude. Also on the regulator panel are an oxygen pressure gage and flow indicator.

Emergency Lever

The red emergency lever on the oxygen regulator panel (refer to Figure 1-77) should be in the center NORMAL position at all times, unless an oxygen pressure increase is necessary. Moving the lever to the EMERGENCY position provides continuous positive pressure to the mask. When the lever is held at the TEST MASK position, oxygen pressure is provided to test the mask for leakage.

WARNING

WHEN POSITIVE PRESSURES ARE REQUIRED, IT IS MANDATORY THAT THE OXYGEN MASK BE WELL FITTED TO THE FACE. UNLESS SPECIAL PRECAUTIONS ARE TAKEN TO ENSURE THAT NO LEAKAGE EXISTS, CONTINUED USE OF POSITIVE PRESSURE WILL RESULT IN RAPID DEPLETION OF THE OXYGEN SUPPLY. THIS RAPID DEPLETION OF OXYGEN COULD RESULT IN EXTREMELY COLD OXYGEN FLOWING TO THE MASK.

NOTE

100% oxygen is not necessarily provided with the lever in the EMERGENCY position.

Diluter Lever

The diluter (white) lever on the oxygen regulator control panel (refer to Figure 1-77) should be in the NORMAL OXYGEN position for normal oxygen use, or at the 100% OXYGEN position for emergency oxygen use.

SUPPLY Lever

The SUPPLY (green) lever on the oxygen regulator panel (refer to Figure 1-77) has two positions, ON and OFF, and is used to shut off the supply of oxygen at the panel. This lever is safetied to the ON position.

Pressure Gage and FLOW Indicator

The pressure gage and FLOW indicator are on the oxygen regulator control panel (refer to Figure 1-77) in each cockpit. The pressure gage shows oxygen system pressure. The FLOW indicator (blinker) consists of an oblong opening in the face of the regulator panel which shows black (no flow) and white (flow) alternately during the breathing cycle. Continuous black indicates that no air/oxygen is being furnished, and continuous white indicates a leak in the system.

Oxygen Quantity Gage

The liquid oxygen quantity gage is on the right forward panel (refer to Figure 1-77). It measures liquid content of the oxygen converter and is calibrated in liters from 0 to 10. Electrical power for the gage is from the XP4 bus. The rear gage requires dc power also by the PP2 bus.

Oxygen System Low Pressure Warning Light

An oxygen low pressure warning light is located on the suit pressure valve control panel (Figure 1-77). This push-to-test shutter dimming light illuminates whenever the system oxygen pressure decreases below 50 psi.

NOTE

The low pressure warning light also may illuminate under the following conditions:

(1) A high consumption rate combined with a converter serviced to or nearly to the full mark (10 liters).
(2) Under prolonged negative G conditions combined with a converter service under (1) above.

Illumination of the light under these circumstances is not a cause for concern.

STANDARD EMERGENCY OXYGEN SUPPLY SYSTEM

The Stand Ox. (Figure 1-76 and Figure 1-78) consists of an emergency oxygen bottle, pressure gage, pressure reducer valve, oxygen regulator, and a valving mechanism on each seat. The valving mechanism is controlled by a striker lever, which turns on the oxygen during seat ejection by striking a pin inserted in the emergency oxygen trip bracket. A manual release knob (green apple) is located on the front on the PEC and is cable-connected to the striking lever. It may be utilized by the pilot in the event the aircraft oxygen system fails.
The oxygen supply available may be 4 minutes or less, contingent on bottle servicing and pilots oxygen requirements. A lanyard attached to the airplane structure disengages the aircraft oxygen supply system by removing the PEC-aircraft portion upon seat ejection.
The PEC-seat portion is designed to stay with the seat, and when the PEC-aircraft portion is disconnected, spring-loaded valve close off the aircraft oxygen system and anti "G" line ports, allowing the emergency oxygen system to function without leakage.
The PEC-pilot portion stays connected to the PEC-seat portion during descent until seat and pilot separation takes place. At that time, the PEC-pilot portion stays with the pilot, until the pilot elects to disconnect it from the oxygen mask.

OXYGEN SYSTEM PREFLIGHT CHECK

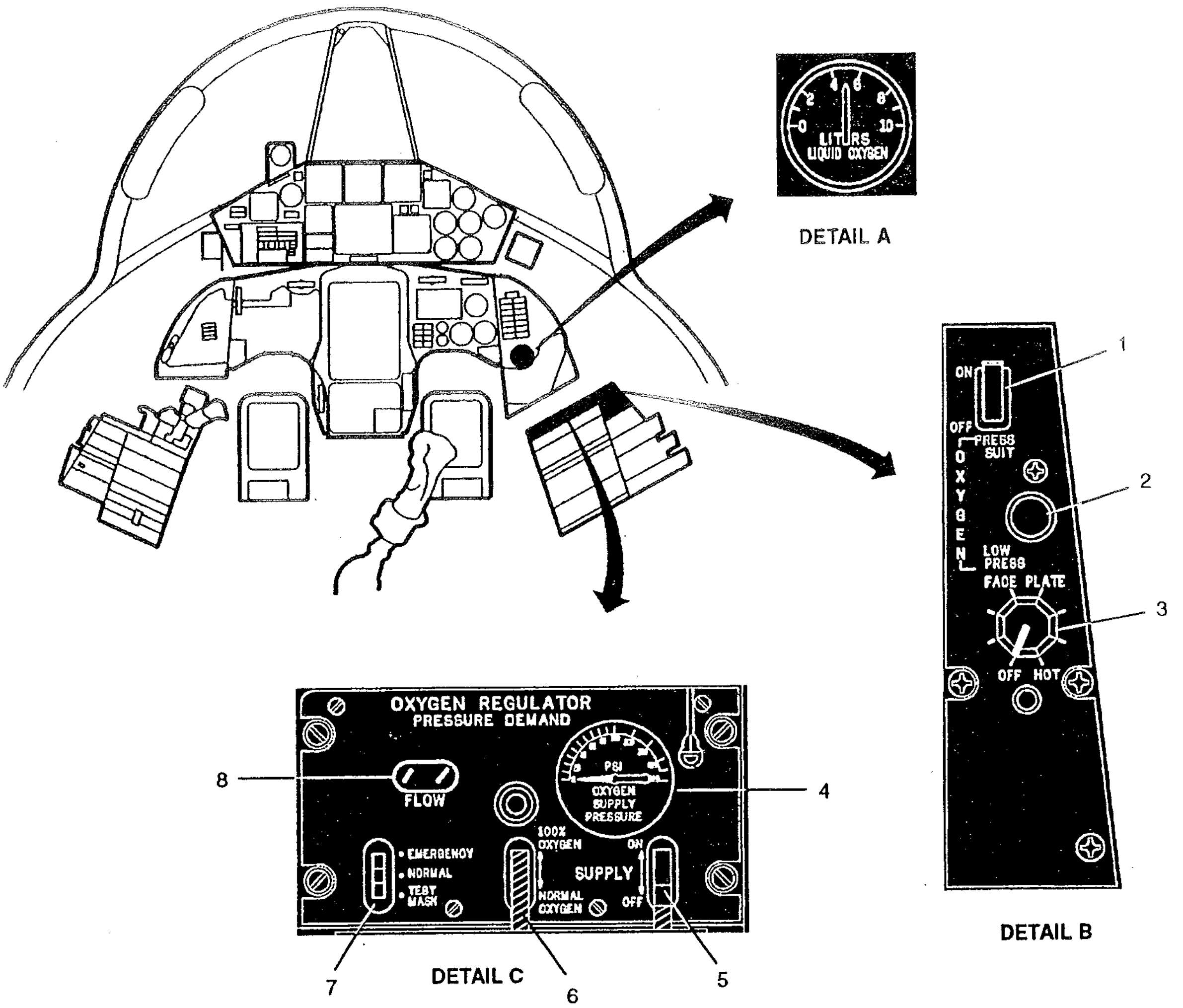
Diluter Demand (Stand Ox)

Before takeoff, perform oxygen system check (PRICE) as follows:

NOTE

- This test procedure is applicable only for an initial preflight check of the system. Inflight or repeated tests made within short periods may produce false or misleading indications.
- The Pressure, Regulator, Indicator. Controls, Emergency (PRICE) oxygen equipment check makes no provision for checking the oxygen mask and protective helmet. Pilots are responsible for ensuring their oxygen masks have been correctly fitted, inspected, cleaned, and repaired by the Base Life Support Section.
- Normal oxygen pressure is 65 to 75 psi as indicated on the control panel and regulator gages when oxygen is being used (flight condition). Normal oxygen pressure when the aircraft is on standby or in preflight condition may vary, depending on the timespan between the filling of the converter and the reading of the gages. For example, if the gage is read approximately 10 minutes after the filling operation has been completed, the gage reads approximately 70 psi. However, if the gage is read several hours after filling, it reads between 100 and 120 psi. This pressure should drop to and remain normal as soon as oxygen is used.
- If an oxygen pressure reading of over 120 psi is observed, an unsatisfactory condition exists and the system should be inspected prior to flight.

OXYGEN PANEL
(FRONT COCKPIT)



A – LIQUID OXYGEN QUANTITY GAGE

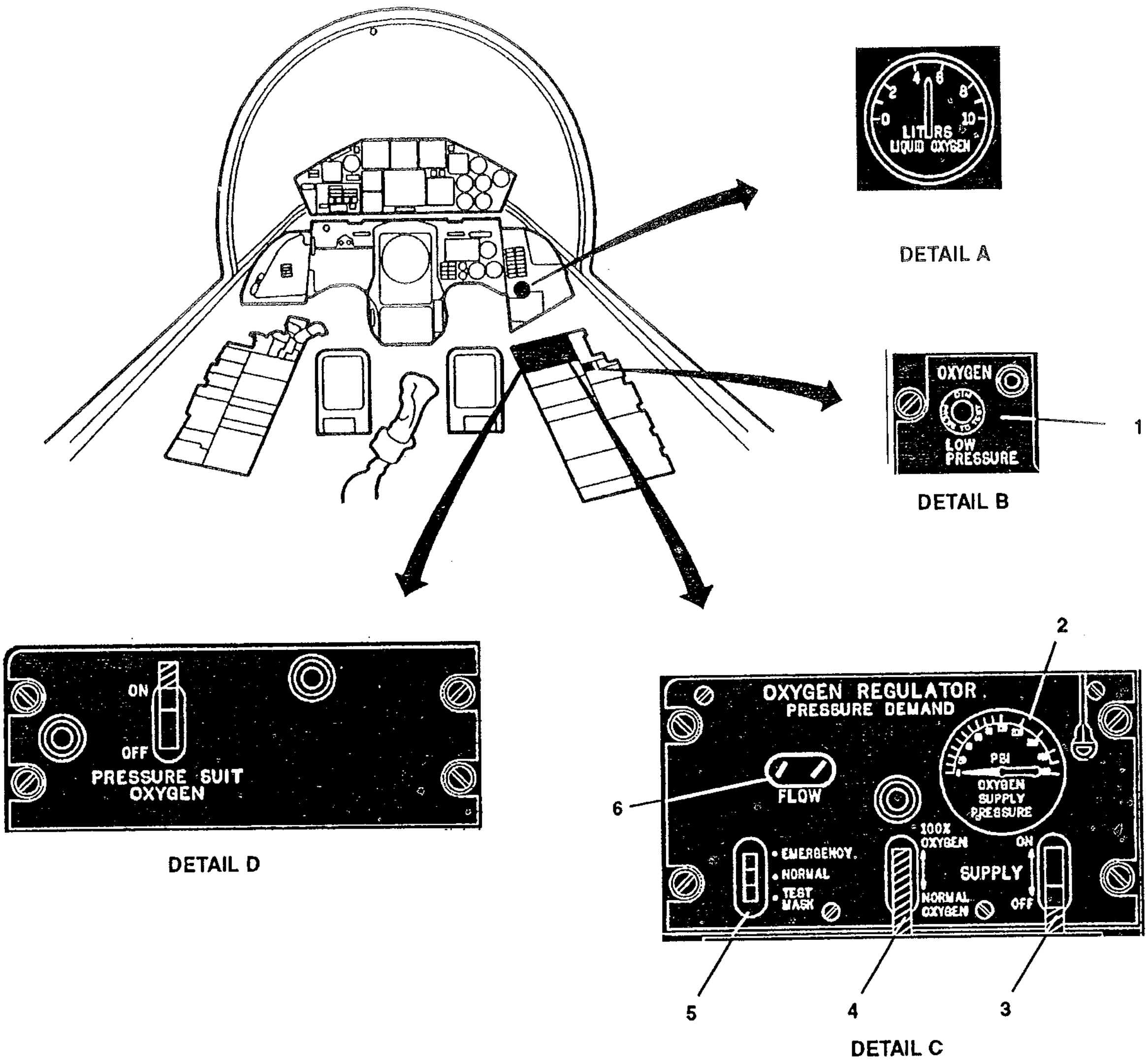
B – RIGHT CONSOLE OXYGEN PANEL
1 PRESS SUIT SWITCH (INOPERATIVE)
2 OXYGEN LOW PRESSURE LAMP
3 FACE PLATE HEAT RHEOSTAT (INOPERATIVE)

C – OXYGEN CONTROL PANEL
4 OXYGEN PRESSURE GAGE
5 OXYGEN SUPPLY LEVER
6 OXYGEN DULUITER LEVER
7 OXYGEN NORMAL/EMERGENCY LEVER
8 OXYGEN FLOW INDICATOR

FA0141

Figure 1-77 (Sheet 1 of 2)

OXYGEN PANEL
(REAR COCKPIT)



A – LIQUID OXYGEN QUANTITY GAGE

B – RIGHT CONSOLE
1 OXYGEN LOW PRESSURE LAMP

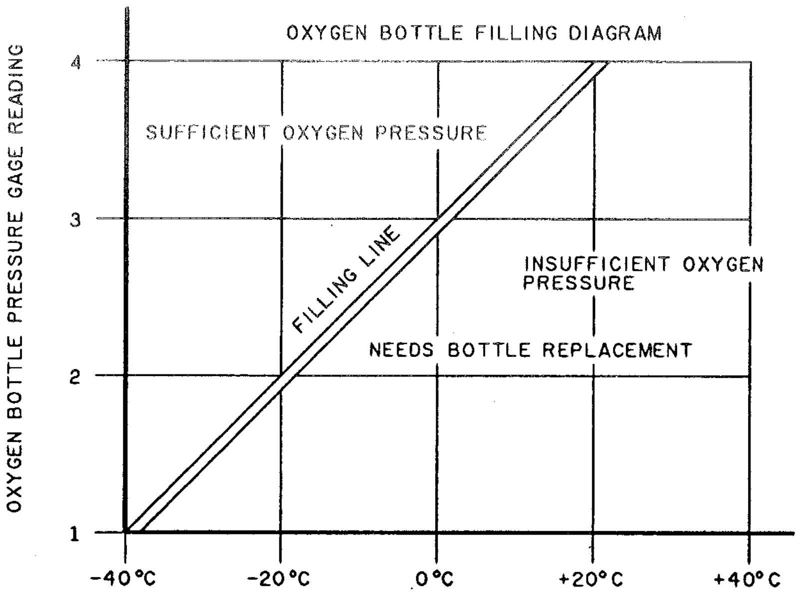
C – OXYGEN CONTROL PANEL
2 OXYGEN PRESSURE GAGE
3 OXYGEN SUPPLY LEVER
4 OXYGEN DILUITER LEVER
5 OXYGEN NORMAL/EMERGENCY LEVER
6 OXYGEN FLOW INDICATOR

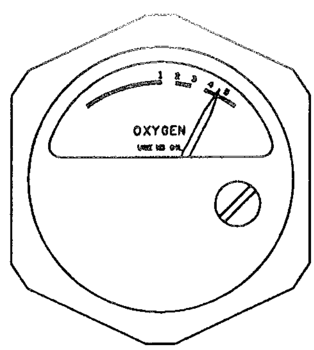
D – PRESSURE SUIT OXYGEN SWITCH (INOPERATIVE)

FA0142

Figure 1-77 (Sheet 2 of 2)

EMERGENCY OXYGEN BOTTLE PRESSURE INDICATION





EXAMPLE:
PRESSURE GAGE INDICATION AT +30°C
AMBIENT TEMPERATURE

NOTE

- THE REQUIRED AMOUNT OF OXYGEN PRESSURE DEPENDS ON AMBIENT AIR TEMPERATURE.
- ENTER THE OXYGEN BOTTLE FILLING DIAGRAM AT THE BOTTOM WITH AMBIENT AIR TEMPERATURE, PROCEED VERTICALLY TO THE FILLING LINE, THEN PROCEED TO THE LEFT TO FIND THE REQUIRED OXYGEN PRESSURE.

FA0102

Figure 1-78

- All connections and checks should be made with the assistance of a personal-equipment man or someone qualified to perform the function.
- Each crewmember must perform his check to ensure proper operation in this cockpit.

(P)

a. Oxygen supply pressure – CHECK 65 TO 120 PSI

1. Low pressure light – OPERATION CHECKED AND LIGHT OUT
2. Oxygen quantity gage – MINIMUM 4 LITERS

(R) (I)

b. Regulator/Indicator – CHECK/SET

1. SUPPLY lever – ON (safetied)

NOTE

If the oxygen system is not to be used and the SUPPLY lever is ON, the regulator will allow positive pressure oxygen flow above 28000 feet cockpit altitude, which will rapidly deplete the oxygen supply.

2. OXYGEN lever – 100%
3. EMERGENCY/NORMAL/TEST MASK lever – EMERGENCY
4. With mask on, breathe normally for a minimum of three cycles; FLOW indicator should show alternately black and white. Hold breath; FLOW indicator should indicate black (no flow). White indicates a leak somewhere in the system and must be corrected before flight.
5. OXYGEN lever – NORMAL (Indicator should remain black; white indicates a leak and shall be corrected before flight)
6. EMERGENCY/NORMAL/TEST MASK lever – NORMAL (Exhale; if breath exhalation is restricted, correct before flight)

(C)

c. Connections – CHECK

1. Regulator hose to regulator connection – SECURE IF EXPOSED
2. Regulator hose – Signs of wear, damage, and missing parts
3. Quick disconnect – 12 to 20 pounds pull to disconnect
4. Survival kit breathing line cap – ON (if not connected to mask to emergency oxygen provisions)
5. Emergency oxygen line to mask connector and anti-G hose to anti-G pants – CONNECTED AND SECURE
6. Check along sides of seat for obstructions

WARNING

PROPER HOOKUP OF PERSONAL LEADS IS ESSENTIAL TO PREVENT ENTANGLEMENT DURING EJECTION. CHECK THAT THE PILOT'S ANTI-G SUIT HOSE AND OXYGEN SUPPLY HOSE WITCH ATTACHED TELE/MIC CABLE IS ROUTED UNDER THE HARNESS WAIST BELT.

(E)

d. Emergency assembly – CHECK

1. Pressure gage – GREEN ARC
2. Green ball handle – SECURE
3. Hose – WEAR AND DAMAGE
4. Connector – SECURE

NORMAL OXYGEN SYSTEM OPERATION

Diluter Demand

a. Oxygen pressure gage – 65-120 psi

b. Liquid oxygen quantity gage – CHECK FOR REQUIRED MINIMUM

c. Diluter demand supply lever – ON

d. Diluter lever – NORMAL OXYGEN

e. Emergency lever – NORMAL

EMERGENCY OXYGEN SYSTEM OPERATION

Diluter Demand

a. Diluter lever – 100% OXYGEN

b. Emergency lever – EMERGENCY

NOTE

Placing regulator lever in 100% OXYGEN position and/or use of emergency lever will rapidly deplete the oxygen supply. Return diluter lever to NORMAL OXYGEN position and lever to NORMAL position when emergency is over.

c. If oxygen regulator becomes inoperative – PULL GREEN APPLE to actuate emergency oxygen supply (oxygen supply for approximately 4 minutes available).

d. Descend to an altitude below 10000 feet

ENGINE BLEED AIR ANTI-G SUIT SYSTEM

On aircraft equipped with MB seat, an engine bleed air anti-G suit system is installed for use with anti-G suit to provide pressurization on critical parts of the pilot's body during flight manoeuvers when positive G force in excess of 1.5 G are encountered.

Engine bleed air from the gun purge duct is routed to an anti-G valve which opens under positive forces in excess of 1.5 G. The amount of pressure applied is regulated and is directly proportional to the applied G force. The anti-G valve is located outboard of the cockpit left console and adjacent to the pilot's seat.

When a G force is applied, a weight moves down, closing the exhaust valve and opening a second stage valve to admit air pressure to the anti-G suit. A relief valve prevents the suit from pressurizing in excess of 9 to 11 psi.

Decreasing G force allows the weight to move up, closing the second stage valve and exhausting the air from the suit through the exhaust valve.

WARNING

- IF THE ANTI-G SUIT FAILS TO DEFLATE AFTER COMPLETION OF A POSITIVE ACCELERATION, DISCONNECT THE ANTI-G SUIT FROM THE PRESSURE SOURCE AT THE CONNECTION TO RELIEVE PRESSURE. IF THE CHECK VALVE IN THE ANTI-G SUIT IS INSTALLED, PLACE THUMB OVER HALF OF CHECK VALVE AND PRESS IN ON THE VALVE. NOTE THAT AIRFLOW MAY BE BLOCKED IF THUMB COVERS MORE THAN HALF OF CHECK VALVE.

- PROPER HOOKUP OF PERSONAL LEADS IS ESSENTIAL TO PREVENT ENTANGLEMENT DURING EJECTION. CHECK THAT THE ANTI-G SUIT HOSE, AND OXYGEN SUPPLY HOSE WITH ATTACHED TELE/MIC CABLE ARE ROUTED UNDER THE HARNESS WAIST BELT.

Anti-G Valve Test Pushbutton

An anti-G valve test pushbutton is provided to test the anti-g system whenever the aircraft bleed air system is pressurized.
The test button is located on the top of the anti-G valve installation on the outboard side of the cockpit left console.
To test the operation of the valve, depress the pushbutton. The valve should open and supply pressure to the anti-G suit connection.

MISCELLANEOUS EQUIPMENT

REAR-VIEW MIRRORS

Two adjustable rear-view mirrors are installed in each cockpit, one on each side of the forward edge of the canopy.

AUXILIARY EQUIPMENT

For information concerning the following auxiliary equipment refer to: Air-Conditioning and Pressurization System, Defogging and Rain-Remover System, Anti-Icing System, Communications and Navigation Equipment, Automatic Pilot, Lighting Equipment, Oxygen System, Air Refueling and Miscellaneous Equipment in this Section.

SERVICING DATA

For general servicing data and location of servicing points refer to servicing diagram (Figure 1-79).

TIRE PRESSURE

WARNING

WHEN SERVICING AIRCRAFT TIRES USE REGULATED AIR PRESSURE NOT TO EXCEED 250 PSI. PERSONNEL SHOULD STAND FORWARD OR AFT OF WHEEL WHEN SERVICING TIRES. OVERPRESSURE MAY CAUSE FAILURE OF TIRE OR WHEEL AND RESULT IN SERIOUS OR FATAL INJURY TO PERSONNEL.

Tire pressure for various aircraft flight configurations are shown in Figure 1-80.
These pressures are based on using 26 × 8.00, 14- or 16-ply tires on main landing gear and 18 × 5,5. 14-ply tires on the nose landing gear.

DRAG CHUTE INSTALLATION

If installation of the drag chute becomes necessary (Figure 1-81), perform the following steps:

a. Release drag chute door and check drag chute mechanism. It is of paramount importance that drag chute hook is properly sequenced before insertion of drag chute riser link. This is accomplished by pulling DRAG CHUTE handle, located in cockpit, to jettison position and then returning it to stowed position. This procedure will assure that hook mechanism is in correct condition for insertion of riser link.
   After following procedure outlined above, drag chute hook will be in open position. To determine if drag chute hook is in proper unlocked condition (see note), use following procedure. Close and fully open the hook with fingers. If hook may be opened against spring tension only, it indicates that hook mechanism is in proper unlocked condition.

SERVICING DIAGRAM

SPECIFICATION		
FUEL	JP-4, JP8, JET A-1 RECOMMENDED, REFER TO FIG. 2-9 AND STRANGE FIELD PROCEDURES	
OIL	MIL-L-7808	NATO-O-148 *
HYDRAULIC FLUID	MIL-H-5606 NATO-H-515	MIL-H-83282 NATO-H-515
RESERVOIR CAPACITIES	NO. 1 – 0.49 US GAL NO. 2 – 1.83 US GAL	
OXYGEN	LIQUID MIL-O-27210	
NITROGEN	MIL-N-6011 GRADE A TYPE 1 (WATER-PUMPED OR DRY AIR)	
WATER	USE DISTILLED WATER OR DEMINERALIZED WATER	

* WHERE NATO-O-148 (MIL-L-7808) IS SPECIFIED AND IS NOT AVAILABLE, NATO-O-149 MAY BE USED OR MIXED WITH O-148 UP TO 50%. IN SUCH A CASE, WHEN O-148 BECOMES AVAILABLE, THE OIL TANK SHOULD BE DRAINED AND FILLED WITH O-148.

STARTING AIR REQUIREMENTS

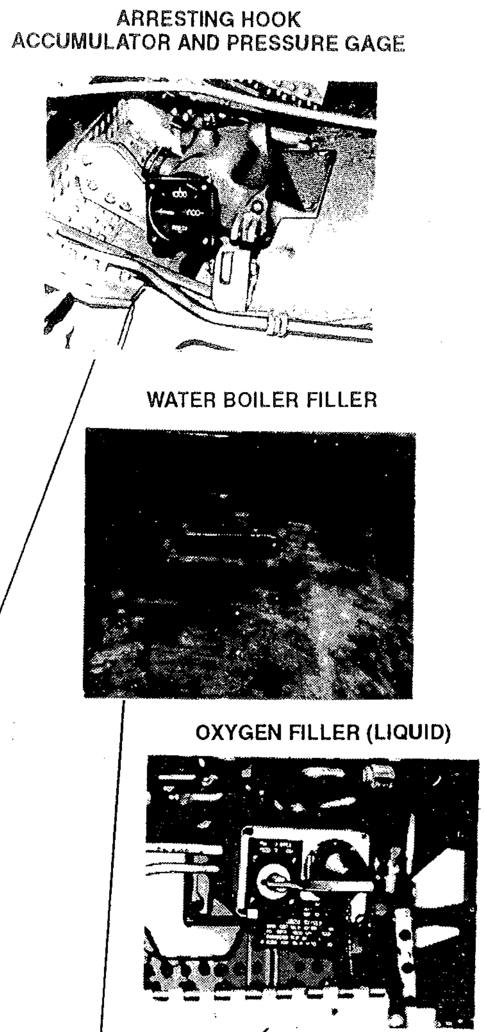
RATED INPUT	350° F AND 45 PSI AT 110 LB/MIN
MAXIMUM PRESSURE INPUT	250° F AND 70 PSI AT 110 LB/MIN

ELECTRICAL REQUIREMENTS

115/200 VOLT, 400HZ, 3-PHASE AC, 28 VDC

OXYGEN REQUIREMENTS

PRESSURE FILLING 30 ÷ 50 PSI



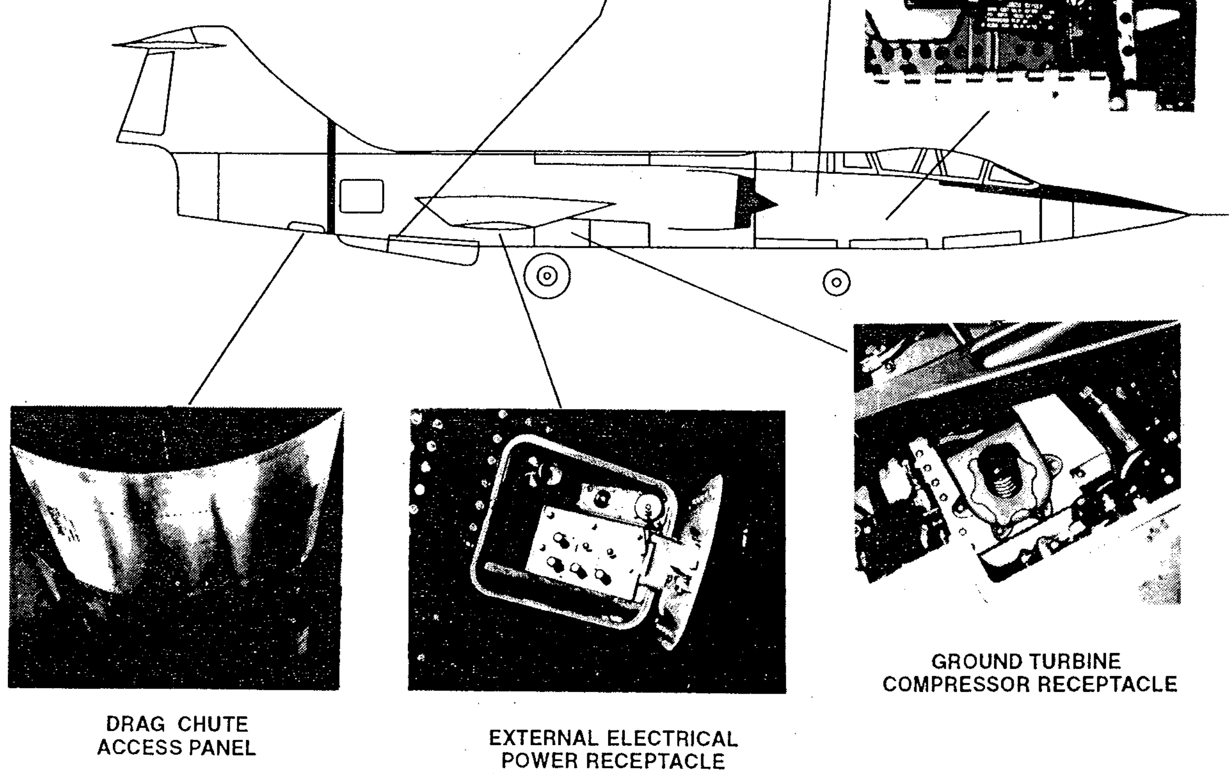


Figure 1-79 (Sheet 1 of 2)

SERVICING DIAGRAM

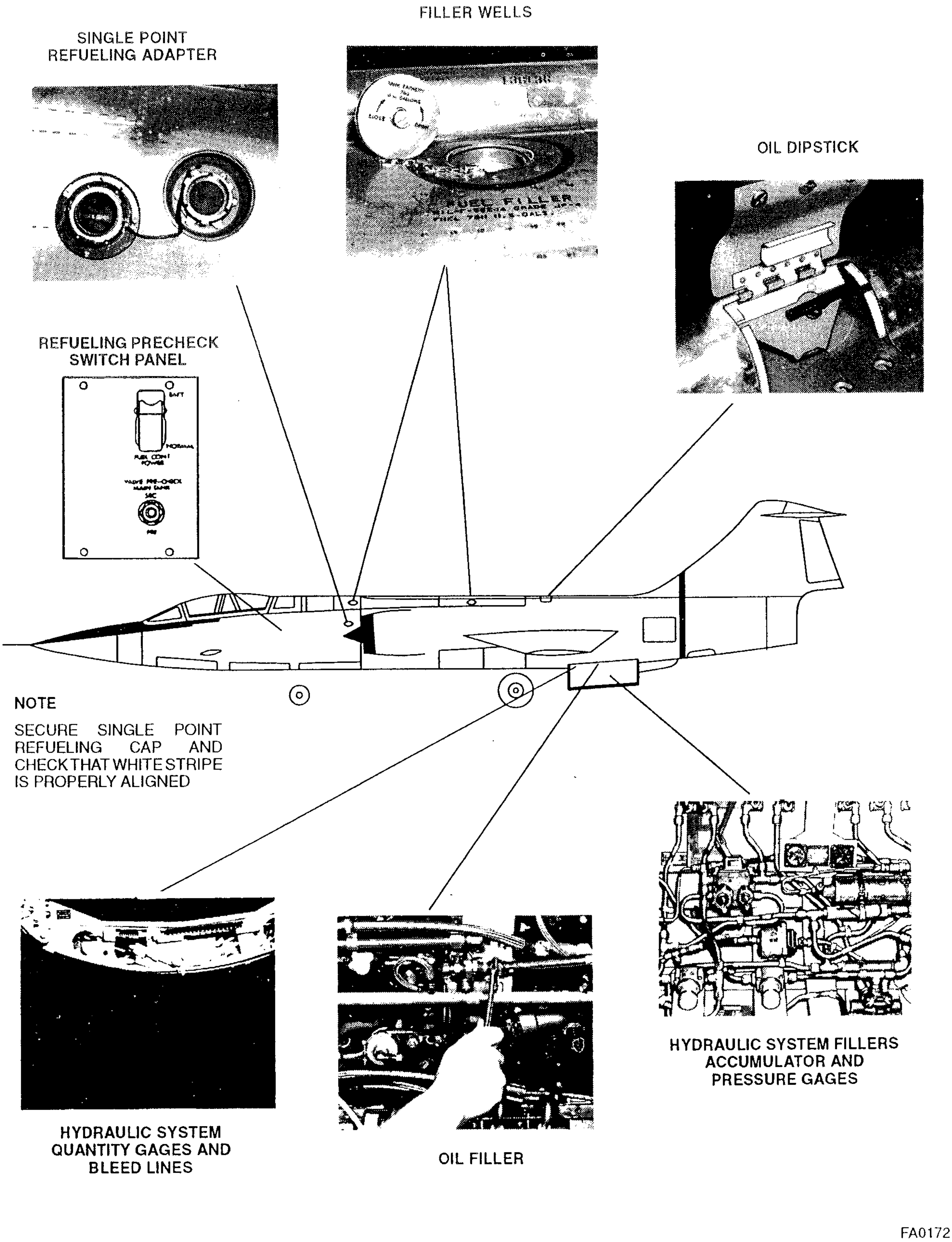


Figure 1-79 (Sheet 2 of 2)

TIRE PRESSURE CHART

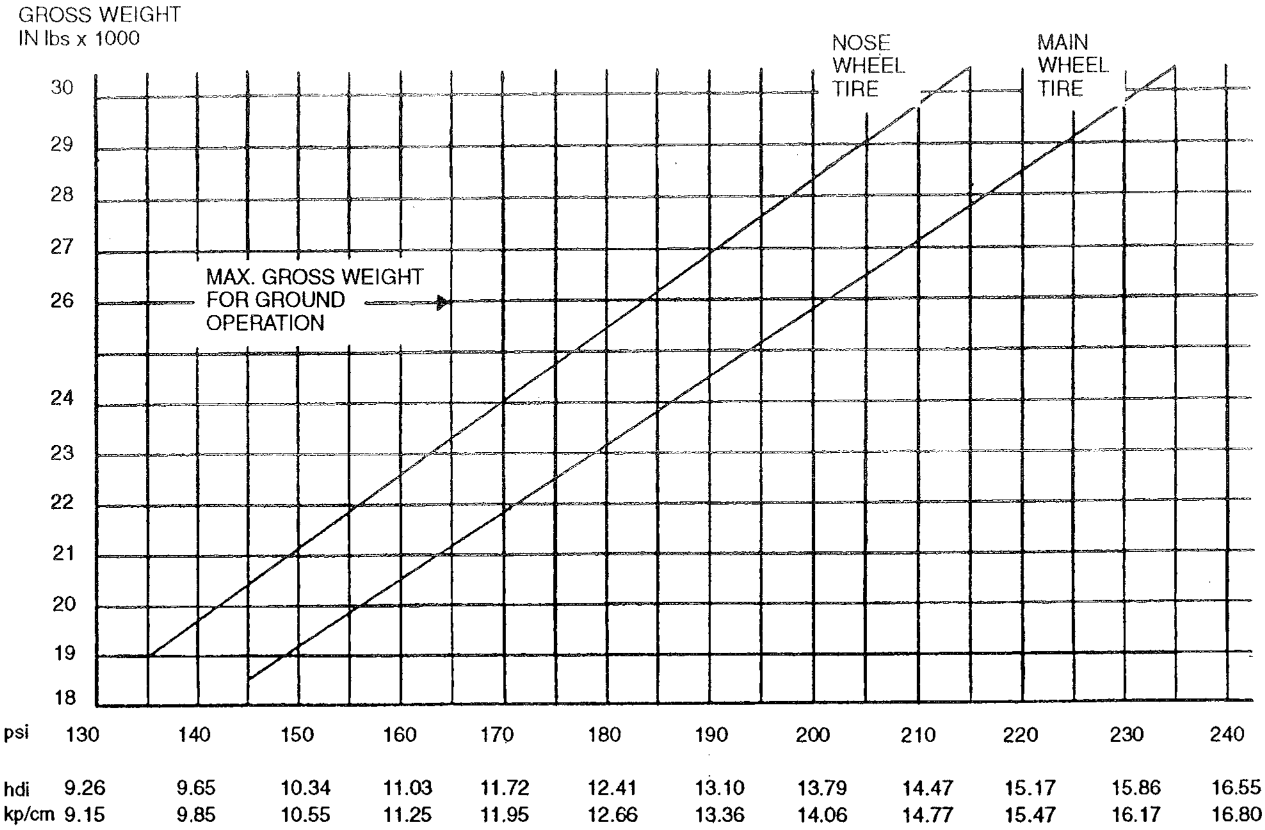


Figure 1-80

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If hook cannot be opened with fingers and is positively locked, it indicates that:

1. Cockpit release handle is not in full stowed position.
2. Cable system is mis-rigged, or
3. Release mechanism is faulty in some respect and must be repaired

NOTE

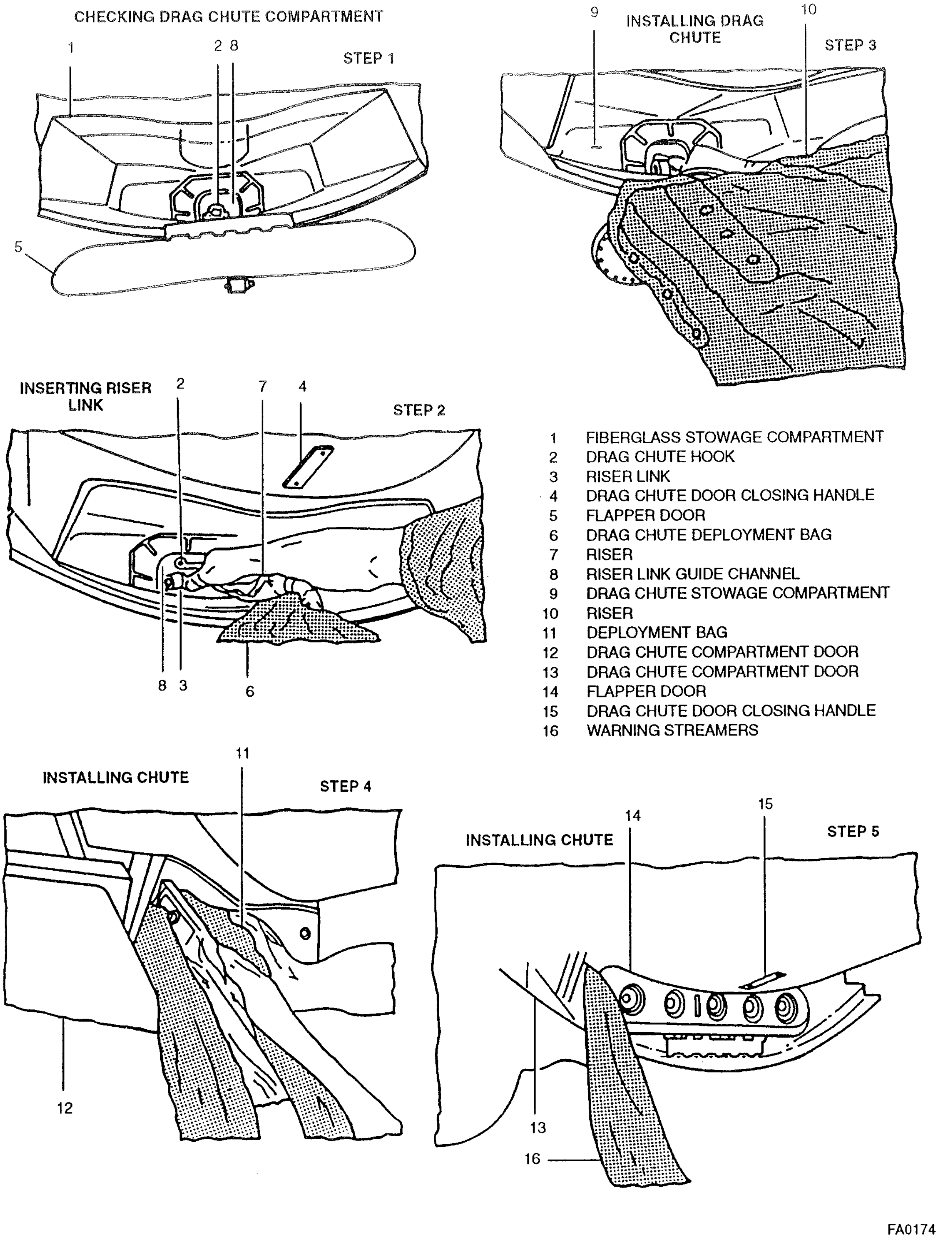
- Position refers to the visible position of the hook.
- Condition refers to the safety feature provided in the hook mechanism to automatically jettison the chute in the event of an inadvertent deployment due to door latch malfunction.
- Due to the lack of leverage and heavy spring tension, opening the hook with the fingers requires considerable manual effort.

b. With printing THIS SIDE UP on uppermost side of bag, and with pilot chute end of drag chute toward rear, arrange riser so that when riser link is installed, riser will not be twisted more that a 90° twist.

Reach into drag chute stowage compartment and insert riser link through riser link guide channel, into mouth of chute hook. Engage riser links, by closing hook over riser link.

c. Lay riser into drag chute stowage compartment, so as to make one condition as shown.

DRAG CHUTE INSTALLATION



1 FIBERGLASS STOWAGE COMPARTMENT
2 DRAG CHUTE HOOK
3 RISER LINK
4 DRAG CHUTE DOOR CLOSING HANDLE
5 FLAPPER DOOR
6 DRAG CHUTE DEPLOYMENT BAG
7 RISER
8 RISER LINK GUIDE CHANNEL
9 DRAG CHUTE STOWAGE COMPARTMENT
10 RISER
11 DEPLOYMENT BAG
12 DRAG CHUTE COMPARTMENT DOOR
13 DRAG CHUTE COMPARTMENT DOOR
14 FLAPPER DOOR
15 DRAG CHUTE DOOR CLOSING HANDLE
16 WARNING STREAMERS

Figure 1-81

CAUTION

MAKE CERTAIN THAT THE BREAKAWAY CORD THROUGH THE RISER TIE DOWN LOOP IS STILL INTACT.

d. Start deployment bag into stowage compartment keeping printing THIS SIDE UP on uppermost side, and keeping riser between forward end of bag and forward end of drag chute cavity. Gently rock bag back and forth and ease it into drag chute compartment. Arrange nylon retention loops so that they are free and stretched toward rear, and insert bag as far forward into compartment as possible.

CAUTION

MAKE CERTAIN THAT THE ARROW ON THE PILOT CHUTE COMPARTMENT FLAP, WHICH MAY BE SEEN AFTER THE BAG IS INSERTED, IS POINTING UP.

e. Arrange warning streamers, of two pins, so that they will not be caught in flapper door and so that two pins are readily attainable for pulling. Release two fasteners which secure drag chute door closing handle to fuselage. Close flapper door carefully. Exert a downward pull on drag chute door closing handle until first click of ratcheting mechanism is felt. (The trailing edge of the main drag chute compartment door moves up approximately one inch with each click.) Ratchet handle a few times until flapper door is engaged with fuselage structure.

CAUTION

THE CABLE MUST COMPLETELY SLACKEN ON EACH RETURN STROKE TO INSURE PROPER RE-ENGAGEMENT OF THE RATCHETING MECHANISM. THE DOOR SHOULD BE CLOSED SLOWLY SO THAT THE FINGERS HOLDING THE FLAPPER DOOR DO NOT GET SQUEEZED BETWEEN THE AFT EDGE OF THE DOOR AND THE FUSELAGE STRUCTURE.

f. Reinsert two fasteners to secure drag chute door closing handle to fuselage.

SECTION II

NORMAL PROCEDURES

TABLE OF CONTENTS

PREPARATION FOR FLIGHT

FLIGHT RESTRICTIONS

Refer to Section V "Operating Limitations" for detailed aircraft and engine limitations.

FLIGHT PLANNING

Refer to the Appendix "Performance Data" to determine the fuel quantity, engine settings and airspeeds required to complete the mission.

EMERGENCY PROCEDURES

Refer to Section III for Emergency Procedures.

TAKEOFF AND LANDING DATA CARDS

Refer to the Appendix "Performance Data" for information necessary to fill-out the takeoff and landing data cards.

MASS AND BALANCE

For maximum gross mass takeoff and landing limitations refer to Section V "Operating Limitations". For mass and balance information refer also to "Manuale di Dati di Peso e Centramento" AER.1F-104(T)GM-5.

PREFLIGHT CHECK

BEFORE EXTERIOR INSPECTION

1. Aircraft engineering status and servicing – Check
2. Lower firing handle swivel guard – UP
3. Radar switches – OFF
4. LDG GEAR – DOWN
5. IN/CDU (front cockpit) – Check DTM inserted
6. Check the following ejection seat safety pins installed:
   - Main gun safety pin
   - Safety pin upper firing handle
   - Safety pin of canopy jettison initiator
   - Safety pin of emergency canopy jettison initiator

NOTE

Cooling is provided by a rear electronic compartment blower motor. Therefore, cooling equipment is not needed.

EXTERIOR INSPECTION

Perform exterior inspection as outlined in Figure 2-1.

WARNING

EACH TIP TANK HAS TWO FILLERS, ONE FOR EACH COMPARTMENT. IF A FLIGHT IS NECESSARY WITH PARTIALLY FILLED TIP TANKS, THE FORWARD COMPARTMENT SHALL BE FILLED FIRST. FILLING THE REAR COMPARTMENT FIRST WOULD RESULT IN A TIP TANK REAR CENTER OF GRAVITY AND THE POSSIBILITY OF TIP TANK FLUTTER. IF PARTIAL REFUELING HAS BEEN ACCOMPLISHED FOR THE MISSION, OR IF INCOMPLETE REFUELING IS SUSPECTED DUE TO TIP TANK FUEL GAGE INDICATIONS ON COCKPIT PREFLIGHT, VISUALLY CHECK THAT THE TIP TANKS HAVE BEEN REFUELED IN ACCORDANCE WITH DECALED INSTRUCTIONS ON THE TANK. IF PARTIAL REFUELING IS REQUIRED, CHECK TO MAKE SURE THAT THE FORWARD COMPARTMENT HAS BEEN REFUELED TO WITHIN 3 INCHES FROM THE TOP IF THERE IS ANY FUEL IN THE REAR COMPARTMENT; OTHERWISE, DO NOT FLY THE AIRCRAFT. IF COCKPIT TIP TANK GAGES HAVE BEEN CHECKED, VISUAL CHECK OF FULLY SERVICED TIP TANKS IS NOT REQUIRED.

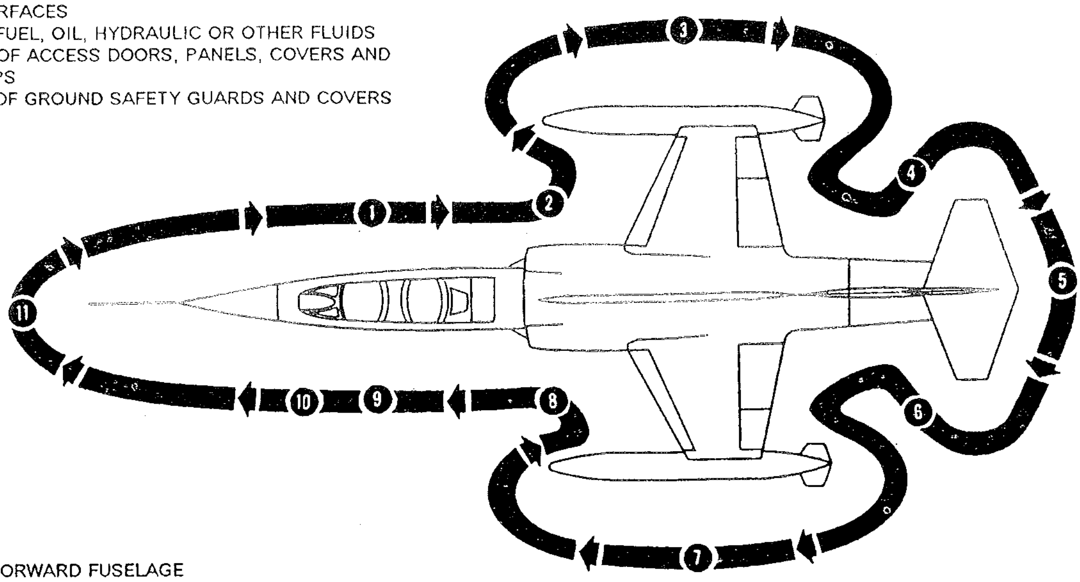
BEFORE ENTERING COCKPIT

1. Canopy – Check for cracks, cleanliness and distortion
2. Safety pins upper firing handle – Check inserted
3. Lower firing handle swivel guard – Check up
4. PEC connecting strap – Check for proper attachment
5. Scissor shackle – Closed and locked
6. Drogue gun safety pin – Removed

EXTERIOR INSPECTION

THE FLIGHT CREW EXTERIOR INSPECTION PROCEDURES ARE BASED ON MAINTENANCE PERSONNEL HAVING COMPLETED ALL POSTFLIGHT AND PREFLIGHT REQUIREMENTS OUTLINED IN THE AER.1F-104A-6; THEREFORE, DUPLICATE INSPECTIONS BY THE FLIGHT CREW HAVE BEEN ELIMINATED, EXCEPT FOR CERTAIN ITEMS REQUIRED FOR FLIGHT SAFETY. THE FLIGHT CREW INSPECTION IS TO CHECK THE AIRCRAFT FOR GENERAL CONDITION AND SHOULD FOLLOW THE PATH SHOWN BELOW. IF AIRCRAFT PREFLIGHT IS ACCOMPLISHED AT A STRANGE FIELD, REFER TO THE DETAILED AIRCRAFT PREFLIGHT UNDER STRANGE FIELD PROCEDURES, CONTAINED IN THIS SECTION.

- CRACKS, DISTORTIONS, LOOSE FASTENERS OR DAMAGE ON ALL SURFACES
- LEAKS OF FUEL, OIL, HYDRAULIC OR OTHER FLUIDS
- SECURITY OF ACCESS DOORS, PANELS, COVERS AND FILLER CAPS
- REMOVAL OF GROUND SAFETY GUARDS AND COVERS



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1. RIGHT FORWARD FUSELAGE
   a. Canopy external locking lever in detent
   b. Electronics compartment circuit breakers in and cover secure, access door closed
   c. Electrical load center circuit breakers – Check
   d. Fuel shut-off closed valve test light – Press, no light
   e. RAT door secure
   f. Air conditioning ram air, no debris
   g. Engine intake duct unobstructed
   h. Engine intake duct inspection door secure
   i. Navigation lights undamaged
2. RIGHT MAIN LANDING GEAR
   a. Landing gear door uplocks, cocked
   b. Ground safety pins removed
   c. Landing gear dump valve – Safetied
   d. Dowlocks in place (knob on drag strut down)
   e. Liquid spring extended according to load (min. 1.3 max 2.4 inches)
   f. Landing lights secure and undamaged
   g. Check for clearance between wheel and tie rod
   h. Wheel brake lines secure, no leakage, self adjuster exposed not less than 1/4 inch
   i. Tires – Inflation, condition
   j. Wheel chocks in place
3. RIGHT WING
   a. Leading edge flap and tip condition
   b. Attachment of external stores secure, general condition
   c. Visually check tip and pylon tank caps, secure; if installed
   d. Pylon and/or tip store pin and flag installed
   e. Aileron and trailing edge flap distortions and conditions
   f. Wing surface condition
4. RIGHT REAR FUSELAGE
   a. Speed brake condition, no leakage
   b. Navigation lights undamaged
   c. Ventral fin condition
5. EMPENNAGE
   a. Vertical and horizontal stabilizer condition
   b. Exhaust nozzle flap linkages and segments secure, no cracks, distortions or oil leaks
   c. Afterburner spray bars, flameholder and liners condition, no cracks or distortions
6. LEFT REAR FUSELAGE
   a. Navigation lights undamaged
   b. Stabilizer servo drain, no leaks
   c. Speed brake condition, no leakage
   d. Drag chute installed, cable connected
   e. Arrestor hook safety look – Removed (if installed)
   f. Hydraulic system accumulator pressures 1000 (±25) psi
   g. Hook accumulator – 740 (±40)
   h. Hydraulic system manual selector valve safetied in number 2 position
   i. Hydraulic system quantity gauges at proper level
   j. Heat exchanger for cracks or discoloration check
   k. BLC Ducts secure
   l. Blow out panel secure
7. LEFT WING
   a. Same as right wing
8. LEFT MAIN LANDING GEAR
   a. Same as right main landing gear
   b. Manual fuel shutoff switch check
   c. Ground-air safety switch clean and undamaged
9. LEFT FORWARD FUSELAGE
   a. Fuel filler caps secure
   b. Engine oil dipstick – Cover secure and flag visible
   c. Navigation light undamaged
   d. Engine intake duct inspection door secure
   e. Engine intake duct unobstructed
   f. Single point filler cap secure and aligned
   g. Refueling switch panel, cover secure
10. NOSE GEAR
    a. Ground safety pin removed
    b. Scissors properly connected
    c. Downlock fully engaged in slot
    d. Taxi light secure and unbroken
    e. Shock strut extended 2 inches minimum (check tape)
    f. Tire – Inflation, condition
11. NOSE SECTION
    a. Pitch sensor vanes free to move
    b. Check windshield thermal sensor and proper installation
    c. Radome latches secure
    d. Infra-red sight window undamaged (if applicable)
    e. Pitot tube – Secure, cover removed, openings clean
    f. Temperature probe, guards removed, opening clean

Figure 2-1

7. Emergency oxygen safety pin – Removed
8. Emergency oxygen pressure – Check
9. Manual override handle linkages (two) – Connected
   Safety pin – Removed

WARNING

- DO NOT STOW ANY ITEMS ON OR UNDER THE EJECTION SEAT.
- IF ANY SAFETY WIRE OR LEAD SEAL IS BROKEN, DO NOT FLY AIRCRAFT UNTIL CLEARED BY MAINTENANCE PERSONNEL.

REAR COCKPIT CHECK (SOLO FLIGHTS)

(The following rear cockpit inspections shall be made before solo flights).

1. Seat secured properly – Check
2. Solo flight apron on rear seat – Installed
3. Circuit breakers – IN
4. STABILITY CONT switches (ROLL, PITCH, YAW) – ON, guarded
5. STICK TRIM-AUX TRIM selector switch – STICK TRIM, guarded
6. FUEL SHUT-OFF switch – ON, guarded
7. RADAR – OFF
8. LANDING LIGHTS and TAXI LIGHT switch – OFF
9. ANTI-SKID switch – ON
10. DRAG CHUTE handle – Stowed
11. MAN LDG GEAR release handle – Stowed
12. EMERGENCY STORES CUT-OUT switch – ON, guarded
13. CANOPY JETTISON handle – Stowed
14. RAM AIR TURBINE handle – Stowed
15. STORM LIGHTS switch – OFF
16. Generator switches – FWD

WARNING

THE REAR GENERATOR SWITCHES MUST BE LEFT ON THE FWD POSITION. THIS IS THE ONLY POSITION THAT ALLOWS OPERATIONS OF THE FRONT COCKPIT GENERATOR SWITCHES.

17. Fresh air scoop lever – Closed. Lever last aft detent

NOTE

Press the button on the lever and make sure lever is in last detent.

18. Oxygen supply lever – ON (safetied)
19. TACAN control panel – As required
20. UHF command radio – As required
21. PITCH SENSOR PITOT HEAT switch – OFF
22. INTERIOR LIGHTS rheostats – OFF
23. Aft canopy – Locked. Visually check the three locking hooks in place and that the canopy handle is rotated fully aft. If canopy is not locked, rotate the canopy external locking lever through its entire arc of travel to an absolute stop above the black line and check in detent.

INTERIOR CHECK

(For Dual Flights items * apply to rear cockpit only)

1. Seat – Adjust

CAUTION

DO NOT OPERATE SEAT ADJUSTMENT MECHANISM FOR MORE THAN 30 SEC WITHIN 10 MINUTES OF TIME.

2. Rudder pedals — Adjust
3. Leg lines — Attach as follows (Figure 2-2):
   a. Route left leg line from inside to outside through lower D-ring, then upward and from outside to inside through upper D-ring of left leg and press into right lock
   b. Route right leg line from inside to outside through lower D-ring, then upward and from outside to inside through upper D-ring of right leg and press into left lock, so that leg lines are crossed before connected to the locks
4. Survival pack lowering line — Connect to life vest or survival pack strap

NOTE

Lowering line must be routed under the left thigh.

5. PEC-pilot portion — Secured
6. Static line for automatic activation of personal locator beacon — Connect (if applicable)
7. Vest emergency radio lanyard — Connect to seat clip (if applicable)
8. Personal leads — Connect. Route the pilot's anti-G suit hose and the oxygen supply hose with attached tele/mic cable under the harness waist belt

WARNING

ALL PERSONAL LEADS MUST BE ROUTED AS DESCRIBED ABOVE IN ORDER TO PREVENT ENTANGLEMENT DURING EJECTION AND/OR INJURY UPON EJECTION.

9. Combined harness — Fasten
   a. Shoulder harness — Locked
   b. Quick release box in position "DON" — Check
   c. Metal lug of negative-g-strap — Insert into quick release box

WARNING

DO NOT ROUTE NEGATIVE-G-STRAP THROUGH THE LOWER FIRING HANDLE.

   d. Lugs of blue lap strap — Connect to quick release box on top of metal lug for negative-g-strap and tighten all straps
   e. Route leg loops through rings on lap straps accomplishing a quarter of a turn, and feed lug of corresponding shoulder strap through end of leg loop into quick release box
   f. Rotate quick release box to "LOCKED" position, insert safety clip and tighten harness

WARNING

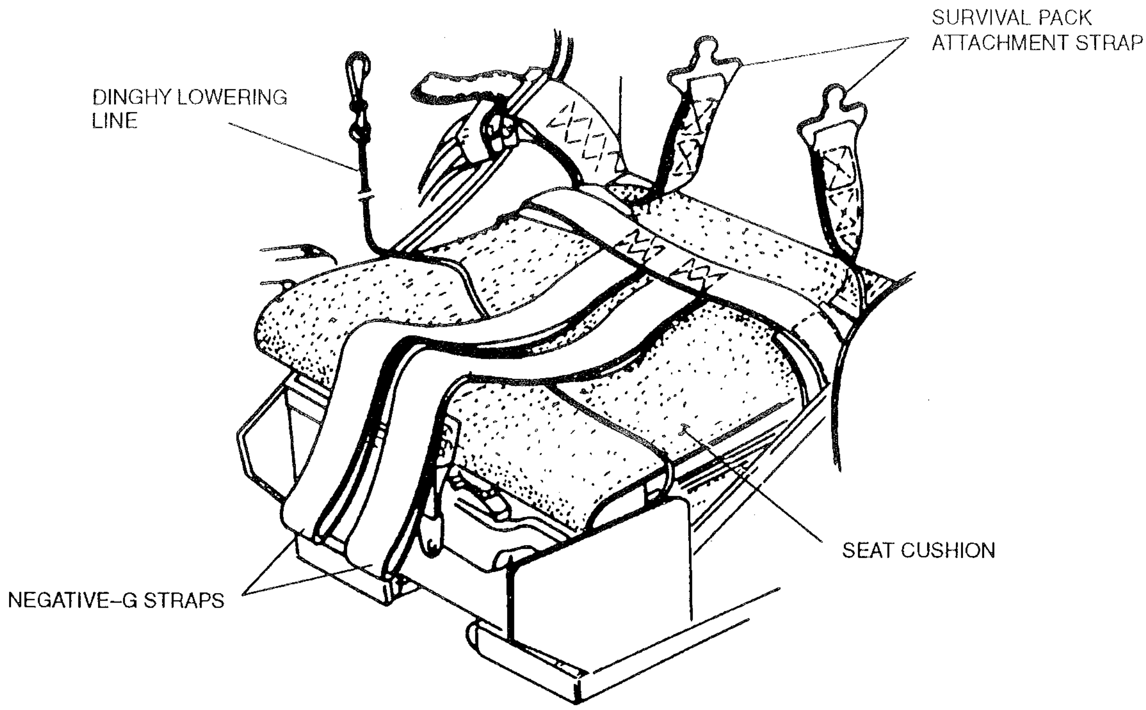
TO AVOID INJURIES UPON EJECTION IT IS IMPERATIVE TO TIGHTEN COMPLETE HARNESS AS TIGHT AS TOLERABLE AND YET COMFORTABLE.

NOTE

After adjustment of complete harness and seat, lean forward and check that rear belts of harness are free and not restricted. Also, check that chute pack restraining strap routing is tight between chute pack and seat pan.

10. Upper firing handle can be reached properly — Check
11. Shoulder harness go-forward lever — Check
12. Emergency UHF radio switch — OFF (test emergency UHF if required, adjust volume as desired)
13. ANTENNA SELECT switch — As desired
14. APC CUTOUT switch — ON, guarded
15. Left console circuit breakers — In, AIL & RUD LIMITER cover down
16. UHF command radio — As required

LEG LINE ATTACHMENT



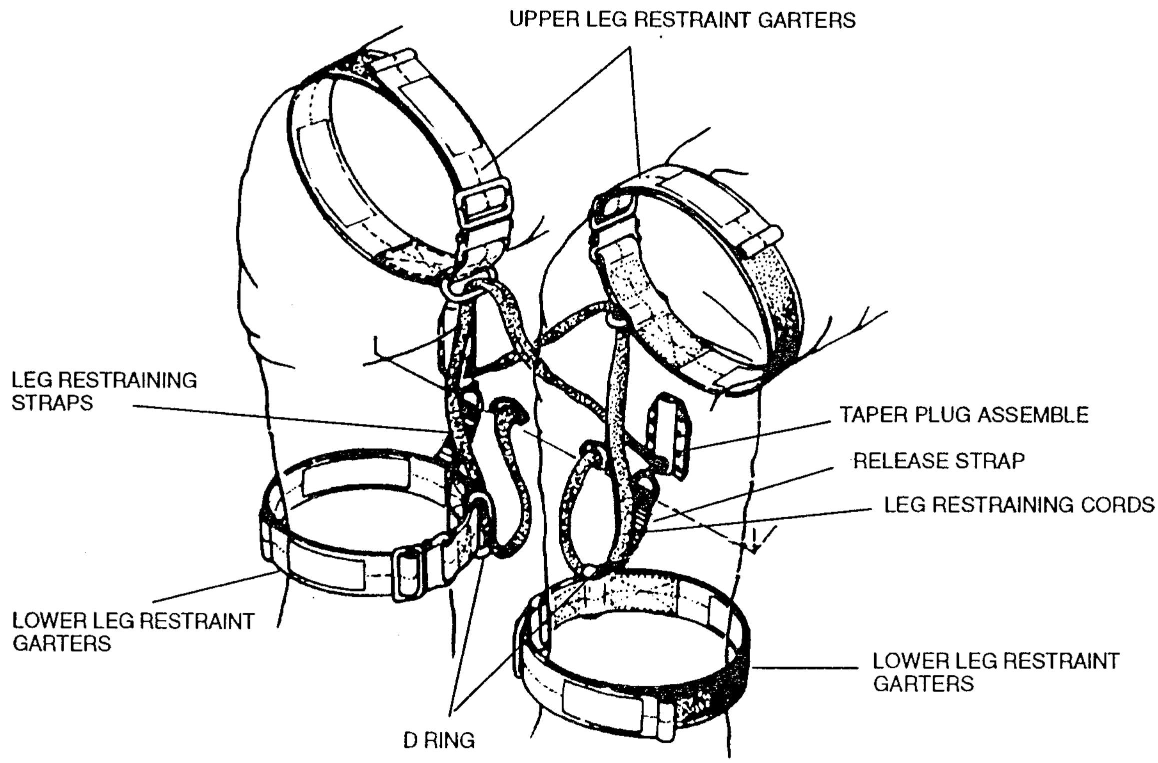


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Figure 2-2

17. STABILITY CONT switches (ROLL, PITCH, YAW) – ON, guarded
18. FUEL SHUT-OFF switch – ON, guarded
19. REFUEL switch – OFF, guarded
20. EXTERNAL switch – BOTH
21. RADAR – OFF
22. STICK TRIM-AUX TRIM selector switch – STICK TRIM, guarded

CAUTION

DO NOT USE PRIMARY OR AUXILIARY TRIM CONTROL WITHOUT HYDRAULIC PRESSURE AS THIS MAY DAMAGE THE TRIM MOTOR.

23. Wing flaps – UP
24. Throttle – Check OFF
25. Speed brakes – IN
26. External power unit – Connected and on

NOTE

Connecting the external power unit will cause the CAUTION light and the following warning lights to be illuminated:
FUEL BOOST PUMP FAILS (if circuit breakers are out)
FIXED FREQ. OUT
GENERATOR NO. 1 OUT
GENERATOR NO. 2 OUT
HYDRAULIC SYSTEM OUT
AUTO PITCH CONT OUT
ENGINE OIL LEVEL LOW
CANOPY UNSAFE
INERTIAL NAV FAULT.

27. Control transfer panel – Front cockpit selection
28. TACAN – Set to REC, channel and mode
29. TCN/SEL mode selector – Press, check SEL caption lit
30. HSI – Check flags out of view
31. C-2G COMPASS function selector switch – MAG and Synchronize
32. IFF master switch – OFF
33. HSI – Check conformity between aircraft and HSI headings
34. TCN/SEL mode selector – Press, check SEL caption extinguished

NOTE

If IN alignment is required before starting engine perform steps 35., 36. and 37. If not, perform steps 3., 4. and 5. in the "Ground Operation - After Start Check" contained in this Section.

35. IN – ALN function, GC or STO mode (refer to "INS Alignment Procedures" contained in this Section). Check INERTIAL NAV FAULT warning light extinguished
36. IN/CDU self test – Check "CDU OK" displayed

NOTE

Abort mission if "CDU FAIL" indication is displayed.

37. IN/CDU – Select and insert initial position, if GC selected
38. Landing gear lever unsafe warning light – Out
39. LG INDICATORS – Lit
40. LANDING LIGHT and TAXI LIGHT switch – OFF
41. ANTI-SKID switch – ON
42. EXT STORES JETTISON button – Check integrity
43. DRAG CHUTE handle – Stowed
44. Hook down light – Press to test
45. LG UNSAFE warning light – Out
46. AIL AND RUD UNLIMITED warning light – Lit
47. Accelerometer – Reset
48. CANOPY UNSAFE warning light and sound check – Move throttle to MILITARY, check

warning light lit and audio warning. Press CANOPY WARNING SOUND CUT-OFF, check audio warning out. Retard throttle to OFF check CANOPY UNSAFE warning light extinguished

49. Airspeed marker – Set as desired

50. Attitude indicator – Check, OFF warning flag in view

51. Standby attitude indicator – Check, cage and quickly release

52. HSI – Check OFF and NAV flags in view

53. Altimeter – ELECT and set

NOTE

The appearance of the PNEU flag, when altimeter is selected in ELECT mode, will indicate a power interruption or a failure in the altimeter.

54. Vertical velocity indicator – Check

55. Clock – Check

56. EMERGENCY NOZZLE CLOSURE handle – Stowed

57. RAM AIR TURBINE handle – Stowed

58. CANOPY JETTISON handle – Stowed

59. RADAR day/night visors – Check for correct locked position

60. External STORES RELEASE selector switch – SAFE

61.* EMERGENCY STORES CUT-OUT switch – OFF, guarded

62. UHF channel frequency indicator – Check

63. MAN LDG GEAR handle – Stowed

64. EXT FUEL QTY IND SEL switch – Check fuel quantity

65. STORM LIGHTS switch – OFF

66. FUEL S/O OPEN VALVE TEST – Press to test, check light lit

67. CANOPY DEFOGGER – As required

68. FUEL QTY/WARNING LIGHTS TEST switch:

    a. Up. Check internal fuel quantity, indication decreasing

    b. Down. Check warning lights and audio warning

69. Generator switches

    a. Front cockpit – ON
    b. Rear cockpit – FWD

WARNING

THE REAR COCKPIT GENERATOR SWITCHES SHALL BE LEFT IN THE FWD POSITION. THIS IS THE ONLY POSITION THAT ALLOWS OPERATION OF THE FRONT COCKPIT GENERATOR SWITCHES.

70. Oxygen system – PRICE check (Refer to Oxygen System Preflight Check in Section I)

    (P)
    a. Pressure – 65 to 120 psi
        1) Low pressure light – Check and light out
        2) Quantity – Minimum 4 liters

    (R)
    b. Regulator/Indicator – Check/set

    (I)
        1) Supply lever – On (safetied)
        2) Oxygen lever – 100%
        3) Emergency/normal/test mask lever – Emergency
        4) With mask on, breathe normally for three cycles minimum. FLOW shall show alternately black and white. Hold breath, FLOW shall remain black
        5) Oxygen lever – Normal (FLOW shall remain black)
        6) Emergency/normal/test mask lever – Normal (exhalation shall not be restricted)

    (C)
    c. Connections – Check (for damage and security)

    (E)
    d. Emergency – Green arc (check before entering cockpit)

71. Fresh air scoop lever – Closed. Lever in last aft detent

CAUTION

KEEP THE FRESH AIR SCOOP LEVER IN CLOSED POSITION DURING THE PREFLIGHT CHECK AND ALL GROUND OPERATION. THIS WILL PROVIDE SUFFICIENT COOLING AIR FOR THE ELECTRONIC EQUIPMENT. IF THE FRESH-AIR SCOOP IS OPENED ON THE GROUND, THE SUPPLY OF COOLING AIR TO THE ELECTRONICS COMPARTMENT IS SHUT OFF AND THE ELECTRONIC EQUIPMENT MAY OVERHEAT.

72. TACAN – As required
73. IFF – As required
74. COCKPIT TEMP mode selector switch – AUTO
75. COCKPIT TEMP rheostat – As required
76. FLYING SUIT switch – As required
77. PITOT-PITCH TEMP PROBE HEATERS switch – OFF
78.* PITCH SENSOR PITOT HEAT switch – OFF
79. ENG DUCT ANTI ICE switch – OFF
80. RAIN REMOVER switch – OFF
81. LIGHT CONTROL/INTERIOR lights – As required
82. PYLON JETTISON switch – OFF, guarded
83.* Right console circuit breakers – In

If INS initialization performed, proceed as follow:

84. IN function selector knob – NAV if requested alignment status reached (refer to "INS Alignment Procedure" contained in this Section). If not, wait for requested ALN status. Check RDY NAV lamp extinguished. Check HSI OFF/NAV flags out of view
85. IN/SEL and TCN/SEL mode selectors – As required
86. Attitude indicator – Check
    a. OFF warning flag out of view
    b. Attitude sphere for proper attitude and freedom from oscillation
    c. Attitude sphere for proper response to trim knob
87. Master IN/CDU – Check format

BEFORE STARTING ENGINE

Before starting the engine, make sure danger areas (Figure 2-3) fore and rear of aircraft are clear of personnel, aircraft, and vehicles.
The boundary layer control outlet for the intake ducts on each side of the lower fuselage will have a strong sunction when the engine is started which may be strong enough to draw articles of clothing or loose equipment into the engine.
When practicable, start engine with aircraft headed into the wind. An external electrical power source should be connected and used.

CAUTION

- STARTER LIMITATIONS ARE AS FOLLOWS:
  1 MINUTE CONTINUOUS OPERATION
  3 MINUTES COOLING PERIOD
  1 MINUTE CONTINUOUS OPERATION
  10 MINUTES COOLING PERIOD
- THE AUTOMATIC STARTING FEATURE SHALL BE USED WHENEVER POSSIBLE. IF THE AUTOMATIC START SYSTEM MALFUNCTIONS, THE MISSION NEED NOT BE ABORTED; HOWEVER, THE MALFUNCTION SHALL BE CORRECTED PRIOR TO THE NEXT FLIGHT. IF THE AUTOSTART CABLE IS NOT CONNECTED, THE PILOT HAS NO CONTROL OVER STARTING AIR IN THE EVENT OF STARTER OVERSPEED.

DANGER AREAS

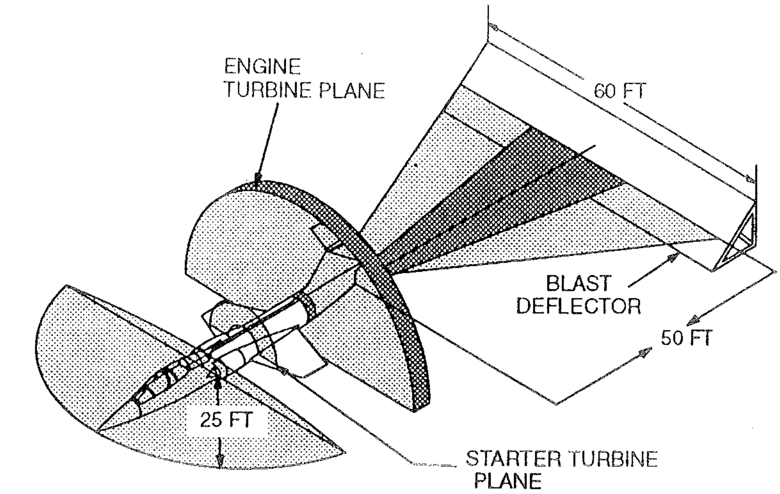
WARNING

THE AREA NEAR THE INTAKE DUCTS AND THE EXHAUST IS VERY DANGEROUS – KEEP CLEAR.

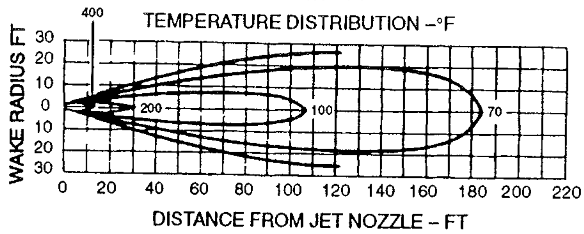
DURING START AND RUNUP AVOID PLANE OF STARTER TURBINE AND ENGINE TURBINE WHEELS.

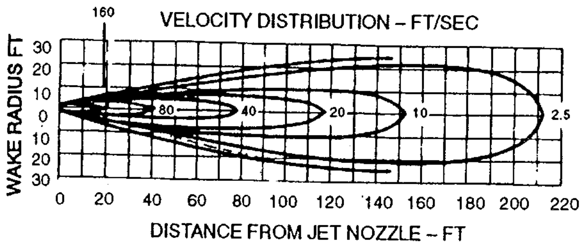
DURING RUNUP, ENGINE NOISE CAN CAUSE PERMANENT DAMAGE TO EARS. WITHIN 100 FEET USE EAR PLUGS. WITHIN 50 FEET USE EAR PLUGS AND PROTECTIVE COVERS.

IF BLAST DEFLECTOR IS NOT AVAILABLE, CLEAR AREA FOR 250 FEET.

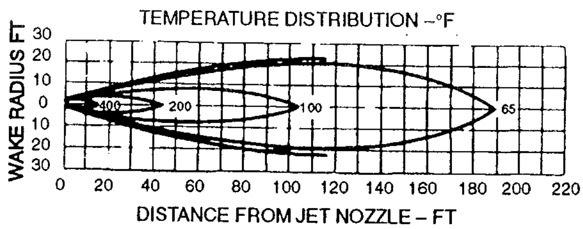


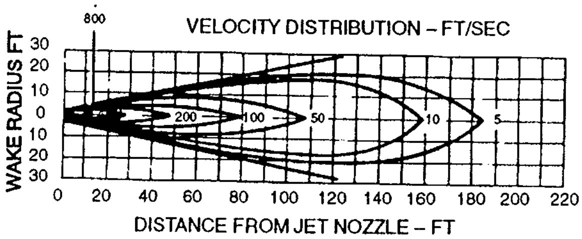
IDLE THRUST



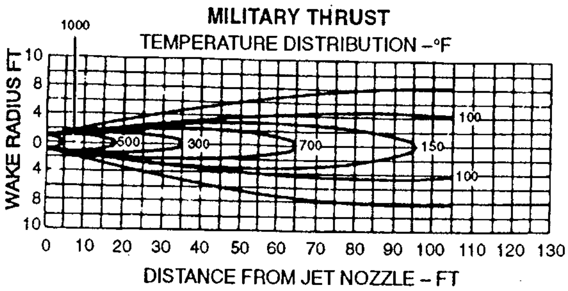


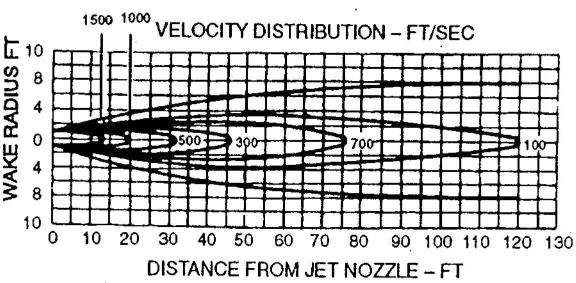
TAXI THRUST



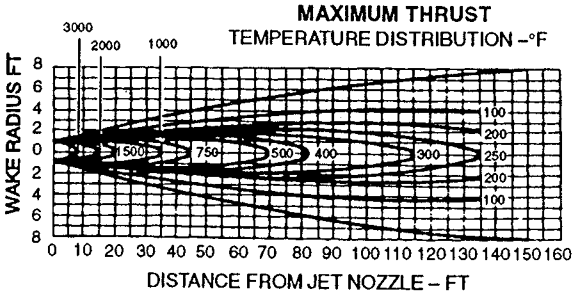


MILITARY THRUST





MAXIMUM THRUST



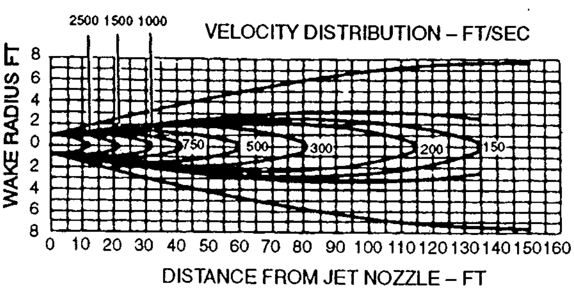


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Figure 2-3

STARTING ENGINE

Basically, three types of starts may be made. These are automatic, manual and battery.
The following chart shows the difference between the starts, and how existing equipment may be utilized to effect a start.

Type of Start	Automatic Start Control Cable Connected	Air Compressor Connected	External Electrical Power Connected
Automatic	Yes	Yes	Yes
Manual	No	Yes	Yes
Battery	Yes or No	Yes	No

CAUTION

DURING ANY START, IT IS IMPERATIVE THAT PILOT AND GROUND CREWMAN COORDINATE THEIR ACTIONS TO PREVENT OVERSPEED OF THE STARTER. PILOT SHALL SIGNAL THE GROUND CREWMAN AT 40% ENGINE RPM TO DISCONNECT EXTERNAL AIR IMMEDIATELY. THIS WILL PREVENT EXCEEDING THE 47% OVERSPEED RPM.

MANUAL START

1. Ground turbine compressor – Connected and ON
2. Start switch – START and release

NOTE

- Use No. 1 ignition system for engine starts on odd-numbered days and use No. 2 ignition system for engine starts on even-numbered days. The alternate usage of ignition systems provides a check on operation of both ignition systems. In case of malfunction of either No. 1 or No. 2 start system, flight should be aborted.
- During cold weather operation, false starts may be encountered; especially on first start of the day. If this condition is experienced, activate both ignitions systems and let RPM build up to 12-14% before advancing throttle to idle range.
- Maximum starting time should not exceed 75 seconds from first indication of RPM until engine reaches idle RPM.

3. Throttle – At 10% RPM MILITARY, then IDLE, check CANOPY UNSAFE warning light, clock act
4. Fuel flow 400-800 pounds per hour – Check at 10 to 12% RPM

CAUTION

- IF FUEL FLOW EXCEEDS 800 POUNDS PER HOUR, A "HOT START" MAY RESULT. IF FUEL FLOW IS LESS THAN 400 POUNDS PER HOUR FOR GROUND STARTS, A "HUNG START" MAY RESULT. IF EITHER OF THESE CONDITIONS OCCUR, THE AIRCRAFT SHALL BE CLEARED BY MAINTENANCE PERSONNEL BEFORE FLIGHT.

- IF COMBUSTION DOES NOT OCCUR BY 20% RPM OR 20 SECONDS, WHICHEVER OCCURS FIRST, AFTER FUEL FLOW INDICATION, OR THE ENGINE FAILS TO ACCELERATE TO NORMAL IDLE RPM, OR THE ENGINE SPONTANEOUSLY ACCELERATES WITH THROTTLE IN IDLE POSITION BEYOND THE NORMAL IDLE RPM (IDLE OVERSPEED CONDITION), OR EXHAUST GAS TEMPERATURE EXCEEDS STARTING LIMITS PROCEED AS INDICATED IN FALSE, HANGING, IDLE OVERSPEED OR HOT START PROCEDURES IN THIS SECTION.

5. Start switches – STOP-START at 40% RPM. At 40% RPM, simultaneously move the No. 1 and No. 2 start switches to the STOP-START position and signal ground crew to stop air flow

CAUTION

IF THE THROTTLE IS UNINTENTIONALLY RETARDED TO OFF, A FLAMEOUT WILL OCCUR IMMEDIATELY. DO NOT REOPEN THROTTLE, AS THE RESULTING FLOW OF UNBURNED FUEL INTO THE ENGINE CREATES A FIRE HAZARD.

6. External electrical power and ground turbine compressor – At idle RPM signal ground crew to disconnect
7. Engine instruments for proper indications – Check:
   a. Nozzle position – 8.5 to 9.5
   b. Tachometer – 67% (±1) RPM

CAUTION

IF THE ENGINE SPONTANEOUSLY ACCELERATES (WITH THROTTLE IN IDLE POSITION) BEYOND 67% (±1) RPM (IDLE OVERSPEED CONDITION) PROCEED IMMEDIATELY AS INDICATED IN FALSE, HANGING, IDLE OVERSPEED OR HOT START PROCEDURES IN THIS SECTION.

   c. Exhaust gas temperature – Normal (320° C to 420° C, not a limit, for reference only)

NOTE

In extremely hot weather with ramp temperatures in excess of 38° C, EGT may increase as high as 500° C. In extremely cold weather with ramp temperatures as low as minus 40° C, EGT may be as low as 120° C.

   d. Oil pressure – 12 psi minimum

NOTE

The ENG OIL LEVEL LOW warning light may flicker at engine speeds below idle. This is acceptable as long as the light is not on for longer than 3 seconds.

   e. Fuel flow – NORMAL (1000 to 1400 lb/hr, not a limit, for reference only)

NOTE

Under extreme temperature conditions fuel flow may be as high as 1600 lb/hr or as low as 700 lb/hr.

AUTOMATIC START

An automatic start is the same as a manual start, except that starting air is applied and shutoff by actuation of the start switch. Check that automatic start control cable is connected.

BATTERY START

A battery start is accomplished with only the air compressor unit connected and with or without the autostart control cable connected. With the autostart control cable connected, the starting procedure is the same as in an automatic start. Without the auto-start control cable connected, the starting procedure is the same as in a manual start.
Stop starting ignition switches, after positive indication of combustion, will not affect starting performance but will improve battery life by disengaging the ignition circuit.

CAUTION

DURING A BATTERY START, THE ONLY ENGINE INSTRUMENTS AVAILABLE UNTIL THE GENERATORS REACH OPERATING SPEED WILL BE THE EXHAUST GAS TEMPERATURE GAGE AND THE TACHOMETER; THEREFORE, EXHAUST GAS TEMPERATURE SHALL BE MONITORED CLOSELY TO PREVENT A POSSIBLE OVERTEMPERATURE CONDITION.

FALSE, HANGING, IDLE OVERSPEED OR HOT START PROCEDURES

1. Throttle – OFF
2. Start switches – STOP-START. Simultaneously move start switches to STOP-START and signal crew to stop air flow
3. Check for absence of fuel in tailpipe

WARNING

WAIT UNTIL THE ENGINE STOPS ROTATING BEFORE CHECKING FOR FUEL IN THE TAILPIPE. IF FUEL IS PRESENT, MOTOR ENGINE.

CAUTION

STARTER LIMITATIONS ARE AS FOLLOWS:
1 MINUTE CONTINUOUS OPERATION
3 MINUTES COOLING PERIOD
1 MINUTE CONTINUOUS OPERATION
10 MINUTES COOLING PERIOD.

4. Attempt restart (unless aircraft shall be cleared by maintenance personnel, refer to Section V)

GROUND OPERATION

AFTER START CHECK

1. Check the following warning lights extinguished: GENERATOR NO. 1 OUT, GENERATOR NO. 2 OUT, HYDRAULIC SYSTEM OUT, AUTO PITCH CONT OUT and ENGINE OIL LEVEL LOW

NOTE

The INERTIAL NAV FAULT warning light may be still lit.

2. Fixed Freq. Reset button – Press for at least 5 sec., check FIXED FREQ. OUT warning light extinguished

NOTE

Check IN function selector knob on NAV. If not perform steps 3., 4. and 5.

3. IN – ALN function, GC or STO mode (refer to INS Alignment Procedure contained in this Section). Check INERTIAL NAV FAULT warning light extinguished
4. IN/CDU self test – Check "CDU OK" displayed

NOTE

Abort mission if "CDU FAIL" indication is displayed.

5. Master IN/CDU – Select and insert initial position, if GC selected
6. ENG/DUCT ANTI-ICE – Check, as required (refer to Section VII for Under Icing Condition Operations). If CAUTION light does not illuminate within 5 seconds after activation of the engine/duct anti-ice system, smoothly advance throttle. CAUTION light should illuminate prior to reaching an engine speed of 80% RPM

CAUTION

ABORT FLIGHT IF WARNING PLACARD REMAINS ILLUMINATED MORE THAN 5 SECONDS AFTER ENGINE/DUCT ANTI-ICE SWITCH IS SET IN OFF POSITION. MAKE NOTATION OF MALFUNCTION IN AIRCRAFT LOG.

PILOT/CREW CHIEF CHECK

With assistance of ground crew, proceed as follows:

1. (At least once per day) Fuel booster pumps check – Check FUEL BOOST PUMP FAIL warning light out
2. Speed brakes – OUT

WARNING

SHOW HANDS (BOTH PILOTS) UNTIL CREW CHIEF GIVES CLOSURE SIGNAL.

a. Pressure indication on No. 2 gage should drop quickly to minimum of 2175 psi, rise momentarily to a maximum of 3300 psi, and return to normal
b. On signal from crew chief, close speed brakes. Pressure indication on No. 2 gage should drop quickly to a minimum of 2175 psi, rise momentarily to a maximum of 3300 psi and return to normal

3. Controls – Check for free movement, in following order, of aileron, horizontal stabilizer, and rudder as follows:

NOTE

Ensure that both number 1 and 2 system gages move when the following checks are accomplished.

a. Move ailerons only through complete cycle. Pressure indications should drop to a minimum of 2600 psi, rise to 3300 psi maximum, and return to normal
b. Move stabilizer only through complete cycle. Pressure indication should drop to a minimum of 2700 psi, rise to 3300 psi maximum, and return to normal
c. Move rudder through maximum travel. Hydraulic pressure should drop, rise, and return to normal

4. Trim – Check operation and set to takeoff, in following order: aileron, horizontal stabilizer, and rudder. Check operation of trim and obtain ground crew verification for proper movement and position.

WARNING

AN IMPROPERLY INSTALLED OR DEFECTIVE TRIM SWITCH IS SUBJECT TO STICKING IN ANY OR ALL ACTUATED POSITIONS, RESULTING IN APPLICATION OF EXTREME TRIM. IF SWITCH DOES NOT RETURN AUTOMATICALLY TO OFF POSITION, ABORT FLIGHT.

CAUTION

TRIM MECHANISM CAN BE DAMAGED BY OPERATING TRIM CONTROLS WITH CONTROL STICK IN A FULL THROW POSITION. MAKE ALL TRIM SYSTEM CHECKS WITH CONTROL STICK IN NEUTRAL POSITION.

NOTE

- The STABILIZER takeoff trim light will remain illuminated when the trim switch is released and the stabilizer is in the takeoff trim position. Have ground crew verify proper trim surface position.
- Leading edge of horizontal stabilizer should be aligned with black T-index on vertical stabilizer.

5. STABILITY CONT switches – Lift cover, OFF then ON then guarded. Check pressure gages flickering and have a ground crew confirmation

Observe respective control-response and hydro system gages indication (flicker). Have operation of dampers verified by crew chief.

NOTE

Turning the yaw damper off and on may not cause control response and hydraulic system gage indication.

6. APC check – Wing flaps UP, stick released
   a. Right vane 4.25 to 4.75 – Shaker
   b. Right vane 5 – No kicker
   c. Right vane 5 – Kicker
   d. Overpower APC – Obtain an aft stick movement of 2 to 3 inch (if force is maintained stick will slowly move to full aft position at approximately ½ inch per second)
   e. APC emergency disengage switch (paddle switch) – Press, check AUTO PITCH CONT OUT warning light lit
   f. Aileron/ rudder limiters – Check limited travel
   g. Left vane up – Shaker
   h. PITOT-PITCH TEMP PROBE HEATERS switch – On for 5 seconds

CAUTION

DO NOT OPERATE FLAPS THROUGH MORE THAN ONE CYCLE AT IDLE RPM. IF REPEATED CYCLES ARE REQUIRED, RPM SHALL BE INCREASED TO 85% OR ABOVE TO PREVENT CONTROL FREEZE.

NOTE

It is recommended, when practical, to allow the flaps to remain in the selected position for a minimum of 30 seconds before reversing flaps travel.

7. RAIN REMOVER switch – Check
8. Wing flaps – LAND have ground crew check BLC air flow (check that flaps extension time does not exceed 20 seconds maximum)
9. Wing flaps – TAKEOFF have ground crew verify flaps position (check that flaps retraction time does not exceed 12 seconds maximum), check absence of BLC airflow

CAUTION

IN ORDER TO PREVENT RADAR ANTENNA DAMAGE THE RADAR MODE SELECTOR SWITCH SHALL BE SET TO SBY.

10. RADAR – SBY
11. Emergency nozzle closure system – Check:
    a. Throttle – IDLE
    b. EMERGENCY NOZZLE CLOSURE handle – Out
    c. Nozzle position indicator – 1.0 to 3.0
    d. EMERGENCY NOZZLE CLOSURE handle – In. Check nozzle returns to IDLE area

NOTE

Movement of handle should be rapid (within one second).

If unable to push handle in and operational conditions permit:

e. Throttle – 85% RPM minimum
f. EMERGENCY NOZZLE CLOSURE handle – In

If still unable to push in:

g. ABORT

12. Brakes – Check (crew chief verifies proper operation)
13. IN function selector knob – NAV if requested alignment status reached (refer to "INS Alignment Procedure" contained in this Section). If not, wait for requested ALN status. Check RDY NAV lamp extinguished. Check HSI OFF/NAV flags out of view
14. TACAN and IFF – As required
15. IN/SEL and TCN/SEL mode selectors – As required
16. Attitude indicator – Check
    a. OFF warning flag out of view
    b. Attitude sphere for proper attitude and freedom from oscillation
    c. Attitude sphere for proper response to trim knob
17. IN/CDU – Check format
18. Parking safety pins in the emergency canopy initiator, canopy jettison initiator, main gun primary firing handle, removed with the aid of the crew chief

CAUTION

THE SAFETY PINS ARE ATTACHED TO TWO RED STREAMERS. AFTER REMOVAL CHECK THAT THE SEVEN PINS ARE ATTACHED TO THE RED STREAMERS AND HAVE NOT BEEN LEFT, BY ERROR, IN THEIR LOCATION.

19. Swivel guard of secondary firing handle – Down

WARNING

FINAL ARMING OF EJECTION SEATS SHOULD NOT BE ACCOMPLISHED INSIDE A SHELTER.

20. External stores safety pins – Removed and shown by ground crew

WARNING

EXTERNAL STORES AUTO DROP SAFETY PINS HAVE TO BE REMOVED IN A CLEAR AREA, WHERE ACCIDENTALLY DROPPED STORES CANNOT ENDANGER GROUND CREW MEMBERS AND/OR MATERIAL. IT MAY BE ADVISABLE TO TAXI CLEAR OF CONGESTED AREAS PRIOR TO REMOVING PINS.

21. UHF HAVE QUICK pre-flight operation – Perform (refer to "UHF HAVE QUICK System Operation" contained in this Section)

BEFORE TAXIING

1. Canopy – Locked

Before taxiing operations the canopy should be full open. When locking the canopy, visually check that the three locking hooks are over the three canopy brackets.

CAUTION

- THE CANOPY CAN BE DAMAGED DURING LOWERING OPERATIONS IF A FIRM GRIP IS NOT MAINTAINED ON THE CANOPY LIFT HANDLE. AS THE CANOPY PASSES OVER TOP DEAD CENTER, THE WEIGHT OF THE CANOPY AND HIGH OR GUSTY WINDS CAN CAUSE THE CANOPY TO SLAM SHUT.
- ANTISKID SYSTEM IS INOPERATIVE AT SPEEDS BELOW 10 KNOTS. MAXIMUM BRAKING AT LOW SPEEDS MAY CAUSE WHEEL LOCKING AND SKIDDING.

2. IN – Check NAV
3. Attitude indicator – Set – 5°
4. Wheel chocks – Removed

TAXIING

(See Figure 2-4 for minimum turning radius and ground clearances).

1. Nosewheel steering – Engage

The nosewheel and rudder pedals must be correctly aligned before engaging nosewheel steering.

CAUTION

- DURING TAXIING, THE SPEED BRAKES SHOULD NOT BE OPERATED WHILE NOSEWHEEL STEERING OR POWER/ANTISKID BRAKES ARE REQUIRED.
- UPON COMPLETION OF THE MOVEMENT OF THE SPEED BRAKES, NOSEWHEEL STEERING BECOMES AVAILABLE; HOWEVER, IT MAY BE NECESSARY TO MOVE THE RUDDER PEDALS TO REENGAGE THE STEERING. THE BRAKES WILL AUTOMATICALLY REVERT TO POWER-ANTISKID.
- TO PREVENT STRUCTURAL DAMAGE, ADDITIONAL DIFFERENTIAL BRAKING SHOULD NOT BE APPLIED IN TURNS WITH NOSEWHEEL STEERING ENGAGE.

2. Brakes:
   a. ANTI-SKID switch OFF – Check standby brakes
   b. ANTI-SKID switch ON – Check normal braking action. Leave ANTI-SKID brake switch ON throughout flight.
3. Flight instrument and navigation equipment – Check
   a. Check HSI for proper indication and operation while taxiing.
   b. Attitude indicators – Check – 5°
   c. Turn and slip indicator – Operating (check for turn needle deflection in the direction of turn while taxiing and ball free in the race)
   d. Vertical velocity indicator – Check for zero setting
   e. Standby magnetic compass – Check heading indication and that the card swings freely and the bowl is full of fluid

MINIMUM TURNING RADIUS AND GROUND CLEARANCES

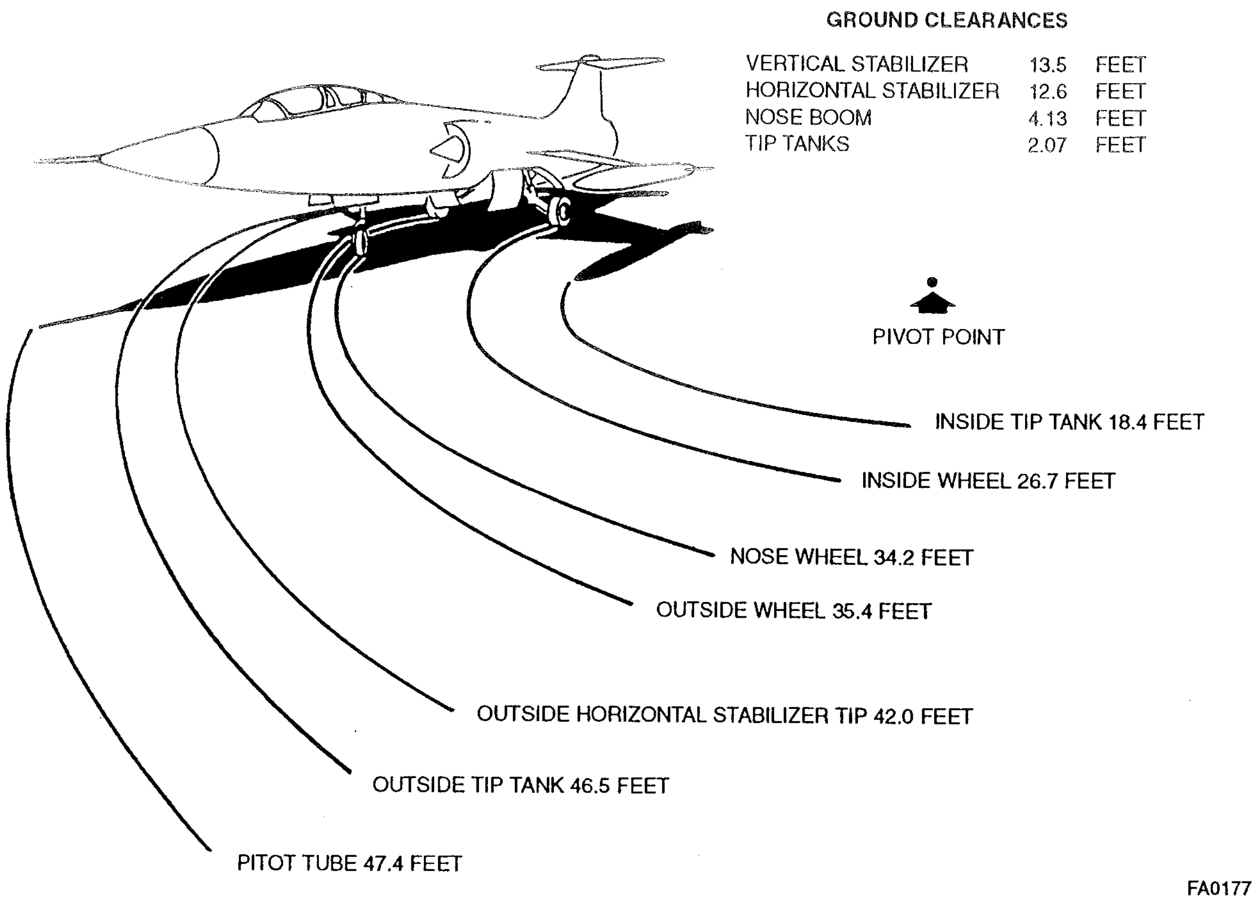


Figure 2-4

f. IN/CDU – Check for proper indications on navigation format (Refer to "IN/CDU Operation" in this Section)

BEFORE TAKEOFF

Perform the following checks:

1. External tanks – Check that fuel transfer can be controlled. Maintain fuel quantity below normal full indication. If correct external fuel feeding cannot be controlled – ABORT

WARNING

OPERATIONAL EXPERIENCE HAS SHOWN THAT INTERNAL FUEL CELL OVERPRESSURIZATION IS POSSIBLE WHEN ASSOCIATED WITH AN OPEN POSITION FAILURE OF THE DUAL AIR PRESSURE REGULATOR AND CERTAIN ACCELERATION AND EXTERNAL FUEL TRANSFER CONFIGURATIONS. THIS OVERPRESSURIZATION CAN RESULT IN FUEL CELL RUPTURE AND/OR AIRCRAFT STRUCTURAL FAILURE.

2. Wing flaps – TAKEOFF, detent
3. Seat equipment – Recheck
   a. Combined harness and leg straps – Tight. Shoulder harness – Locked
4. Speed brakes – Check IN
5. Full stabilizer travel – Check available STABILIZER takeoff trim light – Check

NOTE

To assure full stabilizer travel availability, takeoff trim is required.

6. Oxygen – As required, check quantity, pressure, blinker working, and connections
7. Ejection seat – Recheck
   a. Ejection seat safety pin of upper firing handle – OUT
   b. Swivel guard of lower firing handle – DOWN
8. IFF, TACAN and Radar – Check
9. CANOPY DEFOGGER – Check for operation, operate if required
10. Canopy – Locked (visually check locks). CANOPY UNSAFE warning light – Check extinguished

WARNING

EVEN THOUGH THE CANOPY UNSAFE WARNING LIGHT IS OUT, OBSERVE THAT HOOKS ARE PROPERLY ENGAGED FOR POSITIVE INDICATIONS THAT CANOPY IS LOCKED.

NOTE

Nozzle position indication may drop to 0 momentarily before stabilizing.

ENGINE CHECK

See Figure 1-7 for exhaust nozzle at various throttle settings and see Figure 5-1 for engine limitations. While in takeoff area, align aircraft with runway, check that nosewheel is centered and nosewheel steering engaged (to prevent the aircraft from pivoting to either side in case of one wheelbrake failing), and perform the following checks:

1. Nosewheel steering – Centered and engaged
2. Throttle – Advance throttle rapidly to MILITARY. Check engine acceleration and instruments
   a. Engine acceleration – 10 seconds maximum
   b. RPM – 100% (±1)
   c. EGT – 590° C (±10° C)

NOTE

During steady-state operation, the EGT gauge may fluctuate plus or minus 5° C with peaks of up to 10° C. This is normal providing the peaks (10° C) do not occur more often than once every 5 seconds.

d. Nozzle position – 1.0 to 3.5

NOTE

Nozzle position indication may drop below 1.0 momentarily before stabilizing.

e. Fuel flow – Check. See Figure 2-5
f. Oil pressure – Check placard ±5 psi (±2 psi fluctuation, steady state pressure is allowable)
g. CANOPY UNSAFE warning light – Check out

3. Throttle – Reduce slowly to 80% RPM; check for compressor stall. If compressor stall is encountered, throttle OFF
4. Throttle – Rapidly retard to IDLE stop. Check for stabilized engine operation and minimum fuel flow of 400 lb/hr
5. Pitot heat and engine anti-ice – As required

TAKEOFF

NOTE

- Maximum thrust should be used for takeoff. Check Appendix for takeoff distances.
- The procedures set forth below will produce the results shown in the takeoff charts in the Appendix.

NORMAL TAKEOFF

(Figure 2-6).

1. Nosewheel steering – Engaged
2. Throttle – Military
3. Brakes – Release
4. Throttle – Minimum afterburner (ensure a stabilized afterburner light)

NOTE

A stabilized afterburner light will be indicated by the following:

- RPM may roll-back, then return to normal and stabilize
- EGT will increase to above 600° C then decrease and stabilize between 580° C and 600° C.
- Nozzle position will stabilize between 3.5 and 5.0 in minimum sector afterburning, depending on ambient temperature and altitude.
- Afterburner should light within 3 seconds.

5. Throttle – Maximum thrust
6. Engine instruments – Check

WARNING

DURING AFTERBURNER TAKEOFF, AVOID THROTTLING INTO THE SWITCHOVER POINT TO PREVENT AFTERBURNER BLOWOUT.

MILITARY THRUST FUEL FLOW

GROUND OPERATION

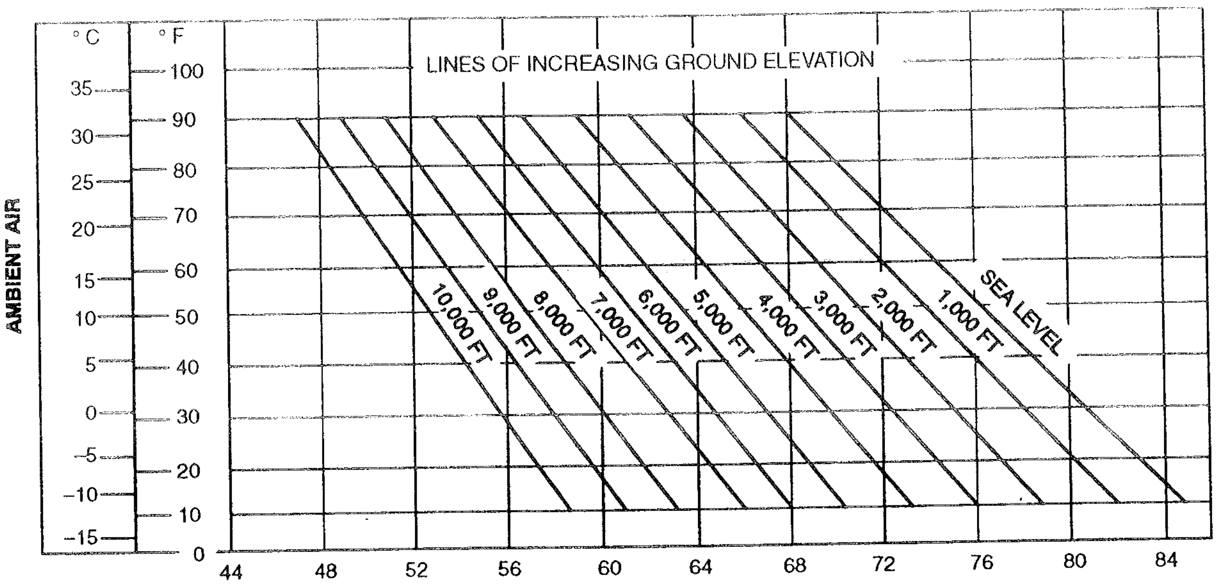


Figure 2-5 FA0178

CAUTION

WITH THROTTLE AT MAXIMUM THRUST, NOZZLE POSITION WILL STABILIZE BETWEEN 7.5 AND 9.5. WITH THROTTLE REMAINING AT MAXIMUM THRUST, AFTERBURNER BLOWOUT WILL BE INDICATED BY A DEFINITE LOSS OF THRUST AND A NOZZLE POSITION READING OF LESS THAN 7.0, ACCOMPANIED BY AN EGT BELOW 500° C. IF BLOWOUT OCCURS, THRUST WILL BE CONSIDERABLY BELOW MILITARY AND TAKEOFF SHOULD BE EITHER ABORTED OR CONTINUED AT MILITARY THRUST, DEPENDING ON SPEED AND REMAINING RUNWAY. IT IS ACCEPTABLE FOR NOZZLE POSITION INDICATOR TO READ 10 OR ABOVE IF EGT REMAINS WITHIN LIMITS.

7. Use nosewheel steering as necessary for directional control

NOTE

The rudder becomes effective at approximately 70 KIAS.

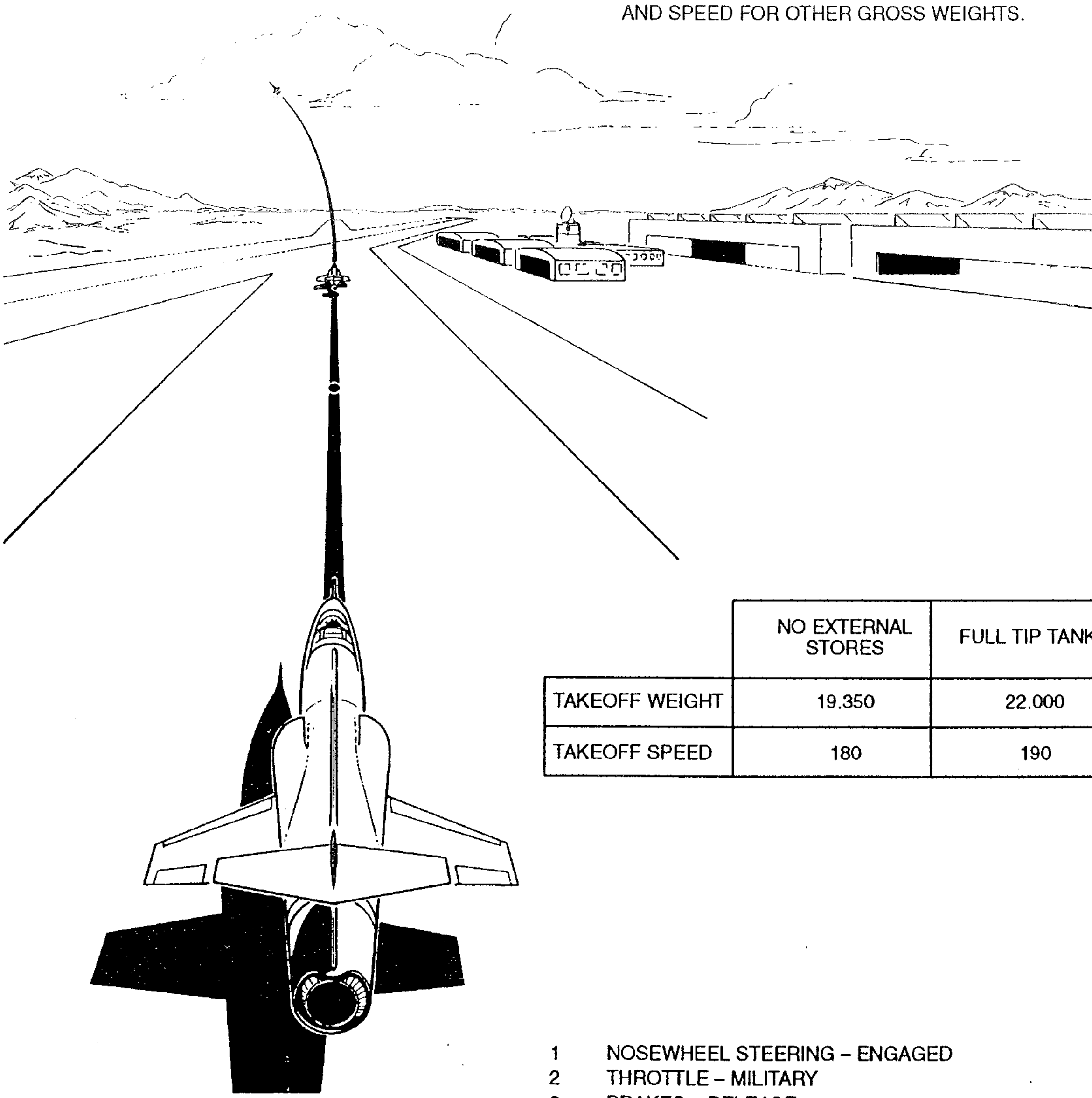
CAUTION

- NOSEWHEEL STEERING SHOULD BE DISENGAGED PRIOR TO NOSEWHEEL LIFT-OFF TO ENSURE PROPER STEERING CLUTCH RELEASE.
- WITH STEERING SYSTEM ENGAGED, SHIMMY DAMPING IS LESSENED. IF NOSEWHEEL SHIMMY IS ENCOUNTERED, RELEASE NOSEWHEEL STEERING.

TYPICAL TAKEOFF

NOTE

- NECESSARY ENGINE INSTRUMENT CHECKS WHEN USING THE AFTERBURNER, MUST BE MADE DURING THE INITIAL PORTION OF TAKEOFF ROLL BECAUSE AIRPLANE CANNOT BE HELD WITH THE BRAKES.
- REFER TO APPENDIX FOR TAKEOFF DISTANCES AND SPEED FOR OTHER GROSS WEIGHTS.



	NO EXTERNAL STORES	FULL TIP TANKS
TAKEOFF WEIGHT	19.350	22.000
TAKEOFF SPEED	180	190

1 NOSEWHEEL STEERING – ENGAGED
2 THROTTLE – MILITARY
3 BRAKES – RELEASE
4 THROTTLE – MINIMUM A/B (ENSURE A STABILIZED AFTERBURNER LIGHT)
5 THROTTLE – MAXIMUM THRUST
6 ENGINE INSTRUMENTS – CHECK
7 USE NOSEWHEEL STEERING AS NECESSARY FOR DIRECTIONAL CONTROL
8 ASSUME TAKEOFF ATTITUDE

FA0179

Figure 2-6

8. Assume takeoff attitude (approximately 5° nose up)

WARNING

IF AFT STICK INPUT IS MADE AT AN AIRSPEED WHICH IS TOO LOW OR IF FULL AFT STICK IS USED PRIOR TO NOSEWHEEL LIFTOFF, THEN THE NOSE MAY FAIL TO ROTATE. IF THIS OCCURS, ABORT.

NOTE

- The takeoff "optimum stick technique" is to anticipate aircraft acceleration to rotate the nose so that takeoff attitude and speed are reached simultaneously. During takeoff roll, hold the stick in takeoff trimmed neutral position to minimize the aerodynamic drag. At 20 knots below the computed takeoff speed smoothly bring the stick back a little over 3 inches to obtain the optimum stabilizer deflection. Once rotation and nosewheel lift-off occurs, further aft stick is unnecessary. Rotation and nosewheel lift-off will normally occur 10 to 20 knots below the computed takeoff speed, except for forward center of gravity configuration. (Refer to Appendix for computed takeoff speed).

- As the stick is pulled smoothly back toward the optimum deflection, the moving stabilizer will actually provide more pitching moment than a static deflection held at a given setting. This stick movement figure is a target value to use as a guide. the actual technique is to feel when the tail lift moment is at maximum effect. (Refer to Section VI, Takeoff Characteristics).

MINIMUM RUN TAKEOFF

Procedure is the same as for normal takeoff with afterburner. Maximum performance takeoff speed is 5 knots less than for normal performance takeoff.

OBSTACLE CLEARANCE TAKEOFF

The procedure to take off and clear an obstacle is the same as for normal takeoff with afterburner; refer to the Appendix for distances to clear a 50-foot obstacle.

CROSSWIND TAKEOFF

In addition to the procedures used for normal takeoff, under gusty crosswind conditions, increase nosewheel lift-off and takeoff speed 5 KIAS for each 10 kts above steady wind velocity. Nosewheel steering may be required in excess of 100 knots if strong crosswinds are present.

AFTER TAKEOFF

1. LDG GEAR – UP

When aircraft is definitely airborne, retract gear.

WARNING

- THE LANDING GEAR MAY BE RETRACTED WHEN THE WEIGHT OF THE AIRCRAFT IS OFF THE NOSE GEAR BUT STILL ON THE MAIN GEAR. THEREFORE, BE SURE THE AIRCRAFT IS DEFINITELY AIRBORNE BEFORE MOVING GEAR HANDLE.

- THE LANDING GEAR AND DOORS SHOULD BE COMPLETELY UP AND LOCKED BEFORE REACHING 260 KIAS; OTHERWISE EXCESSIVE AIRLOADS MAY DAMAGE THE MECHANISM OR PREVENT GEAR RETRACTION.

2. Landing gear warning lights out — Check
3. Wing flaps — UP at 300 KIAS minimum. Check indicator

NOTE

Expect an easily controllable nose-up tendency as the flaps retract.

4. Throttle — As desired (monitor nozzle position indicator)
5. Engine instruments — Check
6. Airspeed — Best climb

Refer to the Appendix for best climb speed. Take care following takeoff to anticipate the high forward acceleration. As climb speed is approached assume the proper climb attitude to ensure maximum performance.

CLIMB

The climb attitude with maximum thrust is extremely steep and until experience is gained, some difficulty in holding the climb schedule will be experienced. Refer to the climb charts for recommended speeds to be used during climb, and for rates of climb and fuel consumption.

1. Fuel quantity — Check; sequence external fuel transfer to maintain internal fuel quantity indication below normal full indication until external tanks are empty

NOTE

Monitor external fuel depletion to assure symmetric tip tank fuel depletion. If an asymmetric fuel condition in excess of 450 pounds is indicated, reduce speed to Mach 0.9 avoiding load factors in excess of 2.0 G's and maintain altitude less than 35000 feet until the asymmetric condition no longer exists. If the asymmetric condition persists, refer to Asymmetric Tip Tank Fuel Load in Section III and to Flight Characteristics in Section VI for flight characteristics under low airspeed conditions.

2. Oxygen diluter lever — Normal
3. Altimeter — Set
4. TACAN — Check
5. Cockpit pressurization — Check

CRUISE

Refer to the Appendix for cruise operating data. The windshield and canopy defogging system should be operated throughout the flight at the highest flow possible (consistent with pilot comfort) so that a sufficiently high temperature is maintained to preheat the canopy and windshield areas. It is necessary to preheat because there is insufficient time during rapid descents to heat these areas to temperatures which prevent the formation of frost or fog.

CAUTION

THE ROLL STABILITY AUGMENTER SHOULD BE TURNED OFF BEFORE REACHING 575 KIAS WITH WING-TIP STORES INSTALLED. WITH THIS CONFIGURATION AND THE ROLL STABILITY AUGMENTER OPERATING, WING TORSIONAL OSCILLATIONS SUFFICIENT TO CAUSE STRUCTURAL DAMAGE MAY BE EXPERIENCED AT HIGH INDICATED AIRSPEEDS.

NOTE

The APC and stick shaker may be checked in flight as follows: while applying a slow stick deflection, note APC indicator reading increase in relation to angle of attack and increasing G force indicating satisfactory system operation from sensing of vane angle. Apply a small rapid stick deflection and note APC indicator reading increase rapidly in relation to the increasing pitch rate, indicating a satisfactory signal from the pitch rate gyro. The stick deflection should be great enough to induce a pitch rate sufficient to actuate the stick shaker.

FLIGHT CHARACTERISTICS

Refer to Section VI for information regarding flight characteristics.

AFTERBURNER OPERATION

Before moving the throttle into the afterburner range, check the nozzle position indicator for normal indication in the Military Thrust range.
Move the throttle smoothly outboard and forward into the afterburner range.
Check exhaust gas temperature, RPM and nozzle position.

CAUTION

IF AN AFTERBURNER LIGHT IS NOT OBTAINED WITHIN APPROXIMATELY THREE SECONDS AT SEA LEVEL OR APPROXIMATELY FIVE SECONDS AT ALTITUDE AFTER THE THROTTLE IS MOVED INTO AFTERBURNER RANGE, MOVE THE THROTTLE INBOARD TO THE MILITARY THRUST POSITION. AFTER 3 TO 5 SECONDS, RETURN TO THE AFTERBURNER RANGE TO RECYCLE THE SYSTEM. AFTER LIGHTOFF IS OBTAINED, MOVE THE THROTTLE FORWARD WITH A POSITIVE MOTION IF MAXIMUM THRUST IS DESIRED.

NOTE

- The fuel flow indicator does not indicate afterburner fuel flow.
- As soon as afterburner thrust is no longer needed, shut down the afterburner by moving throttle aft and inboard to the military thrust position. Monitor the nozzle position indicator to ensure that the nozzle is not fully open and closes normally. During afterburner operation with nozzle failed to open position, reducing the throttle below full afterburner will cause afterburner blowout. Throttle should remain in maximum afterburner until landing is assured.

DESCENT

Refer to the Appendix for recommended descent technique and accomplish the following steps:

1. Engine/duct anti-ice, defogger, and pitot heat – As required
2. Altimeter – Set
3. Fuel quantity – Check (determine final approach speed)
4. Oxygen diluter lever – As required

BEFORE LANDING

Following the procedures set forth below will produce the results shown in the landing chart in the Appendix.

1. Radar – SBY

TYPICAL LANDING PATTERN

(Figure 2-7).

Initial

1. Wing flaps – TAKEOFF, check indicator
2. Airspeed – 325 KIAS
3. Altitude – 2000 feet (AGL)
4. Shoulder harness – Locked

TYPICAL LANDING PATTERN

BASED ON A GROSS WEIGHT OF 16000 lbs

NOTE

ADJUST APPROACH AND LANDING SPEED 5 KNOTS FOR EACH 1000 LB DIFFERENCE IN GROSS WEIGHT OR PORTIONS THEREOFF.

REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR FINAL APPROACH AND TOUCHDOWN SPEEDS FOR OTHER CONFIGURATIONS AND GROSS WEIGHTS.

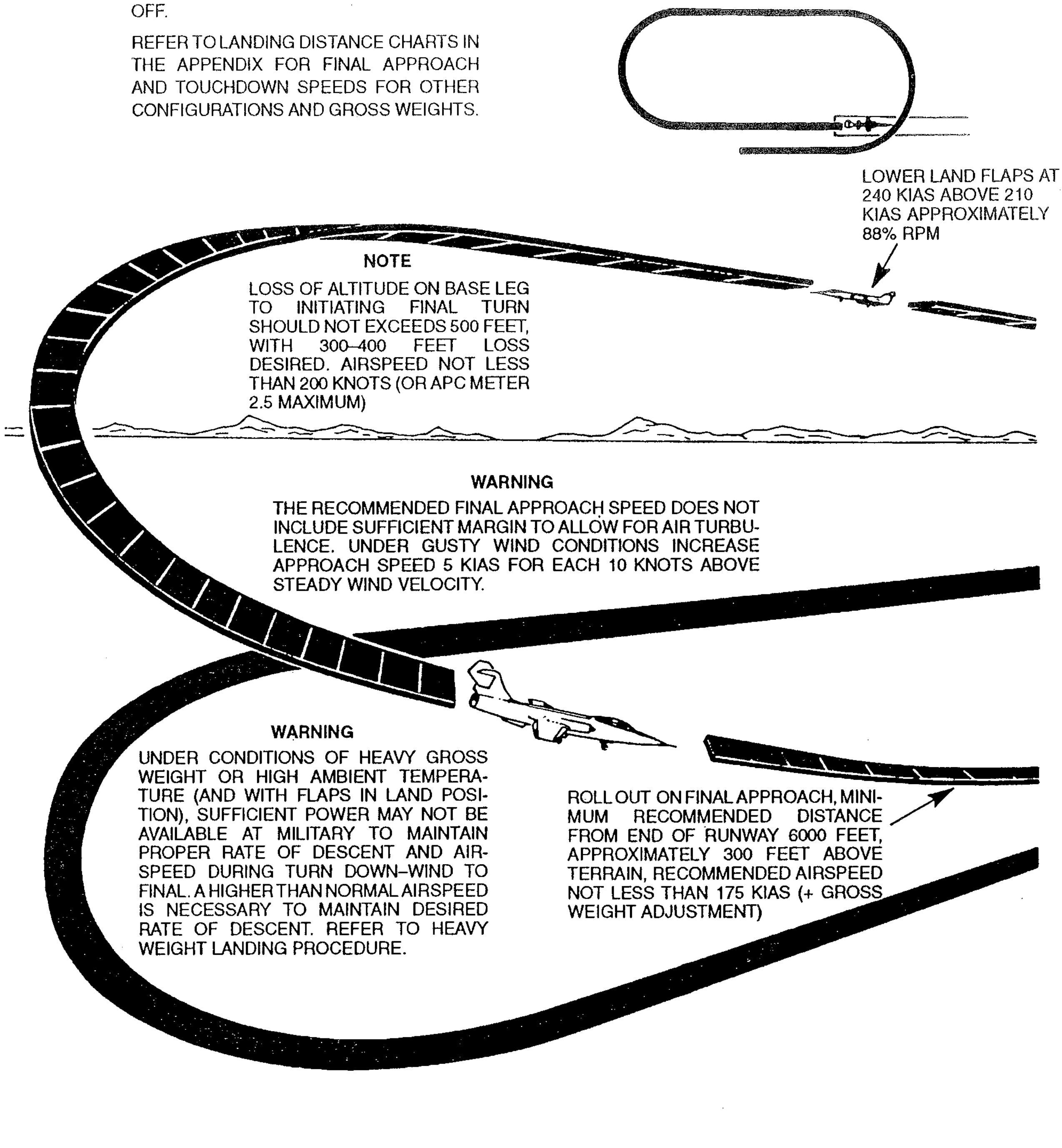
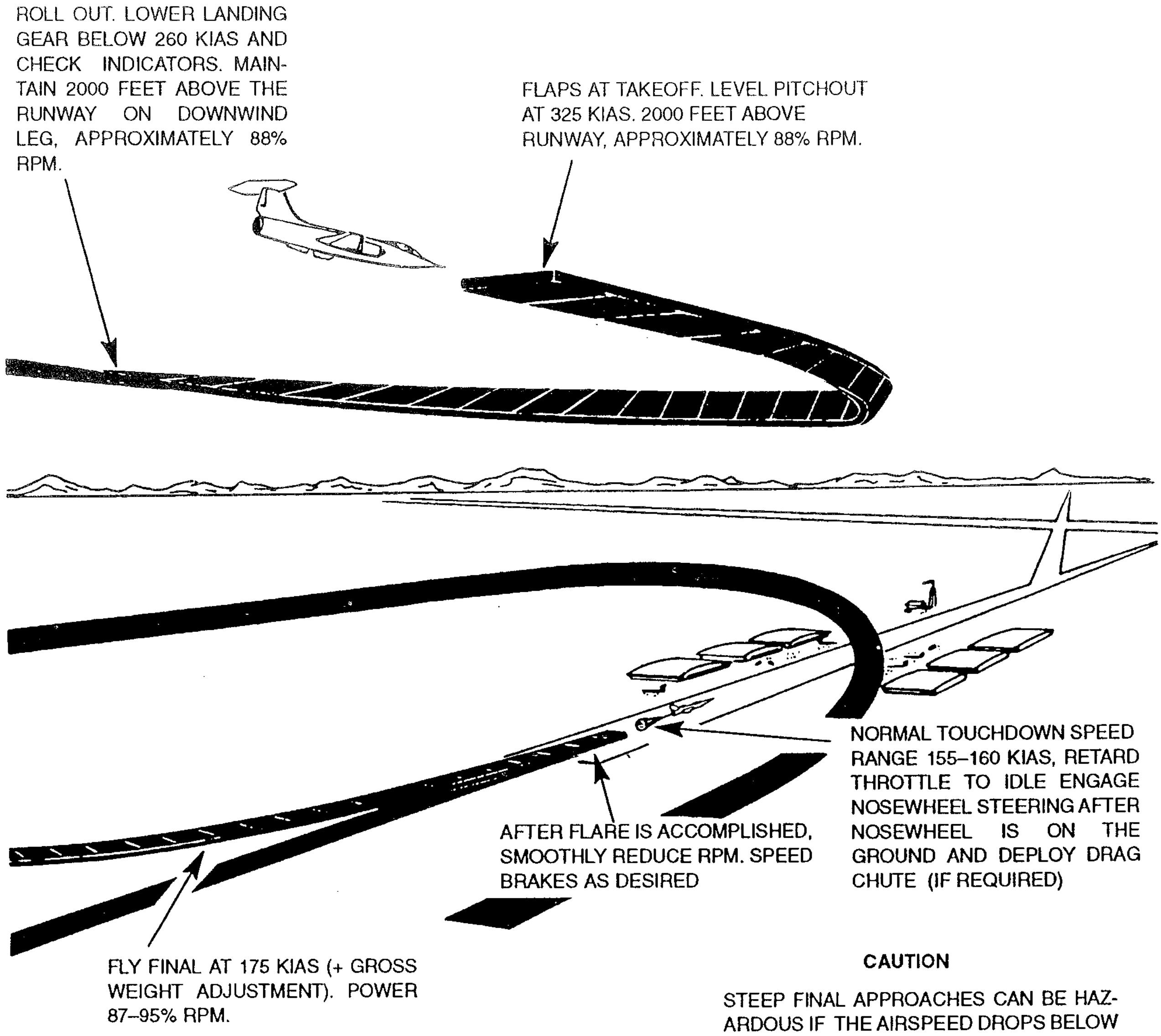


Figure 2-7 (Sheet 1 of 2)

TYPICAL LANDING PATTERN



CAUTION

STEEP FINAL APPROACHES CAN BE HAZARDOUS IF THE AIRSPEED DROPS BELOW NORMAL, TURNS ARE MADE, OR GUSTY WINDS PREVAIL. THESE FACTORS MAY CAUSE AN EXCESSIVE RATE OF SINK WHICH WILL NOT BE RECOGNIZED AND CORRECTED BEFORE CONTACT WITH THE GROUND.

ALL FINAL APPROACHES SHOULD BE MADE WITH POWER, AND ON A GLIDE SLOPE SIMILAR TO THAT FOR ILS/PAR (700–800 FEET PER MINUTE).
THIS SLOPE MAY BE INTERCEPTED AT ANY POINT, BUT SHOULD BE INTERCEPTED AT NOT LESS THAN ONE MILE FROM TOUCHDOWN.

WARNING

DO NOT "CHOP" THROTTLE WHILE AIRBORNE AS ABRUPT LOSS OF LIFT WILL ACCOMPANY THE DECREASE IN BOUNDARY LAYER CONTROL AIRFLOW.

FA0181

Figure 2-7 (Sheet 2 of 2)

Downwind

1. LDG GEAR – DOWN when speed has decreased to less than 260 KIAS, check indicators

NOTE

If the instrument lights rheostat is inadvertently moved out of the OFF position, the warning lights and landing gear position lights may not be visible during daylight operation.

2. Landing lights – On

WARNING

THE CONFIGURATION CHANGE SHOULD BE PERFORMED NOT LOWER THAN 2000 FT AGL IN ORDER TO ALLOW ENOUGH ALTITUDE FOR AIRCRAFT RECOVERY SHOULD A BLC MALFUNCTION OR ASYMMETRIC FLAPS OCCUR.

3. Wing flaps – LAND when speed is below 240 KIAS and above 210 KIAS

Check indicators. Maintain level flight and keep hand on lever until flaps and BLC are known to be functioning normally.

NOTE

A mild roll transient may be experienced on some aircraft as flaps move from TAKEOFF to LAND positions. The cause is attributed to asymmetric differences in boundary layer control systems and will vary in intensity and direction with individual aircraft. Maximum lateral stick displacement should not exceed one inch.

Base Leg Turn

1. Landing gear down and locked – Check
2. Wing flaps selected position – Check
3. ANTI-SKID, AIL AND RUD UNLIMITED lights – Check
4. Airspeed – 200 KIAS minimum (or APC meter 2.5 maximum)

WARNING

UNDER CONDITIONS OF HEAVY GROSS WEIGHT OR HIGH AMBIENT TEMPERATURE (AND WITH FLAPS IN THE LAND POSITION), SUFFICIENT POWER MAY NOT BE AVAILABLE AT MILITARY TO MAINTAIN PROPER RATE OF DESCENT AND AIRSPEED DURING TURN FROM DOWNWIND TO FINAL. A HIGHER THAN NORMAL AIRSPEED IS NECESSARY TO MAINTAIN DESIRED RATE OF DESCENT. REFER TO HEAVY WEIGHT LANDING PROCEDURE.

Final

NOTE

The airspeed listed herein are based on a gross weight of 16000 pounds. Adjust base, final, and landing speeds 5 knots for each 1000 pounds difference in gross weight or portions thereof. Refer to landing speed schedule in the Appendix for the other configurations and gross weights.

1. Final approach:
   Minimum recommended distance from end of runway – 6000 feet
   Recommended airspeed – Not less than 175 KIAS (plus gross weight adjustment)

WARNING

THE RECOMMENDED FINAL APPROACH SPEED DOES NOT INCLUDE SUFFICIENT MARGIN TO ALLOW FOR AIR TURBULENCE. UNDER GUSTY WIND CONDITIONS, INCREASE APPROACH SPEED 5 KIAS FOR EACH 10 KNOTS ABOVE STEADY WIND VELOCITY.

2. Landing gear — Recheck: three green lights
3. Engine speed — Not less than 83% RPM
4. Fly final at 175 KIAS (plus gross weight adjustment)

LANDING

BOUNDARY LAYER CONTROL

The installation of boundary layer control (BLC) to effect lower landing approach and touchdown speeds has resulted in some new flight characteristics and changes in required piloting technique. The pilot should remember at all times when using LAND flaps that the additional lift afforded by BLC is dependent on engine airflow. This lift, therefore, varies with airspeed, altitude, and engine RPM. The greatest effect is realized at low airspeed, low altitude, and engine speeds above 83%, although some effectiveness is still retained at lower power settings. The significance of this is that under the landing condition, especially as touchdown is approached, proper use of the throttle is mandatory to accomplish a smooth reduction in engine RPM so that a smooth reduction in the effects of BLC on lift will result.

LANDING TECHNIQUE

The recommended landing pattern results in a flat powered approach similar to that used for ILS and PAR approaches, carrying approximately 88% RPM until touchdown is approached. A straight-in approach of 6000 feet, minimum, is recommended to simplify the technique and judgment involved in the landing flare. The thrust should be controlled to hold airspeed and sink rate to the recommended values on the final approach (use of the recommended speeds provides ample speed margin from the back side of the power-required curve).

Airspeed response to throttle adjustments is extremely positive and rapid, aiding considerably in establishing a good approach. The high drag of the aircraft in the landing configuration makes it unnecessary to use speed brakes in the landing pattern (especially on the approach). Speed brakes may be used during roundout to aid in controlling touchdown point. The approach should be maintained to establish a flareout just short of the runway. As the touchdown point is approached, flareout rotation should be started, followed by a smooth reduction of thrust to 83% RPM.

An abrupt thrust reduction results in abrupt rolloff tendency and a rapid increase in rate of sink. These characteristics make it necessary to approach touchdown carrying power, and to reduce power to idle as main gear contacts the runway. The smooth thrust reduction reduces the rolloff tendency, thereby making it easy to maintain wings level through the flare as well as to provide positive control of rate of sink. It may seem unnatural to touch down with more than idle thrust; however, with the drag of the landing flaps it is possible to slow down rapidly enough so that Idle thrust need not be used. Adhere to recommended approach and touchdown speeds. If the aircraft is held off to lower speeds lateral stability and control will deteriorate and wing drop tendencies will be experienced. In addition, the high pitch angle required for flight at these low airspeeds will be excessive and can result in tail dragging.

NORMAL LANDING

(Figure 2-7).

1. Normal touch down speed range 155 to 160 KIAS
2. Throttle — Retard to IDLE after touchdown
3. Nosewheel — Lower
4. Rudder pedals — Align
5. Nosewheel steering — Engage

NOTE

- If nosewheel shimmy is encountered, release nosewheel steering and hold weight off nosewheel if possible.

- Ascertain that nosewheel steering is definitely engaged prior to clearing to the turn-off side of the runway.

- At high touchdown speeds or after great temperature changes the ground-air safety switch and/or the nose gear scissor switch may not be activated rendering the nose wheel steering and power brakes inoperative. Increase the weight on the nose gear by pushing the control stick forward to gain nose wheel steering.

6. Drag chute – Deploy (if required)
   To obtain maximum aerodynamic braking, deploy drag chute as soon as nosewheel is on the ground.
   a. Pull drag chute handle straight aft to the stop (about 2 inches)
   b. Hold handle firmly against stop until opening shock of the chute is felt
   c. Release handle

CAUTION

- BECAUSE THE LOCATION OF THE DRAG CHUTE WILL CAUSE A NOSE-DOWN PITCHING MOMENT WHEN DEPLOYED, TO PREVENT DAMAGE TO THE AIRCRAFT, DO NOT DEPLOY THE CHUTE UNTIL ALL THREE GEARS ARE ON THE GROUND.

- DURING THE LANDING GROUND ROLL, DO NOT OPERATE THE SPEED BRAKES WHILE NOSEWHEEL STEERING OR POWER/ANTISKID BRAKES ARE REQUIRED.

- OVERRUNNING ARRESTOR CABLES AT MEDIUM TO HIGH SPEEDS MAY CAUSE INADVERTENT ENGAGEMENT OF ARRESTOR HOOK IN STOWED POSITION. DO NOT LOWER NOSEWHEEL IMMEDIATELY. MAINTAIN A SLIGHT NOSE-UP ATTITUDE UNTIL PAST THE ARRESTING CABLE.

CROSSWIND LANDING

Wind drift may be compensated for by crabbing or the wing down method or a combination of both for approach and landing. In strong crosswinds the crab method or a combination of the two methods is more suitable. A maximum crosswind component of 25 knots is permissible on a dry runway. Landing in excess to this component is not recommended as alignment on the runway is difficult and because such components are greater than those recommended for drag chute deployment.
LAND flaps are used with the maximum crosswind component. The nose should be lowered immediately after touch down and nosewheel steering should be engaged before deploying the drag chute. For dry runway conditions the drag chute may be deployed in 90° crosswinds of 25 knots or 45° crosswinds of 35 knots provided nosewheel steering is engaged. The aircraft tends to weather-vane but directional control can be maintained by nosewheel steering. After landing, some difficulty may be encountered in releasing the drag chute; however, turning the aircraft directly into the wind should solve this difficulty. When landing on wet or icy runways the maximum permissible crosswind component should be adjusted accordingly.
A weather-vaning effect of the drag chute may be sufficient to cause a skid, therefore, the pilot should be prepared to jettison the drag chute. The fact that slick runway conditions reduce braking capability make it desirable to obtain the initial braking effect of the drag chute even though it may be necessary to jettison it later to retain directional control. Under extreme weather conditions when low visibility prevents the pilot from seeing that directional control is being maintained, the drag chute should be jettisoned.

NOTE

- Increase approach and touchdown speed 5 knots for each 10 knots of effective crosswind velocity, if landing with LAND flaps.
- Do not actuate the nosewheel steering button unless the nosewheel and rudder pedals are aligned. If the pedals are deflected when the nosewheel steering button is actuated, clutch friction within the steering system may cause an undesired turn as the pedals are moved to align with the nosewheel.

HEAVYWEIGHT LANDING

When a heavyweight landing must be made, adjust the approach and touchdown airspeeds for gross weight. Refer to the landing charts in the Appendix for the airspeed at any landing gross weight. Fly a wider than normal pattern or make a straight-in approach. This is especially important on approaches under high temperature or high altitude landing conditions.
Rate of descent should be monitored closely and not allowed to become excessive. Be prepared to use afterburning thrust if necessary (refer to Section VI and the Appendix for charts showing the variation of flight performance to expect). Under marginal conditions, a straight-in approach is recommended.
In addition, minimize drag by using a TAKEOFF-flap or gear-up configuration for the approach, changing to the final landing configuration when the landing is assured.
Under certain conditions of forward center of gravity, TAKEOFF flaps must be used for landing. If landing roll distance is a major consideration, use LAND flaps to reduce the touchdown speed and delay extension until the flare is assured.

WARNING

- UNDER HEAVYWEIGHT CONDITIONS, THE AFTERBURNER WILL HAVE TO BE USED IF A GO-AROUND IS ATTEMPTED WITH THE LANDING GEAR EXTENDED AND FLAPS IN THE LAND POSITION.
- WITH MAXIMUM LANDING WEIGHT (21500 LBS) DO NOT EXCEED 300 FT/MIN SINK RATE DURING TOUCHDOWN.

MINIMUM RUN LANDING

For a landing with minimum ground roll, fly the approach so that close control can be exercised over touchdown point and airspeed. Land as near as possible to the end of the runway, touching down at 140 knots for normal landing gross weight.
Use the speed brakes to aid in controlling touchdown point and speed as well as for maximum drag during the rollout.
Plan the chute deployment so that it blossoms as the nosewheel touches down. Smoothly apply antiskid brakes with constantly increasing pedal pressure. If cycling occurs, indicating maximum braking, reduce pedal force.

CAUTION

TO PREVENT DAMAGING BRAKES, TIRES, OR WHEELS DUE TO HEAT, SUFFICIENT TIME MUST BE ALLOWED BETWEEN MAXIMUM EFFORT STOPS FOR COOLING THE BRAKES TO HANDLING TEMPERATURES.

NOTE

Cycling of the antiskid system can be detected by the change in longitudinal deceleration as braking action is automatically released and reapplied by the antiskid system.

LANDING ON SLIPPERY RUNWAYS

To land on wet or icy runways use the same procedure as for a minimum run landing. Leave the flaps at LAND during the landing roll for maximum aerodynamic drag. Refer to the landing charts in the Appendix for information on how stopping distance varies with surface condition. Painted areas on runways, taxiways, and ramps are significantly more slippery than non-painted areas. In addition, painted areas sometimes serve as condensation surfaces and it is possible to have wet, frosty or even icy conditions on these areas when the overall weather condition is dry. When conditions of snow or ice exist, the approach ends of the runway are usually more slippery than other areas because of the melting and refreezing of the ice and snow at this point.

TAKEOFF FLAPS LANDING

TAKEOFF flaps landing may be made from a stright-in approach or an overhead approach.

1. If flying a straight-in approach extend the gear prior to intercept the glide path. Slow to final approach airspeed of not less than 195 KIAS minimum. The approach should be slightly flatter than normal. Touchdown speed should be 165 KIAS minimum. Throttle and speed brakes can be used as necessary to control airspeed and touchdown point.
2. If flying an overhead approach, fly a slightly wider than normal pattern. On base leg, maintain 230 KIAS minimum or APC meter 2.5 maximum whichever occurs first and stay out of shaker. Adjust the final airspeed to 195 KIAS minimum.

NOTE

Power can be adjusted from military to idle, if necessary, as BLC is not operative with TAKEOFF flaps.

TOUCH-AND-GO LANDINGS

After touchdown proceed as follows:

1. Throttle – IDLE
2. Wing flaps – TAKEOFF
3. Throttle – MILITARY
4. Speed brakes – IN
5. Trim – As required

Before going airborne:

6. Flap indicator – TAKEOFF
7. Use normal takeoff technique

GO-AROUND

(Figure 2-8).

Make decision to go around as soon as possible and accomplish the following:

1. Throttle – MILITARY (Maximum thrust if necessary)

WARNING

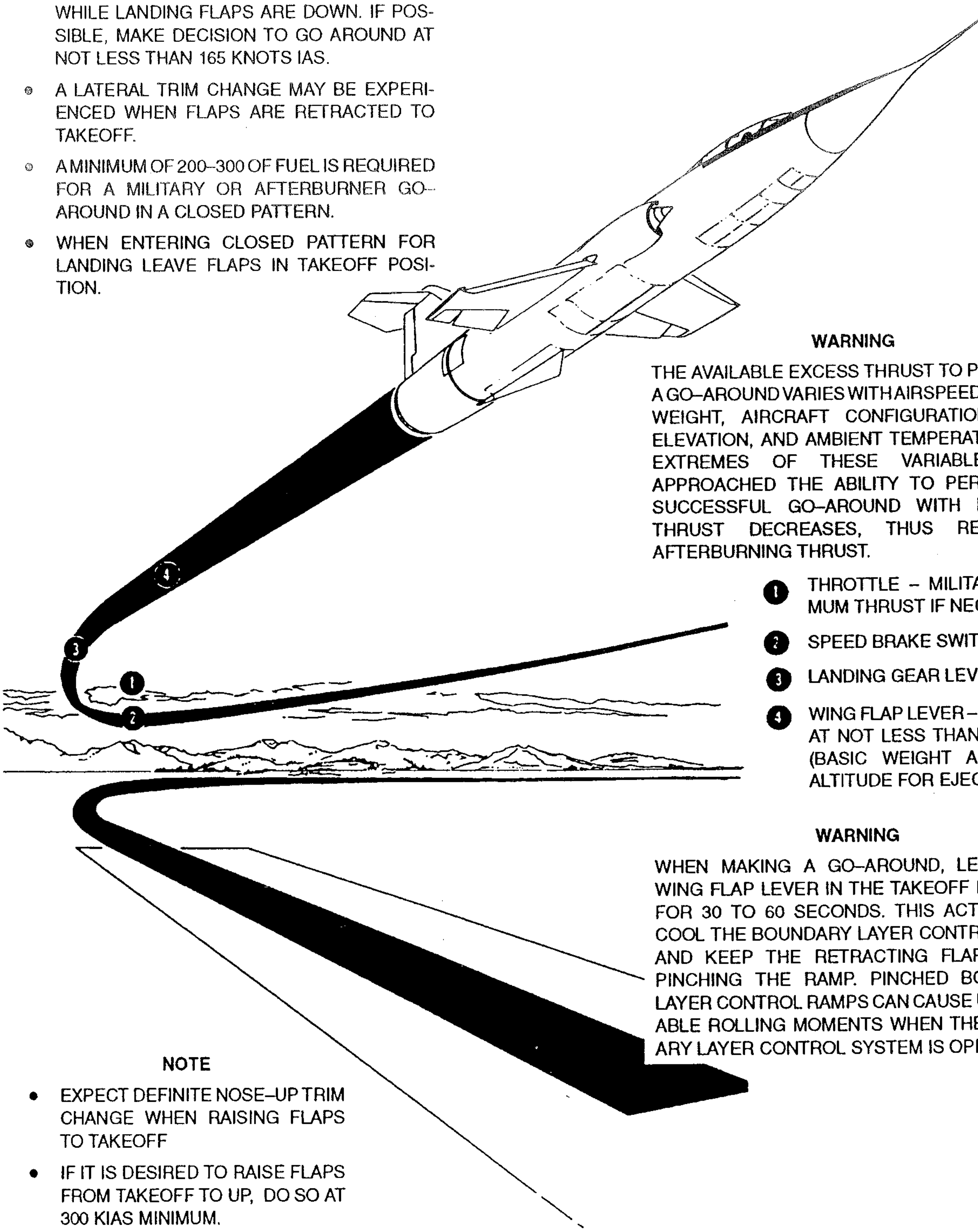
THE AVAILABLE EXCESS THRUST TO PERFORM A GO-AROUND VARIES WITH AIRSPEED, GROSS WEIGHT, AIRCRAFT CONFIGURATION, FIELD ELEVATION AND AMBIENT TEMPERATURE. AS EXTREMES OF THESE VARIABLES ARE APPROACHED THE ABILITY TO PERFORM A SUCCESSFUL GO-AROUND WITH MILITARY THRUST DECREASES, THUS REQUIRING AFTERBURNING. REFER TO SECTION VI AND TO THE APPENDIX FOR ILLUSTRATIONS AND CHARTS SHOWING THE VARIATIONS IN PERFORMANCE TO EXPECT WITH CHANGES IN THESE OPERATING CONDITIONS.

TYPICAL GO-AROUND

BASED ON A GROSS WEIGHT OF 16000 LB. ADJUST APPROACH AND LANDING SPEEDS 5 KNOTS FOR EACH 1000 LB DIFFERENCE IN GROSS WEIGHT OR PORTIONS THEREOF. INCREASE FLAP RETRACTION SPEEDS 5 KNOTS ABOVE COMPUTED FINAL APPROACH SPEED.

NOTE

- THE AIRCRAFT IS SLOW TO ACCELERATE WHILE LANDING FLAPS ARE DOWN. IF POSSIBLE, MAKE DECISION TO GO AROUND AT NOT LESS THAN 165 KNOTS IAS.
- A LATERAL TRIM CHANGE MAY BE EXPERIENCED WHEN FLAPS ARE RETRACTED TO TAKEOFF.
- A MINIMUM OF 200–300 OF FUEL IS REQUIRED FOR A MILITARY OR AFTERBURNER GO-AROUND IN A CLOSED PATTERN.
- WHEN ENTERING CLOSED PATTERN FOR LANDING LEAVE FLAPS IN TAKEOFF POSITION.



WARNING

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1. THROTTLE – MILITARY MAXIMUM THRUST IF NECESSARY
2. SPEED BRAKE SWITCH – IN
3. LANDING GEAR LEVER – UP
4. WING FLAP LEVER – TAKEOFF, AT NOT LESS THAN 180 KIAS (BASIC WEIGHT AND SAFE ALTITUDE FOR EJECTION)

WARNING

WHEN MAKING A GO–AROUND, LEAVE THE WING FLAP LEVER IN THE TAKEOFF POSITION FOR 30 TO 60 SECONDS. THIS ACTION WILL COOL THE BOUNDARY LAYER CONTROL RAMP AND KEEP THE RETRACTING FLAPS FROM PINCHING THE RAMP. PINCHED BOUNDARY LAYER CONTROL RAMPS CAN CAUSE UNDESIRABLE ROLLING MOMENTS WHEN THE BOUNDARY LAYER CONTROL SYSTEM IS OPERATING.

NOTE

- EXPECT DEFINITE NOSE–UP TRIM CHANGE WHEN RAISING FLAPS TO TAKEOFF
- IF IT IS DESIRED TO RAISE FLAPS FROM TAKEOFF TO UP, DO SO AT 300 KIAS MINIMUM.

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Figure 2-8

2. Speed brakes – IN
3. LDG GEAR – UP

When definitely airborne and rate of climb is established.

4. Wing flaps – TAKEOFF, at not less than 180 KIAS (basic weight) at safe altitude for ejection

NOTE

- Select TAKEOFF flaps only after a definite climb is established.
- Expect a definite nose-up trim change when raising flaps to takeoff position.

5. Wing flaps – UP if desired (at not less than 300 KIAS)

WARNING

WHEN MAKING A GO-AROUND, LEAVE THE WING FLAP LEVER IN THE TAKEOFF POSITION FOR 30 TO 60 SECONDS. THIS ACTION WILL COOL THE BLC RAMP AND KEEP THE RETRACTING FLAPS FROM PINCHING THE RAMP. PINCHED BLC RAMPS CAN CAUSE UNDESIRABLE ROLLING MOMENTS WHEN THE BLC SYSTEM IS OPERATING.

AFTER LANDING

NOTE

Drag chute should be jettisoned in the appropriate area as local procedures dictate.

1. Wing flaps – TAKEOFF

NOTE

Leave flaps in TAKEOFF position for a minimum of one minute to ensure sufficient cooling of the flaps after BLC operation.

2. Speed brakes – IN
3. Trim – TAKEOFF
4. Radar – SBY
5. LANDING LIGHT – OFF
6. Ejection seat:
   a. Swivel guard of lower firing handle – UP
   b. Safety pin of upper firing handle – Installed (if desired)

CAUTION

DO NOT INSTALL PIN WHILE CANOPY IS OPEN AND ENGINE IS RUNNING.

7. RAIN REMOVER switch – Check then OFF or OFF if used

NOTE

If rain remover has not been used during flight, turn switch ON for not more than 30 seconds, then OFF to remove condensation and protect rain remover shutoff valve from corrosion. If visible moisture is dissipated before 30 seconds, turn switch OFF.

8. IFF and TACAN – OFF, as desired
9. Heaters and defogger, pitot heat, and engine anti-ice – OFF
10. Canopy – Locked down or full open

CAUTION

WHEN UNLOCKING THE CANOPY, THE PILOT SHALL FIRMLY GRIP THE CANOPY LIFT HANDLE TO AVOID LOSS OF OR DAMAGE TO THE CANOPY DURING HIGH OR GUSTY WINDS. WHEN OPENING THE CANOPY TO THE FULL-OPEN POSITION, A FIRM GRIP ON THE CANOPY LIFT HANDLE IS NECESSARY UNTIL THE CANOPY IS IN THE FULL-OPEN LOCKED POSITION.

ENGINE SHUTDOWN

1. Wing flaps – UP
2. IN terminal error – Record

NOTE

- The IN terminal error may be performed using the OTF technique.
- Terminal error referred to "00" is not reliable.

3. IN – OFF
4. RADAR – OFF
5. All electrical and electronic equipment – OFF
6. Temperature rheostat full hot – If required

NOTE

Leave temperature in full hot position for 30 seconds in order to remove condensation from the air conditioning system.

7. Run engine for 3 minutes at IDLE for proper cooling

NOTE

Operation during taxi can be considered as part of this time.

8. Fast erect standby attitude indicator before engine shutdown, if not in normal position
9. Open canopy

NOTE

Canopy seal keeps inflated if engine shuts-down with canopy in closed position.

10. Throttle – OFF

CAUTION

DO NOT LEAVE THE REAR COCKPIT, PRIOR TO ENGINE SHUTDOWN.

NOTE

Check engine run-down time (will average 45 seconds) and that engine decelerates freely. Listen for any excessive noise during shutdown.

AFTER SHUTDOWN

1. Safety pin of upper firing handle – Installed if not done after landing
2. Swivel guard of secondary firing handle – UP (Check)
3. Remove safety clip and open quick release box of combined harness
4. Dinghy lowering line – Disconnect from life vest
5. Leg lines – Release
6. PEC-pilot portion – Detach
7. Personal leads – Disconnect

8. External stores safety pins — Installed
9. Aircraft forms — Complete

CAUTION

IN ADDITION TO ESTABLISHED REQUIREMENTS FOR REPORTING ANY SYSTEM DEFECTS OR UNUSUAL AND EXCESSIVE OPERATIONS, THE PILOT WILL ALSO MAKE ENTRIES IN AIRCRAFT FORMS TO INDICATE WHEN ANY LIMITS PUBLISHED IN THE FLIGHT MANUAL HAVE BEEN EXCEEDED.

STRANGE FIELD PROCEDURES

If it is necessary to land at an airfield where normal ground support is not available, the pilot will be responsible for performing or at least closely supervising the required aircraft service. The following instructions apply:

1. Complete aircraft preflight inspection and postflight inspection in accordance with the aircraft turnaround procedures in the checklist
2. Use fuel, oil, and hydraulic fluid to service this aircraft as required by the following specifications:
   - Fuel — JP-8, MIL-T-83133 (Refer to Figure 2-9)
   - Oil — MIL-I-7808
   - Hydraulic fluid — MIL-H-5606 (NATO H-515) or MIL-H-83282 (NATO H-537)
     Reservoir Capacity:
     No. 1 — 0.49 US gallon
     No. 2 — 1.83 US gallons
   - Oxygen — Liquid MIL-O-27210
   - Nitrogen — MIL-N-6011 Grade A Type 1 (Water-pumped or dry air)
3. External air source for starting (Refer to Figure 1-79)
4. External electrical power 115 V AC, 400 Hz and 28 V DC (Refer to Figure 1-79)
5. Single-point refueling pressure — 50 psi

If airfield is not a DWP stored in DTM, if possible, do not move the aircraft after IN shutdown. For next flight perform GC alignment whitout depress IP pushbutton on IN/CDU.

NOTE

The navigation performances are degraded.

FLIGHT INTO A FOREIGN COUNTRY WHERE ONLY F-40 FUEL IS AVAILABLE

In case of a planned flight into a foreign country where only F-40 fuel will be available for additional flights, the one-time ferry flight into that specific country is permissible with F-40 fuel altough the main and afterburner fuel controls already set to F-34. In this case the following degradation in engine performance have to be expected:

- Longer time to start and accelerate, with possible missed-starts or start-stalls
- Slower acceleration throughout the operating range
- Lower than normal afterburner thrust

FLIGHT WITH F-34 FUEL AND FUEL CONTROL SETTINGS ON F-40

A one-time flight with F-34 fuel with fuel control settings on F-40 is permissible, provided the following precautions are followed:

a. The flight must be a point-to-point subsonic cruise type
b. Rapid throttle movements are allowed only under emergency conditions
c. Throttle movement during ground operation is as follows:
   - idle to military 6 seconds
   - 85% to military 3 seconds
   - stabilize in military before A/B initiation
   - stabilize in minimum A/B
   - advance to maximum A/B in 3 seconds
d. The above mentioned throttle advance times should be doubled when operating above 20000 ft

FUEL GRADE PROPERTIES AND LIMITS

USE	FUEL TYPE	GRADE	NATO SYMBOL	U.S. MILITARY SPECIFICATION/ COMMERCIAL	SPECIFIC GRAVITY	FREEZE POINT		LIMITS
						°F	°C	
Primary Fuel	Kerosene	JP-8 Jet A-1	F-34 F-35	MIL-T-83133 ASTMD 1655	.840-.775 .840-.775	−58 −53	−50 −47	1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6
Alternate Fuel	Kerosene Wide Cut Gasoline	Jet-A JP-4	None F-40	ASTMD 1655 MIL-T-5624	.840-.755 .802-.751	−40 −72	−40 −58	1, 2, 3, 4, 5, 6 6
Emergency Fuel	Aviation Gasoline (Avgas)	80/87 100/130 115/145	F-12 F-18 F-22	MIL-G-5572 MIL-G-5572 MIL-G-5572	.706 .706 .706	−76 −76 −76	−60 −60 −60	1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6 1, 2, 3, 4, 5, 6

LIMITS

1. Whenever the use of alternate fuel is necessary, the specific gravity setting on the main fuel control and afterburner fuel control shall be adjusted to correspond with a mid-range or average value of the specific gravity of the fuel selected. The specific gravity or average value of authorized fuels can be obtained from the above table. The procedures for changing the specific gravity setting on the fuel controls are pusblished in the aircraft power plant manuals. Whenever the specific gravity adjustments are changed from their standard preset point, an entry will be made in DD Form 781A. This entry can only be cleared when the aircraft is reserviced with the primary fuel and the specific gravity adjustments have been reset to the standard setting.
2. There is no operating time limit with alternate fuels. Use of emergency fuel is restricted to a maximum of 6 hours operation.
3. Airstarts initiated as soon as possible will assure best possible condition for restart.
4. Engine ground and aerial start times will increase when using JP-8, Jet A-1 and Jet-A fuels. Refer to Section III "Emergency Procedures" Figure 3-2 for engine estimated air start envelope.
5. Engine throttle transients, A/B light-off capability and thrust are not degraded by use of JP-8, Jet A-1 or Jet-A. A slight improvement in stall margin will result with the use of JP-8, Jet A-1 or Jet-A.
6. If there is any indication of improper fuel handling procedures, a fuel sample should be taken in a glass container and observed to fogginess, prescence of water, or rust. The primary fuels JP-8 and fuels identified by NATO symbols F-34, F-35 and F-40 contain an icing inhibitor.
7. a. Whenever the use of aviation gasoline is required, the aircraft will be restricted to a one-time flight not to exceed six hours duration. Specific gravity adjustments to the main and afterburner fuel controls are not required. Neigher is the addition of lubricating oil additives required. When using AVGAS there is no restriction on afterburner operation, but the aircraft ceiling is limited to 35M and aircraft velocity shall not exceed subsonic speed at any altitude. In addition to these limitations, certain engine parameters may be degraded under some atmospheric conditions:

   (1) Longer time to start and accelerate with possible missed starts or start stalls.
   (2) Maximum engine rpm and EGT may not be attained.
   (3) Slow acceleration throughout the operating range.
   (4) Reduced engine thrust.
   (5) Reduced aircraft range.

   b. If aircraft exceeds 6 hours of operation of AVGAS, drain aircraft fuel system completely and refuel with primary fuel. Inspect turbine exhaust nozzle area and perform ground run check. If no defects or engine malfunctions are found, release aircraft for flight.

CAUTION

- AVOID FLYING AT ALTITUDES WHERE INDICATED OAT IS BELOW THE FREEZE POINT OF THE FUEL. PRIOR TO USING EMERGENCY COMMERCIAL FUEL, OBTAIN FREEZE POINT FROM VENDOR OR AIRLINE SUPPLYING THE FUEL; THEN FOLLOW THE LIMIT.
- ABOVE 22000 FEET THE TIME FOR ENGINE RESTART MAY BE 15 SECONDS LONGER WHEN USING JP-8/F-34 INSTEAD OF JP-4/F-40.

Figure 2-9

e. Throttle chops are permitted, however, the engine may decelerate slower and idle at higher RPM

CAUTION

CONTINUED OPERATION USING NATO F-34 FUEL WITH THE CONTROL SPECIFIC GRAVITY ADJUSTMENTS SET ON F-40 FUEL IS PROHIBITED.

INS ALIGNMENT PROCEDURE

NOTE

- The procedure contained in this Section are based on the assumption that the DTM has been inserted in the Master IN/CDU before INS switch-on (ALN selection). If DTM is not inserted, the pilot shall be aware of the following:
  - during INS alignment, the IN aligns using the last stored INS PP
  - IN NAV function is available only for IP (00).
- The DTM shall not be inserted/removed during INS alignment and navigation phases.

GYROCOMPASS ALIGNMENT

1. IN function/mode selector knobs – ALN/GC
2. IN – Check FAIL lamp out
3. IN/CDU – (AUTOMATIC POWER ON TEST). After 5 seconds max., check ″CDU OK″ indication displayed.
   If ″CDU FAIL″ indication is displayed: ABORT

NOTE

If ″DTM FAIL″ appears on IN/CDU, switch off the INS and check for correct DTM insertion. Repeat steps from 1 to 3. If ″DTM FAIL″ still displayed the INS aligns on the last stored PP. Only the TACAN steering mode is available.

4. Master IN/CDU rotary switch – Rotate to select requested initial point
5. Master IN/CDU – Press IP within 2 minutes from IN ALN selection

NOTE

In case no DTM Initial Position is available or not selected by the pilot, the system carries out the alignment on the last stored present position. In this case, the navigation subsystem performance is degraded.

If fast GC alignment is requested:

6. IN – Check status 3. Set to NAV when RDY NAV is steady. Check RDY NAV lamp out

If full GC alignment is requested:

6. IN – Check status 1. Set to NAV when RDY NAV flashes. Check RDY NAV lamp out

NOTE

In ALN/GC mode, rotating the rotary switch on Master IN/CDU, allows presentation of stored waypoints.

STORED HEADING ALIGNMENT

NOTE

- Stored heading alignment shall be carried out provided that an INS full GC alignment has been performed before the INS switch-off.
- The aircraft shall not be moved after the last INS switch-off.

- No IN/CDU IP selection is available.

1. IN function/mode selector knobs – ALN/STO
2. IN – Check FAIL lamp out
3. IN/CDU – (AUTOMATIC POWER ON TEST). After 5 seconds max., check for "CDU OK" indication displayed.
   If "CDU FAIL" indication is displayed: ABORT

NOTE

If "DTM FAIL" appears on IN/CDU, switch off INS and check for correct DTM insertion. Repeat steps from 1. to 3. If "DTM FAIL" still displayed the INS aligns on the last stored PP. Only the TACAN steering mode is available.

When RDY NAV flashes:

4. IN – Check STATUS 3. Set to NAV and check RDY NAV lamp out

NOTE

In ALN/STO mode, rotating the rotary switch on IN/CDU will allow presentation of stored waypoints data. Selection of different initial positions is not enabled.

IN/CDU OPERATION

1. DTM – Insert in the IN/CDU (front cockpit) before IN switch-on
2. IN – Switch on
3. IN/CDU – Check no "DTM FAIL" displayed

NOTE

If DTM is not inserted in the IN/CDU, the "NO DTM" is displayed.

After IN/CDU self test:

4. IN/CDU – Check "CDU OK" displayed

NOTE

If failure occurs the "CDU FAIL" is displayed: ABORT.

SELECTION OF DESTINATION WAYPOINT (FLY TO FUNCTION)

1. Master IN/CDU rotary switch – Rotate to select the desired destination waypoint
2. Master IN/CDU rotary switch – Hold for 1 sec to select the destination waypoint

IN/CDU NAVIGATION DATA PRESENTATION

When a destination waypoint is selected on Master IN/CDU:

1. IN/CDU – Press NAV pushbutton. Check navigation parameters displayed
2. IN/CDU – Press PAGE pushbutton to display the desired navigation parameters. After pressing the PAGE pushbutton the third time, the IN/CDU display return to the first page

NOTE

Entering NAV mode from IN ALN, the first page is displayed.

IN/CDU STATION POINT LISTING

1. Master IN/CDU – Press LIST pushbutton. Check destination waypoint coordinates displayed. Rotate the rotary switch to display consequently any other stored point

NOTE

The station identifier number and ICAO label flash during station point listing.

2. Master IN/CDU – Press PAGE pushbutton

NOTE

- The PAGE pushbutton may be pressed up to 2 times.
- By pressing the PAGE pushbutton the information relative to the station height above the sea (except for mark point) and TACAN channel (only for radio station) are displayed.

At any time to resume previous format:

3. Master IN/CDU – Press NAV pushbutton

NOTE

At any time, after pressing NAV pushbutton, the first page of navigation parameter is displayed.

MARK POINT ACQUISITION

NOTE

- Only two mark points may be stored (identifier No. 59 and 60).
- Subsequent mark points acquisition shall delete the previously stored mark points.
- Mark points are not stored into the DTM: after IN switch-off, mark points data are lost.
- After mark point acquisition, the mark point may be used as any other destination waypoint.
- Mark point acquisition shall be carried out with the OTF technique.
- A MARK label on the second row is displayed as mark point acquisition feedback. After 10 seconds the MARK label disappears and the previous format is resumed.

1. Master IN/CDU – Press MARK pushbutton. Check MARK format

ON TOP FIX

NOTE

- OTF procedure is available only for the current destination waypoint.
- During OTF procedure the actuation of NAV or LIST pushbutton on the IN/CDU causes the OTF rejection.
- The MARK/IP and PAGE pushbuttons are disabled.

1. Master IN/CDU – Press FIX/ACC pushbutton. Check fixing error displayed

To accept fixing error:

2. FIX/ACC pushbutton – Press within 30 seconds

To reject fixing error:

3. Wait 30 seconds

   or:

   NAV or LIST pushbutton – Press

SECTION III

EMERGENCY PROCEDURES

TABLE OF CONTENTS

INTRODUCTION

This section covers the operation of the aircraft during emergency/abnormal conditions. It includes discussions of problem indications and corrective actions as well as procedural steps when applicable. Adherence to these guidelines will ensure maximum safety for the pilot and/or aircraft.

The situations covered represent the most probable malfunctions. However, multiple emergencies, weather, or other factors may require modification of the recommended procedures. Accomplish only those steps required to correct or manage the problem.

When dealing with emergency/abnormal conditions, pilots must determine the most correct action using SOUND JUDGMENT, COMMON SENSE, and FULL UNDERSTANDING OF APPLICABLE SYSTEMS.

Critical emergency procedures are presented in **BOLD FACE** capital letters. Pilots shall be able to immediately accomplish these procedures in the pusblished sequence without reference to the checklist.

Three basic rules apply to all emergency situations. The basic rules are not repeated in each of the procedures listed. However, in ALL EMERGENCIES, THE OVERRIDING CONSIDERATIONS SHALL BE TO:

1. Maintain aircraft control
2. Analyze the situation
3. Take proper action

NOTE

- The ground, takeoff, in-flight and landing emergency procedures are sequenced alphabetically as outlined in the Table of Contents.
- Decision factors are provided as a guide in selecting certain procedures.

The terms "Land as soon as possible" (ASAP) and "Land as soon as practical" are used throughout this section. These terms are defined as follows:

Land as soon as possible (ASAP) – An emergency will be declared. A landing should be accomplished at the nearest suitable airfield, considering the severity of the emergency, weather conditions, field facilities, ambient lighting, aircraft gross weight, and command guidance.

Land as soon as practical – Emergency conditions are less urgent, and although the mission is to be terminated, the degree of the emergency is such that an immediate landing at the nearest adequate airfield may not be necessary.

GROUND EMERGENCIES

EMERGENCY ENTRANCE

The procedure to be used by rescue personnel in assisting the pilot from the aircraft following a crash landing is shown in Figure 3-1.

FIRE DURING START

Illumination of the FIRE warning lights and/or other evidence of fire during engine starting is/are an indication of a broken or disconnected fuel line.

1. THROTTLE – OFF
2. **FUEL SHUTOFF – OFF**
3. 1 START 2 SWITCHES – STOP-START
4. Perform GROUND ABANDONMENT procedure

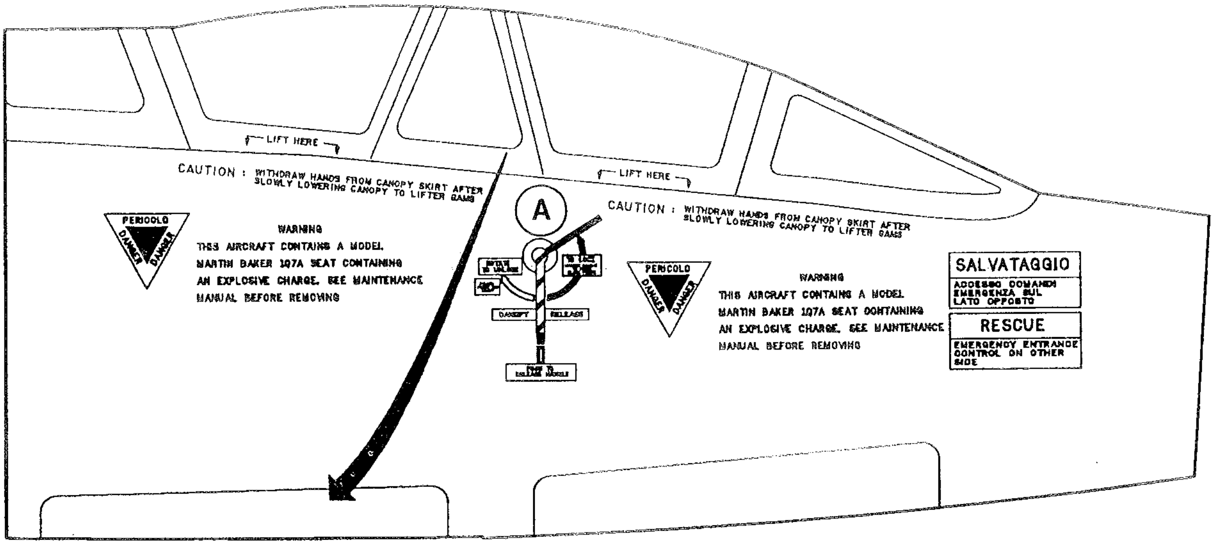
GROUND ABANDONMENT

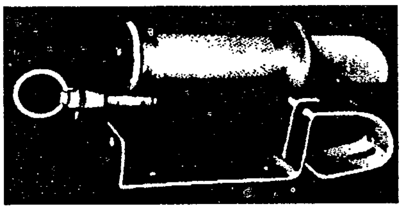
In any situation where rapid abandonment is required, two methods of removing the canopy shall be considered: manual opening and jettison.

If necessary and circumstances permit:

1. **EJECT**

EMERGENCY ENTRANCE





CANOPY BREAKING TOOL (BOTH COCKPIT)

1 IF CANOPY IS LOCKED, UNLOCKED BY ROTATING THE EXTERNAL LOCKING LEVER Ⓐ AFT AND OPEN CANOPY.

2 IF AIRCRAFT IS ON FIRE, JETTISON CANOPY BY PULLING CANOPY JETTISON T–HANDLE Ⓑ

WARNING

THE FORWARD CANOPY WILL FIREFIRST AND THE REAR CANOPY WILL FIRE APPROXIMATELY 3 SECONDS LATER.

3 INSERT SAFETY PIN INTO SEAR OF EJECTION GUN

4 INSERT SAFETY PIN IN UPPER FIRING HANDLE (NOT REQUIRED IF CANOPY IS JETTISONED)

5 ENSURE 0_2 LEVER IS IN NORMAL 0_2 AT 0_2 CONTROL PANEL

6 REMOVE OXYGEN MASK OR OPEN FACE PLATE OF PRESSURE HELMET (UNLESS CREWMEMBER IS IN IMMEDIATE DANGER OF FIRE)

7 RAISE SWIVEL GUARD (NOT REQUIRED IF CANOPY IS JETTISONED)

8 THROTTLE–OFF

9 LEG LINES–RELEASE

10 PEC QUICK DISCONNECT–RELEASE.

11 UNLOCK AND OPEN QUICK RELEASE BOX

12 SURVIVAL KIT LINE – DISCONNECT

13 REMOVE PILOT GENTLY TO AVOID AGGRAVATING POSSIBLE INTERNAL INJURIES

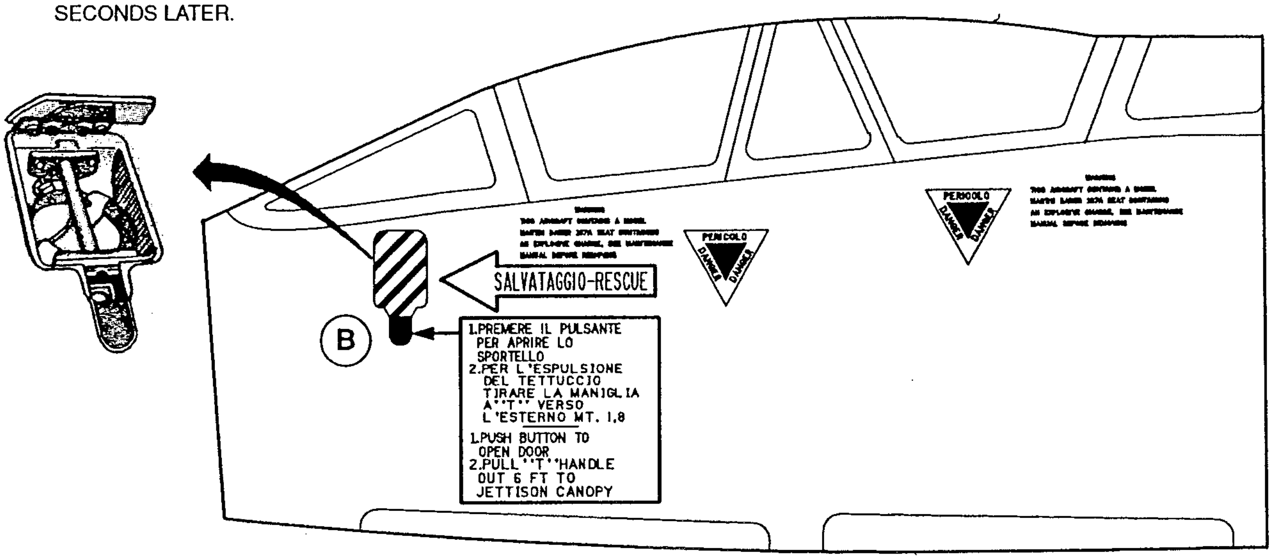


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Figure 3-1

WARNING

THE MINIMUM EJECTION SPEED AT ZERO ALTITUDE IS 60 KIAS TO CLEAR THE CANOPY FROM TRAJECTORY.

If time permits:

1. **SWIVEL GUARD – UP**
2. **LEG LINES – RELEASE**
3. **RELEASE BOX – OPEN**
4. **SURVIVAL PACK LINE – DISCONNECT**
5. **PEC PILOTS PORTION – DISCONNECT**

If time is critical:

6. **CANOPY – JETTISON**

If time is not of prime importance or if no fire exists:

6. **CANOPY – OPEN MANUALLY**

WARNING

- TO AVOID INJURY TO REAR COCKPIT OCCUPANT, THE FRONT CANOPY SHOULD BE JETTISONED FIRST.

- THE INTERNAL CANOPY JETTISON HANDLE IN THE FRONT COCKPIT, WHEN PULLED, WILL JETTISON INSTANTANEOUSLY ONLY THE FRONT CANOPY. IT WILL NOT JETTISON THE REAR CANOPY. THE INTERNAL CANOPY JETTISON HANDLE IN THE REAR COCKPIT, WHEN PULLED, WILL JETTISON ONLY THE REAR CANOPY, 3 SECONDS AFTER ACTUATION. IT WILL NOT JETTISON THE FRONT CANOPY.

NOTE

If survival pack line is not disconnected, pilot will still be able to evacuate the cockpit, as the connected line allows approximately 15 feet of movement. Pulling the manual override handle will also release leg lines and PEC pilot portion, but it should be noted that pilot will be pushed forward by released parachute pack (weight 30 pounds) and personal leads may cause entanglement during evacuation.

TAKEOFF EMERGENCIES

ABORT/BARRIER ENGAGEMENT

During takeoff an emergency may present a situation where groundspeed, aircraft gross weight, existing barrier, runway conditions and other such factors will not permit a safe barrier engagement. In this situation, the only possible course of action may be a ground-level ejection.
Refer to Figure 3-2 for maximum barrier engagement speeds; refer to EJECTION for minimum ejection airspeeds.

If abort is required:

1. **THROTTLE – IDLE**

WARNING

NOSEWHEEL STEERING AND ANTISKID BRAKES BECOME INOPERATIVE WITH THROTTLE OFF. REQUIRED BRAKE PEDAL PRESSURE WILL BE GREATER THAN NORMAL TO EFFECT BRAKING.

CAUTION

ENGINE SPEEDS ABOVE IDLE MAY CAUSE DRAG CHUTE FAILURE.

MAXIMUM BARRIER ENGAGEMENT SPEEDS

AIRCRAFT WEIGHT-POUNDS	BAK-9 (1) (6)	BAK-12 (1) (6)	AAE-44B-2C (1) (2) (3) (4) (5)	AAE-44B-2D (1) (2) (3) (5)
	MAXIMUM ENGAGEMENT SPEED - KNOTS			
14000	188	181	185	135
15000	187	180	185	135
16000	186	179	185	135
17000	185	178	185	135
18000	184	177	184	135
19000	183	176	184	135
20000	182	175	184	130
21000	181	174	184	130
22000	180	173	183	130
23000	179	172	183	130
24000	178	171	183	125
25000	177	170	182	125
26000	176	169	182	125
27000	175	168	181	125
28000	174	167	180	125

NOTE

(1) TRY TO CONTACT THE BARRIER AT THE RUNWAY CENTERLINE

(2) THE ARRESTMENT GEAR MAY BE ENGAGED UP TO 7.5 METERS (25 FEET) OFF CENTER

(3) WHEN THE ARRESTMENT GEAR IS ENGAGED OFF CENTER, THE AIRCRAFT MAY BE DISPLACED TO ONE SIDE AS FAR AS THE EDGE OF THE RUNWAY PAVEMENT

(4) WHEN ENGAGEMENT SPEED FOR AAE-44-B-2C EXCEEDS 100 KNOTS, THE AIRCRAFT WILL BE PULLED REARWARD AFTER ARRESTMENT

(5) LIMITING FACTOR IS NOSE GEAR STRENGTH

(6) LIMITING FACTOR IS DESIGN HOOK STRENGTH

Figure 3-2

If necessary:

2. EXTERNAL STORES – JETTISON

Retain stores if the aircraft can be safely stopped within the remaining runway. Refer to EXTERNAL STORES EMERGENCY/SELECTIVE JETTISON procedure.

WARNING

IF POSSIBLE, JETTISONING OF EXTERNAL STORES SHOULD TAKE PLACE 3000 FEET PRIOR TO BARRIER ENGAGEMENT.

NOTE

- External stores may create a fire hazard when the aircraft stops in the barrier.
- Jettisoned pylon tanks may strike the adjacent main landing gear but will not alter the course of the aircraft.
- Stores should be jettisoned prior to landing in those cases of a known emergency.
- Retain empty tip tanks if they are the only external store.

3. **HOOK RELEASE – PRESS**
4. **DRAG CHUTE – DEPLOY**

WARNING

IF PYLON STORES ARE JETTISONED AFTER DRAG CHUTE DEPLOYMENT, THEY MAY DAMAGE OR COLLAPSE THE DRAG CHUTE.

5. Brakes – APPLY

If barrier engagement is required:

6. Brakes – Release prior to engagement

CAUTION

A LOCKED WHEEL, REGARDLESS OF TIRE STATE, WILL SNAG OR CUT THE ARRESTING HOOK CABLE.

7. Barrier – Contact at 90° angle and as close as possible to the centerline

NOTE

Correction for yaw after hook engagement should be avoided to prevent nose gear damage from side loads.

If aircraft becomes uncontrollable:

8. Throttle – OFF
9. FUEL SHUT-OFF switch – OFF
10. Perform GROUND ABANDONMENT procedure when aircraft has stopped

WARNING

DO NOT UNFASTEN THE COMBINED HARNESS UNTIL THE AIRCRAFT HAS COME TO REST.

AFTERBURNER BLOWOUT

Afterburner blowout is indicated by:

– An abnormally low EGT (less than 500° C)
– Nozzle position reading of less than 7.0
– A definite loss of thrust

If decision is made to stop:

1. ABORT

If takeoff is continued:

1. THROTTLE – MILITARY
2. EXTERNAL STORES – JETTISON
3. Engine instruments – MONITOR

At safe position:

4. Throttle – Minimum practical

WARNING

LOSS OF AFTERBURNER COULD BE AN EARLY INDICATION OF ENGINE FAILURE OR FIRE. DO NOT RECYCLE AFTERBURNER.

5. Confirm fire

If on fire:

6. Eject

If fire cannot be confirmed:

6. Land ASAP

AFTERBURNER SURGE

Afterburner surge is indicated by:

- Unstable feel
- EGT oscillation
- Nozzle position indicator oscillation

If decision is made to stop:

1. ABORT

If takeoff is continued:

1. THROTTLE – MILITARY

WARNING

COMPRESSOR STALL AND FLAMEOUT MAY OCCUR IF THROTTLE IS NOT MOVED OUT OF AFTERBURNER DURING SURGE, PREFERABLY BEFORE MORE THAN TWO CYCLES OF SURGE. DO NOT RECYCLE AFTERBURNER.

2. EXTERNAL STORES – JETTISON (IF NECESSARY)
3. Land as soon as practical

CANOPY OPEN/LOSS/BROKEN

If decision is made to stop:

1. ABORT

If takeoff is continued:

1. If canopy is not locked: JETTISON
2. Throttle – MAINTAIN TAKEOFF THRUST UNTIL A SAFE POSITION

CAUTION

DO NOT CHANGE POWER SETTING. THROTTLE MOVEMENT MAY PRECIPITATE AN ENGINE STALL IF ENGINE HAS SUSTAINED FOREIGN OBJECT DAMAGE. ENGINE STALL MAY OCCUR DUE TO PRESSURE CHANGES ACCOMPANYING AFTERBURNER THROTTLING OR SHUTDOWN, AS WELL AS IN CHANGING RPM.

3. Airspeed – KEEP BELOW 300 KIAS to minimize windblast and noise, use speedbrakes as necessary

If necessary:

4. External stores – Jettison

5. Land as soon as practical, using a PRECAUTIONARY PARTIAL POWER PATTERN procedure

ENGINE FAILURE

Engine failure is defined as a complete power failure which, in the pilot's judgment, makes a restart impossible or inadvisable. Examples are engine seizure or explosion.

If decision is made to stop:

1. **ABORT**

If takeoff is continued:

1. **EXTERNAL STORES – JETTISON**

NOTE

> Maximum altitude gain can be achieved by jettisoning stores prior to zoom. The later in the zoom the stores are jettisoned, the less additional altitude will be gained.

2. **ZOOM, IF POSSIBLE AND EJECT**

WARNING

> EJECT WHILE THE AIRCRAFT HAS A POSITIVE RATE OF CLIMB IN A WINGS-LEVEL ATTITUDE. DO NOT PULL UP TOO RAPIDLY AS THE AIRCRAFT WILL STALL BEFORE ALTITUDE HAS BEEN GAINED, EVEN THOUGH THE INITIAL AIRSPEED SEEMED TO BE ADEQUATE. AT 240 KIAS, IT IS POSSIBLE TO ZOOM NEARLY 400 FEET WITH A DEAD ENGINE.

ENGINE OIL LEVEL LOW

ENGINE OIL LEVEL LOW light on indicates oil depletion to 6.4 pints or less.

If decision is made to stop:

1. **ABORT**

If takeoff is continued:

1. **THROTTLE – MAINTAIN TAKEOFF THRUST AND CLIMB**

If necessary:

2. **EXTERNAL STORES – JETTISON**

At safe position:

3. **THROTTLE – MILITARY**
4. **NOZZLE HANDLE – OUT**
5. Land ASAP, using a PRECAUTIONARY PARTIAL POWER PATTERN procedure

EXHAUST NOZZLE FAILS OPEN WITHOUT INDICATION OF ENGINE OIL SYSTEM MALFUNCTION (AFTERBURNING)

The most probable cause is failure of the nozzle area control amplifier or control alternator.

Indicated by:

- A slight reduction in thrust
- EGT approximately 500° C to 550° C
- An increase in nozzle area to approximately 10

NOTE

> Failure will probably not be detected. Afterburner will continue to operate provided throttle is not retarded to sector/core range. As long as afterburning is maintained, immediate corrective action is not required.

If decision is made to stop:

1. **ABORT**

If takeoff is continued:

1. **THROTTLE – MAINTAIN TAKEOFF THRUST**

Sufficient thrust will be available with full afterburning to climb to a safe altitude and to establish a position from which a safe landing with an open nozzle can be accomplished.

At safe position:

2. Throttle – MILITARY
3. Nozzle handle – Out

If nozzle closes:

4. Land as soon as practical (monitor EGT with throttle adjustment)
5. After touchdown: Nozzle handle – In

If nozzle fails to close:

4. If necessary: EXTERNAL STORES – JETTISON
5. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

EXHAUST NOZZLE FAILS OPEN WITHOUT INDICATION OF ENGINE OIL SYSTEM MALFUNCTION (NON-AFTERBURNING)

The most probable cause is failure of the nozzle area control amplifier or control alternator.

Indicated by:

- Significant thrust decrease
- EGT approximately 350° C
- An increase in nozzle area up to approximately 10

WARNING

SUFFICIENT THRUST IS NOT AVAILABLE TO COMPLETE TAKEOFF OR MAINTAIN LEVEL FLIGHT WITH ANY CONFIGURATION.

If decision is made to stop:

1. ABORT

If takeoff is continued:

1. NOZZLE HANDLE – OUT

If necessary:

2. EXTERNAL STORES – JETTISON

If nozzle fails to close:

3. THROTTLE – RAPIDLY TO MAX A/B

A 3 to 5 second delay may occur before A/B light is obtained. A/B lights with nozzle failed open are not assured, but the probability of successful light increases as altitude decreases.

If A/B lights:

4. NOZZLE HANDLE – IN

At safe position:

5. THROTTLE – MILITARY
6. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

If nozzle closes:

3. Land as soon as practical
4. After touchdown: Nozzle handle – In

If A/B fails to light:

4. The decision to abort/continue flight/eject must be determined by the pilot using appropriate procedures

EXTERNAL STORES EMERGENCY/SELECTIVE JETTISON

To jettison all external stores during an emergency:

1. EXT STORES JETTISON – Press

If a selective jettison has to be carried out or if a store fails to jettison following an emergency jettison:

1. STORES RELEASE selector switch – As required
2. External stores selector button – Press
3. Droppable stores release button – Press

NOTE

Refer to Section V for external stores jettisoning limit speed.

An unsuccessful jettison of a store is indicated by the relevant external stores selector buttons and position indicators light still lit after the emergency and selective jettison attempt and/or by visual check. If the hang-up condition persists a landing with a hang store has to be made.

WARNING

FOLLOWING AN ATTEMPTED JETTISON OPERATION, ANY STORE WHICH DOES NOT SEPARATE FROM THE AIRCRAFT SHOULD BE CONSIDERED UNLOCKED AND SUSCEPTIBLE OF INADVERTENT RELEASE. THE PILOT SHALL BE PREPARED TO EXECUTE AN IMMEDIATE GO-AROUND SHOULD RELEASE OF STORE OCCUR DURING LANDING.

FIRE

Illumination of the FIRE warning lights during takeoff requires immediate action. The exact procedure to follow varies with each set of circumstances and depends upon altitude, airspeed, length of runway, overrun clearing remaining and location of populated areas.

If decision is made to stop:

1. **ABORT**

When aircraft is under positive control:

2. Throttle – OFF

If takeoff is continued:

1. **THROTTLE – MAINTAIN TAKEOFF THRUST TO SAFE POSITION**

If necessary:

2. **EXTERNAL STORES – JETTISON**

At safe position:

3. **THROTTLE – MINIMUM PRACTICAL SETTING**

NOTE

At safe altitude, retarding throttle to minimum practical may be beneficial in eliminating fire.

4. **CONFIRM FIRE**

Confirm fire by any possible means such as report from ground, other aircraft, engine instruments, FIRE warning lights, fire WARNING LIGHTS TEST, smoke in cockpit or visible smoke trail behind aircraft.

NOTE

- If FIRE warning lights go out upon power reduction and illuminates during testing, a fire is not likely in the areas where fire detectors are installed.
- If the FIRE lights do not illuminate during testing, an explosion may have resulted in serious damage in the areas where fire detectors are installed with a great possibility of fire.

5. **IF ON FIRE – EJECT**

If fire cannot be confirmed but FIRE warning lights remains on:

5. Fresh air scoop – OPEN
6. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure, continuously checking for fire

LANDING GEAR LEVER DOWN-LOCK MALFUNCTION

If gear lever will not move to UP when airborne:

1. Wing flaps – Keep TAKEOFF
2. Airspeed – Keep below 260 KIAS
3. DOWN LOCK MECH OVERRIDE – Press
4. LDG GEAR – UP

LANDING GEAR RETRACTION FAILURE

If gear lever warning light remains on after lever is UP:

1. Wing flaps – Keep TAKEOFF
2. Airspeed – Keep below 260 KIAS
3. LDG GEAR – Recycle once at lowest practical airspeed (takeoff speed + 30 KIAS)

CAUTION

IF GEAR IS DAMAGED, RECYCLING MAY CAUSE MORE DAMAGE AND PREVENT GEAR FROM LOCKING IN DOWN POSITION.

If warning light still remains on:

4. LDG GEAR – DOWN
5. Land as soon as practicable

NOSEWHEEL SHIMMY

1. Nosewheel steering – Release

If decision is made to stop:

2. Hold weight off nose gear
3. Throttle – IDLE. If shimmy is severe and conditions permit: Throttle – OFF

NOTE

If throttle is moved to the OFF position, nosewheel steering, antiskid brakes, and UHF radio will be lost.

If necessary:

4. External stores – Jettison
5. HOOK RELEASE – Press
6. Drag chute – Deploy

If takeoff is continued:

2. LDG GEAR – UP immediately after airborne
3. Throttle – Maintain takeoff thrust and climb

At safe position:

4. Check for FOD

If damage is indicated or a nose gear flat tire is suspected:

5. Land as soon as practical using NOSE GEAR FLAT TIRE procedure

TIRE FAILURE

The following procedure is recommended when a tire fails during takeoff run.
The recommended technique applies to all gross weights and aircraft configurations. Directional control of the aircraft becomes more difficult as aircraft gross weight increases. With main gear tire failure and less than 150 knots, an abort is recommended.

If decision is made to stop:

1. **ABORT**

If takeoff is continued:

1. **THROTTLE – MAINTAIN TAKEOFF THRUST TO SAFE POSITION**

If necessary:

2. **EXTERNAL STORES – JETTISON**

If nose gear tire failed:

3. **LDG GEAR – UP**

WARNING

WITH NOSE GEAR TIRE FAILURE, RETRACT GEAR IMMEDIATELY AFTER AIRBORNE AS ENGINE MAY BE SUBJECTED TO FOREIGN OBJECT DAMAGE.

4. Land as soon as practical, using NOSE GEAR FLAT TIRE procedure

If main gear tire failed:

3. **LDG GEAR – DO NOT RETRACT**
4. Tire – Check for fire (another aircraft/tower)
5. Land as soon as practical, using MAIN GEAR FLAT TIRE procedure

If no fire is evident and mission dictates that gear retraction is necessary:

5. ANTI-SKID – OFF
6. Brakes – Apply
7. LDG GEAR – UP
8. ANTI-SKID – ON

IN-FLIGHT EMERGENCIES

AFTERBURNER FAILS TO LIGHT OR BLOWS OUT

Afterburner failure is indicated by:

- an abnormally low EGT
- nozzle position reading of less than 7.0
- a definite loss of thrust

If afterburner fails to light or blows out during flight:

1. **THROTTLE – MINIMUM PRACTICAL**

WARNING

LOSS OF AFTERBURNER COULD BE AN EARLY INDICATION OF ENGINE FAILURE OR FIRE. DO NOT RECYCLE AFTERBURNER.

At safe position:

2. **CONFIRM FIRE**

If on fire:

3. **EJECT**

If fire cannot be confirmed:

3. Land ASAP

AIR START/STALL CLEARING

Indication of compressor stall will be:

- Loss of thrust normally begins with a chug or pop, followed by mild vibration
- EGT – usually 700° C to 800° C. EGT may be as low as 600° C. If stall persists, may rise above 800° C
- Nozzle wide open
- RPM will decrease and hangup in the 70 to 85% range
- If inlet duct shall is recognized, check for sideslip, correct sideslip with rudder and reduce speed
- For detailed engine compressor stall data, refer to Section I

The procedure of **Throttle – Off, Start Switches – Start, Throttle – Idle,** provides the best known conditions for engine relight and power recovery, and in many cases, is the only way a steady state engine stall can be cleared. This procedure also reduces the change of igniter plug wetting during an airstart. The probability of relight is very high as fuel flow needs to be stopped only momentarily to clear the back pressure that is sustaining the stall. The more rapidly the procedure is accomplished, the less the RPM will decay before relight.

At high altitude (above 25000 feet) to avoid the discomfort of loss of cabin pressurization, stall clearing may be attempted by retarding the throttle to idle instead of off. If engine fails to start, RPM hang up, or stall persists, use the normal Air Start/Stall Clearing procedure.

When engine flameout or compressor stall is recognized, and time and altitude permits, take immediate corrective action:

1. **THROTTLE – OFF**
2. **1 START 2 SWITCHES – START AND HOLD**
3. **THROTTLE – IDLE.**

WARNING

- DO NOT ALLOW AIRSPEED AND ALTITUDE TO DETERIORATE BELOW SAFE EJECTION PARAMETERS.
- IF RELIGHT IS OBTAINED AT APPROXIMATELY 350 KIAS, WITH ENGINE SPEED OF 20% RPM, IT MAY TAKE UPWARD OF 20 SECONDS FOR ENGINE PARAMETERS OF RPM, EGT AND FUEL FLOW TO RECOVER TO FLIGHT IDLE VALUES.

- DO NOT ADVANCE THROTTLE ABOVE IDLE RANGE UNTIL FLIGHT IDLE IS OBTAINED. THEN, ADVANCE THROTTLE SMOOTHLY AND SLOWLY UNTIL THRUST SUFFICIENT TO MAINTAIN FLIGHT HAS BEEN OBTAINED.

- LAND AS SOON AS POSSIBLE.

4. EGT and RPM – Monitor for relight indication (be alert for RPM hangup)

   – Repeated air start attempts, without allowing sufficient time for engine parameters to recover to flight idle values, will result in termination of an otherwise successful air start

   – Relight is indicated by EGT rise and normally by RPM increase. RPM may continue to decrease after relight occurs.

   – Fuel flow will not indicate below 20% RPM.

   – 45 seconds of continuous ignition is provided after each actuation of the start switches

   – After relight, the time required for RPM to increase to idle speed is a function of airspeed, altitude, and RPM at light-off. If a hung start occurs, an acceleration to 350 KIAS, altitude permitting, may be required to assist engine acceleration to idle. If relight can be accomplished at 35% RPM or above (time from cutoff/flameout of 10 seconds or less), then engine speed recovery to idle will require 5 seconds or less. If engine speed is allowed to decay below 20% RPM (20 seconds or more from cutoff/flameout to light-off), then recovery time from light-off to idle can be as long as 20 seconds at altitudes below 10000 feet, as much as 40 seconds at an altitude of 35000 feet, and may be up to 60 seconds with alternate fuel in slow/hung start area of the air start envelope (Figure 3-3).

   – If RPM decreases below flight idle before relight is obtained, an RPM hangup can occur at 70 to 80% following relight. EGT will be moderate but rising abnormally, a slight high frequency vibration may also be felt. FOD or a closed nozzle may aggravate this condition.

   – EGT between relight and idle varies with airspeed, altitude, and engine RPM at time of lightoff. EGT at idle RPM can be as low as 120° C.

If relight is not obtained:

5. RAT handle – Pull. Reaccomplish steps 1 through 4. Refer to RAT EXTENDED FLIGHT procedure

If relight and stable flight idle conditions are obtained:

5. Throttle – Slowly advance until required engine thrust is obtained. Attempt no further throttle movement until landing is assured

NOTE

If max. obtainable RPM is 94%, a cold shift has occurred. Thrust will be equal to, or greater than 94% normal. If the CIT is + 38° C or less, there are no restrictions on throttle movement in idle to military range. Afterburner operation should be initiated only in emergencies.

If adequate thrust is obtained:

6. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

If adequate thrust is not obtained:

6. Eject

AUTOPITCH CONTROL SYSTEM FAILURE

Indicated by:

– Illumination of the AUTO-PITCH CONT OUT warning light

– Inadvertent kicker operation

1. Overpower APC kicker, if necessary, to maintain flight attitude. Use emergency disconnect switch (paddle switch)

ESTIMATED AIR START ENVELOPE (JP-4)

NOTE

- INITIATE AIR START PROMPTLY UPON PROBLEM RECOGNITION TO PREVENT EXCESSIVE RPM DECAY
- ——— AIR START ENVELOPE BASED ON F–4/J79–15/17 FLIGHT TEST RESULTS (FTC–TR–73–4) CONSIDERED ANALOGOUS TO F–104/J79–11
- – – – MOST FAVORABLE AIR START ENVELOPE BASED ON GENERAL ELECTRIC ECP E–79–F94
- (350 KIAS) IF HUNG START OCCURS AN ACCELERATION TO 380 KIAS, ALTITUDE PERMITTING, MAY BE REQUIRED TO ASSIST ENGINE ACCELERATION TO IDLE

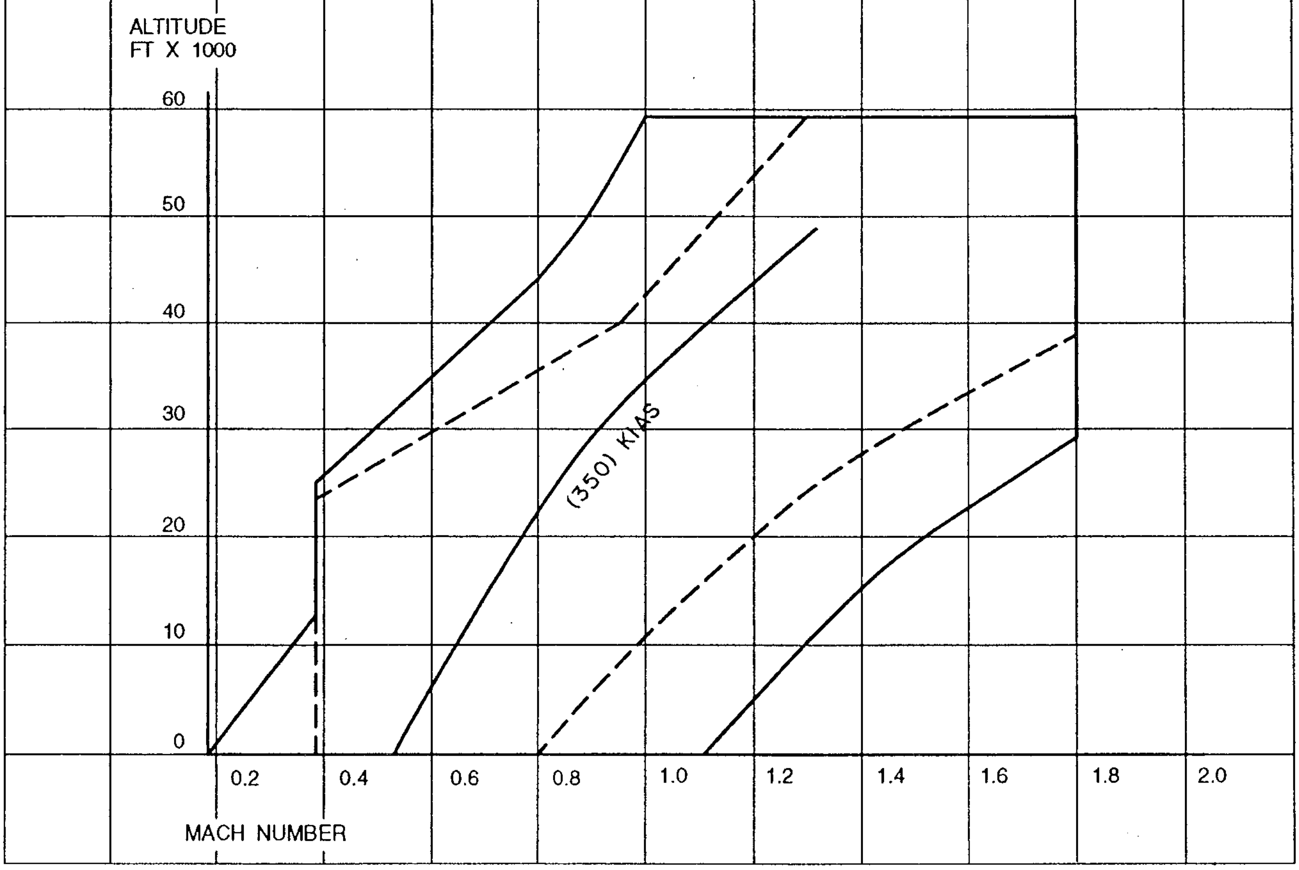

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Figure 3-3 (Sheet 1 of 2)

ESTIMATED AIR START ENVELOPE (JP-8)

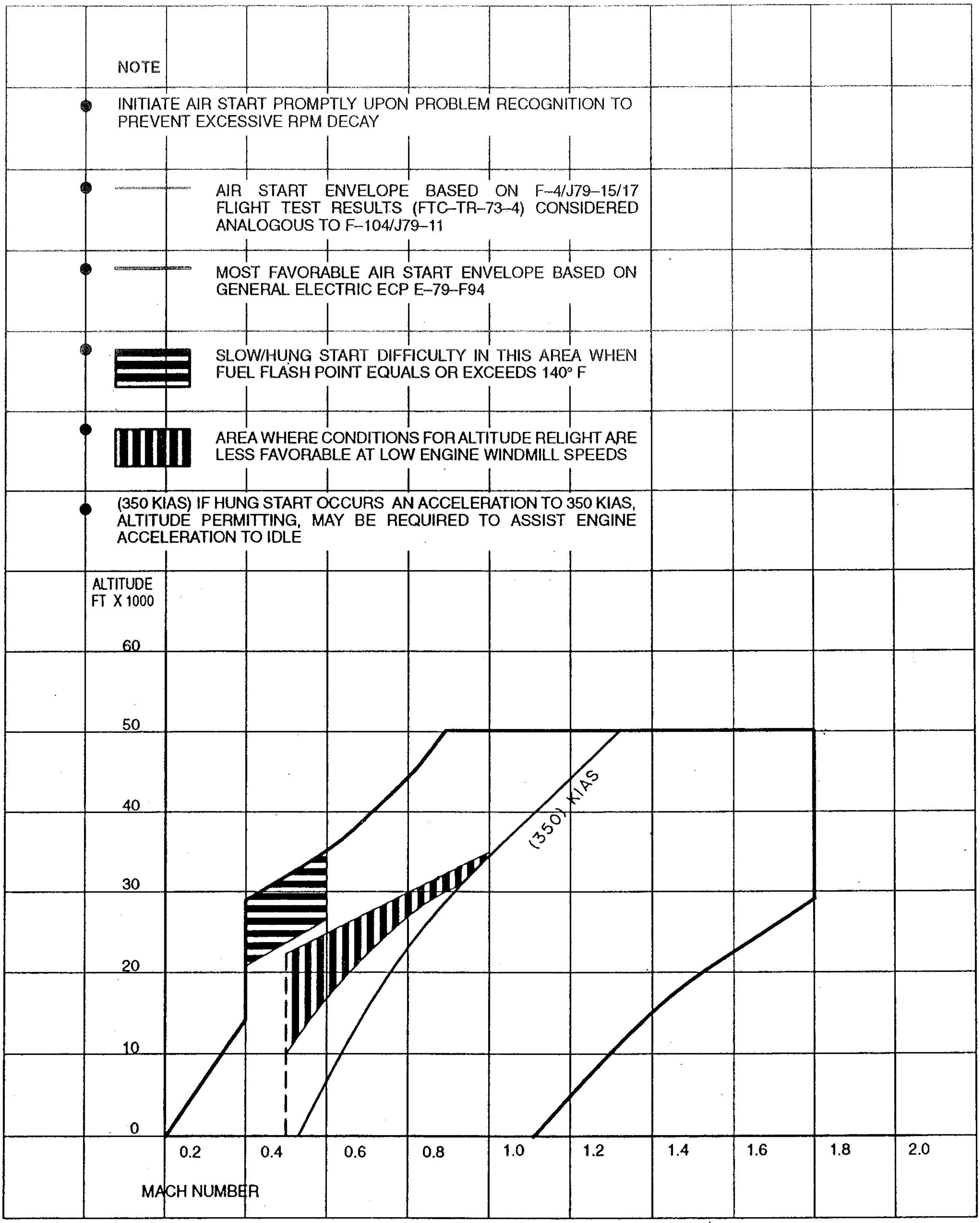

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Figure 3-3 (Sheet 2 of 2)

NOTE

Expect resistance and slow stick movement when overpowering the APC kicker. The APC actuator incorporates an orifice which restricts motion of the stick, after the APC kicker has been activated. Under extreme conditions it may be necessary to activate the emergency disconnect switch (paddle switch) with the left hand; this will preclude the necessity to release the existing back pressure on the control stick.

2. APC CUTOUT switch – OFF

NOTE

The APC kicker will be deactivated automatically whenever the AUTO-PITCH CONT OUT warning light is illuminated. Turning the APC CUTOUT switch off is an added safety measure.

3. Observe the stick shaker boundary as kicker will be inoperative

NOTE

- Any transient in electrical supply will usually cause momentary illumination of AUTO-PITCH CONT OUT warning light (example: transfer from XP5 to XP4 bus). This is a characteristic rather than a malfunction of the system. It is not necessary to switch the APC off in this case. High angle-of-attack maneuvering should be avoided as condition may occur again.

- If transfer of electrical power from the fixed-frequency to the variable-frequency bus caused the APC operation, the FIXED FREQ. OUT warning light will illuminate. Resetting back to the fixed-frequency bus can cause momentary APC operation. However, if safe flight conditions can be assured the APC kicker may be reactivated at the pilot's discretion.

- Momentary loss of primary DC power causes transient and limited kicker operation. This may occur when No. 2 AC WF generator, for any reason, goes off line.

Stick Shaker Failure

1. STICK SHAKER circuit breaker – Pull

If the stick shaker circuit breaker is pulled to deactivate the stick shaker for any reason, the AUTO-PITCH CONT OUT warning light will illuminate. Since the AUTO-PITCH CONT OUT warning light is illuminated, further visual indication of APC kicker malfunction will be lost.
Under low altitude conditions, with gear and flaps UP wherein the stick shaker circuit breaker has been pulled, continue as follows:

2. APC CUTOUT switch – OFF
3. Exercise extreme care to avoid abrupt maneuvers, low airspeeds, or maneuvers requiring operation at high angle-of-attack
4. Land – Using a PRECAUTIONARY STRAIGHT-IN PATTERN procedure

BLEED AIR DUCT SEPARATION/FIRE WARNING LIGHTS/INADVERTENT T2 RESET

If fire warning lights illuminate during flight and there are no other indications of fire, a possible cause is leakage of hot compressor air from BLC or other bleed air ducts into the engine compartment where it can impinge against the fire warning detectors. One indication of this type of failure is the lights are likely to go out after a minute or two of operation at IDLE RPM, because retarding the throttle reduces the volume and temperature of compressor bleed air. Further indications of this type of failure depend on the location of the leak and may include loss of cockpit pressurization, illumination of ENG/DUCT ANTI-ICE ON light, a severe rolloff tendency when the flaps are moved to LAND, and T_2 reset actuation.
If the ENG/DUCT ANTI-ICE ON light illuminates without having the system switched on, and stays on even after pulling the ENGINE/DUCT ANTI-ICE circuit breakers (rear cockpit only), a leak within the engine/duct anti-ice system has occurred.

The rolloff tendency is an indication of BLC duct separation on one side resulting in asymmetric BLC airflow. BLC duct separation should be checked with land flaps at a safe altitude and airspeed (240 KIAS maximum).
T_2 reset actuation may occur if duct failure permits hot compressor air to contact the CIT sensor. In this case, the RPM could increase to 103.5% (±1.0) and remain in the T_2 reset range regardless of throttle position.
(During T_2 reset operation, retarding the throttle below military will not reduce RPM, but will reduce thrust because the exhaust nozzle will open).

Inadvertent T_2 reset may be caused by:

- Leakage of hot compressor air from BLC or other bleed air ducts into the engine compartment
- Fuel leakage in main fuel control (servo line (MFC-CIT)
- Fire

Inadvertent T_2 reset may be assumed with the following instrument readings:

- RPM: 103.5% (±1)
- CIT: below 70° C

If bleed air duct separation is suspected by the illumination of the fire warning lights without any other indications of fire and/or by a severe rolloff during flap selection and/or inadvertent T_2 reset:

1. Throttle – Minimum practical: confirm fire
2. Oxygen – 100%
3. Airspeed – Keep above 300 KIAS as long as possible
4. LDG GEAR – Keep us as long as practical

NOTE

It is advisable to keep the gear retracted as long as practicable to keep the secondary airflow bypass flaps open.

WARNING

IF A BLC DUCT SEPARATION IS INDICATED, DO NOT PLACE THE WING FLAP LEVER TO THE LAND POSITION IN THE LANDING PATTERN BECAUSE THIS MAY RESULT IN A SEVERE ROLLOFF DUE TO ASYMMETRIC AIRFLOW OVER THE FLAPS.

5. Land ASAP using PRECAUTIONARY STRAIGHT-IN PATTERN procedure

Immediately after touchdown:

6. Throttle – IDLE (OFF for fire)
7. Use optimum braking
8. Use nosewheel steering to position aircraft on runway centerline for possible barrier engagement

If T_2 reset persists:

9. Throttle – OFF

NOTE

Nosewheel steering and antiskid/power brakes will be inoperative when engine speed drops below 65%. Directional control shall then be maintained by the standby brakes and rudder. Standby brakes may be used above 65% by turning off the ANTI-SKID switch.

10. DRAG CHUTE – DEPLOY, observing drag chute limits

CAUTION

HIGH EXHAUST GAS VELOCITY COMBINED WITH EXCESSIVE AIRSPEED MAIN FAIL THE DRAG CHUTE; THEREFORE, DEPLOY THE CHUTE AT MINIMUM PRACTICAL AIRSPEED.

CANOPY OPEN/LOSS/BROKEN

If canopy is not locked:

1. Canopy – Jettison
2. Attain safe position
3. Airspeed -- 300 KIAS or less
4. Land as soon as practicable

WARNING

- IN ANY CASE OF LOSS, OPENING OR BROKEN CANOPY, THE ENGINE MAY SUSTAIN FOREIGN OBJECT DAMAGE. THEREFORE, BE PREPARED TO FOLLOW STALL CLEARING PROCEDURE.
- LOSS OF CANOPY IN SUPERSONIC FLIGHT MAY CAUSE SUFFICIENT PRESSURE CHANGE AS ENGINE INLET DUCTS TO CAUSE COMPRESSOR STALL OR ENGINE FLAMEOUT. FOREIGN OBJECT DAMAGE MAY NOT BE INVOLVED. THEREFORE, AIR START/STALL CLEARING PROCEDURES SHOULD BE PERFORMED AS APPLICABLE.

COCKPIT PRESSURIZATION MALFUNCTION

Separation of the hot air duct at the refrigerator inlet can cause a loss of cockpit pressurization, damage to wiring, and subsequent failure of the electrical system.
The electrical system failure can occur within a few minutes from hot compressor air melting the insulation on the wiring in the electrical compartment located behind the cockpit.
If the fresh air scoop is opened promptly after the cockpit pressure loss is recognized, the bleed air shutoff valve will shut off the hot air and prevent failure of the electrical system.

If cockpit pressurization reduces substantially and the cause is not apparent:

1. Start immediate descent to 25000 feet (or lower if circumstances permit)

If complete pressure loss is experienced or if at a safe flight level for depressurization:

2. Fresh air scoop lever -- Open
3. Oxygen regulator – Adjust as necessary

EJECTION

NOTE

Minimum safe ejection altitudes at various degrees of dive angles are shown in Figure 3-4.

Ejection Altitude

A safe ejection can be carried out from zero altitude at zero speed when the aircraft is in a level attitude. To increase the escape it is strongly recommended to use optimum altitude and speed whenever possible especially when considering the extremely high sink rate possible in this type of aircraft.
When ejection are carried out at altitudes above 16400 ft, parachute deployment will be delayed by the barostat until the seat and man have descended to approximately 16400 ft.
During this time the seat will be stabilized by the drogue system but may rotate about its vertical axis. This rotation is not abnormal and the pilot should wait for the parachute to be automatically deployed at approximately 16400 ft.

MINIMUM SAFE EJECTION ALTITUDES AT VARIOUS DEGREES OF DIVE ANGLES

WARNING

ALTITUDES ARE CALCULATED FROM PULLING FIRING HANDLE AND THREREFORE MAKE NO ALLOWANCE FOR PILOT'S REACTION TIME OR ANY PREEJECTION ACTION.

CALCULATIONS ARE MADE ASSUMING A WINGS LEVEL ATTITUDE SO ADDITIONAL ALTITUDE IS REQUIRED IF THE AIRCRAFT IS BANKED AND/OR HAS A HIGH SINK RATE.

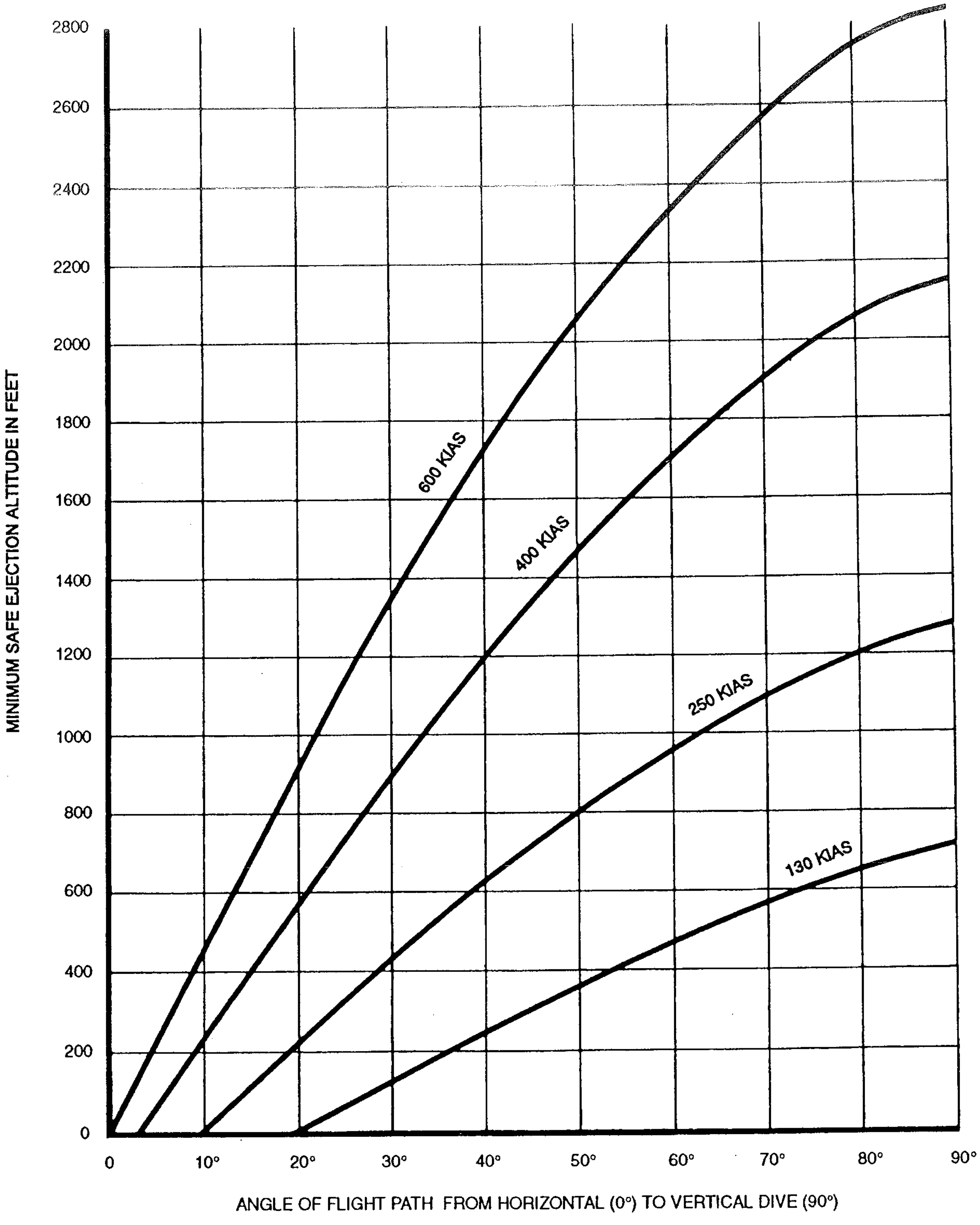


Figure 3-4

WARNING

- FIGURE 3-4 DOES NOT PROVIDE ANY SAFETY FACTOR FOR PILOT REACTION TIME, EQUIPMENT MALFUNCTION, DELAY IN SEAT SEPARATION, BANK ANGLE, HIGH SINK RATE, ETC., AND SHOULD NOT BE USED AS A BASIS FOR DELAYING EJECTION BELOW 2000 FEET ABOVE THE TERRAIN. ACCIDENT STATISTICS EMPHATICALLY SHOW A PROGRESSIVE DECREASE IN SUCCESSFUL EJECTIONS AS ALTITUDES DECREASE BELOW 2000 FEET.
- UNDER LEVEL FLIGHT CONDITIONS, EJECT AT LEAST 2000 FEET ABOVE THE TERRAIN WHENEVER POSSIBLE. IF IN A SPIN, EJECT AT LEAST 15000 FEET ABOVE THE TERRAIN WHENEVER POSSIBLE.

Ejection Attitude

During any low altitude ejection, the chances for successful ejection can be greatly increased by zooming the aircraft (the speed permits) to exchange airspeed for altitude. Ejection should be accomplished while the aircraft is in a positive climb. This will result in a more nearly vertical trajectory for the seat and crewmember, thus providing more altitude and time for seat separation and parachute deployment.

Ejection Speed

If possible, eject at the lowest practical airspeed ("lowest practical" would be that speed below which level flight cannot be maintained). Below 120 KIAS airflow is not sufficient to ensure rapid parachute deployment on aircraft not utilizing force-deployed parachutes. Therefore, it becomes extremely important during low-altitude ejection to obtain at least minimum required airspeed to ensure complete parachute deployment at the greatest height above the terrain. During high-altitude ejection, observing this minimum airspeed becomes less important since there is adequate time (in the form of altitude) for parachute deployment.

WARNING

- AT SEA LEVEL, WINDBLAST WILL EXERT MEDIUM FORCES ON THE BODY UP TO 450 KIAS SEVERE FORCES CAUSING FLAILING AND SKIN INJURIES BETWEEN 450 AND 600 KIAS AND EXCESSIVE FORCES ABOVE 600 KIAS.
- THE MINIMUM EJECTION SPEED AT ZERO ALTITUDE IS 60 KIAS TO CLEAR THE CANOPY FROM TRAJECTORY.

Before Ejection

WARNING

OCCUPANT OF THE REAR SEAT SHOULD EJECT FIRST TO PREVENT INJURY BY COLLISION WITH FRONT CANOPY OR BY EFFECTS OF FRONT SEAT ROCKET BLAST. SEAT/CHUTE/-MAN ENTANGLEMENT MAY OCCUR WITH LESS THAN TWO SECONDS DELAY BETWEEN BOTH SEATS EJECTION.

If time and conditions permit:

1. Advise other occupant of decision to eject
2. Reduce speed
3. Transmit location and intention to nearest radio facility and turn IFF to EMER position
4. Head aircraft toward unpopulated area. If time and conditions permit throttle IDLE prior to ejection

5. Lower visor
6. Leave feet on rudder pedals

Ejection

1. ASSUME EJECTION POSITION
2. PULL EITHER FIRING HANDLE WITH BOTH HANDS

The choice of which firing handle to be used depends upon pilot preference and the circumstances surrounding the ejection.
If the upper handle is used, grasp the upper handle with both hands, knuckles facing forward, and pull the firing handle. Make sure the elbows pass and stay close to both sides of the body and the handle is pulled to its full extent.
If the lower firing handle is used, grip handle with one hand and grasp wrist with the other hand. The ejection sequence remains the same.

WARNING

- DURING CRITICAL LOW ALTITUDE EJECTION INVOLVING ROLLING MANEUVERS, AN ATTEMPT SHOULD BE MADE TO HOLD WINGS LEVEL OR REDUCE THE ROLLING ANGLE WHILE EJECTING, THIS CAN BEST BE ACCOMPLISHED BY FLYING AIRCRAFT WITH ONE HAND AND EJECTING BY MEANS OF LOWER HANDLE WITH OTHER HAND.

- IF THE SELECTED FIRING HANDLE MALFUNCTIONS, PULL THE OTHER FIRING HANDLE. IF THE UPPER FIRING HANDLE MALFUNCTIONS, DO NOT RELEASE IT WHILE PULLING THE LOWER FIRING HANDLE AS IT MAY BECOME ENTANGLED IN THE SEAT DROGUE CHUTE DURING THE EJECTION SEQUENCE.

NOTE

- The lower firing handle should be used if due to injuries, acceleration forces, or other circumstances the upper firing handle can not be reached and/or time dictates an immediate action.

- If the canopy fails to jettison, the pilot will eject through the canopy.

After Seat/Man Separation

1. CANOPY – CHECK
2. MASK – REMOVE (if landing on water)
3. SAFETY CLIP – REMOVE
4. SURVIVAL PACK – RELEASE

If landing on water and, if applicable, on terrain:

5. LIFE VEST – INFLATE

If landing on terrain is confirmed and survival pack was released:

6. DINGHY LOWERING LINE – DISCONNECT

Upon touchdown:

7. RELEASE BOX – ROTATE AND DEPRESS
8. EMERGENCY RADIO – CHECK

Water Landing – Dinghy Failure

When landing on water and dinghy has not inflated automatically or fails to inflate automatically:

1. MANUAL RELEASE HANDLE – PULL

If dinghy still fails to inflate:

2. COVER FLAP – OPEN
3. SURVIVAL PACK RELEASE ASSEMBLY – CHECK OPEN

Manual Seat Separation After Ejection

If seat automatic sequence fails:

1. MANUAL OVERRIDE HANDLE – PULL
2. SEAT – ABANDON

When clear of the aircraft and below 15000 feet:

3. PARACHUTE D-RING – PULL

Bail-Out without Ejection Seat

If seat fails to eject:

1. **OTHER FIRING HANDLE – PULL TO ITS FULLEST EXTENT**
2. **MANUAL OVERRIDE HANDLE – PULL** (this action will cause separation from seat)

   If time permits:

   – **LEG STRAPS – UNLACE**

   – **PEC DISCONNECT – CHECK**
3. **CANOPY – JETTISON**
4. **AIRCRAFT – ABANDON**

When clear of the aircraft and below 15000 ft:

5. **PARACHUTE D-RING – PULL**

ELECTRICAL FIRE

Circuit breakers and fuses protect most of the circuits and tend to prevent electrical fires; however, if electrical fire occurs, proceed as follows:

1. Oxygen diluter lever – 100%
2. Descend to 25000 ft or lower if circumstances permit
3. Fresh air scoop lever – Open
4. Generator switches – OFF-RESET

NOTE

Switching No. 2 AC WF generator off, causes the transient and limited kicker operation. This may be prevented if contemporarily the APC disengage switch (paddle switch) is also held pressed.

5. Clock – Hack (calculate fuel remaining using fuel flow indicator)
6. All electrical accessory switches – OFF
7. Generator switches – ON (one by one needed)
8. Operate only those units necessary for safe flight and landing
9. Return generator switches to OFF-RESET position when operation is complete
10. Land ASAP

WARNING

- ALL FUEL BOOST PUMPS ARE NOT AVAILABLE WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED AND, WHEN REMAINING FUEL (INTERNAL) IS BELOW 1000 LBS, A FLAMEOUT COULD OCCUR IF RAT FAILS TO POWER NO. 3 FUEL BOOST PUMP. MOREOVER FLIGHT WITH NO. 3 FUEL BOOST PUMP POWERED BY RAT AND 1000 LBS. OR LESS INTERNAL FUEL REMAINING, SHOULD BE PERFORMED IN NOSE UP FLIGHT ATTITUDE WITH EASY COORDINATED MANEUVERS AND NEGLIGIBLE DECELERATION FORCES. FLIGHT WITH NOSE DOWN AT THIS FUEL LOAD MAY ALLOW THE NO. 3 FUEL BOOST PUMP TO CAVITATE RESULTING IN SUBSEQUENT ENGINE FLAME OUT. ILLUMINATION OF THE FUEL BOOST PUMPS WARNING LIGHT UNDER THESE CONDITIONS MAY NOT PROVIDE ADEQUATE WARNING TO TAKE CORRECTIVE ACTIONS AND PREVENT AN ENGINE FLAME OUT.

- LANDING WITH RAT EXTENDED AND 1000 LBS. OR LESS REMAINING FUEL SHOULD BE PERFORMED FOLLOWING "PRECAUTIONARY STRAIGHT-IN PATTERN" PROCEDURE.

CAUTION

- WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED, THE EXTERNAL FUEL QUANTITY INDICATION ARE NOT AVAILABLE AND FUEL DATA ARE FROZEN. WHEN RAT IS EXTENDED, THE INTERNAL FUEL QUANTITY INDICATION IS AVAILABLE EXCEPT WHEN FLAPS/SLATS ARE OPERATED: IN THIS CONDITION FUEL QUANTITY DATUM IS TEMPORARILY FROZEN.
- WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED, NO WARNING LIGHTS (EXCEPT FIRE) ARE AVAILABLE UNTIL RAT IS EXTENDED.

ELECTRICAL POWER SUPPLY SYSTEM FAILURE

NOTE

Refer to Figure 3-5 for a summary table of Electrical Power Supply System failure.

Single AC (WF) Generator Failure

Following a single generator failure the XP3 bus and utilities are lost.

NOTE

Momentary loss of primary DC power, following a No. 2 AC WF generator failure, causes transient and limited kicker operation.

If either generator-out warning light illuminates, perform following procedure:

1. Corresponding generator switch OFF-RESET then ON (check generator warning light extinguished)

If failure confirmed:

2. Failed generator – OFF-RESET
3. Land as soon as practicable

Double AC (WF) Generator Failure

Following a double generator failure the XP1, XP2, XP3, XP4 and XP7 busses as well as PP1, PP2 and PP3 busses and utilities are lost. The indications will be:

- Main UHF not available
- Flap position indicators: BARBER POLE
- The only warning lights available: FIRE

CAUTION

THE GENERATOR NO. 1 OUT AND GENERATOR NO. 2 OUT WARNING LIGHTS SHALL BE LIT, ON THE WARNING LIGHTS PANEL, ONLY FOLLOWING RAT EXTENSION.

NOTE

The PRIMARY DC BUS OUT shall be also lit, only following RAT extension.

If transmitting:

1. Radar – SBY
2. Generator switches – OFF-RESET, then ON (check generator warning lights extinguished)

If only one generator is restored:

3. Failed generator – OFF-RESET
4. Land as soon as practicable

If both generators are not restored:

3. Generator switches – OFF-RESET

ELECTRICAL POWER SUPPLY SYSTEM FAILURES SUMMARY TABLE

NOTE: REFER TO THE APPLICABLE FOLDS-OUT FOR UTILITIES LOST

TYPE OF FAILURE	WARNING INDICATION(S) ON WARNING LIGHTS PANEL	BUSSES AVAILABLE AND POWER SOURCES											
		XP1	XP2	XP3	XP4	XP5	XP6	XP7	PP1	PP2	PP3	PP4	PP5
WF GENERATOR 1 FAILURE	GENERATOR NO. 1 OUT	X XP2	X GEN2		X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
WF GENERATOR 2 FAILURE	GENERATOR NO. 2 OUT	X GEN1	X XP1		X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
WF GENERATORS 1 AND 2 FAILURE	NONE					X GEN3	X XP5					X BATT1	X BATT2
WF GENERATORS 1 AND 2 FAILURE (RAT EXTENDED)	GENERATOR NO. 1 OUT GENERATOR NO. 2 OUT PRIMARY DC BUS OUT (1)				X RAT	X GEN3	X XP5			X TRU2 (2)	X TRU2	X TRU2	X TRU2
FF GENERATOR FAILURE	FIXED FREQ OUT	X GEN1	X GEN2	X XP1	X XP2	X XP4	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
XP1 BUS SHORT CIRCUIT	GENERATOR NO. 1 OUT		X GEN2		X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
XP2 BUS SHORT CIRCUIT	NONE	X GEN1				X GEN3	X XP5	X XP5				X BATT1	X BATT2
XP2 BUS SHORT CIRCUIT (RAT EXTENDED)	GENERATOR NO. 2 OUT PRIMARY DC BUS OUT (1)	X GEN1			X RAT	X GEN3	X XP5	X XP5		X TRU2 (2)	X TRU2	X TRU2	X TRU2
XP3 BUS SHORT CIRCUIT	GENERATOR NO. 1 OUT		X GEN2		X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
XP4 BUS SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1		X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X BATT1	X BATT2
XP5 BUS SHORT CIRCUIT	FIXED FREQ OUT	X GEN1	X GEN2	X XP1					X TRU1	X PP1	X PP1	X BATT1	X BATT2
26 V AC TRANSFORMER FAILURE	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3		X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
XP6 BUS SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3		X XP5	X TRU1	X PP1	X PP1	X TRU2	X TRU2
XP7 BUS SHORT CIRCUIT	FIXED FREQ OUT	X GEN1	X GEN2	X XP1					X TRU1	X PP1	X PP1	X BATT1	X BATT2
TRU1 FAILURE	PRIMARY DC BUS OUT	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5		X TRU2 (2)	X TRU2	X TRU2	X TRU2
TRU2 FAILURE	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X BATT1	X BATT2

Figure 3-5 (Sheet 1 of 2)

ELECTRICAL POWER SUPPLY SYSTEM FAILURES SUMMARY TABLE

TYPE OF FAILURE	WARNING INDICATION(S) ON WARNING LIGHTS PANEL	BUSSES AVAILABLE AND POWER SOURCES											
		XP1	XP2	XP3	XP4	XP5	XP6	XP7	PP1	PP2	PP3	PP4	PP5
DOUBLE TRU FAILURE	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5				X BATT1	X BATT2
PP1 SHORT CIRCUIT	PRIMARY DC BUS OUT	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5		X TRU2 (2)	X TRU2	X TRU2	X TRU2
PP2 SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5	X TRU1		X PP1	X TRU2	X TRU2
PP3 SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1		X TRU2	X TRU2
PP4 SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1		X TRU2
PP5 SHORT CIRCUIT	NONE	X GEN1	X GEN2	X XP1	X XP2	X GEN3	X XP5	X XP5	X TRU1	X PP1	X PP1	X TRU2	

WARNING

FAILURE OF BATT1 OR BATT2, OR FAILURE OF BOTH BATT1 AND BATT2 CAUSES THE PP4 OR/AND PP5 NOT TO BE SUPPLIED BY THE RESPECTIVE BATTERY.

NOTES:

(1) If No. 1 EMERGENCY DC BUS (PP2) is not energized there will be no warning light indication.

(2) PP2 temporarily inoperative when operating flaps/slats.

Figure 3-5 (Sheet 2 of 2)

4. RAT handle – Pull (refer to RAT EXTEND FLIGHT procedure)

NOTE

Refer to Figure 3-5 for busses available/not available following RAT extension.

5. Land as soon as practicable using RAT EXTENDED FLIGHT and PRECAUTIONARY STRAIGHT-IN PATTERN procedure

WARNING

- ALL FUEL BOOST PUMPS ARE NOT AVAILABLE WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED AND, WHEN REMAINING FUEL (INTERNAL) IS BELOW 1000 LBS. A FLAMEOUT COULD OCCUR IF RAT FAILS TO POWER NO. 3 FUEL BOOST PUMP. MOREOVER FLIGHT WITH NO. 3 FUEL BOOST PUMP POWERED BY RAT AND 1000 LBS. OR LESS INTERNAL FUEL REMAINING, SHOULD BE PERFORMED IN NOSE UP FLIGHT ATTITUDE WITH EASY COORDINATED MANEUVERS AND NEGLIGIBLE DECELERATION FORCES. FLIGHT WITH NOSE DOWN AT THIS FUEL LOAD MAY ALLOW THE NO. 3 FUEL BOOST PUMP TO CAVITATE RESULTING IN SUBSEQUENT ENGINE FLAME OUT. ILLUMINATION OF THE FUEL BOOST PUMPS WARNING LIGHT UNDER THESE CONDITIONS MAY NOT PROVIDE ADEQUATE WARNING TO TAKE CORRECTIVE ACTIONS AND PREVENT AN ENGINE FLAME OUT.
- LANDING WITH RAT EXTENDED AND 1000 LBS. OR LESS REMAINING FUEL SHOULD BE PERFORMED FOLLOWING "PRECAUTIONARY STRAIGHT-IN PATTERN" PROCEDURE.

CAUTION

- WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED, THE EXTERNAL FUEL QUANTITY INDICATION ARE NOT AVAILABLE AND FUEL DATA ARE FROZEN. WHEN RAT IS EXTENDED, THE INTERNAL FUEL QUANTITY INDICATION IS AVAILABLE EXCEPT WHEN FLAPS/SLATS ARE OPERATED: IN THIS CONDITION FUEL QUANTITY DATUM IS TEMPORARILY FROZEN.
- WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED, NO WARNING LIGHTS (EXCEPT FIRE) ARE AVAILABLE UNTIL RAT IS EXTENDED.

Hydraulically Driven Fixed-Frequency Generator Out

If FIXED FREQ. OUT warning light illuminates, the hydraulically driven generator has been disconnected from the XP5, XP6 and XP7 busses. A relay connects variable-frequency power from the XP4 bus to XP5, XP6 and XP7 busses.
Fixed-frequency powered flight instruments operate adequately on wild-frequency; however, the NASARR system will be inoperative and the inertial navigation system operation will be degraded.

1. FIXED FREQ RESET button – PRESS for at least 5 seconds. Check FIXED FREQ OUT warning light extinguished. To prevent possible

stick kicker operation: press and hold paddle switch during reset

If warning light is still illuminated (and conditions permit):

2. RADAR – OFF

WARNING

PROBLEM MAY BE HYDRAULIC RATHER THAN ELECTRIC. IF THE PROBLEM IS HYDRAULIC, CONTINUED ATTEMPTS TO RESET THE FIXED-FREQUENCY GENERATOR COULD RESULT IN LOSS OF HYDRAULIC FLUID.

3. FIXED FREQ RESET button – PRESS for at least 5 seconds. Check FIXED FREQ OUT warning light extinguished. To prevent possible stick kicker operation: press and hold paddle switch during reset

WARNING

IF FIXED FREQUENCY IS NOT RESTORED, NAVIGATION INSTRUMENTS WILL PRESENT INDICATING ERRORS. BEST RPM FOR MINIMUM INDICATING ERRORS IS 85 ± 4%. WITH "FIXED FREQUENCY OUT" WARNING LIGHT STILL DISPLAYED, A SIGNIFICANT PRESSURE DECREASING SHOWN ON HYD SYSTEM 2 COCKPIT PRESSURE INDICATOR WILL EVIDENCE THE HYDRAULIC ORIGIN OF THE FAILURE. IF NO PRESSURE REDUCTION IS SHOWN, THE FAILURE WILL BE ELECTRICAL.

Primary DC Bus Failure/TRU 1 Failure

Following the PP1 DC bus failure or following the TRU 1 failure, the PP1 bus and utilities are lost and the DC PRIMARY BUS OUT warning light is lit on the warning light panels.

If PRIMARY DC BUS OUT warning lights illuminate:

1. Land as soon as practical

WARNING

NOSEWHEEL STEERING AND POWER BRAKES WILL BE INOPERATIVE. AIRCRAFT DIRECTIONAL CONTROL SHALL BE MAINTAINED BY USE OF RUDDER AND/OR STANDBY BRAKES.

ENGINE FAILURE

Engine failure is defined as a complete power failure which, in the pilot's judgment, makes a restart impossible or inadvisable. Examples are engine seizure or explosion.

If glide is required prior to ejection:

1. Throttle – OFF
2. RAT handle – PULL (refer to RAT EXTENDED FLIGHT procedure)
3. Wing flaps – TAKEOFF
4. Speed – 245 KIAS
5. Ejection preparation – Refer to EJECTION procedure

ENGINE OIL SYSTEM MALFUNCTION

Engine oil system malfunction is evidenced by:

– Abnormal oil pressure

– Illumination of ENGINE OIL LEVEL LOW warning light indicates oil depletion to 6.4 pints or less. With depletion to 4.0 pints the nozzle will fail open and complete loss of oil may be imminent with oil starvation of engine bearings. With a complete loss of oil, the engine may operate for 2 minutes at military thrust without detrimental effects to the bearings. Limited experience has indicated the engine should operate for a period of approximately 4 to 5 minutes at 90% RPM before a complete failure occurs. High thrust settings should be avoided when possible to keep temperature and bearings loads at a minimum. Increasing vibration is an indication of bearings failure. Throttle movement will accelerate this failure.

1. **THROTTLE – 83% to 90% RPM**
2. **NOZZLE HANDLE – OUT**

NOTE

- The nozzle may not close under certain conditions of airspeed and altitude. If nozzle does not close, zoom and reduce power to reduce airspeed/nozzle area pressure.
- If the nozzle opens to the 3.2 position, the mechanical locks have engaged. Under this condition the nozzle will not open when the nozzle handle is pushed in.

3. **EXTERNAL STORES – JETTISON (if necessary)**

If nozzle closes:

4. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

CAUTION

AFTER NOZZLE CLOSURE HANDLE HAS BEEN PULLED AND NOZZLE IS CLOSED, DO NOT LIGHT AFTERBURNER. LIGHTING AFTERBURNER WILL RESULT IN AN OVERTEMPERATURE CONDITION AND MAY CAUSE EXCESSIVE RPM DROP OR ENGINE STALL.

After touchdown:

5. Nozzle handle – In

NOTE

If the nozzle handle does not go in, landing roll will increase and the drag may fail at deployment.

If nozzle fails to close:

4. **THROTTLE – RAPIDLY TO MAX A/B**

A 3 to 5 second delay may occur before an afterburner light is obtained. Afterburner lights with the nozzle failed open are not assured, but the probability of successful lights increases as altitude decreases.

If A/B lights:

5. **NOZZLE HANDLE – IN**

At safe position:

6. Throttle – MILITARY
7. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

If extreme vibration/EGT is encountered:

8. **RAT handle – PULL**
9. **THROTTLE – OFF**
10. **EJECT**

WARNING

EXTREME VIBRATION, USUALLY ACCOMPANIED BY A RISE IN EGT, INDICATES ENGINE SEIZURE IS IMMINENT AND COULD LEAD TO EXCESSIVE ENGINE/AIRCRAFT DAMAGE WHICH MAY JEOPARDIZE SUCCESSFUL BAILOUT CAPABILITY.

EXHAUST NOZZLE FAILS OPEN WITHOUT INDICATION OF ENGINE OIL SYSTEM MALFUNCTION (AFTERBURNING)

The most probable cause is failure of the nozzle area control amplifier or control alternator.

Indicated by:

- A slight reduction in thrust
- EGT approximately 500° C to 550° C
- An increase in nozzle area to approx. 10

NOTE

Failure will probably not be detected. Afterburner will continue to operate provided throttle is not retarded to sector/core range. As long as afterburning is maintained, immediate corrective action is not required.

1. Throttle – Maintain MAX A/B

At safe position:

2. Throttle – MILITARY
3. Nozzle handle – Out

If nozzle closes:

4. Land as soon as practicable, monitor EGT with throttle adjustments

If nozzle fails to close:

4. External stores – Jettison (if necessary)
5. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

After touchdown:

6. Nozzle handle – In

EXHAUST NOZZLE FAILS OPEN WITHOUT INDICATION OF ENGINE OIL SYSTEM MALFUNCTION (NON-AFTERBURNING)

The most probable cause is failure of the nozzle area control amplifier or control alternator.

Indicated by:

- Significant thrust decrease
- EGT approximately 350° C
- An increase in nozzle area to approx. 10

WARNING

SUFFICIENT THRUST IS NOT AVAILABLE TO MAINTAIN LEVEL FLIGHT WITH ANY CONFIGURATION.

1. NOZZLE HANDLE – OUT

CAUTION

AFTER NOZZLE CLOSURE HANDLE HAS BEEN PULLED AND NOZZLE IS CLOSED, DO NOT LIGHT AFTERBURNER. LIGHTING AFTERBURNER WILL RESULT IN AN OVER-TEMPERATURE CONDITION AND MAY CAUSE EXCESSIVE RPM DROP OR ENGINE STALL.

NOTE

The nozzle may not close under certain conditions of airspeed and altitude. If nozzle does not close zoom and reduce power to reduce airspeed/nozzle area pressure.

If nozzle closes:

2. Land as soon as practicable, monitor EGT with throttle adjustment

After touchdown:

3. Nozzle handle – In

If nozzle fails to close:

2. **THROTTLE – RAPIDLY TO MAX A/B**

A 3 to 5 second delay may occur before an afterburner light is obtained. Afterburner lights with the nozzle failed open are not assured, but the probability of a successful light increases as altitude decreases.

If A/B lights:

3. **NOZZLE HANDLE – IN**

WARNING

DURING AFTERBURNER OPERATIONS WITH NOZZLE FAILED TO OPEN POSITION, REDUCING THE THROTTLE BELOW FULL AFTERBURNER WILL CAUSE AFTERBURNER BLOWOUT. THROTTLE SHOULD REMAIN IN MAX A/B UNTIL LANDING IS ASSURED.

At safe position:

4. Throttle – MILITARY
5. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure

EXHAUST NOZZLE FAILS TO MECHANICAL SCHEDULE OR SEVERE NOZZLE FLUCTUATIONS OCCUR (AFTERBURNING)

Indicated by:

- EGT of approximately 700° C
- Nozzle indication of 6.5, (with throttle in maximum afterburner position)

At safe position:

1. **THROTTLE – 83% to 90% RPM**
2. **NOZZLE HANDLE – OUT**
3. EGT – Monitor with throttle adjustments

CAUTION

DO NOT LIGHT AFTERBURNER. LIGHTING AFTERBURNER WILL RESULT IN AN OVER-TEMPERATURE CONDITION AND MAY CAUSE EXCESSIVE RPM DROP OR ENGINE STALL.

4. Land as soon as practicable

After touchdown:

5. Nozzle handle – In

EXHAUST NOZZLE FAILS TO MECHANICAL SCHEDULE OR SEVERE NOZZLE FLUCTUATIONS OCCUR (NON-AFTERBURNING)

Indicated by:

- EGT of approximately 700° C
- Nozzle indication of 0.5 (with throttle in military position)

1. **EGT – MONITOR WITH THROTTLE ADJUSTMENTS**
2. **NOZZLE HANDLE – OUT**

CAUTION

DO NOT LIGHT AFTERBURNER. LIGHTING AFTERBURNER WILL RESULT IN AN OVER-TEMPERATURE CONDITION AND MAY CAUSE EXCESSIVE RPM DROP OR ENGINE STALL.

3. Land as soon as practicable

After touchdown:

4. Nozzle handle – In

EXTERNAL STORES EMERGENCY/SELECTIVE JETTISON

To jettison all external stores, using the following procedure:

1. EXT STORES JETTISON – Press

If a selective jettison has to be carried out or if a store fails to jettison following an emergency jettison:

1. STORES RELEASE selector switch – As required
2. External stores selector button – Press. (Relevant position indicators lit)
3. Droppable stores release button – Press

NOTE

Refer to Section V for external stores jettisoning limit speed.

An unsuccessful jettison of a store is indicated by the relevant external stores selector buttons and position indicators light still lit after the emergency and selective jettison attempt and/or by visual check. If the hang-up condition persists a landing with a hang store has to be made.

WARNING

FOLLOWING AN ATTEMPTED JETTISON OPERATION, ANY STORE WHICH DOES NOT SEPARATE FROM THE AIRCRAFT SHOULD BE CONSIDERED UNLOCKED AND SUSCEPTIBLE OF INADVERTENT RELEASE. THE PILOT SHALL BE PREPARED TO EXECUTE AN IMMEDIATE GO-AROUND SHOULD RELEASE OF STORE OCCUR DURING LANDING.

FIRE

1. **THROTTLE – MINIMUM PRACTICAL**
2. **CONFIRM FIRE**

Confirm fire by any possible means such as report from other aircraft, engine instruments, FIRE warning lights, fire warning test circuit, smoke in cockpit or visible smoke trail behind aircraft.

NOTE

- If FIRE warning light goes out upon power reduction and illuminates during testing, a fire is not likely in the areas where fire detectors are installed.
- If the FIRE lights do not illuminate during testing, an explosion may have resulted in serious damage in the areas where fire detectors are installed with a great possibility of fire.

If on fire:

3. **EJECT**

If fire cannot be confirmed but FIRE warning lights remain on below 25000 ft:

3. Fresh air scoop lever – OPEN
4. Land ASAP using PRECAUTIONARY PARTIAL POWER PATTERN procedure, continuously checking for fire

FUEL SYSTEM FAILURE

External Fuel Transfer Failure

When tip and pylon tanks are installed, fuel cannot be transferred from the tip tanks until the pylon tanks are empty.
If fuel fails to transfer from the pylon tanks, the external fuel tanks transfer circuit breaker on the left console can be opened to deenergize the circuit to the transfer valves allowing the valves to open and fuel to transfer.
If opening the circuit breaker fails to correct the malfunction, it may be necessary to jettison the pylon tanks to transfer fuel from the tip tanks.
If tip tanks only are installed and fuel fails to transfer, opening the external tank fuel transfer circuit breaker will allow the transfer valves to open and fuel to transfer.
When the tip tanks have emptied the circuit breaker should be closed to prevent internal tank fuel entrapment.

If external fuel fails to transfer:

1. EXT TANK FUEL TRANS circuit breaker – PULL

NOTE

- Monitor external fuel quantities for simultaneous decrease. If internal quantity increases, push circuit breaker in as tanks will probably continue to feed.
- If the decrease in external fuel quantities is not simultaneous, immediately reset the circuit breaker to prevent an increasing fuel unbalance.

2. Reset circuit breaker after affected tanks are empty

If steps 1. and 2. do not establish external fuel transfer:

3. EXT TANK FUEL TRANS circuit breaker – RESET

4. Throttle – MILITARY

5. Descend to 10000 feet or below

6. Fresh air scoop lever – Open

NOTE

Monitor external fuel quantities for simultaneous decrease.

If necessary to get tip tanks to feed:

7. Pylon tanks – JETTISON

NOTE

With empty tip tanks and more than residual fuel in pylon tanks, adhere to airspeed limitation of 500 knots.

External Tip Tank Asymmetrical Fuel Load

If asymmetric fuel load exceeds 450 lbs, take immediate corrective action to stop external fuel transfer:

1. External fuel transfer – Stop

2. Do not exceed Mach 0.9, 30° bank, 2.0 g

3. Minimum speed – 265 KIAS

WARNING

USING FULL UNLIMITED AILERON OR RUDDER TRAVEL IN MANEUVERS ABOVE 300 KIAS CAN RESULT IN STRUCTURAL DAMAGE AND POSSIBLE AIRCRAFT LOSS.

If roll control becomes marginal:

4. "G" load factor – Reduce

5. AIL RUD LIMITER switch – Lift cover UNLTD

NOTE

When performing a dual flight, the full unlimited aileron and rudder travel, may be obtained by pulling the RUD AIL LIMIT CONT circuit breaker located in the rear cockpit left console.

6. Land as soon as practical using ASYMMETRIC TIP TANK FUEL LOAD procedure

7. If necessary: Tip tanks – Jettison

HYDRAULIC SYSTEM FAILURE

WARNING

DURING ZERO OR NEGATIVE "G" FLIGHT, PRESSURE IN ONE OR BOTH HYDRAULIC SYSTEMS MAY BE TEMPORARILY LOST IF AIR IS PRESENT IN THE SYSTEM. IF LOSS OF HYDRAULIC SYSTEM PRESSURE IS EXPERIENCED AT ZERO OR NEGATIVE "Gs", IMMEDIATELY RETURN THE AIRCRAFT TO POSIVITE "G" CONDITIONS. IF THIS MANEUVER IS NOT SUFFICIENT TO RESTORE THE HYDRAULIC PRESSURE, ASSIMILATE THIS FAILURE AS NO. 1 AND/OR NO. 2 HYDRAULIC SYSTEMS OUT AND PROCEED CONSEQUENTLY.

NOTE

Failure of one hydraulic system while operating at higher airspeeds can cause a significant reduction in flight control effectiveness and can restrict control stick movement to the extent that the stick appears to be locked.

WARNING

IF LOSS OF PRESSURE ON ONE HYDRAULIC SYSTEM OR RESTRICTED CONTROL STICK MOVEMENT IS EXPERIENCED AT SPEEDS GREATER THAN MACH 1.0, IMMEDIATELY DECELERATE TO 400 KIAS OR MACH 0.9, WHICHEVER IS LESS, BY REDUCING THROTTLE TO IDLE AND EXTENDING SPEEDBRAKES, IF POSSIBLE. THESE ACTIONS WILL IMPROVE FLIGHT CONTROL EFFECTIVENESS AND ENSURE ADEQUATE LOAD FACTOR CAPABILITY FOR MANEUVERING. FLIGHT CONTROL RESPONSE WILL BE SLOWER WITH ONE HYDRAULIC SYSTEM INOPERATIVE.

No. 1 System Out

Indicated by:

- HYDRAULIC SYSTEM OUT warning light lit and No. 1 low system pressure
- AUTO PITCH CONT OUT warning light lit
- Yaw damper inoperative

1. No. 2 pressure – Monitor for the remainder of flight

WARNING

THE HYDRAULIC SYSTEM OUT WARNING LIGHT WILL NOT INDICATE A SECOND FAILURE. THE REMAINING GAGE MUST BE MONITORED TO DETERMINE IF SUBSEQUENT FAILURE OCCURS.

2. Land ASAP using PRECAUTIONARY STRAIGHT-IN PATTERN procedure

No. 2 System Out

WARNING

CLOSE SPEED BRAKES IF A HYDRAULIC FAILURE IS IMMINENT. THE SPEED BRAKE CANNOT BE CLOSED WITHOUT NO. 2 HYDRAULIC SYSTEM PRESSURE.

Indicated by:

- HYDRAULIC SYSTEM OUT warning light lit and low No. 2 pressure
- FIXED FREQ OUT warning light lit
- Pitch and roll dampers inoperative
- Items operated by the utility hydraulic system inoperative

1. No. 1 hydraulic system pressure gage – Monitor for the remainder of flight
2. Land ASAP using PRECAUTIONARY STRAIGHT-IN PATTERN procedure
3. Extend gear with the MAN LDG GEAR handle

CAUTION

- NOSEWHEEL STEERING IS INOPERATIVE WITH NO. 2 HYDRAULIC SYSTEM FAILED. ATTEMPTING TO USE NOSEWHEEL STEERING WITH NO. 2 HYDRAULIC SYSTEM FAILED OR FLUCTUATING WILL CAUSE AIR TO ENTER THE SYSTEM RESULTING IN VIOLENT SHIMMY WHICH CAN CAUSE SUBSTANTIAL DAMAGE.

- POWER BRAKES ARE INOPERATIVE WITH NO. 2 HYDRAULIC SYSTEM FAILED. STANDBY BRAKES MUST BE USED FOR DIRECTIONAL CONTROL AND BRAKING. USE CAUTION TO AVOID A BLOWN TIRE. IF A CABLE ARRESTMENT IS NECESSARY, USE RUDDER AND STANDBY BRAKES TO STEER 90° TO THE CABLE.

No. 1 and No. 2 Hydraulic Systems Out

1. RAT handle – Pull (above minimum recommended airspeed), refer to RAT EXTENDED FLIGHT procedure

NOTE

The RAT provides approximately 1/6 normal hydraulic flow. If the stick is moved rapidly, the controls may freeze until sufficient pressure builds up.

2. No. 1 hydraulic system pressure gage – Monitor

If pressure builds up:

3. Land ASAP using PRECAUTIONARY STRAIGHT-IN PATTERN procedure
4. Landing gear – Extend WITH MAN LDG GEAR handle

CAUTION

- NOSEWHEEL STEERING IS INOPERATIVE WITH THE NO. 2 HYDRAULIC SYSTEM FAILED. ATTEMPTING TO USE NOSEWHEEL STEERING WITH NO. 2 HYDRAULIC SYSTEM FAILED OR FLUCTUATING WILL CAUSE AIR TO ENTER THE SYSTEM RESULTING IN VIOLENT SHIMMY WHICH CAN CAUSE SUBSTANTIAL DAMAGE.

- POWER BRAKES ARE INOPERATIVE WITH NO. 2 HYDRAULIC SYSTEM FAILED. STANDBY BRAKES MUST BE USED FOR DIRECTIONAL CONTROL AND BRAKING. USE CAUTION TO AVOID A BLOWN TIRE. IF A CABLE ARRESTMENT IS NECESSARY, USE RUDDER AND STANDBY BRAKES TO STEER 90° TO THE CABLE.

NOTE

Maximum hydraulic fluid available is reduced; however, flow is sufficiently high for safe flight and moderate maneuvers necessary for landing

If pressure fails to provide adequate flight control response:

3. EJECT

Dive recovery

With one hydraulic system inoperative, the maximum available stabilizer hinge moment is reduced by one-half. This result in a considerable reduction in load factor capability at speeds above Mach 1.0. This reduced load factor capability greatly increases the altitude required for low altitude dive recovery, and increases the necessity to reduce thrust to idle and use speedbrakes, if available, to effect a safe dive recovery.

WARNING

SPEEDBRAKES, CANNOT BE EXTENDED WITHOUT NO. 2 HYDRAULIC SYSTEM PRESSURE.

Figure 3-6 shows the low altitude load factor capability with only one hydraulic system operative for the most critical loading configuration for the aircraft. Note that above Mach 1.0 (525 knots IAS at 15000 feet) recovery is not possible unless power is reduced and/or drag increased by use of speedbrakes even from relatively shallow dives.

LOAD FACTOR CAPABILITY WITH ONE HYDRO SYSTEM OPERATING AS LIMITED BY STABILIZER HINGE MOMENT

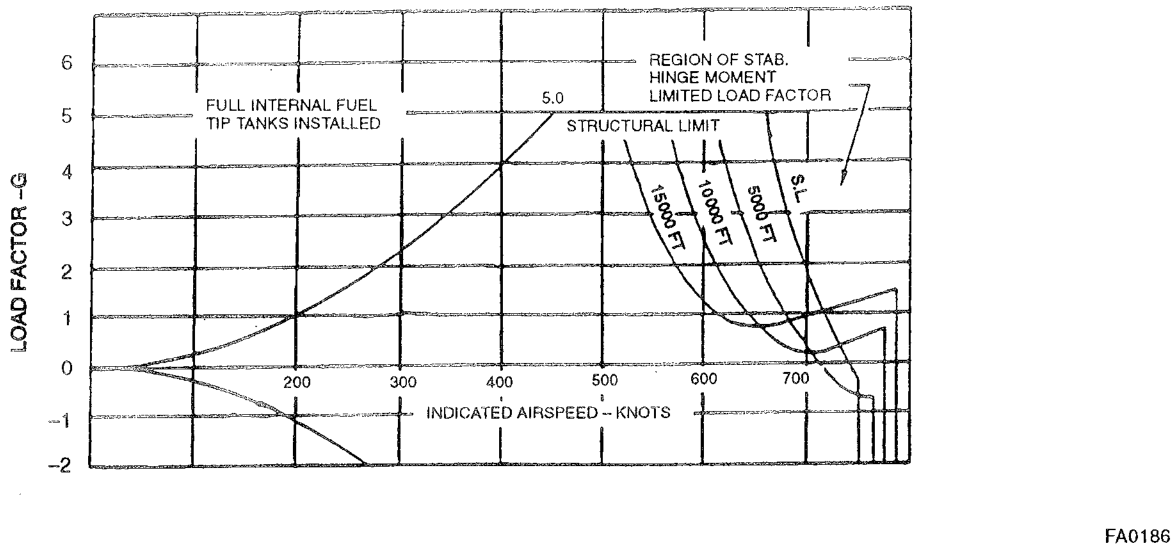


Figure 3-6

IGV CLOSURE

Indicated by:

- Abnormally low fuel flow
- Severe thrust reduction

NOTE

> EGT will be normal due to nozzle modulating open; however, if nozzle is full open EGT will then increase. RPM will be normal for throttle setting.

1. Throttle – MAX A/B
2. EGT – Monitor and retard throttle in A/B range as necessary to avoid overtemperature

If necessary:

3. External stores – Jettison
4. Establish a precautionary pattern or land with A/B

OXYGEN SYSTEM FAILURE

If symptoms of hypoxia or anoxia are noted:

1. Mask connections – Check
2. Green apple – Pull
3. Oxygen supply lever – OFF (normal oxygen may be contaminated)
4. Descent immediately to below 10000 ft
5. Radio – Call MAYDAY and state intentions
6. IFF – EMER

If below 10000 ft:

7. Fresh air scoop lever – Open
8. Mask – Disconnect (after bottle is empty and supply stops)
9. Land ASAP

NOTE

> The emergency oxygen bottle should supply oxygen for about 4 minutes.

PITCH-UP RECOVERY AND SPIN PREVENTION/SPIN RECOVERY

Pitch-Up Recovery and Spin Prevention

Subsonic 1-G/accelerated pitch-up is preceded by buffet, stick shaker, lateral instability, and kicker. If the kicker warning is ignored and the angle-of-attack

DIVE RECOVERY WITH ONE HYDRO SYSTEM OPERATING

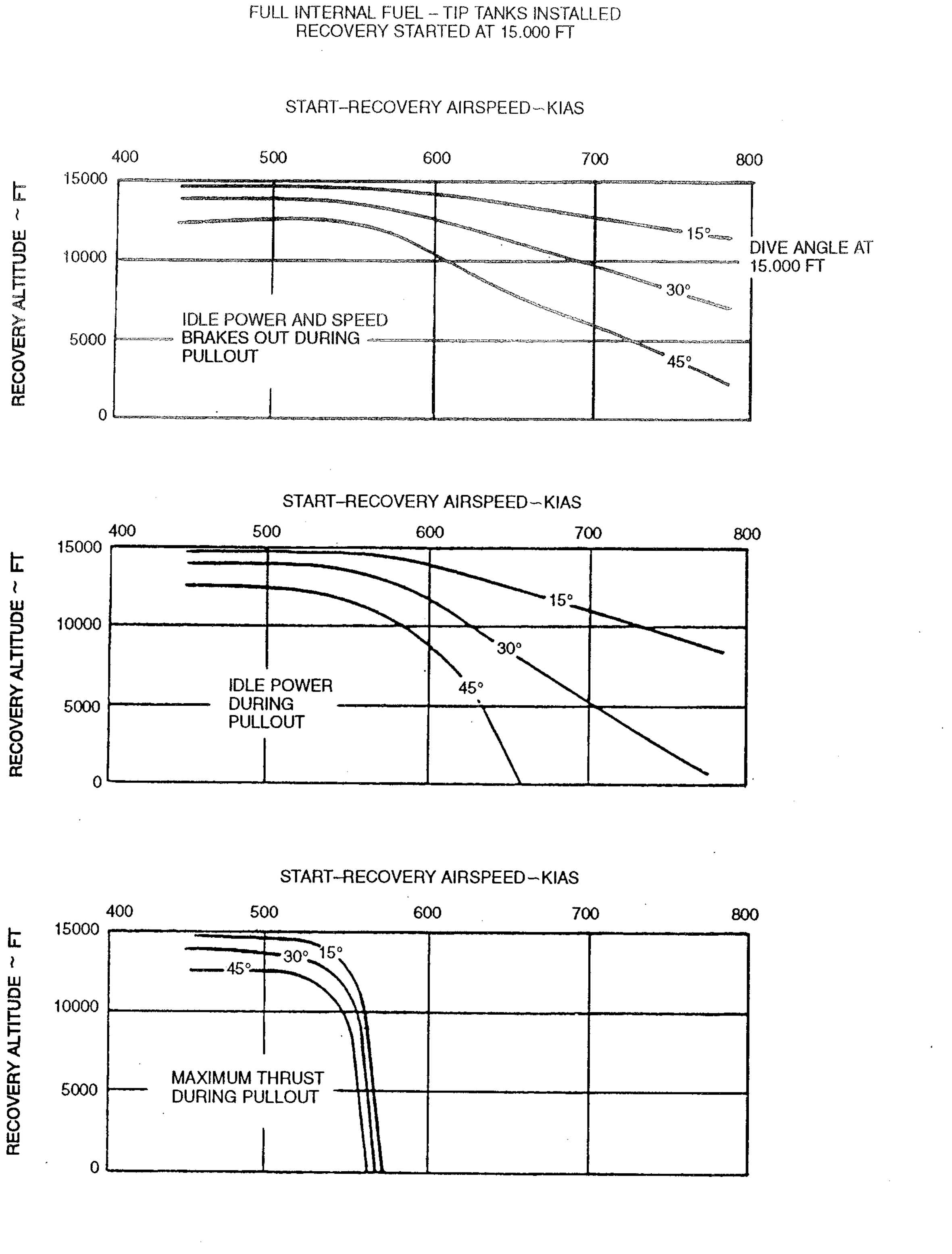


Figure 3-7

is increased beyond the kicker boundary the aircraft will pitch up and enter post-stall gyrations of uncontrolled motions about all axis.
During these motions the turn needle will oscillate around the mid-position. Entry into a spin mode may be prevented if immediate corrective action is taken. Refer to Section VI for detailed pitch-up information.

WARNING

A FULLY-DEVELOPED PITCH-UP MAY REQUIRE 15000 FEET BEFORE RECOVERY IS COMPLETE.

NOTE

The pitch-up differs from a developed spin. During pitch-up, the turn needle will oscillate around the mid-position, where during a spin, the turn needle will indicate a steady direction.

At first indication of pitch-up, if time and altitude permits:

1. **STICK – FULL FORWARD WITH FULL NOSE-DOWN TRIM**
2. **RUDDER AND AILERONS – NEUTRAL**

If extended:

3. Gear, wing flaps, and speed brakes – Retract

If oscillations stop and airspeed begins to increase:

4. Start a gradual pullout observing APC limitations and avoiding excessive aircraft buffet
5. Wing flaps – TAKEOFF (observe airspeed limitations)

If turn needle indicates a steady full deflection:

4. Maintain full forward stick and perform SPIN RECOVERY procedure

Spin Recovery

The normal spin is characterized by pronounced oscillations in pitch, roll, and yaw. The flat spin is characterized by no oscillations about any axis. The turn needle will always be pegged in the direction of the spin. Accomplish spin recovery procedures immediately to prevent early oscillatory spin from developing into flat spin from which flight controls may not effect recovery. Refer to Section VI for detailed spin information.

If aircraft enters a spin, and time and altitude permits:

1. **THROTTLE – IDLE**
2. **RUDDER – FULL OPPOSITE TO SPIN ROTATION (opposite turn needle)**
3. **STICK – FULL FORWARD, WITH FULL NOSEDOWN TRIM AND FULL IN THE DIRECTION OF SPIN (with turn needle)**

If above 25000 ft:

4. Drag chute – DEPLOY

If extended:

5. Gear, wing flaps and speed brakes – Retract

If rotation stops:

6. Drag chute – Jettison as nose swings down

WARNING

USE FORWARD STICK TO PREVENT ABRUPT NOSE-UP PITCH AS CHUTE IS JETTISONED OR FAILS.

NOTE

Chute panels will burn out at RPM above 90% and shear link should fail above 225 kts.

7. Aileron and rudder controls – Neutralize
8. Start gradual pullout, observe APC limitations and avoid excessive aircraft buffet
9. Wing flaps – TAKEOFF (observe airspeed limitations)

If engine has stalled or flamed out:

10. Perform AIR START/STALL CLEARING procedure

If spin rotation has not stopped by 15000 ft AGL:

6. **EJECT**

Inverted Spin Recovery

1. NEUTRALIZE ALL CONTROLS

If spin rotation has not stopped by 15000 ft AGL:

2. EJECT

NOTE

Refer to Section VI, Flight Characteristics, for further information on pitch-up and spin characteristics/prevention, and use of drag chute.

RAT EXTENDED FLIGHT

The ram air turbine (RAT) is available for emergency electrical and hydraulic power when the engine-driven power sources are lost. Extension of the RAT with the engine running may, under certain conditions, adversely affect engine operation.

WARNING

- FLIGHT WITH RAT EXTENDED AND WING FLAPS IN LAND POSITION WILL RESULT IN A STRONG RIGHTROLL TENDENCY AT SPEEDS BELOW 160 KIAS. THE ROLL IS CAUSED BY TURBULENT AIRFLOW FROM THE RAT OVER THE INBOARD RIGHT WING SECTION. FULL AILERON MAY BE REQUIRED TO STOP THE ROLLOFF AND MAINTAIN WINGS-LEVEL FLIGHT; THEREFORE, NEIGHTER EXTEND RAT WITH LAND FLAPS EXTENDED NOR EXTEND WING FLAPS TO LAND WITH RAT EXTENDED.
- ALL FUEL BOOST PUMPS ARE NOT AVAILABLE WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF OR FAILED AND, WHEN REMAINING FUEL (INTERNAL) IS BELOW 1000 LBS. A FLAMEOUT COULD OCCUR IF RAT FAILS TO POWER NO. 3 FUEL BOOST PUMP. MOREOVER FLIGHT WITH NO. 3 FUEL BOOST PUMP POWERED BY RAT AND 1000 LBS. OR LESS INTERNAL FUEL REMAINING, SHOULD BE PERFORMED IN NOSE UP FLIGHT ATTITUDE WITH EASY COORDINATED MANEUVERS AND NEGLIGIBLE DECELERATION FORCES. FLIGHT WITH NOSE DOWN AT THIS FUEL LOAD MAY ALLOW THE NO. 3 FUEL BOOST PUMP TO CAVITATE RESULTING IN SUBSEQUENT ENGINE FLAME OUT. ILLUMINATION OF THE FUEL BOOST PUMPS WARNING LIGHT UNDER THESE CONDITIONS MAY NOT PROVIDE ADEQUATE WARNING TO TAKE CORRECTIVE ACTIONS AND PREVENT AN ENGINE FLAME OUT.
- LANDING WITH RAT EXTENDED AND 1000 LBS. OR LESS REMAINING FUEL SHOULD BE PERFORMED FOLLOWING "PRECAUTIONARY STRAIGHT-IN PATTERN" PROCEDURE.

Because several inadvertent RAT extensions have been experienced, an operating envelope with the engine running and the RAT extended is provided below.

a. The RAT may be extended in level flight without affecting engine operation within the following airspeed limits:

Altitude	Airspeed Limits (KIAS)	
	Min	Max
Up to 30000 feet	None	550
Above 30000 feet	350	

b. Normal air starts with the RAT extended can be made at all altitude up to 35000 feet.

c. Maneuverability is satisfactory with the No. 1 and No. 2 hydraulic pumps inoperative and the RAT supplying hydraulic pressure up to 550 KIAS.

d. Spiral climbs and descents may be made without affecting normal engine operation or aircraft maneuverability.

e. Best range with the RAT extended and 3000 pounds of fuel remaining is realized by cruising at 0.82 Mach at 27000 feet. Range will be approximately 170 nautical miles per 1000 pounds of fuel used.

f. Factors such as G's, yaw, abrupt maneuvers, or rapid throttle movements may induce engine instability, stalls, or flameouts with the RAT extended, especially above 30000 feet. Below 30000 feet, 45° banks do not affect engine operation.

g. With RAT extended, approximately the same glide distance can be obtained at 245 KIAS and TAKEOFF flaps as with 275 KIAS and no flaps. However, the slower speed will result in a lower rate of descent.

Deploy the RAT under the following circumstances:

– Double hydraulic failure

– Double generator failure

– Seized engine

– Dead engine descent in weather

NOTE

If a flameout or engine stall occurs when the RAT is extended, accomplish normal AIR START/STALL CLEARING procedure.

When flying with the RAT extended, avoid abrupt or uncoordinated maneuvers and move throttle slowly and only when necessary.
Do not attempt afterburner lights unless absolutely necessary.
Land as soon as practical and use thrust as required.
If possible, fly a PRECAUTIONARY STRAIGHT-IN approach.

NOTE

- Extending the RAT powers hydraulic system No. 1, energizes the XP4 bus, both PP2 and PP3 busses, and both PP4 and PP5 busses. In addition, if the hydraulic generator is not operating, the RAT energizes the XP5 and XP6 busses. Illumination of the warning lights indicates RAT operation.

- The leading and trailing edge flaps are sequenced to extend separately to the TAKEOFF position only when using RAT-driven generator for electrical power. Therefore, the LAND or UP position should never be selected under this conditions as it may stall the generator. Keep airspeed above 200 KIAS until flaps have reached TAKEOFF position. The wing flap position indicator will be inoperative.

SMOKE OR FUMES IN COCKPIT

If smoke or fumes enter the cockpit proceed as follows:

1. **OXYGEN DILUTER LEVER – 100%**
2. Descend to 25000 feet or lower if circumstances permit
3. Fresh air scoop lever – OPEN

If smoke persists:

4. Land ASAP; if necessary: Canopy – Jettison
5. If it can be determined that smoke is caused by an electrical fire, use ELECTRICAL FIRE procedure

WARNING

- ALL FUEL BOOST PUMPS ARE NOT AVAILABLE WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF AND, WHEN REMAINING FUEL (INTERNAL) IS BELOW 1000 LBS, A FLAMEOUT COULD OCCUR IF RAT FAILS TO POWER NO. 3 FUEL BOOST PUMP. MOREOVER FLIGHT WITH NO. 3 FUEL BOOST PUMP POWERED BY RAT AND 1000 LBS. OR LESS INTERNAL FUEL REMAINING, SHOULD BE PERFORMED IN NOSE UP FLIGHT ATTITUDE WITH EASY COORDINATED MANEUVERS AND NEGLIGIBLE DECELERATION FORCES. FLIGHT WITH NOSE DOWN AT THIS FUEL LOAD MAY ALLOW THE NO. 3 FUEL BOOST PUMP TO CAVITATE RESULTING IN SUBSEQUENT ENGINE FLAME OUT. ILLUMINATION OF THE FUEL BOOST PUMPS WARNING LIGHT UNDER THESE CONDITIONS MAY NOT PROVIDE ADEQUATE WARNING TO TAKE CORRECTIVE ACTIONS AND PREVENT AN ENGINE FLAME OUT.

- LANDING WITH RAT EXTENDED AND 1000 LBS. OR LESS REMAINING FUEL SHOULD BE PERFORMED FOLLOWING "PRECAUTIONARY STRAIGHT-IN PATTERN" PROCEDURE.

CAUTION

- WHEN BOTH WILD FREQUENCY GENERATORS ARE OFF, THE EXTERNAL FUEL QUANTITY INDICATION ARE NOT AVAILABLE AND FUEL DATA ARE FROZEN. WHEN RAT IS EXTENDED, THE INTERNAL FUEL QUANTITY INDICATION IS AVAILABLE EXCEPT WHEN FLAPS/SLATS ARE OPERATED: IN THIS CONDITION FUEL QUANTITY DATUM IS TEMPORARILY FROZEN.

- WHEN BOTH WILD FREQUENCY GENERATOR ARE OFF OR FAILED, NO WARNING LIGHTS (EXCEPT FIRE) ARE AVAILABLE UNTIL RAT IS EXTENDED.

STABILITY AUGMENTATION SYSTEM FAILURE

Failure in any one of the stability augmentation system channels (roll, pitch or yaw) may cause control system oscillation. One or all three of the STABILITY CONT circuits can be disengaged:

As required:

1. ROLL/PITCH/YAW – OFF

THROTTLE CONTROL SYSTEM MALFUNCTION

If an uncommanded decrease or increase in RPM is experienced, a failure in the cable of the throttle control system may have occurred.
During a throttle control system malfunction, the EGT and nozzle will correspond to stabilized engine RPM. If a throttle control system malfunction occurs, the RPM may increase or decrease, depending on the location of the cable failure. The most probable failure location is in the advance-throttle cable and will result in a decrease in RPM. If the engine is at or near Military power when this failure and RPM decrease occurs, the engine will probably roll back to approximately 92% RPM.

If the engine is at a lower power setting when the failure occurs, the RPM will decrease further and the engine may flame out. If this failure has occurred and the throttle is subsequently pulled back, RPM will be further reduced with no capability to increase it with throttle advancement. If a sudden decrease in RPM is noted with no change in throttle setting, note other engine instruments (EGT, nozzle) to determine if the problem is a compressor stall or throttle control system failure.

If a throttle control system failure is suspected:

1. Throttle — SLOWLY ADVANCE and check for any change in engine instrument readings

WARNING

DO NOT RETARD THROTTLE FROM ORIGINAL POSITION UNTIL TYPE OF MALFUNCTION MAY BE DETERMINED. IF THE ADVANCE-THROTTLE CABLE HAS FAILED, ANY REDUCTION IN THROTTLE POSITION WILL ALSO REDUCE ENGINE RPM/THRUST, WITH NO CHANCE OF RECOVERY.

NOTE

If any engine instrument(s) change as a result of the throttle advance, some other type of engine component failure has occurred and advance-throttle cable failure can be discounted.

If there is no engine instrument(s) change due to throttle advance:

2. Throttle — MAX A/B

NOTE

Placing the throttle in the max A/B position with an advance-throttle cable failure will not increase RPM or select afterburner. This procedure will cause a release of tension on the retard-throttle cable to the main fuel control unit, decreasing the chance of any further reduction in RPM/thrust.

If sufficient thrust is available:

3. Land ASAP

WARNING

DO NOT RETARD THE THROTTLE BELOW MAX AFTERBURNER POSITION UNTIL LANDING IS ASSURED, AS ENGINE SHUTDOWN AT A MID-THROTTLE POSITION IS POSSIBLE.

If throttle retard fails to shut off engine:

4. FUEL SHUT-OFF switch — OFF

If sufficient thrust is not available:

3. Eject

TRIM FAILURE OR RUNAWAY TRIM

If trim fails:

1. STICK TRIM-AUX TRIM selector switch Lift guard, AUX TRIM
2. AUX TRIM CONTROL — Use as necessary

NOTE

Maximum nose-down stabilizer travel is dependent upon trim setting. In the event of stick trim button failure, resulting in full nose-up trim, the auxiliary trim switch will have to be used to decrease the nose-up to gain full nosedown capability of the stabilizer.

If trim still malfunctions:

3. TRIM CONTROL circuit breaker (rear cockpit) — Pull

If trim failure has caused full nose-down trim:

4. Land as soon as practicable, using PRECAUTIONARY STRAIGHT-IN PATTERN procedure (TAKEOFF flaps mandatory)

WARNING

IN THE EVENT OF TRIM FAILURE RESULTING IN FULL NOSE-DOWN TRIM, AVAILABLE NOSE-UP STABILIZER TRAVEL WILL BE REDUCED BY 4°. IN THIS CASE A STRAIGHT-IN TAKEOFF FLAPS LANDING IS MANDATORY.

INS FAILURE

Failure of the INS is indicated by the illumination of the amber INERTIAL NAV FAULT warning light on the warning lights panel and by the amber FAIL lamp on the IN control panel. If the aircraft was flying in the INS steering mode, the relevant "SEL" caption, on the navigation steering mode selector pushbutton extinguishes.

The following utilities are lost:

- IN/CDU
- attitude indications (pitch and roll)
- horizontal situation indicator (the data on the HSI will be available again when TACAN steering mode is selected by pressing TCN/SEL pushbutton)

The TACAN steering mode is available provided that the TACAN is set to on.
On HSI all data are TACAN outputs except for the magnetic heading which is derived from the C-2G.
On HSI actual track indication is hidden by the relative bearing indication.

1. TACAN – On, check/set channel and mode
2. TCN/SEL mode selector – Press, check SEL caption lit
3. C-2G – Check/set MAG
4. C-2G – Crosscheck magnetic heading with standby compass

APPROACH AND LANDING EMERGENCIES

AIRSPEED SYSTEM FAILURE

The best procedure for a landing approach with questionable airspeed indication is to fly formation with another aircraft. However, if another is not available, the APC meter can be used to indicate equivalent angle-of-attack which can be used to accomplish a safe approach for all conditions of weight and drag.

1. At a safe altitude, fly a 1-G stall approach to stick shaker action with gear and land flaps extended
2. Note the APC meter reading at beginning of stick shaker action
3. Subtract 2 1/2 from the APC meter stick shaker value and hold this meter number on final approach
4. If unable to perform the stall check, fly to hold an APC meter reading of one

APPROACH-END ARRESTMENTS

Approach-end arrestments reduce the exposure time to which the pilot and aircraft are subjected during a landing roll with adverse directional control; therefore, approach-end engagements can be made when landing with one main gear up or unlocked, or whenever a directional control problem after touchdown is anticipated.

NOTE

- Landing with a blown main tire does not present a critical directional control problem.
- Refer to ABORT/BARRIER ENGAGEMENT procedure (MAXIMUM BARRIER ENGAGEMENT SPEEDS).

Approach-end arrestments are practical only when the barrier has at least 1000 feet of runway ahead of the barrier and a clear approach. Make sure the the MA-1A (chain type)/all net type barrier has been removed prior to landing. If possible, burn excess fuel and jettison external stores to reduce landing gross weight and minimize fire hazard.

However, if landing with one main gear up or unlocked, retain the empty tip tank and pylon tanks to cushion wing drop. In any event, retain empty tip tanks if they are the only external stores.

1. Shoulder harness – Check locked
2. HOOK RELEASE – Press. If time permits, confirm hook extension by other aircraft or tower
3. Make straight-in flat approach at minimum practical landing speed. Plan touchdown on runway or hard surfaces overrun at least 500 to 1000 feet short of barrier. Refer to Figure 3-2 for maximum barrier engagement speeds

Immediately after touchdown:

4. Lower nosewheel to runway
5. Throttle – IDLE

CAUTION

THE NOSEWHEEL MUST BE ON THE RUNWAY PRIOR TO BARRIER ENGAGEMENT, OTHERWISE THE NOSE GEAR MAY FAIL AS IT CONTACTS THE RUNWAY.

6. Nosewheel steering – Engage. Contact barrier as close as possible to 90° angle

CAUTION

DO NOT USE THE BRAKES. A LOCKED WHEEL MAY SNAG OR CUT THE CABLE.

NOTE

The cockpit canopy provides protection from flash fire; therefore, do not jettison the canopy.

If decision is made to shutdown engine:

7. Throttle – OFF
8. FUEL SHUT-OFF switch – OFF

When aircraft stops:

9. Perform GROUND ABANDONMENT procedure

ASYMMETRIC TIP TANK FUEL LOAD

Adequate control is available for landing with one external tank full and one external tank empty under smooth air conditions. Lateral control can be improved by using TAKEOFF flaps.
Consideration should be given to the added aileron requirements under strong or gustly crosswind conditions before attempting a landing with an asymmetric fuel load.
A crosswind from the side with the light tank increases the aileron requirements in the same direction as used to balance the heavy tank. Low speed control should be tested prior entering the landing pattern. If lateral control appears marginal for the landing pattern, the tanks should be jettisoned.

1. Plan landing pattern so that crosswind is from the heavy tank side

NOTE

If it is not possible to land with crosswind on the heavy tank side, landing is still possible with a crosswind component of up to 5 knots on the light tip tank side.

2. Maintain at least 260 KIAS until established for STRAIGHT-IN approach (5 NM/1000 ft AGL – 400/500 ft/min)
3. Maintain the speeds and flap configuration listed below during approach and touchdown

Speed Configuration	Flap Position	Approach KIAS	Touchdown KIAS
TIP	T/O	225	196
TIP + PYLON	T/O	235	204
TIP	LAND	215	188

If crosswind from light tank side is greater than 5 KIAS:

2. Tip tanks – Jettison
3. Perform normal approach and landing

BARRIER ENGAGEMENT

Refer to ABORT/BARRIER ENGAGEMENT procedure under TAKEOFF EMERGENCIES.

BELLY LANDING

The decision to eject or to accomplish a belly landing must remain with the pilot. Belly landing should be considered.

1. Empty tip and/or pylon tanks – Retain
2. Fuel – Burn down to minimum safe level
3. Shoulder harness – LOCKED; seat belt and harness – TIGHT
4. Helmet visor – Check DOWN
5. AIL RUD LIMITER switch – Lift cover UNLTD

NOTE

When performing a dual flight, the full unlimited aileron and rudder travel, may be obtained by pulling the RUD AIL LIMIT CONT circuit breaker located in the rear cockpit left console.

6. Perform straight-in-pattern: gear handle – UP; wing flaps – LAND; establish flat final

At touchdown:

7. Throttle – OFF
8. Drag chute – Deploy
9. FUEL SHUT-OFF switch – OFF

When aircraft stops:

10. Perform GROUND ABANDONMENT procedure

BEST GLIDE

Figure 3-8 shows the glide distance obtainable with a windmilling or seized engine. The recommended configuration is with TAKEOFF flaps and 245 KIAS. Add 5 KIAS for each 1000 pounds weight above 16000 pounds.
Approximately the same distance can be obtained by gliding with flaps up at 275 KIAS; however, the rate of descent with TAKEOFF flaps is approximately 1000 feet per minute less due to the lower speed for the same glide ratio.

The data shown in the chart are for RAT extended since this configuration represents the highest drag and is necessary for flap extension under any engine inoperative condition and for hydraulic power under a seized engine condition. Gliding without RAT extended increases these distance approximately 2 nautical miles per 10000 ft of altitude.

NOTE

Unless the engine is damaged, the windmilling engine will produce sufficient hydraulic pressure to operate the flight control system.

BOUNDARY LAYER CONTROL MALFUNCTION

NOTE

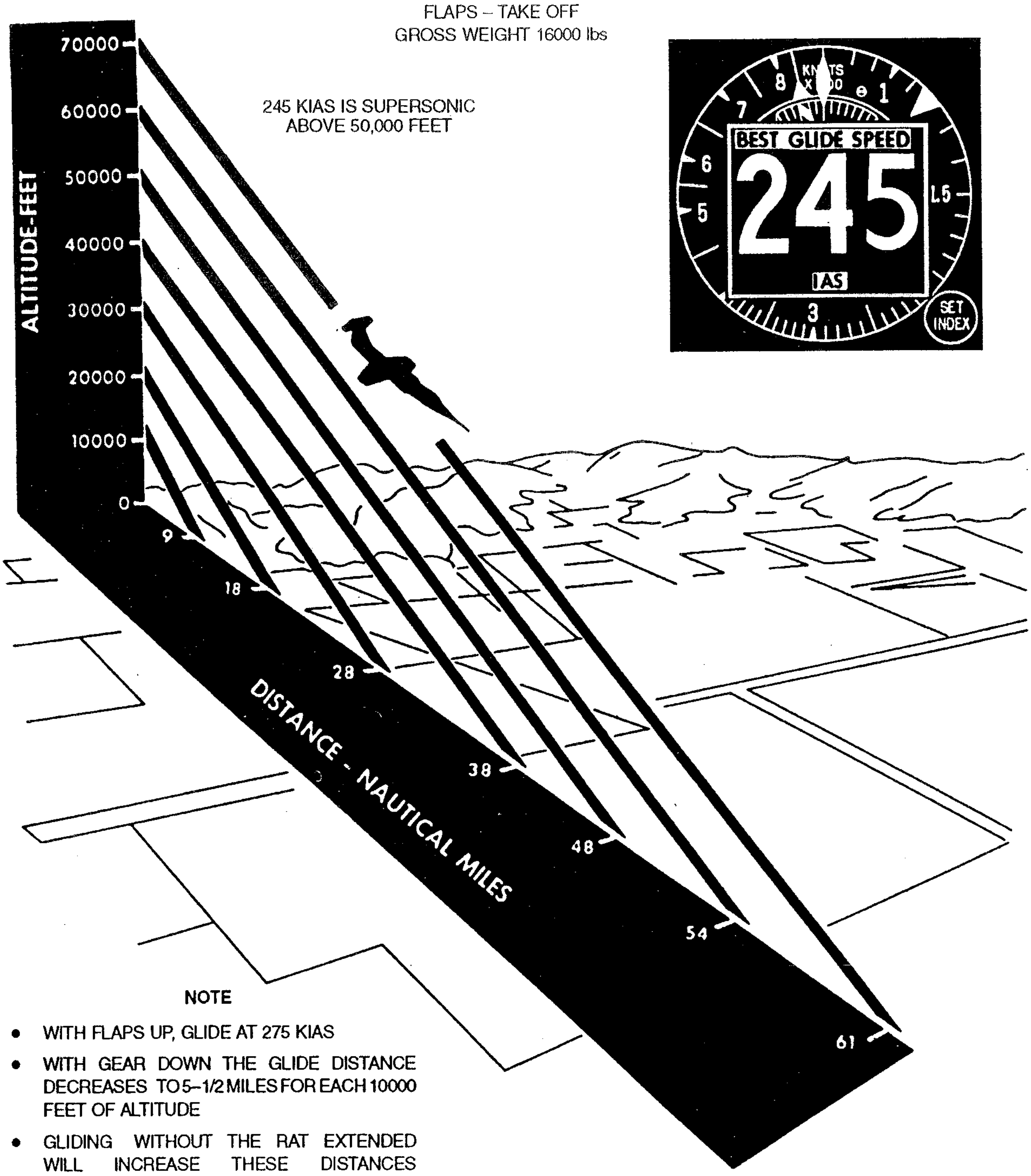
The airspeeds listed herein are based on a gross weight of 16000 pounds. Adjust approach and landing speeds 5 knots for each 1000 pounds difference in gross weight or portions thereof. Refer to landing speed schedule in the Appendix for other configurations and gross weights.

If strong rolling tendency is experienced as wing flaps travel to LAND position:

1. Wing flaps – TAKEOFF
2. Throttle – Adjust to minimum safe setting to reduce the effect of asymmetric BLC while flaps are returning to TAKEOFF
3. Fly final approach at not less than 195 KIAS with wing flaps in TAKEOFF position
4. Touch down at 165 KIAS minimum

MAXIMUM GLIDE DISTANCE

ZERO WIND – STRAIGHT-LINE GLIDE
ENGINE WINDMILLING OR FROZEN
EXTERNAL STORES – NONE OR TIP TANKS INSTALLED
GEAR UP – RAT EXTENDED
FLAPS – TAKE OFF
GROSS WEIGHT 16000 lbs



NOTE

- WITH FLAPS UP, GLIDE AT 275 KIAS
- WITH GEAR DOWN THE GLIDE DISTANCE DECREASES TO 5-1/2 MILES FOR EACH 10000 FEET OF ALTITUDE
- GLIDING WITHOUT THE RAT EXTENDED WILL INCREASE THESE DISTANCES APPROXIMATELY 2 NAUTICAL MILES PER 10000 FEET OF ALTITUDE

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Figure 3-8

FLAP FAILURE

NOTE

When either set of flaps are operating on one actuator it is possible to overload the actuator, causing flap drive disengagement and a barber pole indication. Should disengagement occur during extension in flight, the flap lever should be returned to the previously selected position and left there until landing. When the trailing edge flaps are operating on one motor, landing should be accomplished using TAKEOFF flaps. Extension to TAKEOFF flaps may be accomplished at any speed below 370 KIAS. Retraction of the flaps with one motor can be accomplished in all cases except retraction of leading edge flaps from LAND to TAKEOFF at speeds in excess of 230 KIAS.

No-flap landings require high approach and touchdown speeds, during which the maneuverability of the aircraft around the pitch-axis is very critical. Therefore, a successful no-flap landing can best be executed if a flap approach is made. This eliminates the possibility of misjudging the rotation, which could result in an uncontrollable rate of descent.

Trailing Edge Flaps

If trailing edge flaps fail to extend (regardless of position of leading edge flaps):

1. Reduce fuel load to 3000 lbs or less

WARNING

THIS PROCEDURE IS HAZARDOUS WITH MORE THAN 3000 LBS FUEL LOAD.

If landing cannot be delayed and tanks contain fuel:

2. External tanks – Jettison (refer to EXTERNAL STORES EMERGENCY/SELECTIVE JETTISON procedure).
3. Fly a long, flat final approach of at least 5 nm, beginning at 1000 feet AGL with gear extended
4. Final approach speed – 230 KIAS minimum at 16000 lbs (adjust for gross weight). Maintain appr. 800 fpm rate of descent
5. APC reading 2.5 maximum
6. At approximately 1 nm from beginning of runway, start decreasing rate of descent to arrive over beginning of runway with a near-landing attitude

CAUTION

DO NOT AT ANY TIME ALLOW THE RATE OF DESCENT TO EXCEED 2000 FEET/MINUTE AS RECOVERY WILL REQUIRE A MINIMUM OF 400 FEET ALTITUDE.

7. Touchdown speed – 195 KIAS minimum
8. Lower nose and deploy drag chute, observing drag chute limits

CAUTION

DO NOT DEPLOY DRAG CHUTE AT ENGINE SPEEDS ABOVE IDLE RPM.

If leading edge flaps fail:

1. If only the leading edge flaps fail and the trailing edge flaps can be lowered to the LAND position (BLC available), normal pattern and touchdown speeds can be used.
2. If trailing edge flaps can be extended only to the TAKEOFF position, fly final approach at not less than 195 KIAS and touchdown at not less than 165 KIAS.

Emergency Wing Flap Operation

Partial flap extension is obtainable for landing when alternating current is being furnished under emergency conditions by the ram air turbine. With the RAT extended, wing flap extension is obtained by placing the wing flap lever in the TAKEOFF position. The flaps are then extended in sequence (trailing edge first) to the TAKEOFF position, thereby reducing the RAT momentary electrical load.

Asymmetric Flaps

It is possible that an asymmetric wing flap condition occurs any time the flaps configuration is changed. When the asymmetry condition exceed 3.5° (±1°) the asymmetry detector switch will open the trailing edge flap control circuit thus stopping flap movement. The same switch will exclude the aileron and rudder limiters so that full aileron and rudder travel will be available even if the landing gear is UP.
At the same time the FLAP ASYMMETRY and the AIL AND RUD UNLIMITED lights will illuminates. Under most conditions lateral control will be sufficient to maintain level flight and to land. The most critical time for flap asymmetry to occur is during the landing pattern.

If asymmetric wing flap condition occurs:

1. Immediately return flap lever to previous position

WARNING

- DO NOT ATTEMPT TO RECYCLE FLAPS BECAUSE THE ASYMMETRIC CONDITION MAY INCREASE.
- IF FULL SURFACE TRAVEL HAS BEEN APPLIED WITH LIMITER ENGAGED IT WILL NOT BE POSSIBLE TO ATTAIN ADDITIONAL TRAVEL WHEN THE LIMITER IS DISENGAGED. THIS IS CAUSED BY THE CONTROL INPUT INTERFERING WITH THE LIMITER STOP WHICH WILL HOLD IT IN THE LIMITED POSITION. THEREFORE, IT WILL BE NECESSARY TO RELAX THE CONTROL INPUT TOWARD NEUTRAL SLIGHTLY SO THAT THE LIMITER WILL RETRACT.
- IF AT ANY TIME THE FLAPS SHOULD RETURN WITHIN ASYMMETRY LIMITS, THE FLAPS WILL TRAVEL TO THE POSITION SELECTED BY THE FLAP LEVER.
- THERE MIGHT BE SOME ASYMMETRIC CONDITIONS IN WHICH FULL AILERON TRAVEL MAY NOT BE SUFFICIENT TO RETAIN LATERAL CONTROL.

2. Climb to a safe altitude, evaluate lateral control and determine a safe landing speed

NOTE

- Jettison external stores if aileron control is marginal.
- With the limiters engaged, aileron travel is restricted to ±10° and rudder travel to ±6°. With the limiters disengaged aileron and rudder travel is ±20°.

3. Perform a STRAIGHT-IN approach with airspeed adjusted for wing flaps setting and gross weight

If at any time the aircraft becomes uncontrollable:

2. Eject

LANDING GEAR EMERGENCY EXTENSION

If gear indicators do not show gear down and locked after lever is DOWN and gear indicator lights test was positive:

1. LDG GEAR – Recycle once (260 KIAS maximum)

If gear still indicates unsafe:

2. LDG GEAR – DOWN
3. LANDING GEAR CONT circuit breaker – Pull and reset (after 10 seconds minimum)

If gear still indicates unsafe:

4. Maneuver – Pull g's and yaw/roll aircraft

If gear still indicates unsafe and it cannot visually be determined that three gears are clear of wheel wells:

2. LDG GEAR – UP

NOTE

Pulling MAN LDG GEAR handle with LDG GEAR lever DOWN will require a force exceeding 90 pounds.

3. MAN LDG GEAR handle – PULL

CAUTION

- GEAR CANNOT BE RETRACTED IN FLIGHT AFTER BEING LOWERED BY MEANS OF MANUAL LANDING GEAR RELEASE HANDLE.
- AIRSPEED MUST BE BELOW 225 KIAS BEFORE THE NOSE GEAR WILL LOCK DOWN. HOWEVER, IT IS RECOMMENDED THAT THE LOWEST PRACTICABLE AIRSPEED FOR CONFIGURATION AND GROSS WEIGHT BE USED TO REDUCE NOSE GEAR EXTENSION TIME.

4. LDG GEAR – DOWN
5. ANTI-SKID – OFF

CAUTION

- GEAR EXTENSION BY MEANS OF THE MANUAL LANDING GEAR HANDLE WILL RENDER THE ANTI-SKID AND POWER BRAKES INOPERATIVE. NOSEWHEEL STEERING WILL OPERATE IF HYDRAULIC PRESSURE IS NORMAL.
- IF THE PP2 BUS HAS FAILED, THE EMERGENCY INTERPHONE WILL BE DEENERGIZED WHEN GEAR IS EXTENDED.

If gear still indicates unsafe and it visually (another aircraft or tower) can be determined that all three gears are clear of their wheel wells and hydraulic No. 2 pressure is normal:

6. Land as soon as practicable using PRECAUTIONARY STRAIGHT-IN PATTERN procedure
7. Do not shut down engine before groundcrew has secured gear

If nose gear remains in wheel well, or is partially extended:

6. Refer to NOSE GEAR UP procedure

If one main gear remains up or in an intermediate position:

6. Refer to ONE MAIN GEAR UP OR UNLOCKED procedure

MAIN GEAR FLAT TIRE

1. Perform STRAIGHT-IN approach
2. Touch down on side of runway opposite flat tire. Keep load off flat tire as long as possible
3. Nosewheel – Lower
4. Nosewheel steering – Engage (as soon as possible)
5. Drag chute – Deploy

NOSE GEAR FLAT TIRE

1. Perform STRAIGHT-IN approach with minimum practical fuel load

After touchdown:

2. Nose gear – Hold off
3. Nose – Lower slowly (at minimum of 130 KIAS)

After nosewheel contacts runway:

4. Drag chute – Deploy
5. Engine – Shutdown at pilot's discretion (FOD), using either throttle or FUEL SHUT-OFF switch

NOSE GEAR UP

If required:

1. External stores — Jettison, retain empty tip and pylon tanks

If time and conditions permit:

2. Far-end hook type barrier — Request removal

WARNING

OPERATIONAL EXPERIENCE HAS DETERMINED THE RUNWAY FAR-END HOOK TYPE BARRIER MUST BE REMOVED WHEN LANDING WITH NOSE GEAR UP.

3. Fuel — Burn down to minimum practical load

Before landing:

4. Shoulder harness — LOCKED; seat belt and harness — TIGHT
5. Helmet visor — DOWN
6. Perform STRAIGHT-IN approach and landing

After landing:

7. Nose — Lower slowly at minimum of 130 KIAS (if necessary, light braking application can be made with nose gear held off)

After nose contacts runway:

8. Drag chute — Deploy
9. Engine — SHUTDOWN at pilot's discretion (FOD) using either throttle or FUEL SHUT-OFF switch
10. Apply brakes, using differential braking to maintain directional control
11. Perform GROUND ABANDONMENT procedure

ONE MAIN GEAR UP OR UNLOCKED

If one main gear remains up or in an intermediate position after all procedures to extend have failed:

1. Elect to eject or land
   (Decision to land should include consideration of availability of long, wide runway with adjoining unobstructed runout area, conditions of surface and area adjacent to runway and weather conditions. Also approach-end arrestment may be made with BAK-9/12 or the 44B-2C or the 44B-2D cable, refer to APPROACH-END ARRESTMENT procedure, MAXIMUM BARRIER ENGAGEMENT SPEEDS)

If decision is made to land and time and conditions permit:

2. External stores — Jettison
3. Fuel — Burn down to minimum practical load

Before landing:

4. Shoulder harness — LOCKED; seat belt and harness TIGHT
5. Helmet visor — DOWN
6. AIL RUD LIMITER switch — Lift cover UNLTD

NOTE

When performing a dual flight, the full unlimited aileron and rudder travel, may be obtained by pulling the RUD AIL LIMIT CONT circuit breaker located in the rear cockpit left console.

7. APC CUTOUT switch — OFF
8. Perform straight-in approach and landing, touch down on side of runway opposite to faulty gear
9. Throttle — IDLE

After nosewheel is on runway:

10. Nosewheel steering — Engage
11. Drag chute — Deploy
12. Keep load off faulty gear as long as possible
13. Brakes — As required, use nosewheel steering and brakes for directional control

CAUTION

- NOSEWHEEL STEERING AND POWER/ANTISKID BRAKES WILL NOT BE AVAILABLE, IF THE LEFT MAIN GEAR IS UP.

- GEAR EXTENSION BY MEANS OF MANUAL LANDING GEAR RELEASE HANDLE WILL RENDER ANTISKID AND POWER BRAKES INOPERATIVE. REQUIRED BRAKE PEDAL PRESSURE WILL BE GREATER THAN NORMAL FOR EFFECTIVE BRAKING.

After nosewheel steering is no longer effective:

14. Engine – SHUTDOWN at pilot's discretion (FOD) using either throttle or fuel shutoff switch
15. Perform GROUND ABANDONMENT procedure

Landing Gear Lever Physical Obstruction

If physical obstruction makes it impossible to place landing gear lever down:

1. DOWNLOCK MECH OVERRIDE and LDG GEAR uplock button – Press simultaneously

If still unable to place landing gear lever down:

2. MAN LDG GEAR – Pull
3. LANDING GEAR CONT circuit breaker – Pull and reset (after 10 seconds minimum)

POWER BRAKE/ANTISKID SYSTEM MALFUNCTION

The brake system will transfer automatically to the standby brake system if the power brake/antiskid system malfunctions. Under this condition the ANTI-SKID warning light will be illuminated when aircraft is on the ground and it may be necessary to pump the foot pedals to obtain adequate braking action; pedal forces will be heavier than normal.
There is another condition that can occur which will render the power brake/antiskid system inoperative and yet fail to transfer the system to the standby brakes. This resulting nobrake condition exists when the ground-air safety switch circuit malfunctions. The antiskid system relies on the ground-air safety switch signal indicating an on-the-ground condition before the brakes can be actuated. This touch-down safety feature prevents landing with the brakes applied.
If the ground-air safety switch circuit indicates an in-the-air condition when the aircraft is actually on the ground, antiskid brakes cannot be applied. Such a condition will be characterized by high brake pedal forces, a lack of deceleration, and possible loss of nosewheel steering. In order to regain braking action a manual transfer to the standby brake system must be made by moving the antiskid switch to OFF.

If ANTI-SKID warning light illuminates on ground:

1. ANTI-SKID – OFF
2. Brakes – Pump to obtain adequate braking

CAUTION

WITH INOPERATIVE POWER BRAKES, RELEASE PRESSURE ON BRAKE PEDALS PRIOR TO SWITCHING TO STANDBY BRAKES, OTHERWISE TIRES MAY BE BLOWN.

PRECAUTIONARY PARTIAL POWER PATTERN

This procedure is illustrated in Figure 3-9. The overhead pattern offers the most accurate control of the touchdown point and should be used when possible.
The pattern can be performed anytime when range and bearing data to touchdown is available (GCI, GCA, TACAN, F-15, DF, IN/CDU and HSI or visually). Exact distance to touchdown is vitally important and GCA should be used where available.

NOTE

- The airspeed listed is based on 16000 lbs gross weight and zero wind. Aircraft weight, field elevation and wind conditions must be considered to determine if airspeed and the pattern needs adjustment. Adjust approach and landing speeds 5 KIAS per 1000 lbs difference in gross weight or portions thereof. Refer to landing speed schedule in Appendix for other configuration and gross weights.

- This landing pattern may be used with an open nozzle and 100% RPM. Thrust with nozzle failed open is approx. 83%.

1. Throttle – 83% RPM (open nozzle 100% RPM)
2. LDG GEAR – UP
3. Wing flaps – TAKEOFF
4. Airspeed – 260 KIAS (adjust as necessary)
5. Speed brakes – As necessary to maintain 260 KIAS
6. Descent – Establish and maintain 1 to 2 ratio (1000 ft every 2 nm/1800 to 2000 fpm)
7. Entry altitude – 16000 ft AGL (overhead of touchdown point)

NOTE

The final approach may be intercepted at any point and from any direction, but not less than 2 nm from touchdown.

8. Fly a 30° banked turn to downwind key at 11500 ft AGL
9. Base leg key – 8000 to 9000 ft AGL, approx. 11 nm from touchdown
10. Fly a 30° banked turn to final key – 4000 ft AGL, 8 nm from touchdown
11. Descent – Check 1 to 2 ratio

If descent ratio cannot be maintained:

12. Eject

If descent ratio is maintained and landing is assured:

12. LDG GEAR – DOWN at pilot's discretion

PRECAUTIONARY STRAIGHT-IN PATTERN

NOTE

The airspeed listed is based on 16000 lbs gross weight and zero wind. Aircraft weight, field elevation and wind conditions must be considered to determine if airspeed and the pattern needs adjustment. Adjust approach and landing speeds 5 KIAS per 1000 lbs difference in gross weight or portions thereof. Refer to landing speed schedule in Appendix for other configurations and gross weights.

1. Wing flaps – TAKEOFF (in some cases LAND flaps could be used at pilot's discretion)
2. Airspeed – 20 KIAS above normal final approach speed for gross weight or as required for a specific condition
3. Speed brakes – As necessary
4. LDG GEAR – DOWN at pilot's discretion

PRECAUTIONARY HIGH RPM/THRUST PATTERN

This procedure is to be used for:

- Inadvertent T_2 reset
- Stuck throttle
- IGV closure
- A/B thrust requirement

If all efforts to adjust thrust fail or A/B thrust is required:

1. Fuel – Burn down to highest acceptable landing weight

WARNING

FUEL CONSUMPTION WITH THROTTLE IN AFTERBURNER MAY EXCEED 500 POUNDS PER MINUTE.

2. Speed – Reduce (using speed brakes, G-forces and climb, as necessary) to permit extension of flaps and landing gear

NOTE

With speed brakes out, flaps at TAKEOFF and gear down, speed may be reduced to 230 KIAS for LAND flap extension by pulling 2 to 3 G in a turn and/or climbing.

3. Establish a flat final approach. A flat final approach will result in a slower airspeed. With speed brakes, LAND flaps, and gear down at military power, 190 to 200 KIAS is typical (230 to 240 KIAS is typical with afterburner). Speed and glide slope may be controlled somewhat by modulation of speed brakes

At touchdown:

4. FUEL SHUT-OFF switch – OFF (IGV closure or A/B landing: Throttle – IDLE)

WARNING

ENGINE FLAMEOUT TIME MAY VARY FROM 1 TO 5 SECONDS DEPENDING ON AIRSPEED AND POWER SETTING.

NOTE

If fuel is shut off prior to touchdown, a right rolloff will be experienced as engine torque is lost, however this rolloff is easily controlled. Sufficient hydraulic pressure will be available for the flight controls to control touchdown.

5. Drag chute – Deploy, observing drag chute limits

NOTE

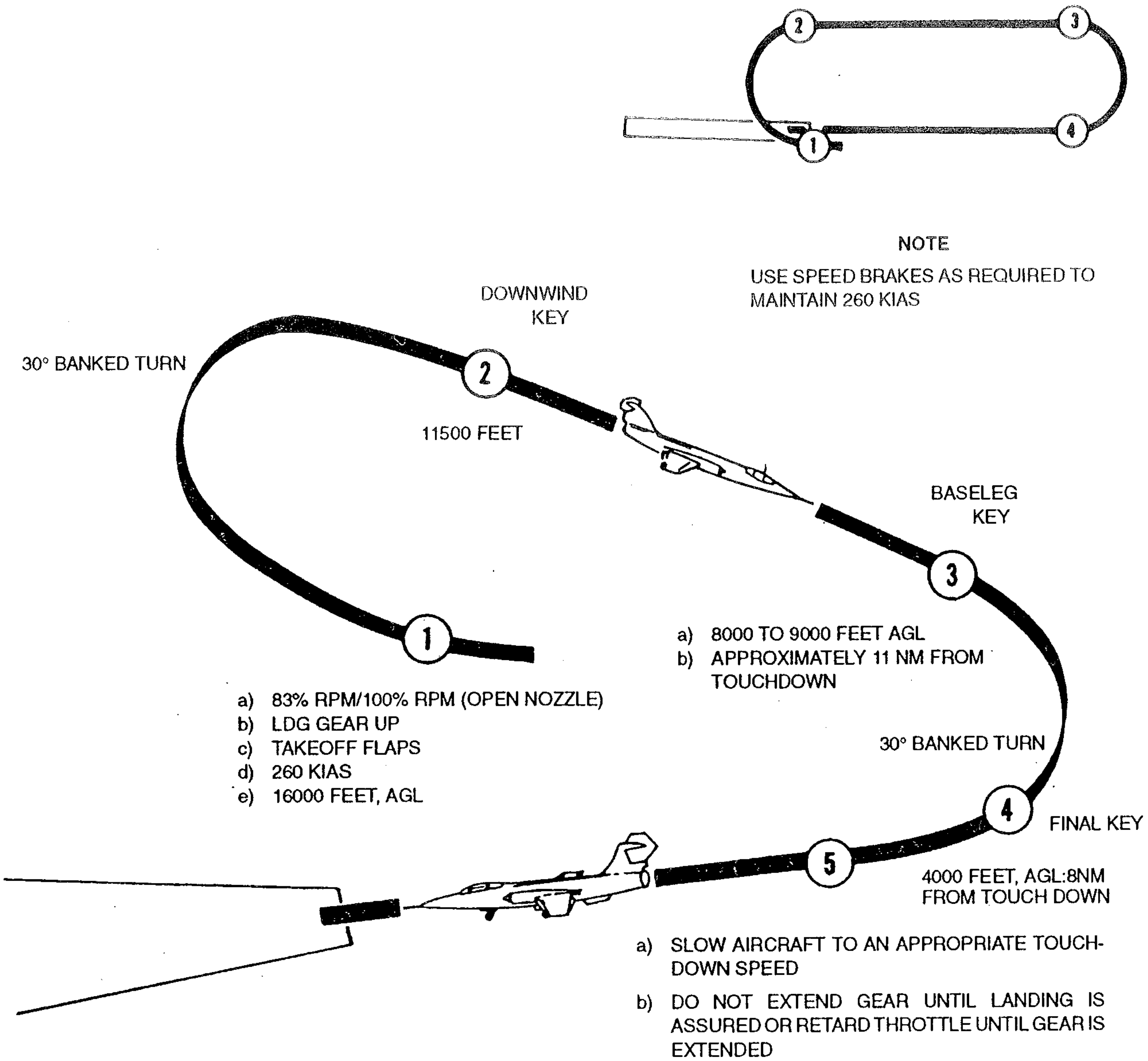
With FUEL SHUT-OFF switch OFF, nosewheel steering and power brakes are inoperative. Directional control must be maintained with use of standby brakes.

If barrier engagement is to be made:

6. HOOK RELEASE – Press. Refer to ABORT/ BARRIER ENGAGEMENT procedure

PRECAUTIONARY PARTIAL POWER PATTERN

MAY BE USED FOR GCI, GCA AND TACAN PENETRATIONS
GROSS WEIGHT 16000 LBS



NOTE

THIS LANDING PATTERN MAY BE USED WITH AN OPEN NOZZLE AND 100% RPM

FINAL APPROACH MAY BE INTERCEPTED AT ANY POINT AND FROM ANY DIRECTION, BUT NOT LESS THAN 2 NM FROM TOUCHDOWN

NOTE

ON GLIDE PATH, MAINTAIN A 1 TO 2 RATIO BETWEEN ALTITUDE AND DISTANCE FROM TOUCHDOWN. DECREASE ALTITUDE 1000 FEET FOR EACH 2 NAUTICAL MILES–TRAVELED

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Figure 3-9

SECTION IV

CREW DUTIES

NOT APPLICABLE.

SECTION V

OPERATING LIMITATIONS

TABLE OF CONTENTS

INTRODUCTION

This section includes the engine and aircraft limitations that must be observed during normal operation. Cognizance must be taken of the instrument markings (Figure 5-1) since they represent limitations that are not necessarily repeated in the text.
For instrument/indicator readings refer to front cockpit values.

ENGINE LIMITATIONS

Refer to the instrument markings illustration (Figure 5-1) and Maximum Allowable Airspeed paragraph, in this section.

THRUST DEFINITIONS AND TIME LIMITS

Military Thrust

Military thrust is obtained with a full (100% RPM) non-afterburning throttle setting. There are no time limits for inflight operation at this throttle setting.

Maximum Thrust

Maximum thrust is obtained with a full afterburning throttle setting and has no time limits for inflight operation.

NOTE

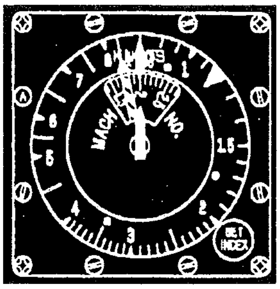
Ground operation is limited to 90 seconds at Military thrust or above but can be repeated after retarding the throttle to IDLE, then advancing to 80-82% RPM for a 2-minute cooling run.

AFTERBURNER SHUTDOWN LIMITATIONS

Gradual afterburner shutdown is required in certain areas of the aircraft flight envelope. It is intended to allow the aircraft to decelerate from high speeds to medium speeds before the engine exhaust nozzles close. This prevents the exhaust nozzles from becoming overpressurized due to peak pressure between maximum thrust and military thrust. Refer to Maximum Airspeed Limitations (Figure 5-4) for gradual afterburner shutdown speeds in relation to altitude.

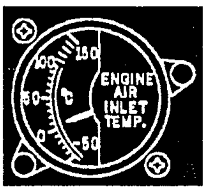
INSTRUMENT MARKINGS

AIRSPEED AND MACH NUMBER INDICATOR



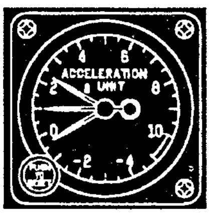
YELLOW 240 IAS, Maximum with flaps in LAND position

ENGINE AIR INLET TEMPERATURE



RED 100° C Maximum
GREEN –70° to 100° C operating range

ACCELEROMETER



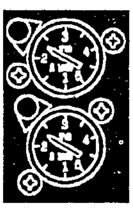
SYMMETRICAL MANEUVER LIMITS

+6.4G Maximum with no external stores and 4000 lb or less fuel remaining
–2.7G Maximum with no external stores and 4000 lb or less fuel remaining.

WARNING

Acceleration limits vary with airspeed and external stores configuration. Refer to maximum allowable acceleration and airspeed figures in this section.

HYDRAULIC SYSTEMS PRESSURE



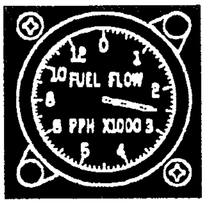
<table>
<tr><td rowspan="5">YELLOW</td><td rowspan="5">400–2800 psi</td><td>No. 1 System</td><td>No. 2 System</td></tr>
<tr><td colspan="2">Permissible with high flow demands on system</td></tr>
<tr><td></td><td>400–2175 psi Flight Controls have priority over utility system.</td></tr>
<tr><td colspan="2">Shows malfunction with no flow demands on system</td></tr>
<tr><td colspan="2"></td></tr>
<tr><td>GREEN</td><td>2800–3200 psi</td><td colspan="2">Normal</td></tr>
<tr><td rowspan="2">YELLOW</td><td rowspan="2">3200–3850 psi</td><td colspan="2">Permissible surge during rapid control surface movement</td></tr>
<tr><td colspan="2">Shows malfunction with control surface static</td></tr>
<tr><td>RED</td><td>3850 psi</td><td colspan="2">Maximum</td></tr>
</table>

FA0103

Figure 5-1 (Sheet 1 of 2)

INSTRUMENT MARKINGS

FUEL FLOW

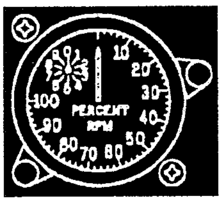


RED	400 lb/hr Minimum
GREEN	400–12000 lb/hr Normal operating range

NOTE

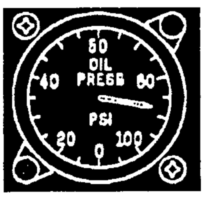
A fluctuation of ± 500 lb per hour about the mean value is permissible if an EGT fluctuation outside of limits does not accompany it.

TACHOMETER



RED	66% Minimum
GREEN	85–104.5% Normal operating range. 101% Maximum without T_2 reset. 104.5% Maximum with T_2 reset. 94% Minimum with T_2 cutback.
RED	107% Maximum permissible overspeed. 105–107% 5–second maximum

OIL PRESSURE

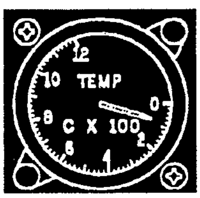


RED	Minimum IDLE (12 psi)
GREEN	Continuous operation (12 psi to placard psi +5 psi)
RED	100% RPM (Placard psi ± 5 psi)
YELLOW	Permissible range with T_2 reset (Placard psi +11 psi –5 psi)
RED	Maximum with full T_2 reset (Placard psi +11 psi).

NOTE

- The markings on the gage shown are for the purpose of illustration only. Oil pressure gage markings vary with each engine/airframe installation. Refer to engine oil pressure write up in section I.
- There is no maximum allowable limit for oil pressure variations during high "G" flight maneuvers.
- Placard psi is the pressure shown on the oil pressure record card.

EXHAUST GAS TEMPERATURE

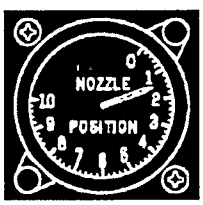


RED	50° C Minimum
GREEN	50° – 600°C Steady–state operation
RED	600° C Maximum steady–state
YELLOW	600° – 1000° C Time limited
RED	1000° C Maximum allowable overtemperature (requires turbine replacement if exceeded).

NOTE

- Operation in excess of the time limit will reduce engine life.
- Transient temperatures may peak to 760° C if the EGT pointer is in constant motion.

NOZZLE POSITION INDICATOR



RPM	NOZZLE INDICATOR
Idle	8.5 to 9.5
Military	1.0 to 3.5
Maximum A/B	7.5 to 9.5
Minimun Sector A/B	3.5 to 5
A/B Blowout	Less than 7
Nozzle Handle Pulled	1.0 to 3.0

NOTE

- It is acceptable for the nozzle position indicator to read 10 or above provided EGT remains within limits.
- Nozzle position indicator fluctuations up to one pointer width are normal.

FA0104

Figure 5-1 (Sheet 2 of 2)

ENGINE AIR INLET TEMPERATURE

The engine air inlet temperature limit is a function of outside air temperature and flight Mach number. Therefore, the speed at which the limit temperature is reached varies with altitude, and from day to day at a given altitude.
Note on Figure 5-4 that this limit (represented by dotted black lines) is encountered at Mach 1.92 at high altitude when the ambient temperature is 10° C higher than standard. Consequently under these conditions this limit rather than the limit Mach number restricts the maximum permissible speed of the aircraft.

T_2 RESET LIMITATION

Due to temperature lag and/or sensor location T_2 reset can start as low as 70° C. After reset the RPM should be 103.5% (±1).

ENGINE OIL LEVEL LOW WARNING LIGHT

The nozzle hydraulic pump and/or nozzle actuation system will suffer damage from insufficient oil. An entry in the aircraft form must be made reporting all occurrences of warning light illumination of 1 second or longer and the approximate duration.

ENGINE ANTI-ICING OPERATION LIMITS

The engine anti-icing system may be operated when the indicated CIT is 10° C or less and at any speed up to 350 KIAS or Mach 1.0, whichever is lower. After flying in moderate to heavy icing for 2 minutes or more, reduce thrust (where practical) to 88% to minimize inlet duct ice ingestion damage to the engine.
Should it be necessary to fly in known icing conditions at low altitude and at low thrust settings (80 to 86% RPM), the engine power should be increased to 100% RPM every 5 minutes to ensure that adequate anti-icing air circulation is available at the engine compressor front frame. This thrust increase should be maintained for approximately 30 seconds.

GROUND STARTING LIMITS

Maximum temperature during engine start is 705° C (Figure 5-2).

CAUTION

WHENEVER THE MAXIMUM TEMPERATURE IS EXCEEDED, ABORT. ENTER MAXIMUM TEMPERATURE REACHED AND DURATION IN AFTO FORM 781.

EGT LIMITS

For EGT limits, except ground starting, refer to Figure 5-3.

NOTE

If EGT tends to exceed limits, when operating at high Mach number, retard throttle to reduce EGT within limits. Throttle retarding will not limit performance because in this condition engine provides excees thrust.

STARTER LIMITATIONS

Starter limitations are as follows:

1 minute continuous operation
3 minutes cooling period
1 minute continuous operation
10 minutes cooling period

ROLL STABILITY AUGMENTER LIMITATIONS

The roll stability augmenter should be turned off before reaching 575 KIAS with any wing store installed.
With the above stores installed and the roll stability augmenter operating, wing torsion oscillations sufficient to cause structural damage may be experienced at high indicated airspeeds.

EGT GROUND STARTING LIMITS

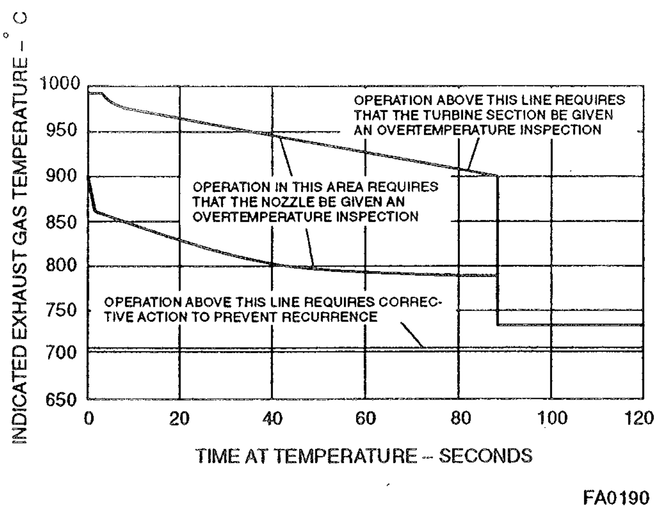


Figure 5-2

EGT LIMITS

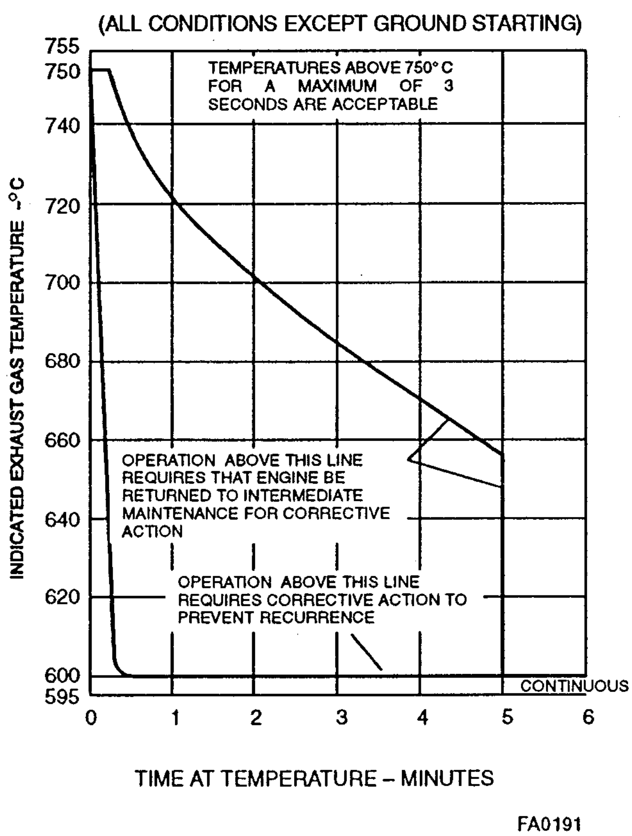


Figure 5-3

AIRSPEED LIMITATIONS

Landing gear transient operation	260 KIAS
Landing gear emergency operation	225 KIAS

WARNING

AIRSPEED FOR EMERGENCY GEAR OPERATION MUST BE BELOW 225 KIAS OR THE NOSE GEAR WILL NOT LOCK DOWN.

CAUTION

IF THE LIMITS FOR LANDING GEAR EXTENSION ARE EXCEEDED, RUPTURE OF THE MAIN LANDING GEAR DOOR ACTUATING CYLINDER AND SUBSEQUENT COMPLETE LOSS OF HYDRAULIC FLUID AND LOW PRESSURE OF NO. 2 HYDRAULIC SYSTEM MAY OCCUR.

Landing gear down and locked — 295 KIAS

Wing flaps:

TAKE OFF

- During extension — 450 KIAS or MACH 0.8 (There is no Mach limitation if 330 KIAS is not exceeded)
- With flap extended or during retraction — 450 KIAS or MACH 0.8 (There is no Mach limitation if 360 KIAS is not exceeded)

LAND — 240 KIAS

Drag chute operation, Irving VPCR — 200 KIAS

MAXIMUM AIRSPEED LIMITATIONS

NO EXTERNAL STORES

SEE FIGURE 5-7 FOR AIRSPEED LIMITS AND OTHER CONFIGURATIONS

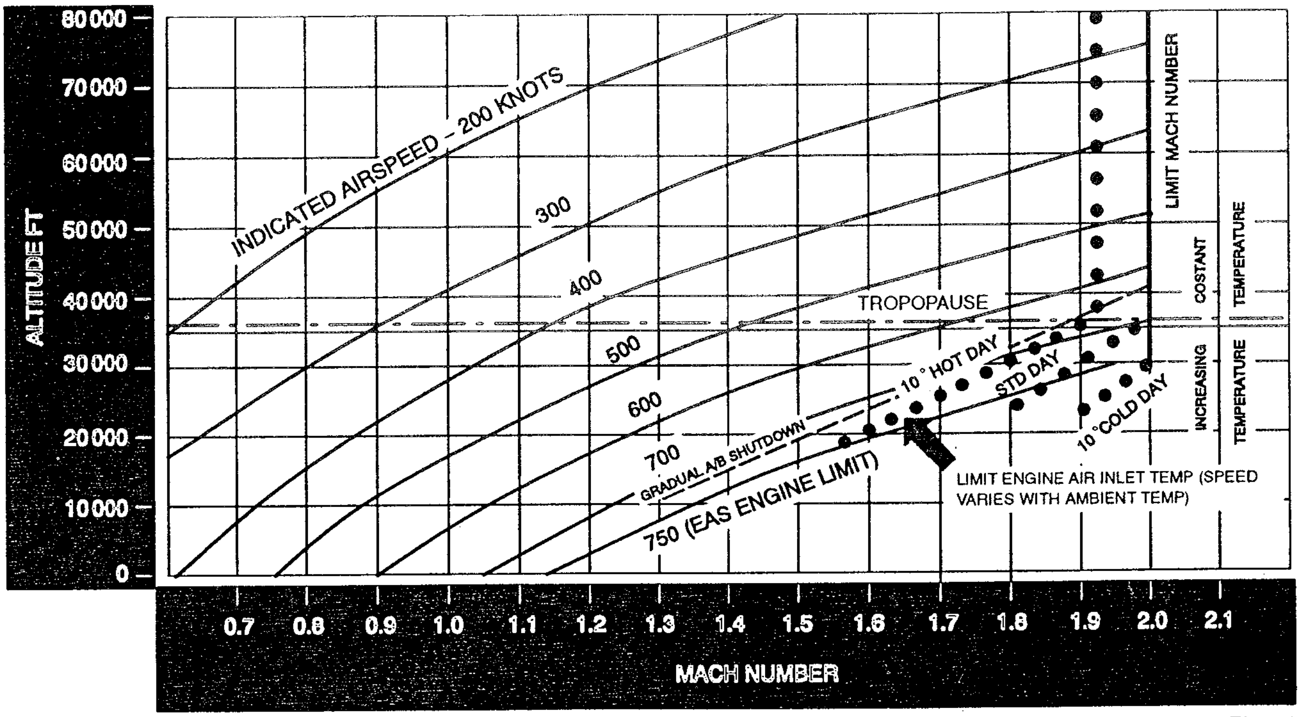


Figure 5-4

NOTE

Operational experience has shown that the maximum recommended airspeed for drag chute deployment is 180 KIAS.

Windshield rain remover operation 295 KIAS

CAUTION

- DO NOT UNLOCK THE CANOPY IN FLIGHT.
- ALTHOUGH AIRCRAFT MAY BE TAXIED WITH CANOPY IN FULL OPEN POSITION, CARE MUST BE EXERCISED TO AVOID FAST TAXI OVER BUMPY STRIPS BECAUSE HIGH VERTICAL LOADS MAY RESULT WHICH CAN DAMAGE THE CANOPY MECHANISM. IN STRONG CROSSWINDS IT IS POSSIBLE FOR THE CANOPY TO SLAM SHUT.

RAT EXTENSION LIMITS

The RAT can be extended in level flight without affecting engine operation within the following airspeed limits:

Altitude	Airspeed Limits (KIAS)	
	Minimum	Maximum
Up to 30000 feet	None	550
Above 30000 feet	350	

MAXIMUM ALLOWABLE AIRSPEED

The maximum allowable airspeed for any external stores configuration is presented in Figure 5-7. Also the limits without external stores, with flaps up, are shown in Figure 5-5.

EXTERNAL STORES JETTISON LIMITS

See Figure 5-6 for the external stores jettison limits.

PROHIBITED MANEUVERS

PITCHUP AND SPINS

Intentional pitchup and spins are prohibited because of the high loads they impose on the aircraft. These loads can be of sufficient magnitude to cause structural damage.

CAUTION

EXERCISE EXTREME CARE TO AVOID ABRUPT MANEUVERS OR LOW INDICATED AIRSPEEDS WHEN THE APC WARNING LIGHT IS ON OR WHEN THERE ARE OTHER INDICATIONS THAT THE APC SYSTEM IS INOPERATIVE.

RESTRICTED MANEUVERS

NEGATIVE "G" FLIGHT

Due to limited oil distribution to the variable exhaust nozzle closure system, engine oil system and fuel starvation during prolonged negative-"g"/inverted flight, the aircraft is limited to:

a. 20 seconds of negative "G" flight

b. 10 seconds of zero "G" flight

WARNING

- ILLUMINATION OF THE "FUEL BOOST PUMP FAIL LIGHT" DURING NEGATIVE "G"/INVERTED FLIGHT, INDICATES THAT FUEL SUPPLY TO THE BOOST PUMPS IS INTERRUPTED AND AN ENGINE FLAME-OUT IS LIKELY TO OCCUR WITHIN A VERY SHORT TIME. IMMEDIATELY RECOVER TO POSITIVE "G" CONDITIONS AND DECREASE THE FUEL FLOW, IF POSSIBLE, IF FLAME-OUT OCCURS, THE TIME REQUIRED TO RECOVER ENGINE FUEL FLOW WILL CAUSE A DELAY OF ENGINE RELIGHT.

- DURING NEGATIVE "G"/INVERTED FLIGHT WITH LESS THAN FULL INTERNAL FUEL AND FUEL FLOW HIGHER THAN 6500 PPH, A FLAME-OUT MAY OCCUR IN LESS THAN 10 SECONDS.

OPERATING FLIGHT LIMITS

FOR SYMMETRICAL FLIGHT IN SMOOTH AIR – GEAR AND FLAPS UP
NO EXTERNAL LOAD WITH LESS THAN 4000 LB FUEL REMAINING

REFER TO FIGURE 5-7 FOR ACCELERATION LIMITS AT OTHER LOADING

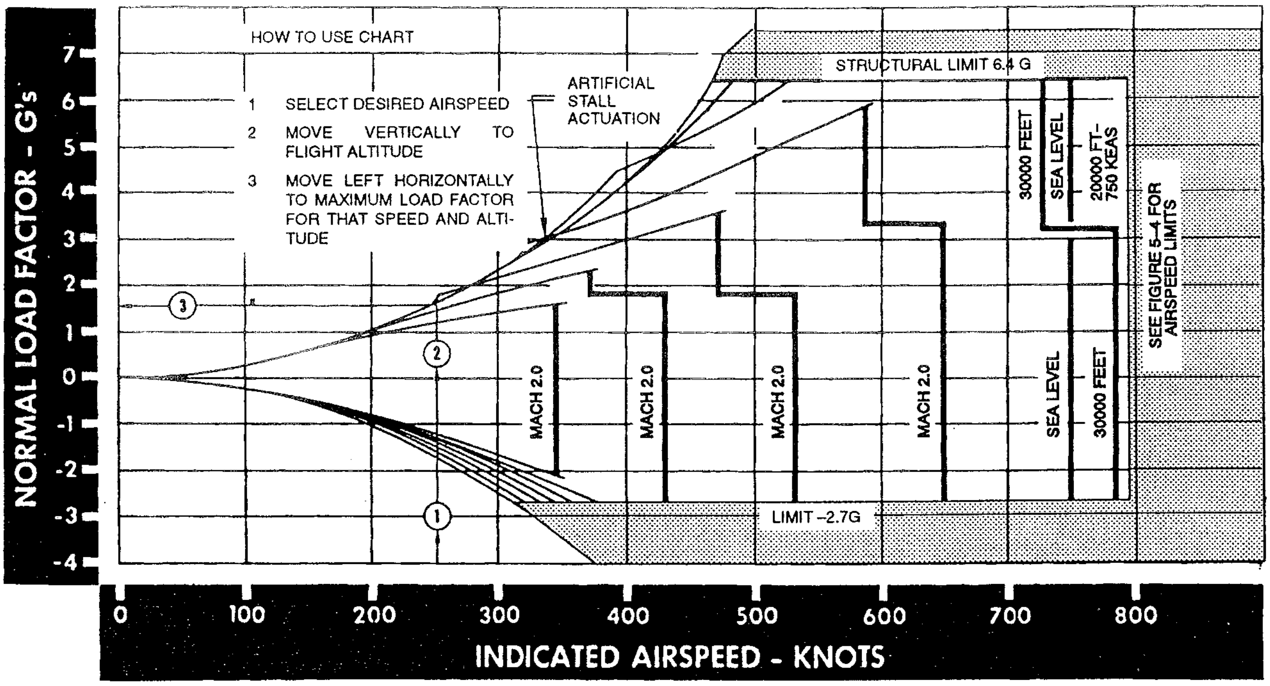


Figure 5-5

FA0192

EXTERNAL STORES JETTISON LIMITS

STORE	STATION	LIMIT
FUEL TANKS	WING PYLON WING TIP	MACH 1.5 MACH 0.9 NOTE IN AN EMERGENCY, EMPTY TIP TANKS MAY BE JETTISONED AT SUPERSONIC SPEEDS. IF POSSIBLE HOWEVER, JETTISON TANKS AT LESS THAN MACH 1.5.
WING PYLONS	WING PYLON	MACH 0.9
NOTE WHEN POSSIBLE, EXTERNAL STORES SHOULD BE JETTISONED IN 1 G FLIGHT.		

Figure 5-6

- DURING NEGATIVE "G"/INVERTED FLIGHT, A COMPLETE LOSS OF OIL CAN BE EXPERIENCED IN 30 SECONDS OR LESS AND RESULT IN NOZZLE OPEN FAILURE AND/OR ENGINE OIL STARVATION. IF NEGATIVE "G"/INVERTED FLIGHT IS EXPERIENCED, RECOVER TO POSITIVE "G", INCREASE PITCH 5° TO 10° AND/OR ACCOMPLISH A DECELERATION MANEUVER. MONITOR THE "OIL LOW LEVEL" WARNING LIGHT AND NOZZLE SYSTEM FOR INDICATION OF OIL LOSS AND/OR NOZZLE OPEN FAILURE.

AILERON ROLL LIMITATIONS

To avoid inertial coupling and high structural loads approaching limit values, aileron rolls are subject to the following restrictions.

WING FLAPS RETRACTED

Entry Load Factor of 0.5 "G" and Greater

Full deflection rolls are limited to 360°. Below 1 "G" with pitch or yaw stability augmenters inoperative, full deflection 360° rolls are prohibited.

Entry Load Factor Less Than 0.5 "G"

Full deflection 360° rolls are prohibited. All rolls below 0.5 "G" load factor must be executed with extreme caution.

TAKEOFF (MANEUVERING) FLAPS EXTENDED

1. Rolls are limited to 360°. High roll rates can develop from moderate aileron displacement
2. Either the pitch or yaw stability augmenter must be operative for all rolling maneuvers
3. Aileron rolls are prohibited for entry load factors of less than 1 "G"

NOTE

Application of some back stick pressure during rolls helps to terminate the roll more rapidly and make a smoother transition to normal flight.

ACCELERATION LIMITATIONS

Maximum allowable acceleration limits for any external configurations are presented in Figure 5-7.

WARNING

THE AIRCRAFT IS RESTRICTED TO 2.0 "G" ABOVE MACH 1.6 WHEN THE YAW STABILITY AUGMENTER IS IN OFF OR INOPERATIVE.

CAUTION

THE INSTRUMENT MARKINGS OF FIGURE 5-1 ARE THE MAXIMUM LOAD FACTOR THAT MAY BE UTILIZED WITHOUT EXTERNAL STORES AND 4000 POUNDS OF FUEL OR LESS. THEREFORE COGNIZANCE MUST BE TAKEN OF THE LIMITS LISTED IN FIGURE 5-7 WHEN OPERATING WITH EXTERNAL STORES OR HIGHER FUEL LOADINGS.

"G"-limit for symmetrical maneuvering such as straight pullups "G"-limits for unsymmetrical maneuvering such as rolling pullouts or rolling pushovers are shown in Figure 5-7.

In rolling pullouts or rolling pushovers, lower "G"-limits must be observed because of the higher structural loads that are imposed on the aircraft.

MANEUVERING BOUNDARIES

An operating flight limits diagram is shown in Figure 5-5. This diagram shows the maximum maneuvering load factors over the operating range of speeds and altitudes of the aircraft. It includes the load factor and airspeed combination at which the stall occurs (refer to Stall, in Section VI) as well as the maximum allowable structural "G" limits for the no-external-stores configuration.
Also it will be noted in the diagram that for altitudes of 30000 feet and above, bends in the vertical lines represent additional limits to prevent flight into areas where the vertical tail structure may be overloaded.
Use of the diagram is illustrated. Since the maximum allowable airspeed depends on different limitations at various altitudes, the diagram for any particular altitude is cut off at the airspeed at which the applicable limit is encountered.
For example, this is indicated at the higher altitudes shown on the figure by the vertical lines labeled Mach 2.0.
The diagram does not include the engine air inlet temperature airspeed limitation; therefore, a reference to the airspeed limitations of Figure 5-4, in conjunction with this diagram, is necessary for establishing the complete limits.
Use of Figure 5-4 will also permit determination of the corresponding Mach number for any airspeed and altitude combination if the maximum load factor for a particular Mach number is desired.

CENTER-OF-GRAVITY LIMITATIONS

With any combination of standard external load items the center-of-gravity remains within limits.

CAUTION

WING-TIP STORES APPROVED FOR USE ON THIS AIRCRAFT MEET CERTAIN REQUIREMENTS AS TO BALANCE AND AERODYNAMIC STABILITY. PILOTS SHOULD MAKE CERTAIN THAT STANDARD WING-TIP STORES ARE USED OR THAT REPLACEMENT COMPONENTS CONFORM TO THE STANDARD AERODYNAMIC CONFIGURATION AND CENTER-OF-GRAVITY.

CENTER-OF-GRAVITY EXCURSIONS

The following CG position excursions, in percent of MAC (Mean Aerodynamic Chord), are applicable to all operative configurations, both for landing gear UP and DOWN and to any wing flaps configurations:

Max aft 16.5%

Max fwd 4%

WEIGHT LIMITATIONS

The maximum allowable gross weights are:

26000 pounds for ground operation

21500 pounds for landing

SINK RATE LIMITATIONS

With maximum landing weight (21500 lbs) do not exceed 300 ft/min sinkrate during touchdown.

MAXIMUM CROSSWIND COMPONENT LIMITATION

Maximum crosswind component limit is 25 knots (see Figure A8-1).

MAXIMUM ALLOWABLE ACCELERATION AND AIRSPEED LIMITS

AIRCRAFT CONFIGURATION		ACCELERATION LIMITS (G)										AIRSPEED LIMITS
		SYMMETRICAL MANEUVERS				UNSYMMETRICAL (ROLLING) MANEUVERS						OBSERVE EAS, IAS, MACH NUMBER, ENGINE AIR INLET TEMPERATURE LIMIT OR SLOW LIGHT, WHICHEVER OCCURS FIRST
WING TIP STORES	PYLON STORES	MACH NUMBER	ALTITUDE (FEET)	TAKEOFF WEIGHT **	COMBAT WEIGHT ***	MACH NUMBER	ALTITUDE (FEET)	TAKEOFF WEIGHT **		COMBAT WEIGHT ***		
NONE	NONE	BELOW 1.8	ALL	+6.2 −2.6	+6.4* −2.7	BELOW 1.6	ALL	+4.1		+4.2*		750 KEAS OR MACH 2.0
		1.8 TO 2.0	UP TO 40000	+3.2 −2.6	+3.2 −2.7	1.6 TO 2.0	UP TO 40000	+3.2	(1)	+3.2	(1)	
			ABOVE 40000	+1.9 −2.6	+1.9 −2.7		ABOVE 40000	+1.9		+1.9		
TANKS	NONE	BELOW 1.6	ALL	+5.0 −2.0	+5.0 −2.0	BELOW 1.6	ALL	+3.3		+3.3		750 KEAS OR MACH 1.8
		1.6 TO 1.8	UP TO 40000	+2.7 −2.0	+2.7 −2.0	1.6 TO 1.8	UP TO 40000	+2.7		+2.7		
			40000 TO 48000	+2.0 −2.0	+2.0 −2.0		40000 TO 48000	+2.0	(1)	+2.0	(1)	
			ABOVE 48000	+1.5 −2.0	+1.5 −2.0		ABOVE 48000	+1.5		+1.5		
NONE	TANKS	TO 1.5	ALL	+5.0 −2.0	+5.0 −2.0	TO 1.5	ALL	+3.3	(1)	+3.3	(1)	750 KEAS OR MACH 1.5
TANKS												
ANY OF THE ABOVE CONFIGURATIONS WITH TAKEOFF FLAPS	DURING FLAP EXTENSION	– FOR POSITIVE ACCELERATION LIMITS SEE APPLICABLE EXTERNAL STORES CONFIGURATIONS ABOVE – THE NEGATIVE ACCELERATION LIMIT IS 1.0 G										450 KIAS OR MACH 0.80. THERE IS NO MACH LIMITATION IF 330 KIAS IS NOT EXCEEDED
	FLAPS EXTENDED OR RETRACTING											450 KIAS OR MACH 0.80. THERE IS NO MACH LIMITATION IF 360 KIAS IS NOT EXCEEDED

Figure 5-7 (Sheet 1 of 2)

MAXIMUM ALLOWABLE ACCELERATION AND AIRSPEED LIMITS

* ON AIRCRAFT POST MOD T.O.1F-104-2030 WITH 2000 LB OR LESS INTERNAL FUEL REMAINING THE MAXIMUM ACCELERATION LIMITS ARE:

SYMMETRICAL MANEUVERS = +7.33 G; −3.0 G

UNSYMMETRICAL MANEUVERS = +4.5 G

** TAKEOFF WEIGHT IS DEFINED AS MORE THAN 4000 LB INTERNAL FUEL OR FUEL IN EXTERNAL TANKS (WHEN INSTALLED)

*** COMBAT WEIGHT IS DEFINED AS 4000 LB OR LESS INTERNAL FUEL AND EMPTY EXTERNAL TANKS (WHEN INSTALLED)

(1) FOR ROLLING PUSHOVERS SEE AILERON ROLL LIMITATIONS

NOTE

1. THE MINIMUM SYMMETRICAL MANEUVER LIMIT IS 0.0 G WITH LAND FLAPS EXTENDED
2. REFER TO EXTERNAL STORES JETTISON LIMITS FOR MAXIMUM ALLOWABLE AIRSPEED AND ACCELERATION LIMITS AT WHICH EXTERNAL STORES MAY BE RELEASED
3. UNLESS OTHERWISE NOTED, LIMITS ARE BASED ON FLIGHT TESTS AND STATIC LOAD TESTS. VALUES SHOWN ARE MAXIMUM PERMISSIBLE
4. BARE WING PYLONS DO NOT AFFECT LIMITS; OBSERVE LIMITS FOR OTHER APPLICABLE STORES
5. REFER TO SECTION VI (FLIGHT CHARACTERISTICS) FOR LATERAL CONTROL REQUIREMENTS IN ASYMMETRIC CONFIGURATIONS
6. ACCELERATION THROUGH MACH 1.6 SHOULD BE DISCONTINUED IF YAW OSCILLATIONS WITH BALL DISPLACEMENT OF MORE THAN ONE-HALF BALL OCCURS.

WARNING

DO NOT EXCEED 500 KIAS WITH EMPTY TIP TANKS AND MORE THAN RESIDUAL FUEL IN PYLON TANKS.

Figure 5-7 (Sheet 2 of 2)

SECTION VI

FLIGHT CHARACTERISTICS

TABLE OF CONTENTS

INTRODUCTION

This section presents a description of the flight characteristics and performance capabilities of the aircraft. The operational speed and altitude capabilities include level-flight and climb speed of Mach 2.0. The aircraft is capable of attaining altitudes in excess of 90000 feet. Flight characteristics and handling qualities are excellent throughout this large range of operating speeds and altitudes. Low-speed flight characteristics are conventional and transition from subsonic to supersonic speed is made with negligible trim changes.

MACH NUMBER

Except for possibly the low-speed stall, the flight characteristics are generally a function of Mach number rather than indicated airspeed. Flight characteristics are therefore more easily associated with Mach number than IAS. Increasing altitude at a constant indicated airspeed result in increasing Mach number; therefore, for each altitude there is a different indicated airspeed for the same Mach number. The effect of airspeed at a given Mach number is simply to vary the magnitude of a particular flight characteristic. At high indicated airspeeds, a given flight characteristic generally is more pronounced. For these reasons, reference to flight speed generally will be made in terms of Mach number rather than airspeed. The airspeed for a desired Mach number at any altitude may be determined by referring to Figure 5-4. Enter the bottom of the chart with the Mach number, move vertically to the flight altitude (using the scale on the left), and read IAS from the curved airspeed lines. Interpolate as necessary between lines.

AIRSPEED AND ALTITUDE ERRORS

The airspeed system was designed and developed to minimize the altimeter error at low altitude throughout the airspeed range. In-flight calibration of this system established that a small error exists, being affected by aircraft attitude at subsonic speeds. The error is small enough to be disregarded during takeoff and landing and normal flight at low altitudes. In 1 "G" level flight at cruise speed the altimeter indicates an altitude higher than the true pressure altitude. On entering a level turn at low altitude, the altimeter will indicate a loss of altitude of 200 to 300 feet, depending on the aircraft load factor used. In level turns at high altitude during supersonic flight, the altimeter will also indicate a loss in altitude of about 300 feet. In order to provide altitude corrections ensuring terrain clearance during low-altitude operation, calibrations are presented in the Appendix, Part. 1.

AIRCRAFT CONFIGURATION

The overall configuration of the aircraft was chosen with an emphasis on high Mach number flight while maintaining conventional landing and takeoff characteristics. Since the appearance of this aircraft is somewhat unconventional, a brief discussion of some of the aerodynamic aspects of its configuration is given before describing the flight characteristics.

NEGATIVE DIHEDRAL

With the exception of the short span of the thin, straight wing, the configuration feature arousing the most interest on the aircraft is the 10° negative wing dihedral. The negative dihedral actually resulted from the solution to the problem of providing satisfactory longitudinal flight characteristics over a wide speed range. The empennage arrangement will be discussed first and then the negative wing dihedral.

EMPENNAGE

The empennage of the aircraft, aside from its very high effectiveness, is conventional in most respects. The unconventional aspect of the empennage is the location of the horizontal stabilizer on top of the vertical fin. This location of the horizontal stabilizer was determined from extensive wind tunnel tests of the tail located in many positions from below the fuselage to its present high position. These tests showed that the high position gave the best stability and control characteristics about the pitch axis over the wide operating range of the aircraft. This tail configuration gives a minimum of transonic trim changes and provides high stabilizer effectiveness throughout the speed range. In addition, the high location provides the minimum drag at supersonic speeds.

DIHEDRAL EFFECT

The position of the horizontal stabilizer atop the fin makes it act as an end plate to the vertical fin. The effective aspect ratio of the vertical fin is thus greatly increased, raising the center of pressure of the side load on the fin higher than would be the case for a low horizontal stabilizer position. This high-side center-of-pressure location on the fin results in a relatively large rolling moment in a sideslip condition. Comparison of the vertical fin height to the wing semi-span shows that the fin is almost as important in producing roll as the wing. Thus, the high center of pressure resulting from the horizontal stabilizer location provides the equivalent dihedral effect of 15° to 20° of positive wing dihedral angle. Negative dihedral of the wing is then introduced to reduce the net positive dihedral effect to that equivalent to 5° to 10° of dihedral. It is to be emphasized that despite the negative wing dihedral, the aircraft possesses a normal positive dihedral effect, as the pilot will immediately detect from the position of the stick in maintaining a steady sideslip at low speeds.

WING

The wing is of basically straight planform in order to minimize drag at very high Mach numbers. The ailerons are conventional outboard ailerons. The flaps comprise leading-edge and trailing-edge flaps. The full-span leading edge flap is deflected for landing and takeoff in order to delay flow separation over the sharp leading edge at the higher angles of attack. The inboard trailing edge flaps incorporate boundary layer control. The boundary layer control system directs high-energy air over the trailing edge flaps in the LAND position, thereby delaying airflow separation on the flap. This permits the use of larger flap deflections than normally would be possible, with a resulting increase in lift at high angles of attack. The combination of leading and trailing edge flaps results in normal approach and landing touchdown speeds, while maintaining the capability of very high speed flight with the basic wing.

MANEUVERING FLAPS

Maneuvering capabilities of the aircraft may be improved by the use of "maneuvering flaps". "Maneuvering flaps" are the wing flaps lowered to TAKE-OFF position with the gear retracted. This permits an increase of approximately 1 "G" in the available load factor over that with the aircraft flaps up, thereby improving the turn radius of the aircraft.

WARNING

THE ANGLE AT WHICH THE ANGLE-OF-ATTACK SENSING VANES ENERGIZE THE SHAKER OR KICKER IS DEPENDENT UPON THE FLAP LEVER SETTING AND NOT THE ACTUAL POSITION OF THE FLAPS. IN NORMAL OPERATING CONDITION AFTER FLAP LEVER HAS BEEN SELECTED FROM UP TO TAKEOFF (APC BOUNDARY VARIATION) AND BEFORE SAFE TAKEOFF INDICATION IS ACHIEVED A TIME INTERVAL OF 7/8 SECONDS EXISTS. THEREFORE INCREASING THE LOAD FACTOR (AOA) DURING THIS TRANSIENT CAN CAUSE A PITCH-UP WITH NO PREVIOUS SHAKER OR KICKER OPERATION.

FLIGHT CONTROLS

STABILIZER

The fully powered horizontal stabilizer is an extremely powerful longitudinal control and provides excellent maneuvering characteristics at all flight speeds. Due to the high response rates possible with this type of control, use caution in rapid maneuvering, especially at high indicated subsonic airspeeds, until you are familiar with its effectiveness. The artificial feel system provides satisfactory stick forces under all conditions with good centering and excellent incremental control qualities.

AILERONS

The ailerons are fully powered, are capable of producing extremely high roll rates, and have stick forces supplied by feel springs. The action of the feel springs results in essentially constant stick forces for any amount of deflection, regardless of airspeed. Aileron travel is approximately $\pm 10°$ of travel with gear up to reduce effectiveness and avoid inertial coupling tendencies in rolls over the wide operating limits of the aircraft.

When the gear is down the available aileron travel is increased to $\pm 20°$ to permit more effective control such as is necessary during approach and landing. Aileron travel is also unlimited when the gear is up and a trailing edge flaps asymmetry, as detected by the flap asymmetry detector system, occurs.
When the ailerons are unlimited, the AIL AND RUD UNLIMITED warning light, located on the left part of the upper main instrument panel, will illuminate.

RUDDER

The rudder is a fully powered irreversible control surface. Feel forces are provided by springs up to the power limit of the system. In order to maintain tail loads below structural values, the rudder incorporates two limits.
With the gear extended, rudder travel is $\pm 20°$ (± 2). On retracting the gear, the rudder travel is automatically limited to $\pm 6.0°$ from trim position. Directional trim control is provided with the rudder surface with $\pm 4.0°$ deflection for trimming. Rudder travel is also unlimited when the gear is up and a trailing edge flaps asymmetry, as detected by the flap asymmetry detector system, occurs.
When the rudder is unlimited, the AIL AND RUD UNLIMITED warning light, located on the left part of the upper main instrument panel, will illuminate.

STABILITY AUGMENTERS

The handling characteristics and dynamic response of the aircraft about all three aircraft axes are greatly improved through the use of stability augmenters. The yaw and roll augmenters provide effective damping of "dutch-roll" motion and the pitch augmenter provides longitudinal damping, resulting in a steady and effective weapon platform.
To obtain optimum handling characteristics, the yaw and pitch augmenters should be in use at all times and the roll augmenter should be on at all times except as limited by installed tip stores (refer to Section V). Figure 6-1 and Figure 6-2 graphically illustrate the effect of these augmenters, showing typical characteristics with and without the augmenters in operation.
Should failure occur in the yaw augmenter system, the surface will remain stationary or drift slowly to any position within its control range and yaw damping will be noticeably reduced. Failure of the roll augmenter system does not reduce the effectiveness of the aircraft but results in more sensitivity to roll disturbances in turbulent air. Flight may be continued, and in smooth air little or no effect will

EFFECT OF STABILITY AUGMENTER ON ROLL AND YAW DYNAMIC FLIGHT CHARACTERISTICS

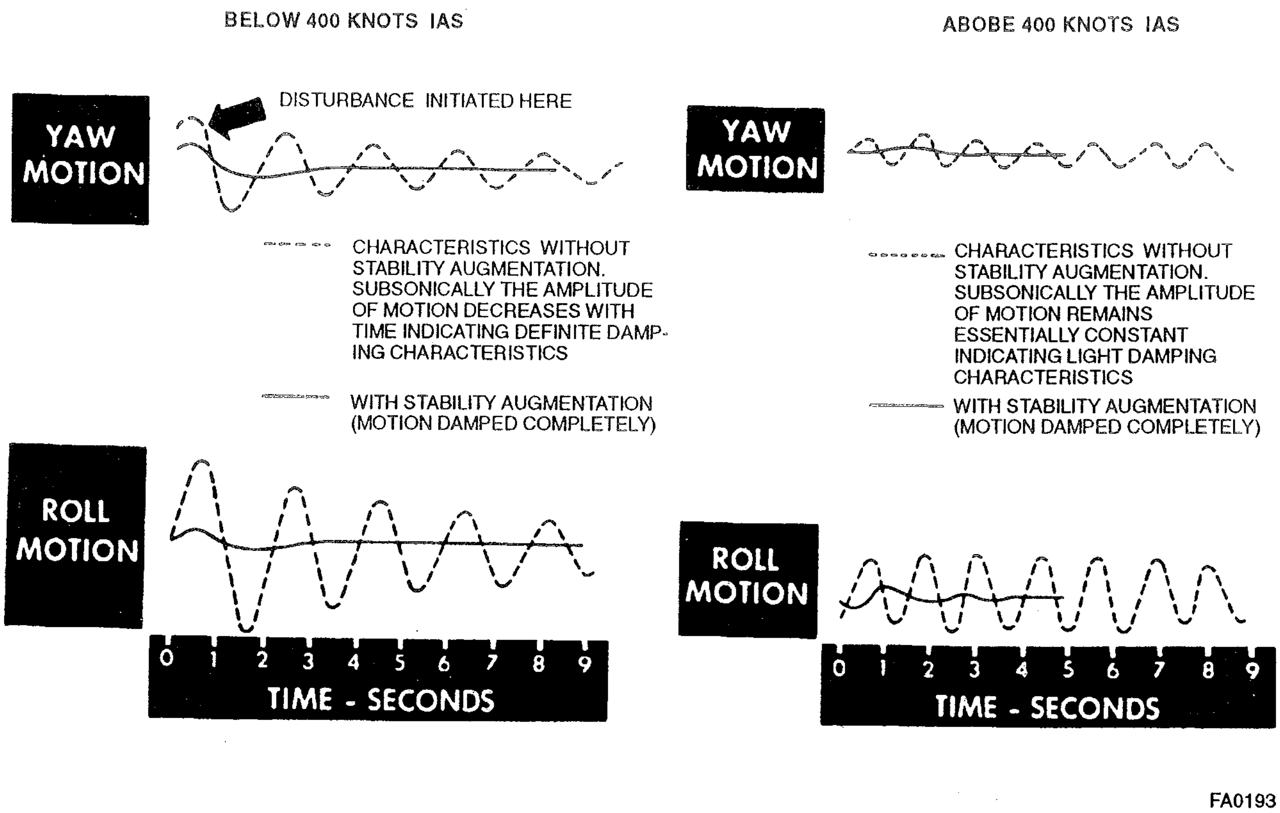


Figure 6-1

be noticed. This is because the yaw augmenter is sufficient to dampen effectively the "dutch-roll" even though the roll augmenter is inoperative. Loss of the pitch stability augmenter will be evident primarily in tracking maneuvers in that effective pitch control is impaired.

SPEED BRAKES

Speed brakes effectiveness is proportional to airspeed. At high airspeeds they produce tremendous drag without objectionable buffeting and only a mild nose-up trim change below Mach 1.75. A mild nose-down trim change is experienced at higher speeds. Combined with engine power reduction the speed brakes provide exceptionally rapid deceleration, particularly at the lower altitudes.

AIR INTAKE SCOOPS

The air intake scoops are of the high-supersonic type. The conical ramps leading up to the inlet cause oblique, or slanting shock waves at high supersonic speeds rather than vertical or normal shocks. These oblique shock waves decelerate the airflow before it enters the inlet, thereby reducing the intensity and resultant pressure losses through the normal shock at the inlet.
These scoops, combined with the internal characteristics of the ducts accomplish efficient compression of the air and maintain a high level of pressure at the engine inlet over the wide operating range of the aircraft.

TAKEOFF CHARACTERISTICS

Aircraft handling qualities and response during takeoff are excellent. Upon aft stick application the nose will initially raise until the nose strut is fully extended and then the aircraft will rotate to the takeoff attitude. The minimum speed at which the aircraft will begin to rotate and assume the takeoff attitude is dependent on the stabilizer deflection, the rate at which back stick is applied, aircraft gross weight and the aircraft center of gravity.
In general, the speed at which rotation and subsequent nosewheel lift-off occur will increase as the

EFFECT OF PITCH STABILITY AUGMENTER ON DYNAMIC FLIGHT CHARACTERISTICS

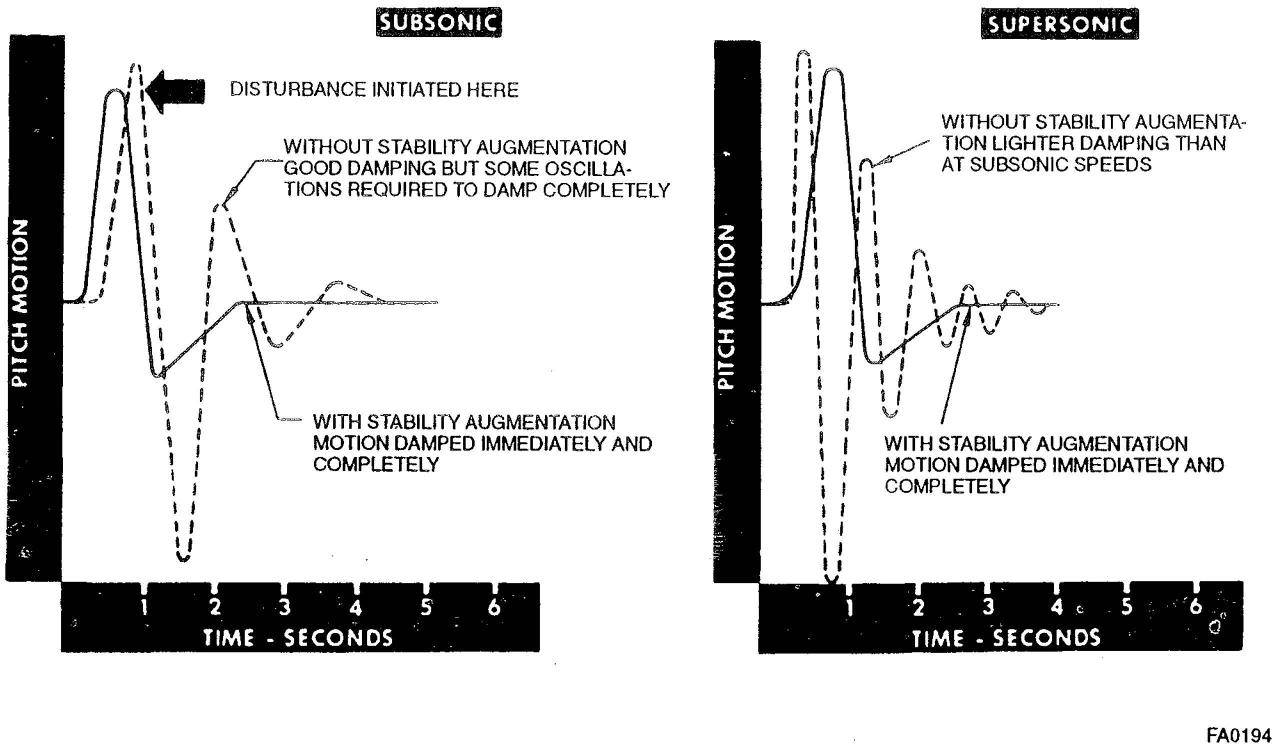


Figure 6-2

aircraft weight is increased and as the center of gravity is moved forward. The addition of external fuel tanks or wing pylon armament will result in an increase in takeoff weight but will result in a more aft center of gravity than without external stores.

The amount of stabilizer deflection and trim setting will have a large effect on rotation and nosewheel lift-off. Wind tunnel tests have determined that a smooth slightly over 3-inch aft stick input results in a more optimum nose up moment on the aircraft than if the stabilizer were at full travel.

If a stick input is made too early in the takeoff, i.e., too low an airspeed, this dynamic effect can cause the nose to raise slightly then fall back again once the inertial (dynamic) effect bleeds off.

This will give the sensation of an ineffective stabilizer or "light stick". Because of this, it is important that the correct amount of aft stick be initiated at the speed at which the stabilizer will develop sufficient lift to raise the nose and achieve takeoff attitude.

The speed at which the nosewheel leaves the runway should always be less than or equal to the computed takeoff speed. (Refer to Appendix for computed takeoff speed).

A lowered or binding nose gear strut will affect and increase the speed at which the nose will begin to rotate, however it will not affect the speed at which the nosewheel leaves the runway.

The maximum amount of aircraft nose up stabilizer travel of 17° leading edge down can be obtained only when the stabilizer is trimmed between 1° and 11° leading edge down. The optimum takeoff trim setting is 5° leading edge down. If the trim is set at less than 1° leading edge down, full back stick will not provide the full 17° of travel.

If the trim shifts or is positioned full nose down, 3 inches aft stick travel will be far short of the optimum stabilizer deflection angle (12.5°) required to achieve nosewheel lift-off. If the trim shifts or is positioned full nose up, then the 3-inch aft stick input will provide a stabilizer deflection of 17° versus the optimum 12.5° (Figure 6-3).

TAKEOFF TECHNIQUE

For the takeoff "optimum stick technique", refer to Section II, Normal Takeoff.

HORIZONTAL STABILIZER POSITION VERSUS CONTROL STICK POSITION AT VARIOUS TRIM CONDITIONS

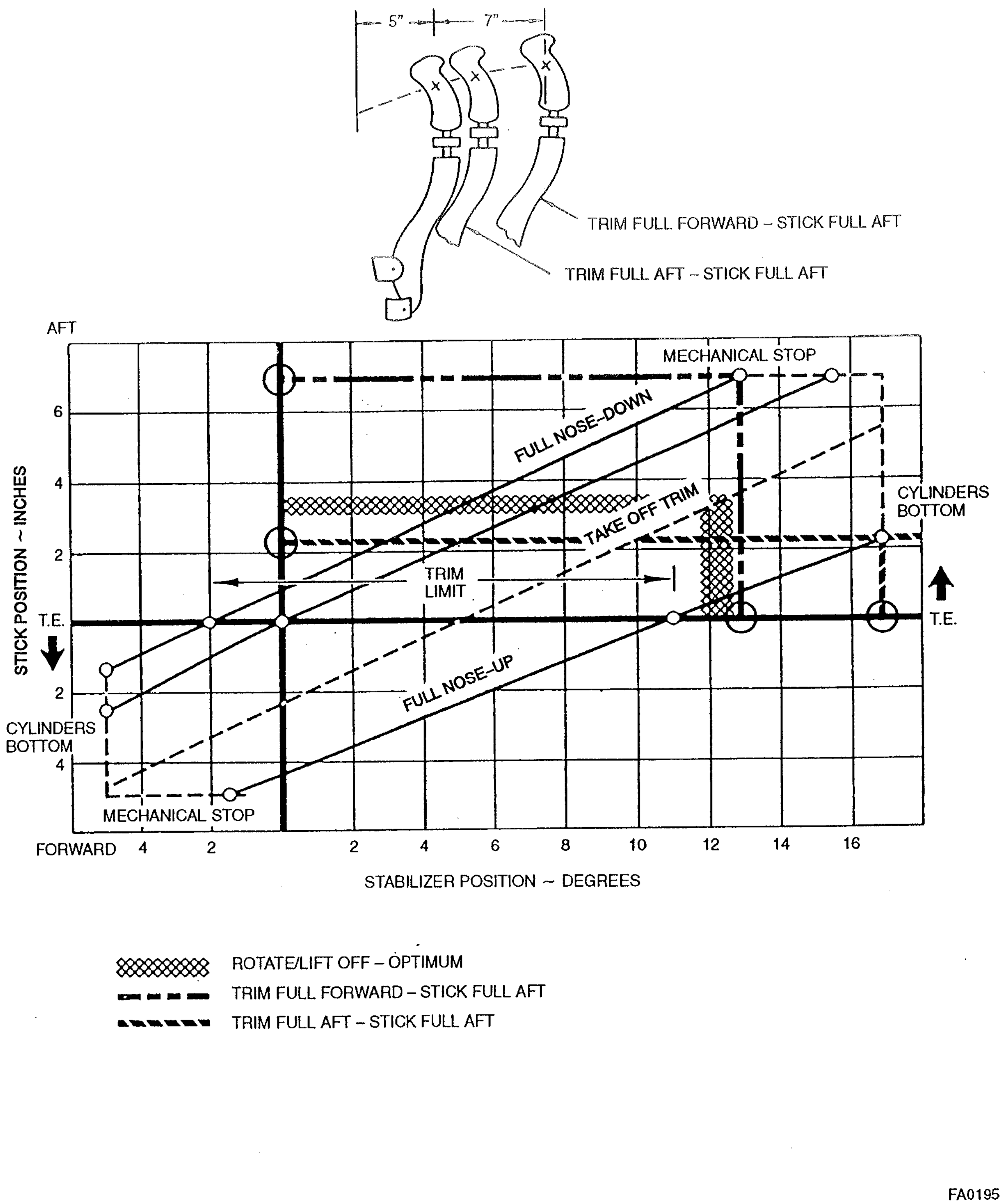


Figure 6-3

STALLS

The airflow characteristics associated with the high-fineness-ratio fuselage and the sharp leading edge, and the low-aspect-ratio wing at high angle of attack combined with the high horizontal tail position, result in a pronounced pitchup characteristic in the fully stalled condition. Beyond a certain point, this pitchup is uncontrollable and results in severe gyration of the aircraft and a large loss in altitude before recovery to level flight.

At high indicated airspeeds, structural failure of the aircraft will result under the excessive airloads due to the large angles of pitch and yaw encountered in such a maneuver.

In addition to these characteristics at the stall, it is possible to develop stall angles of attack very readily and rapidly in abrupt maneuvers, such as quick pullups, even though relatively small amounts of stabilizer are used. This results from the combination of high stabilizer effectiveness and the high inertia in pitch of the modern supersonic aircraft. Under these conditions the usual stall warnings are inadequate to prevent an excessive angle of attack.

Because of these characteristics, this aircraft incorporates an automatic pitch control system which provides adequate warning by initiating corrective action at the proper time to prevent reaching an angle of attack high enough to cause pitchup under any operating condition.

AUTOMATIC PITCH CONTROL (APC)

The APC system provides stall warning in the form of a stick shaker followed by kicking the stick forward abruptly. This action provides pilot warning and automatically applies aircraft nose-down stabilizer to initiate pilot follow-through. It is, in effect, both a built-in buffet warning and an artificial stall that occurs ahead of the aerodynamic stall. The APC operational boundaries are actuated by two stick shaker channels, and a single kicker channel which also drives the APC meter in the cockpit.

One stick shaker channel responds to angle-of-attack only, sensed by the vane located on the forward left side of the fuselage.

The second stick shaker channel provides similar response utilizing the vane on the forward right side of the fuselage.

The third response channel provides kicker operation signals, also sensed by the vane on the right forward fuselage. In addition, pitch rate is used in combination with the right forward fuselage vane shaker and kicker channels to provide an anticipatory function during maneuvering flight.

The left and right vanes provide shaker operation sufficiently before kicker operation to warn of the impending stall. The left vane actuates the stick shaker from low speed to approximately 1.3 Mach number. The right vane actuates the stick shaker at higher Mach numbers.

When TAKEOFF flaps are selected, the angle at which the angle-of-attack sensing vanes energize the stick shaker or kicker is automatically increased, permitting operation at a higher aircraft angle of attack flight.

WARNING

THE ANGLE AT WHICH THE ANGLE-OF-ATTACK SENSING VANES ENERGIZE THE SHAKER OR KICKER IS DEPENDENT UPON THE FLAP LEVER SETTING AND NOT THE ACTUAL POSITION OF THE FLAPS. IN NORMAL OPERATING CONDITION AFTER FLAP LEVER HAS BEEN SELECTED FROM UP TO TAKEOFF (APC BOUNDARY VARIATION) AND BEFORE SAFE TAKEOFF INDICATION IS ACHIEVED A TIME INTERVAL OF 7/8 SECONDS EXISTS. THEREFORE INCREASING THE LOAD FACTOR (AOA) DURING THIS TRANSIENT CAN CAUSE A PITCH-UP WITH NO PREVIOUS SHAKER OR KICKER OPERATION.

In maneuvering flight such as a turn or dive pullout, the aircraft experiences a pitch rate about its pitch axis. This is necessary in order to change the flight direction. Under steady-state turn or pull-up conditions, the pitch rate will be constant and there will be no change in aircraft angle-of-attack. However, when such a maneuver is initiated, there is a change in angle-of-attack and an angular pitch acceleration. Under these conditions the aircraft moment of inertia and the angular rate of pitch change will result in momentum to continue pitching.

The desired "g" will be overshot unless it is anticipated and back stick is relaxed prior to reaching the desired "g". The amount of overshoot will depend on the magnitude of the rate of change of angle-of-attack generated; consequently, the higher the rate, the more overshoot. This means that, when back stick is applied abruptly to initiate a pullup, the stick must be moved forward much earlier to stop the nose up pitching than in a slow entry pullup.

When this characteristic is related to the APC system, it may be seen that the shaker and kicker also must anticipate in order to prevent the aircraft overshooting into the pitch-up flight regime. The shaker and kicker are set to operate as a function of aircraft pitch rate as well as angle-of-attack to provide this anticipatory function.

Under steady-state turning or pullup conditions, the aircraft will have a pitch rate but the rate of angle-of-attack change will be zero. Under these conditions, there will be no overshoot into pitch up; therefore, the anticipatory function of the pitch rate input would penalize the aircraft maneuverability.

In order to avoid this loss of maneuverability due to the pitch rate input to the APC, the pitch rate signal is "washed out" prior to summing with the angle-of-attack vane. Washout is accomplished such that any given constant aircraft pitch rate signal into the APC will be reduced to ¾ of the aircraft pitching rate in ½ second.

Therefore, the APC provides pitch-up protection by anticipating the overshoot in any type of maneuver entry such that the angle-of-attack never exceeds the maximum safe value.

As long as the aircraft is operated beyond the safe maneuvering range, the kicker will continue to operate and continue to force the stick forward. The kicker moves the stick forward to slightly ahead of neutral. The amount of aircraft nose-down stabilizer that is applied is therefore dependent on the stabilizer trim setting.

Under normal operating conditions, this amount of stabilizer ensures adequate nose down corrective action. If the aircraft is trimmed close to a high angle-of-attack condition, additional pilot follow through corrective action is necessary. The kicker is inoperative with flap handle in LAND or with gear DOWN. Thus, the kicker feature is available for high speed maneuvering with flaps UP or TAKEOFF flaps but is inoperative when the gear is down to prevent undesirable kicker operation during takeoff and landing.

Figure 6-4 shows the maneuvering boundary of the kicker and stick shaker in terms of Mach number, and Figure 6-5 presents the kicker boundary in terms of indicated airspeed.

MANEUVERING BOUNDARIES OF AUTOMATIC PITCH CONTROL SYSTEM

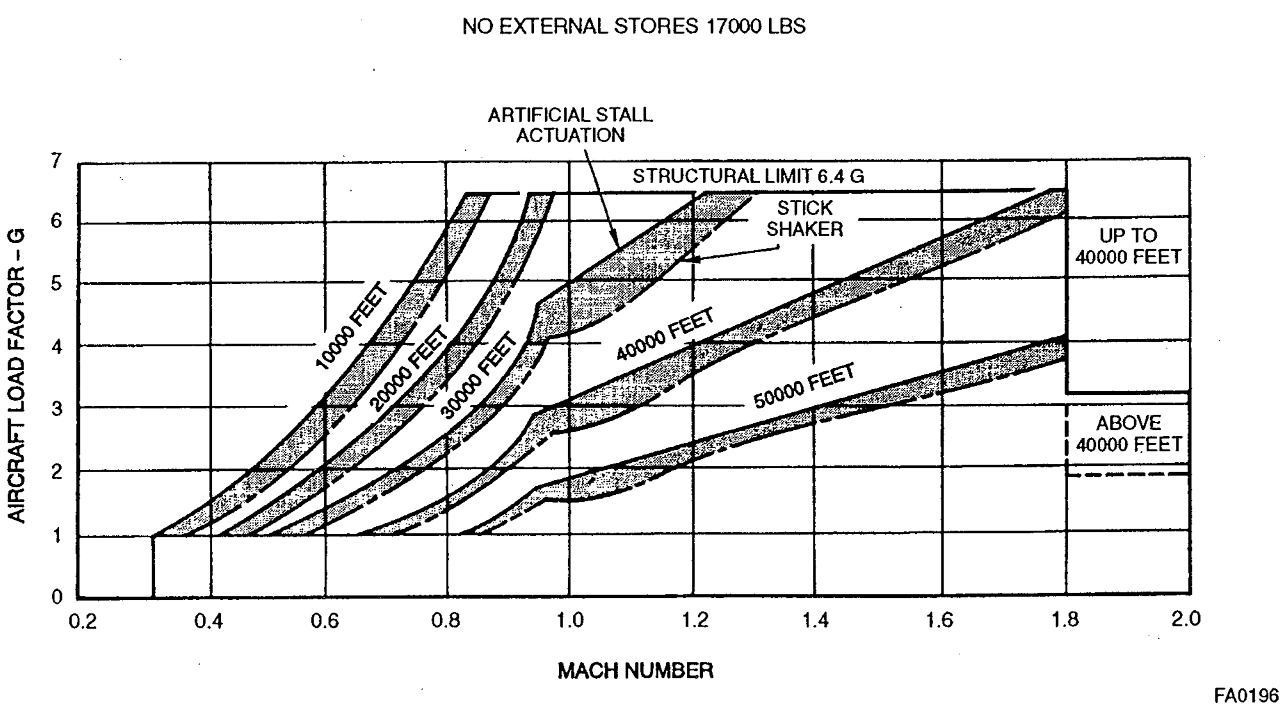


Figure 6-4

MINIMUM OPERATING SPEEDS — KNOTS IAS

(MINIMUM CONTROL SPEED IN PARENTHESIS)

ALTITUDE SEA LEVEL TO 10000 FT	Bank Angle Load Factor	GEAR AND FLAPS UP POWER ON OR OFF			TAKEOFF FLAPS GEAR UP OR DOWN POWER ON OR OFF			LAND FLAPS GEAR UP OR DOWN POWER ON		
		0° 1.0G	40° 1.3G	60° 2.0G	0° 1.0G	40° 1.3G	60° 2.0G	0° 1.0G	40° 1.3G	60° 2.0G
Any configuration not including wing-tip stores (except missile launchers)	23500 Pounds	245 (220)	285 (250)	345 (310)	210 (200)	240 (225)	295 (280)	180 (170)	205 (195)	255 (240)
	21500 Pounds	235 (210)	270 (240)	330 (295)	200 (190)	230 (215)	280 (265)	175 (160)	200 (180)	245 (225)
	19500 Pounds	225 (200)	260 (230)	315 (280)	190 (180)	220 (205)	270 (225)	165 (150)	190 (170)	235 (210)
	15000 Pounds	195 (175)	225 (200)	280 (245)	170 (160)	195 (180)	225 (220)	145 (130)	165 (150)	205 (185)
Any configuration including wing-tip stores	23500 Pounds	235 (215)	275 (245)	335 (300)	200 (190)	235 (220)	285 (270)	175 (165)	200 (190)	250 (235)
	21500 Pounds	225 (205)	260 (235)	320 (290)	195 (185)	225 (210)	275 (260)	170 (155)	195 (175)	240 (220)
	19500 Pounds	215 (195)	250 (225)	305 (275)	185 (175)	215 (200)	260 (245)	160 (150)	185 (170)	230 (210)
	15000 Pounds	190 (170)	220 (195)	270 (240)	160 (150)	185 (175)	230 (215)	140 (130)	160 (150)	200 (180)
Any configuration including tip tanks	28000 Pounds	245 (225)	285 (260)	350 (320)	215 (205)	250 (235)	305 (290)	190 (175)	220 (200)	270 (245)
	23500 Pounds	225 (205)	260 (240)	320 (290)	195 (185)	230 (215)	280 (265)	175 (160)	200 (185)	245 (230)
	21500 Pounds	215 (200)	250 (230)	305 (280)	190 (180)	220 (205)	265 (250)	165 (155)	195 (175)	235 (220)
	19500 Pounds	205 (190)	240 (220)	290 (265)	180 (170)	210 (195)	255 (240)	160 (145)	180 (165)	225 (210)
	15000 Pounds	180 (165)	210 (190)	260 (235)	160 (150)	180 (170)	220 (210)	140 (125)	160 (145)	195 (180)

NOTES

1. Minimum operating speeds are the speeds at which automatic stick-shaker action occurs.
2. Mimimum control speeds are the speeds at which either of the following occur:
   a. Kicker is experienced with flaps UP or with TAKEOFF flaps and gear UP.
   b. Noticeable stability reduction is experienced with TAKEOFF flaps and gear DOWN, or with LAND flaps.
3. Full stall will be encountered if there is further reduction of speed.
4. Speeds in excess of the LAND flap limit are shown for interpolation purposes only.

Figure 6-5

WARNING

AVOID RAPID MANEUVERS DURING PULL-OUTS OR TURNS WHICH INDUCE HIGH PITCH RATES WITH THE APC SYSTEM DEACTIVATED. STAY OUT OF THE STICK SHAKER BOUNDARY AS THERE IS NO WAY OF KNOWING HOW FAR THE BOUNDARY HAS BEEN PENETRATED UNTIL PITCH-UP OCCURS. IF THE STICK SHAKER BOUNDARY IS PENETRATED INADVERTENTLY, REDUCE THE G-LOAD AND INCREASE POWER IF NECESSARY.

The APC meter is adjusted to read a value of 5 at stick kicker for all configurations.
There is no specified adjustment for shaker; however, in steady state flight, typical readings will be about 3.5 with flaps UP subsonic and as high as 4.5 supersonic.
With flaps at TAKEOFF, the meter readings will be about 4.0. LAND flaps is the same as flaps UP.

UNACCELERATED STALLS

Since automatic pitch control actuation is, in effect, a built-in stall, the word stall is used in the following discussions to define the point of automatic pitch control operation.
The low-speed stall is preceded by a rather wide speed range of heavy airframe buffet. This buffeting builds up from light to heavy and then remains heavy with further reduction in airspeed. Speeds below the onset of heavy buffet do not represent an useful operating range of the aircraft. With further reduction in speed, the aircraft becomes laterally unstable.
This lateral instability may increase in intensity or reflect a definite wing drop tendency just prior to the stall. The stick-shaker action will be noticeable under the airframe buffeting condition, indicating that further reduction in speed will result in stall.

WARNING

APPROACHES TO A STALL IN AIRCRAFT CONFIGURATIONS WHERE THE KICKER IS INOPERATIVE SHOULD BE TERMINATED AT STICK-SHAKER ACTION, LATERAL INSTABILITY, OR WING DROP, WHICHEVER OCCURS FIRST. IN ADDITION, IF THE KICKER IS INOPERATIVE THE STALL WARNINGS IN ABRUPT HIGH-PITCH-RATE MANEUVERS ARE INADEQUATE. THEREFORE, THIS TYPE OF MANEUVER MUST BE AVOIDED. THE KICKER IS INOPERATIVE WITH FLAP LEVER IN LAND OR WITH GEAR DOWN.

ACCELERATED STALLS

In the subsonic region, stall characteristics are similar to those described under unaccelerated stalls in that a range of natural airframe buffet and lateral instability warning precedes the stall.
In the transonic region, the speed or "G" range of natural buffet and lateral instability warning gradually reduces and is indicated to be nonexistent above approximately Mach 0.9. In this subsonic range and at all supersonic speeds the stick-shaker warning preceding the stall in normal maneuvering flight provides the only warning prior to the stall.

PRACTICE STALLS

Practice stalls maintained to stick-shaker warning or to the minimum control speeds of Figure 6-5 may be executed at any reasonable altitude; however, 25000 feet should be used for general familiarization with aircraft characteristics. The airspeeds at which the various low-speed flight characteristics occur vary with external store configuration, gross weight, altitude, and load factor; however, typical unaccelerated stall approaches for an aircraft with no external stores and a gross weight of 15200 pounds are described in the following paragraphs.

Increase speeds approximately 5 knots for each additional 1000 pounds of internal fuel weight or pylon/fuselage stores weight. The addition of tip stores will lower the typical speeds due to an incremental increase in aircraft lift. See Figure 6-5 for magnitude.

Gear and Flaps Up

In the clean configuration, the aircraft starts buffeting at approximately 225 knots; this buffeting becomes heavy at 215 knots. The stick-shaker operates below approximately 200 knots. Lateral instability will be experienced at approximately 190 knots, increasing in intensity to kicker operation at 175 knots.

Takeoff Flaps – Gear Up or Down

With TAKEOFF flaps extended, the stick-shaker action is the initial stall warning at 170 knots. Moderate airframe buffet is experienced at 170 knots with a noticeable lowering of overall stability at 160 knots, the minimum control speed. With gear up, the kicker will actuate at 160 knots.

Land Flaps – Gear Up or Down

With LAND flaps, gear up or down, and with sufficient engine RPM for boundary-layer control operation, there is no airframe buffet or significant lateral instability. At 25000 feet with 100% RPM, the stick-shaker action will be felt at 155 knots. As speed is lowered further a noticeable lowering of overall stability will occur at 140 knots, the minimum control speed.

GROUND EFFECT

Due to the ground effect during takeoff and landing, buffet and lateral instability characteristics will not be experienced if the recommended operating speeds are used; however, if the aircraft is lifted off or held off to speeds below the recommended speeds, lateral stability and control will deteriorate and wing-drop tendencies will occur. In addition, the high pitch angles required for flight at these low speeds, will be excessive and can result in tail dragging.

PITCH-UP/SPINS, PITCH-UP RECOVERY AND SPIN RECOVERY

PITCH-UP/SPIN FLIGHT TEST PROGRAM

The flight test program evaluating pitch-up and spins did not investigate flat spins, inverted spins, spins at very high altitude, spins with the engine flamed out, supersonic pitch-ups and spins, nor pitch-ups and spins with external stores.
Subsequent operational experience has shown that if the critical angle-of-attack is exceeded and/or either a nose up or nose down rapid pitch rate is input when approaching the apogee of a high altitude zoom, then the aircraft will digress into a flat spin from which recovery may not be effected using the flight controls.

WARNING

INTENTIONAL PICH-UP AND SPINS ARE PROHIBITED BECAUSE OF THE HIGH LOADS THEY IMPOSE ON THE AIRCRAFT. THESE LOADS CAN BE OF SUFFICIENT MAGNITUDE TO CAUSE STRUCTURAL DAMAGE.

PITCH-UP

This aircraft has a pronounced pitch-up characteristic in the fully stalled condition. Pitch-up is aircraft post-stall-gyrations of uncontrolled motions about all axis following aggravated stall penetration and departure from controlled flight.

Subsonic 1 "G" Pitch-up

Pitch-up entered from subsonic 1 "G" flight is preceded by buffet, stick-shaker, lateral instability, and stick kicker. If the kicker warning is ignored and angle-of-attack is increased beyond the kicker boundary, the aircraft will pitch abruptly to extreme attitudes of 50° to 60° with oscillations in roll and yaw varying in magnitude.

Usually, two or three complete rolls are performed with yaw oscillations of $\pm 60°$ and oscillatory vertical acceleration of $+2$ or -1G.

At the first indication of pitch-up, recovery may be possible and entry into a spin mode prevented by immediately and rapidly moving the stick full forward with full nose down trim. Pitch-up cannot be checked if it is allowed to go beyond the initial stages and as much as 15000 feet may be lost before recovery is complete.

If the internal fuel loading is greater than 2000 pounds, the aircraft is considered spin resistant and will usually oscillate out of control about all three axes until a nose down attitude is attained. If pitch-up has been allowed to develop fully with less than this fuel loading (more aft CG location), the aircraft will probably enter a spin before pitch-up recovery can be made, regardless of control positions.

NOTE

Pitch-up differs from a developed spin. During pitch-up, the turn needle will oscillate around the mid-position; during a spin, the turn needle will be pegged in the spin direction.

Pitch-up Operation Experience

Operational experience has shown that pitch-up recovery can be accomplished with only 2000 to 3000 feet altitude loss, as long as the pitch-up is entered from level or climbing flight attitude and the pitch-up recovery procedures are initiated immediately. The aircraft will usually snap or roll rapidly around the longitudinal axis and oscillate about the lateral and vertical axis until a nose-down attitude is attained. These rolls should not be mistaken for a spin and should cease within two or three rolls. To preclude pitching back into another out-of-control condition, forward stick should be held until all oscillations stop.

When all oscillations have stopped, normal control use will be effective for regaining the desired flight attitude. Under these conditions, even with high bank angles and airspeed around 250 KIAS, the resultant nose-down attitude will be about 20° and 200 to 230 KIAS as aircraft oscillations stop. The situation is much more critical when entry into a pitch-up starts from a nose low attitude and the required altitude for dive recovery is not available (refer to altitude loss in dive recovery, this Section).

Subsonic Accelerated Pitch-up

Pitch-up entered from subsonic accelerated flight is preceded by the same stall warnings; however, the aircraft will progress through buffet and lateral instability more rapidly, pitch-up will be more abrupt, and post pitch-up gyrations will be more violent.

Supersonic Pitch-up

Supersonic pitch-up has not been investigated. Slow rate approaches have been carried into the neutral stability region but not allowed to develop into the uncontrollable region. Natural stall warnings such as buffet or lateral instability are nonexistent; therefore, the stick shaker and kicker provide the only supersonic stall warnings.

WARNING

AIRCRAFT STRUCTURAL FAILURE WILL RESULT FROM THE EXCESSIVE AIRLOADS OF SUPERSONIC PITCH-UP GYRATIONS.

Pitch-up Recovery

At the first indication of pitch-up and if time and conditions permit, entry into a spin mode may be prevented if immediate action is taken to perform the Pitch-up Recovery procedures in Section III.

SPINS

Normal Spin

The normal spin motion is characterized by one complete oscillation in pitch and two oscillations in roll per turn coincident with an amplitude of oscillation in yaw that is more sizeable to the right throughout the spin proper. The peak pitch angle, however, recedes steadily in each turn producing more amplitude in the nose-down direction as the right yaw rate diminishes.

The high angle-of-attack breaks at this time, the autorotation reduces to roll and recovery occurs as the stabilizer becomes fully effective. A spin revolution will result in a loss of 1800 to 2000 feet. Each revolution will take 5 to 6 seconds and result in around 18000 feet per minute rate of descent. With more than 2000 pounds of fuel remaining (fwd c.g.), the aircraft is considered spin resistant and will generally oscillate out of control about all three axes until a nose down attitude is attained. At fuel loadings of less than 2000 pounds (aft c.g.) the aircraft will probably enter a spin following any pitch-up.

If the engine is running, the spin will probably be to the right due to large gyroscopic inertia moment causing a right yaw during pitch-up. The aircraft may spin either left or right depending upon the control inputs and the lateral-directional oscillations during pitch-up. With the engine flamed out, the spin may be either left or right.

Stable Flat Spin

The stable flat spin is characterized by a rotation rate of approximately one revolution every 6 seconds, nose approximately 10° below the horizon, wings level or slightly wing down, and no oscillations about any axis.

WARNING

ACCOMPLISH SPIN RECOVERY PROCEDURES IMMEDIATELY TO PREVENT THE EARLY OSCILLATORY SPIN FROM DEVELOPING INTO A STABLE FLAT SPIN. RECOVERY CAPABILITY FROM A STABLE FLAT SPIN HAS NOT BEEN ESTABLISHED.

Inverted Spins

Inverted spins have not been investigated; however, recovery procedure for inverted spins is to neutralize all controls.

Spin Recognition

Visual cues outside the cockpit, the spin characteristics, and the turn needle (not the ball) should all be used to verify a spin condition and the spin direction. The turn needle will always be pegged in the spin direction.

Recovery Problems

After spin rotation has stopped, altitude required for dive recovery can be as much as 12000 feet due to low initial airspeed. The dive recovery must be gradual to avoid excessive angles-of-attack. It may not be possible to rely on the APC system if RPM has dropped below 20% and the RAT is not extended. The engine may flame out or a compressor stall may occur at entry into or during the spin.

If engine operation is normal advance throttle to give 95 to 100% RPM. If engine stall occurs, retard throttle as required to control EGT. If engine has flamed out, recovery will be more difficult; however, an air start should not be attempted until spin rotation has been stopped and the dive recovery initiated.

If normal recovery technique fails or a stable fault spin has developed, deploy the drag chute. If the chute is used, it should be deployed above 25000 feet. This is to provide recovery by 5000 feet since a loss of 6000 to 8000 feet may occur from the time of deployment to drag chute failure, and a loss of 12000 feet during dive recovery. There is no maximum altitude limit for chute deployment. Deploy the chute at or above 40000 feet in spins entered at high altitude.

As soon as rotation stops, stabilize the aircraft in a vertical nose-down attitude. If there are persistent pitching oscillations, jettison the drag chute as the nose swings down. Application of forward stick as the chute is jettisoned may be required to prevent abrupt nose-up pitching. Use slight aft stick pressure to recover without exceeding the shaker boundary. If the APC system is inoperative, control attitude by avoiding excessive aircraft buffet.

WARNING

EJECT, IF ROTATION HAS NOT STOPPED BY 15000 FEET AGL.

Spin Recovery

If the aircraft enters a spin and time conditions permit, take immediate corrective action to perform the Spin Recovery procedures in Section III.

Vertical Stall Recovery

In the event extremely steep or vertical climb paths are maintained to the point where normal recovery appears questionable, the recommended procedure is to neutralize the controls and allow the maneuver to "peak out". In most cases, the aircraft will acr over the top of its zoom and normal dive recovery can be effected when sufficient airspeed has been regained. In the extreme case of a vertical zoom, a mild "hammer-head" stall-type maneuver will probably be experienced at the apex of the flight path as the aircraft reverses direction and heads back toward the ground. Altitude required for complete recovery to level flight can be as much as 15000 to 20000 feet. In all probability, a pitch-up will not be encountered in a vertical zoom; however, the possibility should not be ignored. In such cases, follow the proper recovery procedure.

PERFORMANCE CAPABILITIES

THRUST AND DRAG

The relatively low drag and the high thrust-to-weight ratio of this aircraft result in high performance capabilities. This relationship of thrust and drag results in supersonic level-flight speeds over a wide range of altitude. An understanding of the thrust and drag relationship will permit optimum utilization of these capabilities.

Figure 6-6 shows typical thrust and drag variations over the flight range of speed and altitude.

At a given speed, the difference between thrust and drag represents the excess thrust that is available to produce rate of climb, accelerate the aircraft from one speed to another, or to maneuver without losing speed. The intersections of the thrust and drag lines are the points where thrust and drag are equal and therefore represent the stabilized level-flight conditions. The thrust lines shown are for the maximum thrust operation. Obviously, at lower power settings the entire level of thrust is lowered, resulting in less excess thrust and lower stabilized level-flight speeds. Near sea level, level-flight speeds are only slightly supersonic even at maximum afterburning thrust. Maximum rate of climb under these conditions occurs at high subsonic speed.

At intermediate altitudes the thrust and drag curves do not intersect within the permissible speed range, and maximum level-flight speed is limited only by engine and airframe design limitations. In this altitude range, sufficient excess thrust is available at all flight speeds for climb, acceleration, and maneuvering flight without loss of speed.

As altitude is increased, due to the relative shape of the two curves, drag exceeds thrust at subsonic speed at a lower altitude than at supersonic speed. This results in a considerable variation in the power-limited ceiling with flight speed. Near the subsonic ceiling the thrust exceeds the drag in two regions, a subsonic and a supersonic region. This means that in order to reach supersonic speed at this altitude the proper climb schedule must be used. For example, if you climbed to this altitude at a subsonic speed the aircraft would not accelerate past the transonic drag rise.

The only way to become supersonic in this case would be to lose altitude, accelerate, and make a supersonic climb to the desired altitude. In the supersonic speed range, the excess power (excess thrust multiplied by velocity) increases with flight speed, resulting in the most excess power at Mach 2.0. At high Mach numbers, dependent on ambient temperature, an increase in engine RPM (due to T_2 reset) occurs resulting in an increase in thrust. This is also illustrated in Figure 6-5. At this speed maximum supersonic rate of climb and constant speed maneuvering load factor are obtainable. The maximum level-flight altitude will be obtained also at this speed as it is the last point at which the thrust and drag curves touch.

FLIGHT CONDITIONS WITH OPEN NOZZLE (NON-AFTERBURNING)

For flight conditions with open nozzle (non-afterburning) see Section III.

PERFORMANCE ENVELOPE

The performance envelope (the speed and altitude capability of the aircraft) contains three separate and distinct regions. These are illustrated in Figure 6-7. The Military thrust region is limited in speed and altitude by the available thrust and by the low-speed stall. The right-hand boundary is the maximum level-flight speed without afterburning. This boundary can be extended in dives to about Mach 1.3.

The Maximum thrust region represents the level-flight speed versus altitude capability with full afterburning thrust. Level-flight speeds are limited under standard atmospheric temperature conditions only by engine compressor restrictions from approximately 10000 feet to altitudes well over 50000 feet. Note the increase in ceiling of approximately 10000 feet obtainable at high supersonic speeds. The ceiling is also dependent upon aircraft gross weight

THRUST AND DRAG

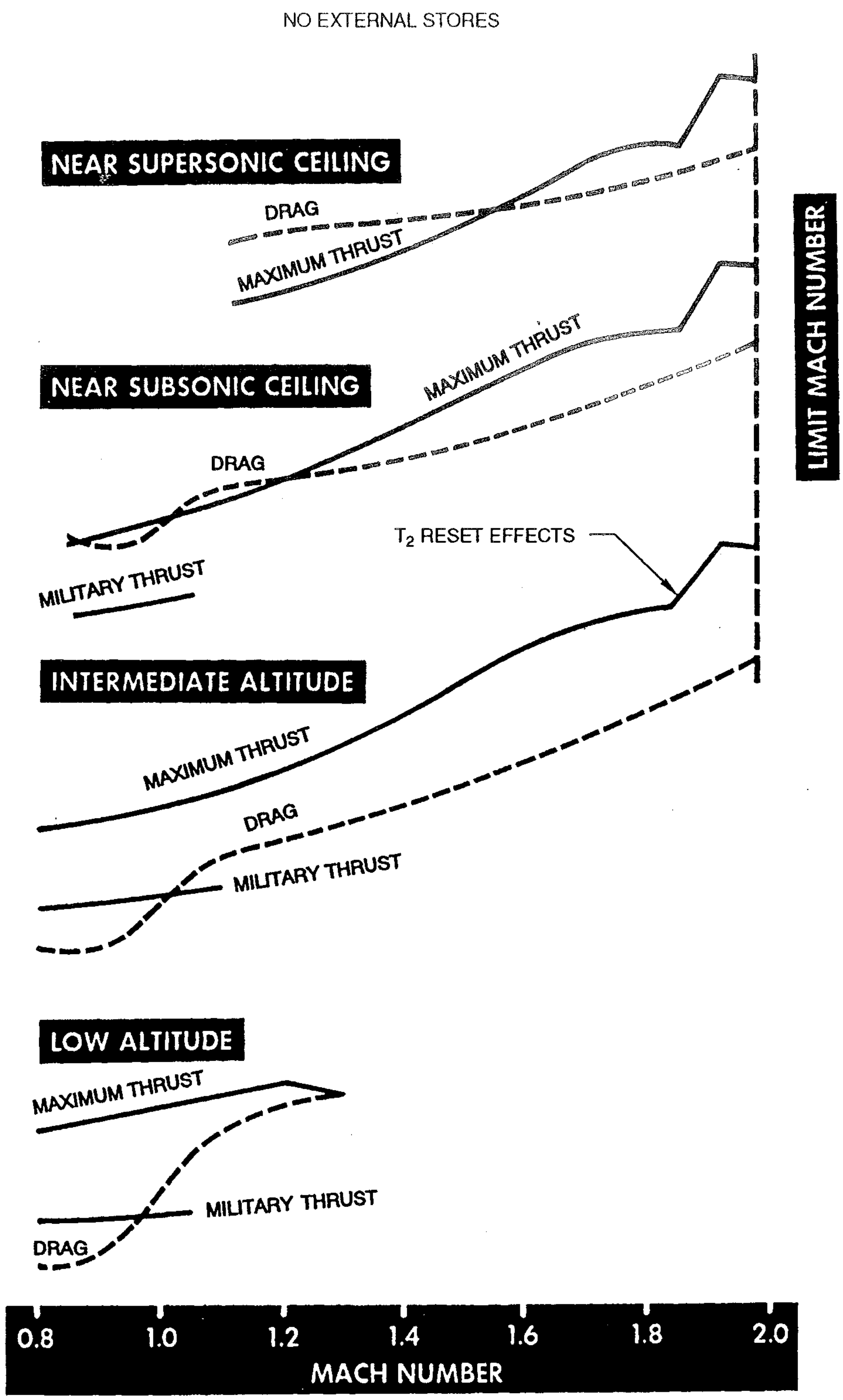


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Figure 6-6

FLIGHT ENVELOPE

NO EXTERNAL STORES
GROSS WEIGHT – 16000 LB.

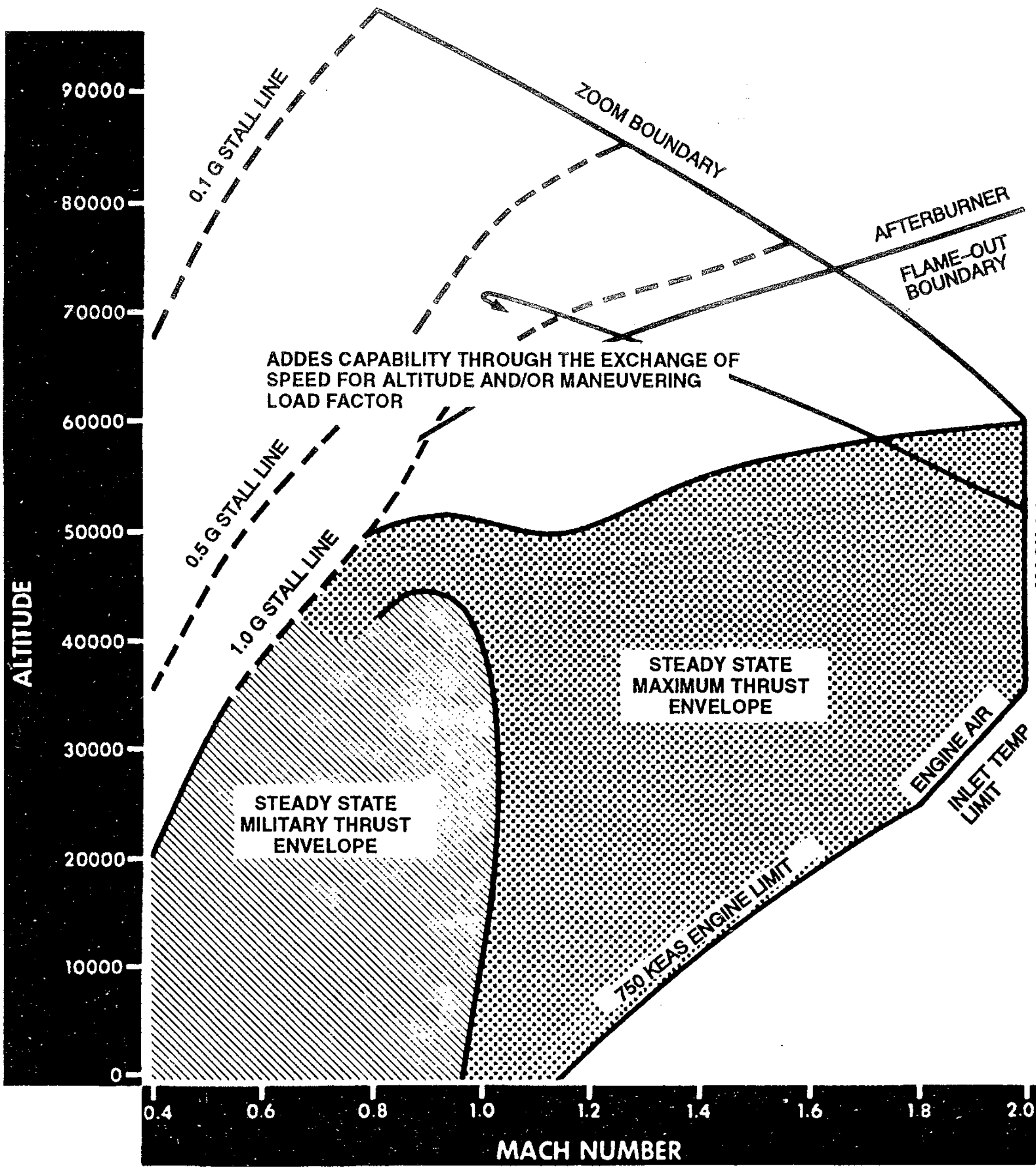

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Figure 6-7

and ambient temperature and will therefore vary slightly from day to day. The upper, or zoom path, region is where the aircraft is trading airspeed for altitude.
The aircraft possesses a high level of energy at high speed and can greatly increase its altitude capability by zooming to altitudes as great as 30000 feet above the power-limited ceiling. In this region the drag exceeds the thrust available from the engine and steady speeds cannot be maintained without loss of altitude. Therefore, careful flight planning is necessary for optimum utilization of zoom capabilities.

CLIMB

At Maximum thrust, the high rate of climb at best climb Mach number results in a steep flight path angle. Care should be exercised following takeoff to anticipate the high forward acceleration of the aircraft as climb speed is approached and to assume the proper climb attitude to ensure maximum performance. Figure 6-8 shows relative angles for the various climb configurations.

FLIGHT PATH ANGLE AT BEST CLIMB MACH

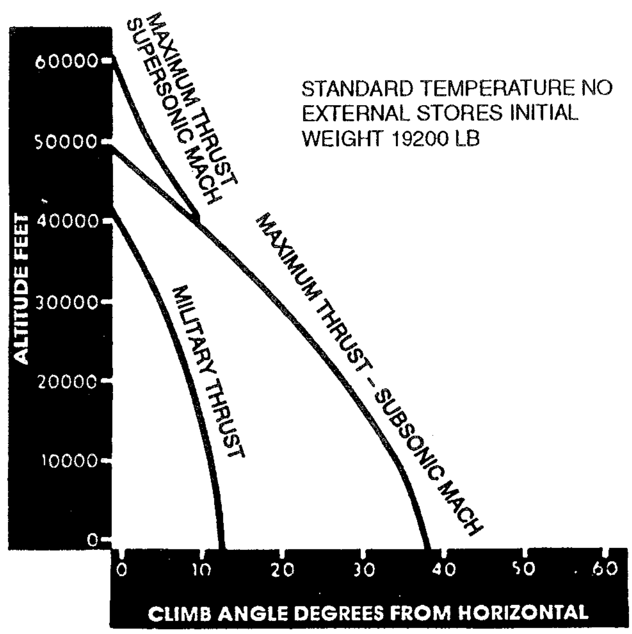

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Figure 6-8

ZOOM CLIMBS

WARNING

THE MAXIMUM PERFORMANCE ZOOM CLIMB IS A HAZARDOUS MANEUVER WHICH CAN RESULT IN A SPIN AND SHOULD BE PERFORMED ONLY IF THE OPERATIONAL REQUIREMENT WARRANTS.

A zoom climb forms an important aspect of the performance capability of the aircraft. It is the quickest way to reach a higher altitude once an adequate level of total energy has been achieved. The high speed capability of the aircraft permits a wide flexibility in changing altitude in zooming flight in relatively short distances and permits zooming to altitude far in excess of the thrust-limited ceiling. Zoom climbs can be initiated from any point within the thrust envelope of the aircraft.
The altitude reached and the final speed are dependent upon the speed and altitude at the start of the zoom. Generally speaking, the higher the initial speed and altitude, the higher will be the airspeed upon reaching a given altitude.

In zooms started from 40000 to 45000 feet, and a Mach number of 2.0, approximately 4000 feet may be gained with proper pilot technique, for each 0.1 Mach number loss. Zooms made from a lower speed or higher altitude give a smaller ratio of height gained to speed lost. This ratio may decrease to as little as 2000 feet gained for each 0.1 Mach number lost from the thrust-limited ceiling. Maximum altitude is obtained in a zoom initiated from maximum permissible speed in the 40000 foot altitude range.
In this altitude range sufficient excess thrust is available to permit rotation of the flight path to a steep climb angle with a minimum bleed-off in speed. In zooms initiated at or near the thrust-limited ceiling of the aircraft, the speed loss in rotating to steep angles is greater due to the absence of excess thrust to hold speed. Practice is required to perfect pilot technique.
Zooms to intercept a target should be preplanned as much as possible since the intercept altitude, desired intercept speed, etc. will dictate the technique required. It is essential to position the aircraft correctly before the zoom is started because further maneuvering after the zoom is initiated will decrease the energy available for the zoom.
Therefore, the approximate time and distance required must be known to conserve energy and assure closure on the target. In tactical situations, zoom climbs will probably not be initiated until

target detection has been obtained on the radar indicator. A pullup to boresight will be accomplished with a boresighted flight path followed from that point on to firing range.
In this case, the technique to be used is automatic since the flight path is dictated by the boresight requirements. In practice zooms from the Mach 2.0 thrust ceiling to a preselected altitude, it is best to make a pullup of approximately 1.5 "G", attaining a maximum climb angle of approximately 20°.
The pushover to the preselected altitude should be started after approximately one half the desired altitude gain. In zooms to above 65000 feet, afterburner blowout will occur as the minimum operating pressure level of the afterburner is crossed. The speed and altitude combination for this boundary is approximated in Figure 6-7. Actually, the boundary is a fairly wide band because the blowout point is affected by individual afterburner performance and is also sensitive to pullup or pushover technique. Also, if the flight path is leveled prior to afterburner blowout, the afterburner will blowout as speed bleeds to the blowout boundary.
As the zoom continues above the afterburner blowout boundary, the minimum fuel flow that can be supplied by the fuel control will become greater than required to maintain full RPM. When this happens the nozzle will open and EGT and RPM will increase. To avoid exceeding limit EGT, the throttle may have to be retarded to OFF.

CAUTION

THROTTLE REDUCTION TO IDLE OR POSSIBLE ENGINE SHUTDOWN MAY BE NECESSARY IN ORDER TO CONTROL EGT. IF A SHUTDOWN IS NECESSARY A RESTART CAN BE ACCOMPLISHED DURING THE DESCENT FOLLOWING THE ZOOM.

NOTE

At very high altitudes following high Mach or sustained afterburner operation, cooling airflow is reduced due to low airspeeds, it is possible for the aft section temperature to increase enough to cause illumination of the fire warning lights.

If the throttle is retarded to OFF, the engine will be windmilling. Should the windmill RPM drop below approximately 65% RPM, the under frequency relays will cut No. 1 and No. 2 generators off the buses.
However, the hydraulically-driven generator, and those items powered by the hydraulic generator, will continue to operate down to 20% RPM. Zooms accomplished to attain maximum altitude are initiated, as stated previously, from 40000 feet of wherever in this altitude range maximum Mach number is permissible.
An initial pullup of 2.0 "G" to 2.5 "G" (or limits shown in Section V) should be used to increase climb angle. As the climb progresses, increasing back stick will be required to hold the nose up; however, at no time should the stick shaker stall warning be exceeded.

NOTE

Do not trim out aft stick forces after pullup is initiated. This will insure adequate nose down control for pushover as top of zoom is approached.

A maximum climb attitude of 45° will be reached at about 50000 feet. Greater climb attitudes should not be used because indicated airspeed may drop to too low a value at the apogee. Beginning no later than 60000 feet a small nose down pitch rate must be established to maintain aircraft attitude in a safe relation to the ballistic flight path.
Airspeed should not be permitted to decrease below 100 KIAS or Mach 1.0 to assure adequate control over the top of the zoom. This is done by maintaining angle of attack well below the shaker boundary.
Although indicated airspeed over the top may fall below normal 1 "G" stall speed, the aircraft load factor will be less than 1, and consequently aircraft angle of attack will be well below stall attitude. Zoom climbs initiated from 40000 feet might reach 80000 feet if unchecked.
Because of the pressure lag in the airspeed system, the indicated readings will lag the actual flight path. This becomes obvious when the peak of the zoom has been passed as indicated by the airspeed increasing while the altimeter still shows a climb. Altimeter range limitations are exceeded in maximum altitude zooms, resulting in a maximum reading of 85000 to 87000 feet.

Prior to making a high altitude zoom, a pressure check on the high altitude suit should be made as well as cinching down the helmet because cockpit pressurization can be expected to decrease, causing suit inflation if engine stall is encountered above the afterburner blowout point.
The natural descent angle from the peak altitude of a zoom can be expected to be as steep, or nearly, as the climb angle.

GO-AROUND

The excess thrust available for go-around varies with aircraft configuration, airspeed, gross weight, field elevation and ambient temperature. In the landing configuration, Military thrust may be inadequate for go-around (or even approach) as extremes of these variables are approached; afterburning thrust will then be necessary.

Figure 6-9 shows the effect of temperature, altitude, and weight on go-around capability in terms of maximum speed and rate of climb available with Military thrust.
However, before a rate of climb can be attained, several seconds may be required to make the transition from the downward flight path. Altitude lost during this time may exceed 100 feet, depending upon the steepness of the glide slope.
Therefore, flat approaches are essential and the pilot should be ready to apply immediate power under marginal conditions.
In addition, under marginal conditions, a straight-in approach with TAKEOFF flaps or gear-up configuration is recommended. Change to final landing configuration only after the landing is assured. Determination of marginal conditions can be made readily before flight from the Rate-of-Climb for Go-Around Chart in the Appendix.

CHANGE IN GO-AROUND CAPABILITY WITH AIRSPEED, ALTITUDE, TEMPERATURE AND WEIGHT

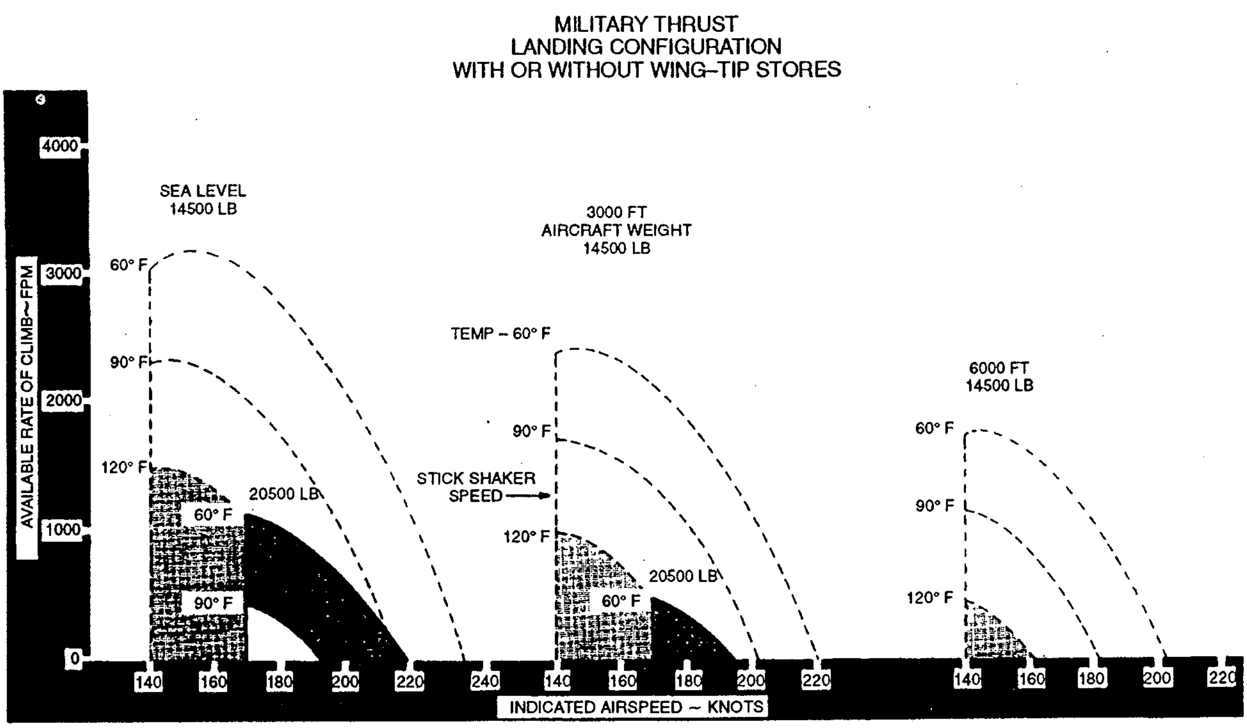

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Figure 6-9

LEVEL FLIGHT CHARACTERISTICS

LOW SPEEDS

Low-speed handling characteristics are good. Longitudinal and lateral control are positive and effective down to the minimum usable speeds. Sideslip may be used as desired. The rudder should not be used to pick up a low wing. Due to the high location of the rudder, the initial rolling moment is in a direction to lower the low wing further.

CRUISE AND TRANSONIC SPEEDS

At cruise speeds and in the transonic speed range, excellent stability and control characteristics are exhibited. There are no tendencies for wing drop or any significant trim changes when operating in or passing through this speed range. Longitudinal stability is positive except for a mild nose-down trim change experienced around Mach 0.88 to 0.90. This is so slight that it will be barely perceptible. At altitudes above 25000 feet, an undamped lateral – directional oscillation of approximately ½ cycle per second frequency may occur in the speed range from Mach 0.94 to 0.98. The oscillation is due to formation of unstable shock patterns on the empennage at these speeds.
Increasing or decreasing speed to values outside the Mach 0.94 to 0.98 range will eliminate the oscillation. In the 450 to 550-knot speed region below 25000 feet small amplitude residual oscillations of approximately 1 cycle per second frequency may be experienced on some aircraft. This oscillation is much smaller than that experienced at Mach 0.94 to 0.98 and in most aircraft it will be barely perceptible. This characteristic is considered normal although it is evidence of less than optimum operation of the yaw stability augmenter in the individual aircraft.

NOTE

During flight, a cockpit pressure drop occurs between 0.93 and 1.0 Mach. This can result in a cockpit altitude increase as high as 2000 feet and is caused by shockwaves passing over the static sensing ports during transonic flight.

SUPERSONIC FLIGHT

As speed is increased from Mach 1.05, a slight nose-down trim change will be noticed up to approximately Mach 1.5. The normal nose-up trim change is experienced with further increase in speed. Although there is this variation in trim over the supersonic speed range, the magnitude of the change is very small as indicated in Figure 6-10.
In addition, at any speed in this range positive maneuvering stability will be evident in that a back pressure on the stick control is always necessary to increase the normal acceleration, and vice versa (Figure 6-13). A directional trim change which requires increasing trim with increasing Mach number may be experienced or detected on some aircrafts. Generally, the sideslip is to the left, giving left ball displacement. In most cases, the directional trim is adequate; however, some cases may be experienced in which a small ball displacement will persist near design speed with full trim applied.

NOTE

If yaw oscillations with ball displacement of more than one-fourth ball occur above Mach 1.6, discontinue further acceleration to prevent a possible engine compressor stall.

PITCH TRIM CHANGES

NO EXTERNAL STORES

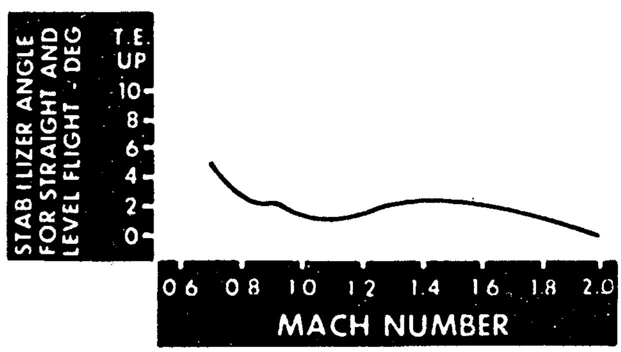

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Figure 6-10

LEVEL FLIGHT SPEED ACCELERATION

Operation of this aircraft in the supersonic region requires an acceleration from subsonic speed to the desired supersonic speed. This acceleration is a component of any mission utilizing the high-speed and high-altitude capabilities of this aircraft.
Figure 6-11 shows the difference in time, fuel, and distance requirements to accelerate at various altitudes. The variation with altitude is emphasized in Figure 6-12 which shows the optimum altitude for the minimum fuel is 35000 feet. This will vary somewhat with the existing atmospheric temperature but usually occurs in the 35000 to 40000-foot altitude range. This is not intended to imply that this range is the only possible acceleration altitude but to emphasize that the best time, fuel, and distance capabilities are available in this range.

LEVEL FLIGHT ACCELERATION

NO EXTERNAL STORES
16500 POUNDS - GROSS WEIGHT

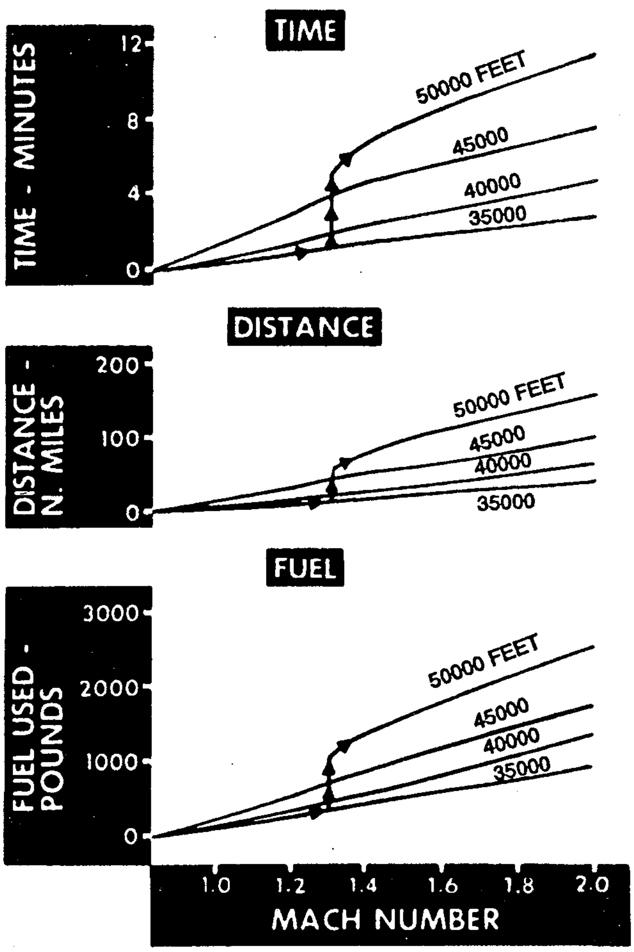

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Figure 6-11

BEST ACCELERATING ALTITUDE

NO EXTERNAL STORES

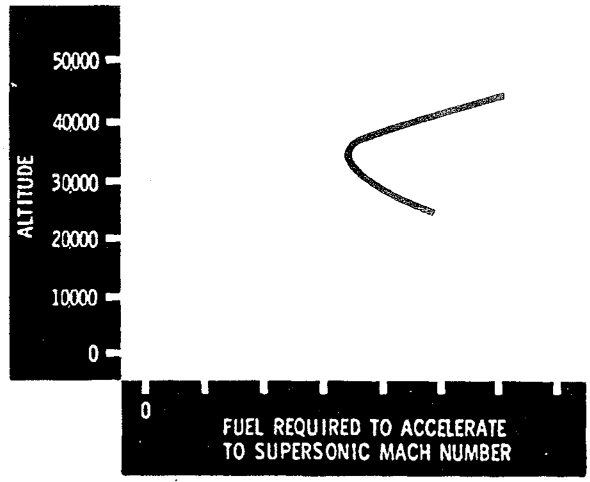

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Figure 6-12

AILERON ROLLS

Aileron roll characteristics within the prescribed limits are excellent. Because of inertial coupling tendencies peculiar to all supersonic configurations, certain restrictions on rolling maneuvers have been imposed to prevent loss of control and/or exceeding structural limits.
These inertial coupling tendencies result from the large pitch and yaw inertia of the supersonic configurations and the angular relationship of the longitudinal inertia axis of the aircraft with the axis of rotation. This angular relationship varies with flight conditions (speed, altitude and load factor). Depending on whether the inertia axis is nose-up or nose-down relative to the flight path or rolling axis, a sideslip is developed during a roll which produces inhibiting or augmenting roll rate.
In extreme cases, if sufficient aileron control power is allowed, this divergent type roll rate due to sideslip can result in excessive rolling velocity, uncontrollable rolls or structural damage and failure. In order to remain within safe limits, the available aileron travel is reduced with gear up to prevent high roll rates. In addition, the amount of continuous rotation is placarded in all configurations to prevent large amounts of sideslip from developing.

MANEUVERING FLIGHT

Maneuvering stick forces are moderate and generally unaffected by Mach number at a given altitude Figure 6-13. Forces will, however, vary with altitude, increasing as altitude is increased.
This is because forces vary with the amount of stabilizer deflection and bob-weight effect, and are not dependent on airloads at the control surface as in the case in unboosted or semiboosted control systems.
As a result, the aircraft will feel more stable and be more comfortable in maneuvers at high altitude where greater stabilizer deflection is required. In turning flight, adverse sideslip is not experienced, and it is not necessary to use the rudder above 300 KIAS.

MANEUVERING STICK FORCES

NO EXTERNAL STORES

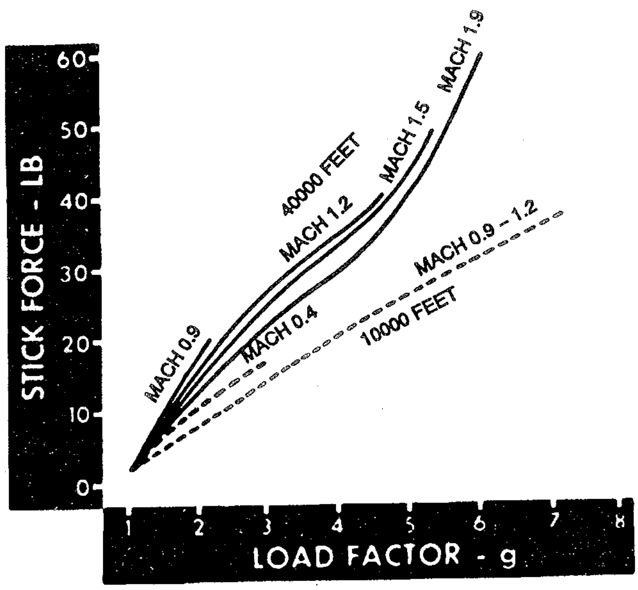

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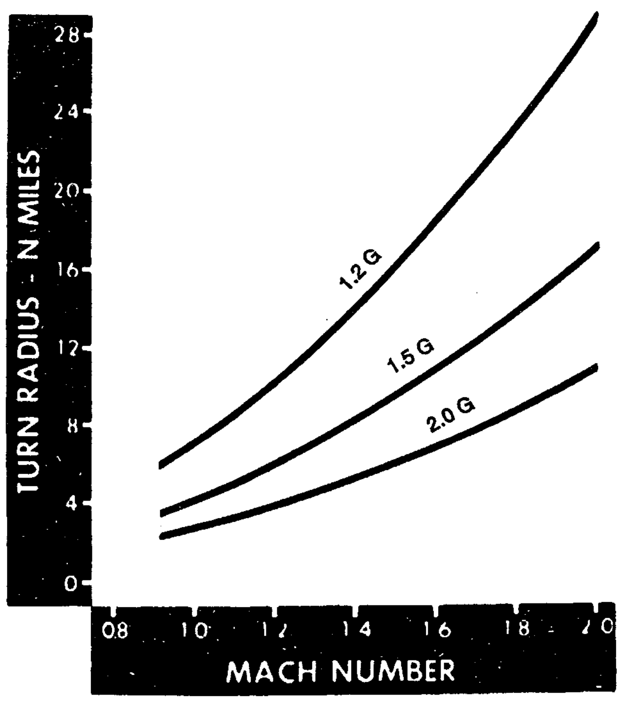
Figure 6-13

TURNING PERFORMANCE

This aircraft has an unusually large amount of excess thrust under normal operating conditions, which gives it excellent constant-speed maneuverability. This maneuverability is somewhat disguised by the relatively large turn radius incurred in high-speed flight.
Figure 6-14 shows this normal increase in turn radius due to Mach number.
Figure 6-15 and Figure 6-16 show the maneuvering capabilities of the aircraft without a speed loss, for various altitudes in terms of both load factor and turn radius.
Notice that at any altitude the available constant speed G's increase with Mach number and are a maximum of 2.0 "G" where excess thrust is greatest. Minimum turn radius, however, is the result of the combined effect of speed and available constant-speed load factor and varies, therefore, considerably with flight condition. Figure 6-15 also illustrates the constant Mach number and constant load factor capability in a climbing turn.
For example, it can be seen that ample constant speed is available to permit a 1.5 "G" turn with sufficient excess thrust remaining to climb quite readily to approximately 50000 feet at which altitude there is only enough excess thrust to maintain the 1.5 "G" turn. Higher load factors than shown on these figures can be utilized to accomplish the desired maneuvering in decelerating flight.

EFFECT OF MACH ON TURN RADIUS

ABOVE 35000 FT



FA0205

Figure 6-14

MAXIMUM CONSTANT SPEED MANEUVER LOAD FACTOR

NO EXTERNAL STORES
16000 POUNDS GROSS WEIGHT

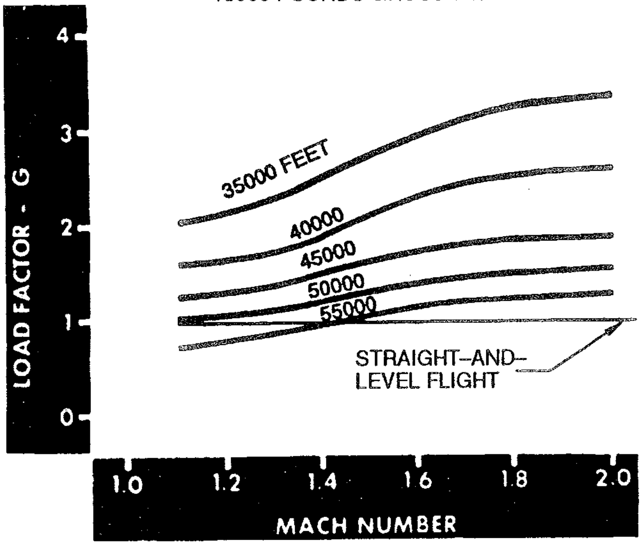


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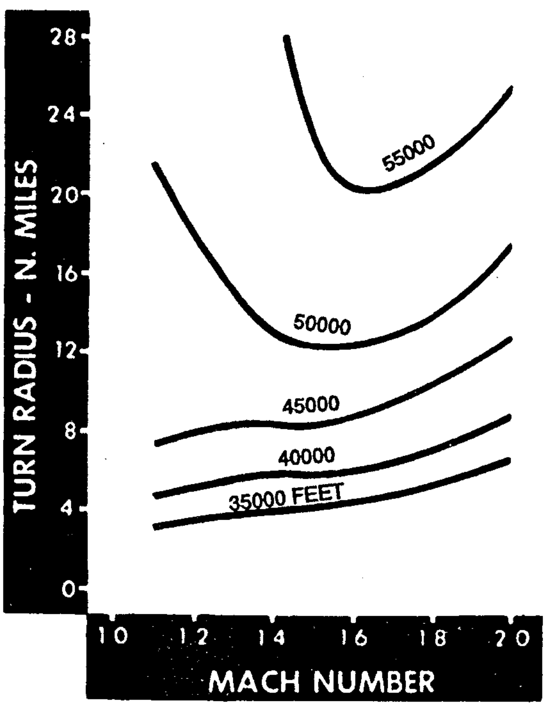
Figure 6-15

MINIMUM TURNING RADIUS AT CONSTANT SPEED

16000 POUNDS GROSS WEIGHT
NO EXTERNAL STORES



FA0207

Figure 6-16

DIVING

As the maximum allowable Mach number in dive and level flight are the same, all of the foregoing comments on stability and control cover the speed range encountered in dives.
This means that flight characteristics during dives at all speeds are excellent and no new or different trim changes will be experienced.
The stick forces remain at comfortable levels and the aircraft is easily controlled. Due to the high rates of descent that are possible, steep dives should be executed with caution. Altitude loss in dive recovery is shown in Figure 6-17 for any combination of speed, dive angle, and pullout load factor.
The altitude loss in recovery is the same for this aircraft for the same condition of speed, "G", dive angle, and altitude as for other aircraft; however, cognizance must be taken of the higher permissible speeds of this aircraft.
Because the maximum permissible speeds can be reached in level flight over a large portion of the altitude range, limit speeds can be reached quite readily; therefore, care should be exercised in diving flight to prevent exceeding limits.

LOW ALTITUDE DIVE RECOVERY

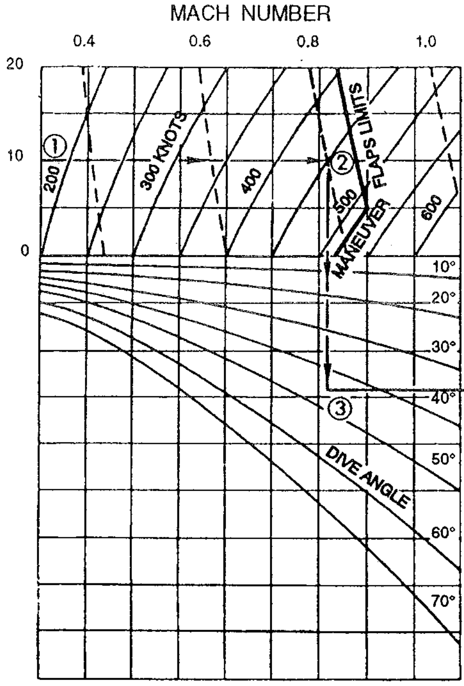
At airspeeds in the 1.1 to 1.4 Mach number range, the aerodynamic shift in stabilizer loading results in a reduction in maximum available stabilizer deflection due to control actuator force capability. In low altitude dive recovery, this maximum available stabilizer hinge moment reduces the pullout load factor capability to less than the structural limit.
Figure 6-18 shows the load factor capability as affected by maximum available stabilizer hinge moment. Aircraft gross weight and forward center of gravity also affect the available load factor. In addition, the stabilizing effect of tip tanks results in a small reduction in available load factor.
Therefore, the data are for the most critical combination for the aircraft likely to be encountered in practical operations.
Figure 6-19 shows the altitude required for recovery from a 45° dive angle at 15000 feet that takes this reduced capability into account. The altitude loss increases with speed and with the power setting used during recovery.

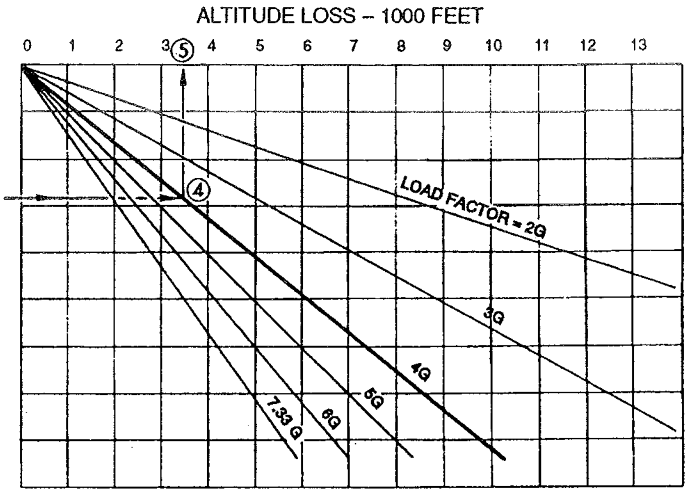
ALTITUDE LOSS IN DIVE RECOVERY

SEA LEVEL TO 20000 FEET

EXAMPLE:
REFER TO FIGURE 5–5 THROUGH 5–7 AND 6–19 WHETHER THE NUMBER OF G'S REQUIRED FOR PULLOUT ARE AVAILABLE.

1 ENTER CHART AT ALTITUDE AT START OF PULLOUT (10000 FEET)
2 MOVE RIGHT AT CONSTANT ALTITUDE TO AIRSPEED AT WHICH PULLOUT IS STARTED (MACH 0.8 450 KNOTS IAS)
3 MOVE VERTICALLY DOWN TO DIVE ANGLE (45°)
4 MOVE TO RIGHT TO PULLOUT LOAD FACTOR (4G)
5 READ ALTITUDE LOST DURING PULLOUT (3500 FEET)

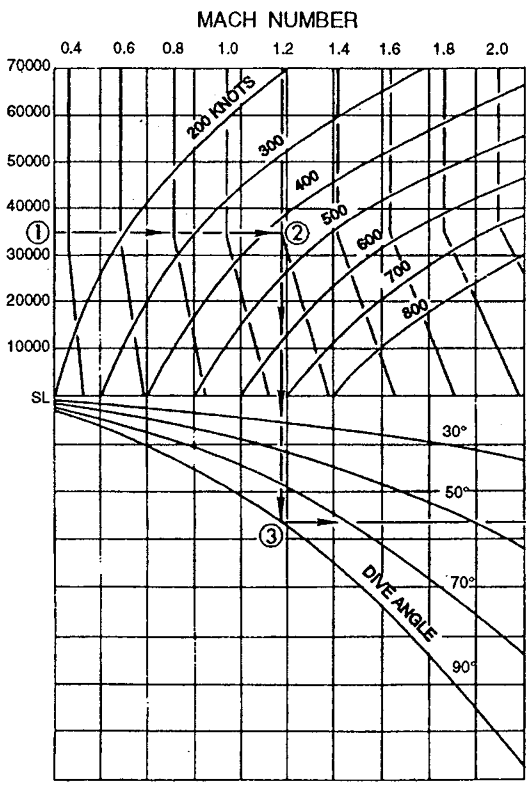


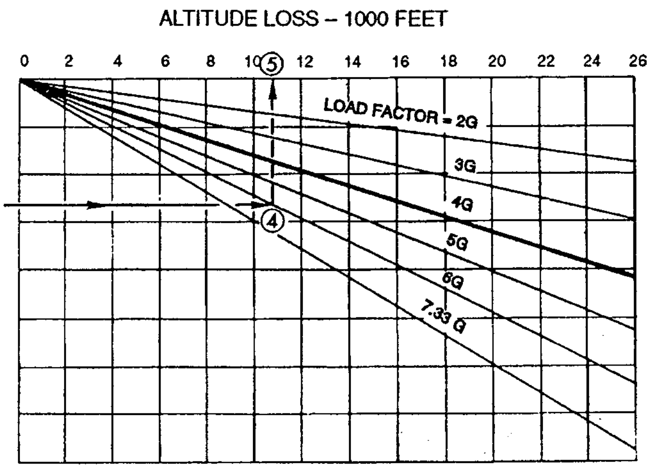


SEA LEVEL TO 70000 FEET

EXAMPLE:
REFER TO FIGURE 5–5 THROUGH 5–7 AND 6–19 WHETHER THE NUMBER OF G'S REQUIRED FOR PULLOUT ARE AVAILABLE.

1 ENTER CHART AT ALTITUDE AT START OF PULLOUT (35000 FEET)
2 MOVE RIGHT AT CONSTANT ALTITUDE TO AIRSPEED AT WHICH PULLOUT IS STARTED (MACH 1.2 430 KNOTS IAS)
3 MOVE VERTICALLY DOWN TO POINT ON CURVE OF DIVE ANGLE (90°)
4 MOVE TO RIGHT TO PULLOUT LOAD FACTOR (6G)
5 READ ALTITUDE LOST DURING PULLOUT (10800 FEET)





FA0212

Figure 6-17

EFFECT OF MAXIMUM AVAILABLE STABILIZER HINGE MOMENT ON LOAD FACTOR CAPABILITY

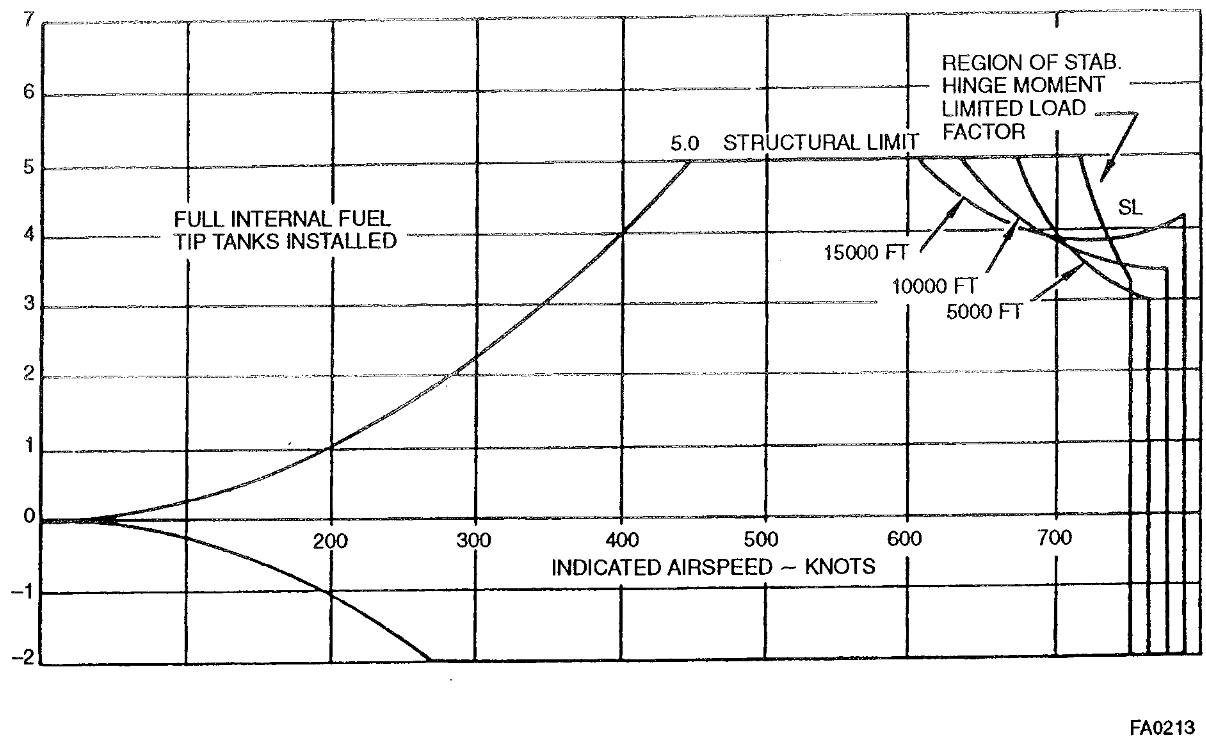


Figure 6-18

Figure 6-20 shows the speed-altitude variation during typical dive recoveries. These curves illustrate or emphasize that at the higher speeds, power must be reduced and speedbrakes extended in order to prevent exceeding limit airspeed during recovery. The altitude loss lines in Figure 6-19 are terminated at the maximum speed at 15000 feet from which recovery can be accomplished without exceeding limit airspeed.
For example, 670 KIAS is the maximum speed at 15000 feet from which recovery can be accomplished without exceeding the 750 KIAS engine limit unless the power setting is reduced below maximum thrust. Obviously, shallower dive angles would result in improved recovery limitation, proper operation of the hydraulic systems should be assured by close monitoring particularly when increasing dive angle and airspeed.
Refer to Section III for recovery capability with only one hydraulic system operating.

FLIGHT WITH EXTERNAL STORES

WARNING

- DO NOT FLY WITH MODIFIED TIP STORES UNLESS THEY MEET THE REQUIREMENTS OF STANDARD TIP STORES AS TO CONFIGURATION AND CENTER OF GRAVITY.

45° DIVE RECOVERY

FULL INTERNAL FUEL – TIP TANKS INSTALLED

RECOVERY STARTED AT 15000 FT

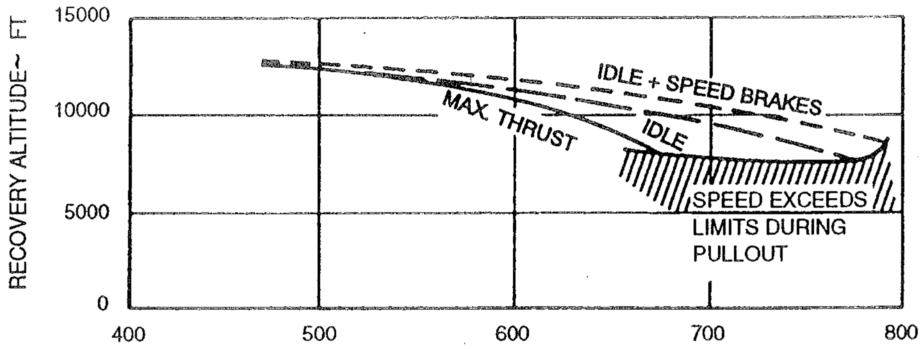


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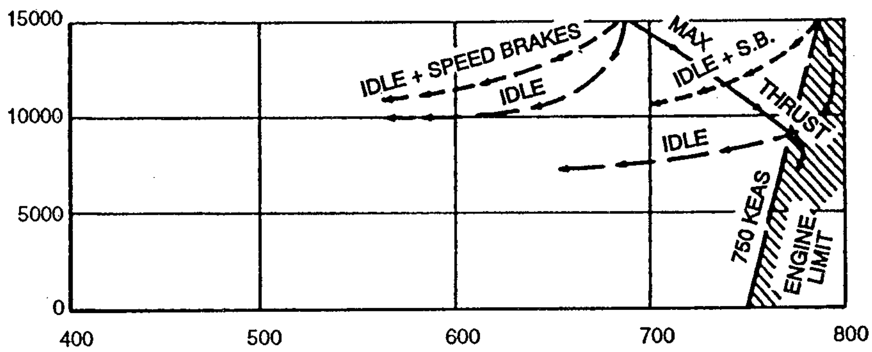
Figure 6-19

SPEED-ALTITUDE PROFILE DURING 45° DIVE RECOVERY

RECOVERY STARTED AT 15000 FT



FA0215

Figure 6-20

- THE ROLL STABILITY AUGMENTER SHOULD BE TURNED OFF BEFORE REACHING 575 KIAS WITH ANY WING STORE INSTALLED ON THE AIRCRAFT. WITH THE ABOVE STORES INSTALLED AND THE ROLL STABILITY AUGMENTER OPERATING, WING TORSIONAL OSCILLATIONS SUFFICIENT TO CAUSE STRUCTURAL DAMAGE MAY BE EXPERIENCED AT HIGH INDICATED AIRSPEEDS.

Flight Characteristics with external stores installed are essentially the same as without external stores. Naturally, performance is degraded in proportion to the increased weight and drag of the individual stores. In rough air, it is normal for the pitching motion of the tip tanks to be large enough to be both seen and felt.

PILOT INDUCED OSCILLATIONS

The F-104 is not prone to pilot induced oscillations; however, rough applications of pitch control can induce some oscillations in pitch. If oscillations are encountered, releasing the stick will allow the inherent aircraft stability to dampen the oscillation. Do not attempt to dampen the oscillations with further pitch inputs.

TIP TANK OSCILLATIONS

Flight tests and operational experience have shown that, under certain conditions, a steady and continued oscillation of the tip tanks can be encountered. This characteristic is usually the result of a high yaw damper setting, but it may also occur if there is a malfunction in the yaw or roll damper. Therefore, if an oscillation occurs, both the yaw and roll dampers should be deactivated to assure elimination of the condition.
Deactivating the pitch damper has no significant effect. The oscillation is most likely to be encountered below 5000 feet at approximately 0.80 Mach number with 400-800 pounds of fuel in each tip tank. It can be mild or severe; however, it is not of sufficient magnitude to cause structural failure. The nose of the tip tank moves up and down at about 4 cycles per second. The aircraft response is noted as a lateral-directional oscillation at the same frequency and is most predominate in yaw. For severe cases, cockpit lateral accelerations may approach ½ "G".
Once the oscillation has started, the intensity can be aggravated by aileron inputs. These inputs can be pilot induced or the result of a defective roll damper. Pilot inputs are inadvertent and result from cockpit side motion experience during the oscillation. They can be minimized by attempting to hold the stick in a fixed position to prevent lateral stick movement. Aileron inputs resulting from a defective roll damper can be considerably larger than those induced by the pilot. Accordingly, the intensity of the oscillation will be more severe.
Generally, oscillation initially encountered in 1 "G" flight will not be aggravated by increase of load factor. Also, if an oscillation is encountered in accelerated flight, the severity will probably be reduced by decreasing load factor. Speed reduction may also be helpful in eliminating a tip tank oscillation. This may be accomplished through the use of throttle and speed brakes.

FLIGHT WITH ASYMMETRICAL LOAD

ASYMMETRIC EXTERNAL TANK FUEL LOAD

Adequate control is available under smooth air conditions for landing with one tip or pylon tank full and one empty. Lateral control can be improved by using TAKEOFF flaps. Consideration should be given to the added aileron requirements under strong or gusty crosswind conditions before attempting to land with an asymmetric fuel load. A crosswind from the side with the light tank increases the aileron requirements in the same direction as used to balance the heavy tank. It is recommended that low-speed control be evaluated prior to entering the landing pattern. If the lateral control appears marginal for the existing landing condition, the tanks should be jettisoned.
Aileron requirements to balance an asymmetric wing loading increase as speed is increased above Mach 1.0 and as altitude is increased. At approximately Mach 1.2 at altitudes above 35000 feet, the available lateral trim is insufficient to maintain wings level flight with 750 lb or more fuel in one tip tank.
In addition, increasing aircraft load factor increases the aileron requirements and lateral control will be insufficient to control bank angle at load factors in excess of 2 "G". Reducing speed to Mach 0.9 and decreasing altitude to below 35000 feet will reduce

the aileron requirements to within lateral trim capacity.
High "G"-loads (more than + 3 "G") may cause the aircraft to roll into the direction of the full tip tank; at low altitude it will become impossible to recover the aircraft safely. Takeoff flaps will decrease the possibility to counteract the rolling movement into the heavy wing. As airspeed decreases, the 3 "G" limit will also decrease.

NOTE

Due to fuel management procedures initiated to prevent overpressurization in the internal fuel system, it is possible that a speed of Mach 1.2 to 1.3 may be attained before tip tank fuel is selected. If this speed occurs before the tip tank fuel is selected and one tip tank fails to feed, an asymmetric tip tank fuel condition can occur wherein one tank is empty and one full approximately 6 minutes after tip tank fuel is selected. This can result in an asymmetric wing load of 1000 pounds.

WARNING

- Do not exceed Mach 0.9 if an asymmetric external tank fuel load exists.
- If asymmetric external fuel load exceeds 450 lb proceed as follows:

  1. Do not exceed 40° bank/2.0 "G"
  2. Avoid abrupt maneuvers
  3. Land according to the prescribed procedures in Section III
  4. Jettison external tanks if aircraft control becomes marginal.

FORMATION FLYING

Because of the rapid aircraft acceleration characteristics and the segmented burning characteristics of the afterburner, a pilot of low proficiency may be quite busy while flying formation; however, after a learning period of two or three flights, a good close formation takeoff and flight can be made.

AFTERBURNER FORMATION TAKEOFFS

Due to the rapid acceleration of the aircraft, it is necessary for the leader to use a throttle setting somewhat less than full throttle to allow the wingman to maintain his position. This should be an indicated nozzle position of 8 minimum.
Large variations in thrust with a relatively small throttle movement are obtained when operating with the afterburner. Because of the brief time involved in takeoff, the wingman may find some difficulty in stabilizing his position. The wingman should become airborne with the leader; takeoff flaps should be retracted on a signal from the leader. If afterburner light-off is accomplished simultaneously between the two aircraft and throttle adjustments made immediately, the wingman will be able to maintain a satisfactory position throughout takeoff with small throttle changes.

AFTERBURNER FORMATION CLIMB

If the wingman is in formation at takeoff, he can easily maintain position throughout a climb with afterburner to subsonic ceiling. The leader must select a setting of less than full throttle to allow the wingman to make throttle adjustments.
Formation join-up from individual takeoffs can be made if the leader turns after takeoff and reduces thrust. A leader who maintains an excessively high throttle setting probably will reach cruise altitude before his wingman can reach formation climb position.

SUPERSONIC FORMATIONS

In close and combat formation at speeds up to Mach 0.9 no critical or safety-of-flight regions of operation are involved. However, as the aircraft reaches supersonic flight, shock waves begin to form. These shock waves attach themselves to such places as the nose, wing, ducts, and the tail. The bow wave and the tail shock wave are parallel and effectively form one wide, disturbed area. This shock wave pattern bends back as the Mach number is increased. If a constant normal formation position is held, the shock wave will begin to envelop the wing aircraft as the speed is increased. When this occurs, the reaction is much the same as if the aircraft had flown through a jet wake.

Yaw and rolling disturbances will develop as the aircraft passes in and out of the shock wave. If the wing aircraft passes completely through the wave on a course parallel to and level with the lead aircraft, the passing aircraft will experience one yaw cycle. As the nose of the wing aircraft enters the wave, the aircraft will yaw toward the leader. As the aircraft becomes enveloped by the wave, it will yaw away; and as it leaves the wave, it will again yaw in, completing one oscillation.

If the overtake rate is in the vicinity of Mach 0.01 or 0.02 per second or if the passage through the wave lasts approximately 2 seconds, there is a possibility of structural damage due to high sideslip angles. If the wing aircraft passes directly above or below the lead aircraft, pitching moments instead of yawing moments will be experienced.

The first movement will be toward the lead aircraft, then away, then back again as passage through the flow field is completed. If the wingman remains behind the shock cone, no difficulty will be experienced. Staying behind the shock cone, may be accomplished by dropping back as Mach number is increased. When flying tactical formation at supersonic speeds, the wing aircraft may receive random actuations of the stick shaker and kicker while crossing through the wave. If kicker operation is experienced, the wing aircraft will fall behind, but that is all.

APC METER

The APC meter is inadequate as a primary flight instrument, however, service experience indicates it is useful as a guide in setting up certain flight conditions at speeds below 0.85 Mach number. Typical flight conditions for which it can be used are speeds for minimum drag and best cruise, and a qualitative indication of the maneuvering margin before APC actuation occurs. Nominal meter readings are shown on Figure 6-21 for various airspeeds and gross weights in steady state flight. Non-steady state conditions can cause higher meter readings depending on pitch rate.

APC meter reading is affected by flap position and wing tip configurations. This effect varies with angle-of-attack, therefore, it is not constant over the full scale range of the APC meter. In the range where most flight operations are conducted, 0 to 1 on the meter, the extension of takeoff flaps will decrease the APC meter reading about 0.6 units. Meter readings with tip tanks will be about 0.5 units lower than for no tip stores for comparable flight conditions.

APPROXIMATE APC METER READING IN STEADY STATE FLIGHT

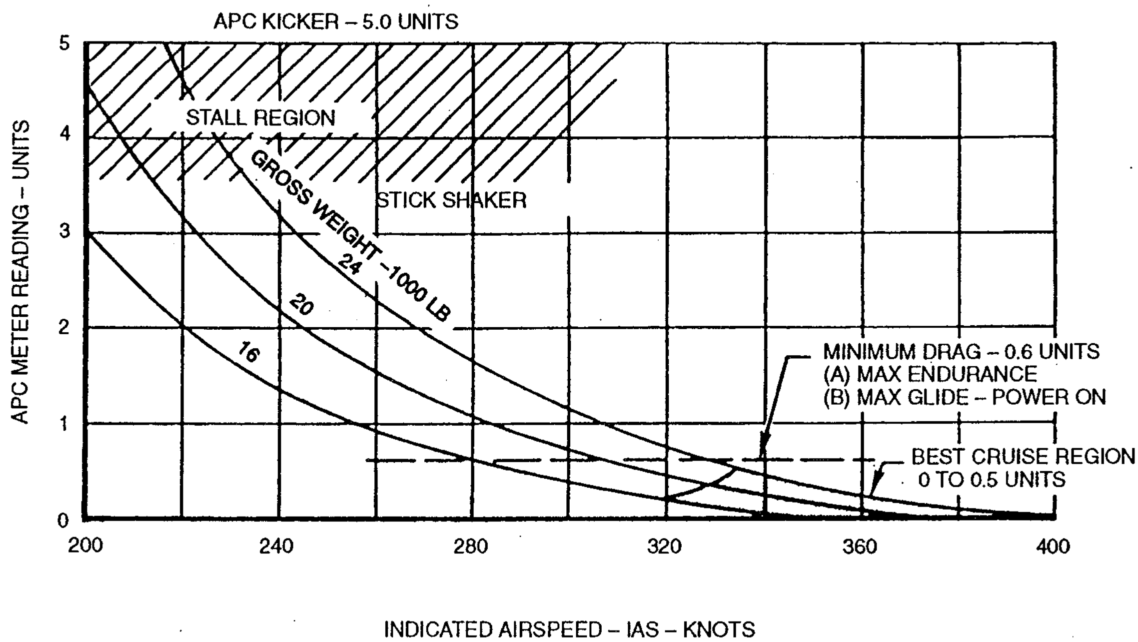
APPROXIMATE APC METER READING IN STEADY STATE FLIGHT

TIP TANKS FLAPS UP

CHART NOT VALID ABOVE 0.85 MACH

NOTES:

1 CHART VALID UP TO 35000 FT
2 WITHOUT TIP TANKS, CHART VALUES SHOULD BE INCREASED A NOMINAL 0.5 UNITS
3 TAKEOFF FLAPS DECREASE METER READING A NOMINAL 0.6 UNITS IN THE 0 TO 1 RANGE
4 METER READINGS CAN VARY AT LEAST 0.25 UNIT AMONG AIRCRAFT



FA0216

Figure 6-21

SECTION VII

ALL WEATHER OPERATION

TABLE OF CONTENTS

INSTRUMENT FLIGHT PROCEDURES

The following procedures and techniques pertain primarily to instrument flight conditions and are in addition to normal procedures. They are based on normal gross weight, clean configuration, and standard day conditions. Because navigation facilities and terrain features are different at each base, this information is intended to serve as a guide to commanders in establishing their own instrument flight procedures.

INSTRUMENT TAKEOFF

An instrument takeoff is essentially the same as a normal VFR takeoff and should be made at maximum thrust.
The pitot-pitch and temperature probe heater switch shall be set to PITOT-PITCH TEMP PROBE prior to takeoff when anticipating an IFR climb-out.

CAUTION

- DO NOT SET THE PITOT-PITCH AND TEMPERATURE PROBE HEATERS SWITCH TO PITOT-PITCH TEMP PROBE POSITION FOR MORE THAN FOUR MINUTES CONTINUOUSLY WHILE THE AIRCRAFT IS ON THE GROUND: OVERHEATING MAY CAUSE THE ELECTRICAL HEATING ELEMENTS TO FAIL.
- DO NOT USE THE RAIN REMOVER FOR TAKEOFF. HIGH POWER SETTINGS AND LOW AIR-SPEED MAY RESULT IN WINDSHIELD DAMAGE.

INSTRUMENT CLIMB

1. Hold 5° to 8° nose-high pitch attitude until climb airspeed is attained.
2. Adjust pitch attitude to maintain desired climb speed. For afterburner climbs, pitch attitude will vary between 25° and 38° nose-up on attitude indicator.

WARNING

WHEN TURN ARE MADE DURING AFTERBURNER CLIMB, THE NOSE WILL TEND TO SCRIBE AN ARC WHICH WILL NOT BE PARALLEL TO THE HORIZON. ALSO, THE PITCH ATTITUDE OF THE AIRCRAFT WILL INCREASE. THIS CONDITION TENDS TO INDUCE VERTIGO; THEREFORE, WHEN MAKING AN AFTERBURNER INSTRUMENT CLIMB, TURNS SHOULD BE KEPT TO AN ABSOLUTE MINIMUM AND SHOULD BE COMPLETED PRIOR TO ACHIEVING CLIMB SPEED AND ATTITUDE.

INSTRUMENT CRUISING FLIGHT

Handling characteristics are such that supersonic instrument cruise flight is possible. Refer to Section V for level-flight characteristics at high speeds. For ease and precision of flight, angles of bank exceeding 30° should be avoided whenever possible.

NOTE

Except a momentary opposite bank indication on the turn-and-slip indicator when starting a turn.

HOLDING

TAKEOFF flaps and 260 KIAS is recommended for holding; however, depending on configuration or formation a speed of 300 KIAS may be necessary. Fuel consumption is as shown in the following table.

Altitude	Approx. Fuel Flow	Approx. RPM
30000 ft	2400 lb/hr	90%
20000 ft	2600 lb/hr	80%

NOTE

Bank angle should not exceed 30° and thrust has to be increased when making level turns to maintain desired airspeed.

JET PENETRATION

Engine and airframe considerations when performing descents include cognizance of inlet guide vane icing. Icing of the inlet guide vanes is most probable at 82% RPM or below, due to inadequate airflow for anti-icing. A minimum of 85% RPM should be maintained when descending in possible icing conditions.

PENETRATION PROCEDURE

Prior to entering the holding pattern or when approaching the initial approach fix, decrease IAS to 260 knots or 300 knots if configuration or formation requires and lower flaps to TAKEOFF. If holding pattern is not to be entered, approach fix at 300 knots with takeoff flaps.

1. Canopy defogger rheostat – As required
2. Pitot, pitch and temperature probe heat switch – ON
3. Engine/duct anti-ice switch – As required

When starting descent:

1. Pitch – Lower nose to establish 300 KIAS
2. Power – Reduce to 85% RPM
3. Speed brakes – As required

NOTE

The average rate of descent will be approximately 3500 feet per minute with speed brakes in and 8500 feet per minute with speed brakes extended. With TAKEOFF flaps and 85 to 88% RPM, the aircraft will stabilize at approximately 250 KIAS in level flight inbound to station.

When leveling off:

1. Pitch attitude – 2000 feet above level-off altitude, reduce rate of descent to approximately 2000 ft/min
2. Speed brakes – IN, if used, when approaching 250 KIAS
3. Power – Adjust to maintain 250 KIAS

When two nautical miles before glide path:

For straight-in approach, set up glide path configuration:

1. Landing gear lever – DOWN
2. Wing flaps lever – LAND
3. Altitude – Descent to and maintain minimum altitude
4. Airspeed – Maintain calculated final airspeed

RADAR APPROACH

1. Maintain 250 KIAS with flaps in takeoff position. Make all turns with 30° of bank.
2. Lower landing gear on base leg or 10 nautical miles out on final if making a straight-in approach. Allow airspeed to bleed off to 230 KIAS and adjust power to maintain this speed.
3. Lower flaps to LAND one nautical mile prior to glide path. Allow airspeed to decrease to calculated final airspeed and adjust power to maintain this airspeed.
4. When intercepting glide path, lower nose to establish calculated rate of descent. Maintain the final approach speed (for 2.5° glideslope and a ground speed of 180 knots the rate of descent will be approximately 800 feet per minute).

MISSED APPROACH

In case of a missed approach, follow the procedure given in Section II for a go-around and follow published missed approach procedure.

ICE AND RAIN

Although this aircraft does not have anti-icing systems for the wing and empennage, flight under icing conditions may be made.
Defogging, rain removal, engine/duct anti-icing, and pitot/pitch/temp/probe heat should be turned on prior to entering an area where icing conditions prevail or are anticipated.
Continued use of the rain-remover system will cause discoloration of the windshield adjacent to the rain-remover nozzle. Successive flights through rain at high speeds may cause erosion of the radome. Therefore, observe the following:

1. Whenever possible, avoid flight in conditions conducive to rapid buildup of ice.
2. Inspect windshield for discoloration after any flight in which the rain remover was used, and have windshield replaced if discoloration is noted.
3. Do not use rain remover for takeoff.

Normal Operation of Defogger and Rain-Remover System

If any portion of the windshield or canopy becomes obscured by moisture, operate the following controls:

1. Canopy defogger knob – FULL
2. Cockpit temp mode selector switch – AUTO
3. Cockpit heat rheostat – HOT
4. Rain-remover switch – RAIN REMOVER (If precipitation obscures forward visibility).

NOTE

Canopy defogging air should be used at the highest possible temperature consistent with pilot comfort. This will minimize the possibility of widshield and canopy fogging caused by extreme temperature differentials accompanying an engine failure or a rapid descent from altitude.

Emergency Operation of Defogger and Rain-Remover Systems

If the windshield cannot be cleared by normal procedures and it is necessary to land without delay, do the following:

1. Canopy defogger knob – CHECK (FULL position)
2. Cockpit temp mode selector switch – MANUAL
3. Cockpit heat rheostat – HOT
4. Rain-remover switch – RAIN REMOVER
5. Engine RPM – MAXIMUM, if fuel and time permit

The above procedure will direct compressor air to the windshield outlets at its maximum available temperature and pressure.

NOTE

If excessive fog, vapor, or visible moisture of any kind enters the cockpit, restricting visibility on takeoff, open the fresh-air scoop.

ENGINE/DUCT ANTI-ICING SYSTEM OPERATION

Icing will occur on the inlet ducts and engine compressor front frame at subsonic speeds only. Ram air temperature rise at supersonic speeds is sufficient to prevent icing.
If the engine is operated above 82% RPM, the anti-icing air temperature is sufficient to prevent rapid ice buildup on the engine front frame and inlet guide vanes.
The engine may safely ingest inlet duct ice at engine speeds less than 88% RPM. At higher engine speeds, the inlet guide vanes may be damaged.
Engine operation is still possible with limited inlet guide vane damage. The requirement for engine anti-icing is a direct function of indicated compressor inlet temperature (CIT).

NOTE

Operation of the anti-icing system at a CIT above 10° C degrades the service life expectancy of the magnesium front frame. If weather conditions indicate a need for anti-icing, the system should be actuated only at a CIT indication of 10° C or below.

Refer also to Section I and V for procedures and operating limitations of the anti-icing and rain-removing equipment.

MODIFIED APPROACH PROCEDURES UNDER KNOWN ICING CONDITIONS WITHOUT ANTI ICING EQUIPMENT

When operational necessity dictates an approach and landing under confirmed icing conditions not exceeding "moderate to heavy", the following procedure for single operating aircraft only should be executed to keep exposure time to an absolute minimum.

NOTE

Normal jet penetration has to be made with a minimum engine RPM of 85%.

When starting descent:

1. Power – Reduce to 85% minimum
2. Pitch – Lower nose to establish 325 KIAS
3. Speedbrakes – As required, when approaching 325 KIAS by adjusting pitch attitude to maintain close to desired instrument rates of descent

When leveling off:

1. Pitch attitude – 3000 feet above level-off altitude, reduce rate of descent to 2000 ft/min
2. Speedbrakes – IN, if used
3. Power – Adjust to maintain 325 KIAS (engine limitations under icing conditions should be observed)

NOTE

- If icing altitude region is known, approach altitude should be chosen accordingly. Deviations to the high side of published approach altitude should not exceed 1000 feet.
- Thoroughly cold-soaked aircraft should, fuel permitting, delay final approach for a minimum of 4 minutes to take advantage of heating by higher ambient temperature. Maintain 325 KIAS.

When three nautical miles before glide path intercept:

Set up glide path configuration in shortest possible time.

1. Speedbrakes – OUT
2. Power – Reduce not below 82% RPM
3. Descent to published altitude, if necessary
4. Landing gear lever – Extend 260 KIAS (maximum)
5. Wing flaps lever – LAND at 240 KIAS (maximum)
6. Rain Removal – ON, if necessary
7. Final approach speed – Maintain as high as practicable for prevailing conditions to ensure safe landing

HYDROPLANING

Dynamic hydroplaning is a condition where the tires of the aircraft are separated from the runway surface by a fluid.
Under conditions of total dynamic hydroplaning, the hydrodynamic pressures between the tires and runway lift the tires off the runway to the extent that wheel rotation slows or actually stops. The major factors in determining when any aircraft will hydroplane are groundspeed, tire pressure, and depth of water on the surface. To a lesser degree, the surface texture, type of tire, and tire tread depth influence the total hydroplaning speed.
Total dynamic hydroplaning in this aircraft with recommended tire pressure and 0.1 inch or more of water or slush on the runway can be expected at the following approximate groundspeeds (KIAS):

	Takeoff Gross Weight	
	23000 lb (tip tanks)	26000 lb (tip and pylon tanks)
Main gear wheels	118	127
Nose wheel	129	132

These speeds will change as tire pressure is varied for takeoff gross weight. Partial dynamic hydroplaning occurs to varying degrees below these speeds. When an aircraft is subjected to hydroplaning to any degree, directional control becomes difficult.
Under total dynamic hydroplaning conditions nosewheel steering is ineffective and wheel braking is nonexistent. In addition to dynamic, two other types of hydroplaning can occur. Viscous hydroplaning can occur on a damp runway and at speeds less than those associated with dynamic hydroplaning, and is caused by a thinfilm of water mixed with tubber deposits and/or dust.
Reverted rubber hydroplaning is caused by a skid which boils the water on the runway, causing the rubber to revert to its natural latex state and seals the tire grooves, delaying water dispersal. Reverted rubber hydroplaning can occur at very low airspeeds.
When possible hydroplaning conditions exist, pilots should be aware of the following:

1. Smooth tires tend to hydroplane with as little as 0.08 inch of water. New tires tend to release hydro-dynamic pressures and will require in excess of 0.2 inches of water depth to hydroplane.
2. Takeoffs with crosswinds on water covered runways should be made with caution. An aborted takeoff on a wet runway initiated at or near hydroplaning speed will require considerably more runway than a dry runway abort and directional control of the airplane will be critical until the speed has decreased below hydroplaning velocity.
3. In the absence of accurately measured runway water depths, pilots may use the following information to determine the possibility of hydroplaning when landing must be accomplished on a wet runway that does not have a porous surface or is not grooved:

    a. Rain reported as LIGHT – Dynamic hydroplaning unlikely, viscous and reverted rubber hydroplaning are possible.

b. Rain reported as MODERATE -- All types of hydroplaning are possible. Smooth tires will likely hydroplane; however, new tires are less likely to hydroplane.

c. Rain reported as HEAVY -- Hydroplaning will occur.

LANDING ON SLIPPERY RUNWAYS

Refer to the applicable Appendix for computing ground roll distance when landing on slippery runways.

WARNING

IF HYDROPLANING CONDITIONS EXIST THE LANDING ROLL WILL BE INCREASED AN INDETERMINATE AMOUNT, THEREFORE, BE PREPARED FOR A DEPARTURE END BARRIER ENGAGEMENT.

TURBULENCE AND THUNDERSTORMS

Flight through turbulence and thunderstorms may result in structural damage to the aircraft, or engine flameout (hail may cause rapid deterioration of the radome).
Engine flameouts have been experienced in jet aircraft which incorporate through-flow inlet systems, due singly or in combination to such factors as the following:

1. Penetration of cumulus buildups with associated high water content.

2. Icing of duct inlet or engine inlet guide vanes.

3. High concentration of ice crystals associated with tops of cumulus clouds.

4. Changes in engine inlet pressure associated with turbulent air penetration.

5. Operating above 40000 feet where the surge margin of the engine is reduced.

The last two factors are significant primarily at low indicated airspeeds. Operating in the sector range of afterburning also increases the possibility of flameout.
This is due to the greater sensitivity of the afterburner, in this range, to flow disturbances. The rapid closure of the exhaust nozzle when the afterburner blows out may in turn cause engine flameout when operating under the marginal conditions mentioned above.

CAUTION

FLYING IN TURBULENCE OR HAIL MAY CAUSE DUCT AIR INLET DISTORTION. AT HIGH ALTITUDES, THIS DISTORTION CAN RESULT IN ENGINE SURGE AND POSSIBLE FLAMEOUT; HOWEVER, NORMAL AIR RESTARTS MAY BE ACCOMPLISHED AS OUTLINED IN SECTION III.

Areas of turbulent air, hailstorms, or thunderstorms should be avoided whenever possible because of the increased danger of engine flameout.
If these areas may not be avoided, the following should be performed:

1. Establish a penetration airspeed of 350 KIAS for a clean aircraft or 275 knots if TAKEOFF flaps are extended. At altitudes where 350 KIAS is higher than normal cruise speed and would penalize range, use the best operating speed for performance. If a climb over the top is attempted, use the recommended climb Mach number to obtain best performance. Above 40000 feet, modify the climb schedule to maintain a minimum of 275 KIAS and use full afterburning in order to ensure adequate engine surge margin.

2. Turn on engine/duct anti-icing and pitot-pitch temp probe heat.

CAUTION

- THE WARNING PANEL LIGHTS ARE DIMMED AUTOMATICALLY WHEN THE INSTRUMENT LIGHTS ARE TURNED ON. SPECIAL CARE SHOULD BE EXERCISED, THEREFORE, TO DETECT ANY WARNING LIGHT ILLUMINATION.
- MONITOR ALL ENGINE INSTRUMENTS CONTINUOUSLY TO ENSURE TIMELY CORRECTIVE ACTION.

Refer to Ice and Rain paragraph in this section and Section I for operation and procedures of the anti-icing and rain-removing systems.

NIGHT FLYING

During night flights this airplane does not prescent any special problems or require any special techniques except during landing. During final approach at night, the runway lights are reflected in the windshield panels and disorientation may occur. Monitor attitude and flight path, ignoring the runway light reflection in the windshield.

COLD WEATHER OPERATIONS

The success of operations at low temperatures depends primarily upon the preparation made during the postflight inspection in anticipation of the requirements for operation on the following day. The procedures outlines should be followed during outdoor operation to expedite the preflight inspection and to ensure satisfactory operation of the aircraft and its systems during the next flight.

BEFORE ENTERING THE AIRCRAFT

1. Check the entire aircraft for freedom from snow and ice.

WARNING

DO NOT ATTEMPT TO TAKE OFF IF THERE IS ANY ICE, FROST, OR SNOW ON THE AIRCRAFT, BECAUSE ANY AMOUNT GREATLY REDUCES LIFT.

2. The pressure of both hydraulic accumulators will drop with temperature. A pressure as low as 700 psi may be expected at – 55° C.

BEFORE STARTING ENGINE

Make normal checks as outlined in Section II.

STARTING ENGINE

Make normal start as outlined in Section II.

NOTE

- During low ambient temperature engine starts, it is normal for engine oil pressure to be excessive for a short period of time. This, however, will not cause engine damage. Initial fuel flow indications are often delayed under the same conditions until after the engine reaches idle RPM, but the start may be readily monitoring using the EGT indicator.
- On icy surfaces the aircraft will slide forward at approximately 84-88% RPM with the brakes locked. Make certain the aircraft is clear before advancing the throttle.

WARMUP AND GROUND CHECK

Use normal procedure.

NOTE

It will require 2-4 minutes to obtain warm air from the defogger. If immediate defrosting or deicing is required, turn rain remover ON until the left windshield and canopy have cleared, then turn rain remover OFF.

1. Check hydraulic pressure, oil pressure and engine instruments. Allow oil pressure to decrease to operating limits before taxiing.
2. Check flight instruments.

WARNING

MAKE SURE ALL INSTRUMENTS HAVE WARMED UP SUFFICIENTLY TO ENSURE NORMAL OPERATION. CHECK FOR SLUGGISH INSTRUMENT INDICATIONS DURING TAXIING.

TAXIING INSTRUCTIONS

The aircraft can be successfully taxied in snow up to 6 inches deep.

1. Taxi at slow speed over rough snow-packed surfaces.
2. Allow more distance than on a cleared surface to bring the aircraft to a stop.

NOTE

Only nosewheel steering should be used in deep snow. Braking will cause the snow to melt and moisture to form on the wheels which may laer freeze.

BEFORE TAKEOFF

Make normal before-takeoff check as outlined in Section II.

NOTE

Canopy defogging air should be operated at highest temperature consistent with pilot comfort at all times. This will minimize the possibility of windshield and canopy fogging caused by extreme temperature differentials accompanying an engine failure or rapid descent from altitude.

TAKEOFF

Be prepared for an increase in takeoff distance if runway is covered with snow.

CAUTION

- ACCELERATION IS VERY RAPID IN COLD WEATHER. THE PILOT SHALL BE ASSURED THAT THE AIRCRAFT IS ROLLING STRAIGHT DOWN THE RUNWAY BEFORE APPLYING AFTERBURNER.
- TAKEOFF ON RUNWAY COVERED WITH HEAVY SLUSH SHOULD BE AVOIDED IN ORDER TO PREVENT DAMAGE CAUSED BY PROJECTION OF MELTING SNOW ON FUSELAGE AND IN WHEEL BAY.

AFTER TAKEOFF

Follow procedure as outlined in Section II. Climb performance will be improved during cold-weather operation at lower altitudes. Follow recommended climb speeds as given in climb charts.

ENGINE OPERATION IN FLIGHT

Engine operation during flight in cold weather should be governed by normal procedures.

LANDING

Use normal procedures. Refer to Landing on Slippery Runways paragraph in Section II.

STOPPING THE ENGINE

The engine is shut down in the normal manner.

BEFORE LEAVING THE AIRCRAFT

Use normal procedures.

HOT WEATHER OPERATIONS

NOTE

- The nozzle may sense an overtemperature condition at EGT's as low as 472° C. This may cause the nozzle to go to full open.
- Inlet temperatures above 52° C may cause engine to idle at higher than normal RPM.

TAKEOFF

Takeoff distances will be longer during high ambient temperatures. Check takeoff distances required for existing conditions by referring to takeoff charts in the Appendix.

CAUTION

IT IS IMPERATIVE THAT TAKEOFF BE MADE AT THE RECOMMENDED SPEED. MORE THAN THE USUAL TAKEOFF DISTANCE WILL BE REQUIRED TO OBTAIN TAKEOFF SPEED WHEN OUTSIDE AIR TEMPERATURE IS HIGH; THEREFORE, EXERCISE CAUTION AGAINST LIFTING OFF THE RUNWAY TOO SOON.

NOTE

During afterburner takeoff under high ambient temperature conditions, EGT limits may be exceeded. Adjust the throttle to maintain EGT within limits.

APPROACH AND LANDING

Monitor rate of descent closely during approach. Do not allow rate of descent to exceed the 700-800 feet per minute recommended during the final portion of the approach. Be prepared to use afterburner, if necessary. Refer to discussion in Section VI and the charts in the Appendix pertaining to variations in performance for changes in temperature, weight and altitude.

TABLE OF CONTENTS

APPENDIX - PERFORMANCE DATA

PART 1

INTRODUCTION

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

SCOPE AND ARRANGEMENT

This appendix contains performance data necessary for accurate preflight planning. A part-type arrangement groups the material as needed for planning general phases of each flight. Descriptive text in each part discusses and explains the use of the charts.

A mission-planning section at the end of the appendix shows how the individual performance charts for each part of a flight may be combined for overall planning purposes. High-speed maneuvering performance data are presented in Part 9. Symbols and definitions used throughout the appendix are listed and define on page A-2.

PERFORMANCE DATA BASIS

Flight planning information shown in this appendix is based on manufacturer flight-test data and is directly applicable to operation of TF-104G-M aircraft with the J79-GE-11B engine. The cruise performance has been corrected to include a 5% operational tolerance.

The material is presented for standard atmospheric conditions as defined by the Standard Altitude Table; however, corrections for nonstandard temperature conditions have been included on the charts where possible.

FUEL AND FUEL DENSITY

All performance and operating weight ranges included in this appendix are based on operation with JP-8 fuel at a nominal fuel density of 6.68 pounds per US gallon. If fuel density is known to be different from the nominal value, enter the performance charts with an aircraft weight corrected to reflect the actual weight of fuel on board.

CONFIGURATION DRAG INDEX

The performance charts include a configuration drag index. Figure A1-1 defines the configuration drag index of some of the operating configurations and provides a quick reference to the applicable charted value of index performance.

Figure A1-2 shows the incremental corrections to be applied to a known basic index (no external stores) to account for other configurations. Performance which is available by dropping external stores is not shown directly by special charts but may be determined by referring to the appropriate charts for the initial and subsequent configuration drag index and use the charts on an incremental basis.

CONFIGURATION DRAG INDEX – SAMPLE PROBLEM

Determine the configuration drag index and the loaded gross weight for a TF-104G-M aircraft equipped with wing-tip tanks and pylon fuel tanks. Internal fuel includes auxiliary fuel tanks.

a. Determine the total of store drag numbers for the configuration (Figure A1-2):

Wing tip tanks	16.0
Fuel tank pylons	6.0
Pylon fuel tanks	46.0
TOTAL (configuration drag index)	68.0

b. Determine the loaded gross weight:

No external stores, zero fuel weight	15056 lb
Internal fuel	4676 lb
Wing tip fuel tanks	2731 lb
Wing pylon tanks and fuel	3230 lb
LOADED GROSS WEIGHT	25693 lb

POSITION ERROR CORRECTIONS

The compensated pitot-static airspeed system is designed to minimize the corrections which are normally required for airspeed, mach number, and altimeter position errors.
The small variation of the system error at subsonic speeds is due to changes in aircraft attitude, which is dependent upon speed, altitude, aircraft weight, and wing-tip stores.
Calibration curves are provided for configurations without wing-tip stores, with tip tanks. Each of the curves shows the effect of speed, altitude, and gross weight. Stores at other stations do not alter aircraft attitude, except for the additional weight, and therefore do not require separate calibrations.

NOTE

The compensated pitot-static airspeed system position error is small and usually may be neglected so that KCAS = KIAS and True Mach number = Indicated Mach number.

MACH NUMBER CORRECTION

Indicated Mach number may be corrected for position error as follows: enter the appropriate wing-tip store configuration calibration curve at the indicated Mach number, pressure altitude, and gross weight. Read the Mach correction (ΔM). Add or subtract the correction, as required, to determine the true Mach number.
For example, cruise operation in a configuration with tip fuel tanks at 0.90 indicated Mach number at 10000 feet and a gross weight of 20000 pounds requires a correction of minus 0.005 Mach. The true Mach number is 0.895 (system reads high – subtract).

AIRSPEED POSITION ERROR CORRECTION

Position error correction is obtained and applied to indicated airspeed to determine calibrated airspeed as follows: enter Figure A1-7 or Figure A1-8 with indicated airspeed (corrected for individual instrument mechanical error) and read the airspeed position error correction. Add or subtract, as appropriate, the correction to obtain the calibrated airspeed from the corrected instrument reading.

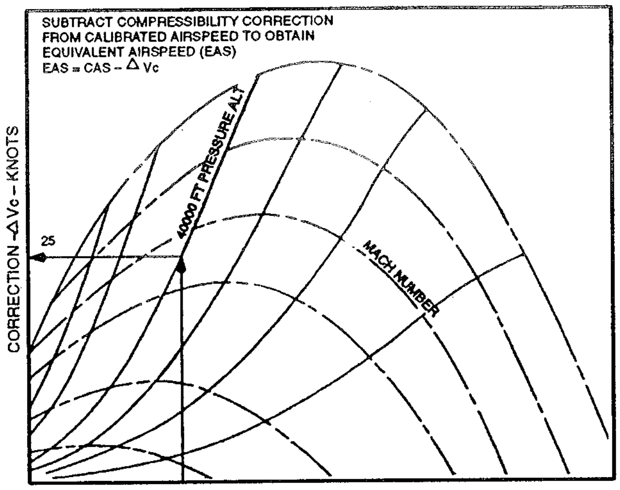
ALTIMETER POSITION ERROR CORRECTION

Correction for altimeter position error may be obtained from Figure A1-3 and Figure A1-4 on page A1-8 for subsonic and supersonic operation at various indicated Mach numbers, from Figure A1-7 and Figure A1-8 for subsonic operation at indicated airspeeds, and from Figure A1-5 and Figure A1-6 for subsonic, low-altitude operation at true Mach numbers.
At low-level the altimeter may indicate an altitude higher than true pressure altitude. Therefore it is important that the appropriate calibration curve be used to establish the flight altitude required, to ensure terrain clearance.
For example, operation in a configuration with tip tanks at an indicated Mach number of 0.95 at sea level and a gross weight of 18000 pounds requires a correction of minus 280 feet. The proper indicated altitude to fly for this condition then shall be at least 300 feet. On entering level turns at low altitude, the altimeter will indicate a loss of 200 to 300 feet, depending upon the load factor in the turn. At high altitudes and supersonic speeds, the altimeter will also lose approximately 300 feet on entering level turns.

AIRSPEED COMPRESSIBILITY CORRECTION

The airspeed compressibility correction may be obtained and applied to the CAS for determination of EAS as follows: enter Figure A1-9 with calibrated airspeed and subtract the compressibility correction value shown on the curve from the calibrated airspeed to obtain equivalent airspeed.

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED



AIRSPEED COMPRESSIBILITY CORRECTION – SAMPLE PROBLEM

Problem: To find equivalent airspeed

Altimeter reading .... 40000 ft

Calibrated airspeed (CAS) .... 300 knots

Airspeed compressibility correction
(use Figure A1-9, find ΔV_c) .... 25 knots

Equivalent airspeed (CAS $-$ ΔV_c) .... 275 knots

AIRSPEED-MACH NUMBER CONVERSION

True Mach number, true airspeed, or calibrated airspeed may be obtained from the Airspeed-Mach Number curves. Figure A1-10 covers the cruise range of the aircraft.

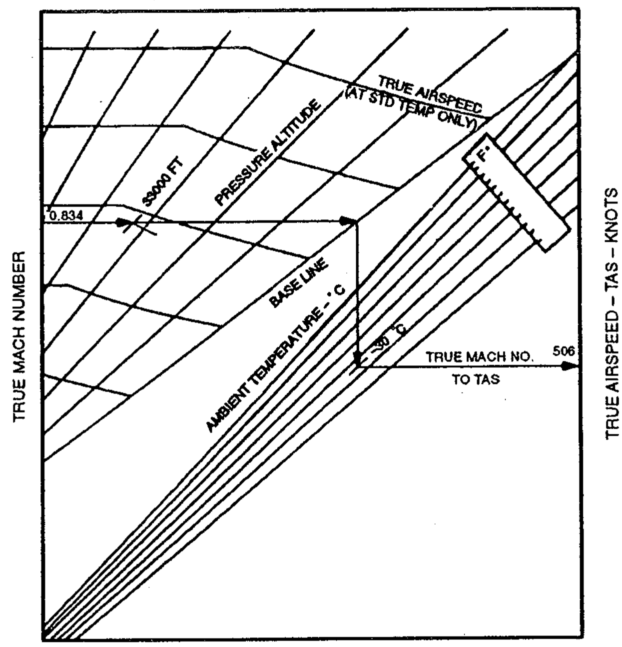
Figure A1-11 provides data for the high-speed operating area. Use of the charts is illustrated by the following sample problems.

AIRSPEED-MACH NUMBER CONVERSION – SAMPLE PROBLEM

Problem 1: To find true airspeed

With true Mach number known, enter Figure A1-10 or Figure A1-11 on the left and proceed horizontally to the altitude desired. If standard temperature exists at that altitude, read the true airspeed from the speed line provided. If nonstandard free air temperature exists, proceed horizontally to the base line, then move down to the desired temperature and read true airspeed from the right-hand scale as follows:

AIRSPEED - MACH NUMBER CONVERSION



CALIBRATED AIRSPEED - CAS - KNOTS

True Mach Number 0.834

Pressure Altitude .... 33000 ft

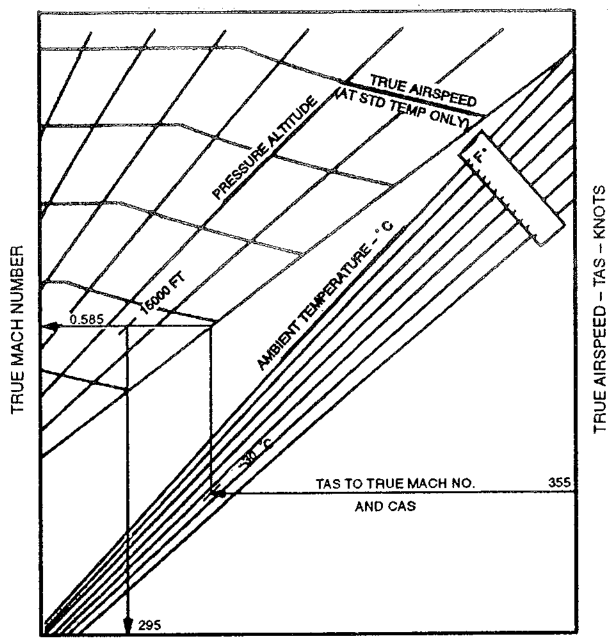
Ambient Temperature .... $-30°$ C

True Airspeed (TAS) .... 506 knots

Problem 2: To find true Mach number and Calibrated Airspeed

Using a procedure similar to steps described above, enter the charts on the right with true airspeed as follows:

AIRSPEED – MACH NUMBER CONVERSION



CALIBRATED AIRSPEED – CAS – KNOTS

FA0316

True Airspeed (TAS)	355 knots
Ambient Temperature	−30° C
Pressure Altitude	15000 ft
True Mach Number	0.585
Calibrated Airspeed (CAS)	295 knots

USE OF CIT TO OBTAIN TRUE AIRSPEED

The US Air Force MB-2A pilot's computer may be used in conjunction with the cockpit CIT (engine air inlet temperature) gage indication to obtain ambient air temperature and true airspeed during flight. Air temperature sensed by the CIT gage is greater than ambient by 100% of the possible adiabatic rise when operating at stable cruise speeds.
By setting an input data of CIT gage reading, C_T of 1.0, calibrated airspeed and calibrated altitude, the MB-2A computer will yield true airspeed, true Mach number, temperature rise and ambient air temperature.

Ambient air temperature may be approximated at constant Mach 0.80, 0.85 or 0.90, using the CIT gage and the following relationships.

Mach	Ambient Air Temp. °C
.80	.9 CIT − 31° C
.85	.9 CIT − 34° C
.90	.9 CIT − 38° C

For example, determine the ambient air temperature in cruising flight at 20000 feet at Mach 0.80. The CIT Engine Air Inlet (Temperature) is 20° C. Ambient temperature is −13° C; .9(20) − 31.

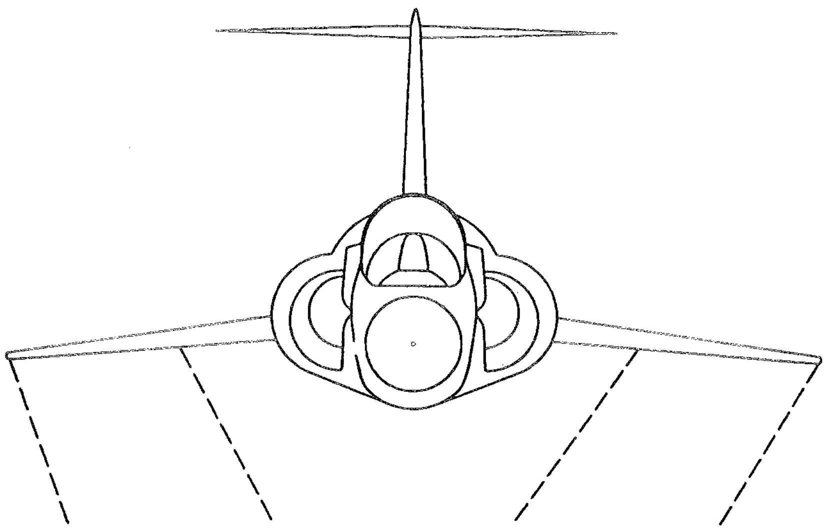
STANDARD ALTITUDE TABLE

The standard altitude table (Figure A1-12) provides reference temperature, pressure, air density, and sonic speed information which may be of assistance in overall flight planning.

STANDARD UNITS CONVERSION TABLE

The standard units conversion table (Figure A1-13) provides a means for direct conversions of temperature from degrees centigrade to degrees Fahrenheit, and of distance and speed from feet, nautical miles, feet per second and feet per minute to metric equivalents of meters, kilometers, meters per second and meters per minute, respectively.

CONFIGURATION DRAG INDEX



EXTERNAL STORES MOUNTED				TAKEOFF WEIGHT (LBS)	CONFIGURATION DRAG INDEX
WING TIP	BL 75	BL 75	WING TIP		
				19732	0
TANK			TANK	22463	16
	TANK	TANK		22962	52
TANK	TANK	TANK	TANK	25693	68

NOTES:
1) Operating mass empty: 15056 lbs
2) Take-off weights computed considering full internal and external fuel tanks (when applicable)
3) Usable fuel:
   - Internal fuel 4676 lbs
   - External fuel 4876 lbs
   - Total usable fuel 9552 lbs
4) 170 gall. external tank (P/N 851717) on wing tip
5) 195 gall. external tank (P/N 791210) on BL 75

Figure A1-1

STORE DRAG NUMBER AND LOADED GROSS WEIGHT

EXTERNAL STORES	SINGLE WEIGHT lbs	STORE DRAG INDEX (Single Item)	
		BL75 PYLON	WING TIP
WING TIP MOUNTED			
Fuel tank (INCLUDES 1135.5 lb fuel)	1365.5		8
WING PYLON MOUNTED			
BL 75 equipped for fuel tank	135.5	3	
Fuel tank (INCLUDES 1302.5 lb fuel)	1479.5	23	

TF104G-M — Operating Mass Empty (includes crew) 15056 lb

USABLE FUEL
- — Main internal fuel 4676 lb
- — External fuel 4876 lb
- — Tip tanks (two) 2271 lb
- — Pylon tanks (two) 2605 lb

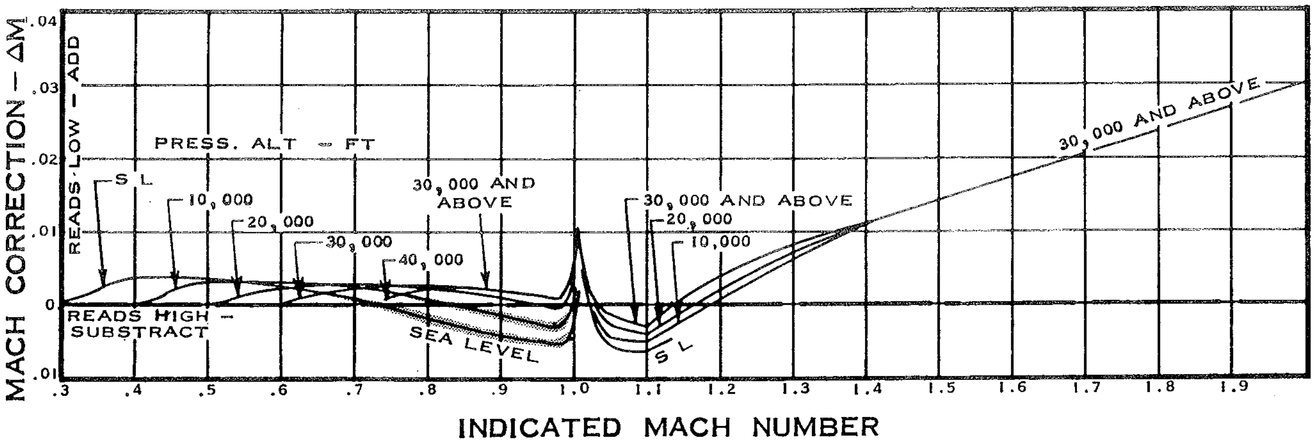
Figure A1-2

MACH NUMBER AND ALTIMETER POSITION ERROR CORRECTION

FLAPS AND GEAR UP — NO WING TIP STORES

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



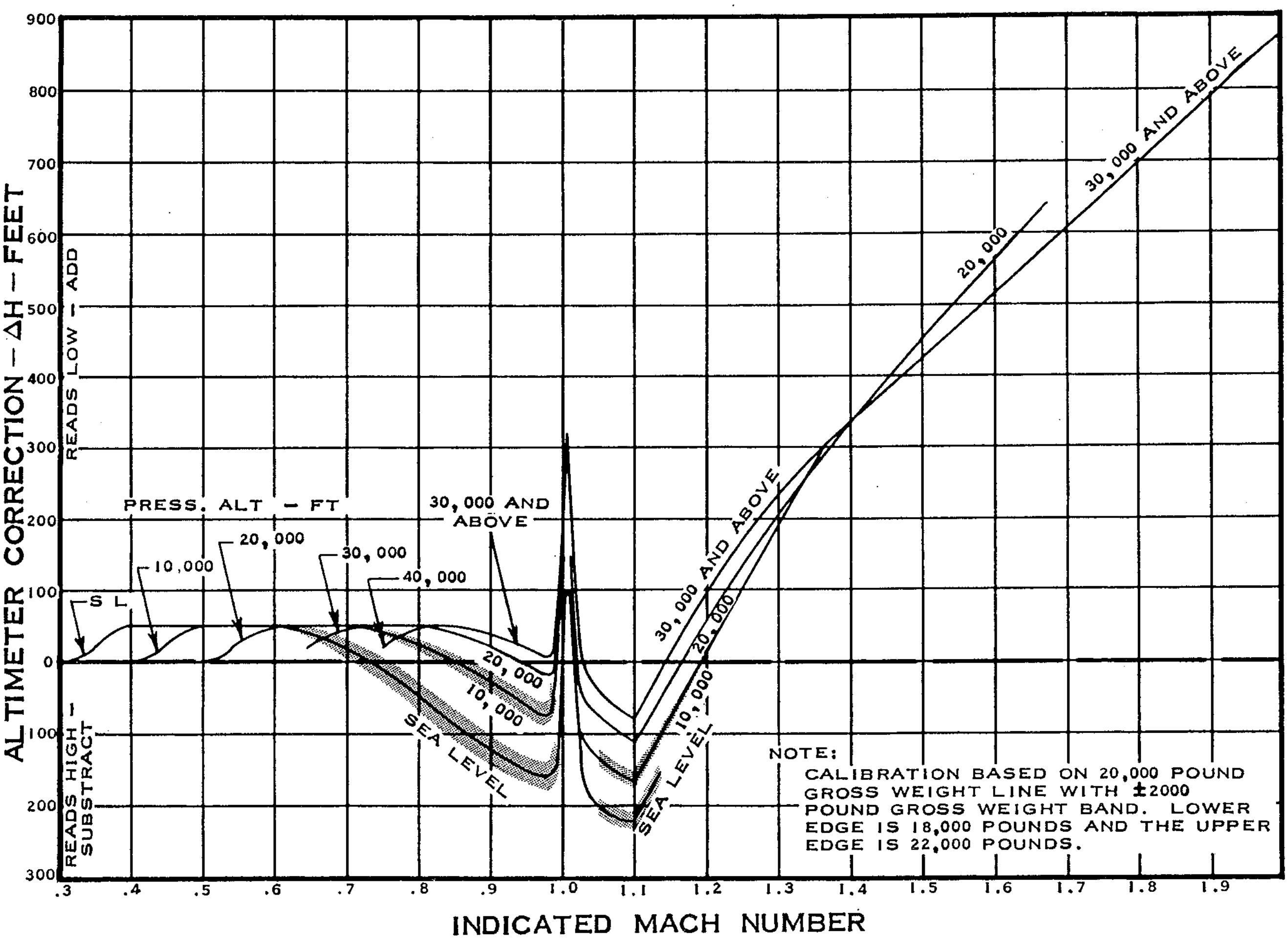


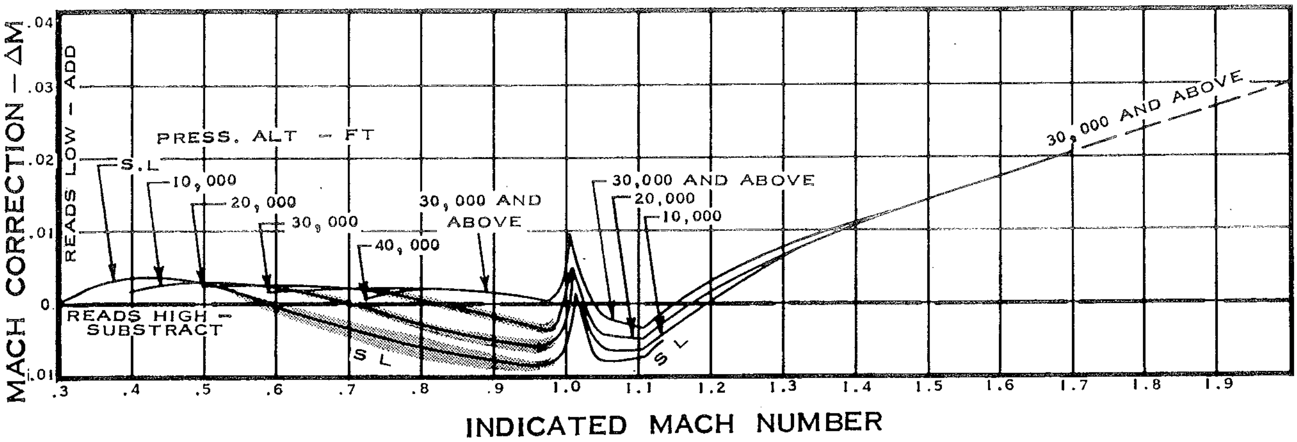
Figure A1-3

MACH NUMBER AND ALTIMETER POSITION ERROR CORRECTION

FLAPS AND GEAR UP — TIP TANKS

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



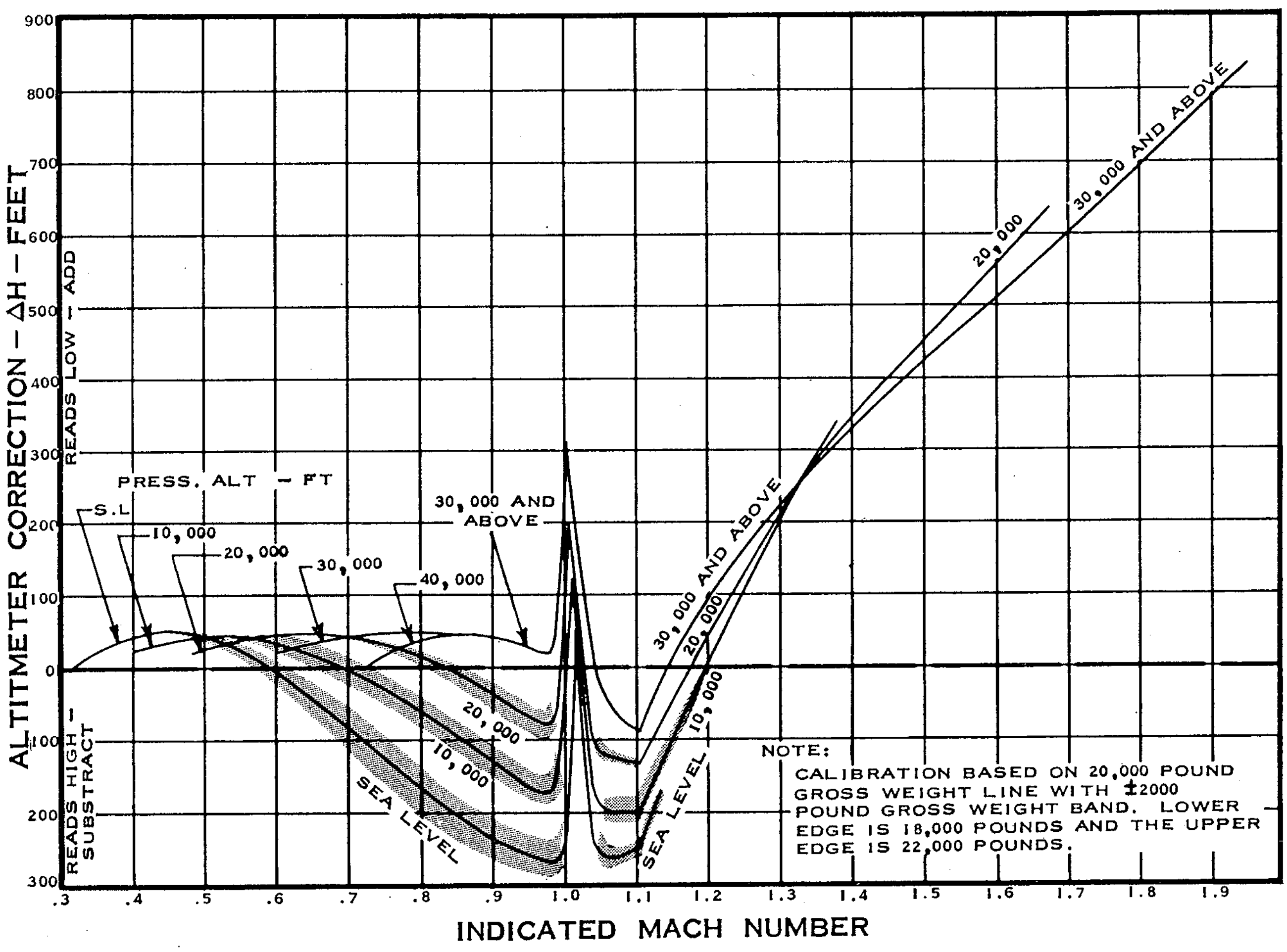


Figure A1-4

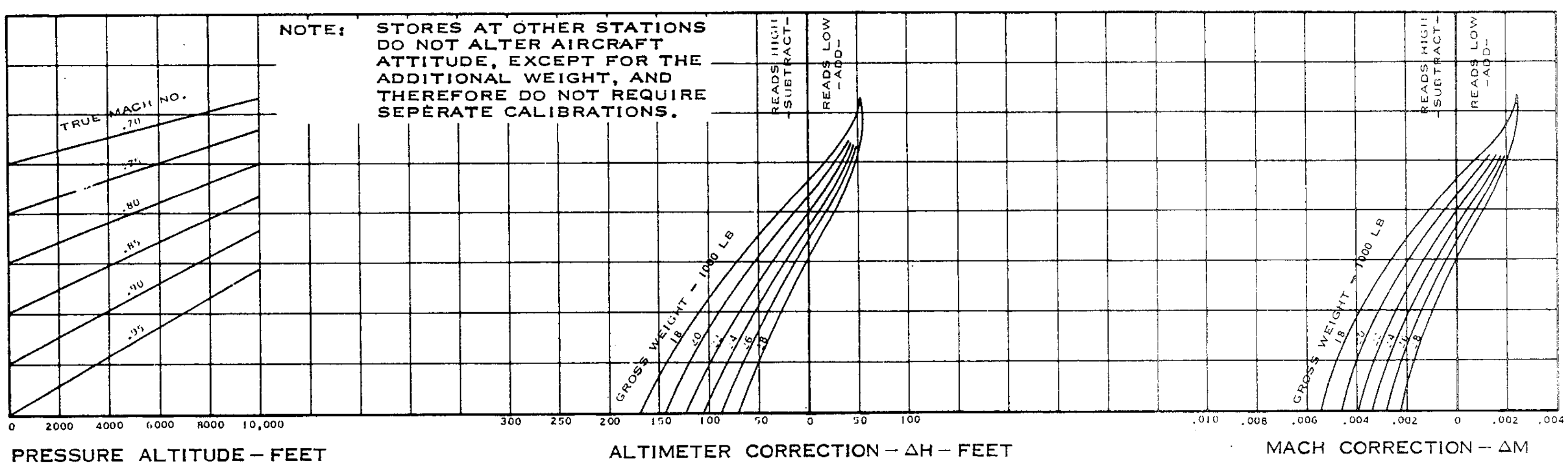
SUBSONIC OPERATION
MACH NUMBER AND ALTIMETER POSITION ERROR CORRECTION – LOW ALTITUDE

NO WING TIP STORES

Model: F-104G
Date: 1 July 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FLAPS UP



TAKEOFF (MANEUVERING) FLAPS

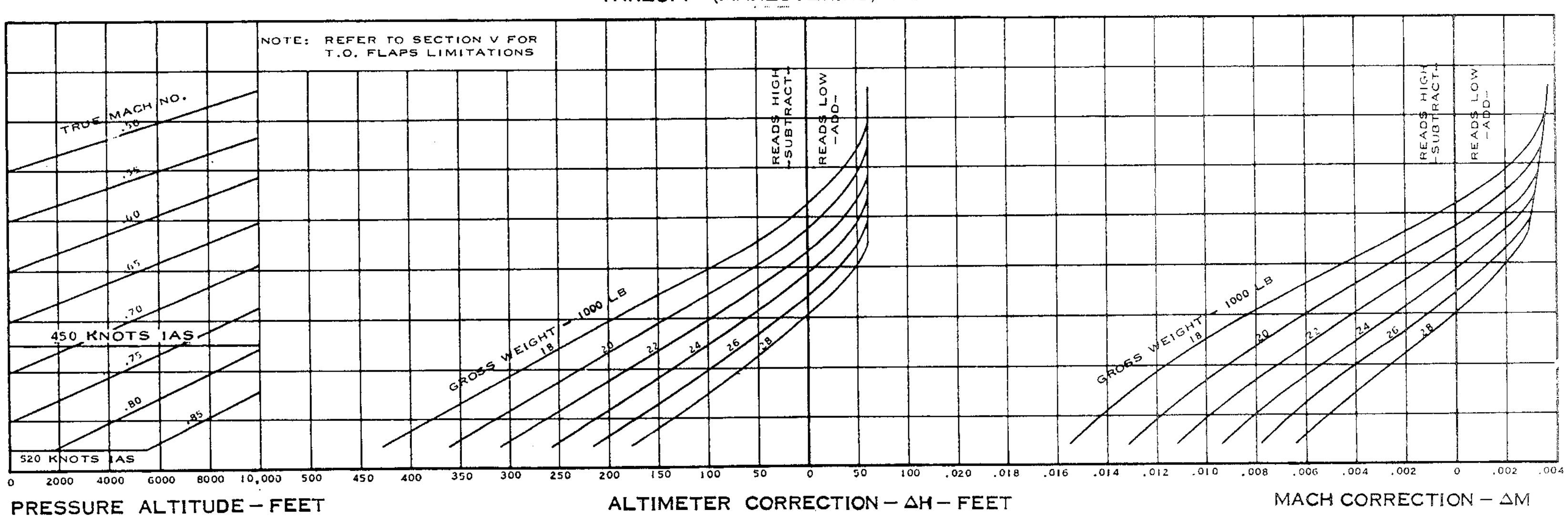


Figure A1-5

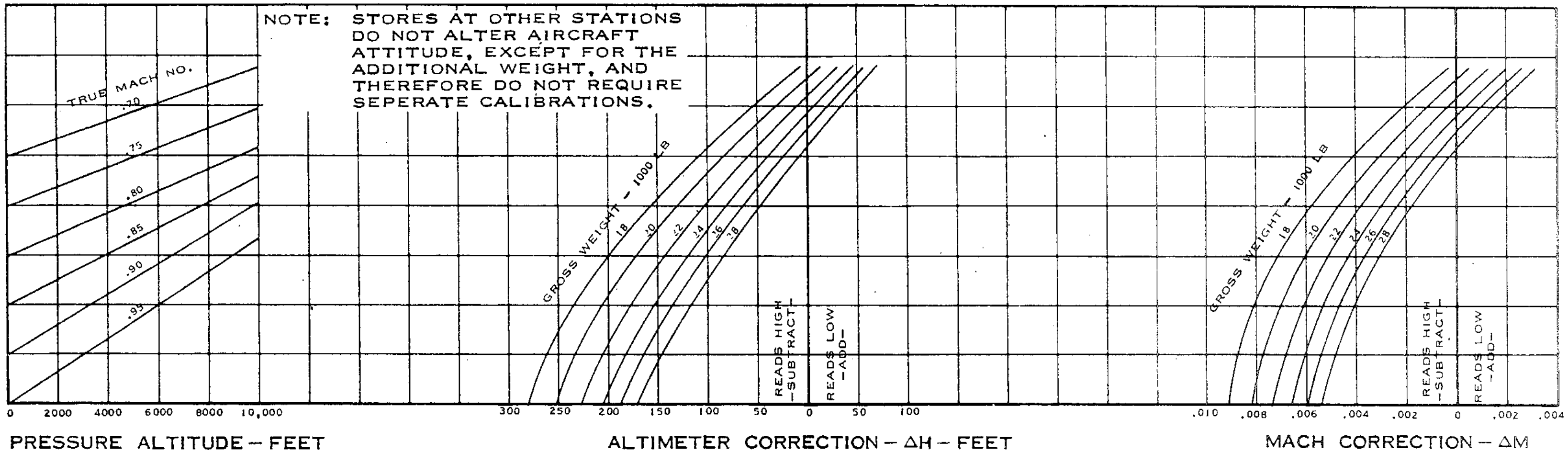
SUBSONIC OPERATION
MACH NUMBER AND ALTIMETER POSITION ERROR CORRECTION — LOW ALTITUDE

TIP TANKS

Model: F-104G
Date: 1 July 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FLAPS UP



TAKEOFF (MANEUVERING) FLAPS

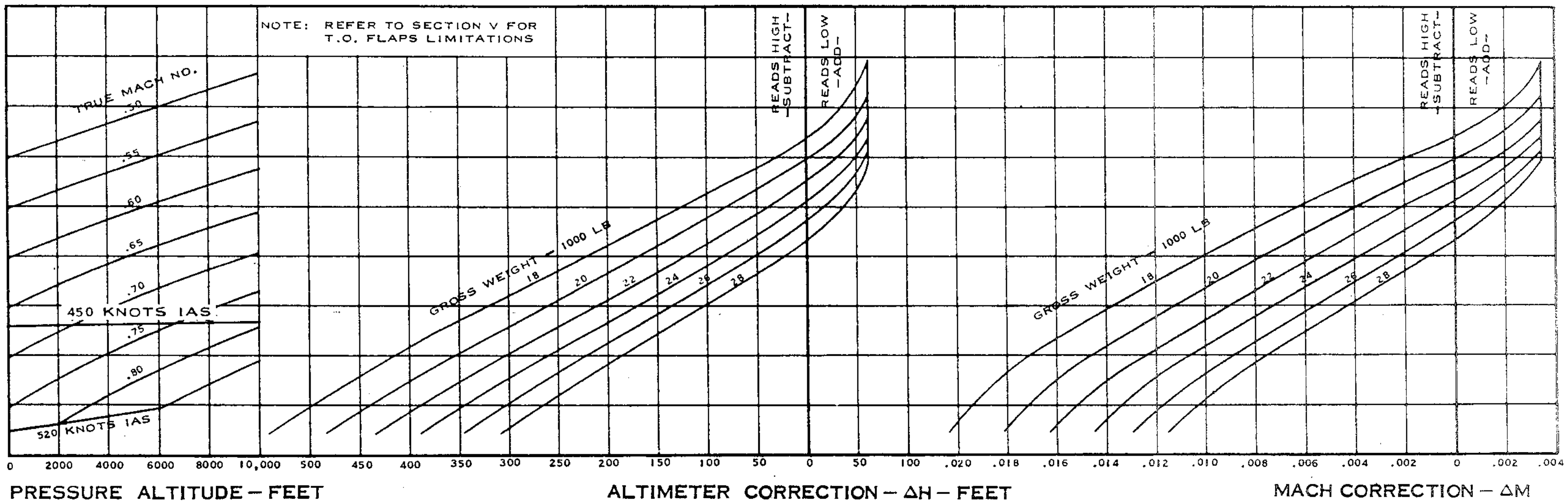


Figure A1-6

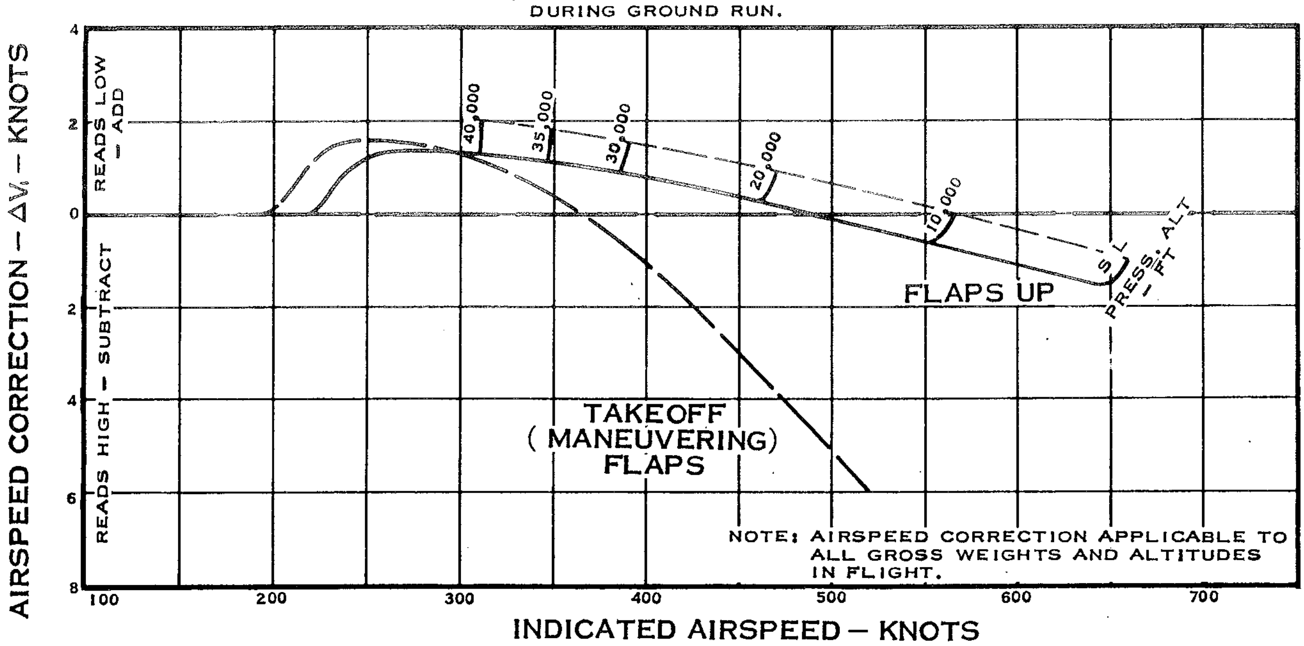
SUBSONIC OPERATION
AIRSPEED AND ALTIMETER POSITION ERROR CORRECTION

NO WING TIP STORES

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE: NO CORRECTION NECESSARY DURING GROUND RUN.



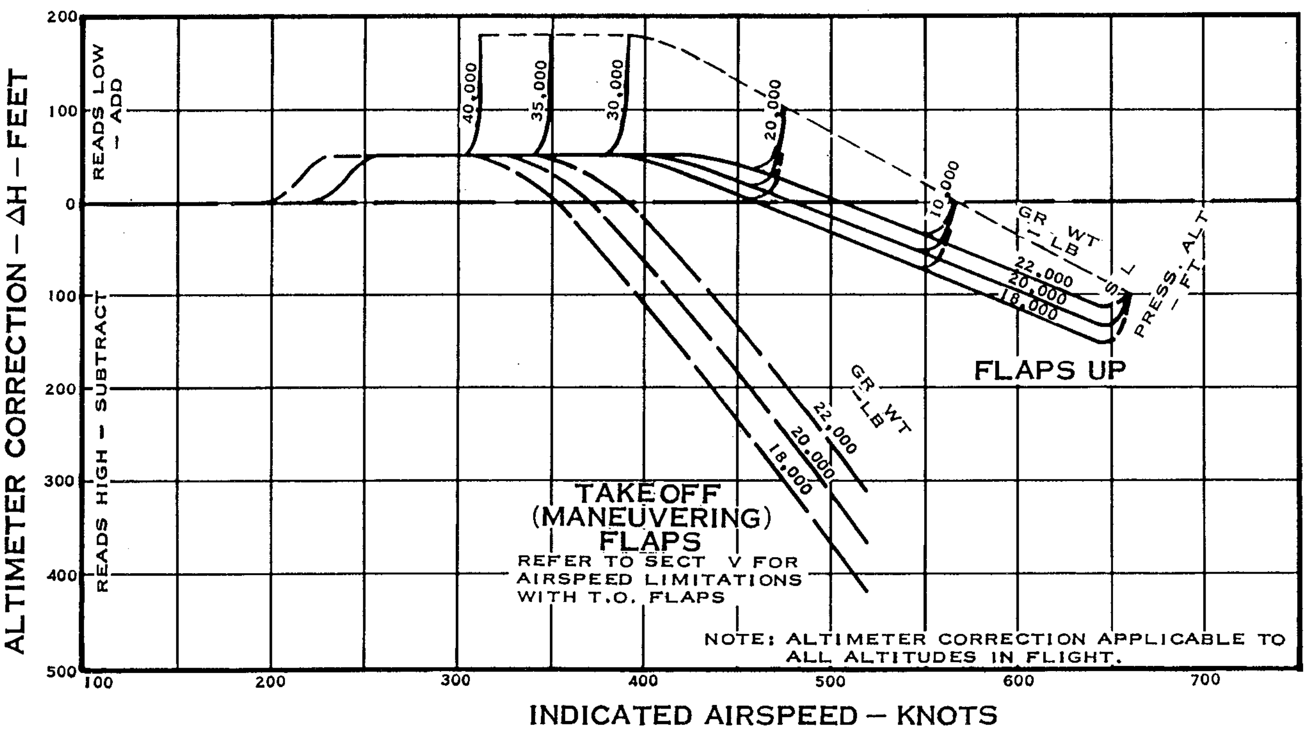


Figure A1-7

SUBSONIC OPERATION
AIRSPEED AND ALTIMETER POSITION ERROR CORRECTION

TIP TANKS

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

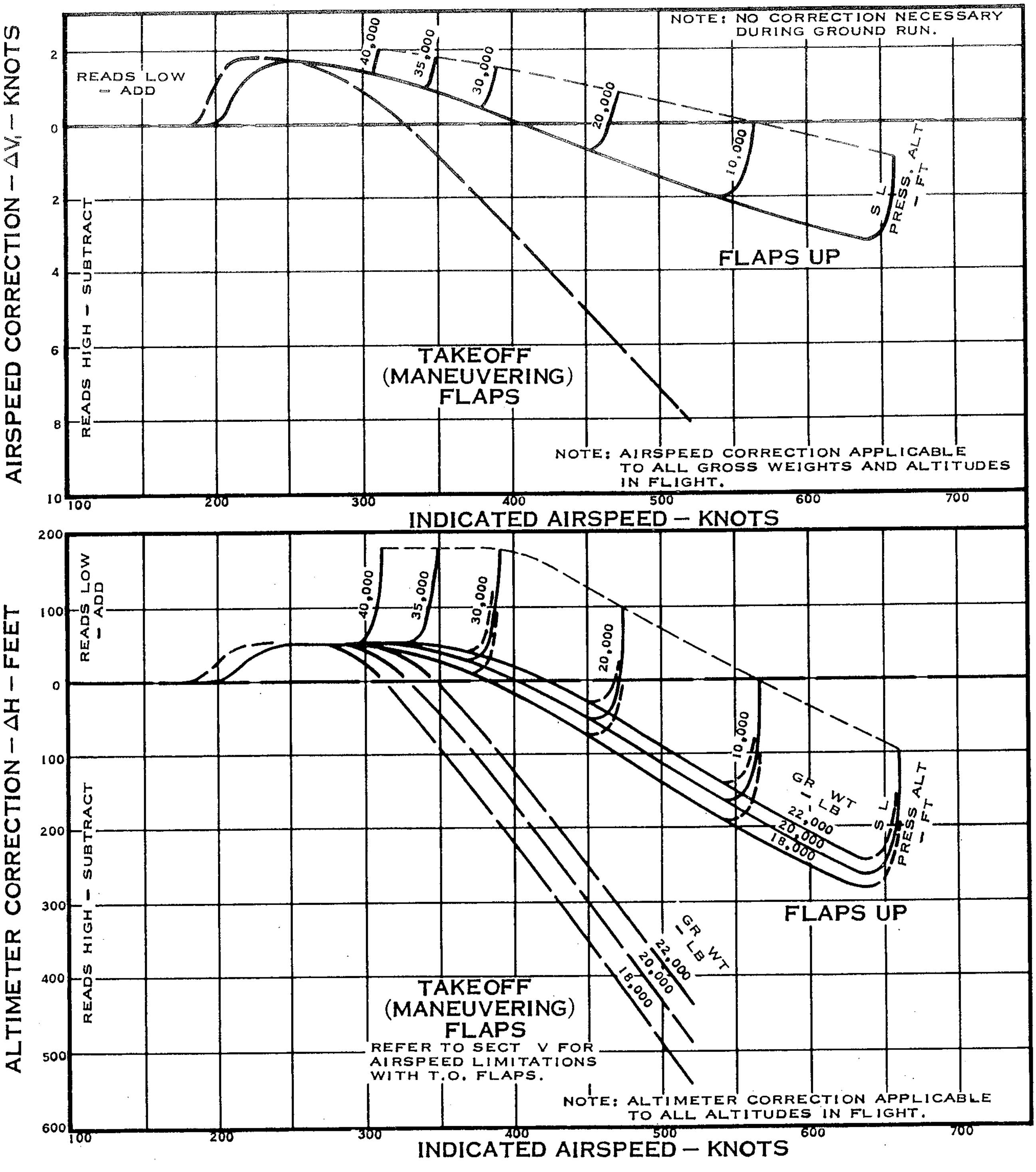


Figure A1-8

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED

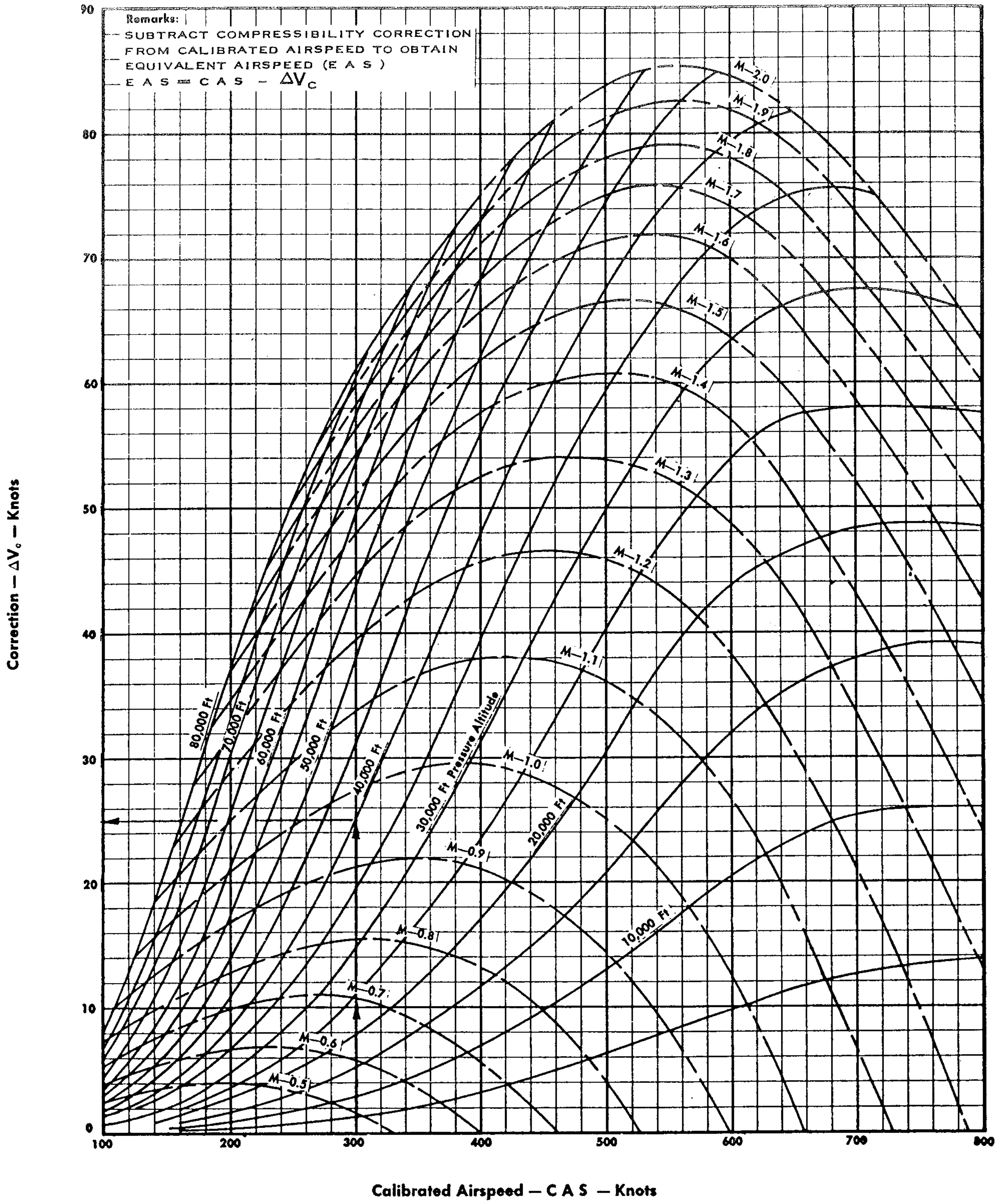


Figure A1-9

AIRSPEED — MACH NUMBER CURVES

0.3 TO 1.2 MACH NUMBER

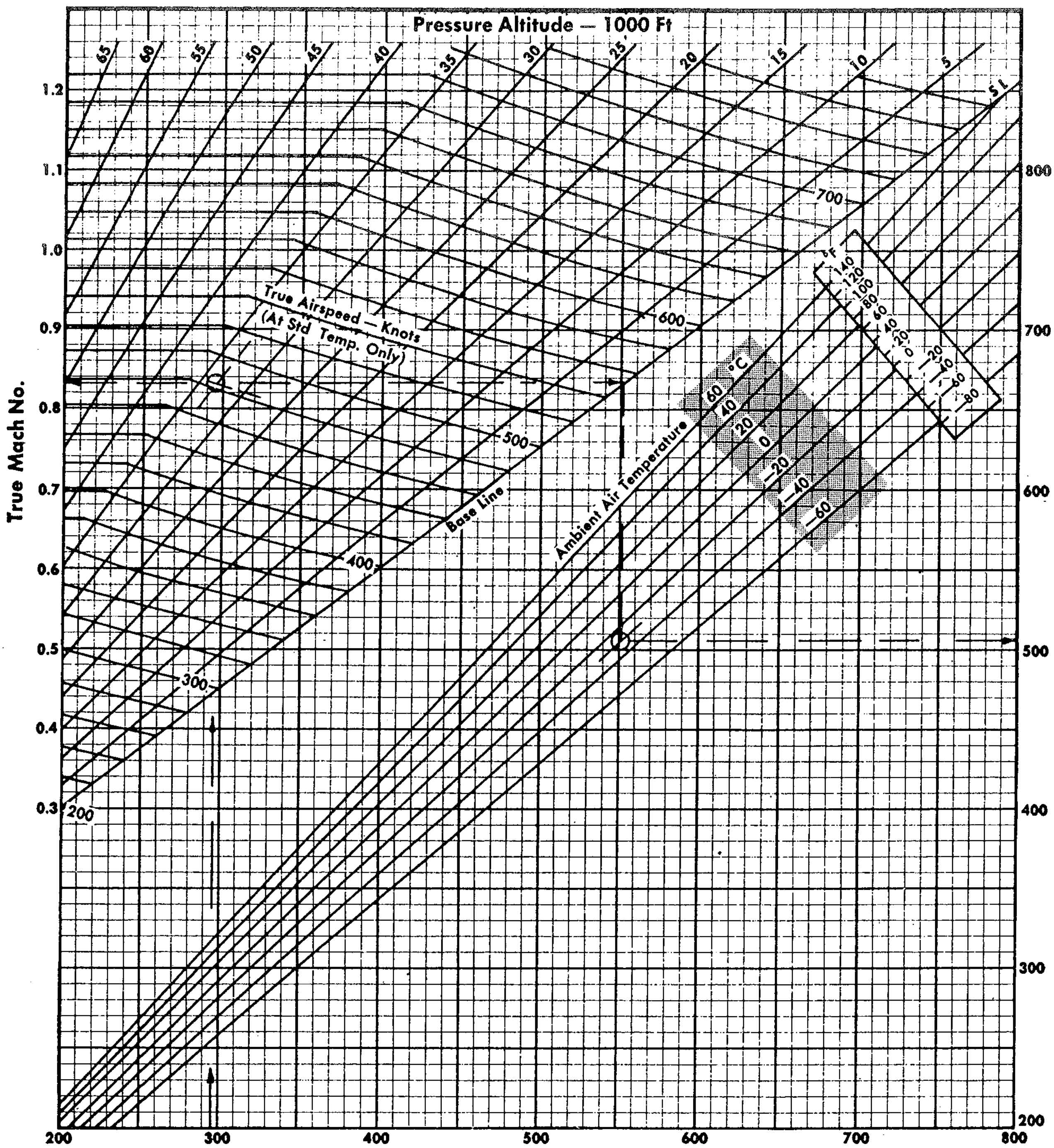


Figure A1-10

AIRSPEED — MACH NUMBER CURVES

0.4 TO 2.2 MACH NUMBER

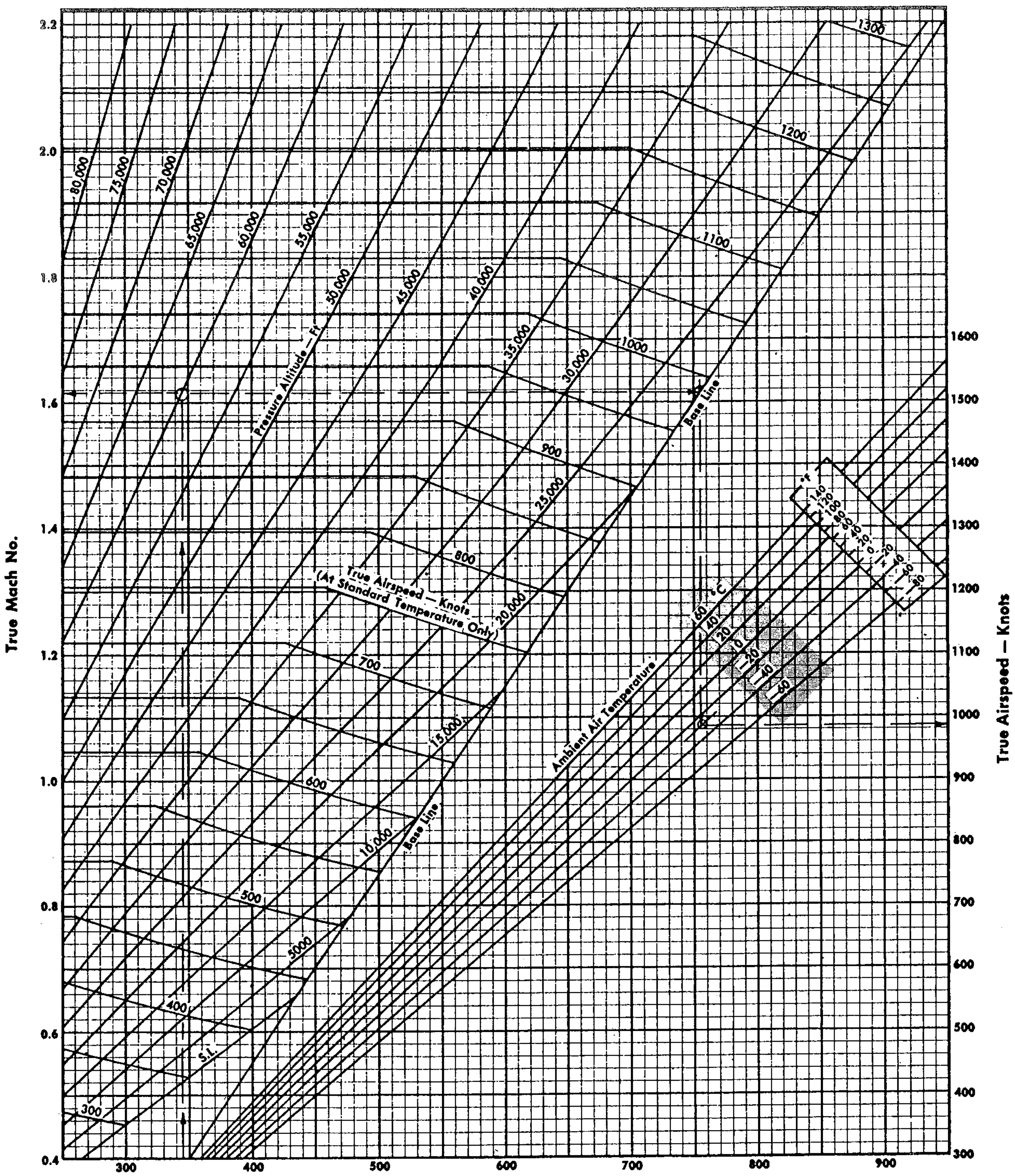


Figure A1-11

STANDARD ALTITUDE TABLE

STANDARD SEA LEVEL AIR:
$T = 59°F$
$P = 29.921$ IN. OF HG

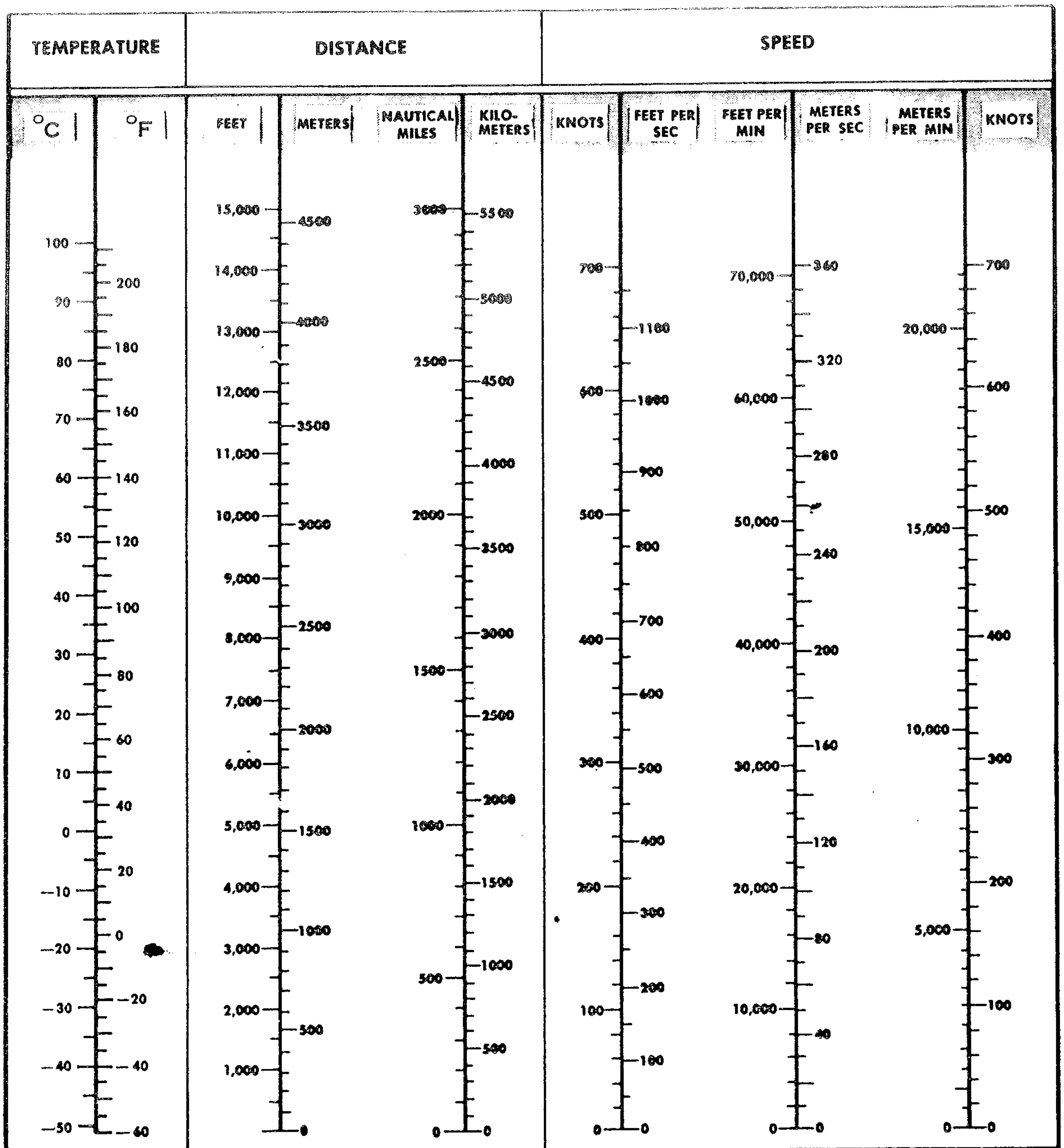
$W = .076475$ LB/CU FT $\rho_o = .0023769$ SLUGS/CU FT
$1''$ OF HG. $= 70.732$ LB/SQ FT $= 0.4912$ LB/SQ IN.
$a_o = 1116.89$ FT/SEC

BASED ON INTERNATIONAL CIVIL AVIATION ORGANIZATION (ICAO) STANDARD ATMOSPHERE
(NACA TECHNICAL REPORT NO. 1235)

ALTITUDE FEET	DENSITY RATIO ρ/ρ_o	$1/\sqrt{\sigma}$	TEMPERATURE DEG F	TEMPERATURE DEG C	SPEED OF SOUND RATIO a/a_o	PRESSURE IN. OF HG	PRESSURE RATIO P/P_o
—2000	1.0598	0.9714	66.132	18.962	1.0064	32.15	1.0294
—1000	1.0296	0.9855	62.566	16.981	1.0030	31.02	1.0147
0	1.0000	1.0000	59.000	15.000	1.0000	29.92	1.0000
1000	.9711	1.0148	55.434	13.019	.9966	28.86	.9644
2000	.9428	1.0299	51.868	11.038	.9931	27.82	.9298
3000	.9151	1.0454	48.302	9.057	.9896	26.82	.8962
4000	.8881	1.0611	44.735	7.075	.9862	25.84	.8637
5000	.8617	1.0773	41.169	5.094	.9827	24.90	.8320
6000	.8359	1.0938	37.603	3.113	.9792	23.98	.8014
7000	.8106	1.1107	34.037	1.132	.9756	23.09	.7716
8000	.7860	1.1279	30.471	— 0.849	.9721	22.22	.7428
9000	.7620	1.1456	26.905	— 2.831	.9686	21.39	.7148
10000	.7385	1.1637	23.338	— 4.812	.9650	20.58	.6877
11000	.7156	1.1822	19.772	— 6.793	.9614	19.79	.6614
12000	.6932	1.2011	16.206	— 8.774	.9579	19.03	.6360
13000	.6713	1.2205	12.640	—10.756	.9543	18.29	.6113
14000	.6500	1.2403	9.074	—12.737	.9507	17.58	.5875
15000	.6292	1.2606	5.508	—14.718	.9470	16.89	.5643
16000	.6090	1.2815	1.941	—16.699	.9434	16.22	.5420
17000	.5892	1.3028	— 1.625	—18.681	.9397	15.57	.5203
18000	.5699	1.3246	— 5.191	—20.662	.9361	14.94	.4994
19000	.5511	1.3470	— 8.757	—22.643	.9324	14.34	.4791
20000	.5328	1.3700	—12.323	—24.624	.9287	13.75	.4595
21000	.5150	1.3935	—15.889	—26.605	.9250	13.18	.4406
22000	.4976	1.4176	—19.456	—28.587	.9213	12.64	.4223
23000	.4807	1.4424	—23.022	—30.568	.9175	12.11	.4046
24000	.4642	1.4678	—26.588	—32.549	.9138	11.60	.3876
25000	.4481	1.4938	—30.154	—34.530	.9100	11.10	.3711
26000	.4325	1.5206	—33.720	—36.511	.9062	10.63	.3552
27000	.4173	1.5480	—37.286	—38.492	.9024	10.17	.3398
28000	.4025	1.5762	—40.852	—40.473	.8986	9.725	.3250
29000	.3881	1.6052	—44.419	—42.455	.8948	9.297	.3107
30000	.3741	1.6349	—47.985	—44.436	.8909	8.885	.2970
31000	.3605	1.6654	—51.551	—46.417	.8871	8.488	.2837
32000	.3473	1.6968	—55.117	—48.398	.8832	8.106	.2709
33000	.3345	1.7291	—58.683	—50.379	.8793	7.737	.2586
34000	.3220	1.7623	—62.249	—52.361	.8754	7.382	.2467
35000	.3099	1.7964	—65.816	—54.342	.8714	7.041	.2353
36000	.2981	1.8315	—69.382	—56.323	.8675	6.712	.2243
37000	.2844	1.8753	—69.700	—56.500	.8671	6.397	.2138
38000	.2710	1.9209	—69.700	—56.500	.8671	6.097	.2038
39000	.2583	1.9677	—69.700	—56.500	.8671	5.811	.1942
40000	.2462	2.0155	—69.700	—56.500	.8671	5.538	.1851
41000	.2346	2.0645	—69.700	—56.500	.8671	5.278	.1764
42000	.2236	2.1148	—69.700	—56.500	.8671	5.030	.1681
43000	.2131	2.1662	—69.700	—56.500	.8671	4.794	.1602
44000	.2031	2.2189	—69.700	—56.500	.8671	4.569	.1527
45000	.1936	2.2728	—69.700	—56.500	.8671	4.355	.1455
46000	.1845	2.3281	—69.700	—56.500	.8671	4.151	.1387
47000	.1758	2.3848	—69.700	—56.500	.8671	3.956	.1322
48000	.1676	2.4428	—69.700	—56.500	.8671	3.770	.1260
49000	.1597	2.5022	—69.700	—56.500	.8671	3.593	.1201
50000	.1522	2.5630	—69.700	—56.500	.8671	3.425	.1145
51000	.1451	2.6254	—69.700	—56.500	.8671	3.264	.1091
52000	.1383	2.6892	—69.700	—56.500	.8671	3.111	.1040
53000	.1318	2.7546	—69.700	—56.500	.8671	2.965	.09909
54000	.1256	2.8216	—69.700	—56.500	.8671	2.826	.09444
55000	.1197	2.8903	—69.700	—56.500	.8671	2.693	.09001
56000	.1141	2.9606	—69.700	—56.500	.8671	2.567	.08578
57000	.1087	3.0326	—69.700	—56.500	.8671	2.446	.08176
58000	.1036	3.1063	—69.700	—56.500	.8671	2.331	.07792
59000	.09877	3.1819	—69.700	—56.500	.8671	2.222	.07426
60000	.09414	3.2593	—69.700	—56.500	.8671	2.118	.07078
61000	.08972	3.3386	—69.700	—56.500	.8671	2.018	.06746
62000	.08551	3.4198	—69.700	—56.500	.8671	1.924	.06429
63000	.08150	3.5029	—69.700	—56.500	.8671	1.833	.06127
64000	.07767	3.5881	—69.700	—56.500	.8671	1.747	.05840
65000	.07403	3.6754	—69.700	—56.500	.8671	1.665	.05566

Figure A1-12

STANDARD UNITS CONVERSION



NOTE:

TO OBTAIN U S GALLONS MULTIPLY LITERS BY 0.264

TO OBTAIN IMPERIAL GALLONS MULTIPLY LITERS BY 0.220

TO OBTAIN INCHES OF MERCURY MULTIPLY MILLIBARS BY .0295

TO OBTAIN POUNDS MULTIPLY KILOGRAMS BY 2.20

Figure A1-13

PART 2

TAKEOFF

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

TAKEOFF PLANNING AND CHARTS

This part includes the charts to be used in predicting Maximum and Military thrust takeoff performance. The takeoff distances are based on normal acceleration and use of the takeoff technique described in Section II. They are applicable to all configurations and weights. Refusal speed and acceleration check data are provided so that a Go, No-Go procedure may be used if desired.

Use of the Go, No-Go concept lets the pilot check predicted airspeed at marked runway distances against actual performance while accelerating. The system assumes that failure to check within allowable tolerance indicates a malfunction that justifies aborting takeoff.

TAKEOFF PERFORMANCE NOMOGRAM

Takeoff performance with Maximum thrust is presented in Figure A2-3 in a form readily usable for "everyday" types of takeoff flight planning. The chart contains the normal performance speed schedule, takeoff ground run distances, and acceleration check speeds for a standardized 2000-foot acceleration check distance. Also included in the chart are ground run distance corrections for wind velocity and runway slope.

The series of shaded and clear areas, bounded by lines of wind speed, indicate areas where normal takeoff speeds exceed the ground-speed tire limit of 239 knots (interpolate as necessary between these lines for the existing wind conditions).

An accurate presentation of true ground-speed for various takeoff conditions is presented in Figure A2-4. Maximum performance technique may be desirable. Maximum performance ground run distances can be obtained by multiplying the normal performance values by 0.95.

To determine the maximum performance acceleration check speed, enter the chart with the maximum performance ground run distance and maximum performance takeoff speed (shaded area at the bottom of the chart).

TAKEOFF SPEED SCHEDULES

Normal takeoff speeds and the speeds at which a 50-foot height should be attained are given in Figure A2-4 for the entire range of takeoff gross weights. The same chart supplies the speeds for maximum performance operations. As noted on the takeoff ground run distance charts, adjust normal takeoff distances by 5½ percent per 5 knots IAS deviation from normal speeds.

Maximum performance speeds may be desirable if operating from runways of marginal length, or necessary if normal procedure results in takeoff groundspeeds exceeding tire limits.

WIND COMPONENT CHART

The takeoff distance and takeoff groundspeed charts incorporate correction grids which allow an accounting for headwind and tailwind effects.

The Wind Component Chart, Figure A2-2, allows reported winds to be converted directly into crosswind and headwind or tailwind components when the reported speed and angle between wind direction and runway heading are known. Use maximum reported gust velocity to find crosswind component.

The takeoff charts do not make allowance for any difference in wind velocity at runway level and at anemometer height.

GROUND RUN DISTANCE

Takeoff ground run is the distance from the start of takeoff roll to the point at which the aircraft becomes airborne.

Takeoff distances are shown on Figure A2-5 and Figure A2-11 for Maximum and Military thrust operation, respectively. The chart apply to takeoff from dry, hard-surfaced runways for most normal variations of operating altitude, temperature, wind, and slope. The shaded areas on these charts indicated those conditions where normal procedures may result in takeoff groundspeeds exceeding limit tire speeds.

When an operating condition results in distance falling in these areas, check takeoff groundspeed, using Figure A2-4. Use of maximum performance speed may be desirable in order to avoid excessive groundspeed (see notes on takeoff distance curves for effect of using other than normal speed schedule).

NOTE

Always check the total distance to takeoff and clear 50 feet (Figure A2-10 and Figure A2-14) to be sure that climbout requirements are adequate.

REFUSAL SPEED

Refusal speed is the highest speed to which the aircraft may be accelerated, assuming normal acceleration, and still be stopped on the runway remaining. The refusal speed charts in this part assume maximum braking after the initial reaction time and prompt deployment of the drag chute if it is to be activated. They account for gross weight, pressure altitude, air temperature, and actual runway length available. They do not consider the added variables of wind or slope.

The data are directly applicable to operation from dry, hard-surface. A correction grid, keyed to a braking coefficient index, is supplied to cover operations on wet, icy, or snow-covered runways. The index may be used if the percentage of available to maximum dry surface wheel braking force may be determined for the existing condition.

Otherwise, use the general ice, snow, and rain criteria provided.

REFUSAL DISTANCE

Refusal distance is the distance required to reach refusal speed with normal acceleration. It is determined from the "Velocity During Takeoff Ground Run" chart after refusal speed has been determined. Enter the "Velocity During Takeoff Ground Run" chart from the top with takeoff speed and find a point opposite the predicted takeoff ground run distance value on the right of the grid (distance includes the period of rotation to takeoff attitude and liftoff).

The nearest guide lines now indicate the speed-distance relationship for the takeoff run. Follow the guide lines down and to the left until above the predicted refusal speed.

Read refusal distance on the acceleration distance scale to the left of the grid (distance reflects the acceleration of the aircraft in ground run attitude, prior to rotation to takeoff attitude).

NOTE

For convenience, normal takeoff speeds have been keyed to takeoff weight at the top of the chart.

ACCELERATION CHECK DISTANCE

The acceleration check distance usually used for light weight operation is the second marker beyond the brake release point. At heavy weights, the acceleration check should be made at the distance 2000 feet short of the go, no-go marker unless the speed at that point would be less than the minimum range of the airspeed indicator. In such cases, make the acceleration check at a point 1000 feet short of the go, no-go marker.

NOTE

The above applies to runways marked in thousands of feet, and to runways marker in 500 meter increments with suitable modification of the procedure. If runways are not marked, an appropriate checkpoint should be located and used for reference.

ACCELERATION CHECK SPEED

The normal speed at the acceleration check marker is determined from the "Velocity During Takeoff Ground Run" chart. Use the same guide line found for obtaining refusal distance, extending it down and to the left until a point is located opposite the desired acceleration check distance. Read acceleration check speed on the bottom scale below this point, increase by the amount of runway headwind component, and use during takeoff. During the takeoff run, compare actual indicated airspeed with the predicted value as the aircraft passes the acceleration checkpoint. If the airspeed is less than normal (minus the allowable tolerance) the takeoff may have to be aborted. It is then mandatory that a careful speed check be made upon reaching the go, no-go marker.

NOTE

If the go, no-go system is not being used, local policy should establish abort criteria and airspeed tolerance allowed at the acceleration checkpoint.

GO, NO-GO DISTANCE

The last marker reached before passing the refusal point shall be used as an abort decision point, because the refusal point usually will not coincide with a marked runway distance. This marker, at which the final decision is made as to whether or not to continue the takeoff, is called the "go, no-go marker". The distance that the aircraft has traveled in reaching the go, no-go marker is called the "go, no-go distance". The takeoff should be aborted if the aircraft is below the "go, no-go speed" at the go, no-go marker.

GO, NO-GO SPEED

The go, no-go speed is the minimum allowable speed at the go, no-go marker. The minimum allowable speed is the normal speed at the go, no-go marker corrected for acceleration tolerance and runway wind component.

The procedure for finding the go, no-go speed and marker number is as follows:

1. Find the runway length available by subtracting the lineup allowance from the runway length (lineup allowance is the distance from the head of the runway to the point where brakes are released).
2. Find the refusal speed and refusal distance.
3. The distance remaining at the refusal point is the available runway length minus the refusal distance.
4. The go, no-go marker is the marker nearest the refusal point where the distance to the end of the runway is equal to or greater than the distance in step 3.
5. The distance remaining at the go, no-go marker is the marker distance at that point plus ½ of the odd figure over exact thousands of feet of runway length.
6. The go, no-go distance (distance from the lineup point to the go, no-go point) is the available runway length minus the distance from step 5. This distance is used to enter the "Velocity During Takeoff Ground Run" chart to find the normal speed at the go, no-go marker.

ACCELERATION TOLERANCE

Acceleration tolerance is the amount that the speed at the go, no-go marker may fall below the normal speed at the marker without endangering the takeoff. The acceleration tolerance is based on the minimum allowable takeoff acceleration which will result in a zero-wind takeoff ground run no more than 90% of the runway length available. Acceleration tolerance data are presented in Figure A2-9 for Maximum and Military thrust operation. The tolerance values should not be exceeded even when excess runway length is available and use of a larger acceleration tolerance might otherwise be permitted. There must be a serious flaw in the aircraft performance and a takeoff should be aborted if the acceleration is inadequate to the exceeding the maximum allowable tolerance shown in Figure A2-9.

RUNWAY WIND COMPONENT

Normal acceleration speeds at the go, no-go point and the acceleration checkpoint shall be corrected for wind. The correction is made by adding the runway headwind component directly to the no-wind speeds. Figure A2-2 present a chart for computing the runway wind component. This chart may be used to obtain crosswind component speeds. Acceleration check speed and go, no-go speed should always be corrected for runway wind component. Takeoff planning procedure may ignore the effect of headwind on ground run distance, thus considering the benefits of headwind as an added safety margin. If this is done a check should be made to be certain the crosswind conditions are not excessive.

TOTAL DISTANCE TO TAKE OFF AND CLEAR A 50-FOOT HEIGHT

The total distance to take off and clear a 50-foot obstacle is the sum of the ground run distance and the air distance traveled while attaining a height of 50 feet above the runway.

Total distance chart are provided for Maximum and Military thrust operation in Figure A2-10 and Figure A2-14.
Distances are shown for a range of altitude, temperature, wind, and slope conditions. The obstacle clearance speeds are shown in Figure A2-4.

RUNWAY MARKING SYSTEM

The runway markers are placed 1000 feet apart along the runway (see Figure A2-1) and are numbered to reflect the number of thousand-foot intervals remaining before the zero-distance marker is reached.
The zero-distance markers are on the ends of the runway when runway length is an exact multiple of 1000 feet; otherwise, the markers are placed inward from the ends of the runway, one-half the excess distance over exact thousands of feet.
In the case of a 7500-foot runway, for example, there is a 500-foot excess over exact thousands of feet; therefore, the zero-distance markers are placed half that distance, or 250 feet, from the ends of the runway, and the No. 5 marker is 5250 feet from the end of the runway.

LINEUP ALLOWANCE

The distance that the aircraft has traveled at any particular marker depends on the length of the runway and the location of the aircraft on the runway when the brakes were released.
The distance from the starting end of the runway to the aircraft at brake release point is called the lineup allowance.
If the lineup distance is 250 feet on a 7500-foot runway, brakes will be released at the No. 7 marker and the aircraft will have traveled 2000 feet at the No. 5 marker.
The term "line-up allowance" should not be confused with "marshalling distance" which is sometimes used to denote the distance between aircraft during formation takeoffs.

RUNWAY MARKING SYSTEM AND GO, NO-GO CONCEPT

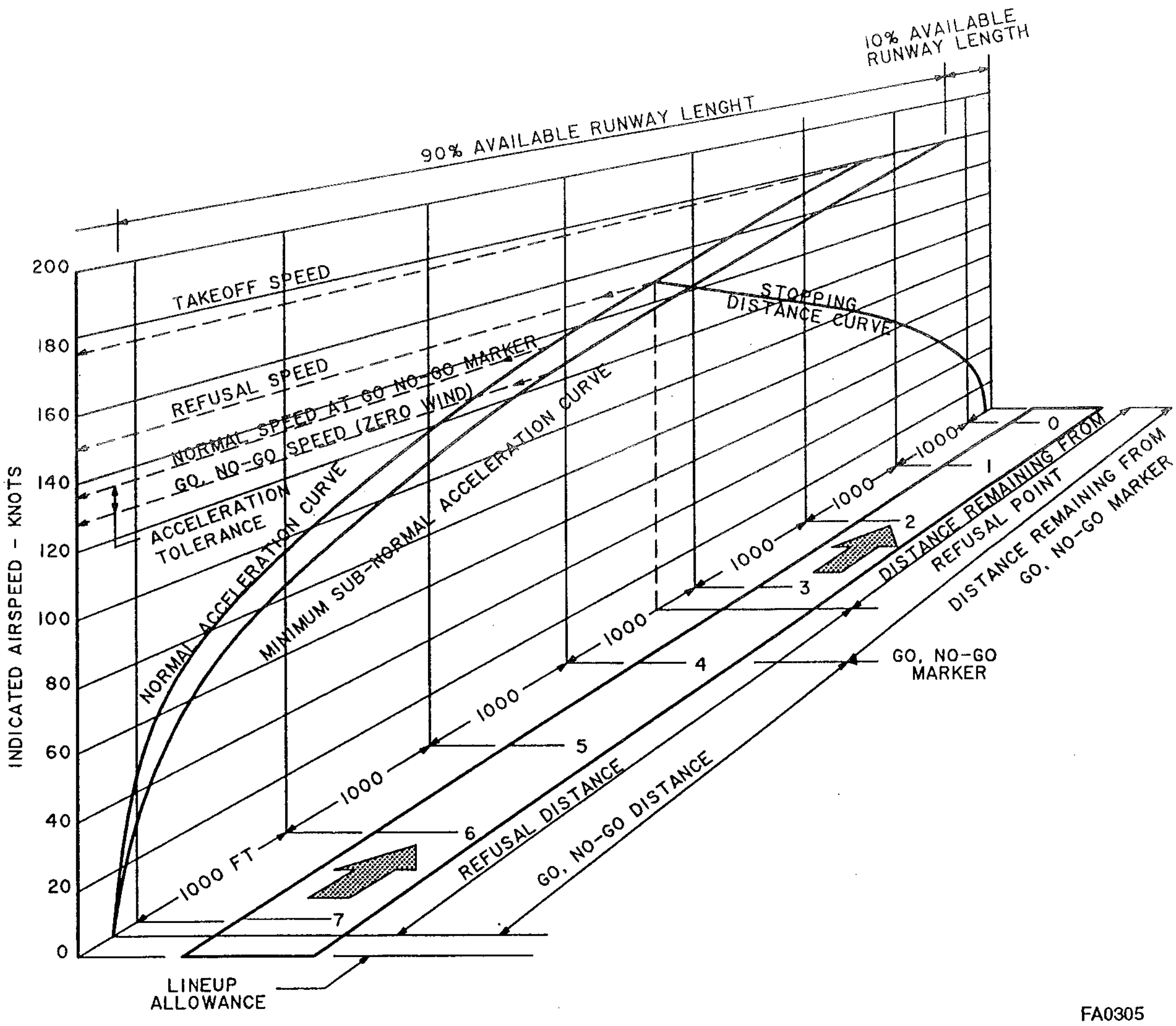


Figure A2-1

TAKEOFF PLANNING PROBLEM

LIGHT WEIGHT

The procedures for accurate takeoff planning are illustrated by two sample problems. The first problem contains the minimum amount of planning necessary; the procedures shown are adequate for normal operation at light weights. The second problem illustrates the procedures necessary for a complete solution when operating at heavy weights or under adverse conditions. The methods used for solution of the examples apply to the actual performance charts discussed. Sample charts are provided for illustration purposes.

Determine takeoff performance for the following conditions:

- Maximum thrust
- Configuration – TF-104G-M with wing tip fuel tanks
- Takeoff gross weight:
  21170 – 150 lb = 21020 lb
- Field ambient air temperature – 40° C
- Field pressure altitude – 3000 ft
- Wind – 20 knots from 060°

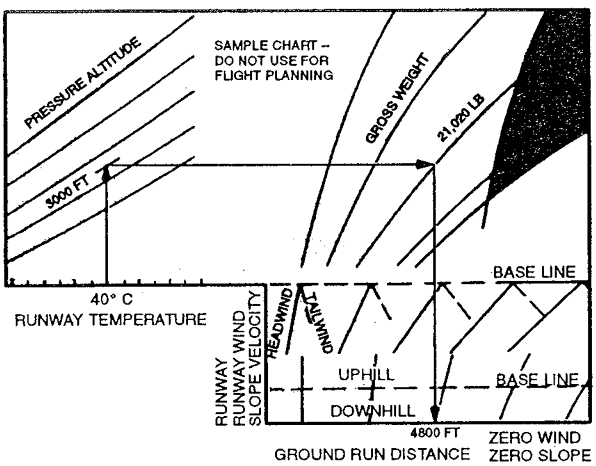
- Runway – 10250 ft long, 035° heading, hard dry surface, 1% downhill slope
- Lineup allowance – 250 feet from end of runway

a. From Figure A2-2, at a reported wind velocity of 20 knots and an angle between wind and runway of 25° (60° – 35°), the runway component is 18 knots headwind.

b. From Figure A2-5 the zero wind, zero slope ground run distance is 4800 ft. Predicted distance with wind and slope is 4100 ft.

c. The normal performance takeoff airspeed (from Figure A2-4) is 189 KIAS. Groundspeed, 190 knots, is less than tire limit speed.

TAKEOFF GROUND RUN DISTANCE – MAXIMUM THRUST



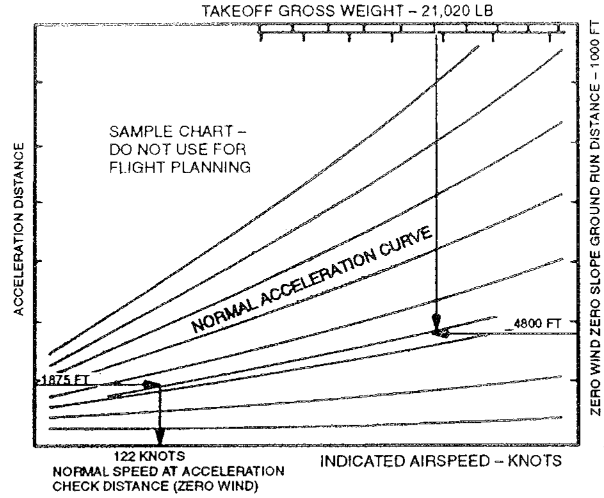
FA0317

d. The distance from the head of the runway to the No. 10 marker is 125 ft (see runway marking system text for explanation). Brakes will be released 125 ft beyond the first marker for a lineup allowance of 250 ft. The remaining runway length available is 10000 ft. The distance to clear 50 feet (Figure A2-10) is 7030 ft.

e. The acceleration check marker is No. 8, the second marker beyond brake release point. The rolling distance to this marker is 1875 ft (2000 ft – 125 ft).

f. Enter Figure A2-8 at the takeoff gross weight of 21020 lb and the zero wind, zero slope takeoff ground run distance (4800 ft) to establish the normal acceleration guide line.

VELOCITY DURING TAKEOFF GROUND RUN – MAXIMUM THRUST
TAKEOFF GROSS WEIGHT – 21,020 LB



g. Still using Figure A2-8, enter the acceleration distance scale at the check distance (1875 ft). At the intersection with the normal acceleration guide line, read the normal speed at the acceleration check distance to be 122 KIAS.

h. The runway wind component (18 knots headwind) added to the normal speed at the acceleration check distance (122 KIAS) is the speed to be expected at the No. 8 marker, 140 KIAS.

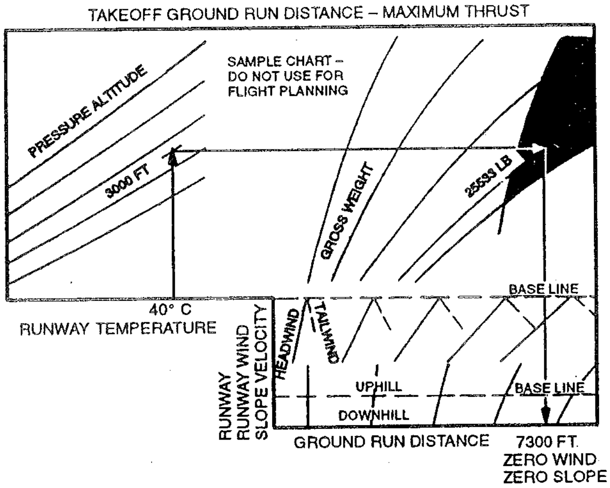
TAKEOFF PLANNING PROBLEM AT HEAVY WEIGHT

Determine takeoff performance for the following conditions:

- Configuration – TF-104G-M with tip fuel tanks and fuel tanks on BL75
- Takeoff gross weight:
  25693 lb – 150 lb = 25533 lb
- Maximum thrust
- Field pressure altitude – 3000 feet
- Field ambient air temperature – 40° C
- Wind – 20 knots from 060°
- Runway – 10250 feet long, 035° heading, dry hard surfaced, 1% downhill slope
- Lineup allowance – 250 feet from end of runway

a. From Figure A2-2, at a reported wind velocity of 20 knots and an angle between wind and runway of 25° (60° − 35°), the runway component of the wind is 18 knots headwind.

TAKEOFF GROUND RUN DISTANCE – MAXIMUM THRUST



FA0318

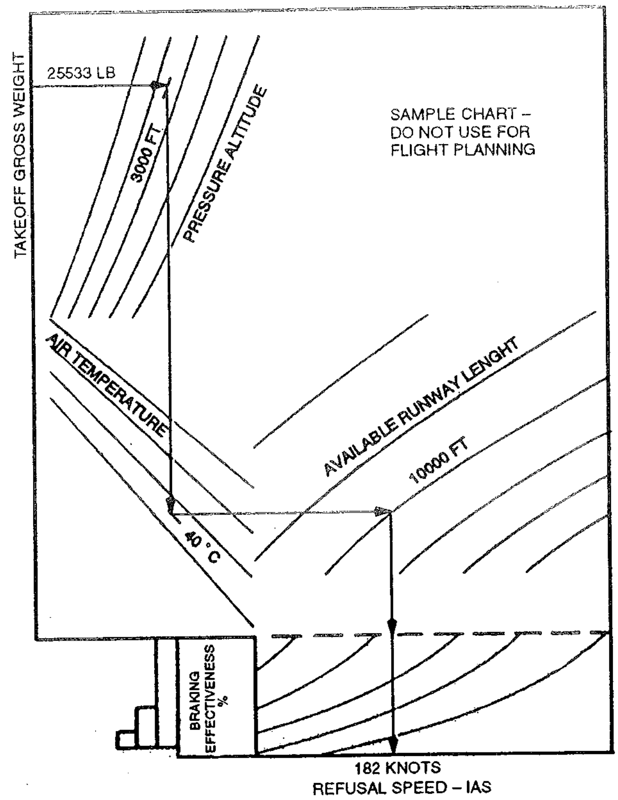
b. From Figure A2-5 the zero wind, zero slope ground run distance, for use in determining the go, no-go requirements, is 7300 ft. Predicted distance with wind and slope is 6300 ft.

c. The groundspeed at lift-off should be obtained in order to determine whether or not a maximum performance technique is advisable. From Figure A2-4, the takeoff groundspeed for the existing conditions will be 212 knots. Normal performance technique will be used. The airspeed at takeoff (also from Figure A2-4) is 209 KIAS.

d. The distance from the head of the runway to the No. 10 marker is 125 feet (see runway marking system for text explanation). Brakes will be released 125 feet beyond the first marker for a lineup allowance of 250 feet. The remaining runway length available is 10000 ft. The distance with wind and slope to clear 50 ft (Figure A2-10) is 11500 ft. (An unobstructed clearway is desirable for this type of operation, but distance to 50 ft can be decreased by reducing the airspeed at 50 ft below normal performance values).

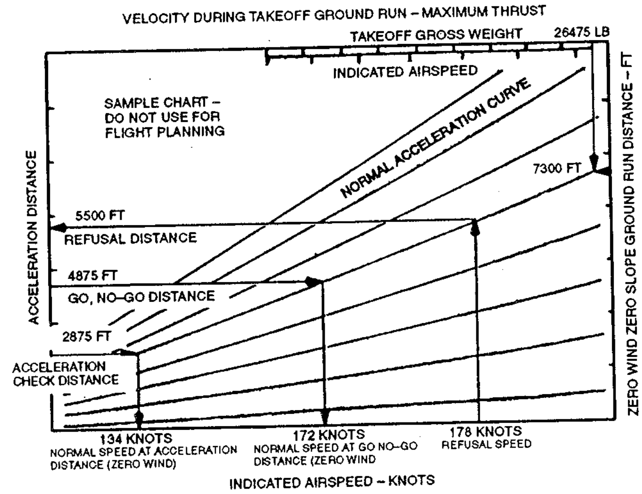
e. The refusal speed, from Figure A2-6 is 182 knots.

REFUSAL SPEED – MAXIMUM THRUST



f. Enter Figure A2-8 at the takeoff gross weight 25533 lb and zero wind, zero slope takeoff ground run distance (7300 ft) to establish the normal acceleration guide line.

VELOCITY DURING TAKEOFF GROUND RUN – MAXIMUM THRUST



FA0319

g. Still using Figure A2-8, enter (from the bottom) at the refusal speed (182 knots). At the intersection with the normal acceleration guide line read the refusal distance, 5500 ft on the acceleration distance scale.

h. The refusal point is reached 5625 ft (5500 + 125 ft) beyond the No. 10 marker, 625 ft beyond the No. 5 marker. The No. 5 marker becomes the go, no-go marker (see Go, No-Go Distance text for explanation). Ground run distance to the go, no-go marker is 625 feet less than to the refusal point, or 4875 ft (5500 − 625 ft).

i. Using Figure A2-8, enter the acceleration distance scale at the go, no-go distance (4875 ft). At the intersection with the normal acceleration guide line, read the normal speed of 172 knots at the go, go-no distance.

j. From Figure A2-9, the allowable speed tolerance is higher than the 10 knots maximum recommended speed tolerance. The zero wind go, no-go speed is 162 KIAS (172 − 10 knots).

k. Adding the headwind component to the go, no-go speed found above provides the final go, no-go speed of 180 KIAS (18 + 162 knots). The takeoff should be aborted if airspeed is below 180 KIAS at the No. 5 marker. Drag chute and maximum wheel braking should be used.

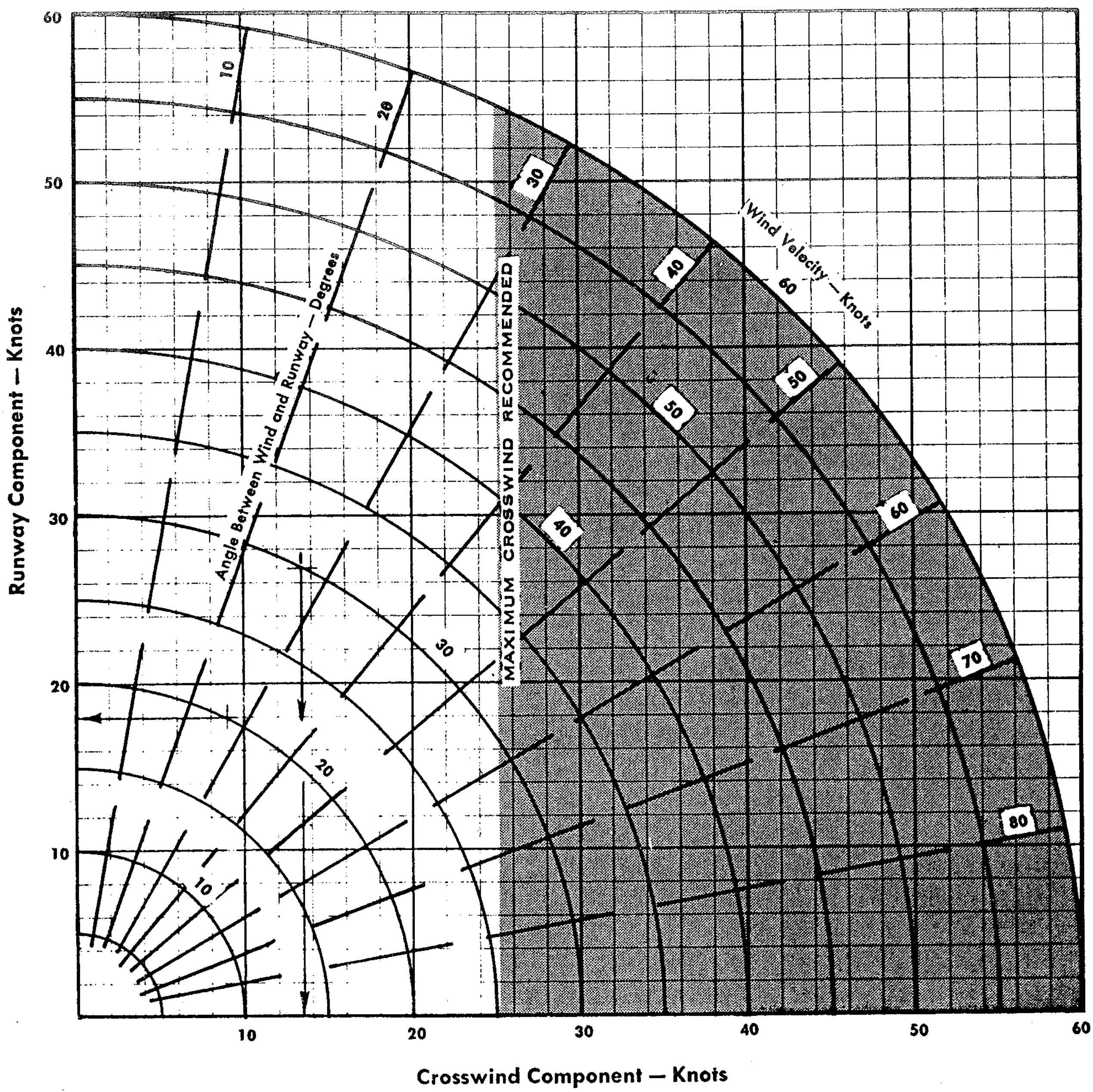
l. The acceleration check marker is reached 2000 ft before the go, no-go marker. Ground run distance to this marker (No. 7) is the go, no-go distance, or 2875 ft (4875 − 2000 ft) from the brake release point.

m. Enter Figure A2-8 with the acceleration check distance, 2875 ft. At the intersection with the normal acceleration guide line read the normal speed at the acceleration marker (zero wind) of 134 knots.

n. The runway wind component (18 knots headwind) added to the normal speed at the acceleration check distance (134 knots) provides the expected speed (152 knots) at the acceleration check distance.

When the acceleration check speed is marginal, the speed at the go, no-go marker should be given close attention. If the altitude for the above problem were 6000 ft with a 10-knot tailwind, Figure A2-4 shows that the takeoff groundspeed would be 253 knots for the normal speed schedule. A maximum performance takeoff is desirable to reduce tire speed. The ground run is reduced 5½ percent as a result. Total distance to reach 50 ft would be approximately the same as for a normal takeoff if the aircraft accelerates to the normal target speed as 50 ft height is reached. The go, no-go and acceleration check analysis are made by the methods described above.

WIND COMPONENT CHART
WITH OR WITHOUT DRAG CHUTE



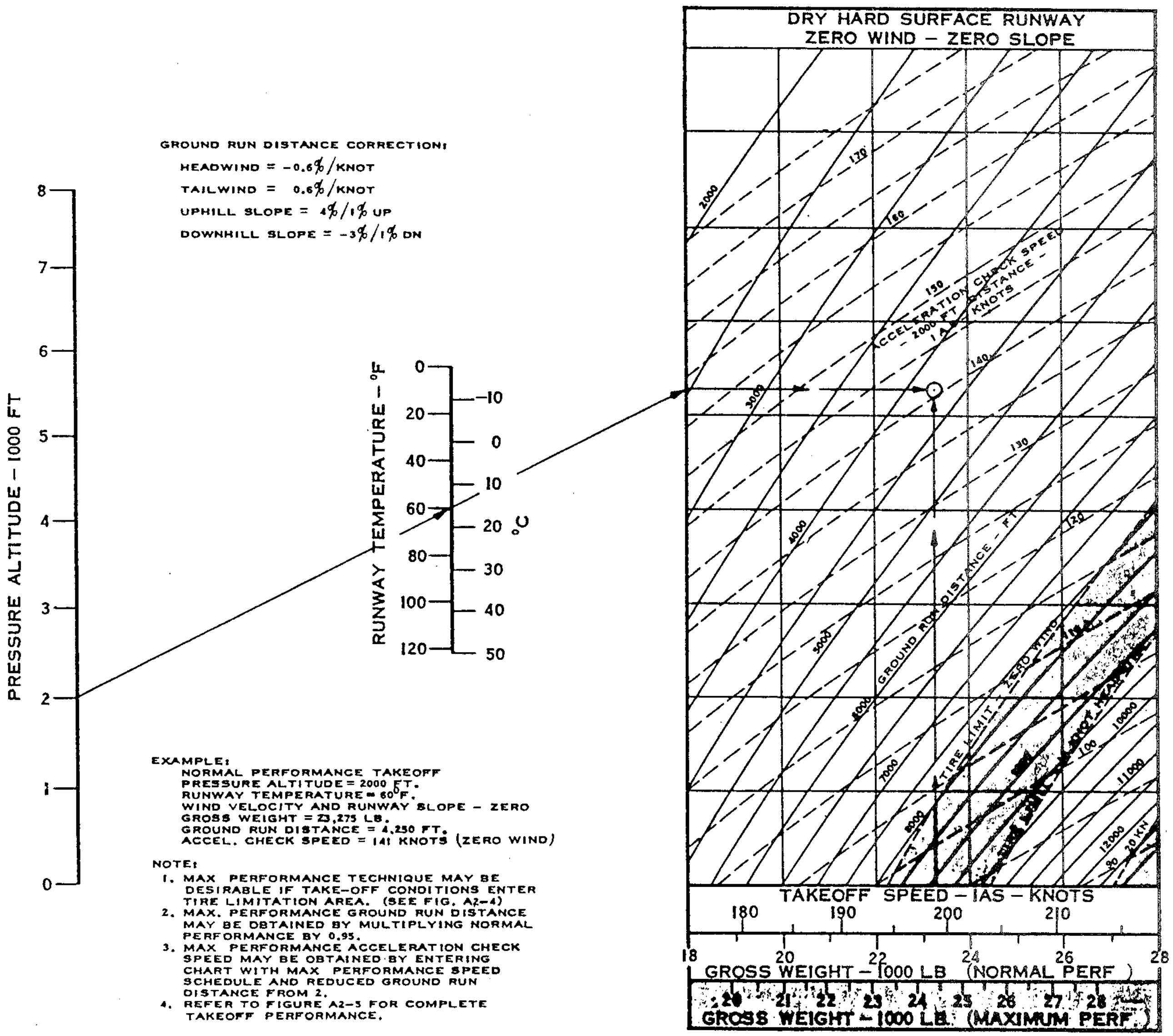
(NOTE: FOR CROSSWIND COMPONENT ENTER CHART WITH MAXIMUM REPORTED GUST VELOCITY)

Figure A2-2

NORMAL PERFORMANCE TAKEOFF GROUND RUN DISTANCE AND 2000 FT ACCELERATION DISTANCE CHECK SPEED MAXIMUM THRUST

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



GROUND RUN DISTANCE CORRECTION:
HEADWIND = -0.6%/KNOT
TAILWIND = 0.6%/KNOT
UPHILL SLOPE = 4%/1% UP
DOWNHILL SLOPE = -3%/1% DN

EXAMPLE:
NORMAL PERFORMANCE TAKEOFF
PRESSURE ALTITUDE = 2000 FT.
RUNWAY TEMPERATURE = 60°F.
WIND VELOCITY AND RUNWAY SLOPE – ZERO
GROSS WEIGHT = 23,275 LB.
GROUND RUN DISTANCE = 4,250 FT.
ACCEL. CHECK SPEED = 141 KNOTS (ZERO WIND)

NOTE:
1. MAX PERFORMANCE TECHNIQUE MAY BE DESIRABLE IF TAKE-OFF CONDITIONS ENTER TIRE LIMITATION AREA. (SEE FIG. A2-4)
2. MAX. PERFORMANCE GROUND RUN DISTANCE MAY BE OBTAINED BY MULTIPLYING NORMAL PERFORMANCE BY 0.95.
3. MAX PERFORMANCE ACCELERATION CHECK SPEED MAY BE OBTAINED BY ENTERING CHART WITH MAX PERFORMANCE SPEED SCHEDULE AND REDUCED GROUND RUN DISTANCE FROM 2.
4. REFER TO FIGURE A2-5 FOR COMPLETE TAKEOFF PERFORMANCE.

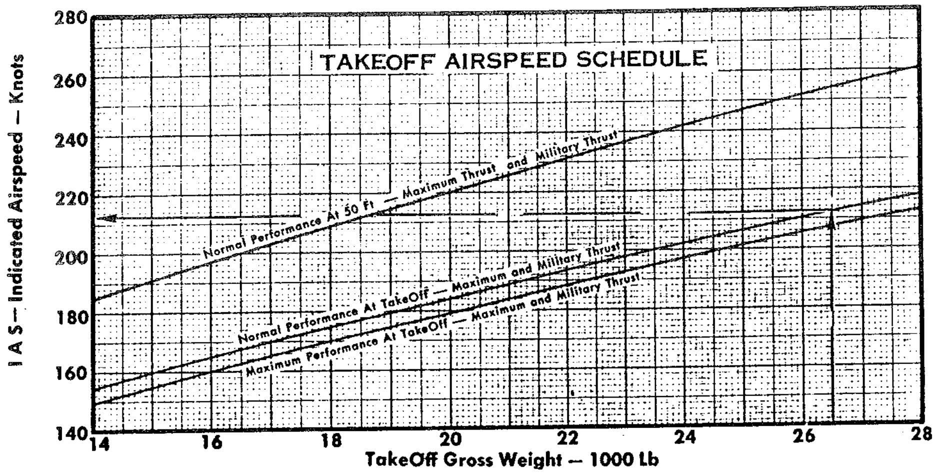
Figure A2-3

TAKEOFF SPEED SCHEDULES AND DETERMINATION OF TECHNIQUE

All Configurations

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



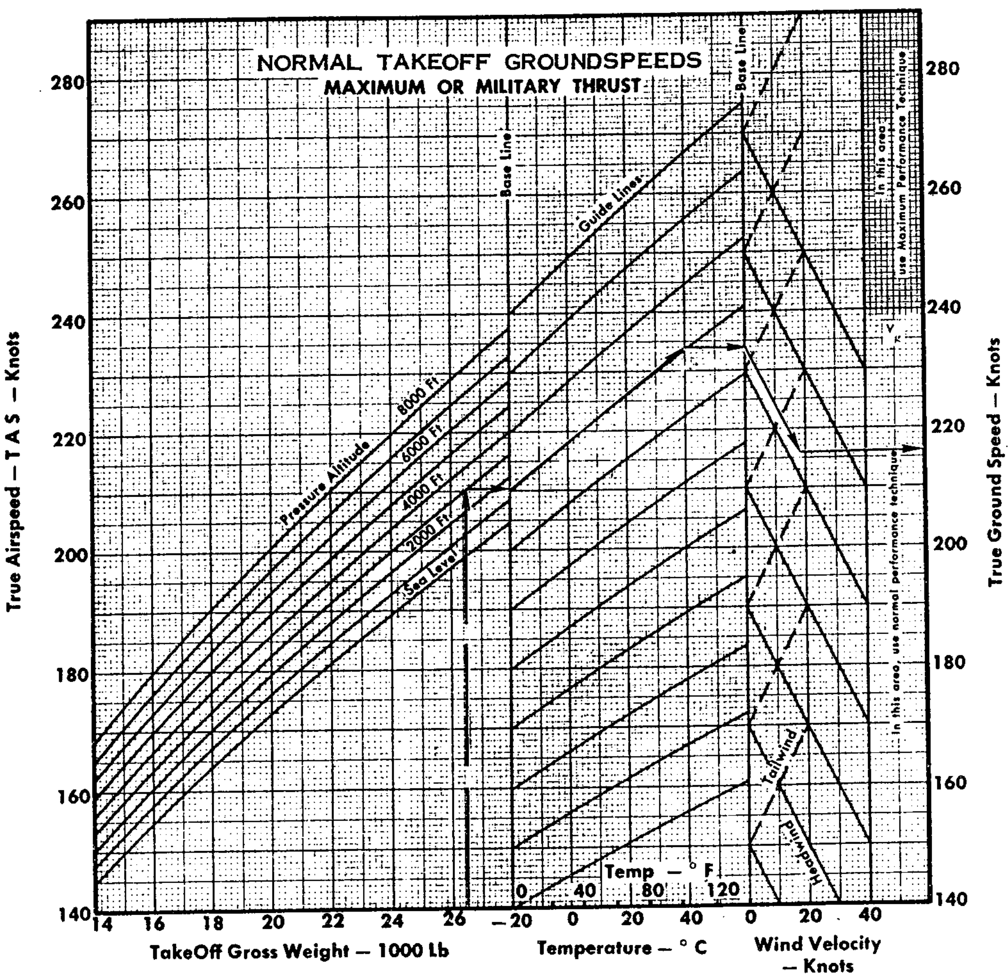


Figure A2-4

TAKEOFF GROUND RUN DISTANCE — MAXIMUM THRUST

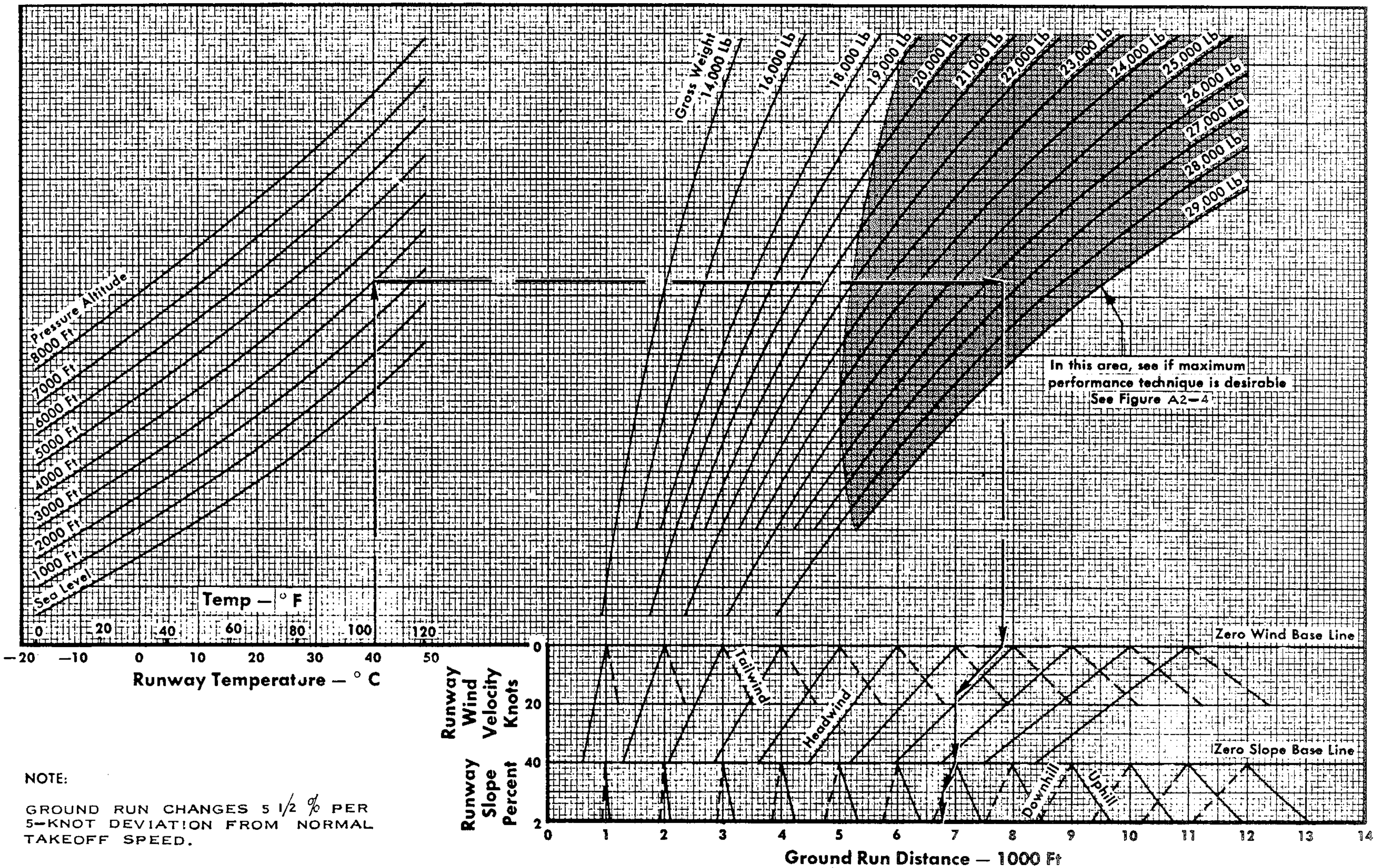
Hard Surface Level Runway

Normal Performance

All Configurations

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE:

GROUND RUN CHANGES 5 1/2 % PER 5-KNOT DEVIATION FROM NORMAL TAKEOFF SPEED.

Figure A2-5

REFUSAL SPEED — MAXIMUM THRUST

WITH DRAG CHUTE

Anti-Skid Brakes　　All Configurations　　18-Foot Drag Chute

Zero Wind　　Zero Slope

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

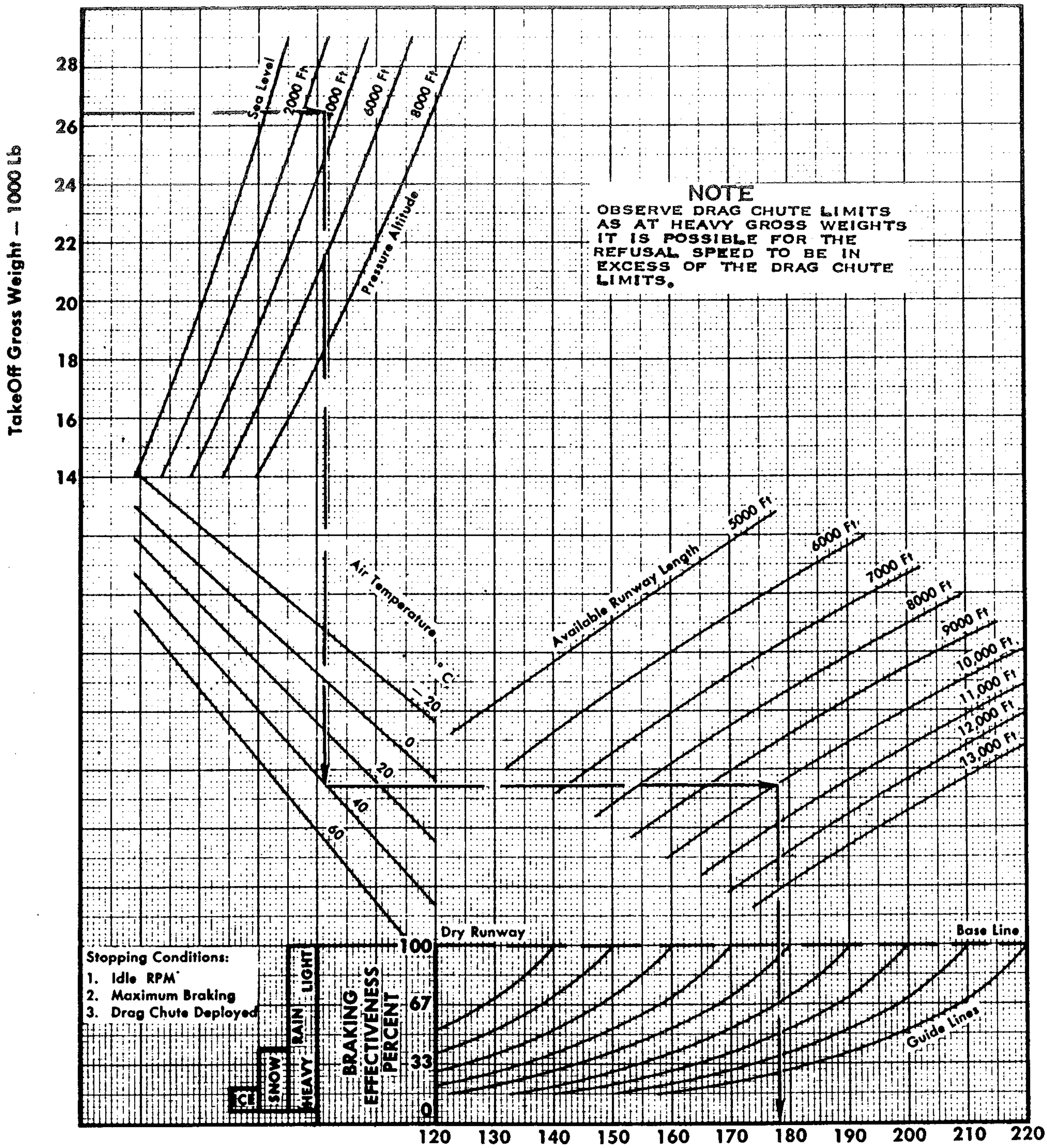


Figure A2-6

REFUSAL SPEED — MAXIMUM THRUST

WITHOUT DRAG CHUTE

All Configurations

Zero Wind Zero Slope

Anti-Skid Brakes

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

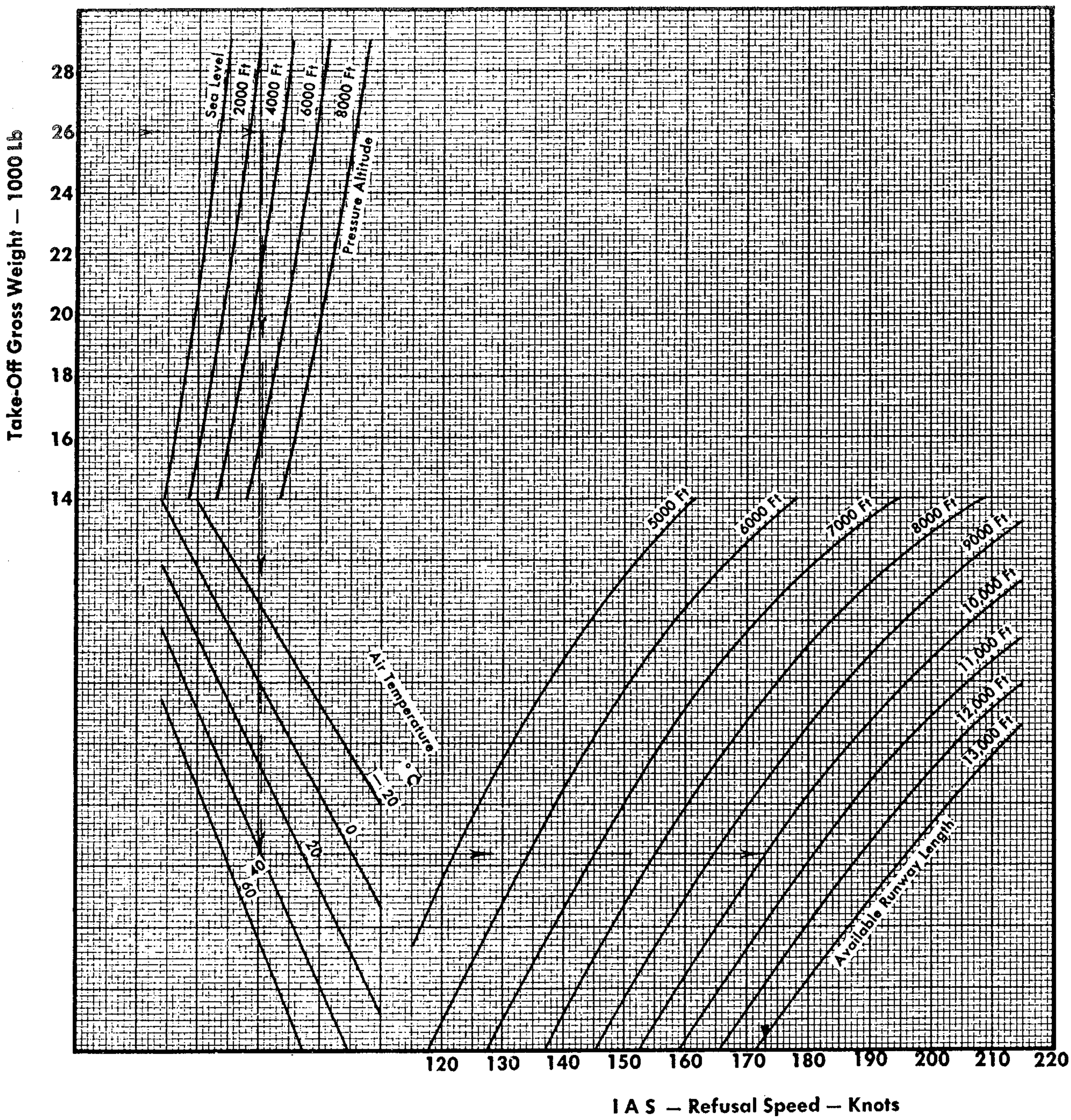


Figure A2-7

VELOCITY DURING TAKEOFF GROUND RUN — MAXIMUM THRUST

All Configurations
Hard Surface Level Runway

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

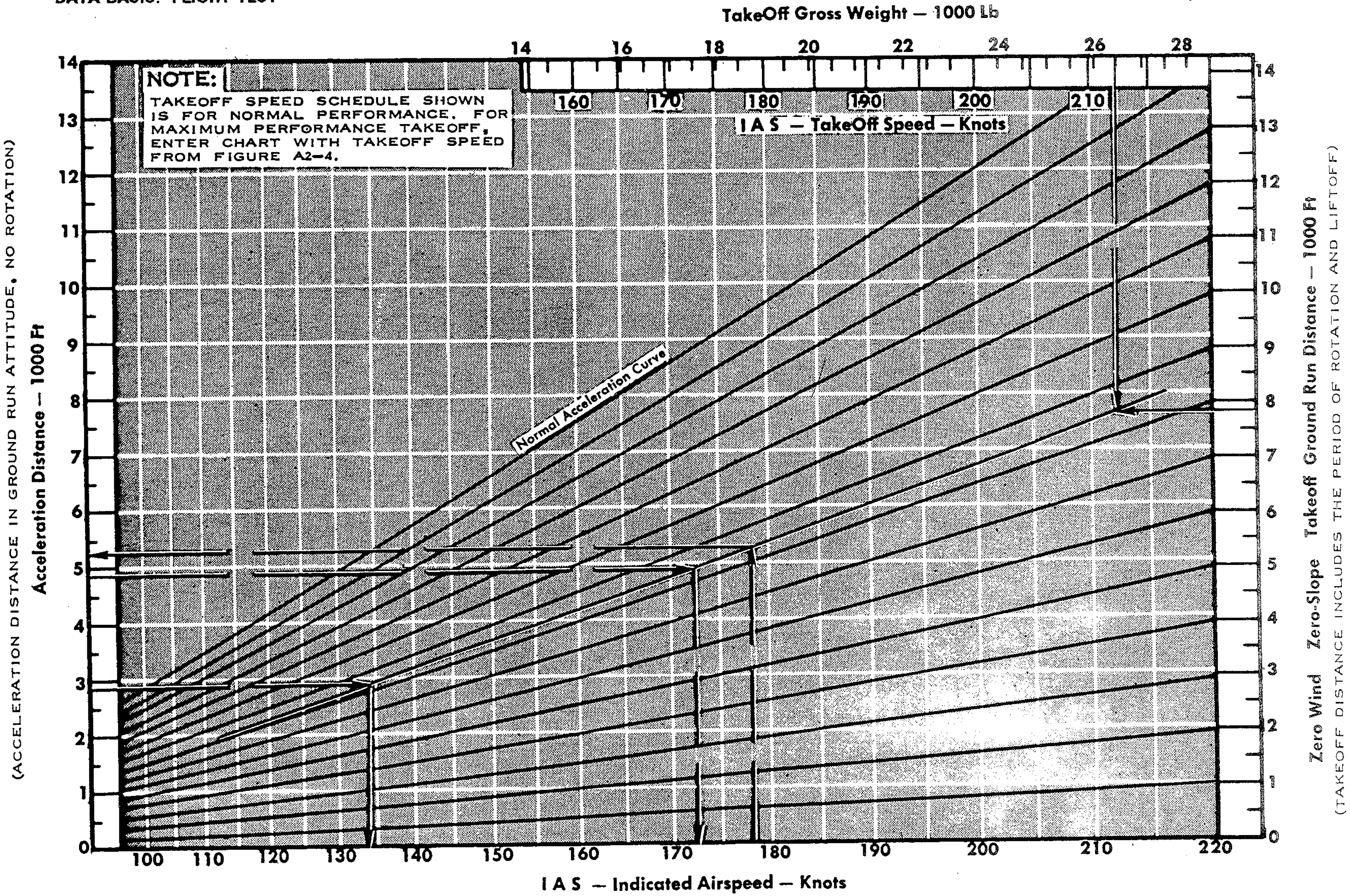


Figure A2-8

AIRSPEED ACCELERATION TOLERANCE

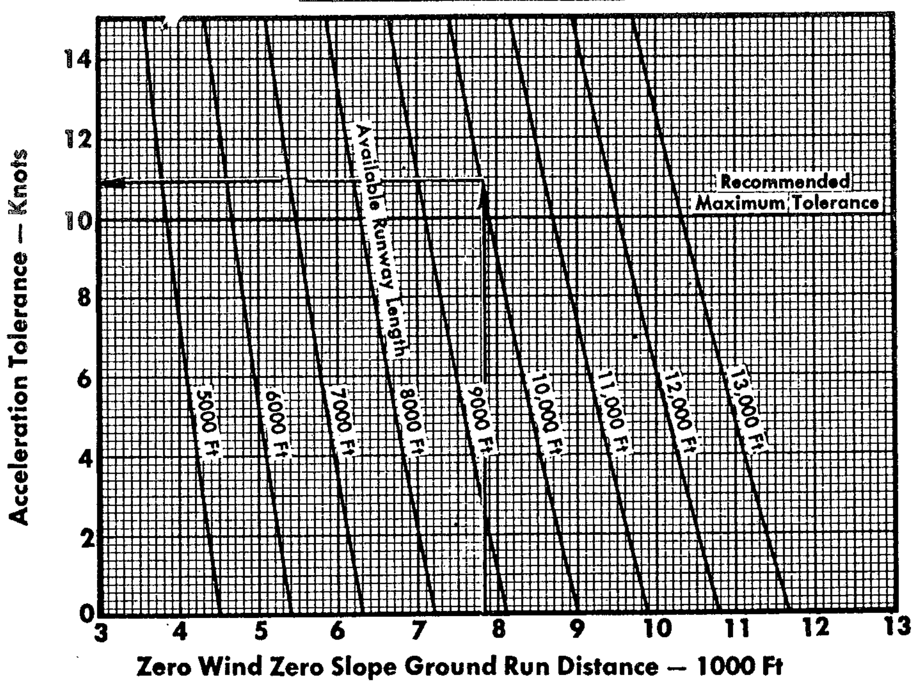
Zero Wind Zero Slope

Dry Hard Surface Runway

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

MAXIMUM THRUST



MILITARY THRUST

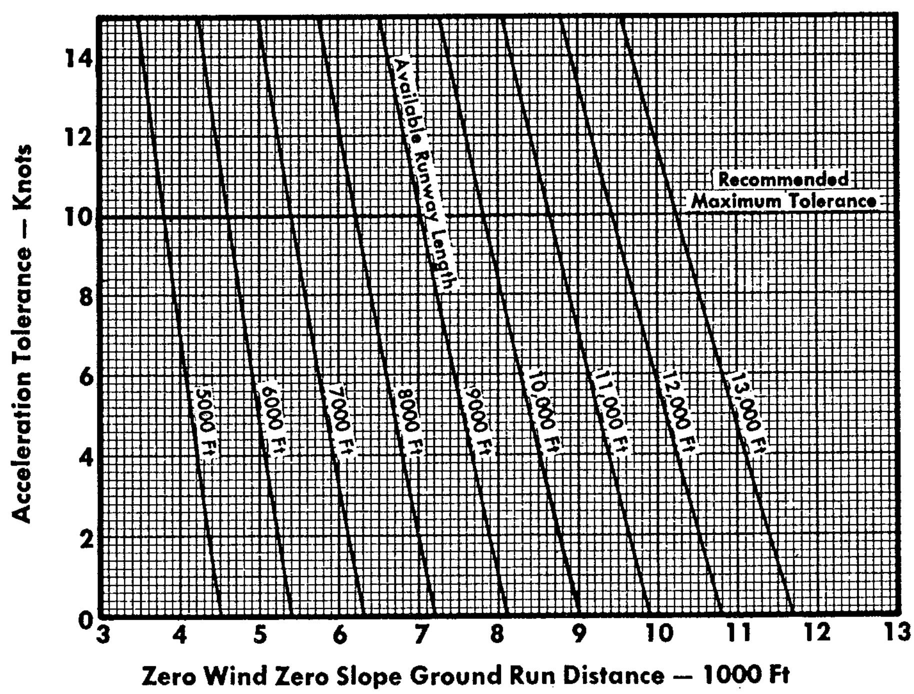


Figure A2-9

TAKEOFF DISTANCE OVER A 50 FOOT OBSTACLE — MAXIMUM THRUST

Normal Performance

Dry Hard Surface Runway

All Configurations

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

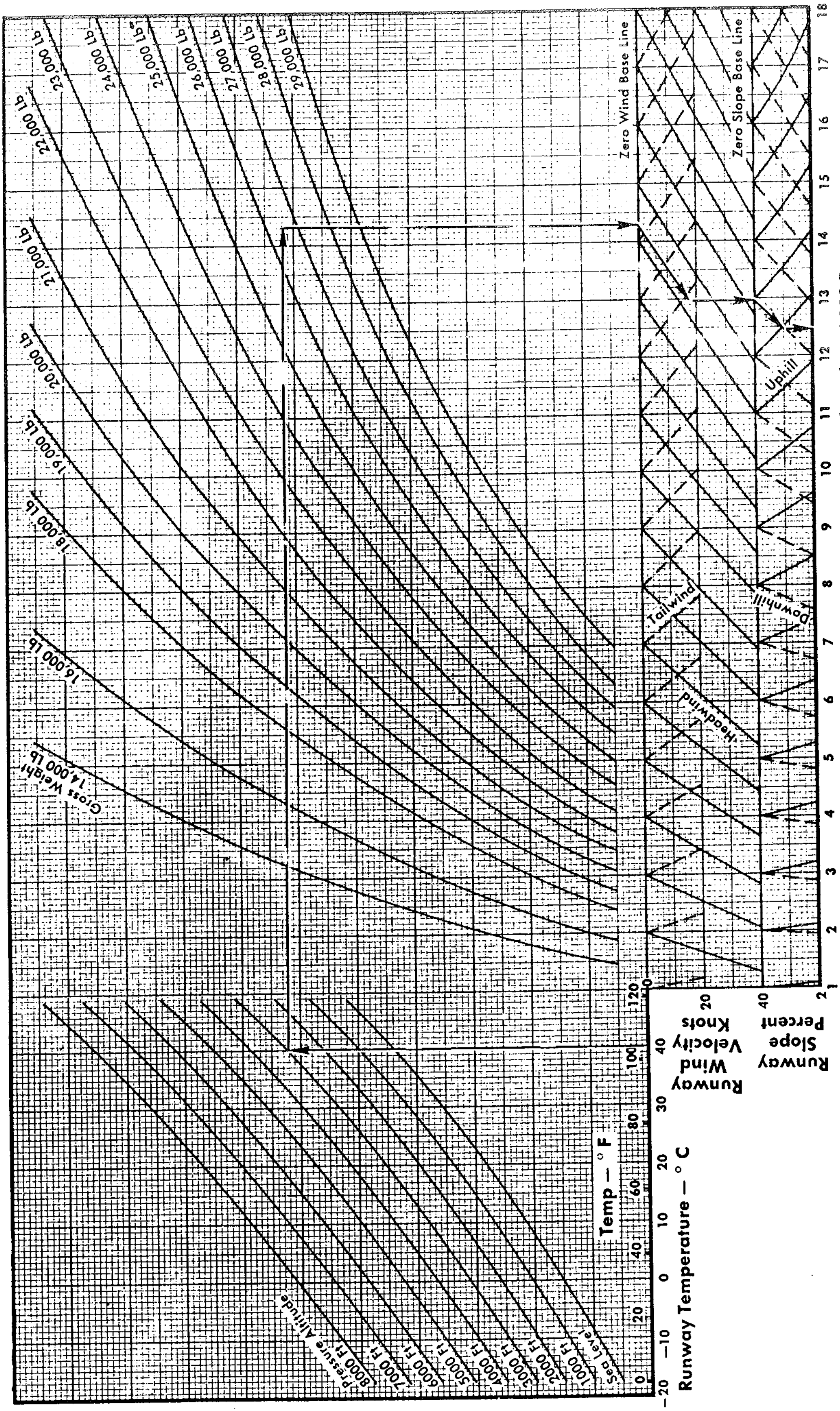


Figure A2-10

TAKEOFF GROUND RUN DISTANCE — MILITARY THRUST

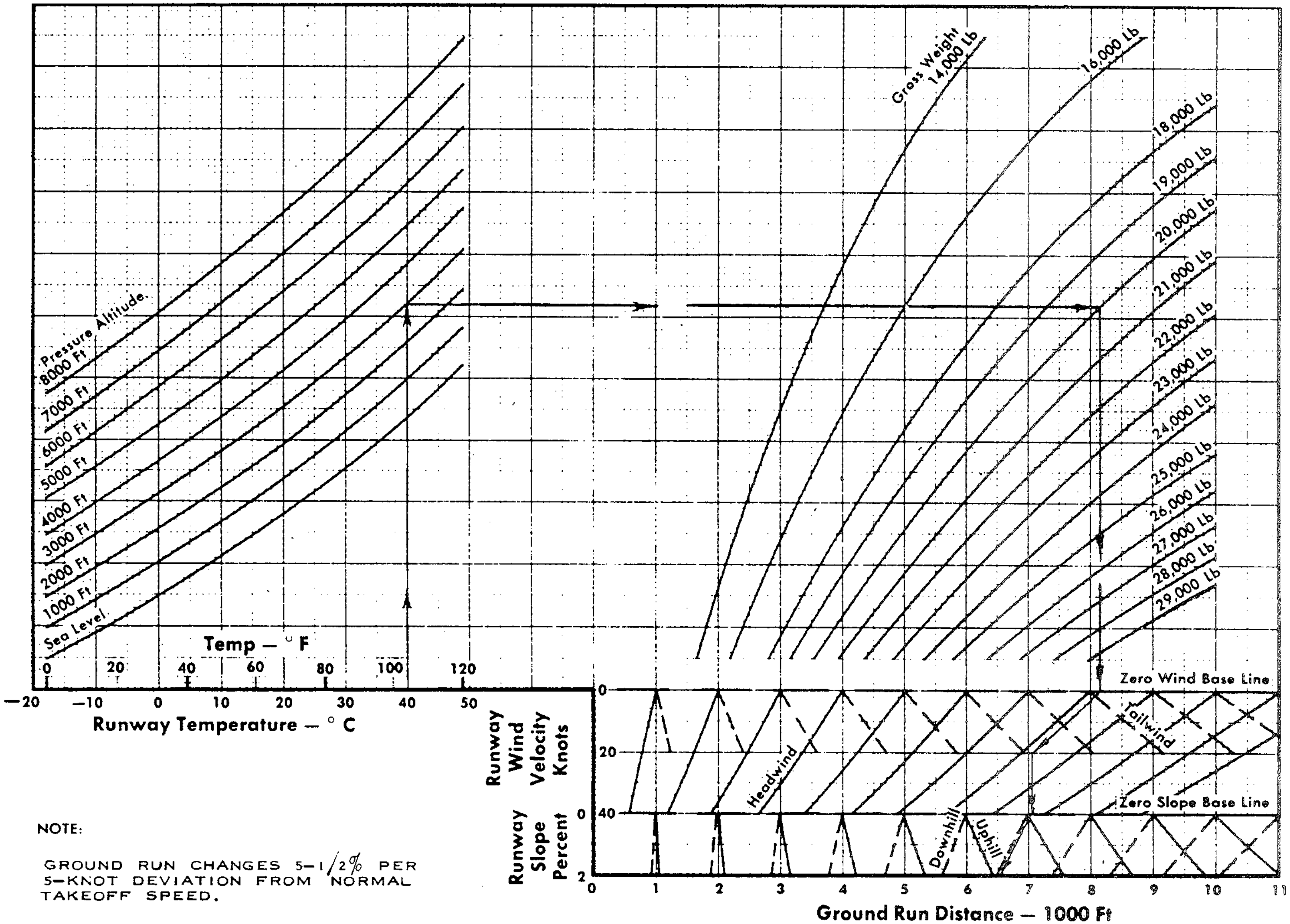
Hard Surface Level Runway

Normal Performance

All Configurations

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE:

GROUND RUN CHANGES 5-1/2% PER 5-KNOT DEVIATION FROM NORMAL TAKEOFF SPEED.

Figure A2-11

REFUSAL SPEED — MILITARY THRUST

WITH DRAG CHUTE

Anti-Skid Brakes   All Configurations   18-Foot Drag Chute

Zero Wind   Zero Slope

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

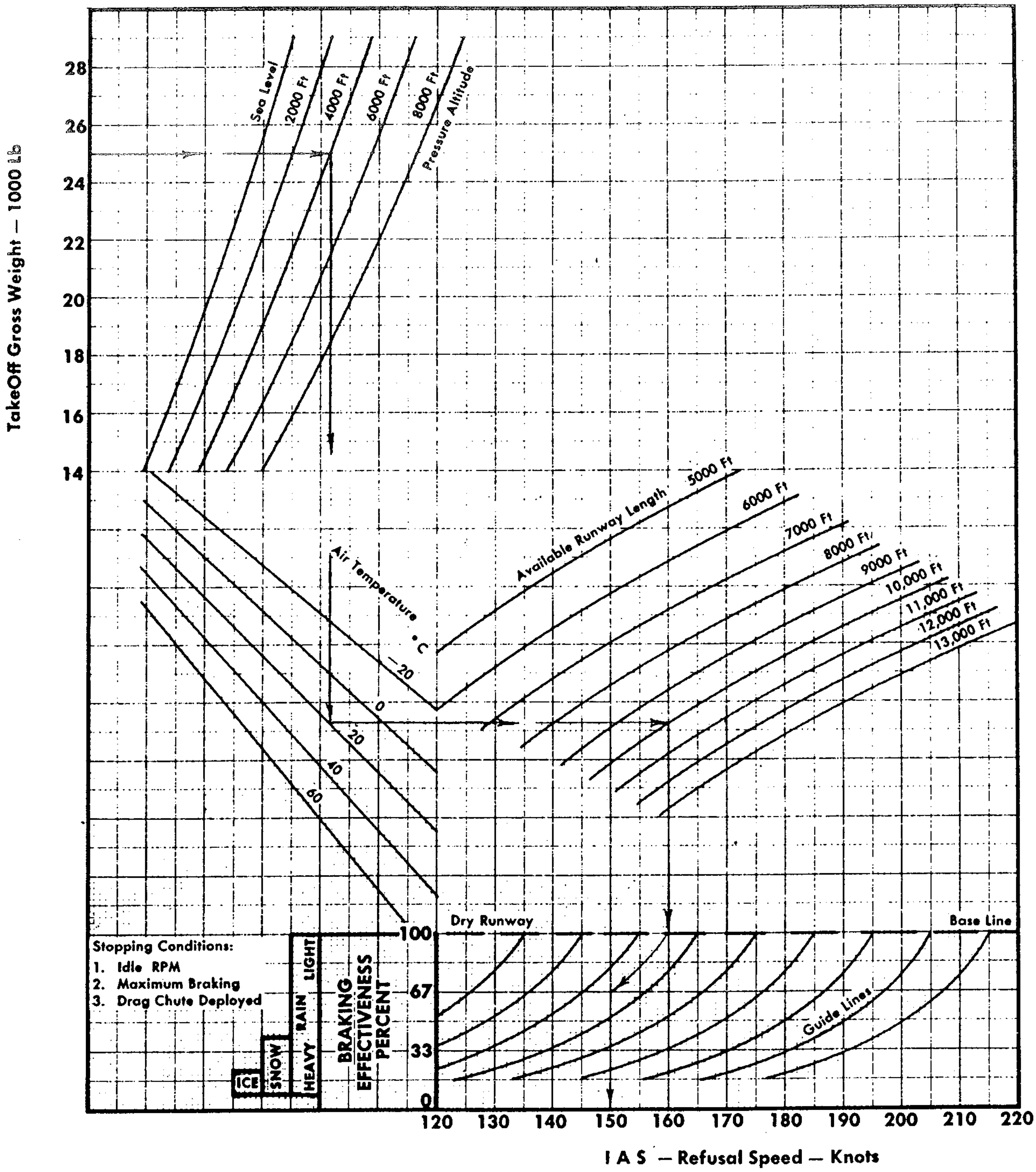


Figure A2-12

VELOCITY DURING TAKEOFF GROUND RUN — MILITARY THRUST

All Configurations

Hard Surface Level Runway

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

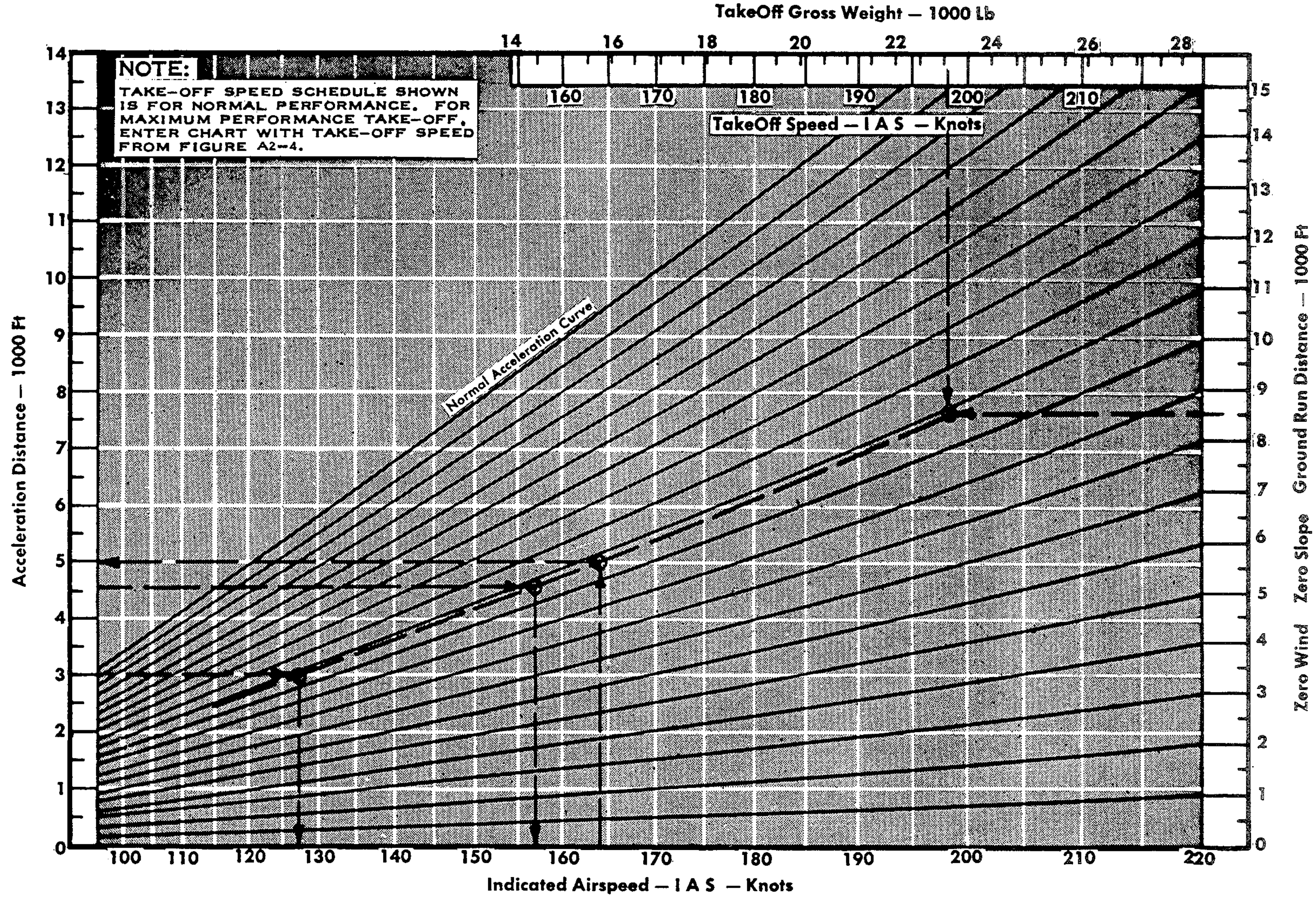


Figure A2-13

TAKEOFF DISTANCE OVER A 50 FOOT OBSTACLE — MILITARY THRUST

Normal Performance   Dry Hard Surface Runway

All Configurations

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

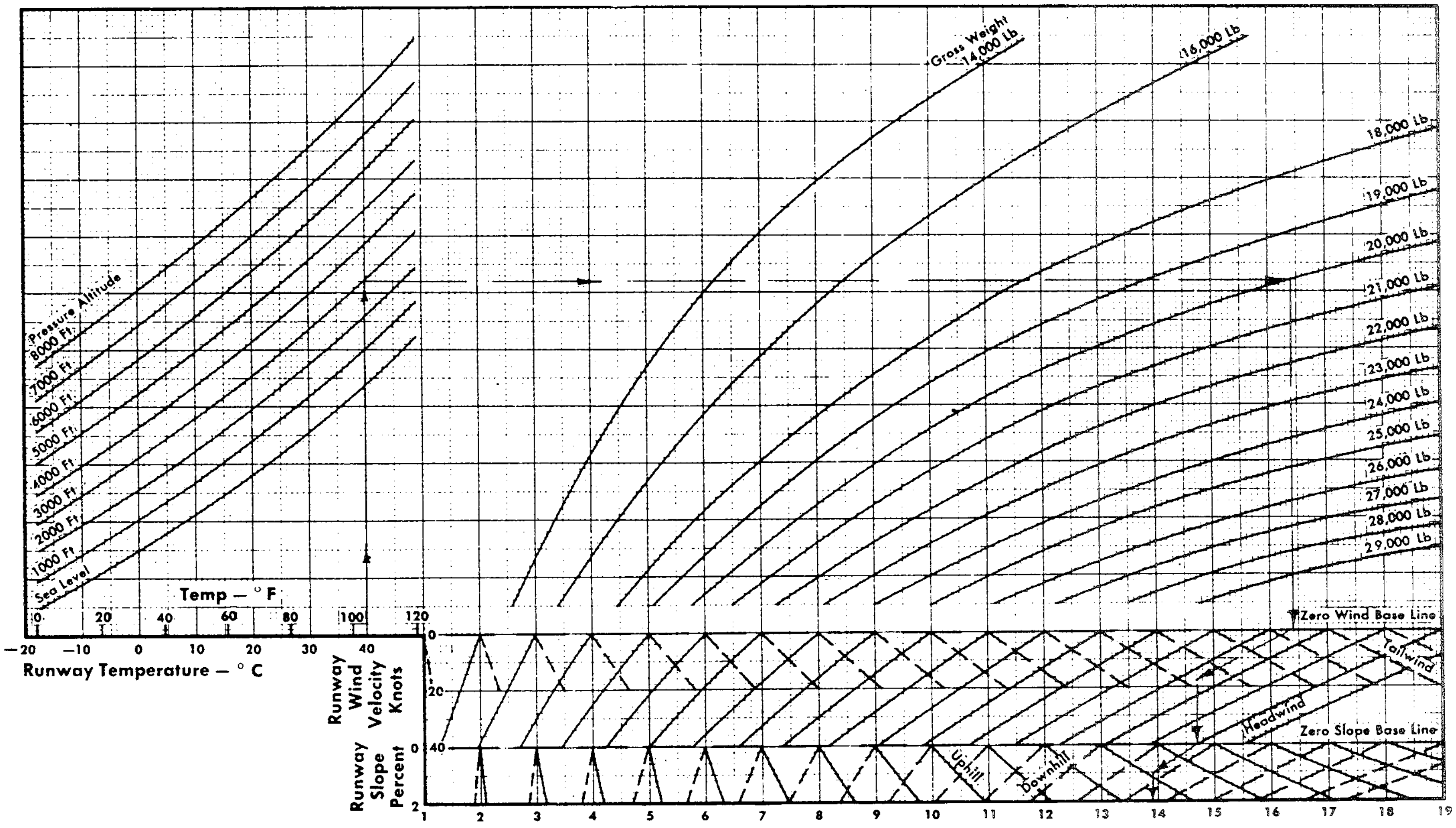


Figure A2-14

PART 3

CLIMB

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

CLIMB CONTROL CHARTS

The climb control charts present data for planning the subsonic climb segments of flight. Performance curves show time, fuel used, and distance traveled. The curves are directly applicable to climbs initiated at or near sea level, and may be used on an incremental basis for climbs started at higher levels. Data are provided for operation at either Military or Maximum thrust settings. Supersonic climb performance data are provided in Part 9.

CLIMB CONTROL INDEX

The climb control charts are indexed into groups of configurations having similar rates of climb. The small differences in operational performance among configurations in a given group may be disregarded. Typical configurations applicable to the performance index presented are illustrated on a chart in Part 1. If a climb performance is desired for a configuration not illustrated, refer to the index system shown in Part 1, determine the configuration drag index and use the appropriate climb control chart.

CLIMB SPEED SCHEDULES

Subsonic Maximum thrust climb control is conducted at speed schedules of Mach 0.925 and 0.90; Military thrust at speed schedules of Mach 0.90 and 0.85. In general, the higher Mach schedule is used for aircraft in configuration with no external stores, tip stores, or tip. When stores are carried on the wing pylons (two stores), or pylons, with or without tip stores, the increase in drag and drag rise characteristics in these configurations necessitates a lower Mach schedule to maintain a level of excess thrust for best performance.
Occasionally, in order to clear local traffic patterns, etc., it may be desirable to initiate Military thrust climbs at speeds lower than those listed. In such cases, start the climb at the speed desired (such as afterburner-off speed) and hold constant IAS until the normal Mach number schedule is obtained at altitude. Then, climb at the recommended Mach number until cruise altitude is reached. Military and Maximum thrust climb performance for alternate schedules of 400 KIAS are included. For Military thrust climb control at Mach 0.90 (configuration drag index of 30; 40 with tip tanks), without pylon mounted stores, read time, fuel used and distance traveled in the climb from the climb control charts at Mach 0.85.

TRANSITION TIME AND FUEL ALLOWANCES

Information regarding fuel used and time to accelerate to climb speed from the point of brake release is shown in Figure A3-1 for a range of gross weight. A table of Mach numbers for 350 KIAS, 400 KIAS and 450 KIAS are also included in the chart to provide for a combination usage of Maximum and Military thrust where the thrust setting is changed at that speed. A table is included on the chart for fuel allowances during various periods of ground maneuver at idle throttle. The values are based on a nominal fuel flow and should be adjusted if service experience indicates another value is more suitable.

DETERMINATION OF EQUIVALENT WEIGHT

The procedure for determining equivalent weight for transition time and fuel allowances is as follows:

1. Determine the OAT (ambient air temperature) expected from meteorological forecasts.

2. Enter the standard altitude table (in Part 1) at the field pressure altitude and obtain the standard-day temperature.

3. Determine the difference between the standard-day temperature (step 2) and the OAT (step 1). This is the Δ temperature.

4. Using the gross weight correction on the chart, calculate the weight increment correction. Δ weight = (Δ temp° C/10° C) × weight factor.

5. Determine the equivalent gross weight by using the suitable non-standard temperature equation.

    a. Hot day: equivalent weight = actual aircraft gross weight + Δ weight.

    b. Cold day: equivalent weight = actual aircraft gross weight − Δ weight.

6. At field pressure altitudes above sea level determine the weight increment correction and, if necessary, add to equivalent gross weight in step 5. Δ weight = (Δ altitude/1000 feet) × weight factor.

CLIMB PERFORMANCE

The subsonic climb control charts are plots of pressure altitude versus distance, time, and fuel used for various gross weights. The charts are read directly for climbs started from sea level, and on an incremental basis for climbs initiated from higher altitudes and terminated at optimum cruise altitudes. The performance includes weight reduction due to fuel consumption and reflects the increase in rates of climb that result.

EFFECT OF FREE AIR TEMPERATURE ON CLIMB PERFORMANCE

The climb charts may be read directly to obtain performance data for standard-day conditions. When the free air temperature is higher than standard (a hot day) the aircraft will perform as though the weight is greater than actual. When the free air temperature is lower than standard (a cold day) the aircraft will perform as though the weight is less than actual.
A Δ temperature grid is included on each climb control chart for the determination of nonstandard-day climb performance. The Δ temperature for the climb is determined as follows:

1. Determine the average OAT (ambient air temperature) expected during the climb to desired altitude.

2. Enter the standard altitude table (in Part 1) at the average pressure altitude during the climb and obtain the standard-day temperature.

3. The difference between the standard-day temperature (step 2) and the average OAT (step 1) is the Δ temperature.

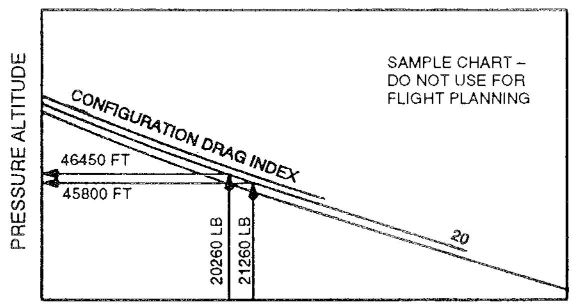
COMBAT CEILING

Combat ceiling (500 fpm rate of climb) for either Maximum or Military thrust, at the appropriate Mach schedule, may be obtained from the summary curves provided. The combat ceiling is based on the actual gross weight at altitude. To determine the climb performance to combat ceiling, estimate the altitude based on the initial climb gross weight (gross weight corrected for fuel consumed during ground maneuver, takeoff and acceleration to climb speed). Refer to the appropriate climb control chart and read the fuel consumed to the estimated altitude. Subtract the fuel consumed from the initial climb gross weight and obtain a corrected combat ceiling for the fuel used in the climb. A sample problem illustrates use of the chart.

SAMPLE PROBLEM

Determine the standard day combat ceiling for a maximum thrust climb from sea level at Mach 0.925. The aircraft has a configuration drag index of 20 and a loaded gross weight of 22000 pounds.

MAXIMUM THRUST COMBACT CEILING



FA0320

a. From Figure A3-1 determine the fuel required for ground maneuver, takeoff and accelerate to climb speed.

   Ground maneuver for 8.5 minutes .... 190 lb
   Takeoff and accelerate to climb speed .. 550 lb
   TOTAL fuel used .... 740 lb

b. Initial climb gross weight is 21260 lb (22000 − 740).

c. Enter Figure A3-2 at 21260 pounds and read an estimated combat ceiling of 45800 feet.

d. Enter Figure A3-6 (sheet 2 of 2) "Maximum Thrust Climb Control", Mach 0.925 at 21260 pounds. Proceed to a pressure altitude of 45800 feet. For a configuration drag index of 20 read climb fuel used as 1000 pounds.

e. Gross weight at the estimated combat ceiling (45800 feet) is 20260 pounds (21260 − 1000).

f. Reenter Figure A3-2 at 20260 pounds and read a corrected combat ceiling, 46450 feet.

CLIMB PERFORMANCE

Sample Problem

Determine the climb performance for a military thrust climb at Mach 0.85 from sea level to optimum cruise altitude. The aircraft has a configuration drag index of 60 and loaded gross weight of 26000 pounds. Ambient temperature at sea level and the average temperature during the climb is 10° C below standard (ΔT = −10° C).

a. For 8.5 minutes of ground maneuvering, maximum thrust acceleration to 350 KIAS and military thrust acceleration to climb Mach 0.85, determine the fuel required and initial climb gross weight.

   Ground maneuver fuel .... 190 lb

   Maximum thrust acceleration gross weight is 25810 (26000 − 190)

   Fuel required to 350 KIAS .... 370 lb

   Gross weight at 350 KIAS .... 25440 lb

   Military thrust acceleration gross weight is 25440 (25810 − 370)

   Fuel required to Mach 0.85 .... 150 lb

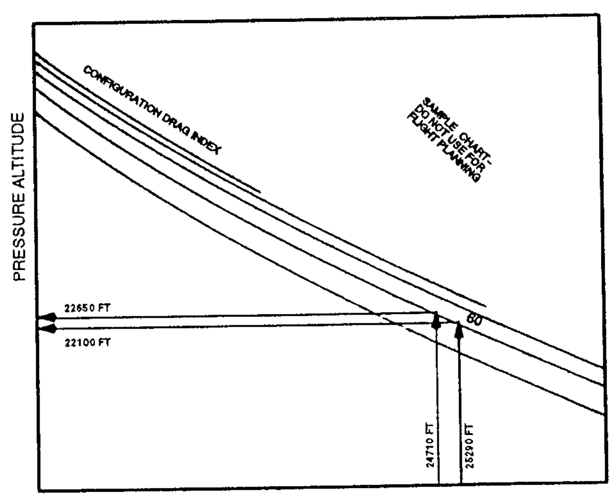
   Gross weight at start of climb .... 25290 lb

b. Enter Figure A3-4, "Optimum Cruise Altitude", at the initial climb gross weight, 25290 lb, and read an estimated altitude, 22100 feet.

c. Enter Figure A3-10 (sheet 2) at the initial climb gross weight, correct for ΔT = −10° C, and proceed to the optimum cruise altitude, 22100 feet. For a configuration drag index of 60, fuel used is 580 pounds. Gross weight at the optimum cruise altitude is 24710 lb (25290 − 580).

d. Reenter Figure A3-4 at 24710 lb and read a corrected optimum cruise altitude, 22650 feet.

OPTIMUM CRUISE ALTITUDE



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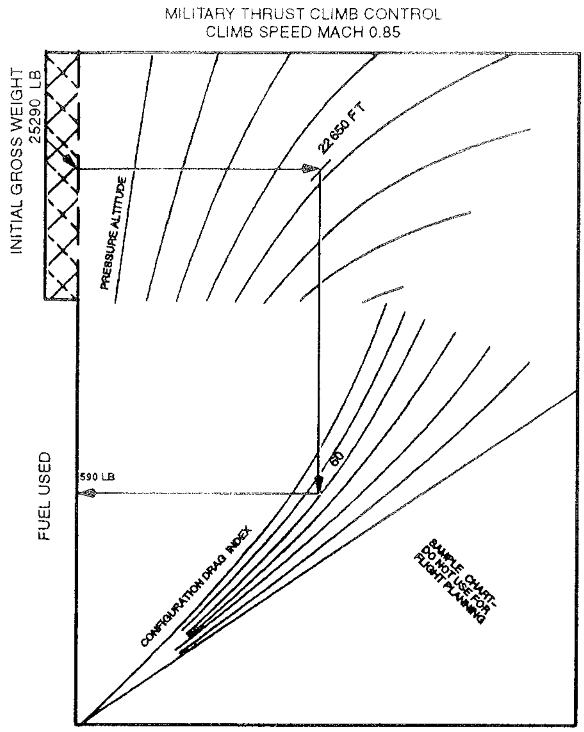

e. Reenter Figure A3-10 at the initial climb gross weight 25290 lb, correct for $\Delta T = -10°$ C, proceed to the optimum cruise altitude, 22650 feet. For a configuration drag index of 60, read time, distance and fuel used.

Time .... 4.7 min.
Distance .... 42 n.m.
Fuel used .... 590 lb.

OPTIMUM CRUISE ALTITUDE

The optimum altitude for maximum range cruise may be obtained from the summary plot provided. The altitude is based on actual gross weight. The altitude shall be corrected for fuel consumed in the climb. Determination of the corrected altitude and the climb performance are illustrated in the sample problem.

CRUISE ALTITUDE – SHORT RANGE

Maximum range operation at distances of approximately 250 nautical miles or less, including a Military thrust climb, may require flight planning at altitudes below optimum cruise altitude (cruise-climb). Under some conditions of heavy gross weight or high configuration drag index, it is more advantageous to terminate the climb at a lower altitude than optimum cruise, to initiate a cruise fuel comsumption less than would otherwise be required at Military thrust to reach optimum cruise altitude. The cruise altitude summary in Figure A3-5, illustrates such an altitude and when used in conjunction with the maximum range constant altitude charts, will provide a prediction of minimum fuel for short range operation.

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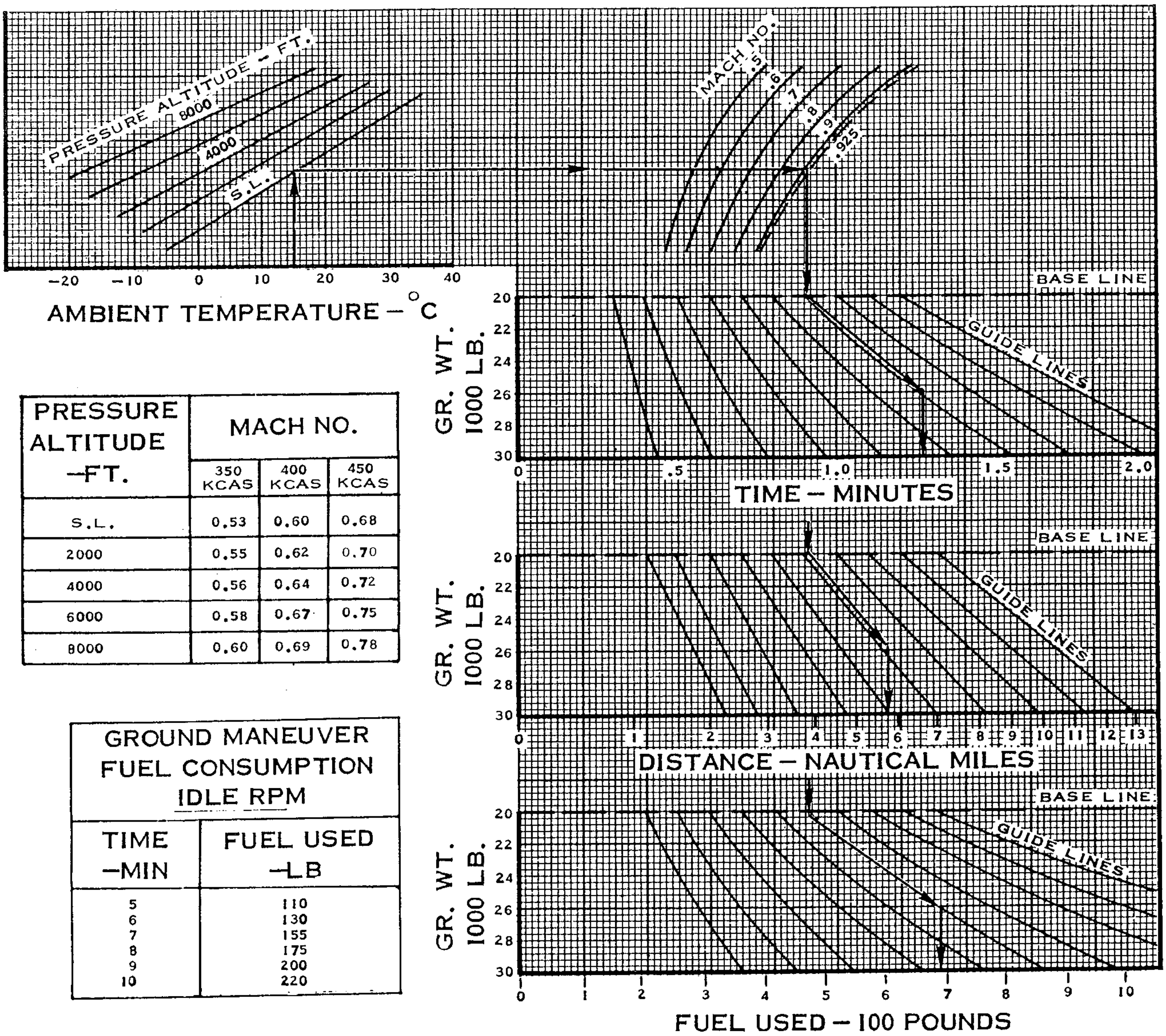
MAXIMUM THRUST

TIME, DISTANCE AND FUEL REQUIRED FROM BRAKE RELEASE TO CLIMB SPEED

Model: F-104G
Date: 15 October 1967
DATA BASIS: FLIGHT TEST



PRESSURE ALTITUDE –FT.	MACH NO.		
	350 KCAS	400 KCAS	450 KCAS
S.L.	0.53	0.60	0.68
2000	0.55	0.62	0.70
4000	0.56	0.64	0.72
6000	0.58	0.67	0.75
8000	0.60	0.69	0.78

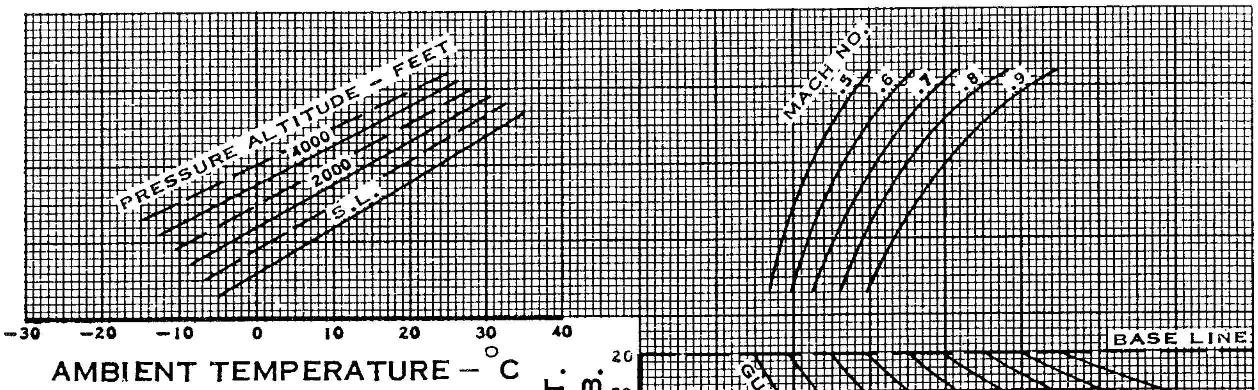
GROUND MANEUVER FUEL CONSUMPTION IDLE RPM	
TIME –MIN	FUEL USED –LB
5	110
6	130
7	155
8	175
9	200
10	220

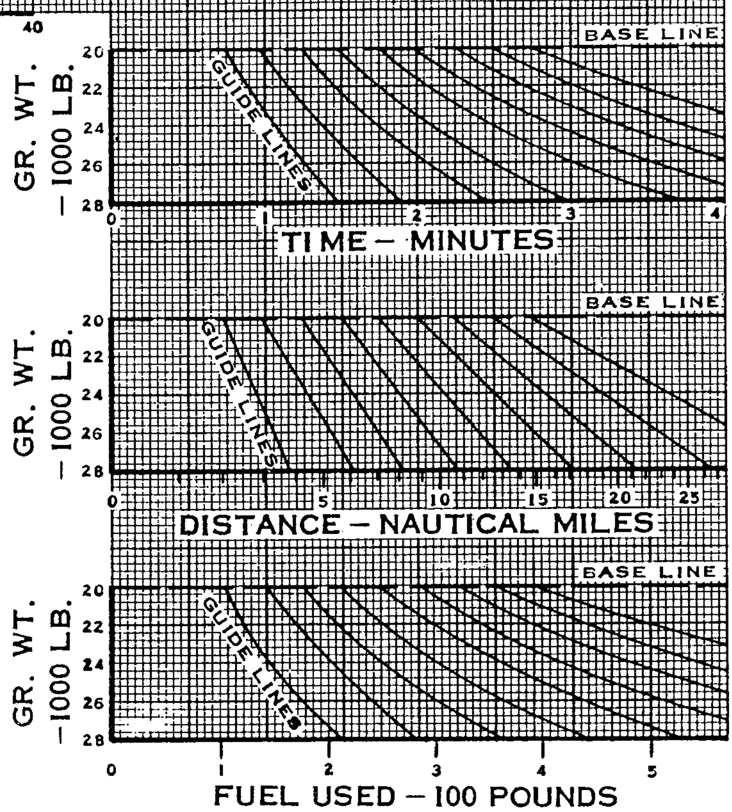
Figure A3-1 (Sheet 1 of 2)

MILITARY THRUST

TIME, DISTANCE AND FUEL REQUIRED FROM BRAKE RELEASE TO CLIMB SPEED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal





PRESSURE ALTITUDE – FT.	MACH NO.		
	350 KCAS	400 KCAS	450 KCAS
S.L.	0.53	0.60	0.68
2000	0.55	0.62	0.70
4000	0.56	0.64	0.72
6000	0.58	0.67	0.75
8000	0.60	0.69	0.78

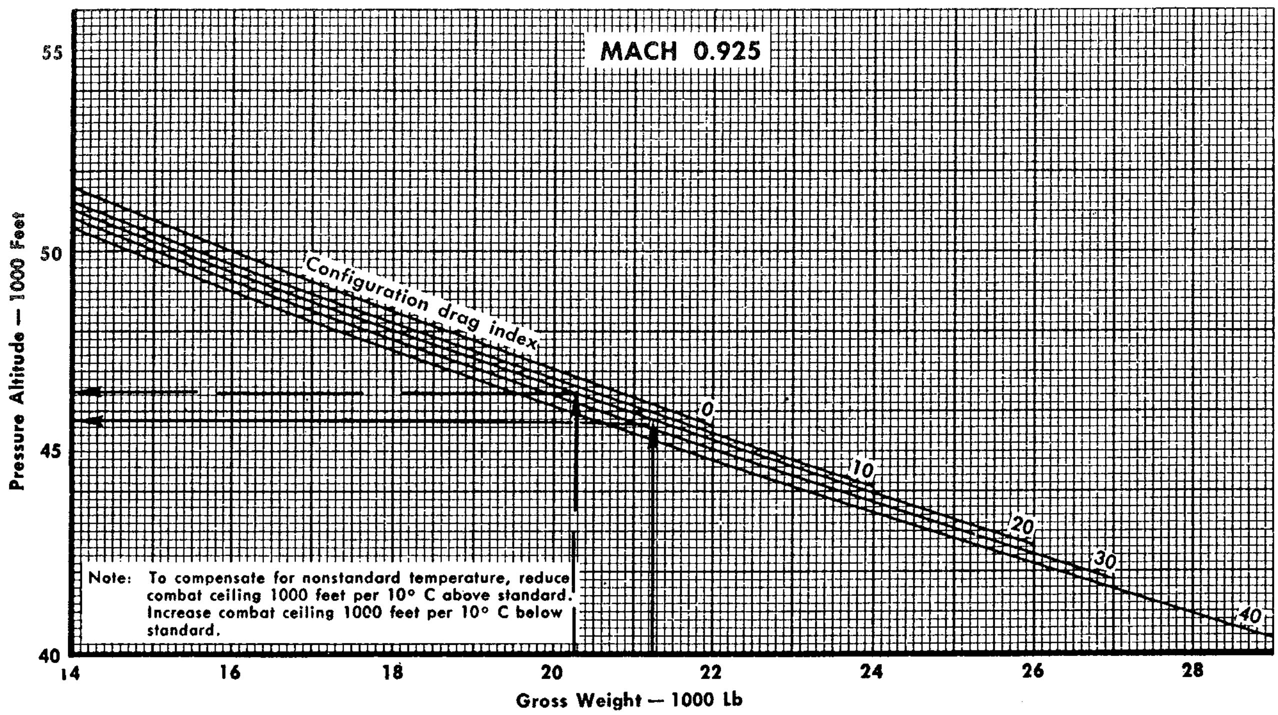
GROUND MANEUVER FUEL CONSUMPTION IDLE RPM	
TIME – MIN.	FUEL USED – LB.
5	110
6	130
7	155
8	175
9	200
10	220

Figure A3-1 (Sheet 2 of 2)

MAXIMUM THRUST COMBAT CEILING

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



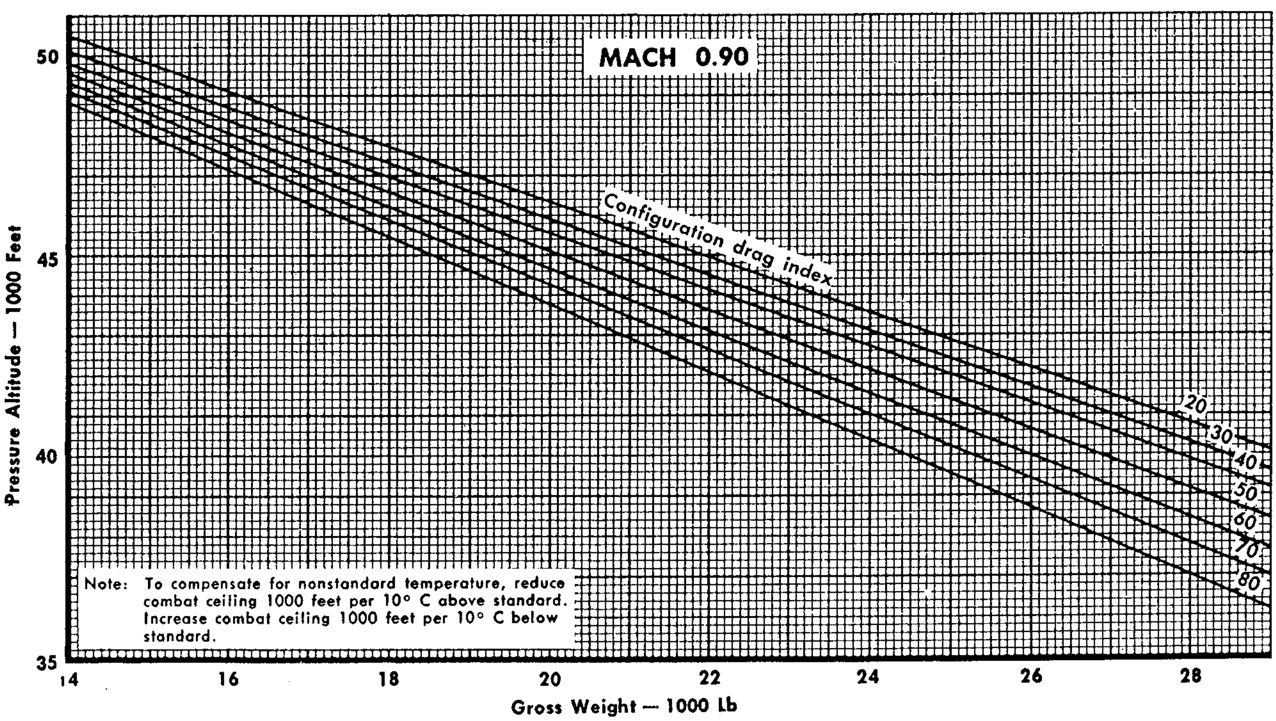
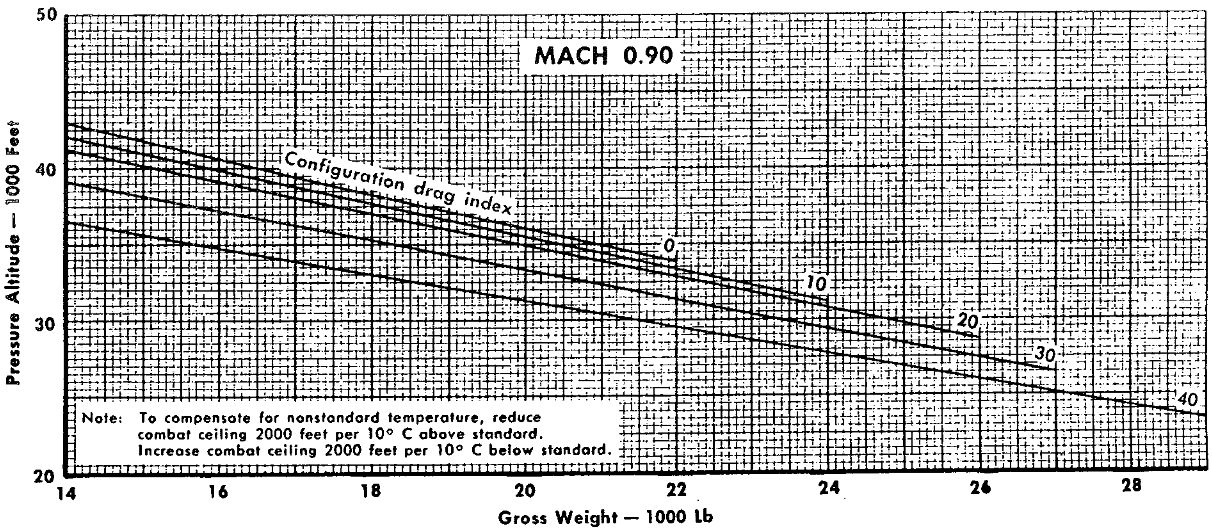


Figure A3-2

MILITARY THRUST COMBAT CEILING

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



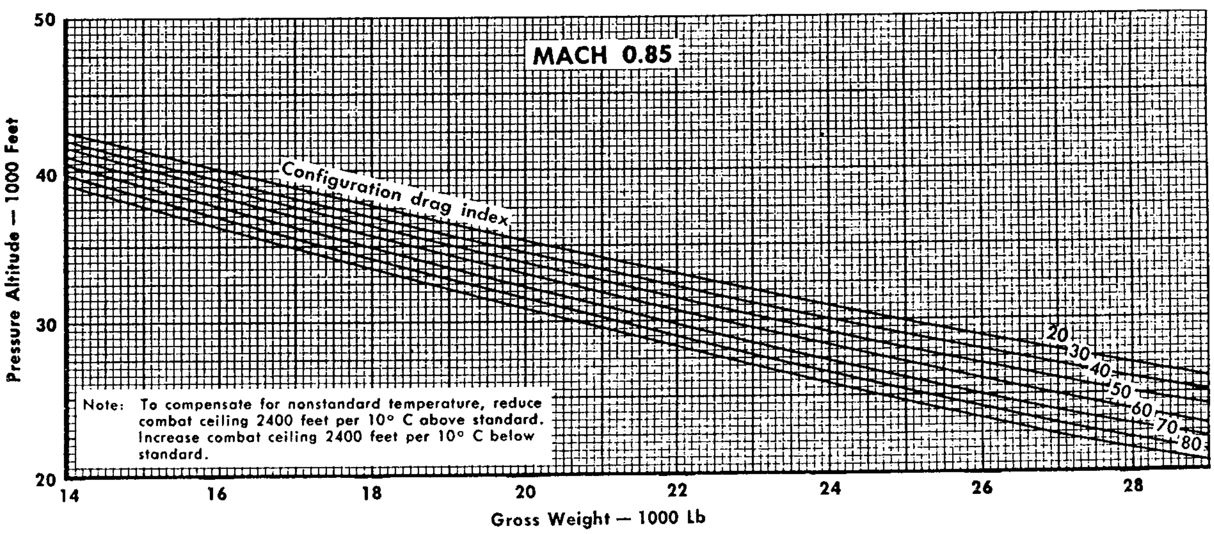


Figure A3-3

OPTIMUM CRUISE ALTITUDE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

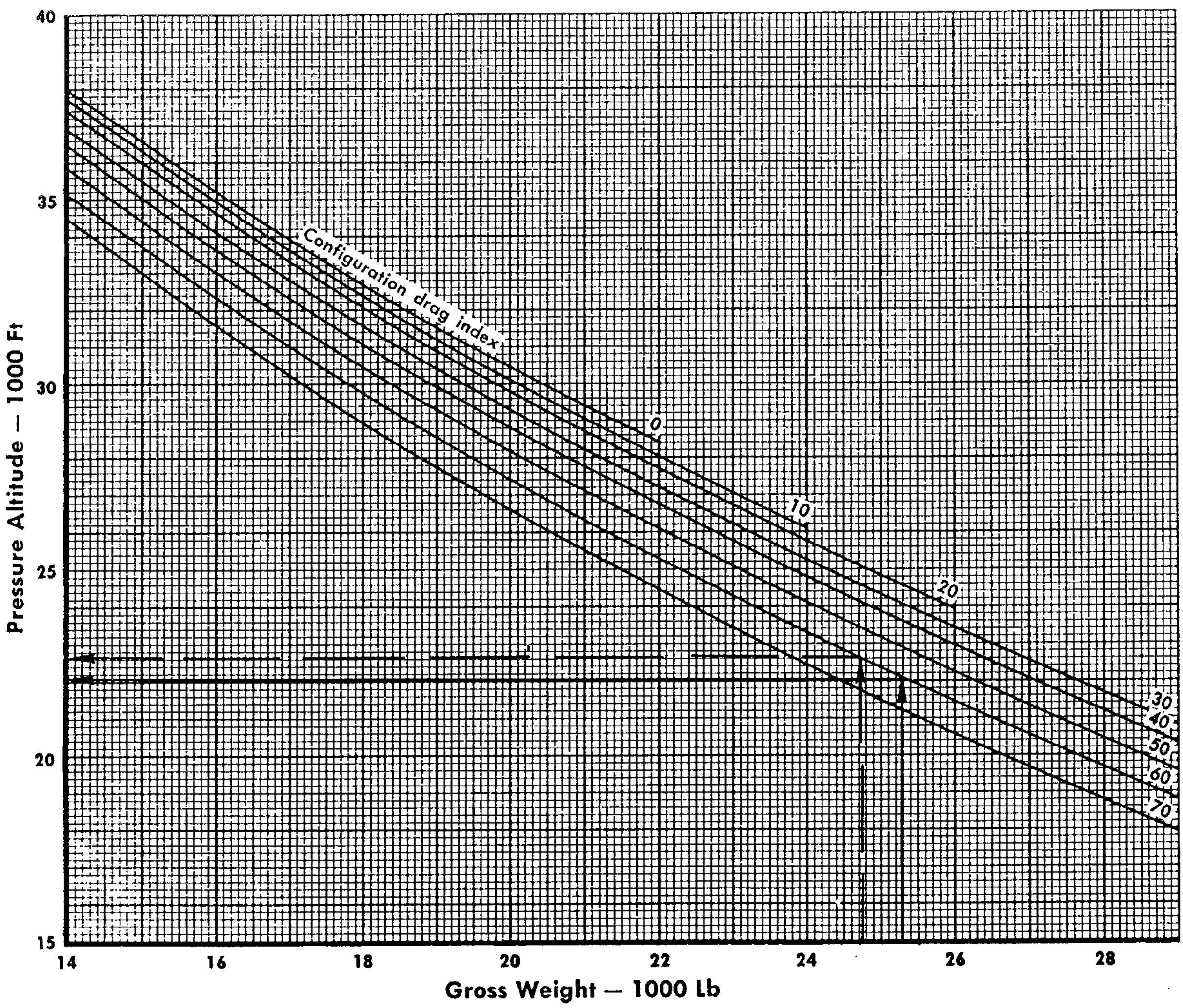


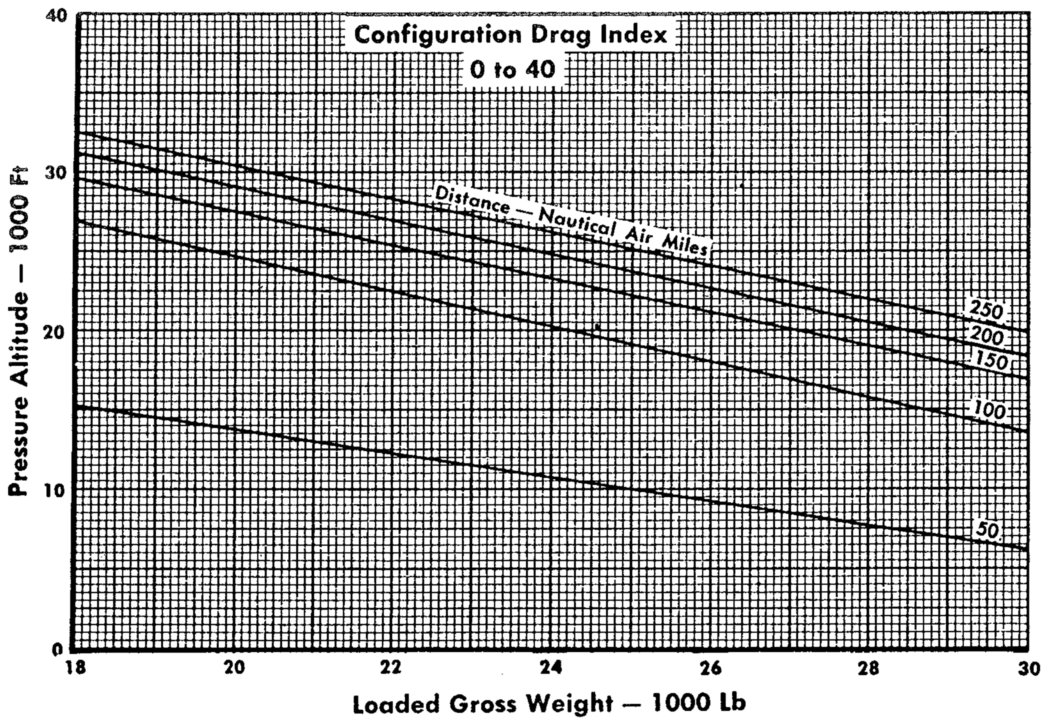
Figure A3-4

OPTIMUM ALTITUDE — MAXIMUM RANGE

CRUISE DISTANCE 250 NAUTICAL AIR MILES OR LESS

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



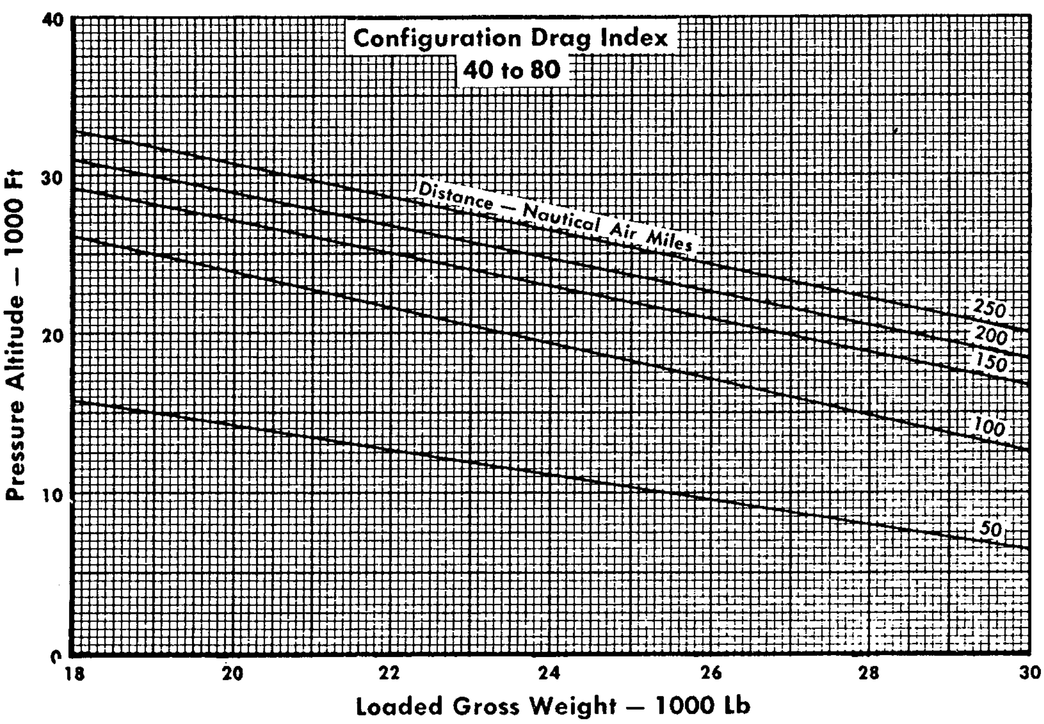
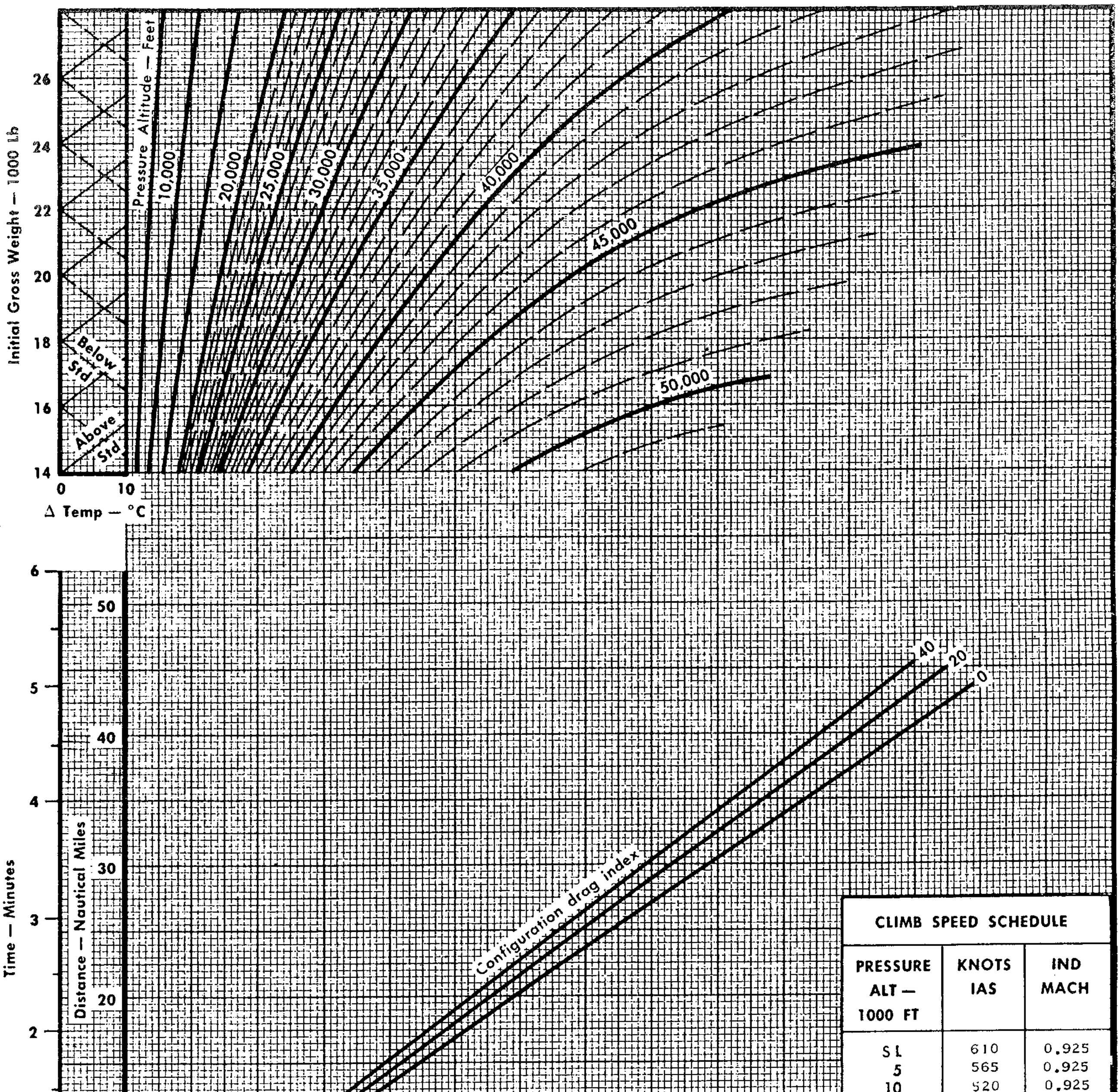


Figure A3-5

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.925

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



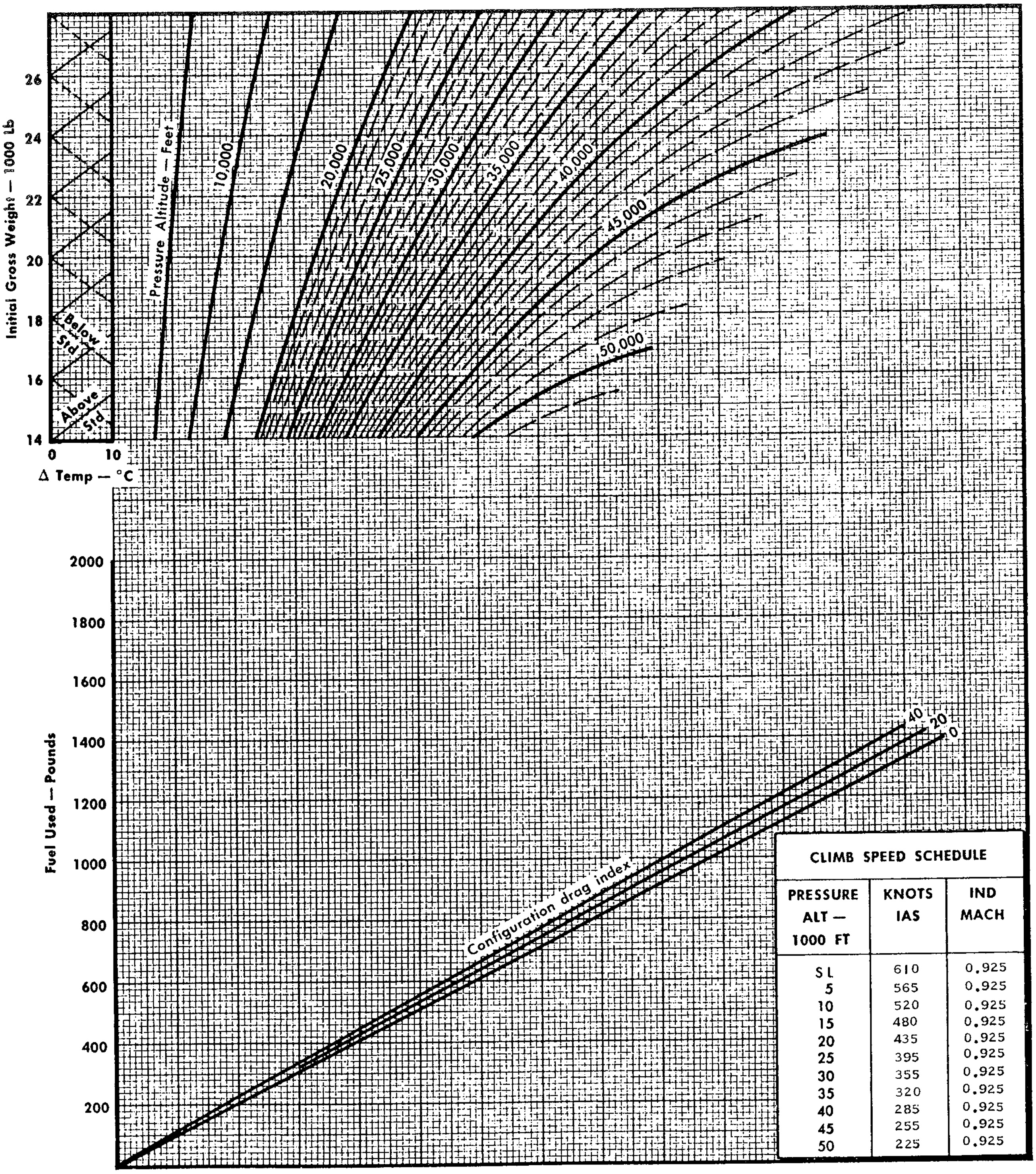
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	610	0.925
5	565	0.925
10	520	0.925
15	480	0.925
20	435	0.925
25	395	0.925
30	355	0.925
35	320	0.925
40	285	0.925
45	255	0.925
50	225	0.925

Figure A3-6 (Sheet 1 of 2)

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.925

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



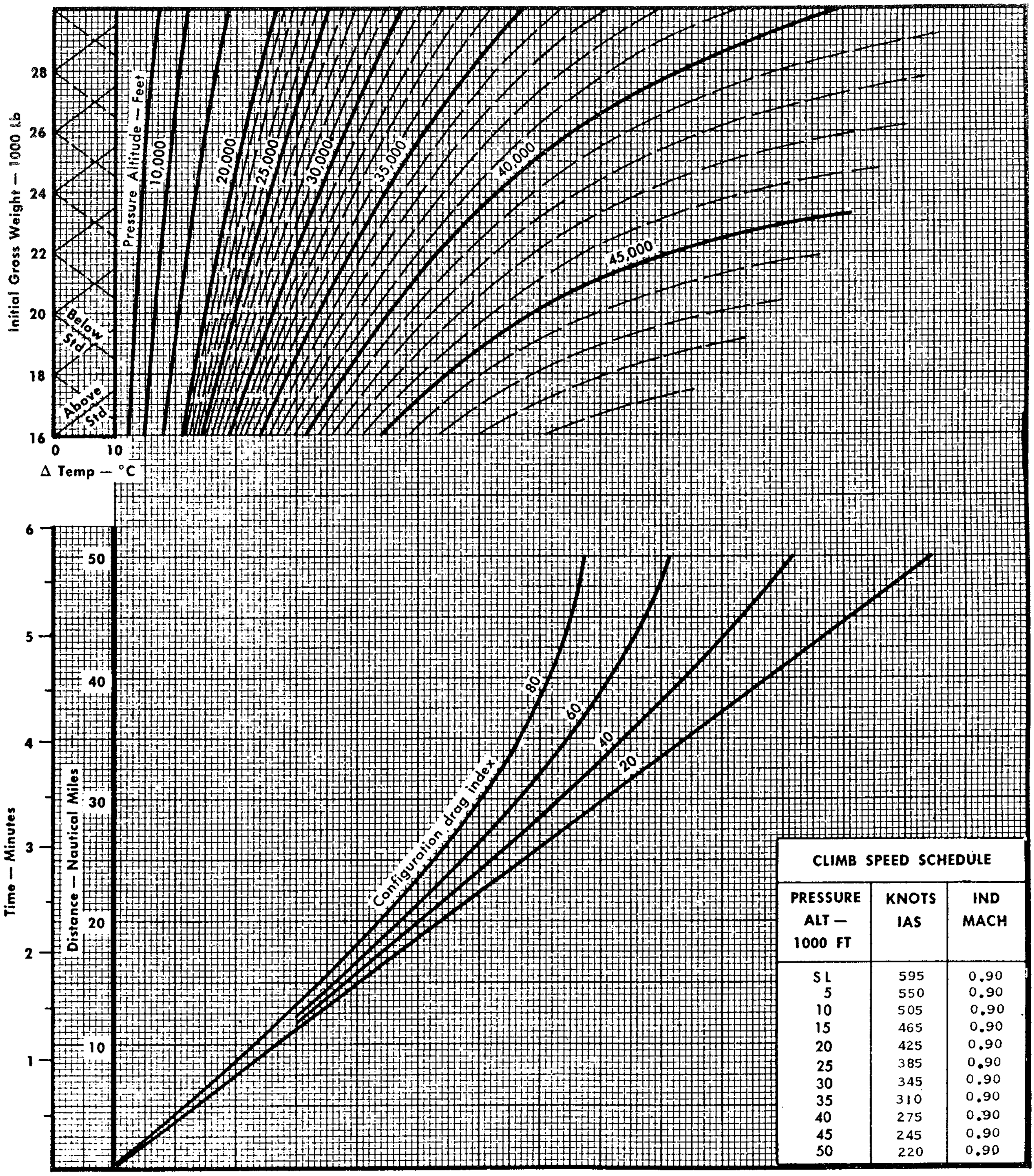
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	610	0.925
5	565	0.925
10	520	0.925
15	480	0.925
20	435	0.925
25	395	0.925
30	355	0.925
35	320	0.925
40	285	0.925
45	255	0.925
50	225	0.925

Figure A3-6 (Sheet 2 of 2)

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.90

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



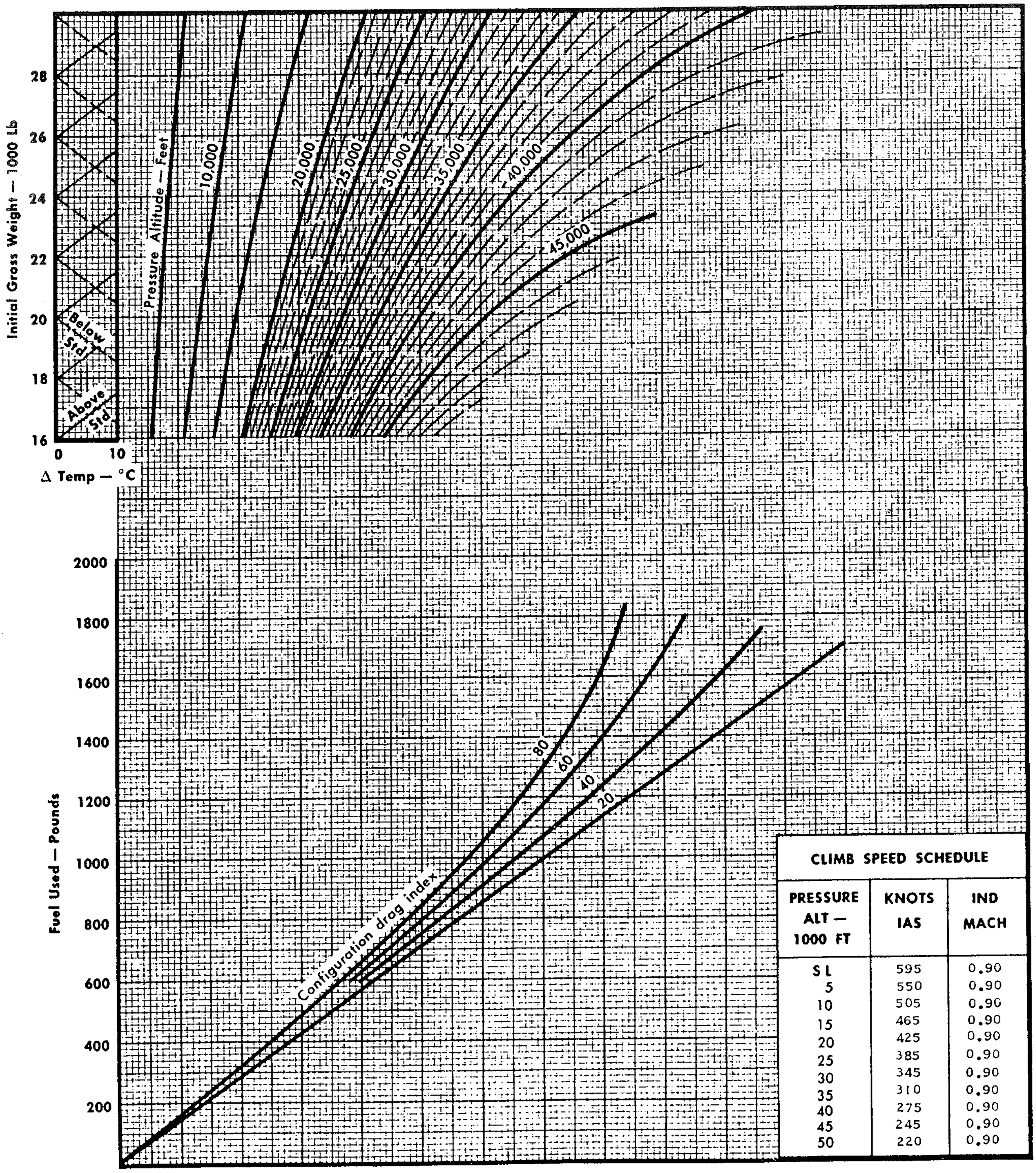
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	595	0.90
5	550	0.90
10	505	0.90
15	465	0.90
20	425	0.90
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90
45	245	0.90
50	220	0.90

Figure A3-7 (Sheet 1 of 2)

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.90

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



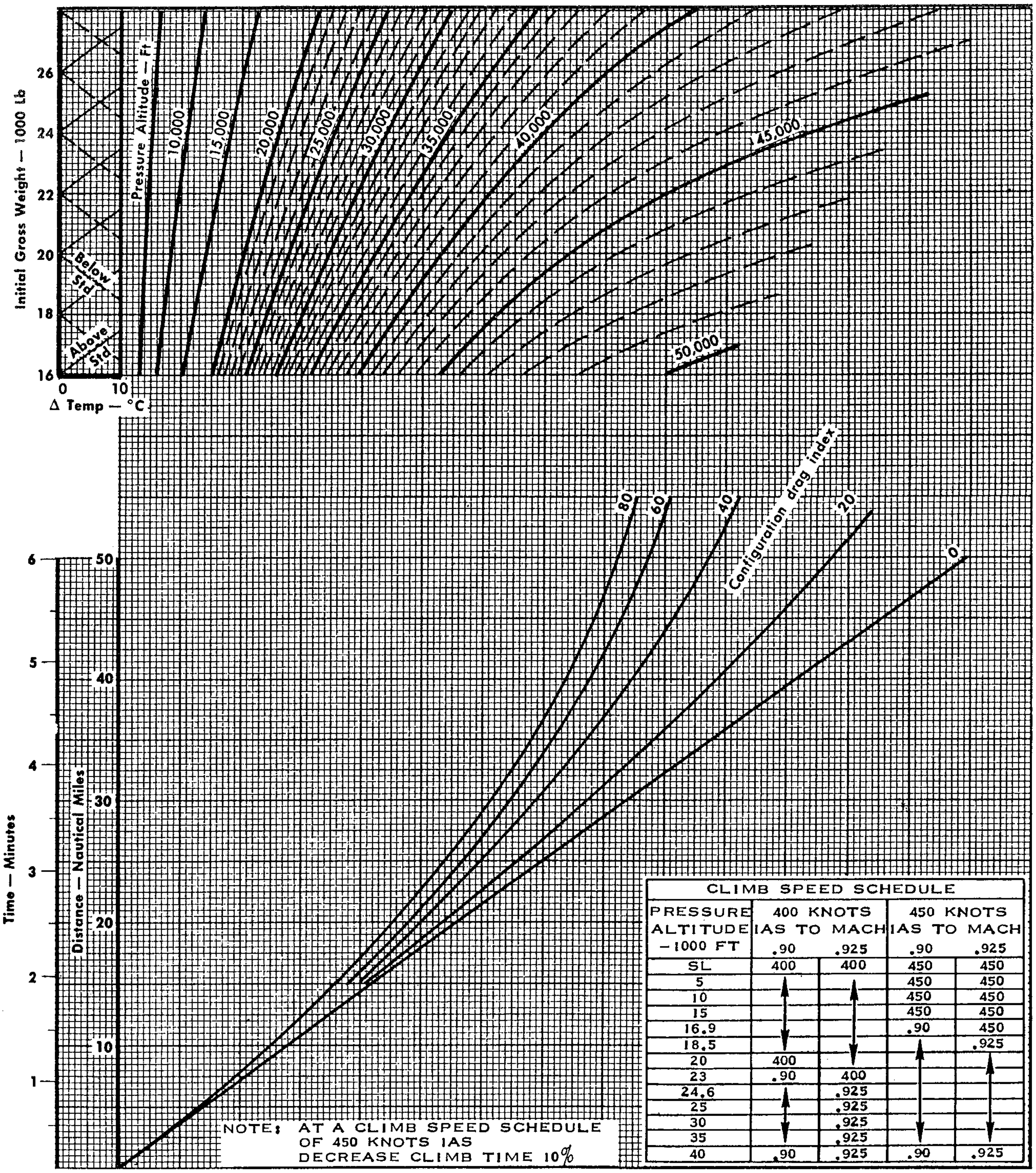
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
S L	595	0.90
5	550	0.90
10	505	0.90
15	465	0.90
20	425	0.90
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90
45	245	0.90
50	220	0.90

Figure A3-7 (Sheet 2 of 2)

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.90 OR MACH 0.925

TIME AND DISTANCE

Model: F-104G
Date: 15 October 1967
DATA BASIS: FLIGHT TEST



CLIMB SPEED SCHEDULE

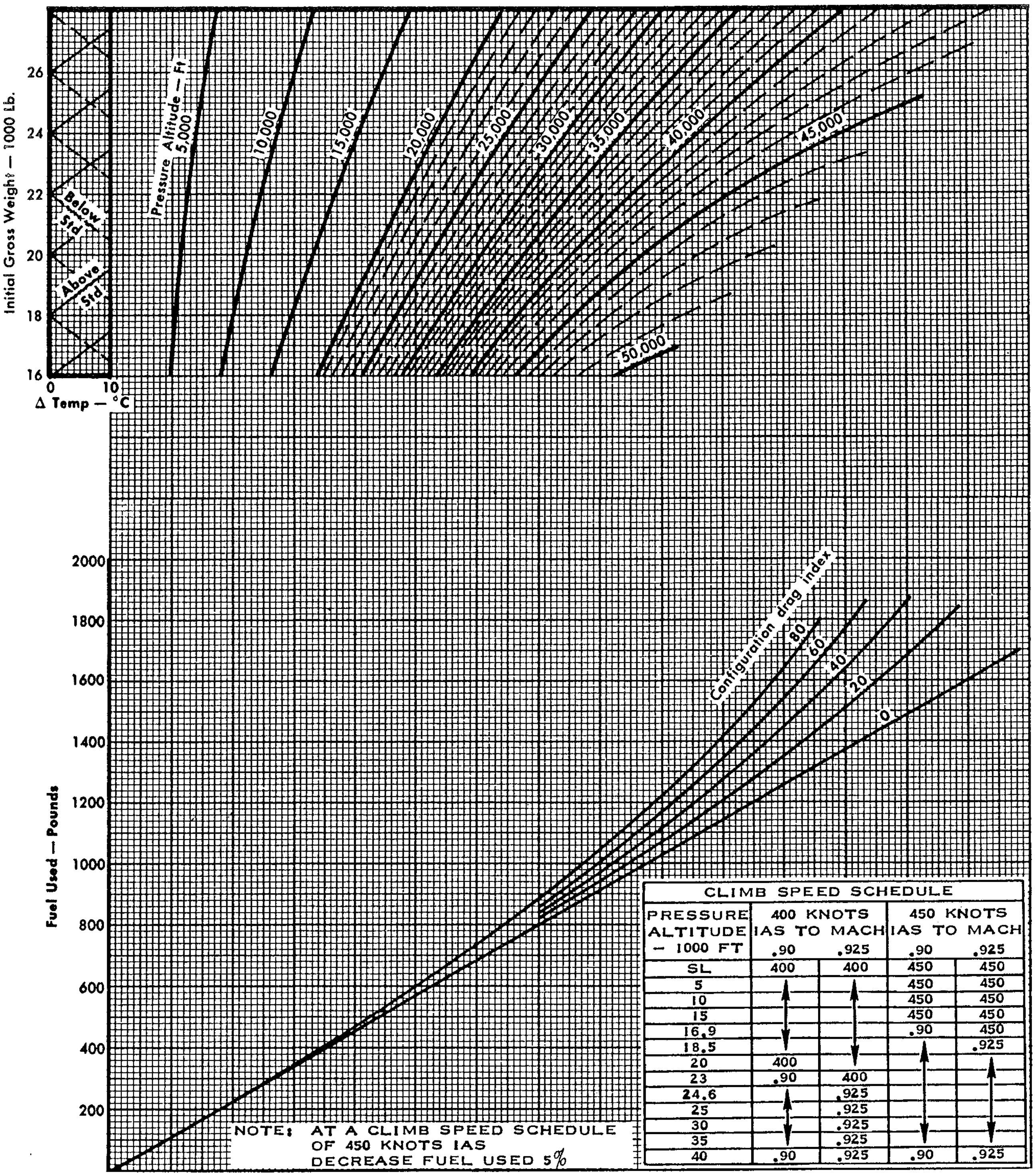
PRESSURE ALTITUDE — 1000 FT	400 KNOTS IAS TO MACH .90	400 KNOTS IAS TO MACH .925	450 KNOTS IAS TO MACH .90	450 KNOTS IAS TO MACH .925
SL	400	400	450	450
5			450	450
10			450	450
15			450	450
16.9			.90	450
18.5				.925
20	400			
23	.90	400		
24.6		.925		
25		.925		
30		.925		
35		.925		
40	.90	.925	.90	.925

Figure A3-8 (Sheet 1 of 2)

MAXIMUM THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.90 OR MACH 0.925

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



CLIMB SPEED SCHEDULE				
PRESSURE ALTITUDE – 1000 FT	400 KNOTS IAS TO MACH		450 KNOTS IAS TO MACH	
	.90	.925	.90	.925
SL	400	400	450	450
5			450	450
10			450	450
15			450	450
16.9			.90	450
18.5				.925
20	400			
23	.90	400		
24.6		.925		
25		.925		
30		.925		
35		.925		
40	.90	.925	.90	.925

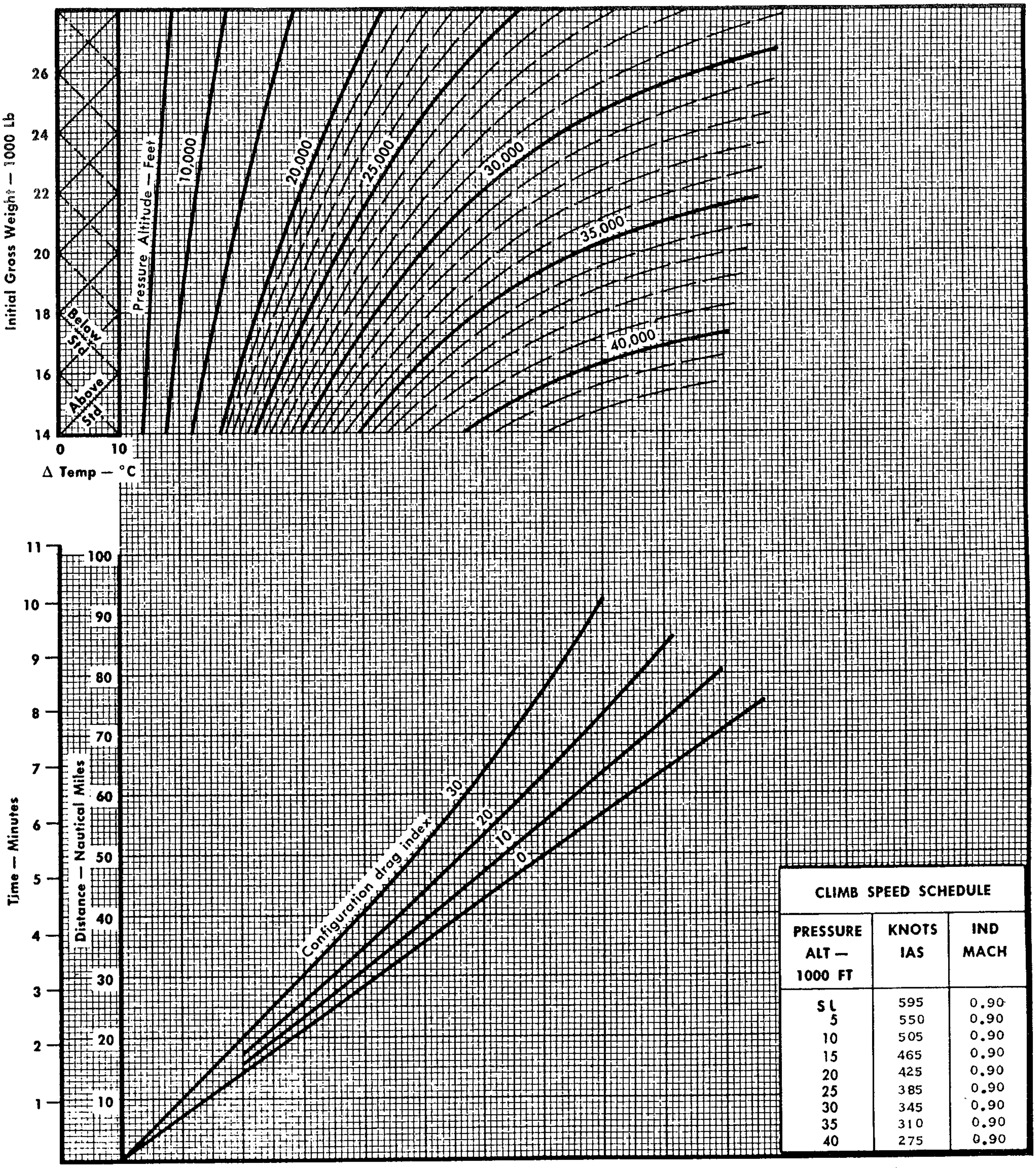
NOTE: AT A CLIMB SPEED SCHEDULE OF 450 KNOTS IAS DECREASE FUEL USED 5%

Figure A3-8 (Sheet 2 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.90

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



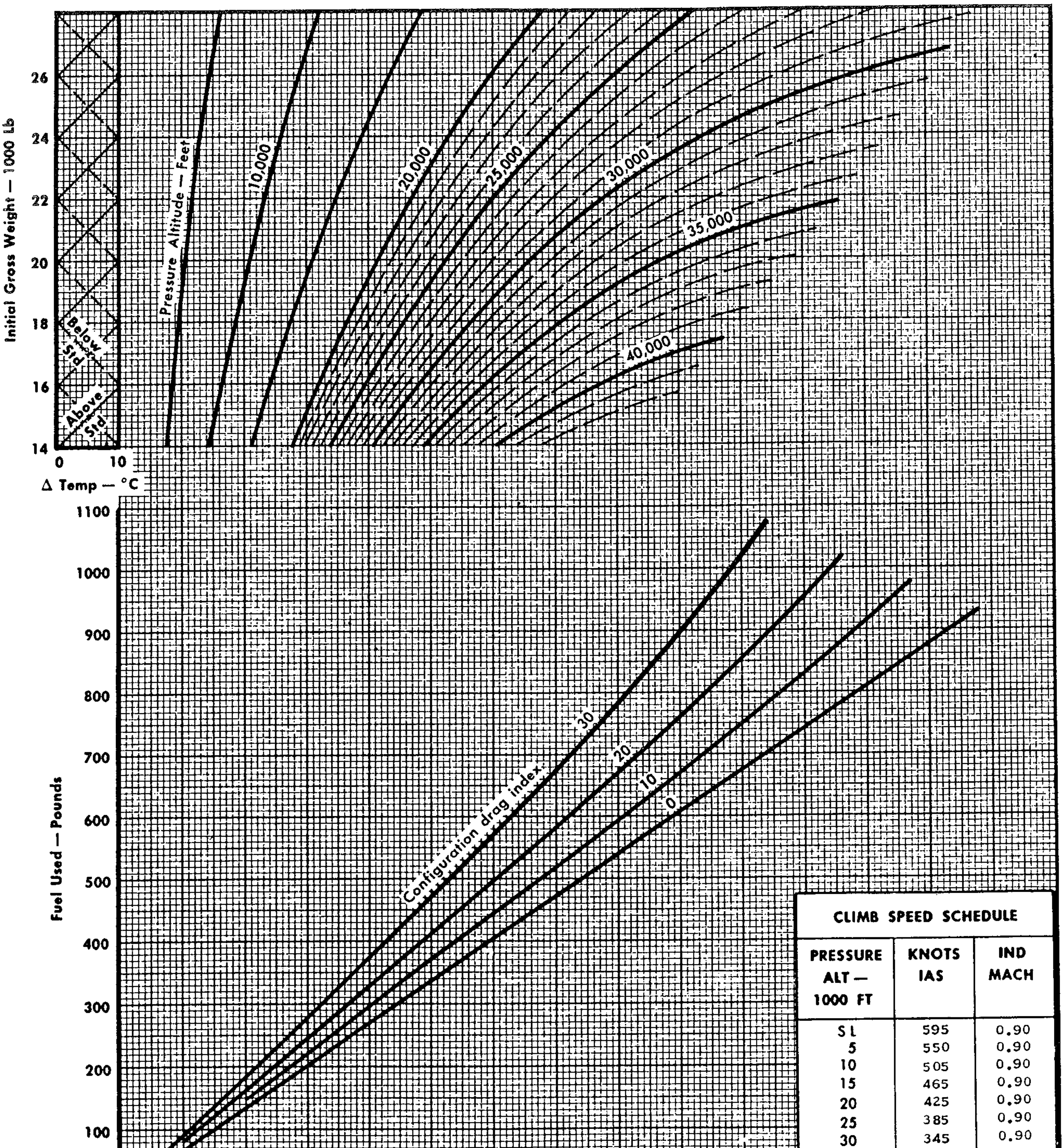
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
S L	595	0.90
5	550	0.90
10	505	0.90
15	465	0.90
20	425	0.90
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90

Figure A3-9 (Sheet 1 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.90

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



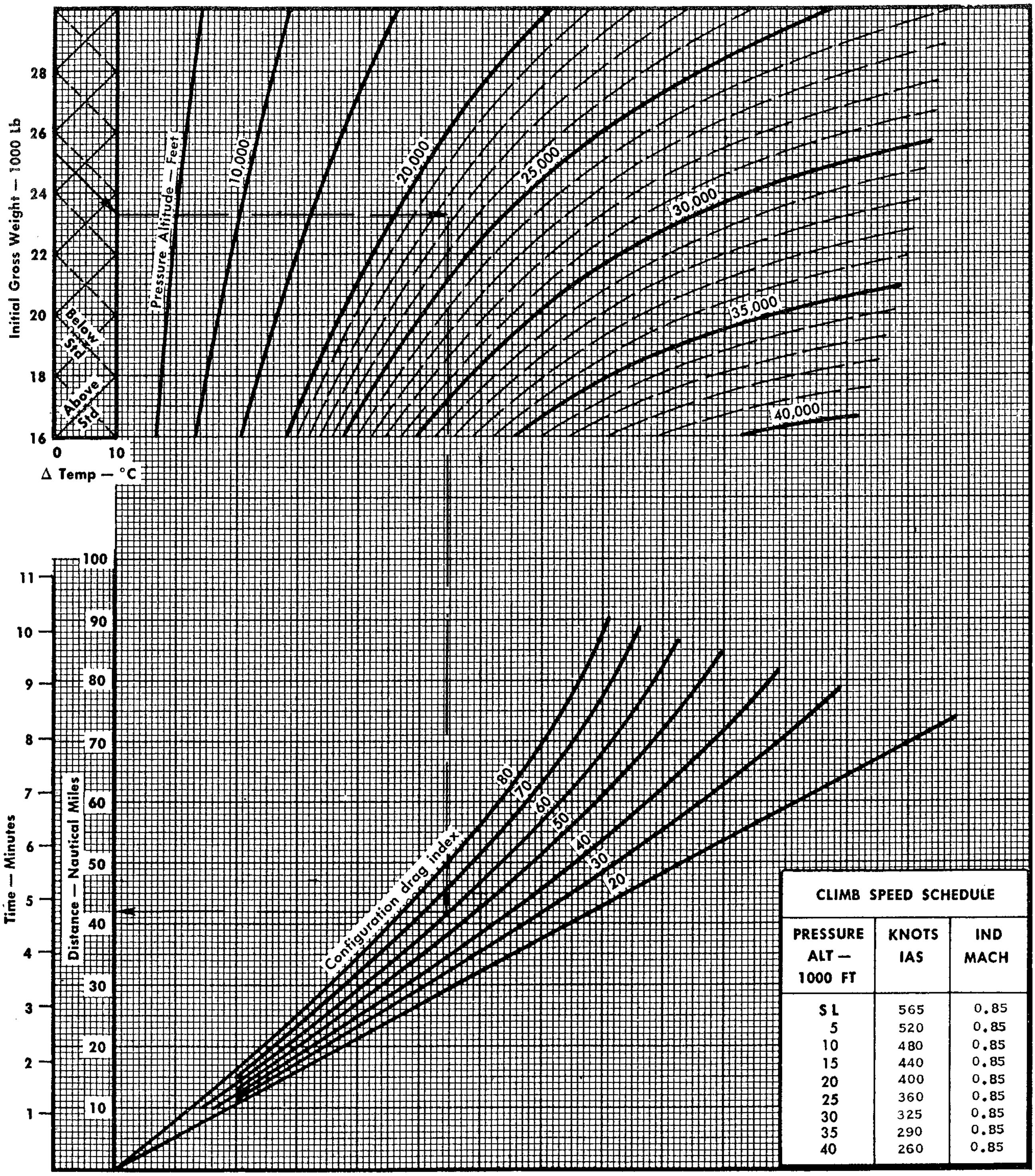
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
S L	595	0.90
5	550	0.90
10	505	0.90
15	465	0.90
20	425	0.90
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90

Figure A3-9 (Sheet 2 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.85

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



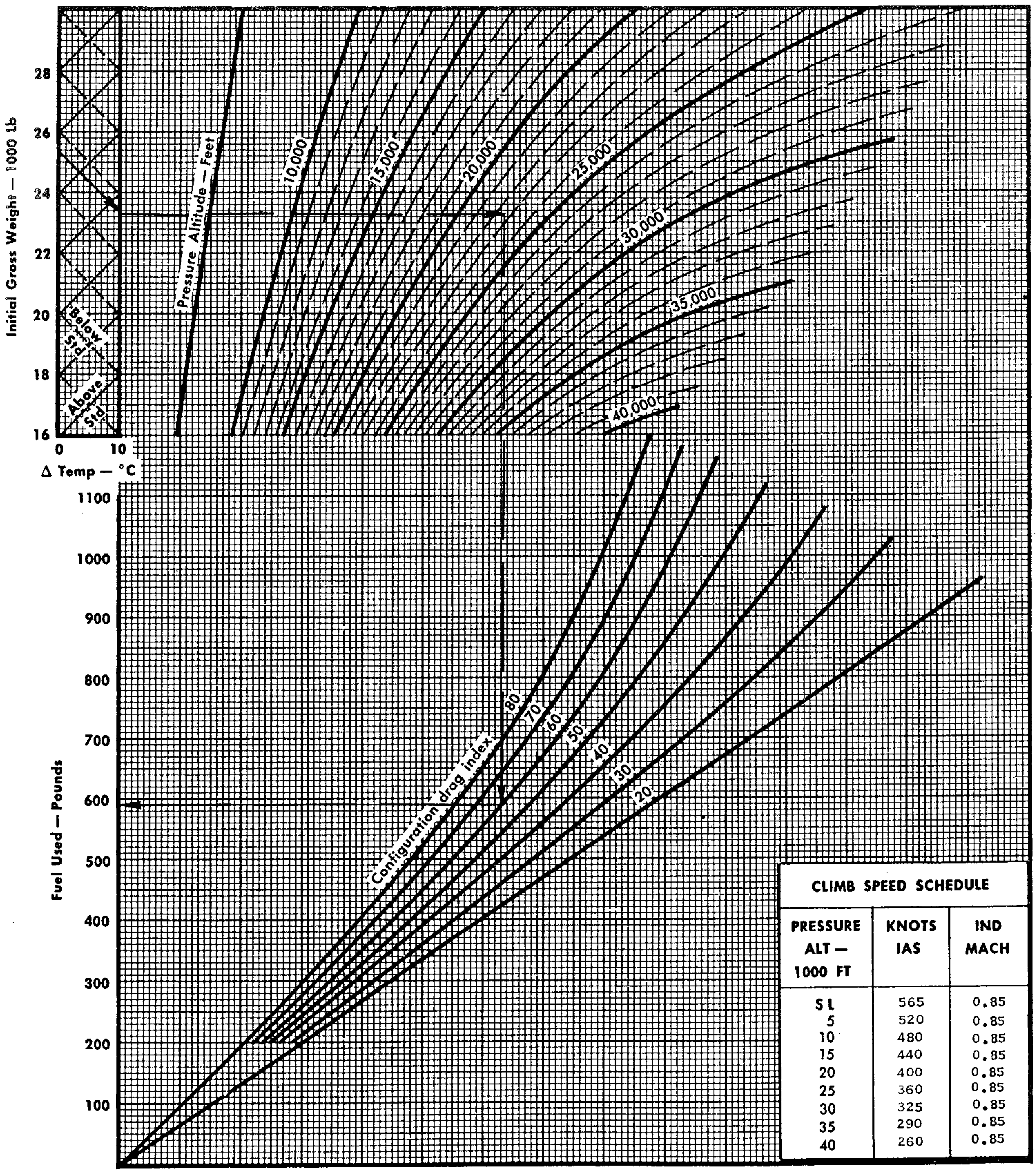
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
S L	565	0.85
5	520	0.85
10	480	0.85
15	440	0.85
20	400	0.85
25	360	0.85
30	325	0.85
35	290	0.85
40	260	0.85

Figure A3-10 (Sheet 1 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED MACH 0.85

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



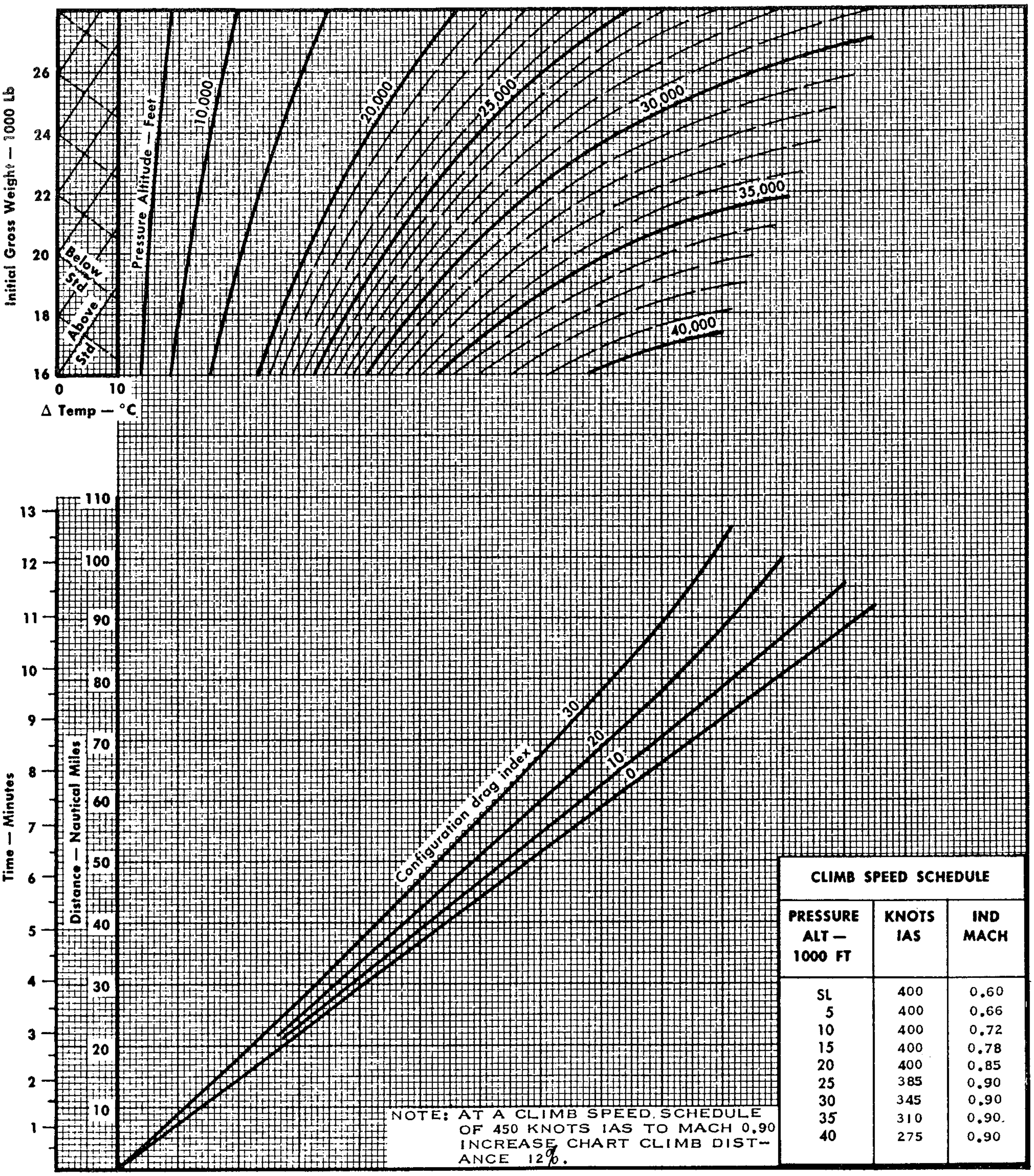
CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
S L	565	0.85
5	520	0.85
10	480	0.85
15	440	0.85
20	400	0.85
25	360	0.85
30	325	0.85
35	290	0.85
40	260	0.85

Figure A3-10 (Sheet 2 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.90

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	**KNOTS IAS**	**IND MACH**
SL	400	0.60
5	400	0.66
10	400	0.72
15	400	0.78
20	400	0.85
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90

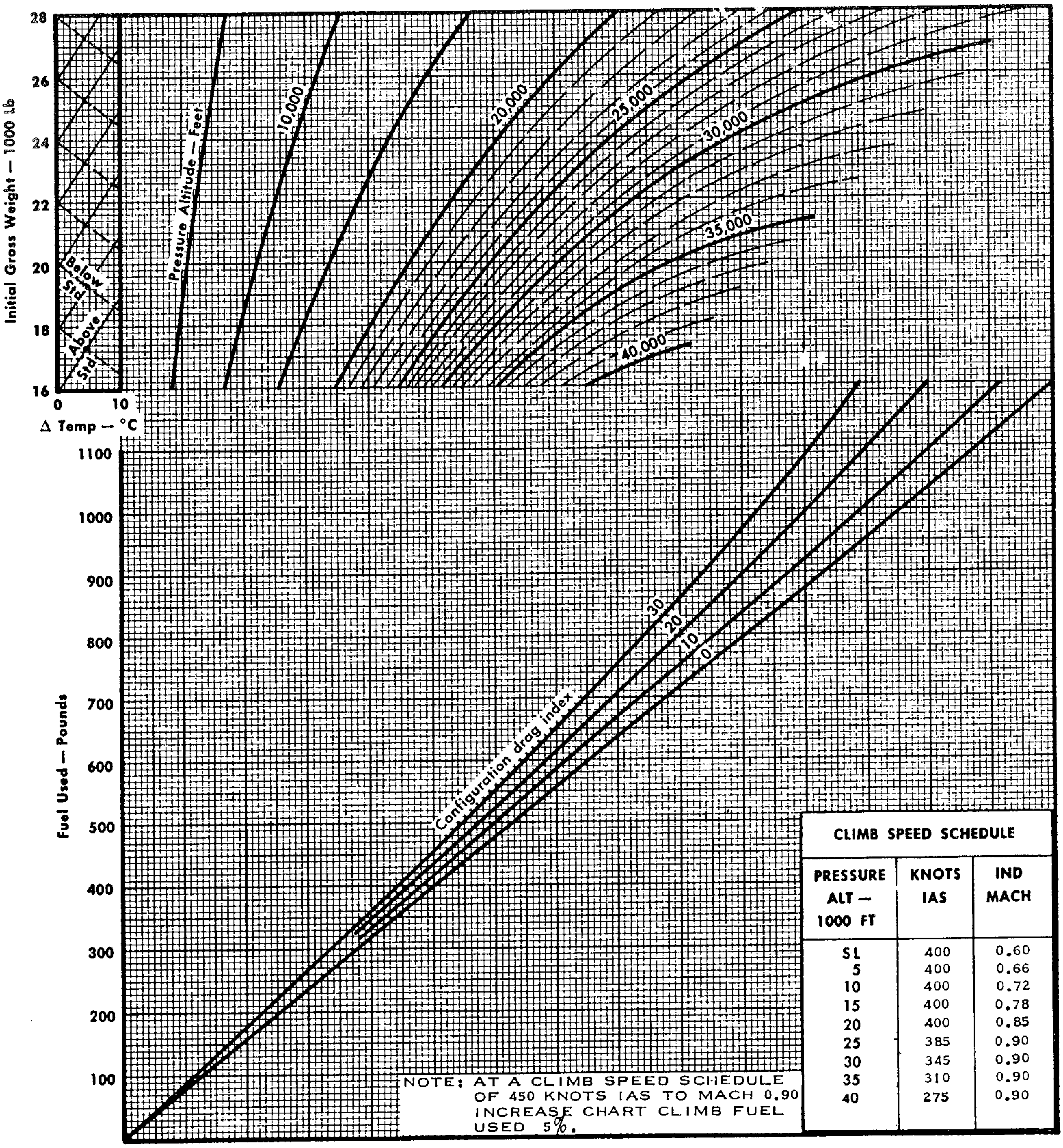
NOTE: AT A CLIMB SPEED SCHEDULE OF 450 KNOTS IAS TO MACH 0.90 INCREASE CHART CLIMB DISTANCE 12%.

Figure A3-11 (Sheet 1 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.90

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE: AT A CLIMB SPEED SCHEDULE OF 450 KNOTS IAS TO MACH 0.90 INCREASE CHART CLIMB FUEL USED 5%.

CLIMB SPEED SCHEDULE

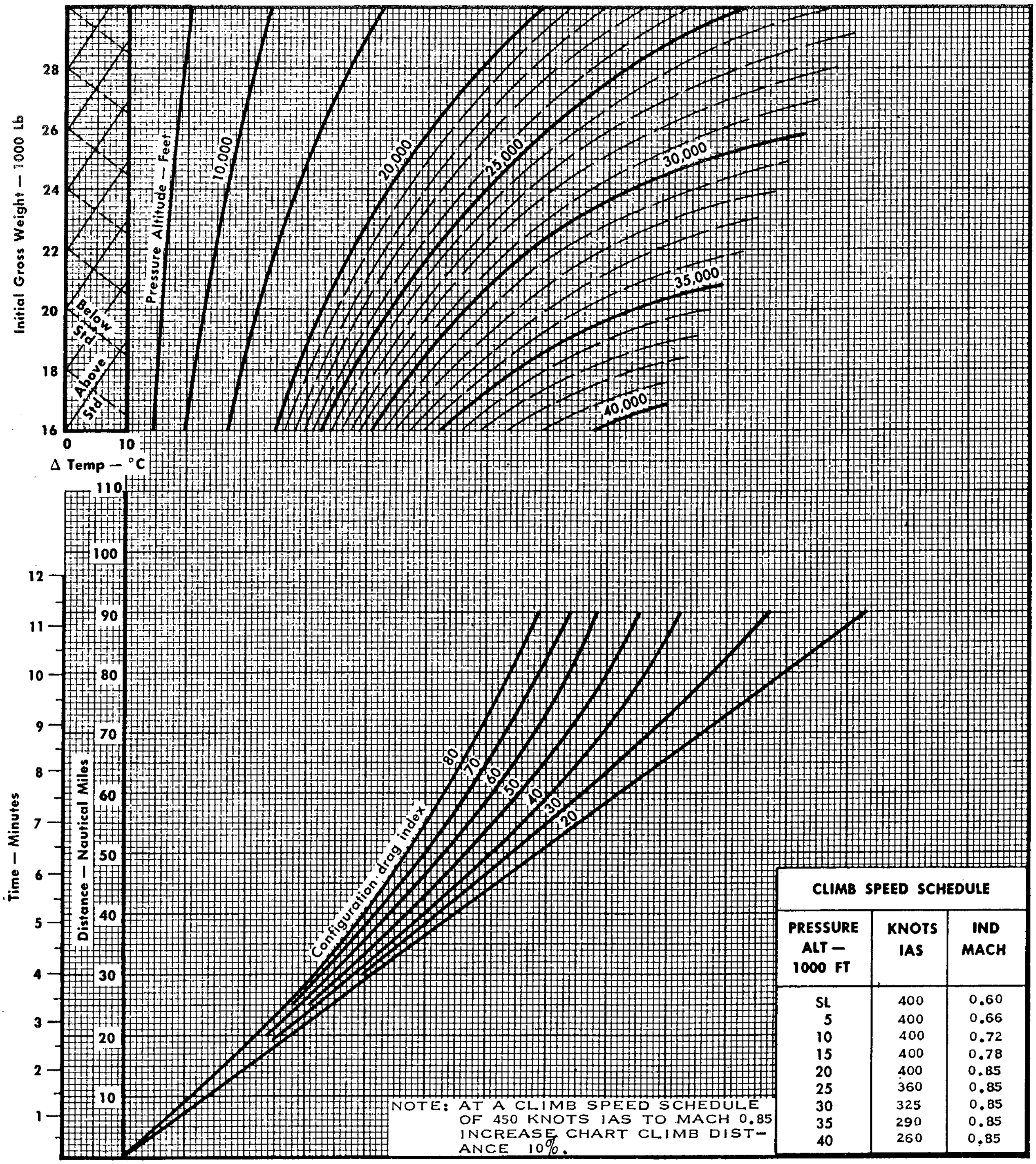
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	400	0.60
5	400	0.66
10	400	0.72
15	400	0.78
20	400	0.85
25	385	0.90
30	345	0.90
35	310	0.90
40	275	0.90

Figure A3-11 (Sheet 2 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.85

TIME AND DISTANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	400	0.60
5	400	0.66
10	400	0.72
15	400	0.78
20	400	0.85
25	360	0.85
30	325	0.85
35	290	0.85
40	260	0.85

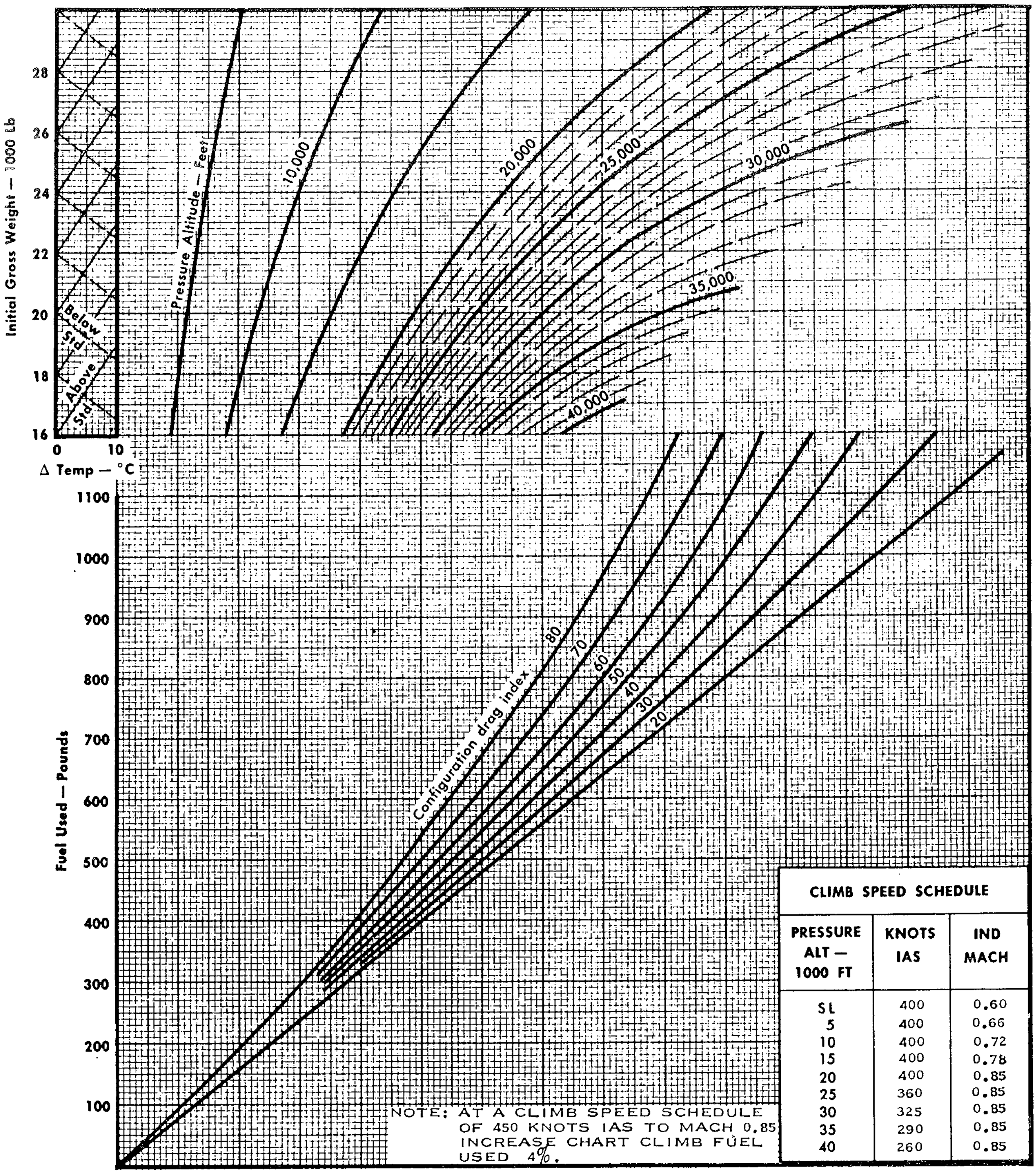
NOTE: AT A CLIMB SPEED SCHEDULE OF 450 KNOTS IAS TO MACH 0.85 INCREASE CHART CLIMB DISTANCE 10%.

Figure A3-12 (Sheet 1 of 2)

MILITARY THRUST CLIMB CONTROL
CLIMB SPEED 400 OR 450 KIAS
TO MACH 0.85

FUEL USED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



CLIMB SPEED SCHEDULE		
PRESSURE ALT — 1000 FT	KNOTS IAS	IND MACH
SL	400	0.60
5	400	0.66
10	400	0.72
15	400	0.78
20	400	0.85
25	360	0.85
30	325	0.85
35	290	0.85
40	260	0.85

NOTE: AT A CLIMB SPEED SCHEDULE OF 450 KNOTS IAS TO MACH 0.85 INCREASE CHART CLIMB FUEL USED 4%.

Figure A3-12 (Sheet 2 of 2)

PART 4

RANGE

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

RANGE DATA PRESENTED

Range data presented in this part show the range available at constant altitude and cruise climb altitude based on operation at the recommended speed schedules.
The data are based on flight tests except as noted on the stores table (Figure A1-2) and include a 5% operational tolerance. Diversion range summary tables are also presented for return performance with low fuel remaining quantities.
If detailed flight planning information is desired for speed schedules other than those defined in this part, refer to Part 6 "Miles Per Pounds Data" for applicable miles-per-pound charts.

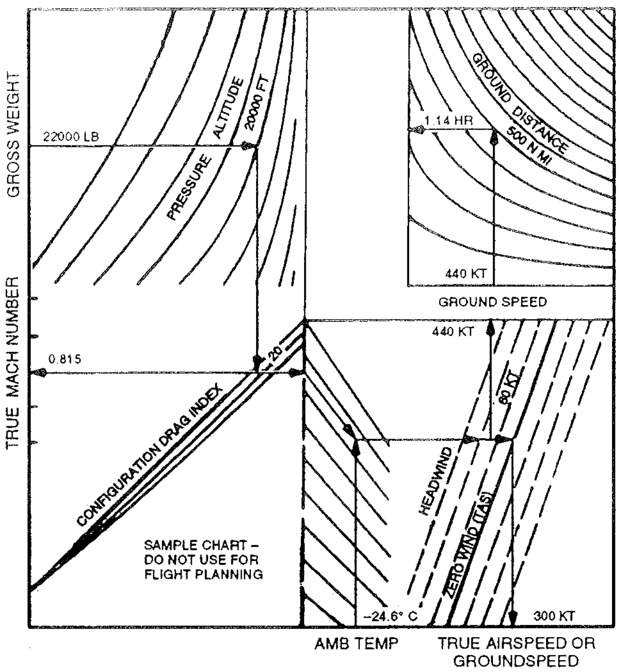
MAXIMUM RANGE PERFORMANCE

The range charts present the level flight cruise and optimum cruise (cruise-climb) range capabilities of the aircraft for all gross weights and configurations. The maximum range cruise Mach number schedule is also included, and shall be used to achieve the charted performance. Altitudes shown assume that the altimeter is set at 29.92 inches Hg (1013.2 mb). To achieve the desired pressure altitude, indicated altitude shall be corrected for altimeter position error and instrument calibration, if known. Optimum altitude for maximum range cruise may be obtained from the chart provided in Part 3 "Climb". As indicated, the altitude is based on actual gross weight and shall be corrected for the fuel consumed during ground maneuver, takeoff and acceleration to climb speed and climb. Use of the maximum range performance charts is illustrated by the following sample problems.

MAXIMUN RANGE – CONSTANT ALTITUDE SAMPLE PROBLEM

Determine the fuel required to cruise a ground distance of 500 nautical miles at an altitude of 20000 feet on a standard day with a 60 knot headwind component. The aircraft has a configuration drag index of 20.
The initial cruise gross weight (gross weight after allowances have been made for ground maneuver, takeoff and acceleration to climb speed and climb to 20000 feet) is 22000 pounds.

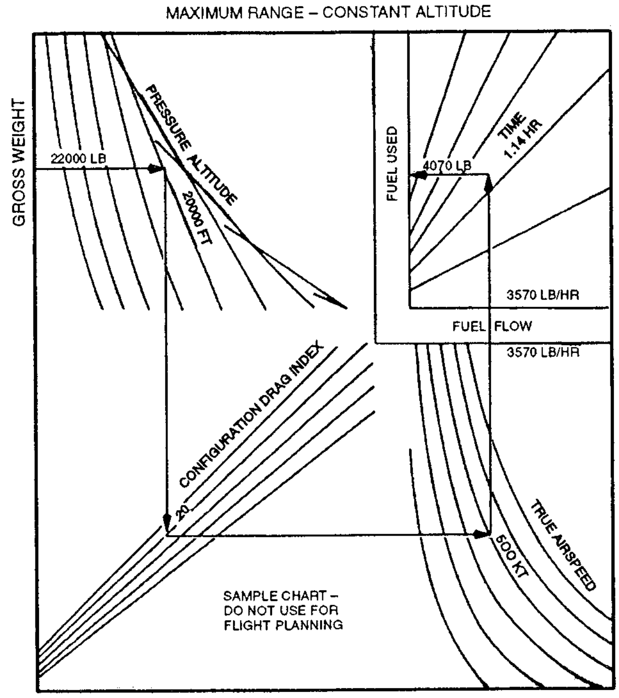
MAXIMUM RANGE – CONSTANT ALTITUDE



FA0321

a. Enter Figure A4-1 (sheet 1) at 22000 lb and 20000 feet. At a configuration drag index of 20, read maximum range cruise true Mach 0.815. Proceed horizontally to the right to the true airspeed base line, follow the guide lines to −24.6° C, standard day temperature at 20000 feet. Continue to the right and read cruise true airspeed of 500 knots. Ground speed with a 60 knots headwind is 440 knots. Vertically upwards at a groundspeed of 440 knots read a flight time of 1.14 hours for a cruise ground distance of 500 nautical miles.

MAXIMUM RANGE – CONSTANT ALTITUDE



FA0322

b. Enter Figure A4-1 (sheet 2) at 22000 lb and 20000 feet. Proceed vertically below this point to a configuration drag index of 20. Horizontally to the right intersect the true airspeed determined in step (a), 500 knots. Vertically above the true airspeed read an initial cruise fuel flow of 3570 lb per hour. Continue vertically to the cruise time from step (a) and read cruise fuel required, 4070 pounds (1.14 x 3570).

The preceding steps assume cruise performance based on the initial conditions for cruise. Obviously, as fuel is consumed, the aircraft weight decreases and the maximum range capability increases. Therefore, when large quantities of fuel are required, it is recommended that performance calculations be made in gross weight increments (fuel consumed) of 2000 pounds or less. The preceding problem should be used as an approximation, then recalculated as follows:

c. Determine the cruise ground run distance for 2000 pounds of fuel consumption. Average gross weight is 21000 lb (22000 + 20000)/2. Follow the same procedures as in step (a), Figure A4-1 (sheet 1) and read:

True Mach number	0.805
True airspeed at −24.6° C	495 kt

d. Enter Figure A4-1 (sheet 2) at 21000 lb, follow the procedures of step (b) and read cruise fuel flow 3430 lb per hour at 495 knots.

e. Cruise time to consume 2000 pounds of fuel is 0.583 hr (2000/3430).

f. Reenter Figure A4-1 (sheet 1) at a ground speed of 435 knots (495 − 60) and read cruise ground distance of 254 nautical miles or (435 x 0.583).

g. From step (b), the approximate additional fuel required to cruise the remaining distance of 246 nautical miles (500 − 254) is 2070 pounds (4070 − 2000).

h. Average gross weight for the segment is 18965 lb (20000 − 2070/2).

i. From Figure A4-1 (sheet 1) read:

True Mach number	0.787
True airspeed at −24.6° C	484 kt
Ground speed (60 kt. headwind)	424 kt
Time to cruise 246 nmi (246/424)	0.58 hr

j. From Figure A4-1 (sheet 2) read cruise fuel flow, 3190 lb per hour at 484 knots true airspeed.

k. Fuel required to cruise remaining distance is 1850 lb (3190 × 0.58).

l. Total time and fuel required to cruise 500 nautical miles is:

Time (0.583 hr + 0.58 hr) .... 1 hr 10 min.
Fuel (2000 lb + 1850 lb) .... 3850 lb

MAXIMUM RANGE – CONSTANT ALTITUDE SAMPLE PROBLEM (EXTERNAL TANKS DROPPED)

Determine the range available at a cruise altitude of 20000 feet on a standard day with a 60 knots headwing component. The aircraft has a configuration drag index of 68 (with tip tanks and pylon tanks installed). The initial cruise gross weight is 24400 lb, 9340 pounds total fuel remaining (1300 pounds remaining in pylon tanks).

Cruise Segment (Pylon Tanks Empty)

a. Find the cruise gross weight at pylon tank fuel burnout.

NOTE

When calculating the point at which external fuel tanks become empty, it is recommended that a nominal allowance of 480 pounds of fuel be added to the last external fuel tank quantity to account for internal fuel depletion during pilot recognition of the external tanks empty, decision and action to jettison. For this problem, the allowance will be made at tip tanks empty.

Cruise gross weight at pylon tanks empty is 23100 pounds (24400 – 1300). Average gross weight 23750 lb.

b. From Figure A4-1 (sheet 1) read:

True Mach number 0.816
True airspeed .... 501 kt
Ground speed .... 441 kt

c. From Figure A4-1 (sheet 2) read:

Fuel flow .... 4000 lb/hr
Cruise time (1300/4000).... 0.325 hr (19.5 min.)

d. Ground distance is 143.5 nautical miles (441 × 0.325).

Cruise Segment (Tip Tanks Empty)

Determine the configuration drag index after pylon tanks are dropped (pylons assumed retained). From Figure A1-2, the store drag number for pylon tanks is 46. Configuration drag index (pylon tanks dropped) is 22 (68 – 46).

e. Initial cruise gross weight after pylon tanks are dropped is 22746 pounds (23100 – 354).

f. Cruise gross weight at tip tank fuel burnout is 19995 pounds (22746 – 2751), including the allowance for empty tanks recognition and action to jettison. Average gross weight is 21370.5 pounds.

g. From Figure A4-1 (sheet 1) read:

True Mach number 0.816
True airspeed .... 501 kt
Groundspeed .... 441 kt

h. From Figure A4-1 (sheet 2) read:

Fuel flow .... 3440 lb/hr
Cruise time ..... (2440/3440) 0.71 hr (42.6 min)

i. Ground distance is 313 nautical miles (441 × 0.71)

Cruise Segment (Internal Fuel)

The configuration drag index after tip tanks are dropped is 6 (22 – 16).

j. Initial cruise gross weight after tip tanks are dropped is 19535 pounds (19995 – 460). Internal fuel remaining 4196 pounds (4676 – 480).

k. Average cruise gross weight during comsumption of 2000 pounds of internal fuel is 18535 (19535 – 2000/2).

l. From Figure A4-1 (sheet 1) read:

True Mach number 0.79
True airspeed .... 485 kt
Groundspeed .... 425 kt

m. From Figure A4-1 (sheet 2) read:

Fuel flow .... 3065 lb/hr
Cruise time ... (2000/3065) 0.653 hr (39.2 min)

n. Ground distance is 277.5 nautical miles (425 × 0.653)

o. Internal fuel remaining is 2196 pounds (4196 − 2000). Average gross weight for a 2000 pound fuel consumption segment is 16535 (17535 − 2000/2).

p. From Figure A4-1 (sheet 1) read:

True Mach number 0.764
True airspeed .... 469 kt
Groundspeed .... 409 kt

q. From Figure A4-1 (sheet 2) read:

Fuel flow .... 2810 lb/hr
Cruise time ... (2000/2810) 0.712 hr (42.7 min)

r. Ground distance is 291.5 nautical miles (409 × 0.712)

s. Internal fuel remaining is 1196 pounds (2196 − 1000). Assume that a 1000 pound fuel reserve is desired, cruise fuel remaining is 196 pounds. Average gross weight for final segment is 15437 pounds (15535 − 196/2).

t. From Figure A4-1 (sheet 1) read:

True Mach number 0.742
True airspeed .... 455 kt
Groundspeed .... 395 kt

u. From Figure A4-1 (sheet 2) read:

Fuel flow .... 2650 lb/hr
Cruise time .... (196/2650) 0.074 hr (4.4 min)

v. Ground distance is 29.2 nautical miles (395 × 0.074)

w. Total range available and cruise time are:

Ground distance .... 1054.7 n.mi.
Time .... 2 hr 43.0 min.

CRUISE CLIMB

The preceding problem (external stores dropped) was conducted at constant altitude. If the same conditions had been planned for a cruise-climb flight, Figure A4-2 would be used. The altitude-gross weight relationship (optimum W/δ) increases with each change in configuration (external stores dropped). Reference to the appropriate configuration drag index on Figure A4-2 (sheet 1) will provide the proper cruise-climb pressure altitude for each gross weight. Climb performance data for the altitude changes required can be determined from the climb control charts in Part 3.

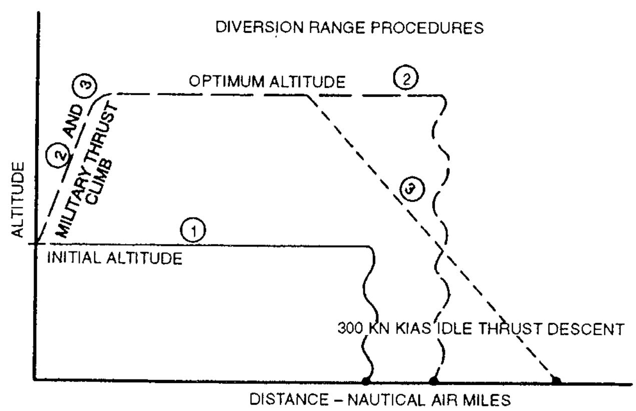
For example, the configuration drag index with tip tanks and pylon tanks is 68, gross weight with pylon tanks empty is 23100 pounds. Cruise-climb altitude at the weight is 25400 feet. The configuration drag index after pylon tanks are dropped is 46. Approximate the cruise-climb altitude with tip tanks as 26300 feet at 23100 pounds. After pylon tanks are dropped a Military thrust climb at M = 0.85 (ISA) would be made from 25400 feet to optimum cruise altitude with tip tanks. Climb performance from Figure A3-10 results in the following:

SL to 25400 ft.	SL to 26300 ft
740 lb.	770 lb.
55 n.mi.	58 n.mi.
6.3 min.	6.6 min.

Time, fuel and distance to climb to optimum cruise altitude with tip tanks would be 0.3 minutes, 30 pounds and 3 nautical miles, respectively. Optimum cruise altitude is 26300 feet and the initial cruise gross weight is 23080 pounds.

Diversion Range Summary Tables

The Diversion Range Summary Tables presented are in a form readily usable when rapid determination of range available with various fuel remaining values is desired. The range available using three flight procedures is presented for fuel remaining values of 750, 1000, 1500, 2000 and 3000 pounds. The procedures, as indicated on each of the tables, are as follows:



FA0323

1. Cruise at initial altitude until over base and descend (speed brakes in).

2. Climb to optimum altitude (indicated by short arrow on left side of initial altitude column), cruise at optimum altitude, until over base, and descend (speed brakes extended).

3. Climb to optimum altitude (indicated by long arrow on right side of initial altitude column), cruise at optimum altitude, and descend on course to base (speed brakes in). Climb, using Military thrust at the speed schedules indicated in the notes (refer to Part 3 for detailed climb performance). Cruise at the recommended Mach numbers. Descent with Idle thrust at 300 KIAS (Figure A7-1). All three procedures allow a 500-lb fuel reserve for a sea level destination.

Use of the tables is indicated in the following example:

Configuration drag index is 40.

Zero fuel weight .... 15000 lb

Fuel remaining .... 3000 lb

Initial altitude .... 15000 ft

Distance to base .... 450 n.mi.

Determine the proper procedure to obtain the required distance to base. The configuration drag index 40 table shows that by using procedure 3 the following information is obtained:

Total distance .... 469 n.mi.

Total time .... 57.6 min.

Optimum altitude .... 35000 ft

Climb at indicated Mach No. 0.85

Cruise at indicated Mach No. 0.85

Start 300 KIAS idle descent with 560 lb of fuel remaining

ACCELERATION TO CLIMB SPEED

The following may be used in obtaining a reasonable estimate of fuel flow and distance traveled in accelerating to climb speed:

Fuel flow .... 100 lb/min

Distance traveled .... 7.5 nmi/min

MAXIMUM RANGE

CONSTANT ALTITUDE CRUISE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

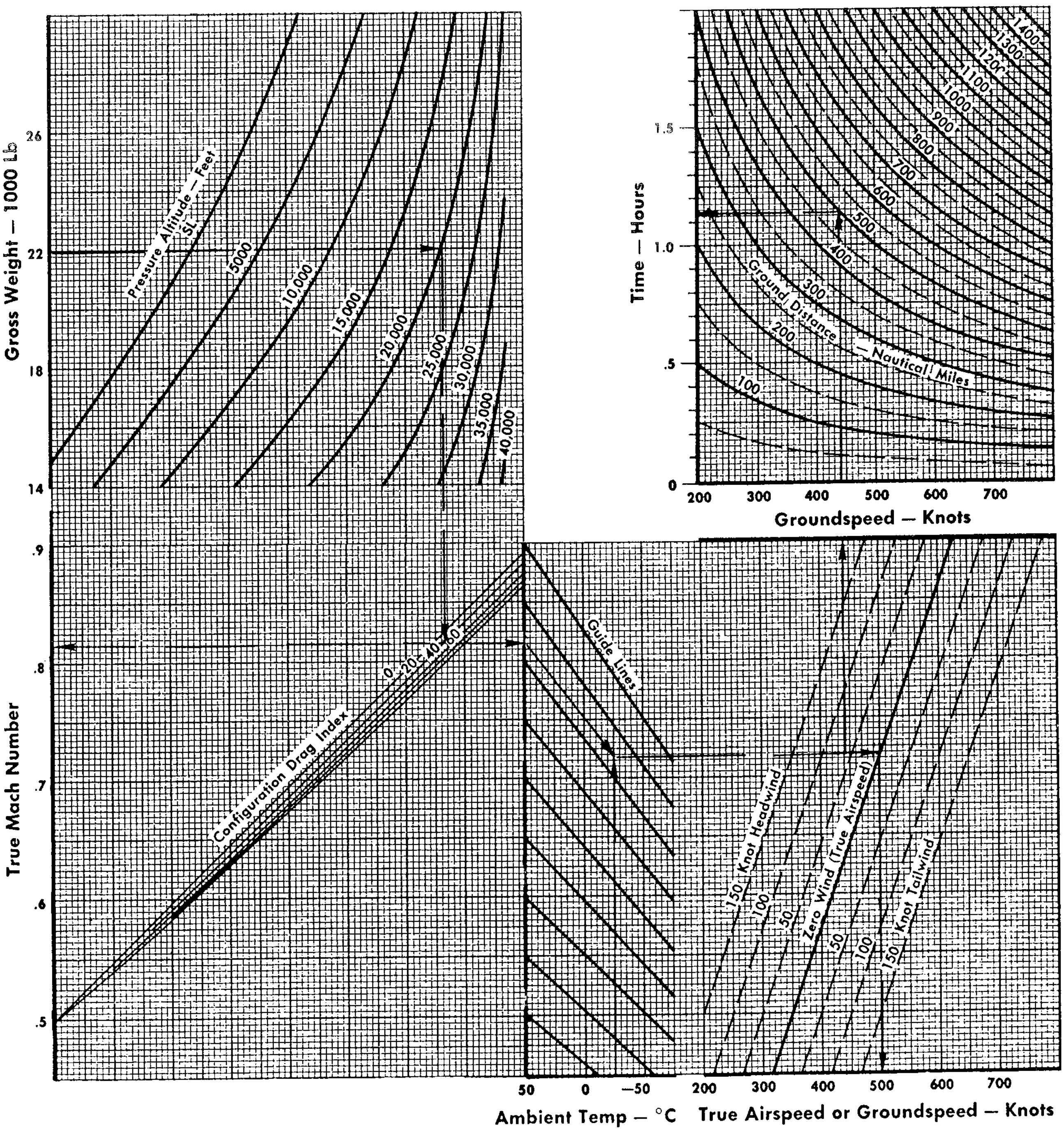


Figure A4-1 (Sheet 1 of 2)

MAXIMUM RANGE

CONSTANT ALTITUDE CRUISE

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

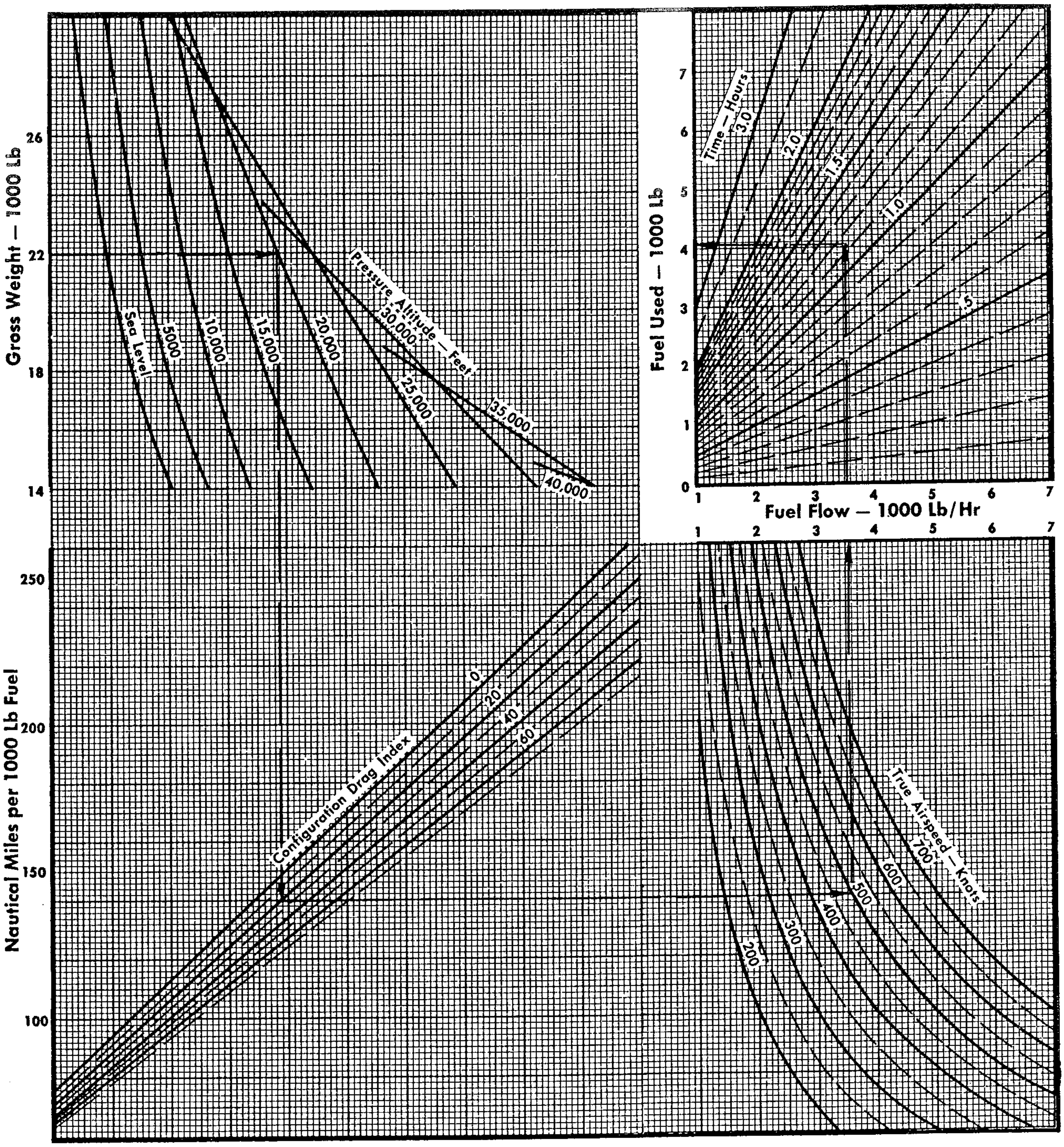


Figure A4-1 (Sheet 2 of 2)

MAXIMUM RANGE

CRUISE - CLIMB

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

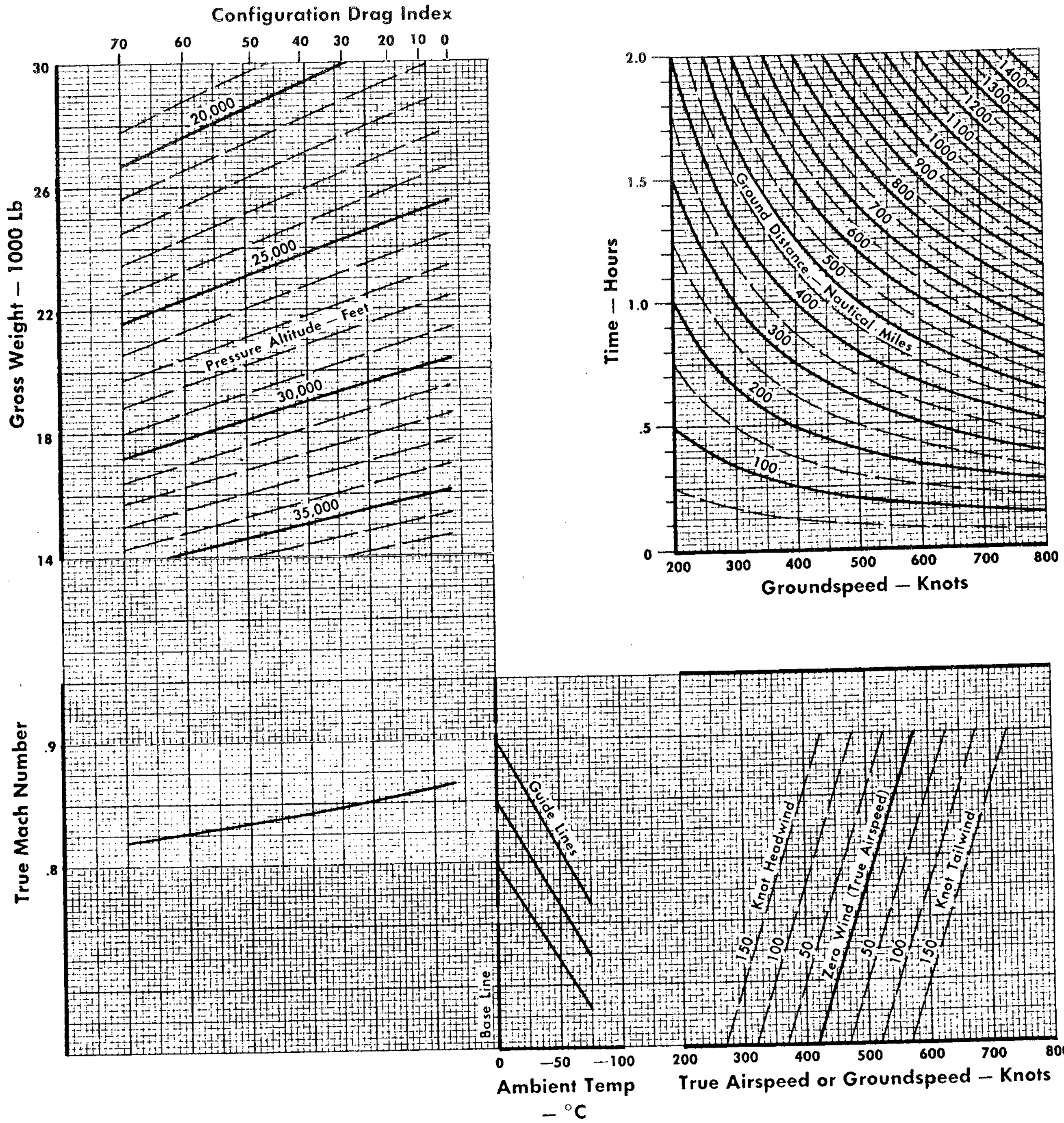


Figure A4-2 (Sheet 1 of 2)

MAXIMUM RANGE

CRUISE - CLIMB

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

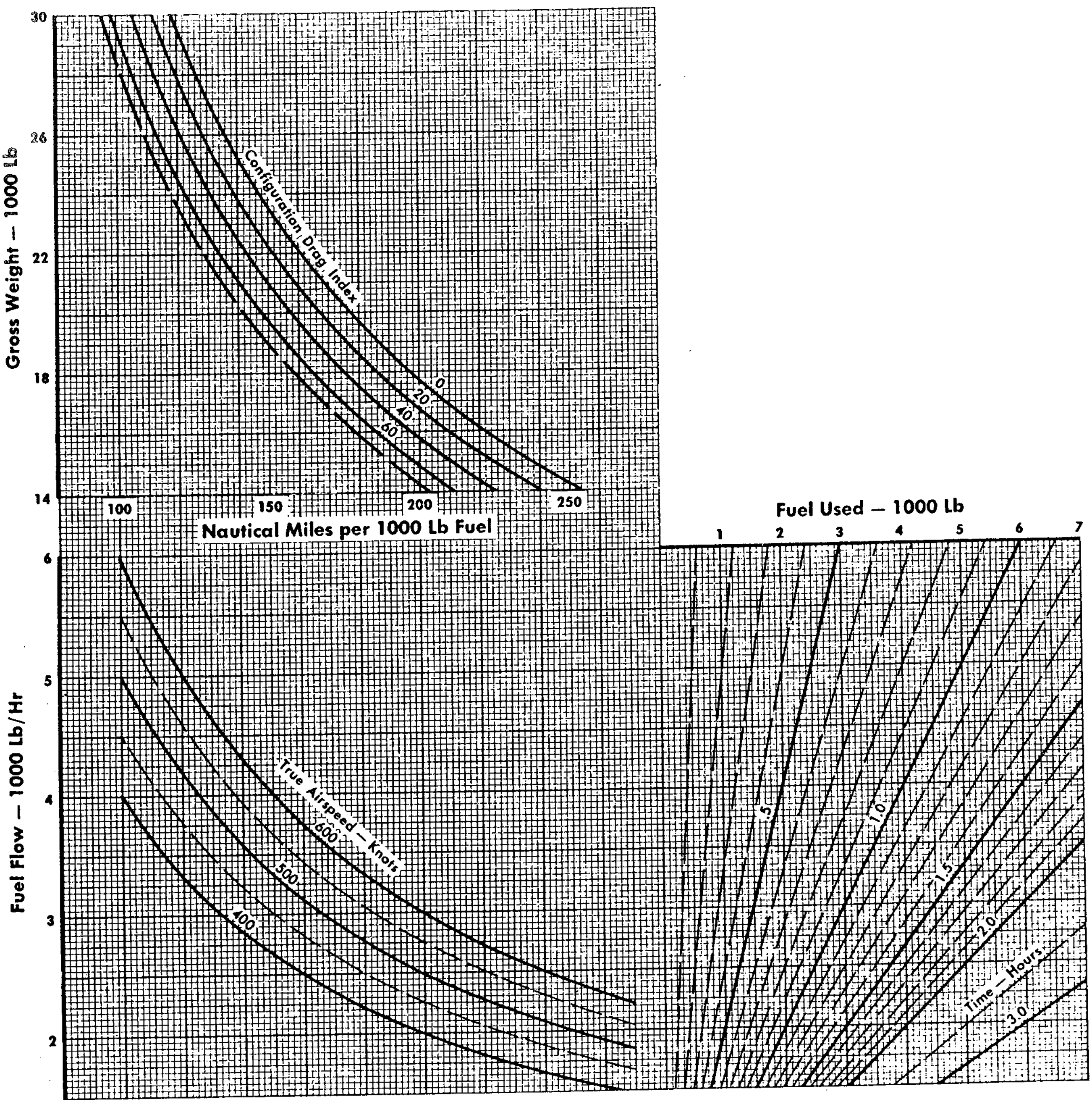


Figure A4-2 (Sheet 2 of 2)

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 14000 LB

Configuration Drag Index 0

Zero Wind Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FUEL	RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL — INITIAL ALTITUDE									PROCEDURE
	1000 FT	S.L.	5	10	15	20	25	30	35	
750 LB	NAUT. MI.	27	28	30	33	36	40	44	45	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	5.0	5.5	6.5	7.5	8.5	9.5	10.5	11.5	
	1000 FT	SL/15	5/20	10/25	15/30	25/35	25/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	27	29	32	36	40	45	50	53	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	5.0	5.0	6.0	6.5	7.0	8.0	9.0	9.5	
	NAUT. MI.	32	40	49	59	66	73	80	86	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	5.0	6.0	7.0	8.0	9.0	10.0	11.0	11.5	
1000 LB	NAUT. MI.	53	58	64	71	79	90	99	104	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	9.5	10.5	11.5	12.5	14.0	16.0	17.5	19.0	
	1000 FT	25/35	30/35	30/35	30/35	30/35	30/35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	64	69	78	86	94	100	106	112	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	10.0	11.0	12.0	13.0	14.0	15.0	16.0	16.5	
	NAUT. MI.	92	99	109	118	126	133	139	145	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	12.0	13.0	14.0	15.5	16.5	17.5	18.0	19.0	
1500 LB	NAUT. MI.	106	116	130	146	165	188	209	221	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	19.0	20.0	22.0	23.5	25.5	28.0	31.0	33.0	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	171	182	192	201	209	216	222	228	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	23.5	25.0	26.5	27.0	28.0	29.0	30.0	30.5	
	NAUT. MI.	204	215	225	234	242	249	256	262	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	25.5	27.0	28.5	29.5	30.5	31.5	32.0	33.0	
2000 LB	NAUT. MI.	158	174	195	220	250	284	316	333	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	28.0	29.0	31.5	34.0	36.5	40.0	44.0	46.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	289	297	305	313	322	328	335	341	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	38.0	39.0	39.5	41.0	41.5	42.5	43.5	44.5	
	NAUT. MI.	322	331	340	348	355	362	368	374	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	39.5	41.0	42.0	43.0	44.0	45.0	45.5	46.5	
3000 LB	NAUT. MI.	259	288	324	367	418	473	524	548	CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:45.0	:47.0	:50.0	:53.5	:58.5	1:03.5	1:09.0	1:12.0	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	502	512	521	529	537	543	550	556	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	1:04.0	1:05.0	1:06.0	1:07.0	1:07.5	1:08.5	1:09.5	1:10.0	
	NAUT. MI.	536	546	555	562	570	577	583	589	USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	1:05.0	1:07.0	1:08.0	1:09.0	1:10.0	1:11.0	1:11.5	1:12.0	
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	
TRUE MACH NO.		.50	.56	.61	.67	.73	.78	.82	.85	
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41	
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0	
	FUEL REM.		515	525	535	545	550	555	560	

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT	MACH
17–19000 LB	30000 FT	0.85
19–21000 LB	30000 FT	0.86

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.90 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 14,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-3

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 14000 LB

Configuration Drag Index 10

Zero Wind — Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FUEL	RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL — INITIAL ALTITUDE 1000 FT	S.L.	5	10	15	20	25	30	35	PROCEDURE
750 LB	NAUT. MI.	26	27	30	32	35	39	43	44	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.5	5.5	6.5	7.5	8.5	9.5	10.5	11.5	
	1000 FT	SL/15	5/20	10/25	15/30	20/30	25/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	26	28	31	34	39	44	48	51	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.5	5.0	5.5	6.0	7.0	7.5	8.5	9.5	
	NAUT. MI.	31	40	47	57	64	72	78	85	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	5.0	6.0	7.0	8.0	9.0	10.0	11.0	11.5	
1000 LB	NAUT. MI.	51	56	62	68	77	87	97	102	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	9.0	10.0	11.5	12.5	14.0	15.5	17.5	18.5	
	1000 FT	20/30	30/35	30/35	30/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	60	65	74	83	90	97	102	109	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	10.0	10.5	11.5	12.5	13.5	14.5	15.5	16.5	
	NAUT. MI.	87	95	104	114	122	130	136	143	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	12.0	12.5	13.5	15.0	16.0	17.0	18.0	18.5	
1500 LB	NAUT. MI.	102	112	126	141	161	182	202	214	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	18.0	19.5	21.0	23.0	25.0	27.5	30.5	32.0	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	163	174	184	196	201	209	215	222	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	22.5	24.0	25.5	26.5	27.5	28.5	29.0	30.0	
	NAUT. MI.	197	208	218	227	235	243	249	255	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	25.0	26.5	27.5	28.5	29.5	30.5	31.5	32.0	
2000 LB	NAUT. MI.	152	168	189	213	243	276	307	324	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	27.0	29.0	30.5	32.5	35.5	39.5	43.0	45.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	277	286	294	303	311	319	325	331	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	36.5	38.0	38.5	39.5	40.5	42.0	42.5	43.0	
	NAUT. MI.	311	320	328	337	345	352	358	365	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	39.0	40.0	41.0	42.0	43.0	44.0	44.5	45.5	
3000 LB	NAUT. MI.	250	278	314	355	405	459	509	533	CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:43.5	:46.0	:49.0	:52.5	:57.0	1:02.0	1:07.5	1:10.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	484	494	504	513	521	528	534	540	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	1:01.5	1:03.0	1:04.0	1:05.0	1:06.0	1:07.0	1:07.5	1:08.5	
	NAUT. MI.	519	530	540	546	554	561	567	574	USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	1:04.5	1:05.5	1:06.5	1:07.0	1:08.0	1:09.0	1:09.5	1:10.5	
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	
TRUE MACH NO.		.54	.59	.65	.71	.76	.81	.84	.87	
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41	
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0	
	FUEL REM.		515	525	535	545	550	555	560	

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT.	MACH
16–18000 LB	30000 FT	0.84
18–20000 LB	30000 FT	0.85
20–22000 LB	25000 FT	0.84
22–24000 LB	25000 FT	0.85

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.90 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 14,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-4

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 14000 LB

Configuration Drag Index 20

Zero Wind | Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FUEL	RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL — INITIAL ALTITUDE									PROCEDURE
	1000 FT	S.L.	5	10	15	20	25	30	35	
750 LB	NAUT. MI.	25	27	29	31	35	38	41	42	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.5	5.5	6.5	7.5	8.5	9.5	10.5	11.5	
	1000 FT	SL/15	5/20	10/25	15/30	20/35	25/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	25	27	30	34	38	42	47	50	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.5	4.5	5.5	6.5	7.0	7.5	8.5	9.5	
	NAUT. MI.	30	38	46	55	63	71	78	84	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	4.5	5.5	6.5	8.0	9.0	10.0	10.5	11.5	
1000 LB	NAUT. MI.	50	54	60	67	76	85	94	99	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	9.0	10.0	11.0	12.5	14.0	15.5	17.0	18.5	
	1000 FT	20/35	25/35	30/35	30/35	30/35	30/35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	57	65	74	79	87	94	100	106	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	9.0	10.5	12.0	12.5	13.5	14.5	15.0	16.0	
	NAUT. MI.	82	90	101	110	119	127	134	139	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	11.0	12.0	13.5	14.5	16.0	17.0	17.5	18.0	
1500 LB	NAUT. MI.	99	109	123	138	157	177	197	208	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	17.5	19.0	20.5	22.5	24.5	27.0	29.5	31.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	162	172	181	190	198	205	210	215	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	23.0	24.0	25.5	26.5	27.5	28.0	29.0	29.5	
	NAUT. MI.	196	206	216	224	232	239	243	249	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	25.0	26.0	27.5	28.5	29.5	30.5	31.0	31.5	
2000 LB	NAUT. MI.	148	164	185	209	238	269	298	315	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	26.0	28.0	30.0	32.5	35.5	39.0	42.0	44.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	272	281	290	299	306	312	316	322	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	36.0	37.5	38.5	39.5	40.0	41.0	41.5	42.5	
	NAUT. MI.	305	315	324	332	340	346	350	356	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	38.0	39.5	40.5	41.5	42.5	43.5	44.0	44.5	
3000 LB	NAUT. MI.	243	271	307	348	396	448	495	516	CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:42.5	:45.0	:48.0	:51.5	:55.5	1:01.0	1:06.5	1:09.0	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	477	486	493	500	507	513	519	525	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	1:01.0	1:02.0	1:03.0	1:03.5	1:04.5	1:05.5	1:06.0	1:07.0	
	NAUT. MI.	511	520	528	536	542	548	554	559	USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	1:03.0	1:04.0	1:05.5	1:06.5	1:07.0	1:08.0	1:08.5	1:09.5	

		S.L.	5	10	15	20	25	30	35
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35
TRUE MACH NO.		.54	.58	.65	.70	.76	.80	.84	.87
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0
	FUEL REM.		515	525	535	545	550	555	560

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT	MACH
16 – 18000 LB	30000 FT	0.84
18 – 20000 LB	30000 FT	0.85
20 – 22000 LB	25000 FT	0.84
22 – 24000 LB	25000 FT	0.85
24 – 26000 LB	20000 FT	0.83

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 14,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT. AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-5

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 14000 LB

Configuration Drag Index 30

Zero Wind | Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL

FUEL	INITIAL ALTITUDE 1000 FT	S.L.	5	10	15	20	25	30	35	PROCEDURE
750 LB	NAUT. MI.	24	26	28	30	34	37	40	41	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.5	5.0	6.0	7.0	8.0	9.0	10.5	11.5	
	1000 FT	SL/15	5/20	10/25	15/30	20/30	25/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	24	26	30	33	37	41	46	48	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.5	5.0	5.5	6.0	7.0	7.5	8.5	9.0	
	NAUT. MI.	29	36	45	53	62	68	75	82	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	4.0	5.5	6.5	7.5	8.5	9.5	10.5	11.5	
1000 LB	NAUT. MI.	48	53	59	65	74	83	92	96	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	9.0	10.0	11.0	12.0	13.5	15.0	16.5	18.0	
	1000 FT	15/30	25/30	25/35	30/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	54	62	70	78	85	91	97	103	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	9.0	10.0	11.0	12.5	13.0	14.0	15.0	15.5	
	NAUT. MI.	82	92	101	110	116	123	130	137	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	11.0	12.0	13.5	14.5	15.0	16.0	17.0	18.0	
1500 LB	NAUT. MI.	96	107	120	135	154	173	193	202	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	17.0	18.5	20.5	22.0	24.0	26.5	29.5	31.0	
	1000 FT	30/35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	153	164	174	183	192	199	202	209	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	21.5	23.0	24.5	25.5	26.5	27.5	28.0	29.0	
	NAUT. MI.	191	200	210	218	226	233	236	243	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	24.0	26.0	27.0	27.5	29.0	30.0	30.0	31.0	
2000 LB	NAUT. MI.	144	160	180	204	232	262	292	306	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	25.5	27.5	29.5	31.5	35.0	38.0	41.5	43.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	260	269	278	287	295	301	306	312	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	34.5	36.0	37.0	38.0	39.5	40.0	40.5	41.5	
	NAUT. MI.	295	305	314	322	330	336	340	347	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	36.5	38.5	39.5	40.5	41.5	42.5	43.0	43.5	
3000 LB	NAUT. MI.	236	264	299	340	388	437	483	501	CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:41.0	:44.0	:47.0	:50.5	:55.0	1:00.0	1:05.0	1:07.5	
	1000 FT	35	35	35	35	35	35	35	35	OPTIMUM ALTITUDE
	NAUT. MI.	458	468	477	484	491	497	501	508	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	:59.0	1:00.0	1:01.5	1:02.0	1:03.0	1:04.0	1:04.0	1:05.0	
	NAUT. MI.	493	503	511	519	525	531	535	542	USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	1:01.5	1:02.5	1:03.5	1:04.0	1:05.0	1:06.0	1:06.5	1:07.5	

		S.L.	5	10	15	20	25	30	35
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35
TRUE MACH NO.		.54	.59	.64	.70	.75	.80	.83	.86
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0
	FUEL REM.		515	525	535	545	550	555	560

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT	MACH
17–19000 LB	30000 FT	0.84
19–21000 LB	25000 FT	0.83
21–23000 LB	25000 FT	0.84
23–25000 LB	25000 FT	0.85
25–27000 LB	20000 FT	0.83

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 14,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT. AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-6

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 15000 LB

Configuration Drag Index **40**

Zero Wind — Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL

FUEL	INITIAL ALTITUDE 1000 FT	S.L.	5	10	15	20	25	30	35	PROCEDURE
750 LB	NAUT. MI.	23	25	27	29	32	35	38	38	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	
	1000 FT	SL/10	5/15	10/20	15/25	20/30	25/30	30	35	OPTIMUM ALTITUDE
	NAUT. MI.	23	25	28	32	35	39	43	44	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.0	4.5	5.0	6.0	6.5	7.5	8.0	8.5	
	NAUT. MI.	27	33	42	50	58	65	72	79	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	4.0	5.0	6.0	7.5	8.5	9.0	10.0	11.0	
1000 LB	NAUT. MI.	46	51	56	63	70	79	86	87	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	8.0	9.0	10.5	11.5	13.0	14.5	16.0	17.0	
	1000 FT	10/25	20/30	25/30	25/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	48	55	63	71	77	84	91	93	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	7.5	8.5	10.0	11.0	12.5	13.0	14.0	14.5	
	NAUT. MI.	71	82	92	102	111	118	124	128	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	9.5	11.0	12.5	13.5	15.0	15.5	16.5	17.0	
1500 LB	NAUT. MI.	92	102	115	130	147	165	180	183	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	16.0	17.5	19.0	21.0	23.0	25.5	27.5	28.5	
	1000 FT	30/35	30/35	30/35	30/35	30/35	30/35	30/35	30/35	OPTIMUM ALTITUDE
	NAUT. MI.	139	149	158	166	173	179	185	189	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	19.5	21.0	22.0	23.0	24.0	25.0	25.5	26.0	
	NAUT. MI.	172	182	191	200	208	211	217	224	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	22.0	23.5	24.5	25.5	26.5	27.0	28.0	28.5	
2000 LB	NAUT. MI.	137	153	173	196	222	249	273	275	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	24.0	25.5	27.5	30.0	33.0	36.0	39.0	40.0	
	1000 FT	30/35	30/35	30/35	30/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	232	242	251	259	267	271	278	281	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	31.0	32.0	33.5	34.5	35.5	36.0	37.0	37.5	
	NAUT. MI.	264	274	283	292	300	307	310	316	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	33.5	34.5	36.0	37.0	38.0	39.0	39.0	40.0	
3000 LB	NAUT. MI.	225	254	287	326	370	414	451	449	CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:38.0	:41.0	:44.0	:47.5	:52.0	:56.5	1:01.0	1:01.0	
	1000 FT	30/35	30/35	30/35	30/35	30/35	30/35	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	412	421	430	437	444	450	456	459	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	:53.0	:54.0	:55.5	:56.5	:57.5	:58.0	:59.0	:59.0	
	NAUT. MI.	443	453	461	469	475	481	485	490	USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	:54.0	:55.0	:56.5	:57.5	:58.5	:59.0	1:01.0	1:01.5	
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	
TRUE MACH NO.		.53	.58	.64	.69	.75	.79	.83	.85	
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41	
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0	
	FUEL REM.		515	525	535	545	550	555	560	

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT	MACH
17 – 19000 LB	30000 FT	0.84
19 – 21000 LB	25000 FT	0.82
21 – 23000 LB	25000 FT	0.83
23 – 25000 LB	20000 FT	0.82
25 – 27000 LB	20000 FT	0.83

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON – COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 15,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-7

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 15000 LB

Configuration Drag Index 50

Zero Wind | Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FUEL	RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL — INITIAL ALTITUDE									PROCEDURE
	1000 FT	S.L.	5	10	15	20	25	30	35	
750 LB	NAUT. MI.	23	24	26	29	32	35	37	36	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.0	5.0	6.0	7.0	8.0	9.0	10.0	10.5	
	1000 FT	SL/10	5/15	10/20	15/25	20/30	25/30	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	23	25	28	31	35	39	42	42	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.0	4.5	5.0	5.5	6.5	7.0	8.0	8.0	
	NAUT. MI.	26	33	40	49	57	64	71	77	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	4.0	5.0	6.0	7.0	8.0	9.0	10.0	10.5	
1000 LB	NAUT. MI.	46	50	55	62	69	77	84	84	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	8.0	9.0	10.0	11.5	13.0	14.5	15.5	16.5	
	1000 FT	10/25	15/30	25/30	25/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	48	56	63	70	77	82	89	90	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	7.5	9.0	10.0	11.5	12.5	12.5	13.5	14.0	
	NAUT. MI.	68	79	90	99	107	114	119	125	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	9.0	10.5	12.0	13.5	14.7	15.5	16.0	16.5	
1500 LB	NAUT. MI.	91	100	113	127	144	162	176	175	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	16.0	17.0	19.0	20.5	23.0	25.0	27.5	28.0	
	1000 FT	30/35	30/35	30/35	30/35	30/35	30/35	30/35	35	OPTIMUM ALTITUDE
	NAUT. MI.	134	144	153	161	167	174	181	181	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	19.0	20.0	21.5	23.0	23.5	24.5	25.0	25.5	
	NAUT. MI.	165	174	183	192	199	204	211	216	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	21.5	22.5	23.5	25.0	26.2	26.5	27.5	28.0	
2000 LB	NAUT. MI.	135	150	169	193	218	245	266	264	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	24.0	25.0	27.0	30.0	32.5	35.5	38.5	39.0	
	1000 FT	30/35	30/35	30/35	30/35	30/35	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	225	235	244	252	260	264	271	274	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	30.0	32.0	33.0	34.0	35.0	35.5	36.5	36.5	
	NAUT. MI.	254	264	274	282	289	293	300	305	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	32.0	33.5	35.0	35.5	36.5	37.5	38.5	39.0	
3000 LB	NAUT. MI.	223	248	282	321	363	407	440	429	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	38.0	40.5	43.5	47.5	51.0	56.0	60.0	59.0	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	398	409	418	426	433	438	445	447	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	51.5	53.0	54.0	55.0	56.5	57.0	57.5	58.0	
	NAUT. MI.	428	439	448	456	463	467	474	478	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	53.5	55.5	56.0	57.5	58.5	59.0	60.0	60.5	

CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35
TRUE MACH NO.		.53	.58	.63	.69	.74	.79	.82	.85
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0
	FUEL REM.		515	525	535	545	550	555	560

WITH OVER 3000 LB OF FUEL REMAINING – USE:

GR WEIGHT	ALT	MACH
18–20000 LB	25000 FT	0.81
20–22000 LB	25000 FT	0.83
22–24000 LB	25000 FT	0.84
24–26000 LB	20000 FT	0.82
26–28000 LB	20000 FT	0.83

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON–COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 15,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-8

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 15000 LB

Configuration Drag Index 60

Zero Wind   Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

FUEL	RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL — INITIAL ALTITUDE									PROCEDURE
	1000 FT	S.L.	5	10	15	20	25	30	35	
750 LB	NAUT. MI.	22	23	26	28	31	34	36	35	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.0	5.0	6.0	7.0	8.0	9.0	10.0	10.5	
	1000 FT	SL/10	5/15	10/20	15/25	20/30	25/30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	22	24	27	30	34	38	41	44	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.0	4.5	5.0	5.5	6.0	7.0	8.0	8.0	
	NAUT. MI.	26	32	40	48	56	63	70	76	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	3.5	5.0	6.0	7.0	8.0	9.0	10.0	10.5	
1000 LB	NAUT. MI.	44	48	54	60	68	76	82	80	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	8.0	9.0	10.0	11.5	12.5	14.0	15.5	16.0	
	1000 FT	10/25	15/30	20/30	25/30	25/30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	46	53	61	68	73	80	87	90	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	7.0	9.0	10.0	11.5	11.5	12.5	13.5	14.0	
	NAUT. MI.	66	77	87	96	104	109	116	122	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	9.0	10.5	12.0	13.0	14.0	14.5	15.5	16.5	
1500 LB	NAUT. MI.	87	97	110	124	141	158	172	167	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	15.0	16.5	18.5	20.5	22.5	24.5	27.0	26.5	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	128	138	147	156	162	170	177	180	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	18.5	19.5	21.0	22.0	23.0	24.0	25.0	25.0	
	NAUT. MI.	157	167	177	185	193	199	206	212	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	20.5	21.5	23.0	24.5	25.0	26.0	27.0	28.0	
2000 LB	NAUT. MI.	130	146	166	188	213	239	260	251	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	22.5	24.5	27.0	29.0	32.0	35.0	38.0	37.5	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	216	226	236	244	250	258	265	268	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	29.5	30.5	32.0	33.0	34.0	35.0	36.0	36.0	
	NAUT. MI.	245	256	265	274	282	287	294	300	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	31.5	33.5	34.0	35.5	36.5	37.0	38.0	38.5	
3000 LB	NAUT. MI.	214	242	275	313	355	398	428	407	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	37.0	40.0	43.0	46.5	50.5	55.0	58.5	56.5	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	386	396	406	413	419	426	433	436	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	50.5	51.5	53.0	54.0	55.0	56.0	56.5	57.0	
	NAUT. MI.	415	425	434	442	448	455	462	468	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	52.5	53.5	55.0	56.5	57.0	58.0	58.5	59.5	
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	WITH OVER 3000 LB OF FUEL REMAINING – USE:
TRUE MACH NO.		.53	.58	.63	.69	.74	.78	.82	.84	
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.		5	10	15	21	27	34	41	
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0	
	FUEL REM.		515	525	535	545	550	555	560	

GR WEIGHT	ALT	MACH
18–20000 LB	25000 FT	0.81
20–22000 LB	25000 FT	0.82
22–24000 LB	20000 FT	0.80
24–26000 LB	20000 FT	0.82
26–28000 LB	20000 FT	0.83

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 15,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-9

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 15000 LB

Configuration Drag Index 70

Zero Wind — Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL

FUEL	INITIAL ALTITUDE 1000 FT	S.L.	5	10	15	20	25	30	35	PROCEDURE
750 LB	NAUT. MI.	21	23	25	27	30	33	35	33	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	4.0	4.5	5.5	6.5	7.5	9.0	9.5	10.5	
	1000 FT	SL/10	5/15	10/20	15/25	20/30	25/30	30	35	OPTIMUM ALTITUDE
	NAUT. MI.	21	24	26	29	33	37	40	39	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	4.0	4.0	5.0	5.5	6.0	7.0	7.5	8.0	
	NAUT. MI.	25	31	38	47	55	62	69	74	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	4.5	4.5	6.0	7.0	8.0	9.0	9.5	10.5	
1000 LB	NAUT. MI.	43	47	53	59	66	74	79	77	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	7.5	8.5	10.0	11.0	12.5	14.0	15.5	16.0	
	1000 FT	10/20	15/25	20/30	25/30	25/30	30	30	35	OPTIMUM ALTITUDE
	NAUT. MI.	45	52	59	65	72	78	84	82	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	7.0	8.5	9.5	10.5	11.5	12.5	13.5	13.0	
	NAUT. MI.	60	72	83	91	99	107	113	118	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	8.5	10.0	11.5	12.5	13.5	14.5	15.5	16.0	
1500 LB	NAUT. MI.	85	95	107	121	138	154	167	161	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	15.0	16.0	18.0	20.0	22.0	24.5	26.5	26.0	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	121	132	141	149	157	165	171	175	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	17.0	19.0	20.5	21.0	22.0	23.0	24.0	24.5	
	NAUT. MI.	151	161	170	178	187	194	201	206	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	19.5	21.0	22.5	23.5	24.5	25.5	26.5	27.0	
2000 LB	NAUT. MI.	127	142	162	183	208	234	252	240	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	22.0	24.0	26.0	28.5	31.5	34.5	37.0	36.0	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	207	217	227	236	242	250	256	260	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	28.5	29.5	31.0	32.0	33.0	34.0	35.0	35.0	
	NAUT. MI.	236	247	256	266	274	279	285	292	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	30.0	31.5	33.0	34.0	35.5	36.0	37.0	38.0	
3000 LB	NAUT. MI.	209	236	269	306	347	389	414	389	CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	35.5	38.0	41.5	45.0	49.5	54.0	57.0	54.0	
	1000 FT	30	30	30	30	30	30	30	30	OPTIMUM ALTITUDE
	NAUT. MI.	371	381	391	399	405	412	419	422	USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	48.0	50.0	51.0	53.0	53.0	54.0	54.5	55.0	
	NAUT. MI.	401	412	421	429	436	443	448	453	USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	50.5	52.0	53.5	54.5	55.5	56.5	57.0	57.5	
CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	
TRUE MACH NO.		.53	.58	.63	.68	.74	.78	.82	.84	
START 300 KIAS IDLE DESCENT WITH ④	MI. REM.		5	10	15	21	27	34	41	
	MIN. REM.		1.0	2.0	2.5	3.5	4.5	5.5	6.0	
	FUEL REM.		515	525	535	545	550	555	560	

WITH OVER 3000 LB OF FUEL REMAINING - USE:

GR WEIGHT	ALT	MACH
18–20000 LB	25000 FT	0.80
20–22000 LB	25000 FT	0.82
22–24000 LB	20000 FT	0.80
24–26000 LB	20000 FT	0.81
26–28000 LB	15000 FT	0.79

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.85 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 15,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT, AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-10

PART 5

ENDURANCE

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

ENDURANCE DATA PRESENTED

Endurance data presented in this part show the endurance available at constant altitude and are based on operation at the recommended speed schedules including a 5% operational tolerance. Performance at other speed schedules may be obtained from the data shown on the miles-per-pound charts contained in Part 6.

ENDURANCE CHART INDEX

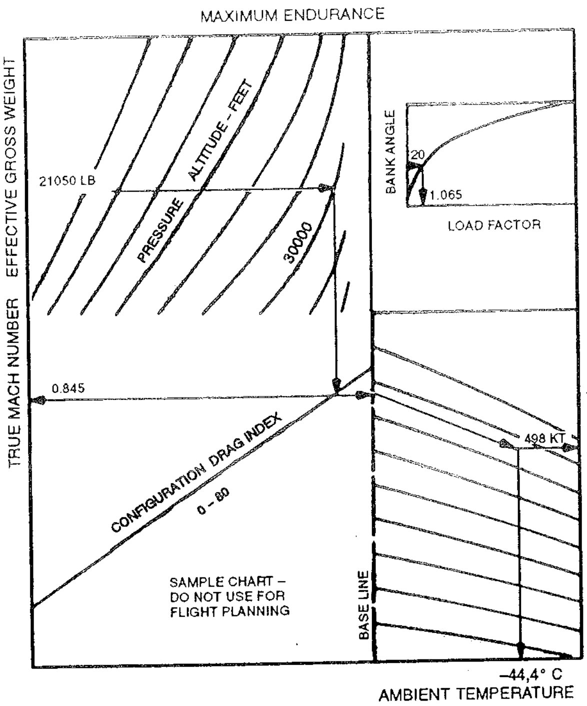
The endurance charts include a configuration drag index. Typical configurations applicable to the index are illustrated on a chart in Part 1. If the performance is desired for a configuration not illustrated, refer to the index shown in Part 1. Determine the configuration drag index, and use the appropriate line on the endurance chart. Endurance performance which is available by dropping external stores is not shown by special charts. Such performance may be determine by referring to the appropriate index line for the initial and subsequent configuration drag index numbers.

MAXIMUM ENDURANCE PERFORMANCE

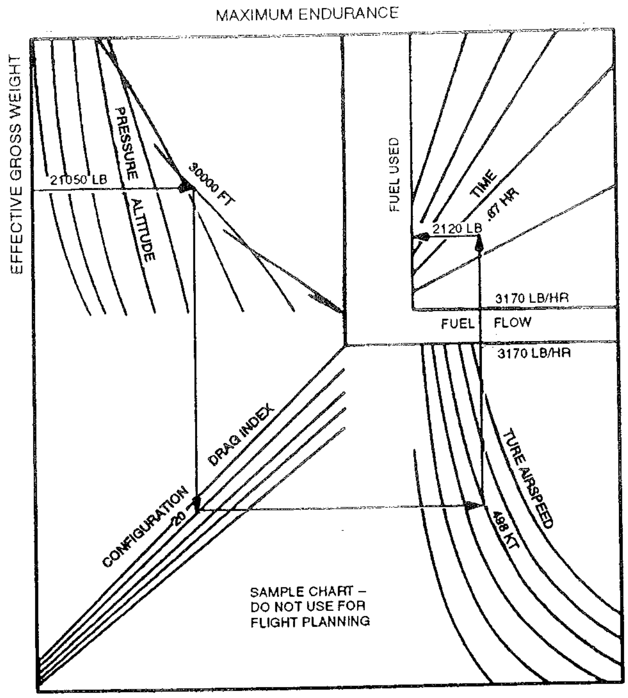
The maximum endurance charts show the loiter time available at a constant altitude for any gross weight. The performance is presented in terms of loiter fuel flow and recommended loiter speed, and does not include climb or descent allowances. The recommended loiter speeds shall be used to obtain the predicted performance.
Endurance performance in turning flight may be estimated on the basis of an effective gross weight calculated from the load factor-bank angle chart provided. Effective gross weight equals gross weight multiplied by the load factor in the turn.

MAXIMUM ENDURANCE PERFORMANCE SAMPLE PROBLEM

Determine the fuel required to loiter 40 minutes in a 20° bank at 30000 feet on a standard day for a configuration with a drag index of 20. The fuel on board at the start is 5000 lb. Assumed zero fuel weight is 14800 pounds.



FA0324



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a. The aircraft gross weight is 19800 lb (14800 + 5000).

b. Enter the load factor-bank angle plot at a bank angle of 20° and read the load factor, 1.065 "G".

c. The initial loiter effective gross weight is 21050 lbs. (19800 × 1.065).

d. Enter Figure A5-1 (sheet 1) at 21050 lb and 30000 feet. Read loiter true Mach number of 0.845. Proceed horizontally to the right to the true airspeed base line, follow the guide lines to −44.4° C, standard day temperature at 30000 feet. Continue to the right and read true airspeed of 498 knots.

e. Enter Figure A5-1 (sheet 2) at 21050 lb and 30000 feet. Proceed vertically below this point to a configuration drag index of 20. Horizontally to the right intersect the true airspeed determined in step (d), 498 knots.

f. Vertically above the point in step (e), read loiter fuel flow, 3170 pounds per hour. Continue to the desired loiter time of 0.67 hr (40 min.). Read fuel used for loiter, 2120 pounds (0.67 × 3170).

The preceding steps assume loiter performance based on the initial condition for loiter. Additional accuracy in the determination of the fuel requirements for loiter for the preceding conditions may be obtained by approximating the average loiter gross weight.

g. Approximate final gross weight is 17680 lb (19800 − 2120).

h. Average gross weight is 18740 lb (17680 + 19800)/2.

i. The average effective gross weight is 19960 lb (18740 × 1.065).

j. Following the same procedures, enter Figure A5-1 at 19960 lb, and 30000 feet loiter true Mach 0.837, loiter true airspeed 493 knots, fuel flow 2960 pounds per hour. Fuel used for .67 hr (40 min.) is 1980 lb (.67 × 2960).

Determine the total endurance time for 5000 pounds of fuel remaining. When total loiter time is required for large quantities of fuel remaining, it is recommended that performance calculations be made in gross weight increments of 2000 pounds or less.

a. Average gross weight for 2000 pound increment from 5000 lb to 3000 lb fuel remaining is, 18800 lb, (19800 + 17800)/2. Effective gross weight is 20000 lb.

b. Loiter speed at 30000 feet is true Mach 0.837, true airspeed 493 knots, fuel flow 2960 lb per hour. Loiter time to consume 2000 lb of fuel is 40.5 minutes (.675 hr), (2000/2960).

c. Average gross weight for 2000 pound increment from 2000 lb to 1000 lb fuel remaining is 16800 lb, (17800 + 15800)/2. Effective gross weight is 17900 lb.

d. Loiter speed at 30000 feet is true Mach 0.812, true airspeed 478 knots, fuel flow 2620 lb per hour. Loiter time to consume 2000 lb of fuel is 45.8 minutes (.763 hr), (2000/2620)

e. Average gross weight for 1000 lb remaining is 15300 lb (15800 + 14800)/2. Effective gross weight is 16300 lb.

f. Loiter speed at 30000 feet is true Mach 0.785, true airspeed 462 knots, fuel flow 2360 lb per hour. Loiter time to consume 1000 lb of fuel is 25.4 minutes (.423 hr), (1000/2360).

g. Total loiter time is 1 hr. 51.7 min. (40.5 + 45.8 + 25.4).

MAXIMUM ENDURANCE

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

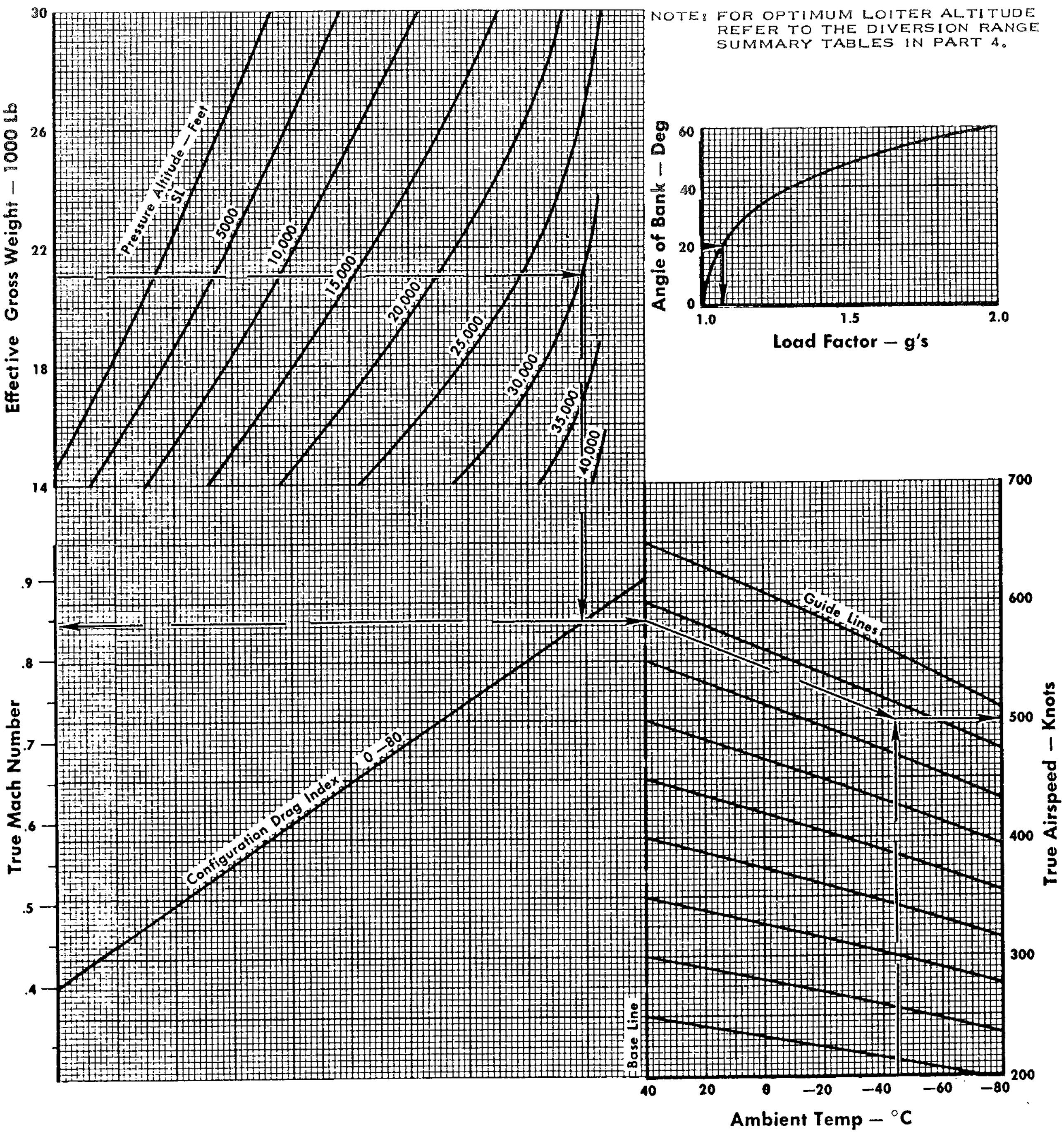


Figure A5-1 (Sheet 1 of 2)

MAXIMUM ENDURANCE

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

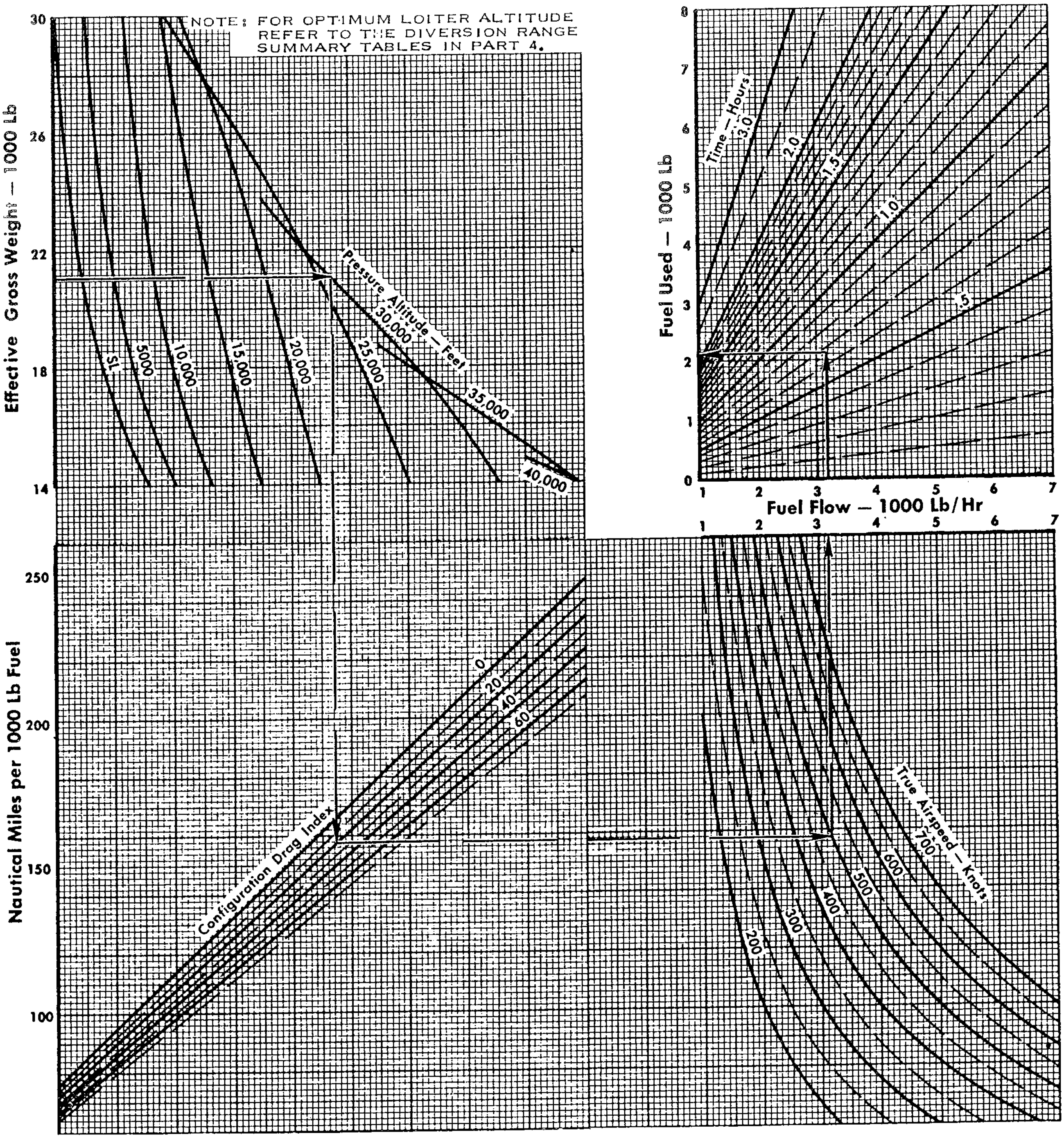


Figure A5-1 (Sheet 2 of 2)

PART 6

MILES PER POUND DATA

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

PURPOSE OF CHARTS

The purpose of the charts is to supply generalized flight planning data applicable to the cruise portion of any type of flight plan. The parametric miles-per-pound charts show subsonic level flight performance for operations between maximum endurance speeds and speeds resulting from use of normal thrust.

MILES PER POUND CHARTS

The nautical miles per pound of fuel charts present cruise data for the speed range from maximum endurance to normal thrust in a parametric form applicable to any altitude or gross weight. The chart parameters are basic plots of nautical miles per 1000 pounds of fuel $\times\ \delta$ versus Mach number for various gross weights divided by δ.
Conversion curves are supplied on the charts to find W/δ for any flight weight and altitude condition, and actual miles per 1000 pounds of fuel from the fuel economy parameter. Each chart includes the specific range available that will result from various speed settings and fuel flows. Engine speeds are not presented, because non-standard air conditions and the operating characteristics of individual engines have a decided effect on the RPM setting required to maintain cruise airspeeds.

The recommended procedure is to cruise at Mach number recommended and monitor fuel flow to compare actual miles per pound with that predicted. A Military thrust operating limit is not shown since Military thrust does not provide a significant speed increase over normal thrust, but does substantially increase fuel flow. Military thrust is considered an operational "pad" to allow maintenance of normal thrust speeds and altitude schedules when higher standard temperature conditions exist. Included are curves of recommended cruise Mach number and speeds for maximum endurance. Specific range is plotted versus true Mach numbers with conversion grids to obtain true airspeed for any ambient temperature.

NOTE

IAS = CAS and Indicated Mach = True Mach, because static position error calibration of the compensated pitot-static airspeed system is small and under normal operating conditions may usually be disregarded.

Miles per Pound Chart Index

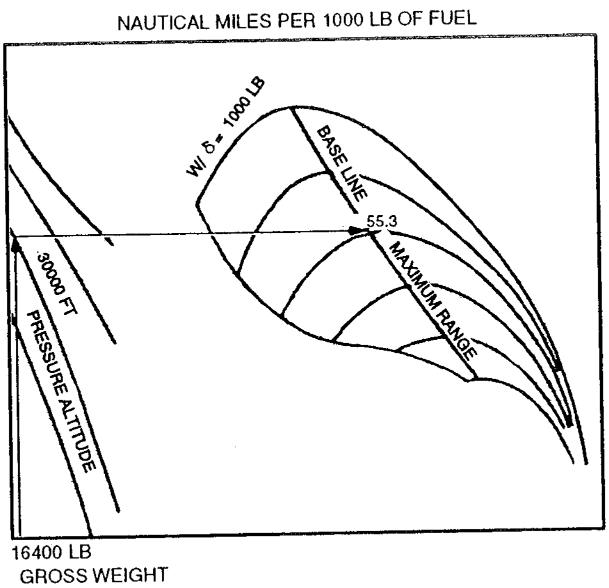
The miles-per-pound charts are arranged in groups of configurations with similar drag characteristics. Typical configurations applicable to the fuel economy presented are illustrated on a chart in Part 1, "Introduction". If the specific range is desired for a configuration not illustrated, refer to the index system in Part 1, determine the configuration drag index, and use the appropriate nautical miles per 1000 pounds of fuel chart.

USE OF MILES PER POUND CHARTS

The miles-per-pound charts may be used to determine range when the speed schedules shown on the range charts are not satisfactory for mission requirements. The condition might arise on a buddy mission for example, when a long range speed schedule shall be selected which is a compromise between the maximum range schedules for two types of aircraft.

MILES PER POUND CHART SAMPLE PROBLEM — RANGE

Determine the range available, time required, and fuel flow to cruise in level flight at a constant true Mach 0.90 at 30000 ft. The initial cruise gross weight is 17000 lb and a 1000-lb reserve is desired. Assume a standard day with a 60-knot headwind component. The aircraft has a configuration drag index of 0, zero fuel weight assumed is 14800 lb.

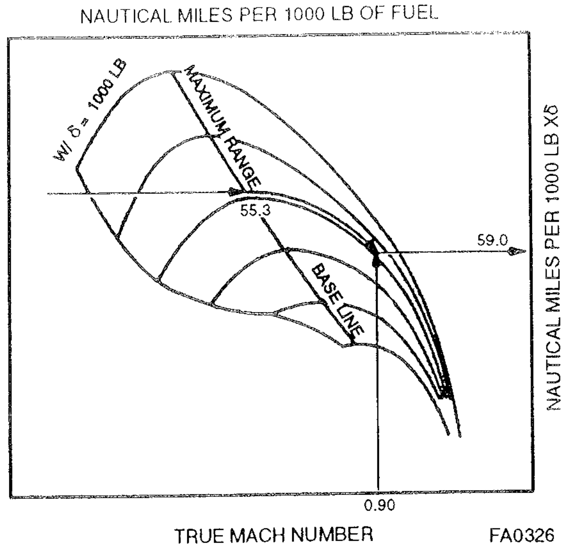


a. Gross weight with a 1000-lb reserve is 15800 lb (14800 + 1000).

b. Average cruise gross weight is 16400 lb (17000 + 15800)/2.

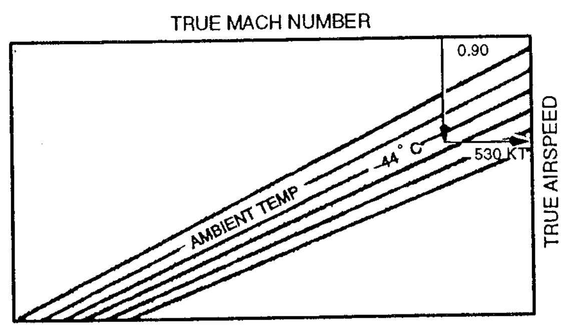
c. Enter the W/δ conversion chart at the average cruise gross weight and the cruise pressure altitude, proceed horizontally right to the base line, and read cruise W/δ of 55300 lb.

d. Enter the parametric fuel economy curve with the cruise true Mach of 0.90, proceed to the cruise W/δ, and read nautical miles per 1000 lb of fuel x = 59.0 nmi/1000 lb × δ.



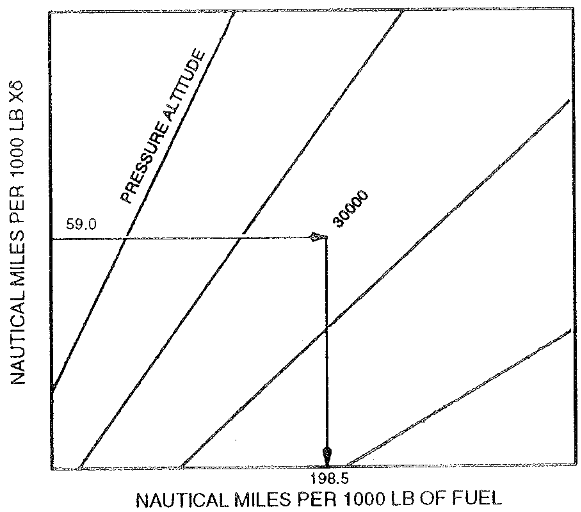
FA0326

e. From the same cruise true Mach number proceed vertically downward to −44° C, the standard temperature at 30000 feet. Horizontally from this point read cruise true airspeed of 530 knots.



f. Groundspeed, V_g, is 470 knots (530 − 60).

g. To determine average air nautical miles per pound of fuel enter chart on right at 59 obtained in step d., proceed to the cruise pressure altitude, and read 198.5 air nautical miles per 1000 lb of fuel.



h. Determine the average ground nmi/1000 lb by using the formula:

$$\frac{\text{ground nmi}}{1000\text{ lb fuel}} = \frac{\text{air nmi}}{1000\text{ lb fuel}} \times \frac{\text{GS}}{\text{TAS}} =$$

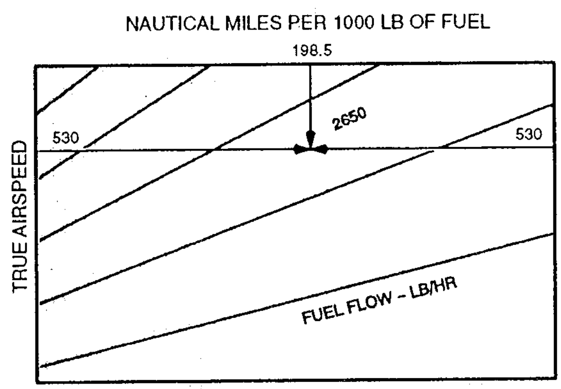
$$198.5 \times \left(\frac{470}{530}\right) = 176 \times \frac{\text{nmi}}{1000\text{ lb}}$$

i. Fuel to be used in cruise is 1200 lb (17000 − 15800).

j. Range available is 211.5 nmi = (176/1000 × 1200).

k. Time required is 27 min (215.5/470).

l. From the air nautical miles per 1000 lb of fuel obtained in step g., continue vertically downward and opposite the TAS read average fuel flow of 2650 lb/hr.



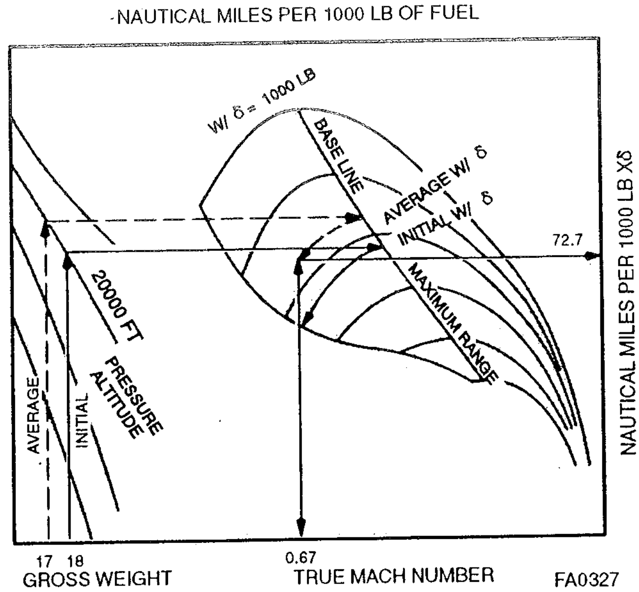
NAUTICAL MILES PER POUND SAMPLE PROBLEM – ENDURANCE

For an initial loiter gross weight of 18000 pounds, determine the loiter time at 20000 feet if 2000 pounds of fuel are available.

a. Determine the initial and average W/δ at the loiter altitude:

18000 lb at 20000 ft = 39200 lb

17000 lb at 20000 ft = 37000 lb



FA0327

b. Find the intersection of the initial W/δ on the maximum endurance schedule on the left side of the fuel economy curve (the recommended loiter speed):

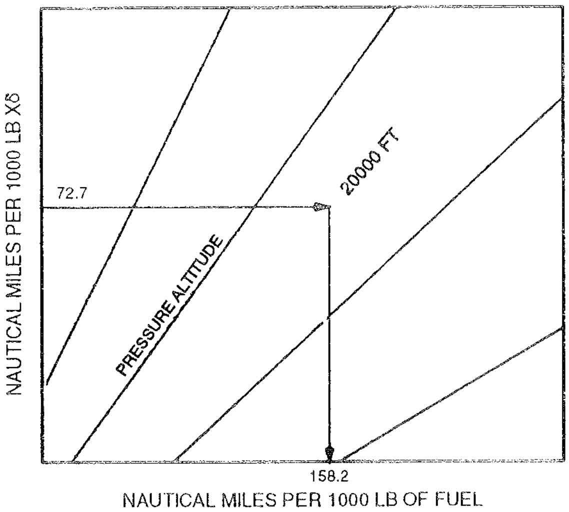
True Mach is 0.67

TAS at 20000 ft is 410 knots

c. Reenter at the average W/δ of 37000 lb. At the intersection of 37000 W/δ and the loiter Mach of 0.67 read 72.7 nautical miles per 1000 lb of fuel × δ.

d. Enter the right hand chart at 72.7 and the loiter altitude and read 158.2 nmi/1000 lb. Continue vertically downward opposite the loiter TAS and read average fuel flow of 2590 lb/hr.

e. Loiter time is 2000 lb fuel/2590 lb/hr = 0.77 hr, or 46 min.

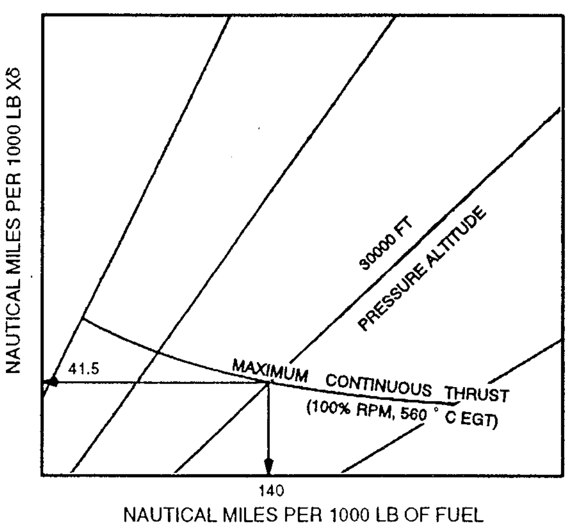


MILES PER POUND CHART SAMPLE PROBLEM – MAXIMUM CONTINUOUS THRUST

Reverse the order of sample problem sequence if the speed available at maximum continuous thrust is desired. Maximum continuous thrust is 100% RPM and an EGT of 560° C. For the 30000-foot example conditions of the range problem, at the intersection of the normal thrust line, the following result is obtained:

Nautical miles per 1000 lb of fuel × δ 41.5

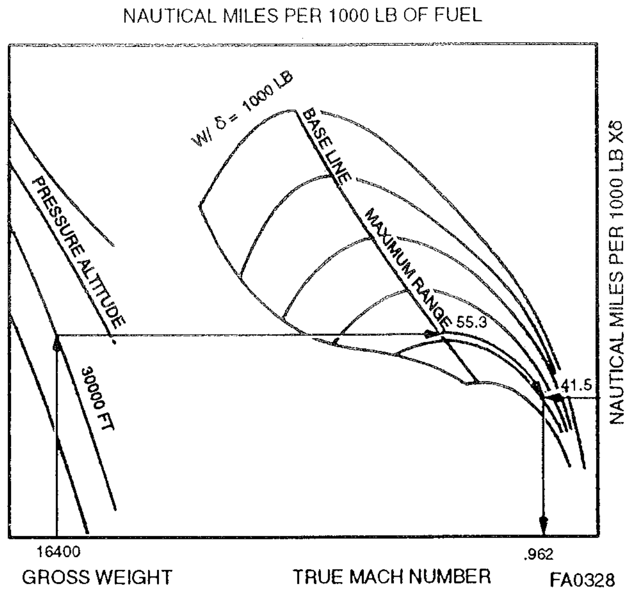
Actual fuel economy (nmi/1000 lb fuel) 140



Enter the left-hand chart at 41.5 (F.E. × δ) intersecting the 55300 lb W/δ and read:

True Mach number 0.962

True airspeed .... 567 knots



Reenter the right hand chart at 567 knots TAS and 140 nmi/1000 lb of fuel and read fuel flow of 4050 lb/hr.

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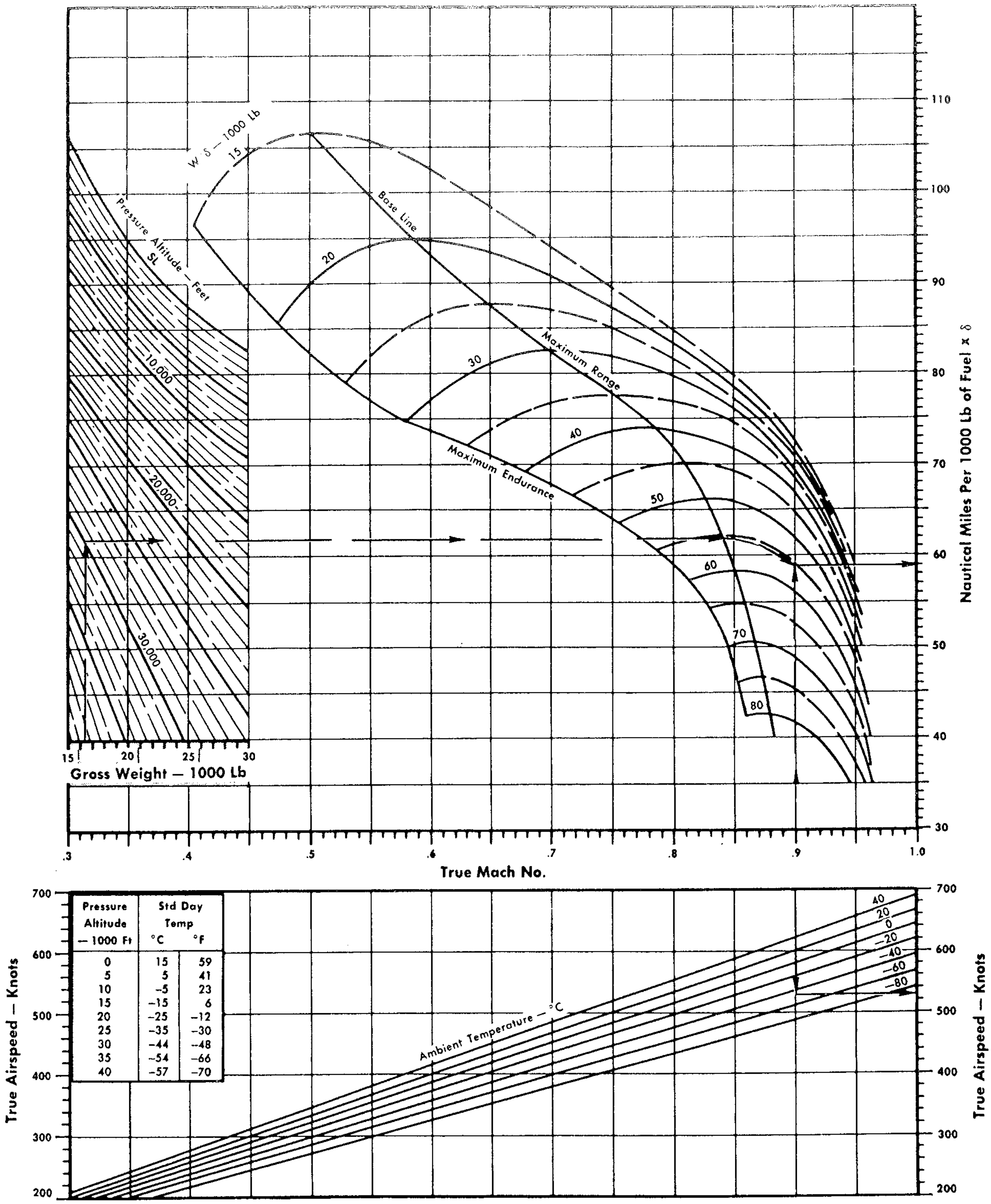
NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 0

Model: F-104G (MAP)
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	Std Day Temp °F
0	15	59
5	5	41
10	-5	23
15	-15	6
20	-25	-12
25	-35	-30
30	-44	-48
35	-54	-66
40	-57	-70

Figure A6-1 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 0

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

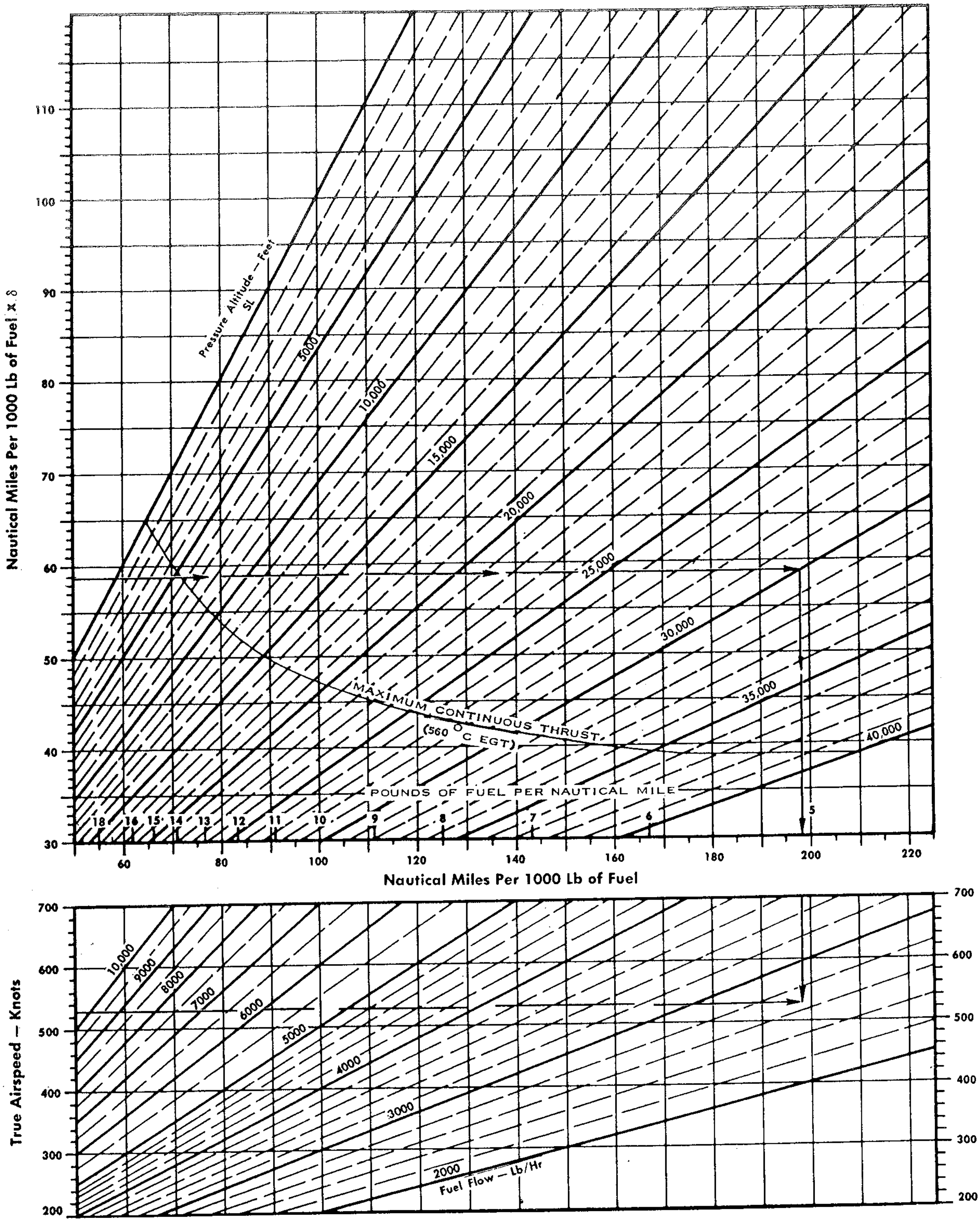
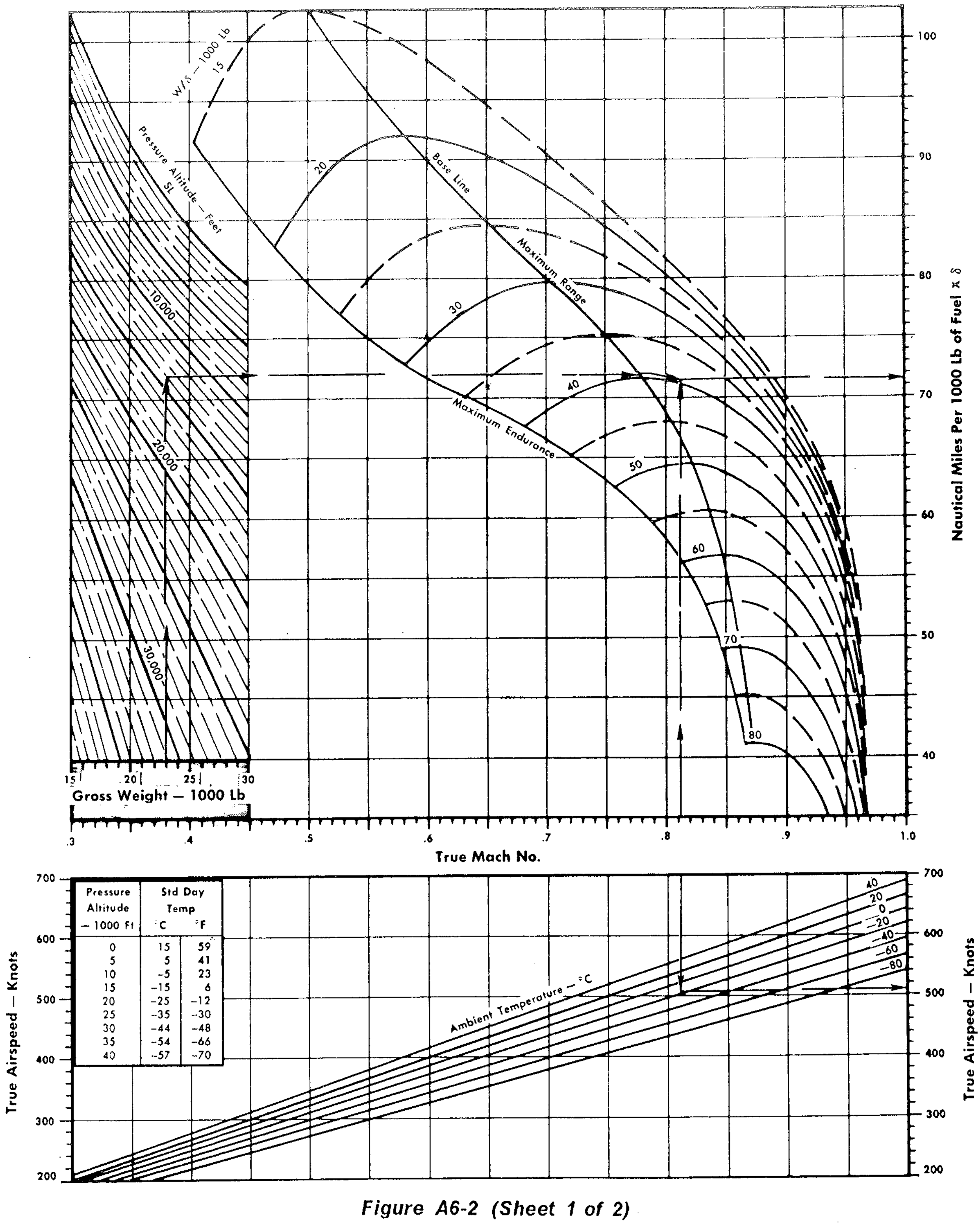


Figure A6-1 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 10

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	°F
0	15	59
5	5	41
10	-5	23
15	-15	6
20	-25	-12
25	-35	-30
30	-44	-48
35	-54	-66
40	-57	-70

Figure A6-2 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 10

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

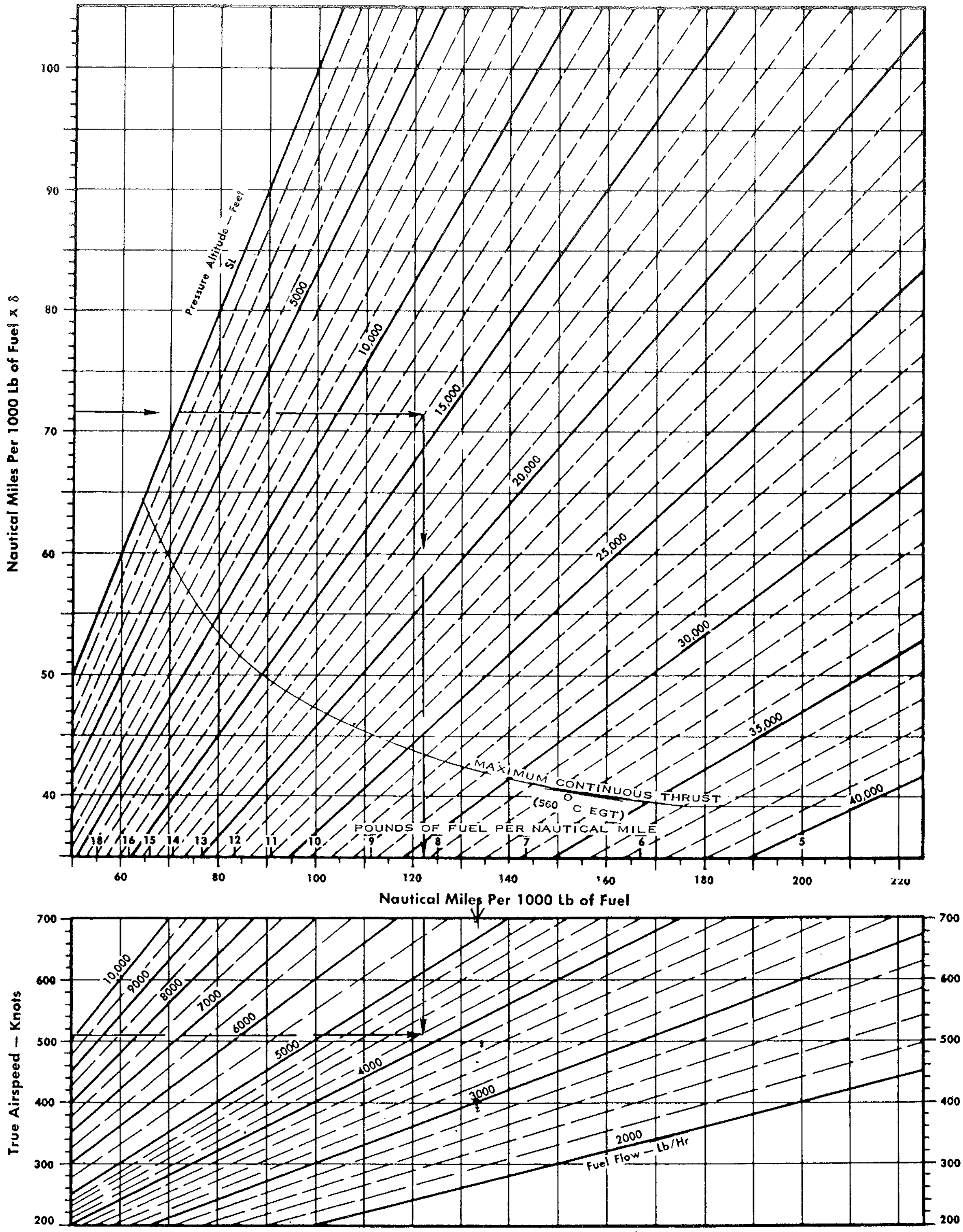
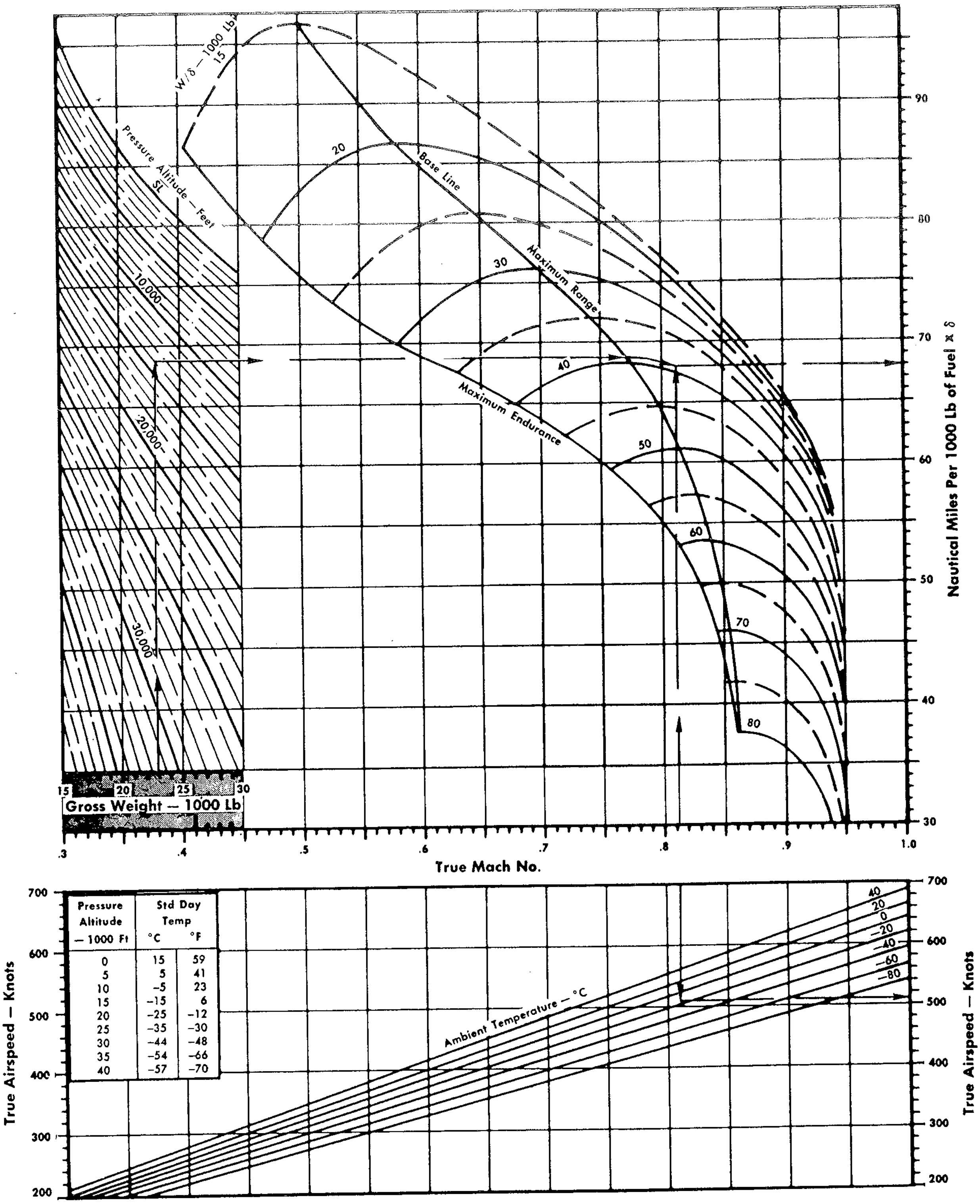


Figure A6-2 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 30

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	Std Day Temp °F
0	15	59
5	5	41
10	−5	23
15	−15	6
20	−25	−12
25	−35	−30
30	−44	−48
35	−54	−66
40	−57	−70

Figure A6-4 (Sheet 1 of 2)

DIVERSION RANGE SUMMARY TABLE

ZERO FUEL WEIGHT 14000 LB

Configuration Drag Index 0

Zero Wind Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

RANGE AND TIME REMAINING WITH 500 LB RESERVE AT SEA LEVEL

FUEL	INITIAL ALTITUDE 1000 FT	S.L.	5	10	15	20	25	30	35		PROCEDURE
750 LB	NAUT. MI.	27	28	30	33	36	40	44	45		CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	5.0	5.5	6.5	7.5	8.5	9.5	10.5	11.5		
	1000 FT	SL/15	5/20	10/25	15/30	25/35	25/35	30/35	35		OPTIMUM ALTITUDE
	NAUT. MI.	27	29	32	36	40	45	50	53		USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	5.0	5.0	6.0	6.5	7.0	8.0	9.0	9.5		
	NAUT. MI.	32	40	49	59	66	73	80	86		USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	5.0	6.0	7.0	8.0	9.0	10.0	11.0	11.5		
1000 LB	NAUT. MI.	53	58	64	71	79	90	99	104		CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	9.5	10.5	11.5	12.5	14.0	16.0	17.5	19.0		
	1000 FT	25/35	30/35	30/35	30/35	30/35	30/35	35	35		OPTIMUM ALTITUDE
	NAUT. MI.	64	69	78	86	94	100	106	112		USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	10.0	11.0	12.0	13.0	14.0	15.0	16.0	16.5		
	NAUT. MI.	92	99	109	118	126	133	139	145		USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	12.0	13.0	14.0	15.5	16.5	17.5	18.0	19.0		
1500 LB	NAUT. MI.	106	116	130	146	165	188	209	221		CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	19.0	20.0	22.0	23.5	25.5	28.0	31.0	33.0		
	1000 FT	35	35	35	35	35	35	35	35		OPTIMUM ALTITUDE
	NAUT. MI.	171	182	192	201	209	216	222	228		USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	23.5	25.0	26.5	27.0	28.0	29.0	30.0	30.5		
	NAUT. MI.	204	215	225	234	242	249	256	262		USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	25.5	27.0	28.5	29.5	30.5	31.5	32.0	33.0		
2000 LB	NAUT. MI.	158	174	195	220	250	284	316	333		CRUISE AT INITIAL ALTITUDE TO BASE ①
	MINUTES	28.0	29.0	31.5	34.0	36.5	40.0	44.0	46.5		
	1000 FT	35	35	35	35	35	35	35	35		OPTIMUM ALTITUDE
	NAUT. MI.	289	297	305	313	322	328	335	341		USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	MINUTES	38.0	39.0	39.5	41.0	41.5	42.5	43.5	44.5		
	NAUT. MI.	322	331	340	348	355	362	368	374		USE OPT. ALT. AND DESCEND ON COURSE ③
	MINUTES	39.5	41.0	42.0	43.0	44.0	45.0	45.5	46.5		
3000 LB	NAUT. MI.	259	288	324	367	418	473	524	548		CRUISE AT INITIAL ALTITUDE TO BASE ①
	HRS : MIN	:45.0	:47.0	:50.0	:53.5	:58.5	1:03.5	1:09.0	1:12.0		
	1000 FT	35	35	35	35	35	35	35	35		OPTIMUM ALTITUDE
	NAUT. MI.	502	512	521	529	537	543	550	556		USE OPTIMUM ALTITUDE UNTIL OVER BASE ②
	HRS : MIN	1:04.0	1:05.0	1:06.0	1:07.0	1:07.5	1:08.5	1:09.5	1:10.0		
	NAUT. MI.	536	546	555	562	570	577	583	589		USE OPT. ALT. AND DESCEND ON COURSE ③
	HRS : MIN	1:05.0	1:07.0	1:08.0	1:09.0	1:10.0	1:11.0	1:11.5	1:12.0		

CRUISE ALTITUDE		S.L.	5	10	15	20	25	30	35	
TRUE MACH NO.		.50	.56	.61	.67	.73	.78	.82	.85	
START 300 KIAS IDLE DESCENT WITH: ④	MI. REM.	5	10	15	21	27	34	41		
	MIN. REM.	1.0	2.0	2.5	3.5	4.5	5.5	6.0		
	FUEL REM.	515	525	535	545	550	555	560		

WITH OVER 3000 LB OF FUEL REMAINING - USE:

GR WEIGHT	ALT	MACH
17-19000 LB	30000 FT	0.85
19-21000 LB	30000 FT	0.86

① FUEL INCLUDED FOR DESCENT AT DESTINATION WITHOUT DISTANCE CREDIT, DIVE FLAPS RETRACTED.
② TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. NO DISTANCE CREDIT FOR DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS EXTENDED.
③ TIME AND FUEL INCLUDED FOR CLIMB TO OPTIMUM ALTITUDE AND DESCENT AT DESTINATION. RANGE INCLUDES DISTANCE FOR ON-COURSE DESCENT TO SEA LEVEL DESTINATION, DIVE FLAPS RETRACTED.
④ DESCENT DATA TABULATED FOR DIVE FLAPS RETRACTED. WITH DIVE FLAPS EXTENDED, USE HALF OF MILES AND MINUTES VALUES.
5. CLIMB AT 0.90 TRUE MACH NO. WITH MILITARY THRUST.
6. SCHEDULES PROVIDE FOR ARRIVAL AT SEA LEVEL WITH 500 LB APPROACH AND LANDING RESERVE.
7. 5% OPERATIONAL FUEL FLOW TOLERANCE INCLUDED.
8. DATA BASED ON 14,000 LB ZERO FUEL WT. DECREASE OPTIMUM ALTITUDE 1,500 FT. AND RANGE 5 PER CENT PER 1000 LB INCREASE IN ZERO FUEL WEIGHT.

Figure A4-3

MAXIMUM RANGE

CRUISE - CLIMB

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

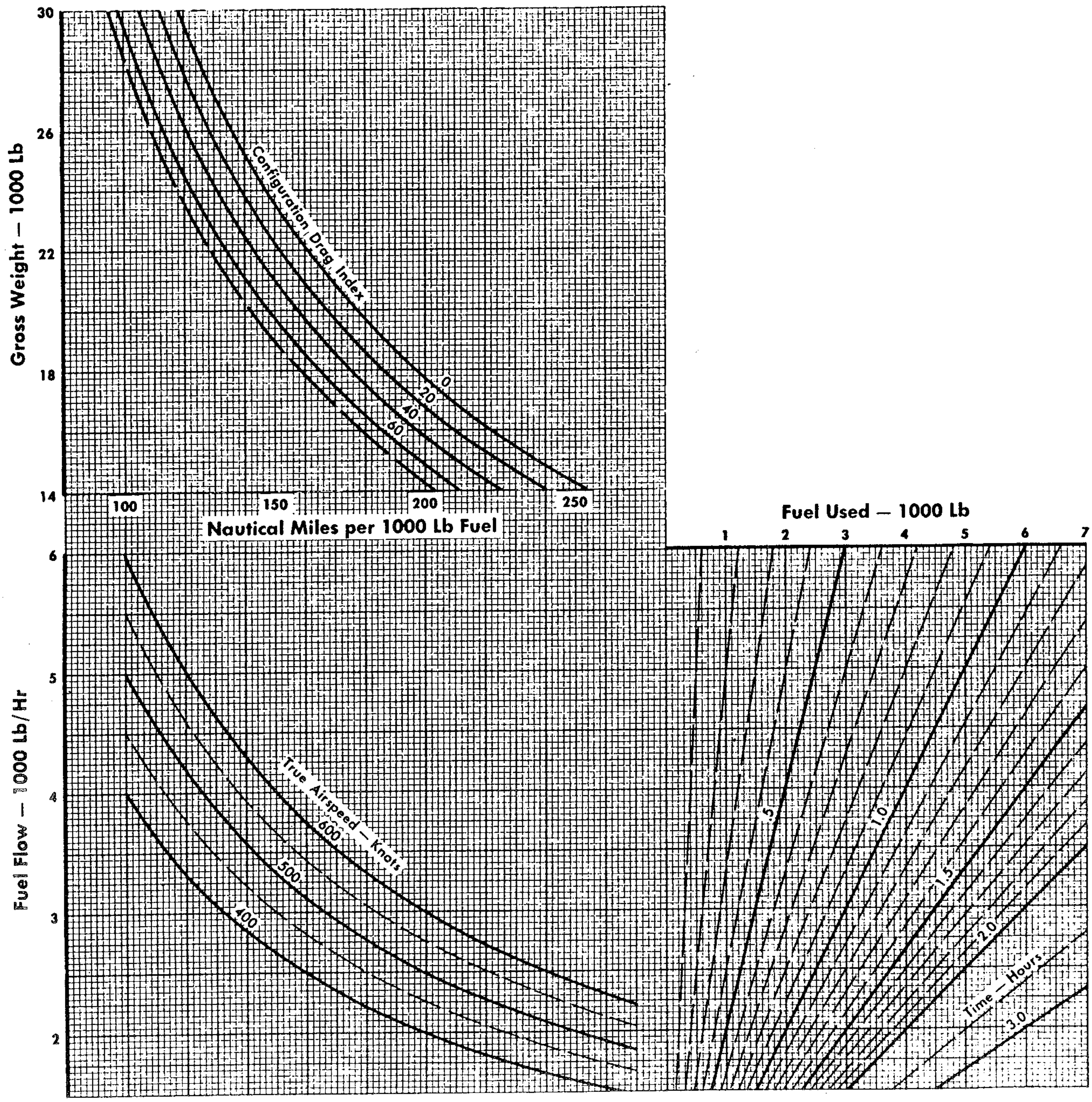


Figure A4-2 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 30

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

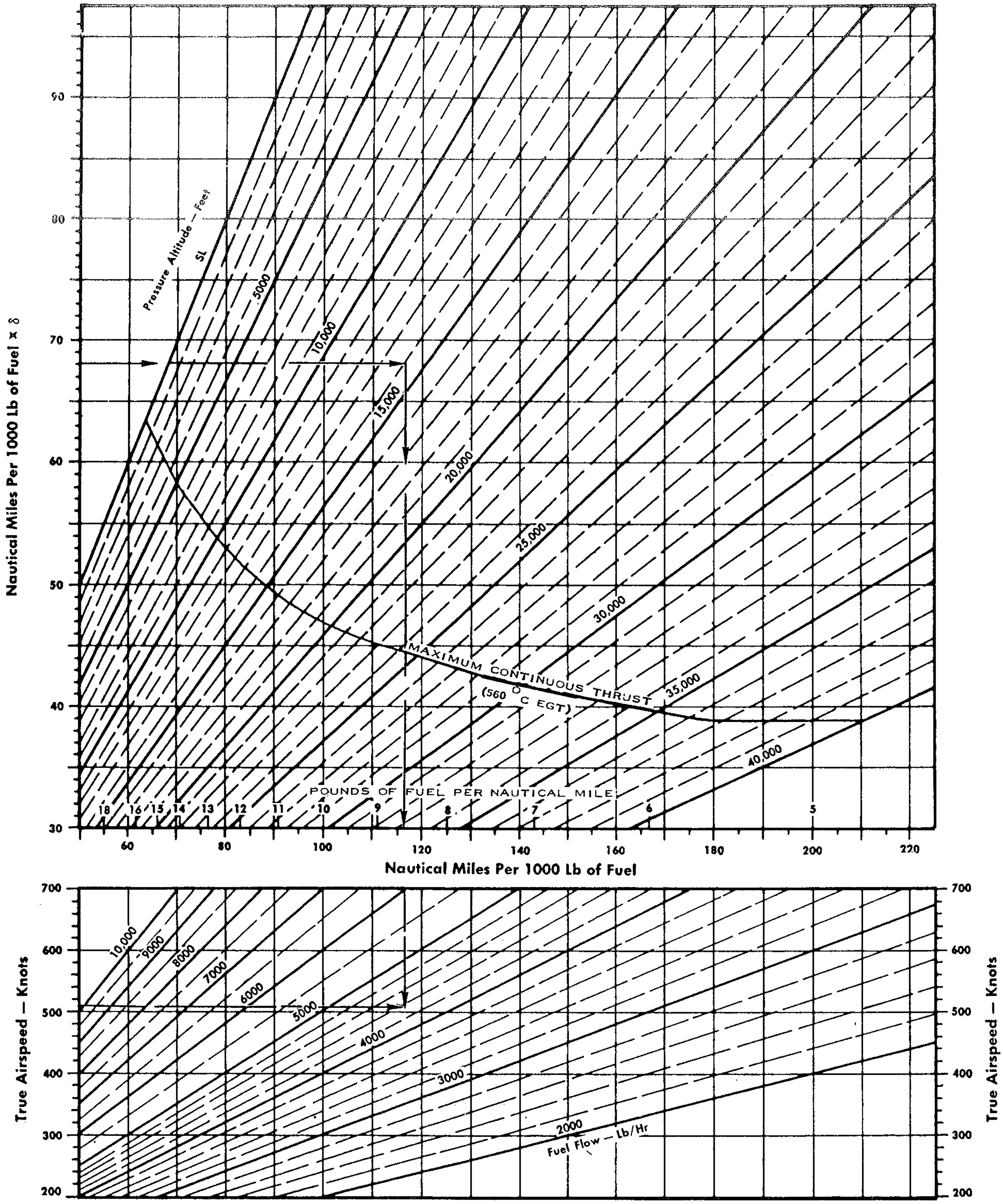
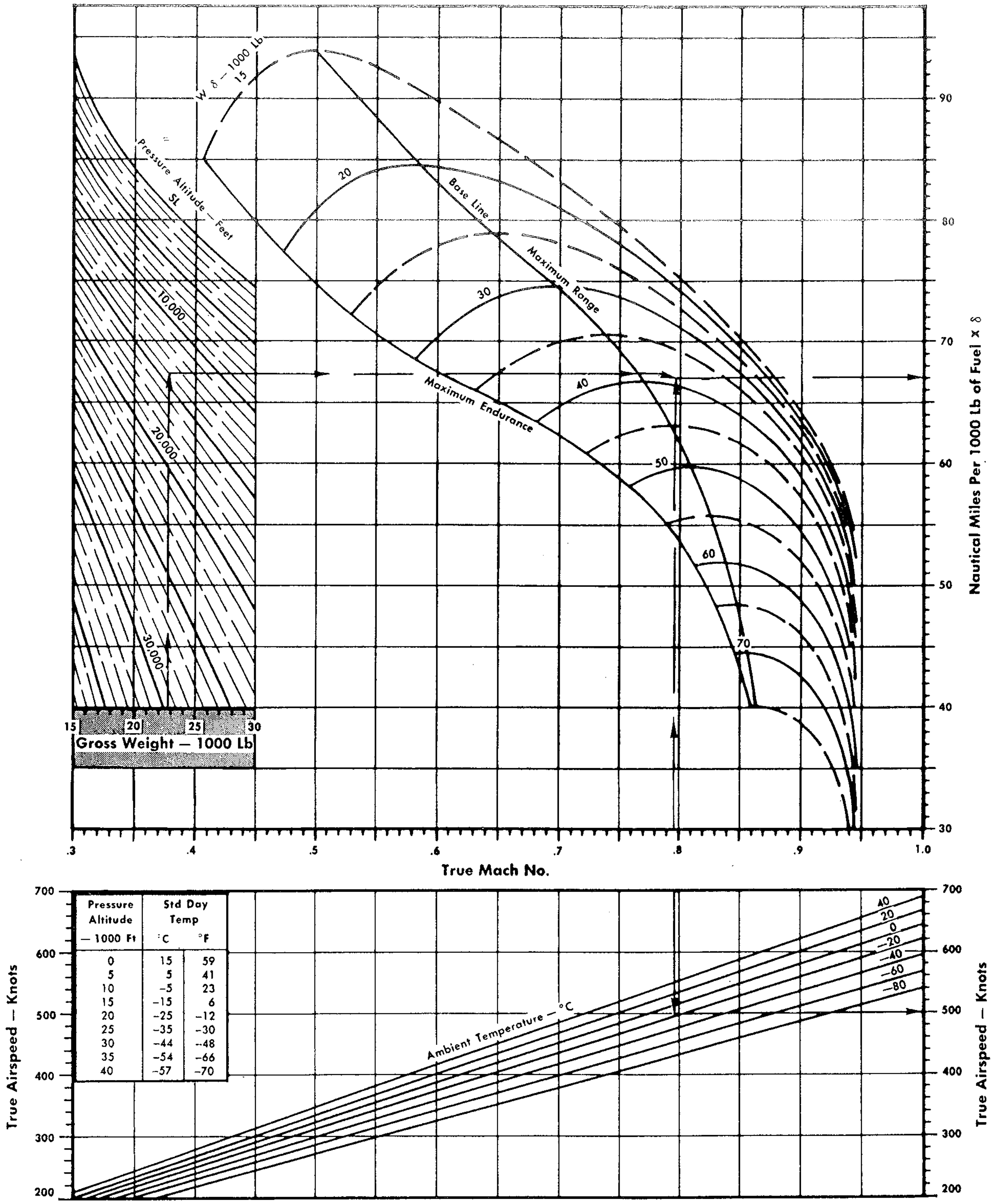


Figure A6-4 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 40

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	Std Day Temp °F
0	15	59
5	5	41
10	-5	23
15	-15	6
20	-25	-12
25	-35	-30
30	-44	-48
35	-54	-66
40	-57	-70

Figure A6-5 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 40

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

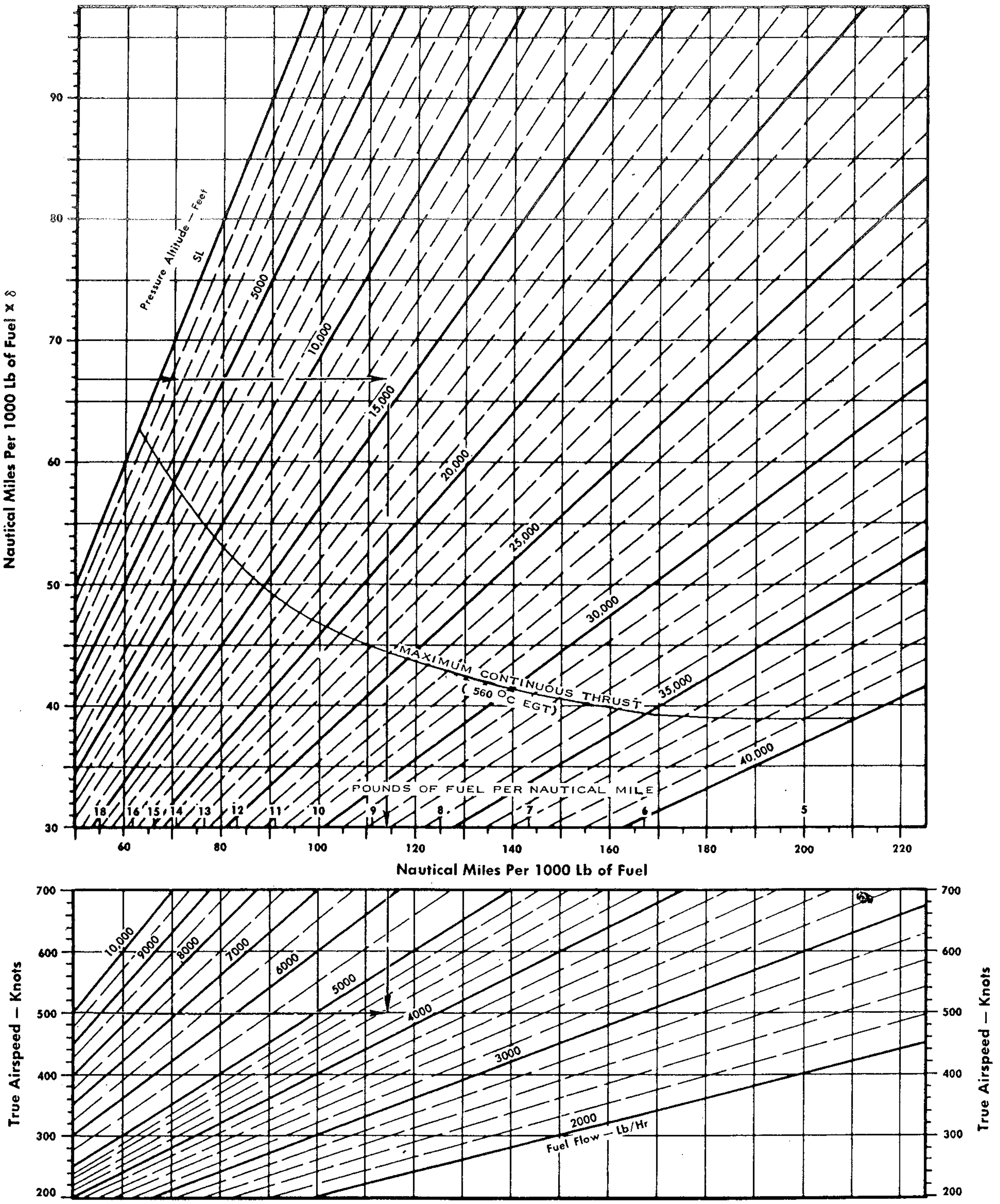
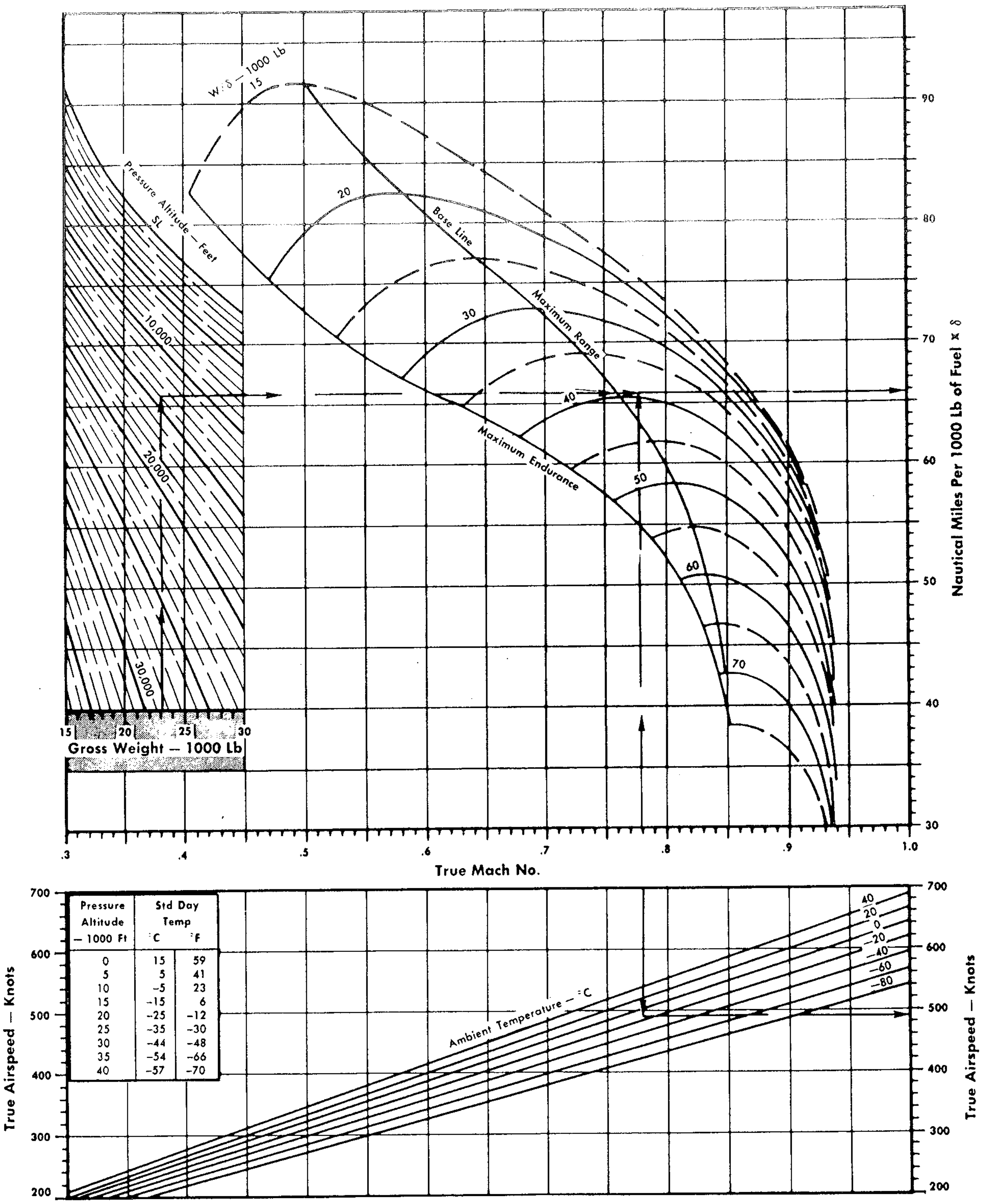


Figure A6-5 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 50

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	°F
0	15	59
5	5	41
10	−5	23
15	−15	6
20	−25	−12
25	−35	−30
30	−44	−48
35	−54	−66
40	−57	−70

Figure A6-6 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 50

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

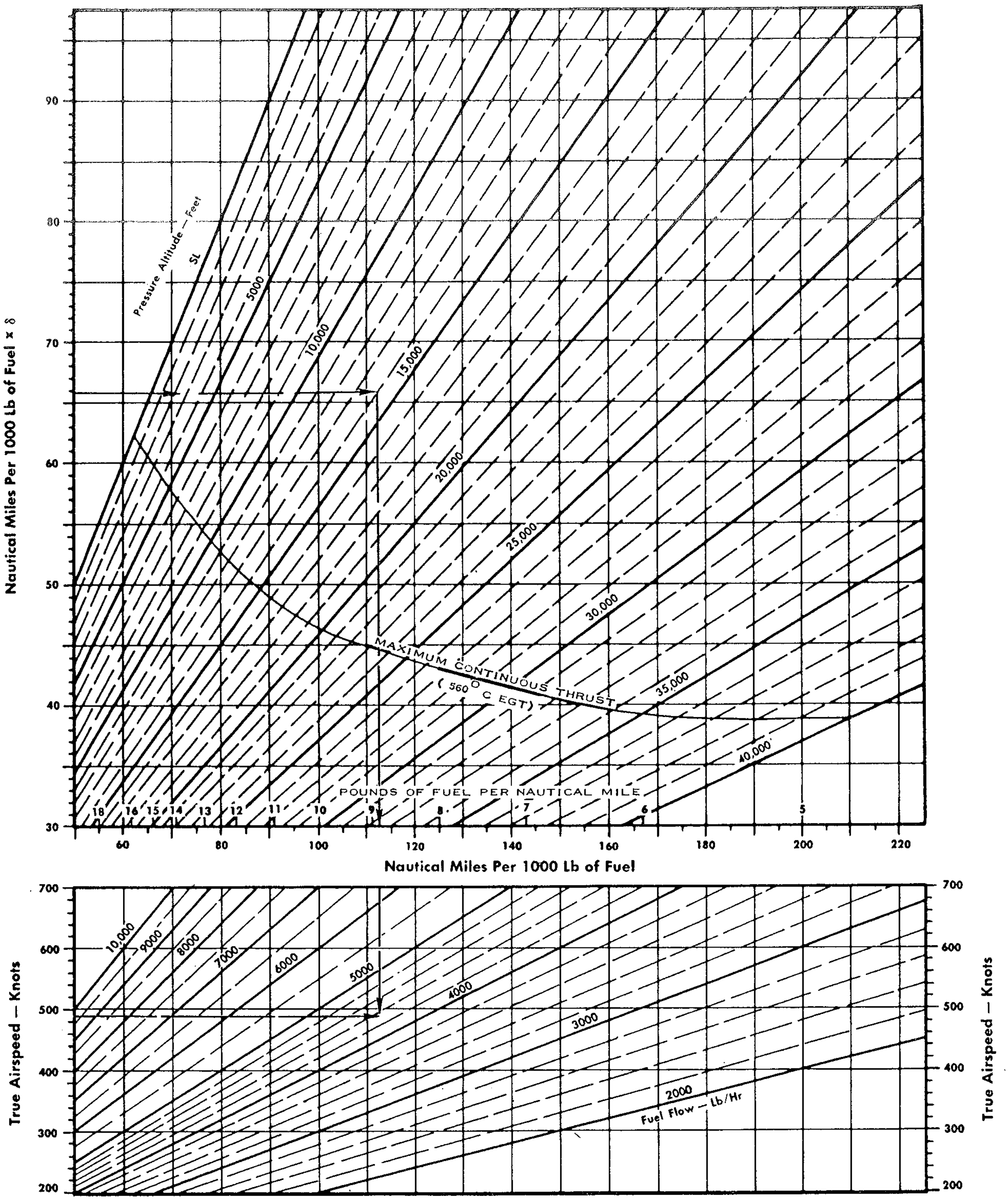
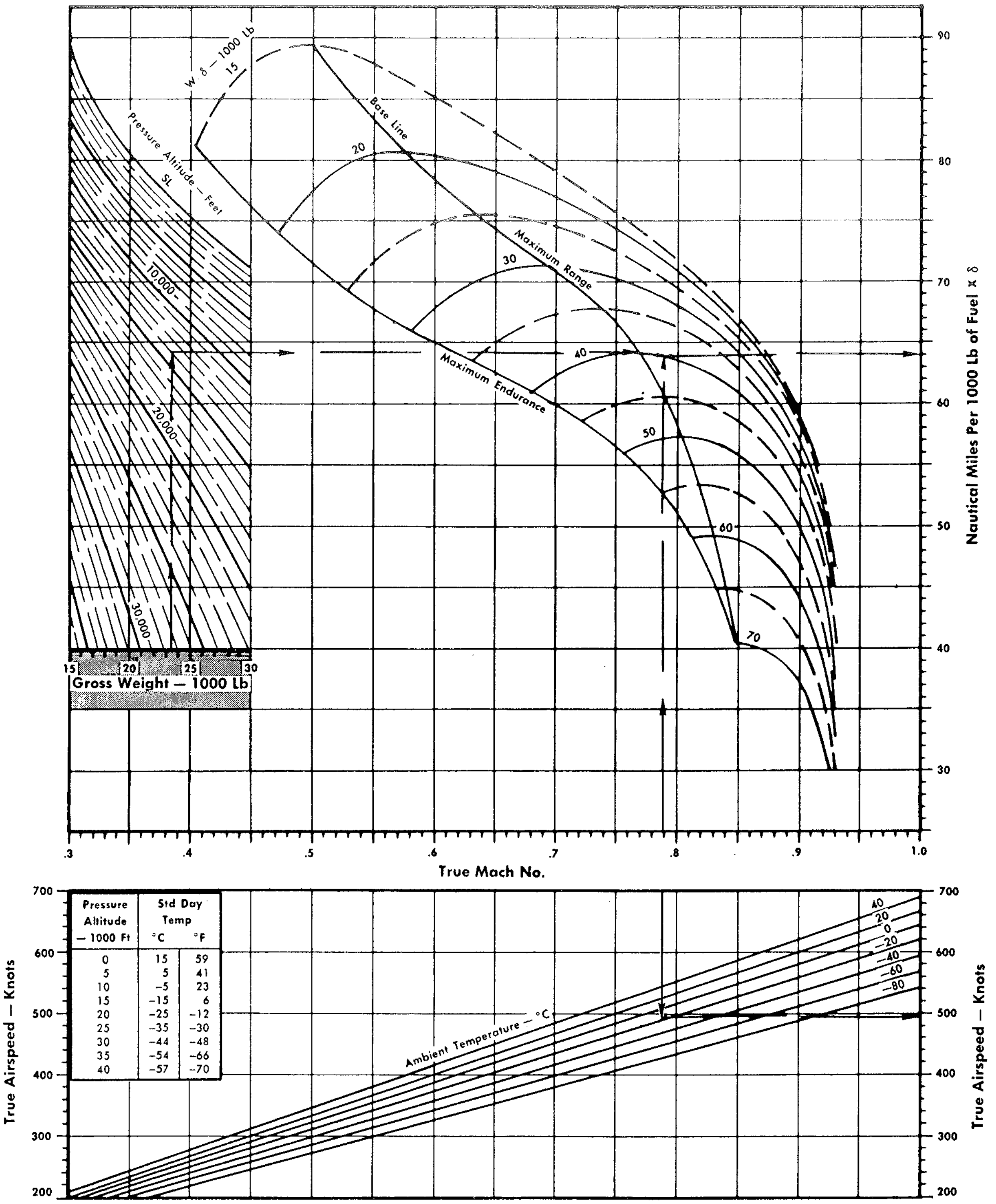


Figure A6-6 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 60

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	Std Day Temp °F
0	15	59
5	5	41
10	-5	23
15	-15	6
20	-25	-12
25	-35	-30
30	-44	-48
35	-54	-66
40	-57	-70

Figure A6-7 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 60

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

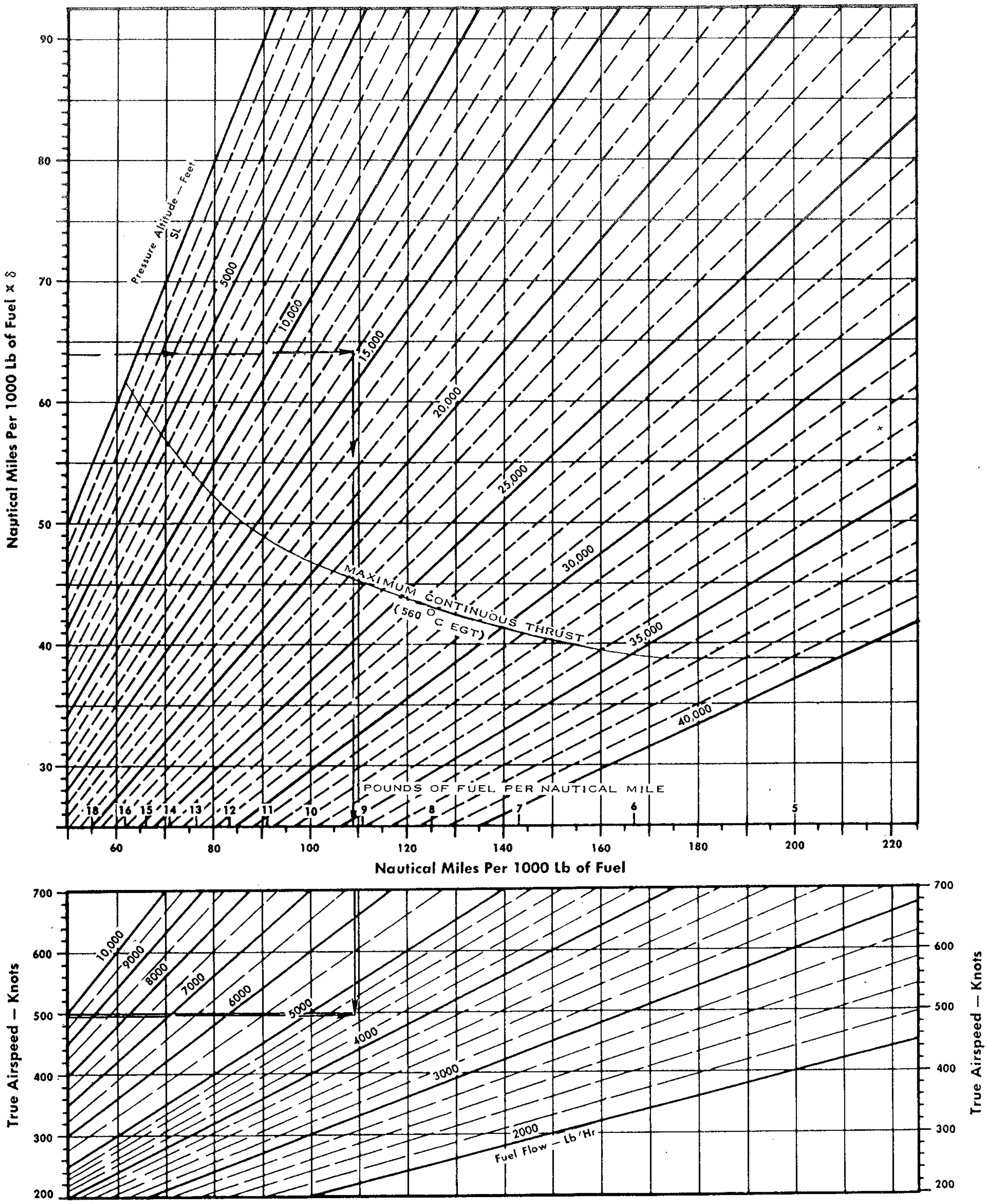
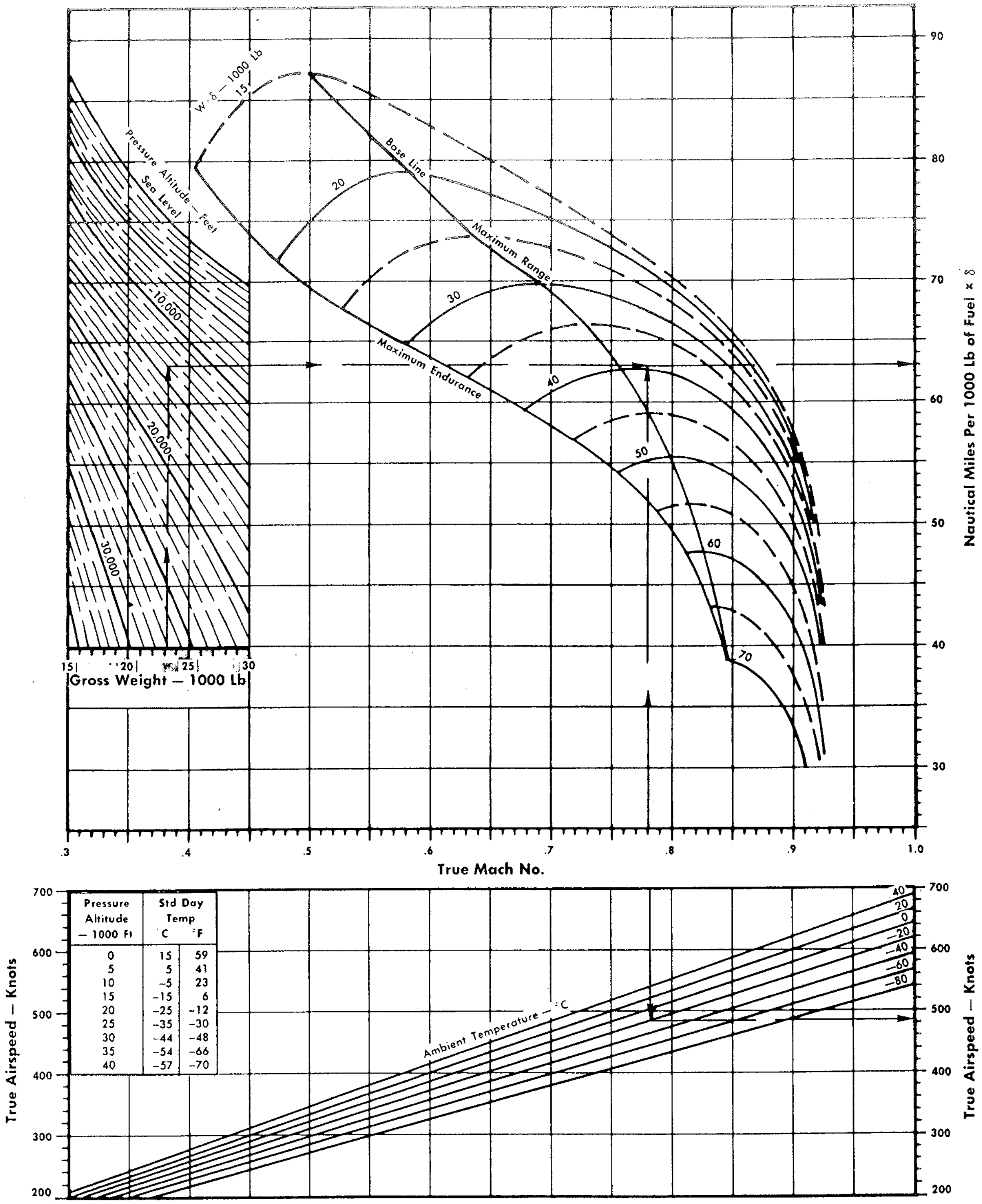


Figure A6-7 (Sheet 2 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 70

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST



Pressure Altitude — 1000 Ft	Std Day Temp °C	Std Day Temp °F
0	15	59
5	5	41
10	-5	23
15	-15	6
20	-25	-12
25	-35	-30
30	-44	-48
35	-54	-66
40	-57	-70

Figure A6-8 (Sheet 1 of 2)

NAUTICAL MILES PER 1000 LB OF FUEL

Configuration Drag Index 70

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

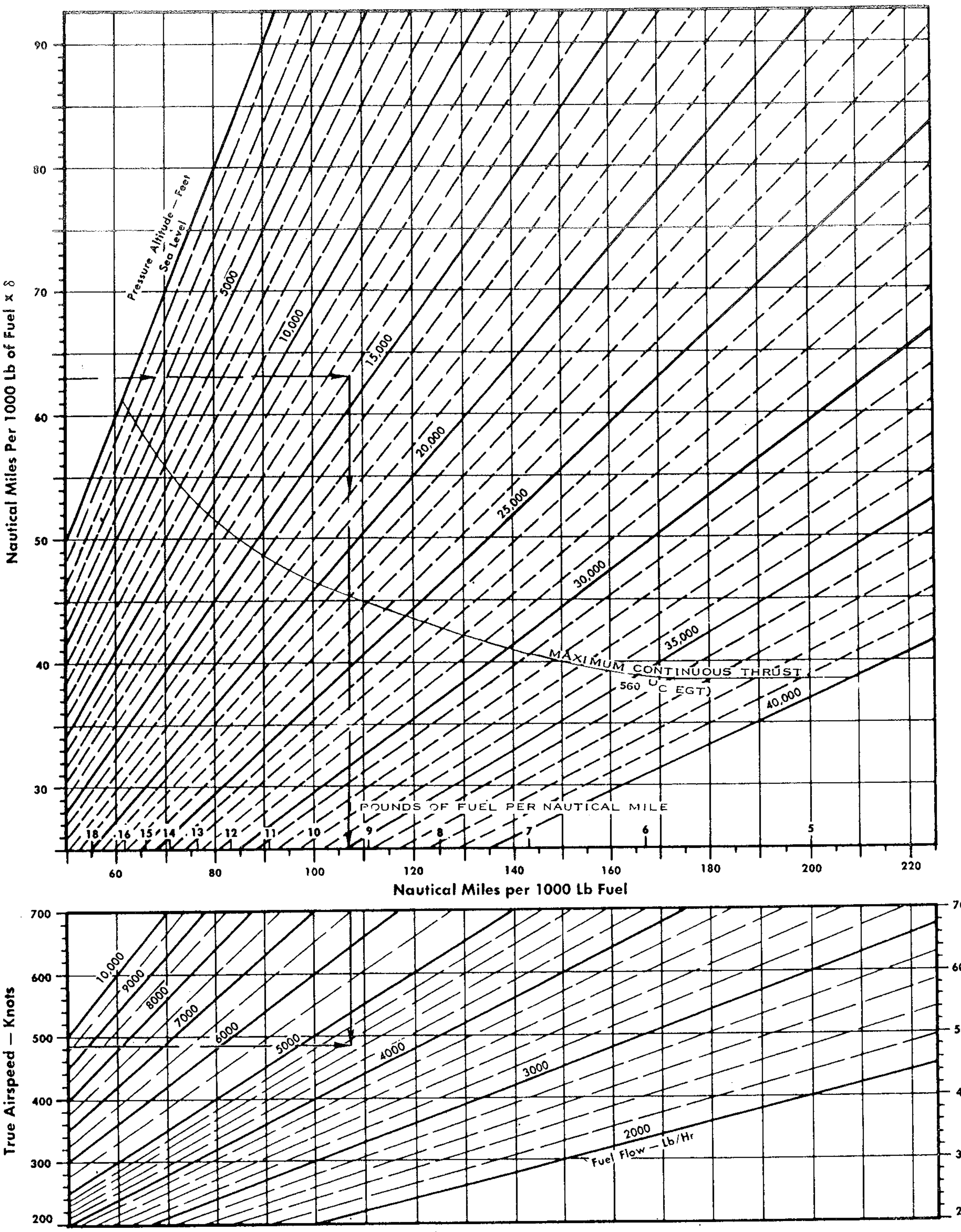


Figure A6-8 (Sheet 2 of 2)

PART 7

DESCENT

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

DESCENT PERFORMANCE

Time, distance and fuel required to descent from altitude using idle throttle setting or 85% RPM are presented in this part. Conditions include landing gear up, speed brakes extended and retracted, wing flaps UP and TAKEOFF.

Read the descent charts directly if sea level is the terminal altitude; otherwise, read the charts on an incremental basis between the initial and final altitudes. (Refer to Section VII "All Weather Operation" for jet penetration descent procedures).

SAMPLE PROBLEM

Find time, fuel used, and distance traveled during an idle decent from 29000 feet to sea level. Speed brakes are retracted.

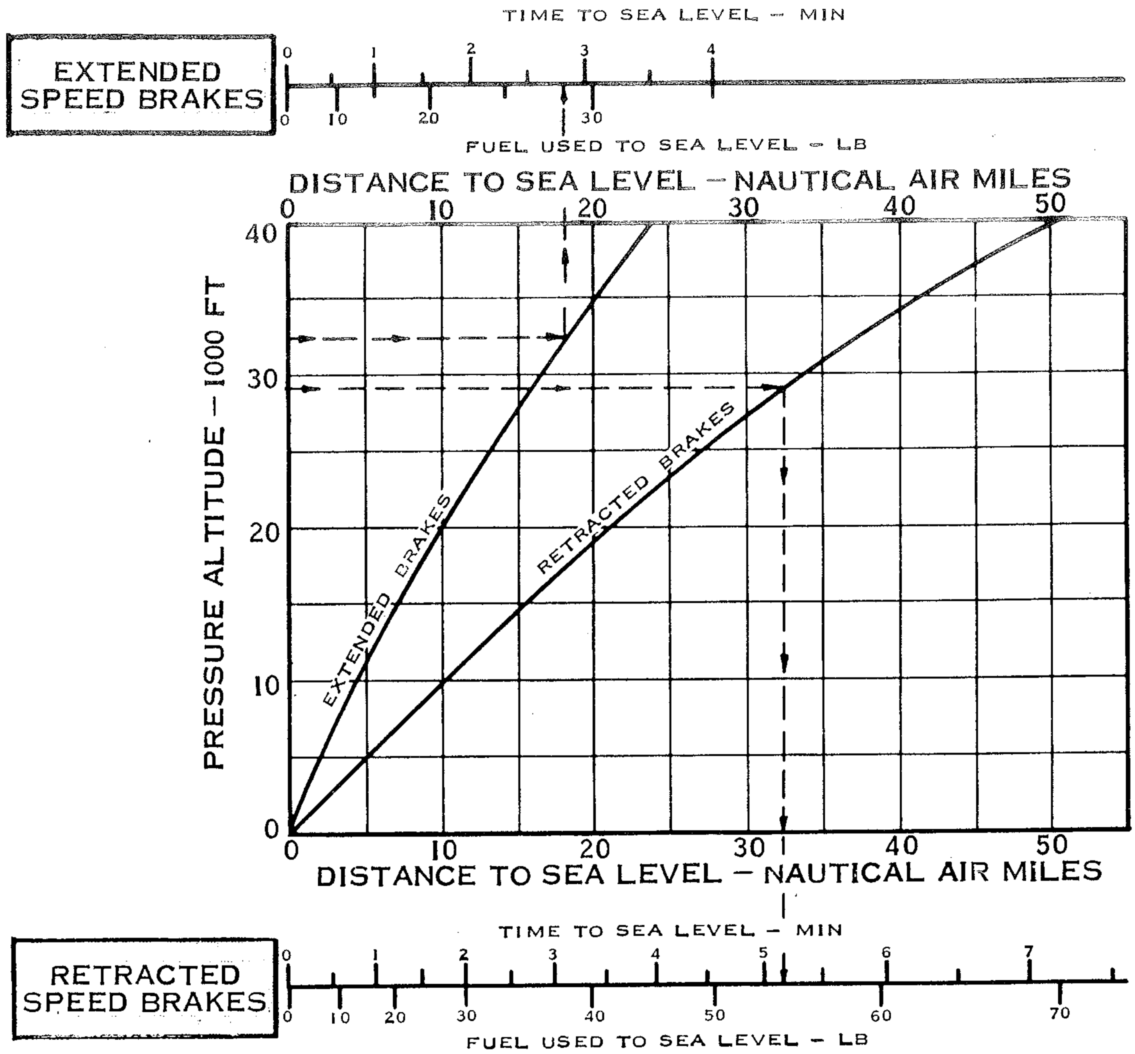
Enter Figure A7-1 at 29000 feet and read 32 nautical miles traveled, 5.2 minutes elapsed time, and 54 pounds of fuel used.

IDLE DESCENT PERFORMANCE — 300 KNOTS IAS

FLAPS AND GEAR UP

Model: F-104G
Date: 15 August 1962
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



REMARKS:

1. SET IDLE THRUST
2. HOLD 300 KNOTS IAS DURING DESCENTS FROM ALTITUDES ABOVE 40,000 FT
3. AVERAGE RATES OF DESCENT:
   SPEED BRAKES RETRACTED – 5,600 FT/MIN
   SPEED BRAKES EXTENDED – 11,150 FT/MIN
4. REFER TO SECTION IX FOR 85% RPM JET PENETRATION DESCENT PERFORMANCE.
5. DATA BASED ON 14,400 TO 16,000 POUND GROSS WEIGHTS.

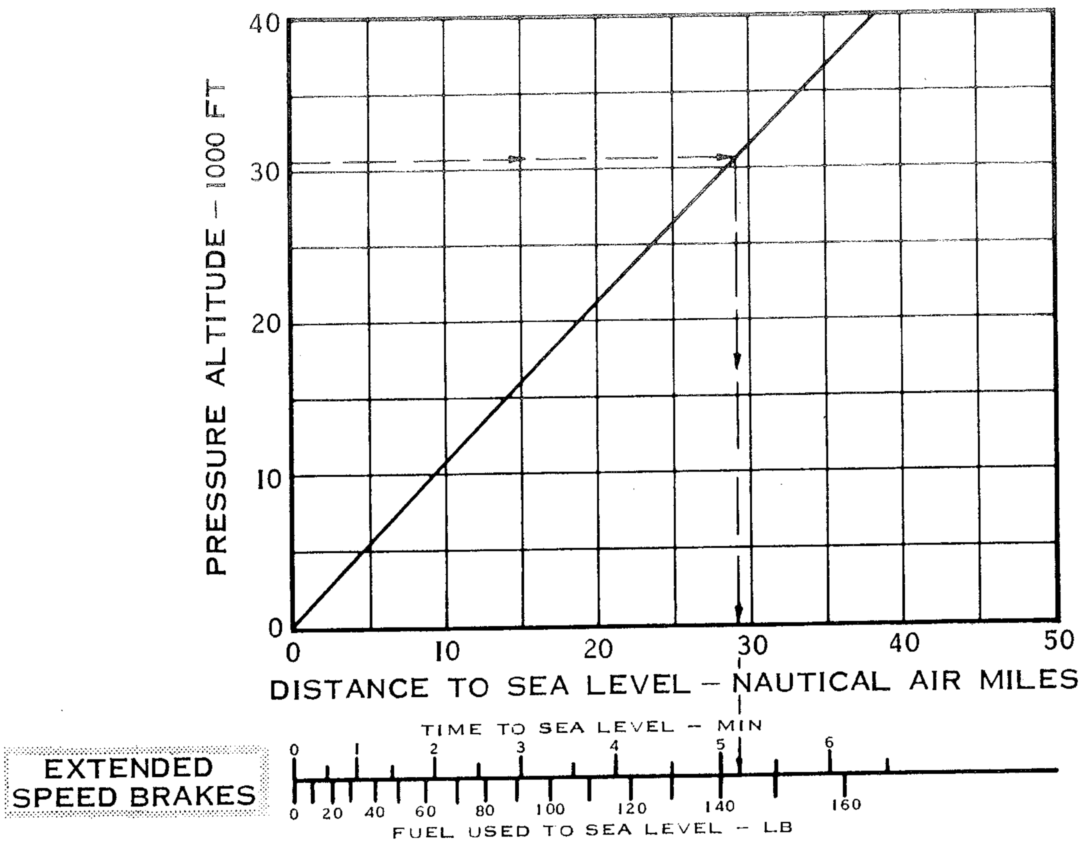
Figure A7-1

85% RPM DESCENT PERFORMANCE — 275 KNOTS IAS

TAKEOFF FLAPS EXTENDED AND GEAR UP

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



REMARKS:

1. SET 85% ENGINE RPM.
2. HOLD 275 KNOTS IAS DURING DESCENT.
3. AVERAGE RATE OF DESCENT – 6500 FT/MIN
4. DATA BASED ON 16,000 TO 25,000 POUND GROSS WEIGHTS.

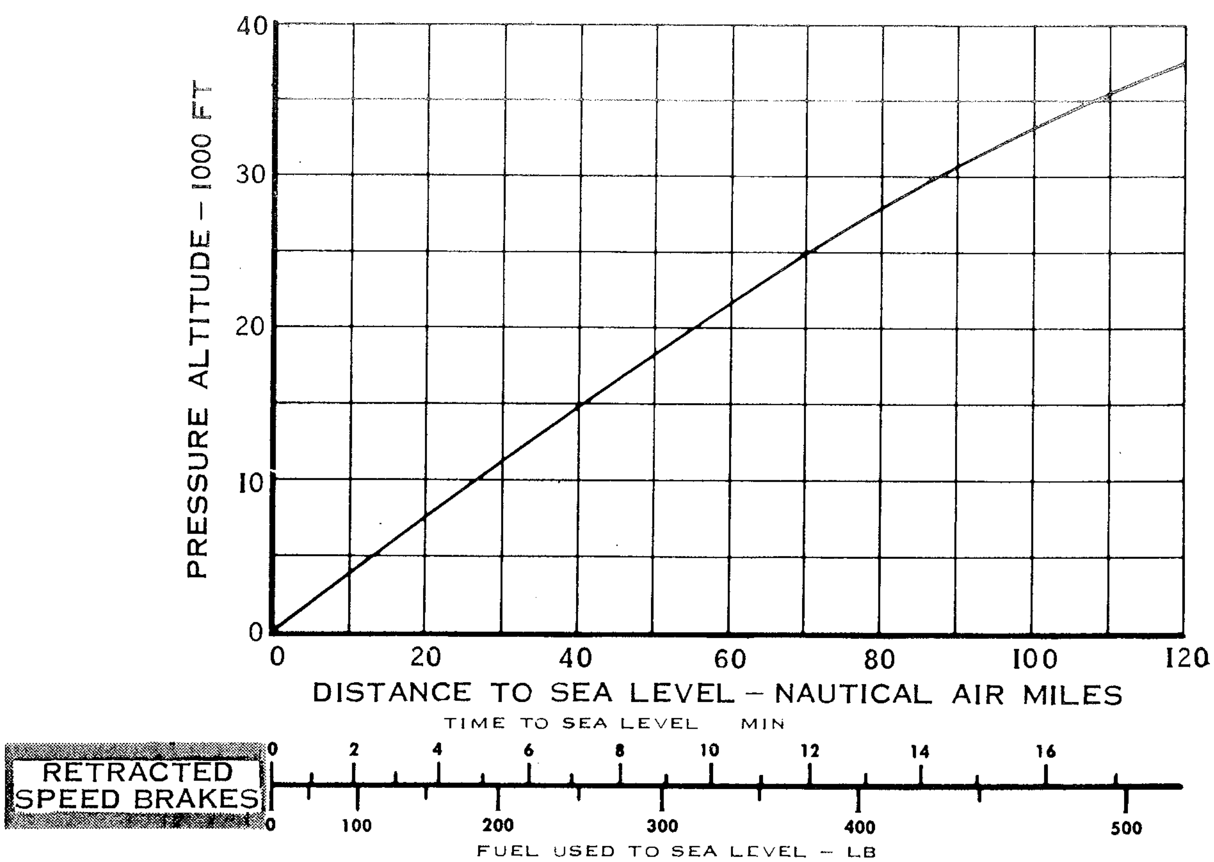
Figure A7-2

85% RPM DESCENT PERFORMANCE — 300 KNOTS IAS

FLAPS AND GEAR UP

Model: F-104G
Date: 1 February 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



REMARKS:

1. SET 85% ENGINE RPM.
2. HOLD 300 KNOTS IAS DURING DESCENT
3. AVERAGE RATES OF DESCENT – 2000 FT/MIN.
4. DATA BASED ON 15,300 TO 16,000 POUND GROSS WEIGHTS.

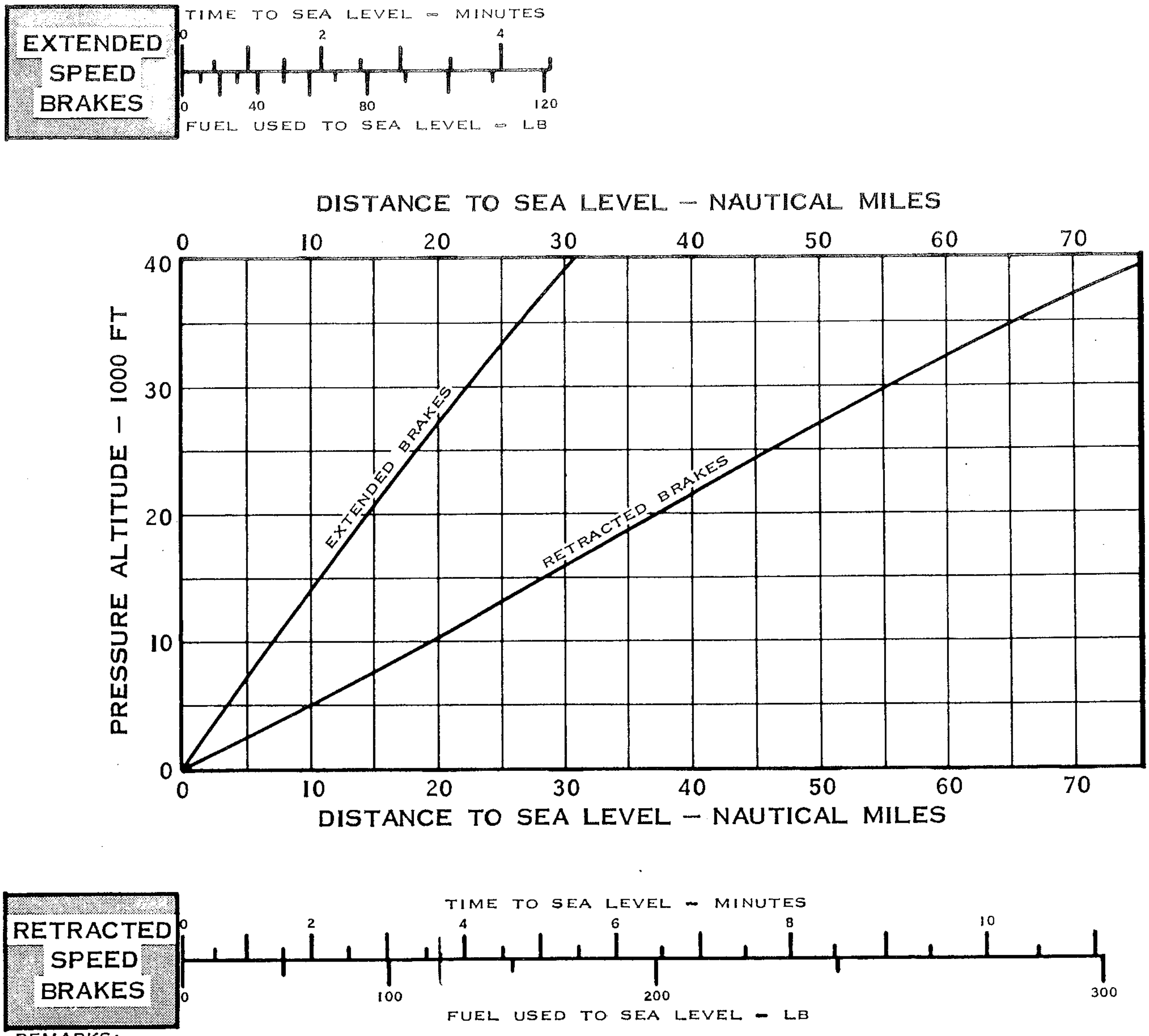
Figure A7-3

85% RPM DESCENT PERFORMANCE — 300 KNOTS IAS

TAKEOFF FLAPS EXTENDED AND GEAR UP

Model: F-104G
Date: 1 October 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



REMARKS:

1. SET 85% ENGINE RPM
2. HOLD 300 KNOTS IAS DURING DESCENT
3. AVERAGE RATES OF DESCENT:
   SPEED BRAKES RETRACTED — 3500 FT/MIN
   SPEED BRAKES EXTENDED — 8500 FT/MIN

Figure A7-4

PART 8

LANDING

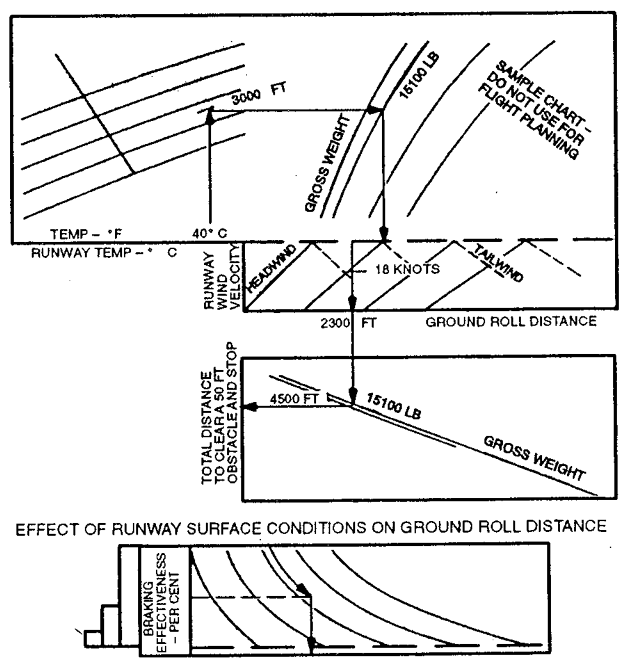
TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

NORMAL PERFORMANCE WITH DRAG CHUTE

Figure A8-3 presents landing ground roll distances and distances required to clear a 50-foot obstacle, land, and stop when the 18-foot drag chute is deployed during ground roll to assist wheel braking. Values shown are directly applicable to operation with landing flaps, hard dry runway surface conditions, and runway wind component speeds from 20-knot tailwinds to 40-knot head-winds. A correction grid is provided for obtaining the ground roll distance under various runway surface conditions. Distances reflect results of flight tests with normal speed schedules for approach and touchdown, and hard wheel braking without skidding for the subsequent stop. Normal performance assumes that an approach flight path slope of approximately 2½° is maintained by use of thrust until the 50-foot obstacle point is cleared, and that thrust is reduced smoothly during the flare to just above Idle at the touchdown point. It also assumes a smooth rotation from touchdown attitude to nosewheel contact, prompt activation of the drag chute with hard anti-skid wheel braking during the transition period, and continuous hard wheel braking with full drag chute effectiveness during the remainder of the stop. Use of the drag chute allows a high rate of deceleration during the early portion of the ground roll: any unnecessary delay in its deployment will add considerably to the stopping distance. However, premature chute deployment before nosewheel contact may result in high structural loads on the aircraft. Use of moderate or light wheel braking will extend the landing roll beyond charted values, but will result in less wear on the brakes and tires. The normal braking force which is used operationally should be judged in terms of the difference between charted distance (for which hard braking was used) and landing roll distances actually available.

LANDING DISTANCE WITH DRAG CHUTE
LANDING FLAPS



FA0329

SAMPLE PROBLEM

Determine the landing ground roll and total distance to clear a 50-foot obstacle and stop for the following conditions:

- Boundary Layer Control – On
- Gross Weight – 15100 lb
- Runway Air Temperature – 40° C
- Field Pressure Altitude – 3000 ft
- Runway Wind Component – 18-knot headwind
- Runway Slope – 1% downhill
- Drag Chute – Deployed
- Runway Surface – Dry and hard

NOTE

> The gross weight of 15100 lb used in this sample problem does not represent any particular configuration. However, it is the weight at which landing speeds must be increased 5 knots for each 1000 lb gross weight above 15100 lb. Refer to Figure A8-2.

a. From Figure A8-3, the ground roll distance from touchdown to stop for a level runway with an 18-knot headwind is 2300 ft.

b. The ground roll distance with a 1% downhill slope is determined by increasing the zero slope value by 2%.
2300 ft + (.02 x 2300 ft) = 2345 ft.

c. For a level runway, total distance to clear a 50-foot obstacle and stop is 4500 ft.

d. Total distance with a 1% downhill slope is determined by increasing the zero-slope total distance by 4%.
4500 ft + (.04 x 4500 ft) = 4680 ft.

e. From Figure A8-2, normal touchdown speed is 150 to 155 KIAS.

f. From Figure A8-2, final approach speed is 170 KIAS.

NORMAL PERFORMANCE WITHOUT DRAG CHUTE

Figure A8-4 shows landing distances to be expected when the drag chute is not deployed and hard wheel braking is used for the stop. Performance is based on the same speed schedules as for normal operation with the drag chute.
As discussed above, if hard braking is not used operationally, the pilot should ensure that landing field lengths in excess of those shown in Figure A8-4 are available.

EFFECT OF RUNWAY SURFACE CONDITIONS

A Runway Condition Reading (RCR) grid is included on each landing distance chart for determining the effect of runway surface conditions on landing ground roll distances. To use the grids, enter on the line labeled "Dry Hard Runway" with the dry hard runway ground roll distance.
Follow the guide lines to a point opposite the RCR given by the base weather station and read the ground roll distance for that condition directly below. If no RCR is available use a value of 12 for wet runways and 5 for icy runways. For an ICAO report of "Good", use and RCR value of 23; for "Medium" use 12; for "Poor" use 5.
General rain, snow, and ice criteria are also provided. Careful judgement shall be used in applying the information shown by the general surface conditions. A light rain on a clean, well-drained runway will probably result in very little reduction in braking effectiveness.
On the other hand, a dirty or dusty runway, under the same rain conditions will probably present nearly the same surface condition as an icy runway. Similarly, an accumulation of water on the runway will produce a tendency for the tires to ride on a thin film of water. This also produces much the same result as an icy runway.
When operating from snow-covered runways, the braking effectiveness shown may only be realized when the runway surface under the snow is dry. Slush or ice under the snow will further reduce the effective braking force. The grids also indicate the effect of brake failure on ground roll distance. If the drag chute is not deployed and brake failure occurs, the idle thrust of the engine is enough to keep the aircraft moving.

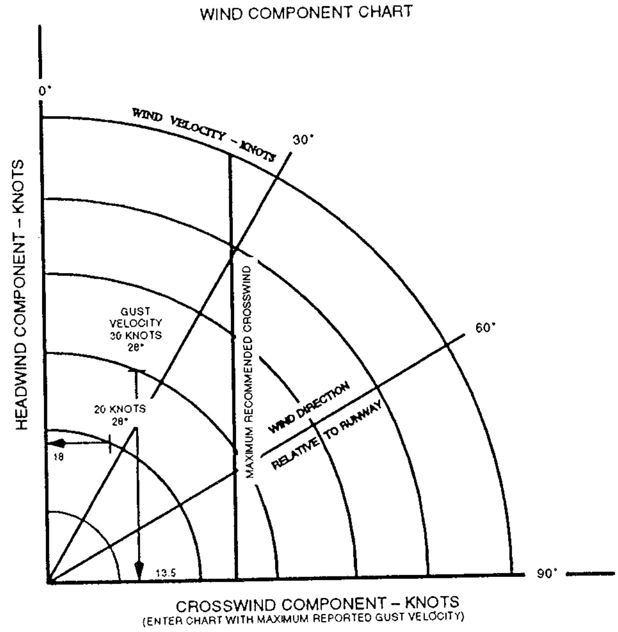
EFFECT OF RUNWAY SLOPE

Landing ground roll distances are increased 2% per 1% downhill runway slope and decreased 2% per 1% uphill slope, operating either with or without drag chute. Total landing distances are increased 4% per 1% downhill slope, decreased 3% per 1% uphill slope.

EFFECT OF WIND – CROSSWIND CONVERSION

Correction grids included on the landing distance charts show the effect of wind components along the runway. Actual surface wind velocities should be used for computing headwind component. Use maximum gust speeds for crosswind components. Wind speeds indicated by an anemometer located more than 50 feet above the surface may be corrected for wind gradient by multiplying reported winds by a factor of 7/10.
Landing with crosswind components at the runway in excess of 25 knots are not recommended. It will be difficult to maintain alignment with the runway at normal touchdown speeds without slipping the aircraft, and such components are greater than recommended for grad chute deployment.
Figure A8-1 illustrates a means for converting wind velocity into runway and crosswind component speeds.

WIND COMPONENT CHART



SAMPLE PROBLEM

The reported wind velocity is 20 knots with gusts to 30, blowing 27° relative to the service runway (wind velocity obtained from an anemometer located adjacent to the runway).

a. From Figure A8-1 determine the crosswind component (using the maximum gust velocity) to be 13.5 knots (this is less than the maximum recommended component of 25 knots).

b. If landings are to be made into the wind, the headwind component (using the reported wind of 20 knots) for the determination of landing distances is 18 knots.

MAXIMUM PERFORMANCE LANDINGS

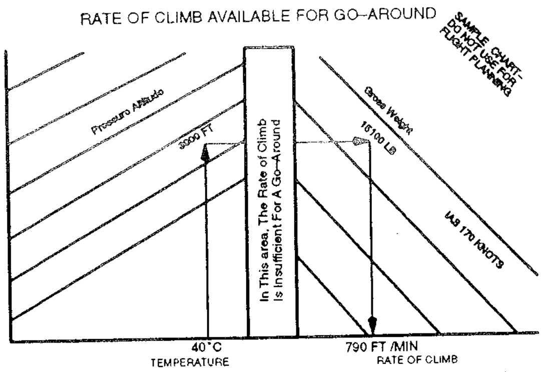
The minimum recommended touchdown speed for maximum performance landings is 5 knots less than that used for normal operation. This will result in landing ground roll distances 5% less than shown for hard braking on Figure A8-3 and Figure A8-4.

LANDING WITH TAKEOFF FLAPS

Landing with TAKEOFF flaps may be desirable, because of external store – fuel load combinations, to maintain go-around capability under heavy weight and high temperature conditions, or for various other reasons. Landing ground roll distances to be expected with hard wheel braking and dry runway conditions may be obtained from Figure A8-5 or Figure A8-6 for operation with or without the drag chute. These values apply when touchdown speeds shown on Figure A8-2 are used. Distances for the descent from 50 feet to the touchdown point will remain substantially the same if final approach slope somewhat greater than 2½% is used to compensate for the slight additional distance required to flare at the higher approach speeds. The need for such adjustment should be judged in terms of terrain clearance requirements.

RATE OF CLIMB AVAILABLE FOR GO-AROUND

Rates of climb available at final approach speeds are given on Figure A8-7 for operation with Military thrust. The data are directly applicable to the wing-tip stores and no-external-stores configurations. Rates of climb available with stores at the pylon and fuselage positions will be slightly less because of the additional drag. Flight characteristics and the effects of speed and configuration changes are described in Section VI. However, note that go-around performance given here for the Military thrust setting applies to the landing configuration only. Rates of climb are not critical with TAKE-OFF flaps, even at high weights and high ambient temperatures.



FA0473

SAMPLE PROBLEM

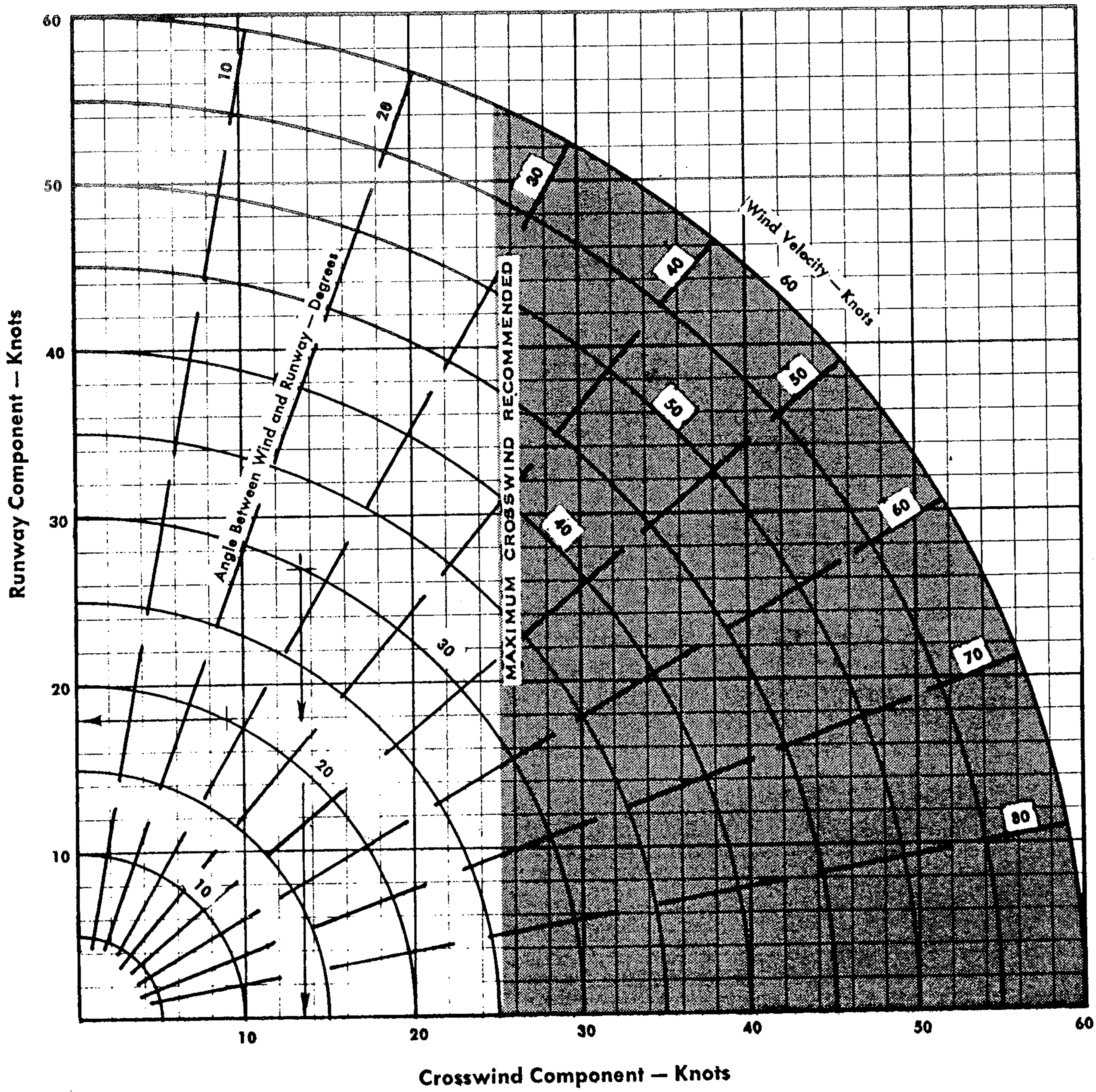
Determine the rate of climb available for go-around with military thrust and landing configuration, for the following conditions:

- Gross Weight – 15100 lb
- Air Temperature – 40° C
- Pressure Altitude – 3000 ft
- IAS – 170 knots

From Figure A8-7 determine the available rate of climb to be 790 fpm.

WIND COMPONENT CHART-LANDING

WITH OR WITHOUT DRAG CHUTE



(NOTE: FOR CROSSWIND COMPONENT ENTER CHART WITH MAXIMUM REPORTED GUST VELOCITY)

Figure A8-1

LANDING SPEED SCHEDULE

All External Stores Configurations

Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

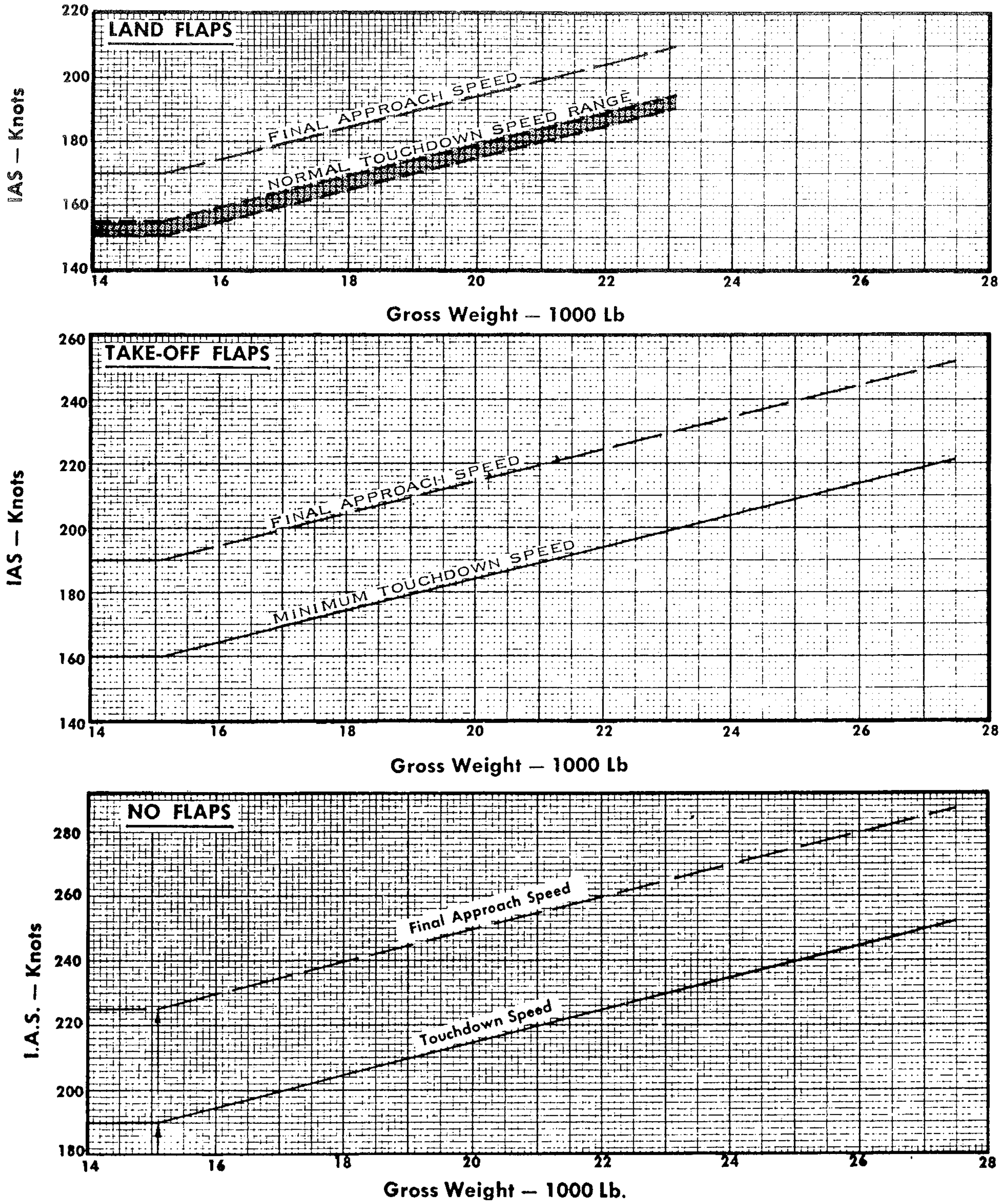


Figure A8-2

LANDING DISTANCE WITH DRAG CHUTE

LANDING FLAPS

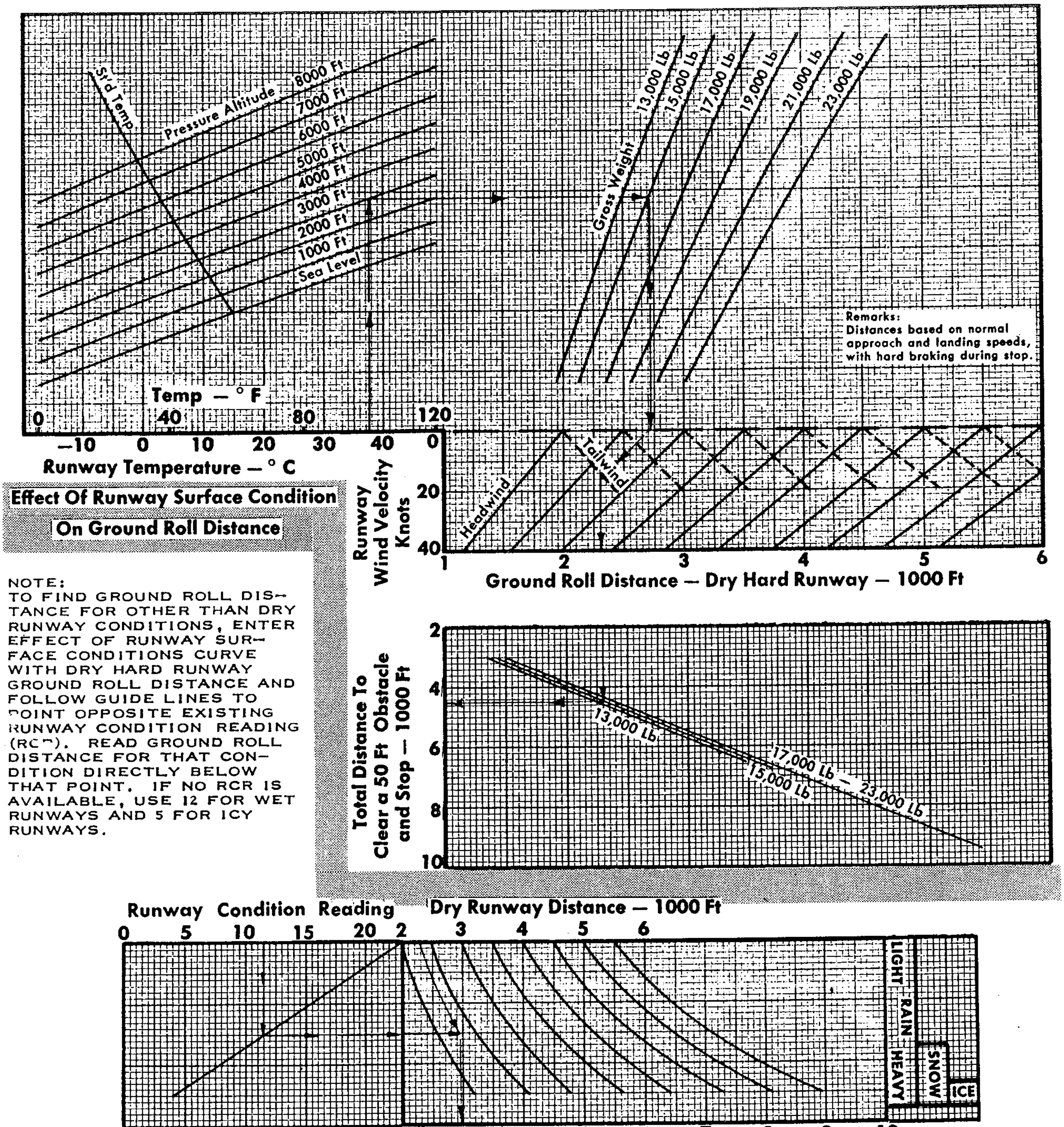
Anti-Skid Brakes — Boundary Layer Control On — 18-Foot Drag Chute

All External Stores Configurations

Level Runway

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE:
TO FIND GROUND ROLL DISTANCE FOR OTHER THAN DRY RUNWAY CONDITIONS, ENTER EFFECT OF RUNWAY SURFACE CONDITIONS CURVE WITH DRY HARD RUNWAY GROUND ROLL DISTANCE AND FOLLOW GUIDE LINES TO POINT OPPOSITE EXISTING RUNWAY CONDITION READING (RCR). READ GROUND ROLL DISTANCE FOR THAT CONDITION DIRECTLY BELOW THAT POINT. IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS.

Figure A8-3

LANDING DISTANCE WITHOUT DRAG CHUTE

LANDING FLAPS

Boundary Layer Control On

All External Stores Configurations

Level Runway Anti-Skid Brakes

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

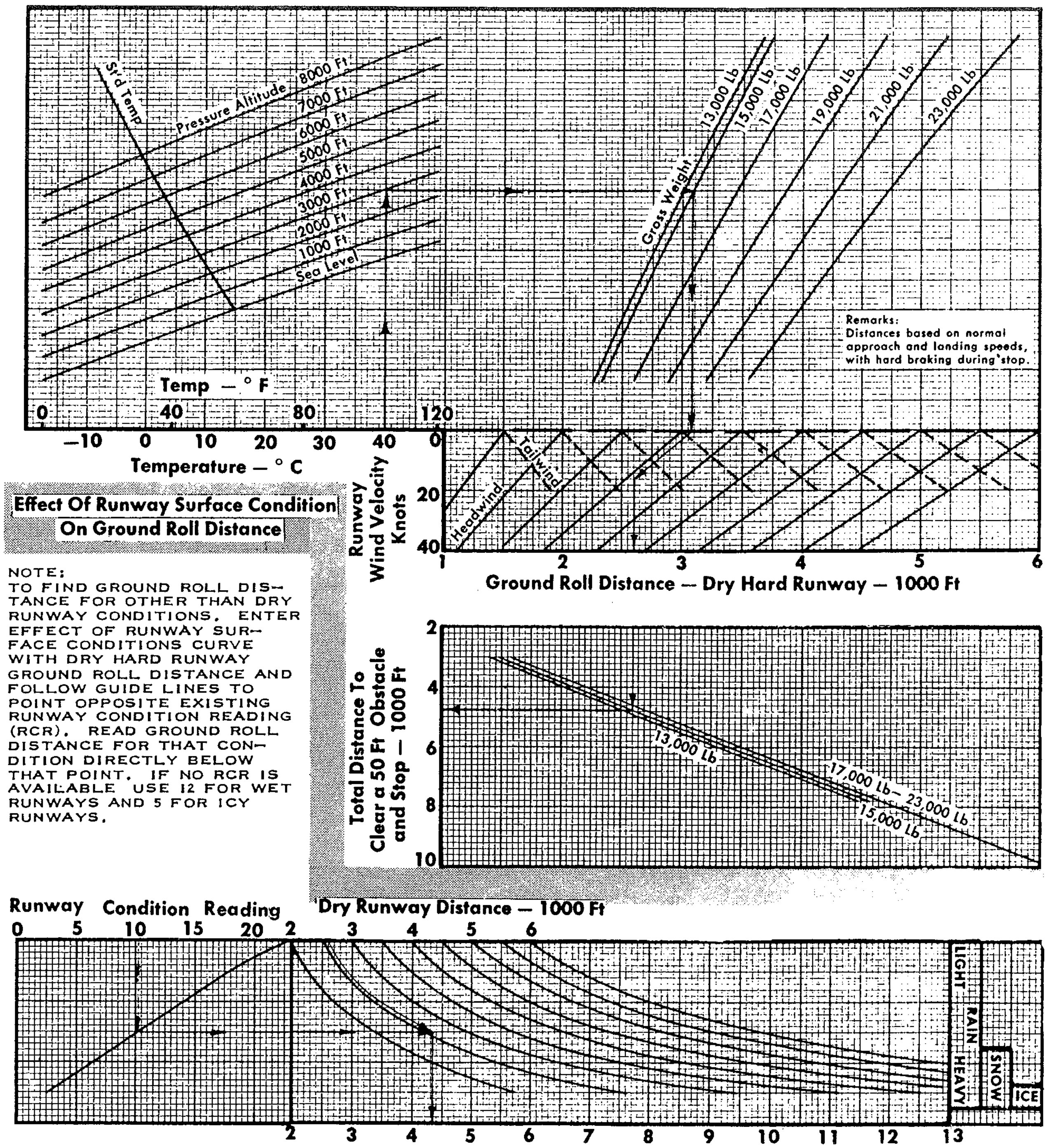


Figure A8-4

LANDING DISTANCE WITH DRAG CHUTE

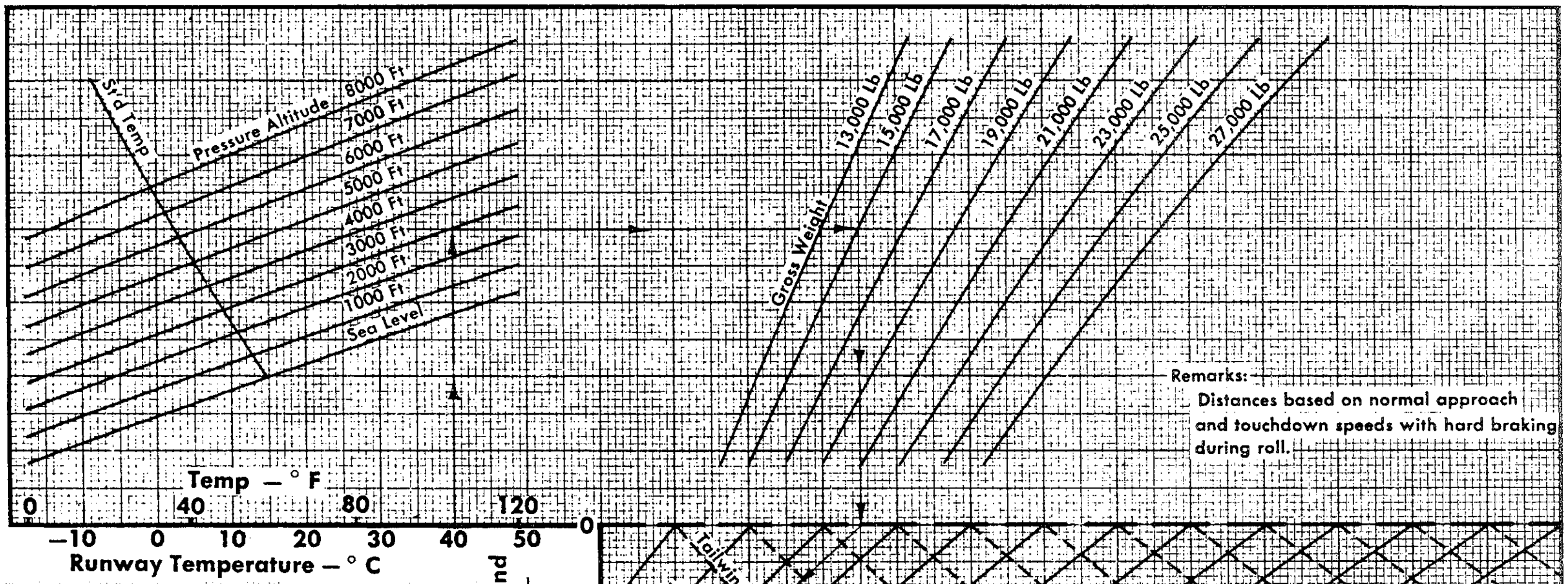
TAKEOFF FLAPS

All External Stores Configurations

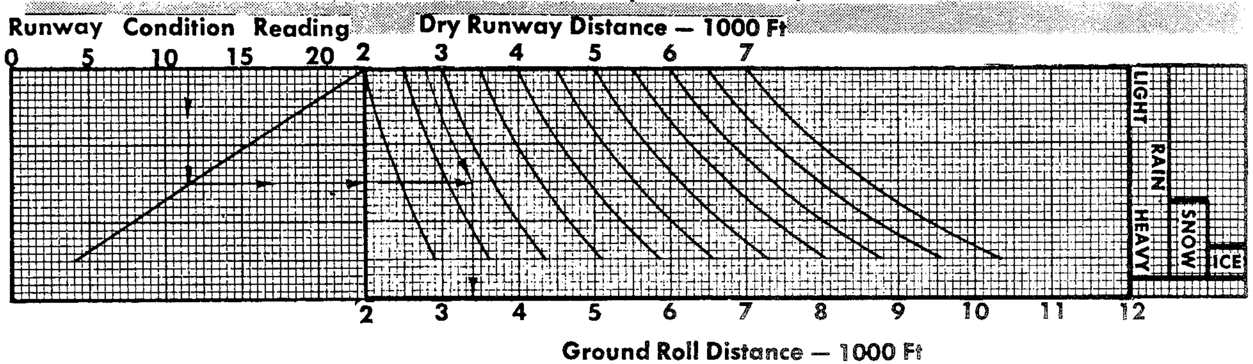
Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Anti-Skid Brakes — Level Runway — 18-Foot Drag Chute

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



EFFECT OF RUNWAY SURFACE CONDITION ON GROUND ROLL DISTANCE



NOTE:
TO FIND GROUND ROLL DISTANCE FOR OTHER THAN DRY RUNWAY CONDITIONS, ENTER EFFECT OF RUNWAY SURFACE CONDITIONS CURVE WITH DRY HARD RUNWAY GROUND ROLL DISTANCE AND FOLLOW GUIDE LINES TO POINT OPPOSITE EXISTING RUNWAY CONDITION READING (RCR). READ GROUND ROLL DISTANCE FOR THAT CONDITION DIRECTLY BELOW THAT POINT. IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS.

Figure A8-5

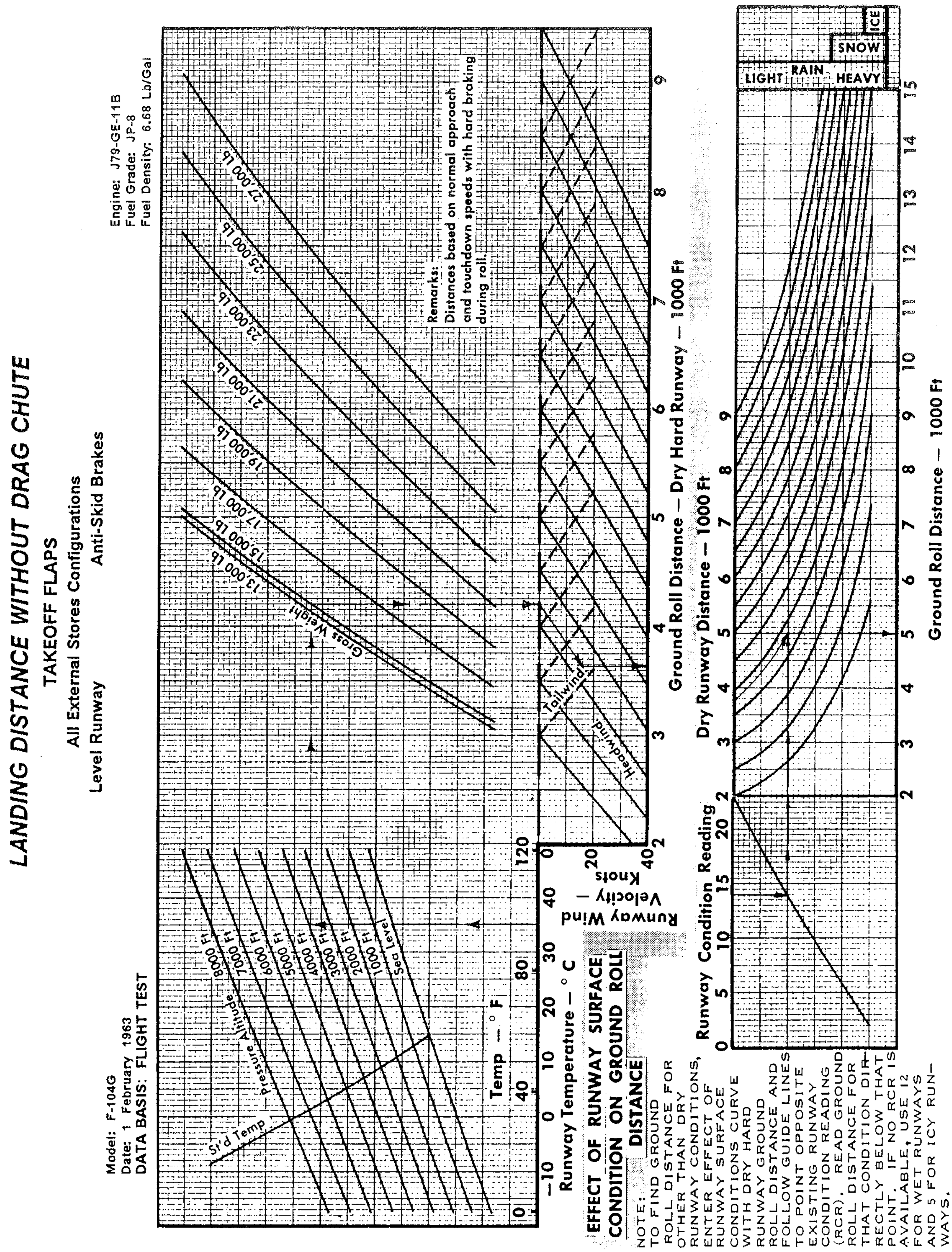

LANDING DISTANCE WITHOUT DRAG CHUTE
TAKEOFF FLAPS
All External Stores Configurations
Level Runway
Anti-Skid Brakes
Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST
Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal
Remarks:
Distances based on normal approach and touchdown speeds with hard braking during roll.
Std Temp
Pressure Altitude
8000 Ft
7000 Ft
6000 Ft
5000 Ft
4000 Ft
3000 Ft
2000 Ft
1000 Ft
Sea Level
Temp — °F
0
40
80
120
Runway Temperature — °C
-10
0
10
20
30
40
Gross Weight
13,000 Lb
15,000 Lb
17,000 Lb
19,000 Lb
21,000 Lb
23,000 Lb
25,000 Lb
27,000 Lb
Tailwind
Headwind
Runway Wind Velocity — Knots
0
20
40
Ground Roll Distance — Dry Hard Runway — 1000 Ft
2
3
4
5
6
7
8
9
EFFECT OF RUNWAY SURFACE CONDITION ON GROUND ROLL DISTANCE
NOTE:
TO FIND GROUND ROLL DISTANCE FOR OTHER THAN DRY RUNWAY CONDITIONS, ENTER EFFECT OF RUNWAY SURFACE CONDITIONS CURVE WITH DRY HARD RUNWAY GROUND ROLL DISTANCE AND FOLLOW GUIDE LINES TO POINT OPPOSITE EXISTING RUNWAY CONDITION READING (RCR). READ GROUND ROLL DISTANCE FOR THAT CONDITION DIRECTLY BELOW THAT POINT. IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS.
Runway Condition Reading
0
5
10
15
20
Dry Runway Distance — 1000 Ft
2
3
4
5
6
7
8
9
LIGHT
RAIN
HEAVY
SNOW
ICE
Ground Roll Distance — 1000 Ft
2
3
4
5
6
7
8
9
10
11
12
13
14
15

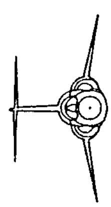
Figure A8-6

RATE OF CLIMB AVAILABLE FOR GO-AROUND — MILITARY THRUST

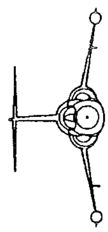
Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Landing Flaps Down/
Landing Gear Down/

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



No External Stores



Tip Tanks

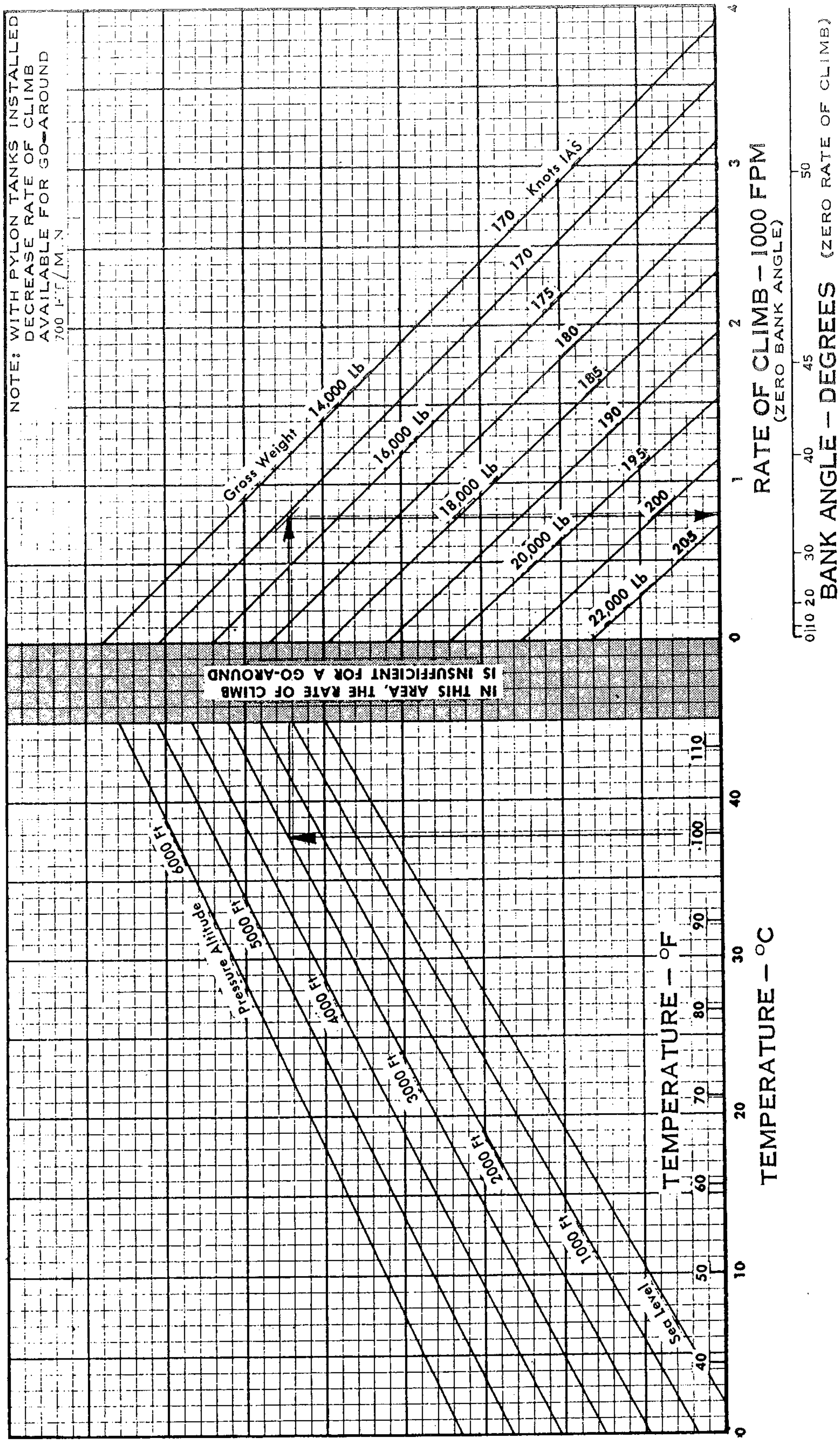


Figure A8-7

PART 9

COMBAT PERFORMANCE

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

INTRODUCTION

This part contains the necessary data for determining aircraft performance supersonic speeds. This typically includes acceleration, climb, turning and supersonic cruise performance. The data shown contain some allowances for normal in-service thrust variations.

MAXIMUM THRUST ACCELERATION

Maximum thrust level flight acceleration time, distance, and fuel required to accelerate from an initial cruise speed of Mach 0.90 are plotted as functions of the final desired Mach number, initial operating weight and ambient temperature. Data are shown for operation without external stores and in configurations with wing pylons tanks and tip tanks.

Maximum acceleration performance occurs at approximately 35000 feet pressure altitude for all configurations and results in minimum fuel required to reach any desired final Mach number.

Ambient temperature above standard and the operating characteristics of individual engines may have a decided effect on acceleration performance. The maximum speed may be limited by compressor inlet temperature (refer to Section V "Operating Limitations") or thrust available.

Engine air inlet temperature is a function of ambient temperature and flight Mach number. This limit is illustrated on each of the charts by a shaded area.

The maximum permissible speed can be read directly from the ambient temperature scale and the Mach number lines shown.

For example, in the no-external-stores configuration at 35000 feet pressure altitude and an ambient temperature of $-40°$ C the maximum Mach number is 1.86. Obviously, as temperature increases above standard, Maximum thrust will decrease and the level of excess thrust available for acceleration will also decrease. Therefore, operation at these temperatures requires careful flight planning because of the additional time required to accelerate and the corresponding increase in fuel used. The sample problem illustrates the use of acceleration charts.

SAMPLE PROBLEM

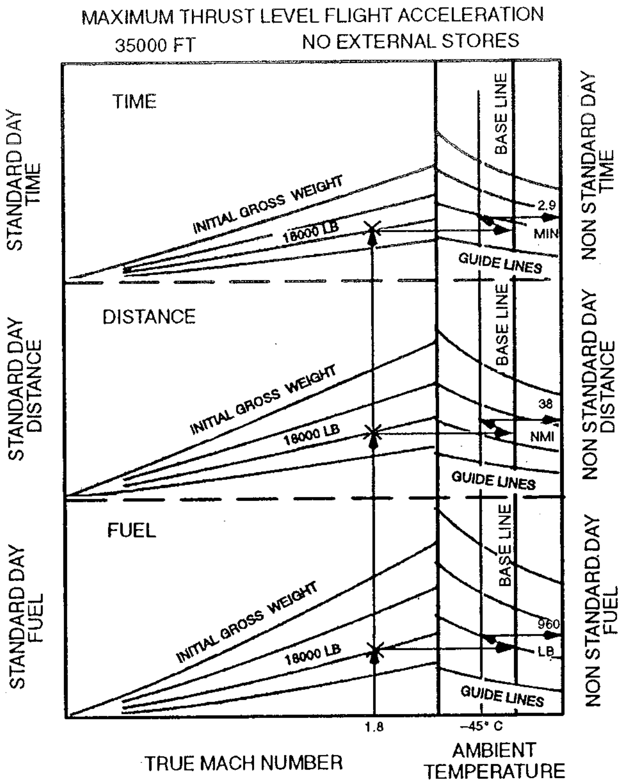
Determine the time, distance and fuel required to accelerate from Mach 0.90 to Mach 1.8 at 35000 feet for operation with no external stores. Assume an initial gross weight of 18000 pounds and an ambient temperature of $-45°$ C.

Enter Figure A9-1 at a final Mach number of 1.8, and at each 18000 lb gross weight position proceed horizontally right to the base line, follow the guide lines to the left until intersecting the ambient temperature, then proceed horizontally right and read as follows:

Time of accelerate .... 2.9 min

Distance to accelerate .... 38 nmi

Fuel to accelerate .... 960 lb



FA0330

MAXIMUM THRUST CLIMB PERFORMANCE

Climb performance curves of time, distance, and fuel to climb from a base altitude of 35000 feet to the combat ceilings are given for several representative Mach numbers. Use of the curves is illustrated by the following sample problem.

SAMPLE PROBLEM

Determine the standard-day climb performance for a Maximum thrust climb at Mach 1.7 from 35000 feet to 55000 feet at an initial gross weight of 18000 pounds. The aircraft has no external stores.

a. Enter Figure A9-12 at the final altitude, 55000 feet and proceed horizontally to the initial gross weight, 18000 lb read:

Time .... 2.35 min

Fuel .... 600 lb

Distance .... 38 nmi

MAXIMUM THRUST TURNING PERFORMANCE

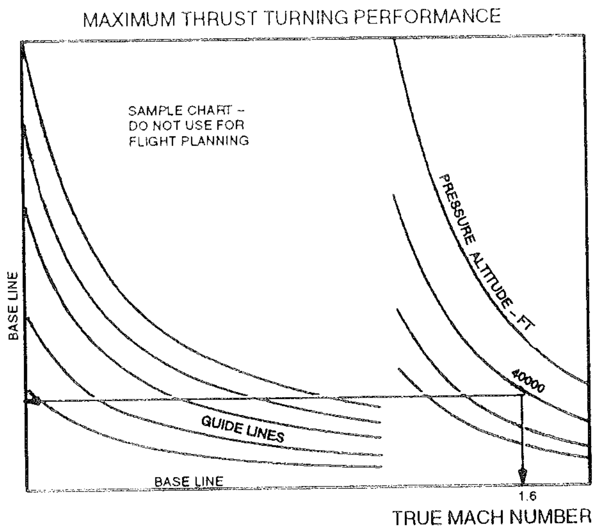
A turn of up to 180° may be needed to complete the intercept, requiring appreciable amounts of time and fuel for the maneuver. Turn radius as well as time and fuel should be considered in overall mission planning.
Maximum thrust turning performance is shown for constant altitude, minimum radius turns where speed and load factor (G's) are also held constant.
Maximum thrust turning performance load factor are shown for operation without external stores and in configurations with wing pylons tanks and tip tanks. Load factor when used in conjunction with the "Turning Performance" chart, Figure A9-27, provides turn radius and time for a 180° turn. The "Maximum Thrust Fuel Consumption" chart, Figure A9-30, provides the fuel flow to calculate the fuel used.

Low altitude turning performance is shown on a separate charts, Figure A9-28. Radius of turn, time to turn 180°, and true airspeed are presented versus pressure altitude, Δ temperature from standard day, load factor (G's) and bank angle for an indicated airspeed range of 300 to 500 knots.

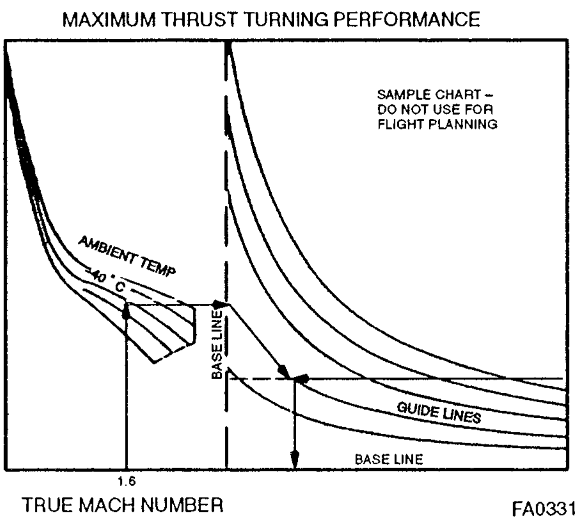
SAMPLE PROBLEM

Determine the 180° turn performance in the no external stores configuration at 40000 feet, ambient temperature of −40° C, true Mach number of 1.6 and gross weight of 18000 pounds.

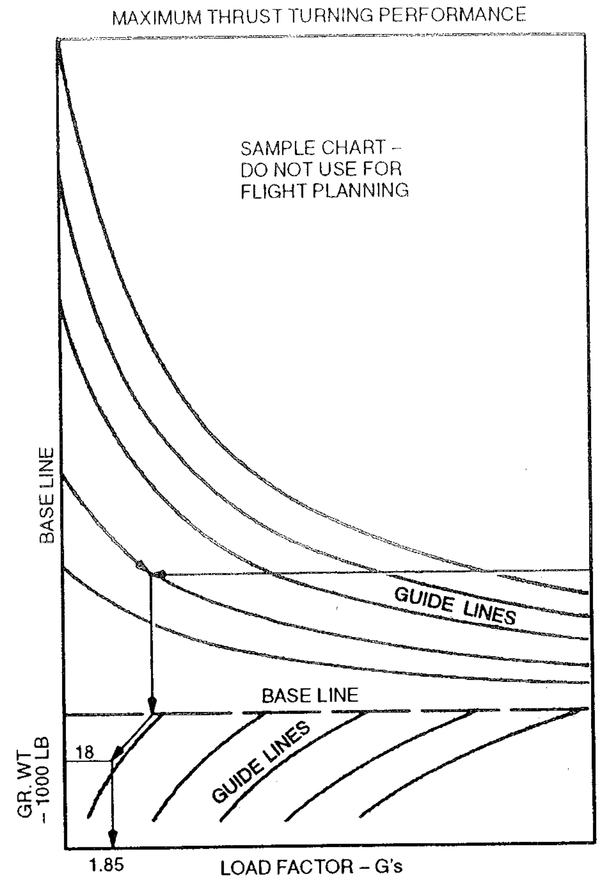
a. Enter Figure A9-23 on the right with 1.6 true Mach number. Proceed up to a pressure altitude of 40000 feet and establish a horizontal reference line to the left through the guide lines.



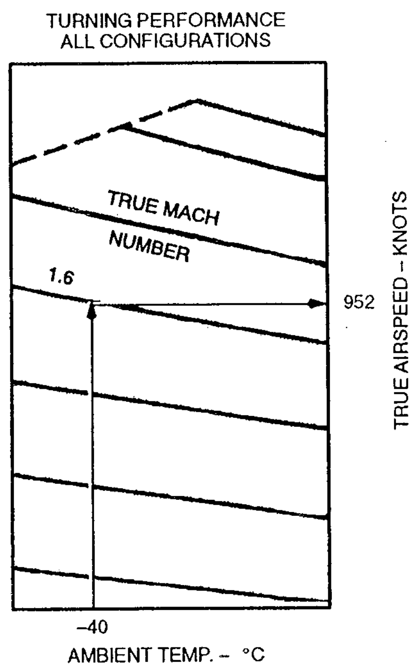
b. Reenter the chart on the left with 1.6 true Mach number. Proceed up to −40° C ambient temperature, then horizontally to the base line. Follow the guide lines to an intersection with the horizontal reference line from step a. If an intersection shall not be made with the horizontal reference line, maneuvering capability is not available for the desired conditions.



c. Proceed down from this point to the base line. Follow the guide lines to the gross weight of 18000 pounds. Directly below this point read, load factor 1.85 "G".

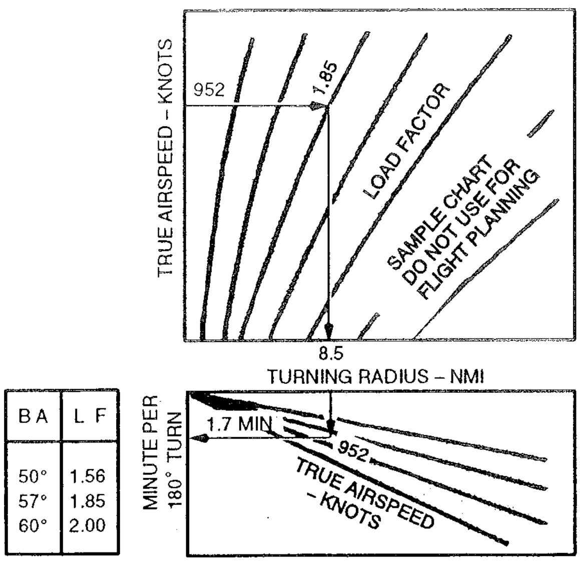


d. Refer to the turning performance chart to determine turn radius, time in 180° turn and bank angle. Enter the chart at the ambient temperature of −40° C and proceed up to 1.6 Mach number. Horizontally to the right read, true airspeed 952 knots.



e. Continue to the right to the load factor determined in step c. (1.85 "G"), directly below this point read turn radius 8.5 nmi.

TURNING PERFORMANCE
ALL CONFIGURATIONS



FA0332

f. Continue downward to the true airspeed from step d. (952 knots) and read 1.7 minutes for a 180° turn. From the table of bank angle and load factor interpolate the bank angle for 1.85 "G" as 57°.

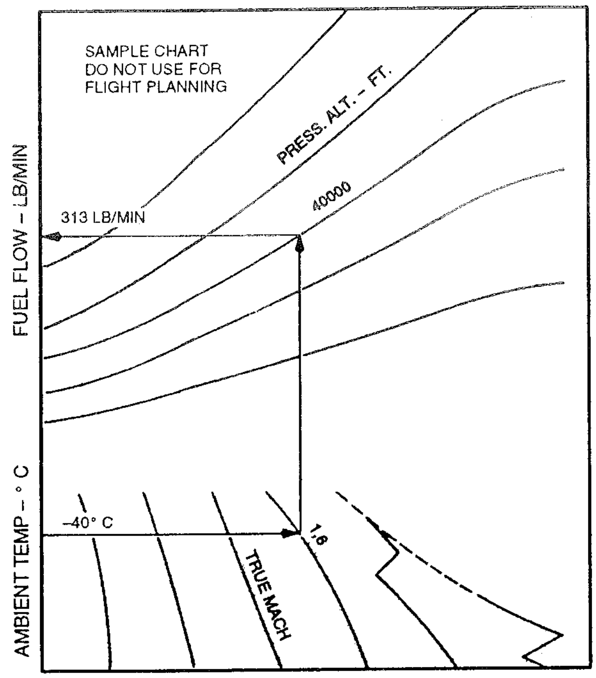
g. Fuel used for the turn is determined from the maximum thrust fuel consumption curve (Figure A9-30). Enter with the ambient temperature of −40° C and true Mach number 1.6. Proceed vertically upward to 40000 feet pressure altitude, at the left read fuel flow, 313 lb/min. Calculate the fuel used in 180° turn (313 × 1.7), or 532 lb.

MAXIMUM THRUST FUEL CONSUMPTION

Fuel consumption for Maximum thrust operation at constant speed and altitude applicable to full afterburning is provided for use in planning combat allowances at or near a target area.
The charts should not be interpreted as providing fuel flows for unaccelerated level flight with less than full afterburner, such as a dash approach to a target.

Figure A9-29 presents fuel consumption from sea level to 30000 feet at standard day temperatures for a Mach range of 0.8 to 1.8. Figure A9-30 presents fuel consumption from 30000 to 60000 feet as a function of ambient temperature and true Mach number.

MAXIMUM THRUST FUEL CONSUMPTION



SAMPLE PROBLEM

Determine the time available for combat at Mach 1.6 and 40000 feet if 3000 pounds of fuel are to be used. The ambient temperature is −40° C.
Enter the maximum thrust fuel consumption chart (Figure A9-30) at −40° C ambient temperature and Mach 1.6.
Proceed vertically upward to 40000 feet pressure altitude and read fuel flow, 313 lb/min. Combat time available is 3000/313 or 9.6 min.

Military Thrust Fuel Consumption

Fuel consumption for military thrust operation at constant speed and altitude is provided in Figure A9-31 for use in planning combat allowances at or near a target area.
Use of the chart is illustrated by the following example.

SAMPLE PROBLEM

Determine the fuel required for combat maneuvering with Military thrust at Mach 0.90 at a pressure altitude of 3000 feet and an ambient temperature that is 19° C (ΔT = + 10° C).
Enter Figure A9-31 at 3000 feet pressure altitude and proceed to Mach 0.90. Vertically below this point intersect the base line in the Δ temperature grid. Follow the guide lines to a Δ temperature above standard of 10° C. Directly below this point read fuel flow of 162 lb/min. Fuel required is 324 lb (2 × 162).

MILITARY THRUST ACCELERATION

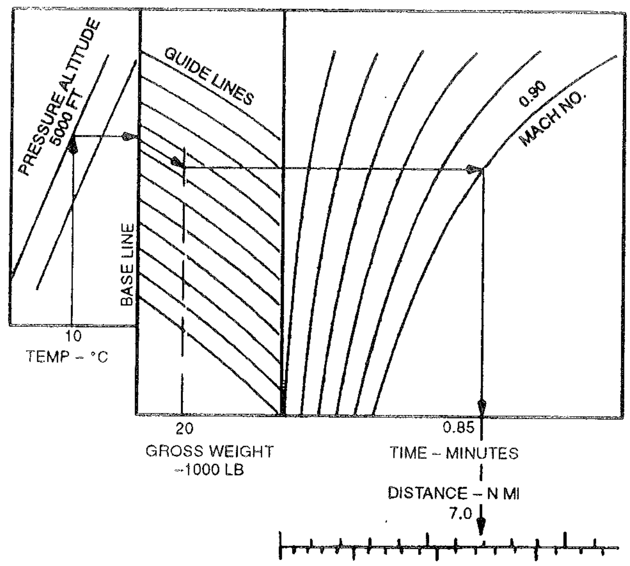
Military thrust level flight acceleration time, distance, and fuel required to accelerate from an initial cruise speed of 400 KIAS or 450 KIAS are plotted as functions of the final desired Mach number or indicated airspeed up to Mach 0.90, initial operating weight, and ambient temperature.
Data are shown for operation in configuration Drag Index (D.I.) 30, 40 and 75.
Determine the index for the configuration desired from the Store Drag Number Table in Part 1, and interpolate the acceleration performance.

SAMPLE PROBLEM

Determine the time, distance and fuel required to accelerate from 400 KIAS to Mach 0.90 at 5000 feet and an ambient temperature of 10° C.
The aircraft configuration is equivalent to Drag Index 30 and the gross weight at the start of the acceleration is 20000 pounds. On Figure A9-32, at the ambient temperature and the pressure altitude, proceed horizontally to the right to the gross weight base line, follow the guide lines to the desired gross weight.
Proceed horizontally to the right to the Mach number and directly below this point read:

Time to accelerate	0.85 min
Distance to accelerate	7.0 nmi
Fuel to accelerate	124 lb

LOW ALTITUDE
MILITARY THRUST LEVEL FLIGHT ACCELERATION
FROM 400 KNOTS IAS
FUSELAGE MD – 6 BOMB

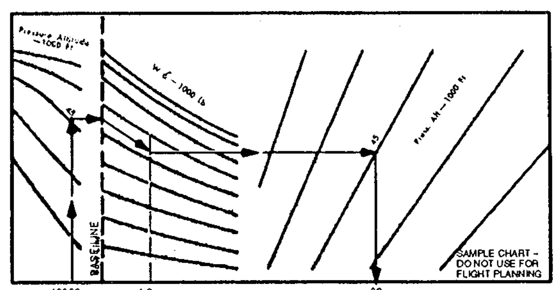


FA0333

AFTERBURNING CRUISE PERFORMANCE

Afterburning cruise performance is provided in Figure A9-36 thru Figure A9-39 for operation with no external stores, and in configurations with wing pylons tanks and tip tanks. The nautical miles per 1000 pounds of fuel used are applicable to supersonic cruising with the afterburner throttled to maintain level flight at constant Mach number from 1.7 to 2.0.

AFTERBURNING CRUISE PERFORMANCE



SAMPLE PROBLEM

Determine the fuel required to make a 50 nautical mile dash at Mach 1.8 and 45000 feet, at a gross weight of 19000 pounds. The aircraft is in the tip tanks configuration.

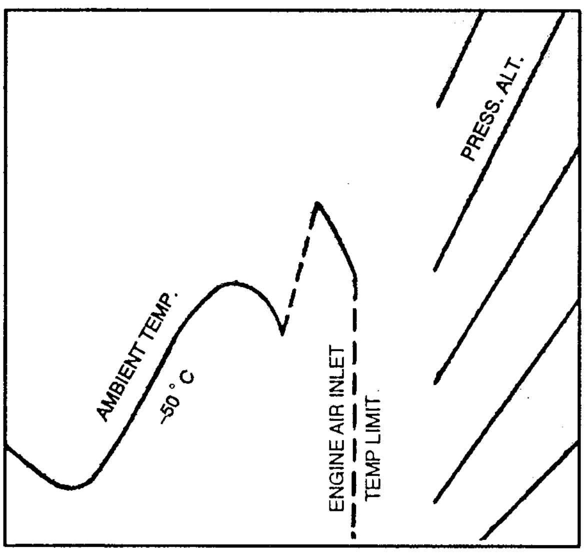
Enter Figure A9-37 at 19000 pounds gross weight and proceed vertically to 45000 feet, then horizontally to the base line. Follow to the right along a constant W/Δ line until it intersects with Mach 1.8. Proceed horizontally to 45000 feet; read 66.5 nautical miles per thousand pounds of fuel.

Fuel used ...... (50 nmi/66.5 nmi/1000 lb) = 752 lb

MAXIMUM THRUST FLIGHT ENVELOPE

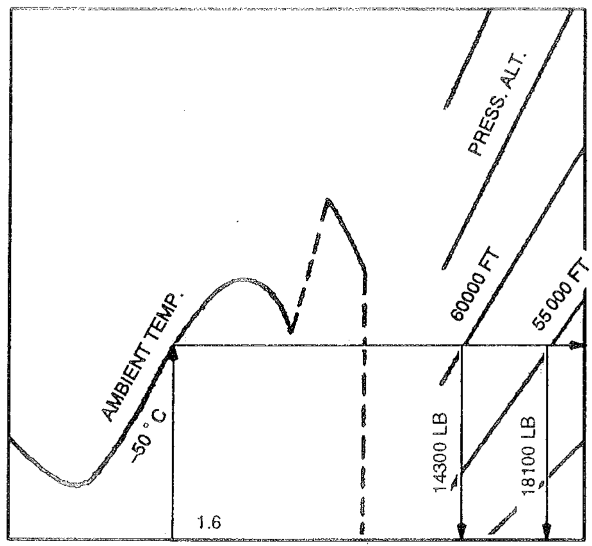
Maximum thrust flight envelopes are provided in Figure A9-40 thru Figure A9-43 for various configurations, plotted as a function of Mach Number, ambient temperature, pressure altitude and gross weight. The envelope illustrate the limits of maximum thrust operation due to thrust available or engine air inlet temperature. As an example, with no external stores (Figure A9-40), at an ambient temperature of − 50° C, the curve defines the operating limits.

MAXIMUM THRUST FLIGHT ENVELOPE



The envelope of gross weight and altitude for a particular Mach Number is obtained by an intersection with the temperature lines. For example, enter at Mach 1.6, proceed to the temperature, then establish a horizontal line to the right through the gross weight and altitude grid.

MAXIMUM THRUST FLIGHT ENVELOPE

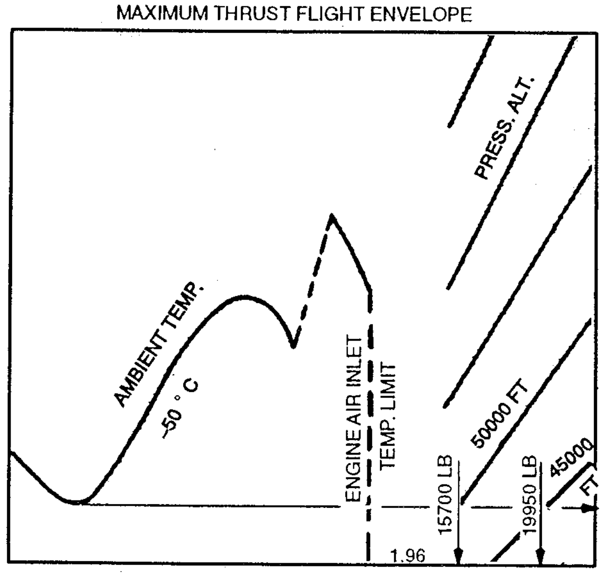


FA0334

Maximum thrust would be required at 60000 feet and 55000 feet for gross weights of 14300 pounds and 18100 pounds respectively. At lighter gross weights at the same altitude or other combinations of weight and altitude below the reference line, the afterburner may be throttled for cruise of excess thrust is available for acceleration, climb or maneuvering.

Maximum thrust acceleration performance initiated at subsonic Mach Numbers may be limited by thrust available. To determine the envelope of altitudes and gross weights for the thrust available, establish a horizontal line tangent to the lowest portion of the ambient temperature line.

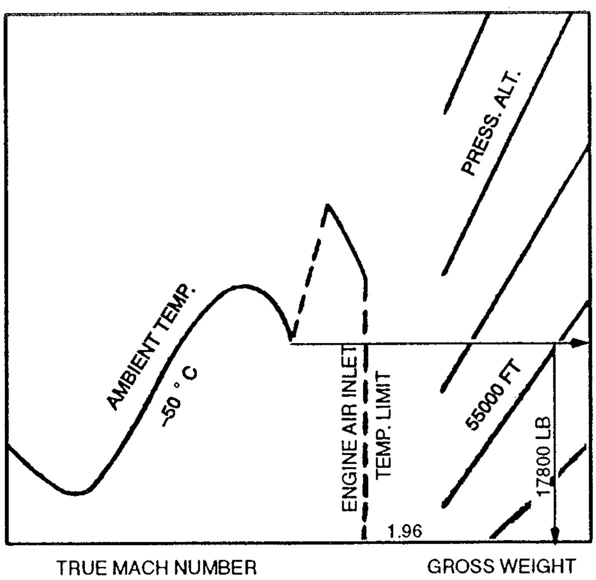
MAXIMUM THRUST FLIGHT ENVELOPE



For the example temperature of $-50°$ C, at 50000 feet the gross weight at the start of the acceleration must be 15700 pounds or less; at 45000 feet, 19950 pounds or less. Maximum Mach Number is limited by engine air inlet temperature at Mach 1.96. Acceleration initiated at subsonic speed or intermediate supersonic speed may also be limited by thrust available, due to the operating characteristics of individual engines at T_2 reset. The envelope of capabilities for this condition is determined in the same manner as the subsonic to supersonic acceleration. Establish a horizontal reference line through the combination of ambient temperature and Mach number at which T_2 reset begins.

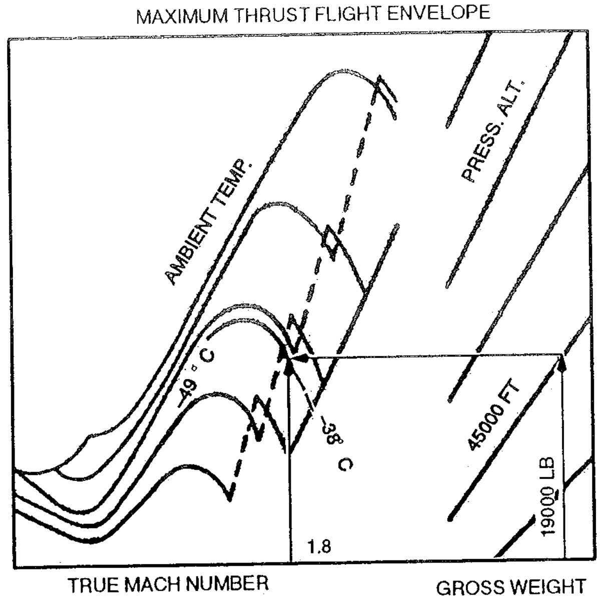
For the example conditions, at 55000 feet the gross weight at the start of the acceleration must be 17800 pounds or less.

MAXIMUM THRUST FLIGHT ENVELOPE



Afterburning cruise performance limitations can also be determined from the flight envelopes. For the sample problem illustrated in afterburning cruise, enter Figure A9-42 at 19000 pound gross weight proceed to 45000 feet, then horizontal to the left to Mach 1.8.

MAXIMUM THRUST FLIGHT ENVELOPE



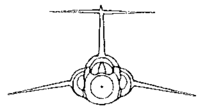
FA0335

Interpolation between the lines of ambient temperature cruising would be limited to a temperature of $-51°$ C without T_2 reset and $-39°$ C in the area of T_2 reset, with maximum thrust being required.

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
35000 FEET

NO EXTERNAL STORES



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE: 1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).

2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

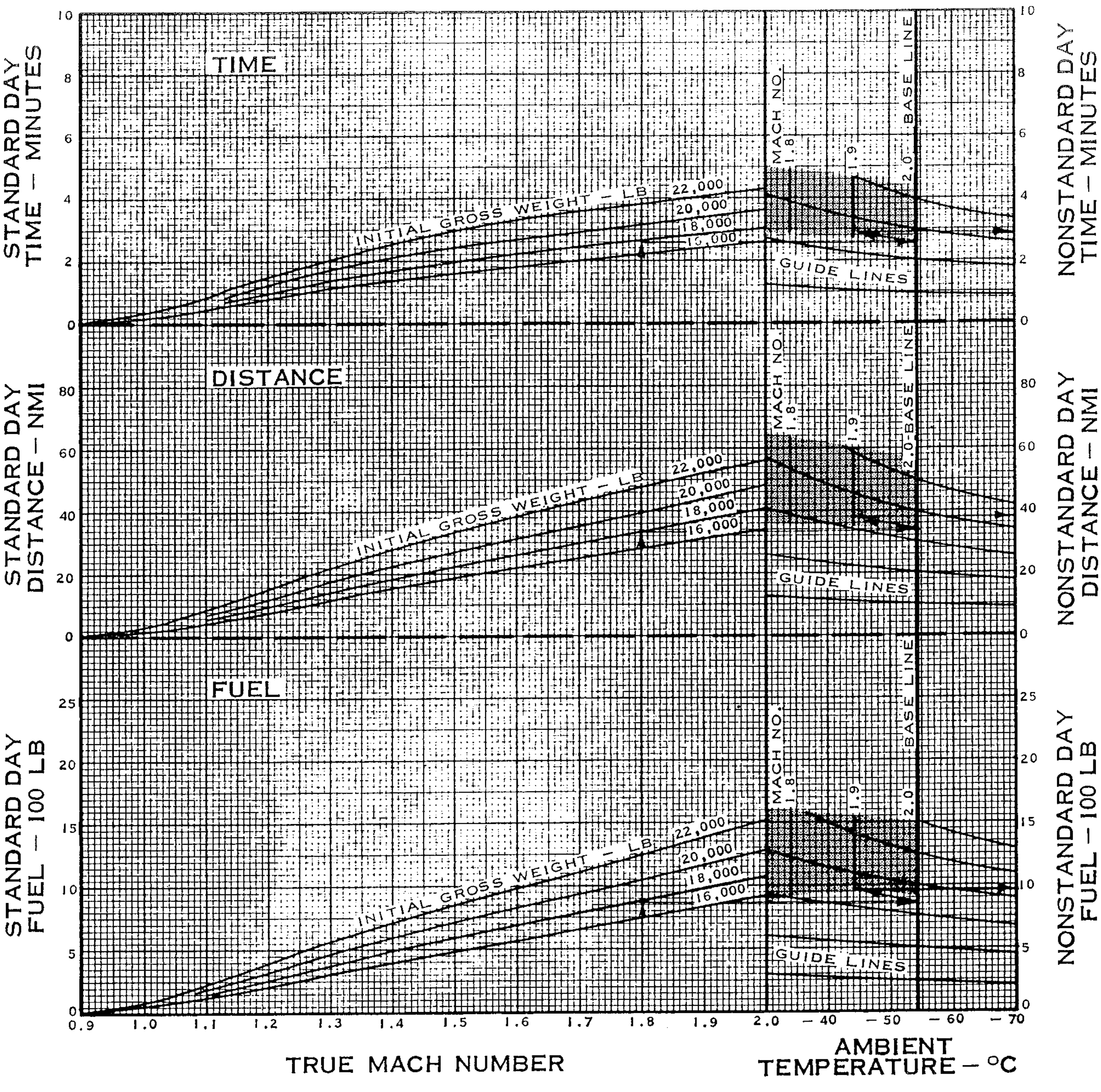


Figure A9-1

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
40000 FEET

NO EXTERNAL STORES

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

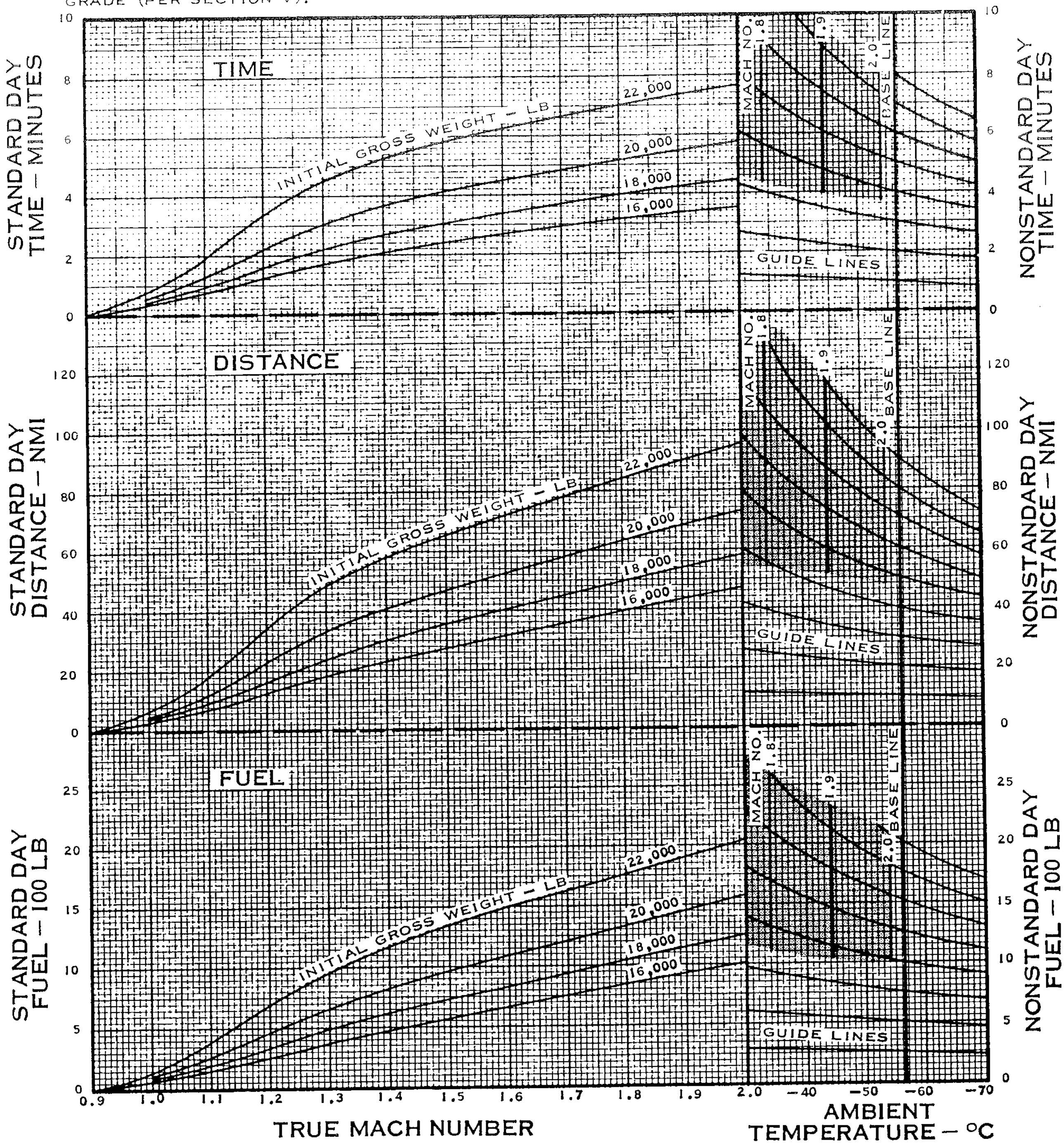
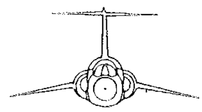


Figure A9-2

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
45000 FEET

NO EXTERNAL STORES



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

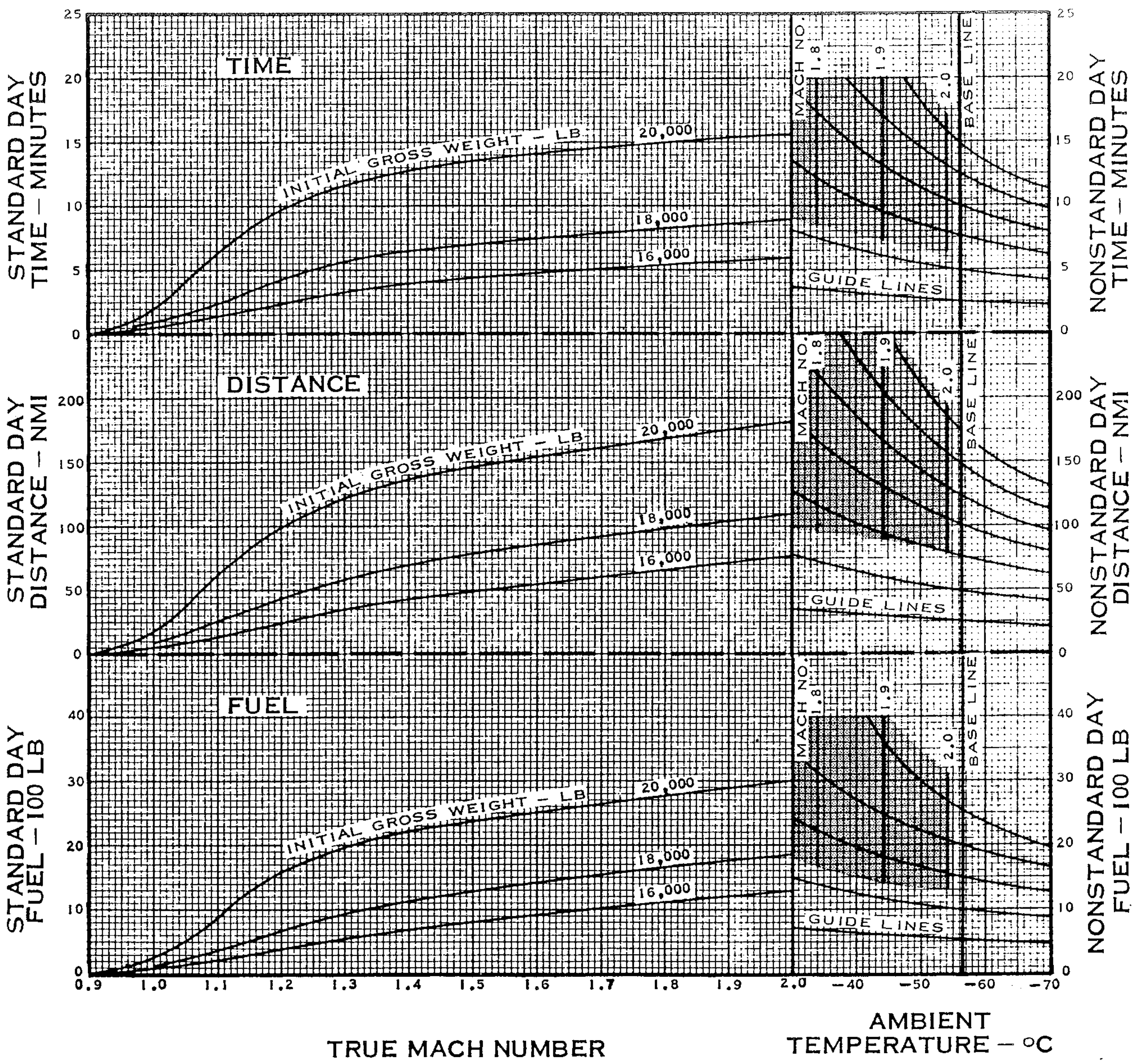
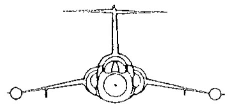


Figure A9-3

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
35000 FEET

TIP TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

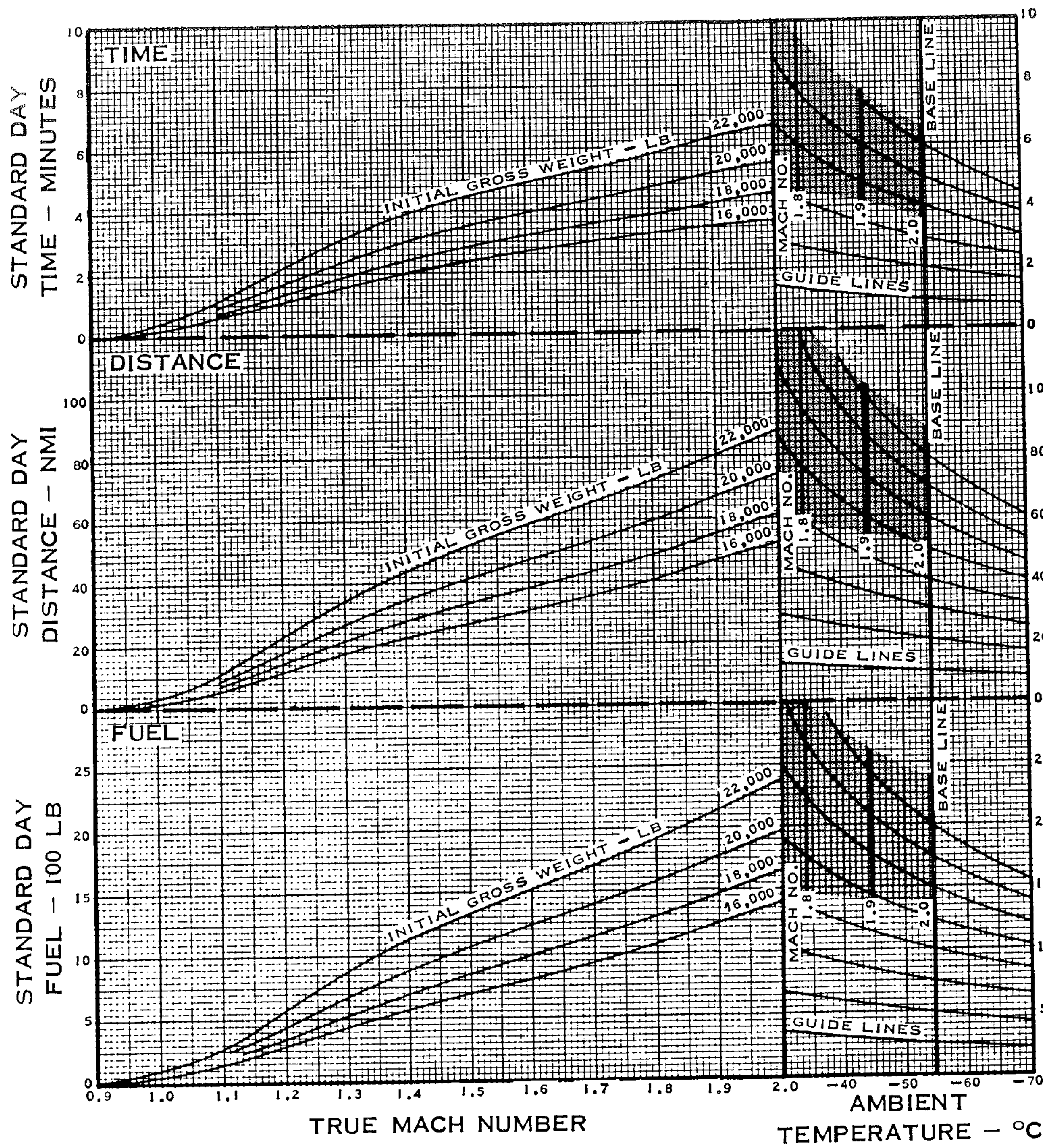
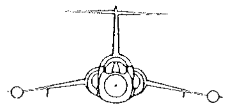


Figure A9-4

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
40000 FEET

TIP TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

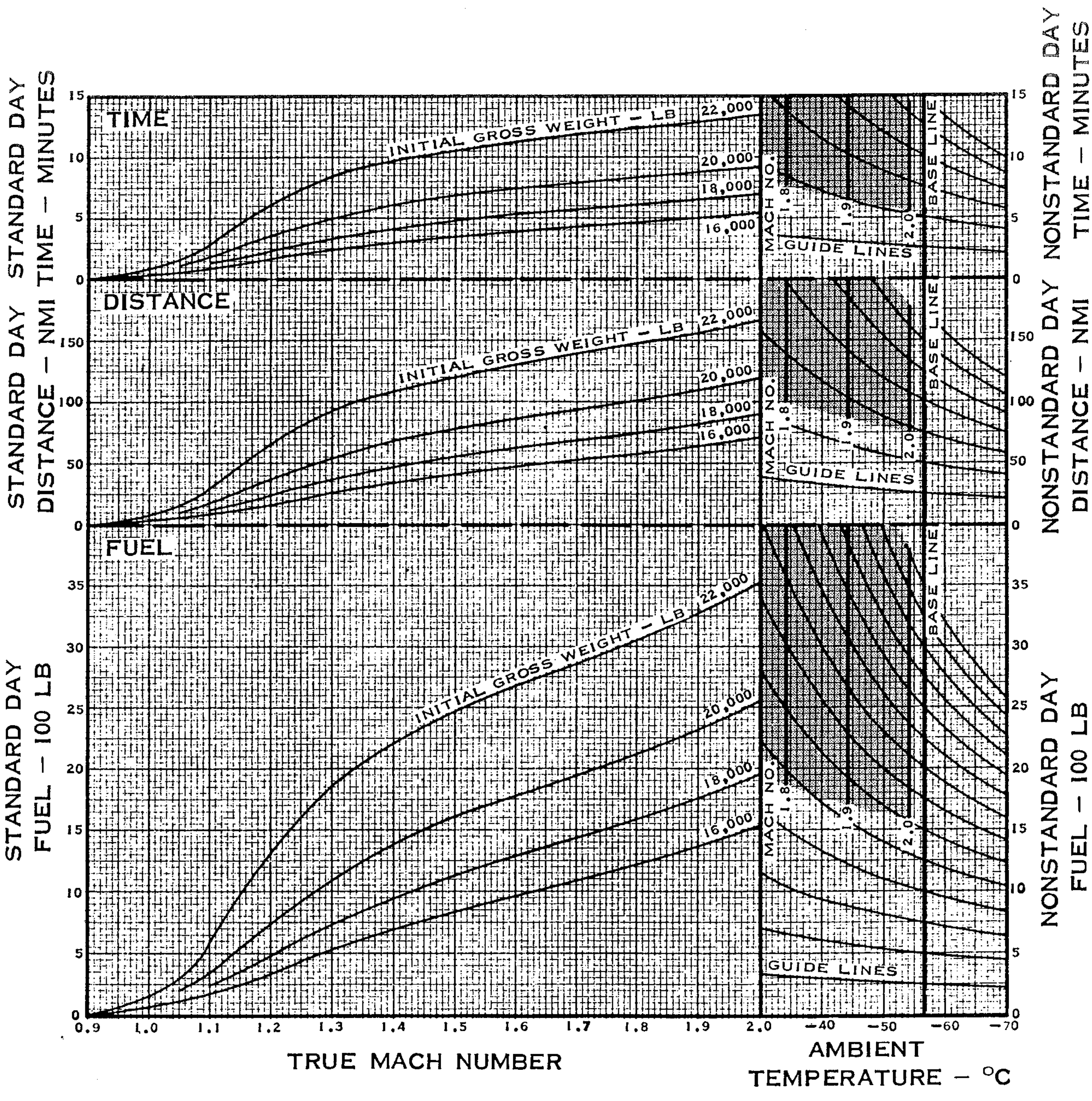


Figure A9-5

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
45000 FEET

TIP TANKS

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

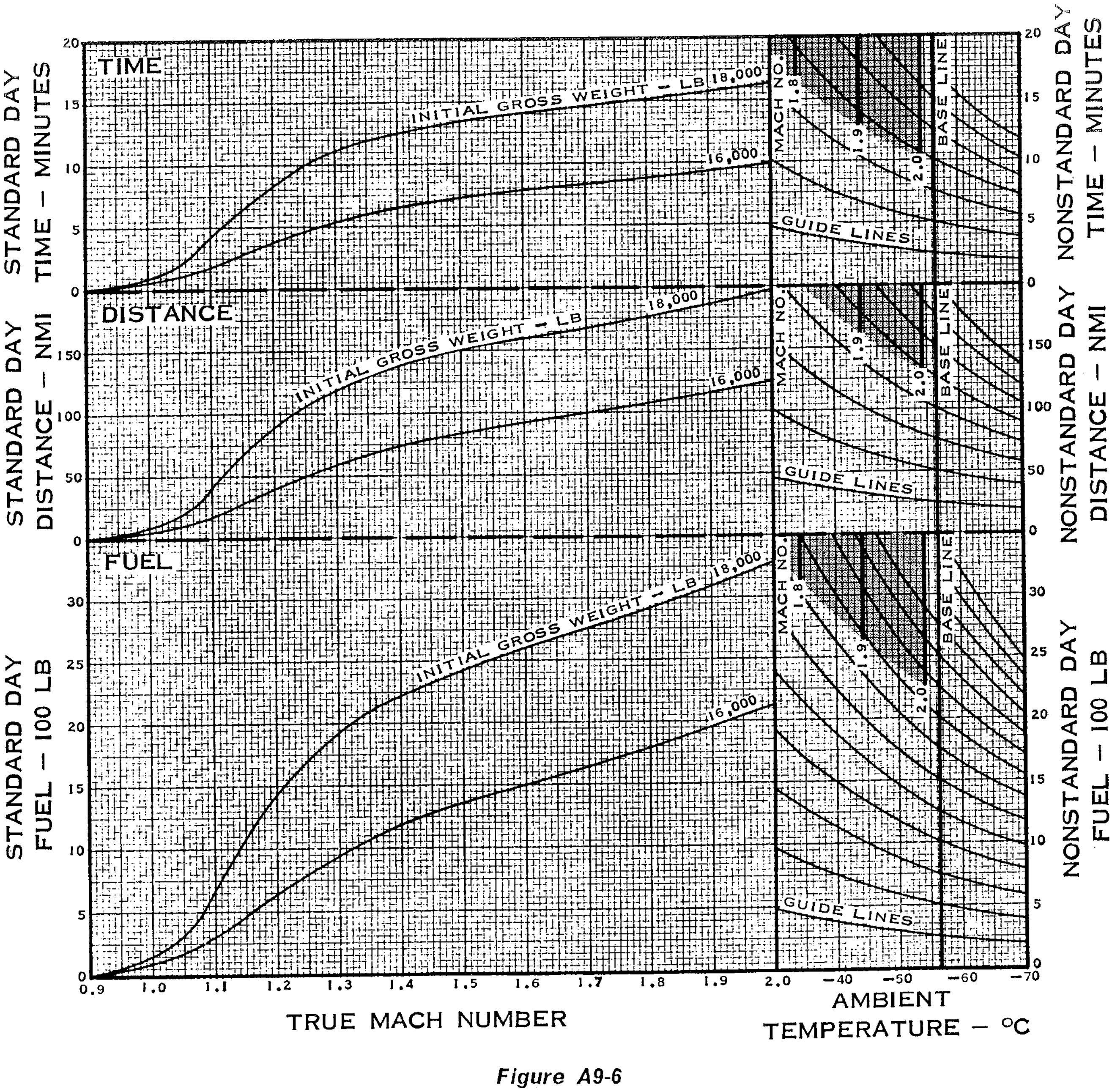
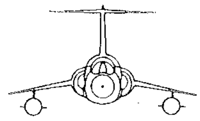


Figure A9-6

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
35000 FEET

PYLON TANKS



Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

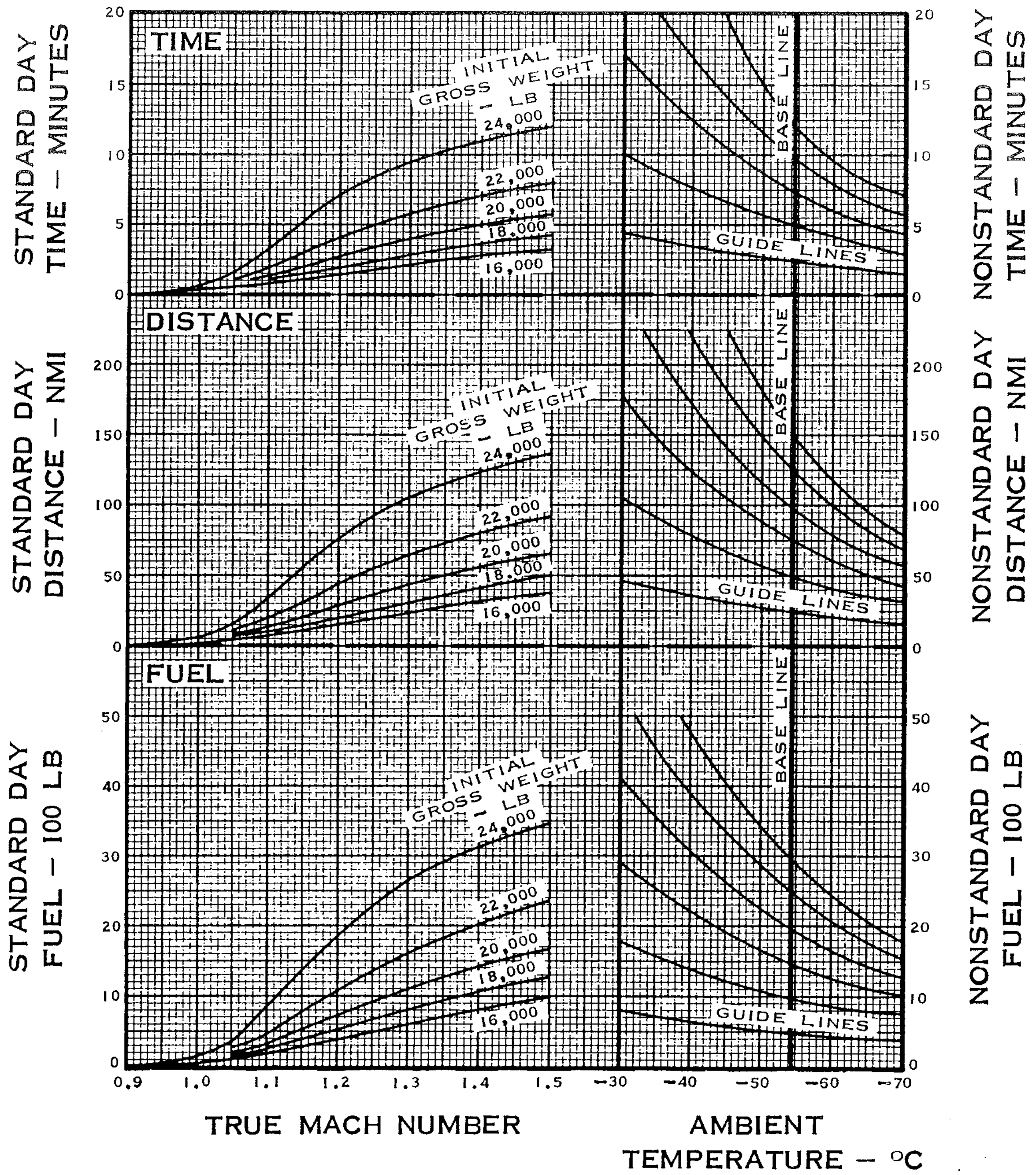
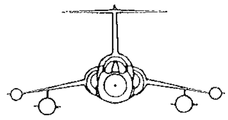


Figure A9-7

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

PRESSURE ALTITUDE
35000 FEET

TIP TANKS AND
PYLON TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

NOTE:
1. DO NOT EXCEED THE ENGINE AIR INLET TEMPERATURE OF 121 DEGREES CENTIGRADE (PER SECTION V).
2. MAXIMUM MACH NUMBER AS LIMITED BY ENGINE AIR INLET TEMPERATURE IS SHOWN IN THE SHADED AREA.

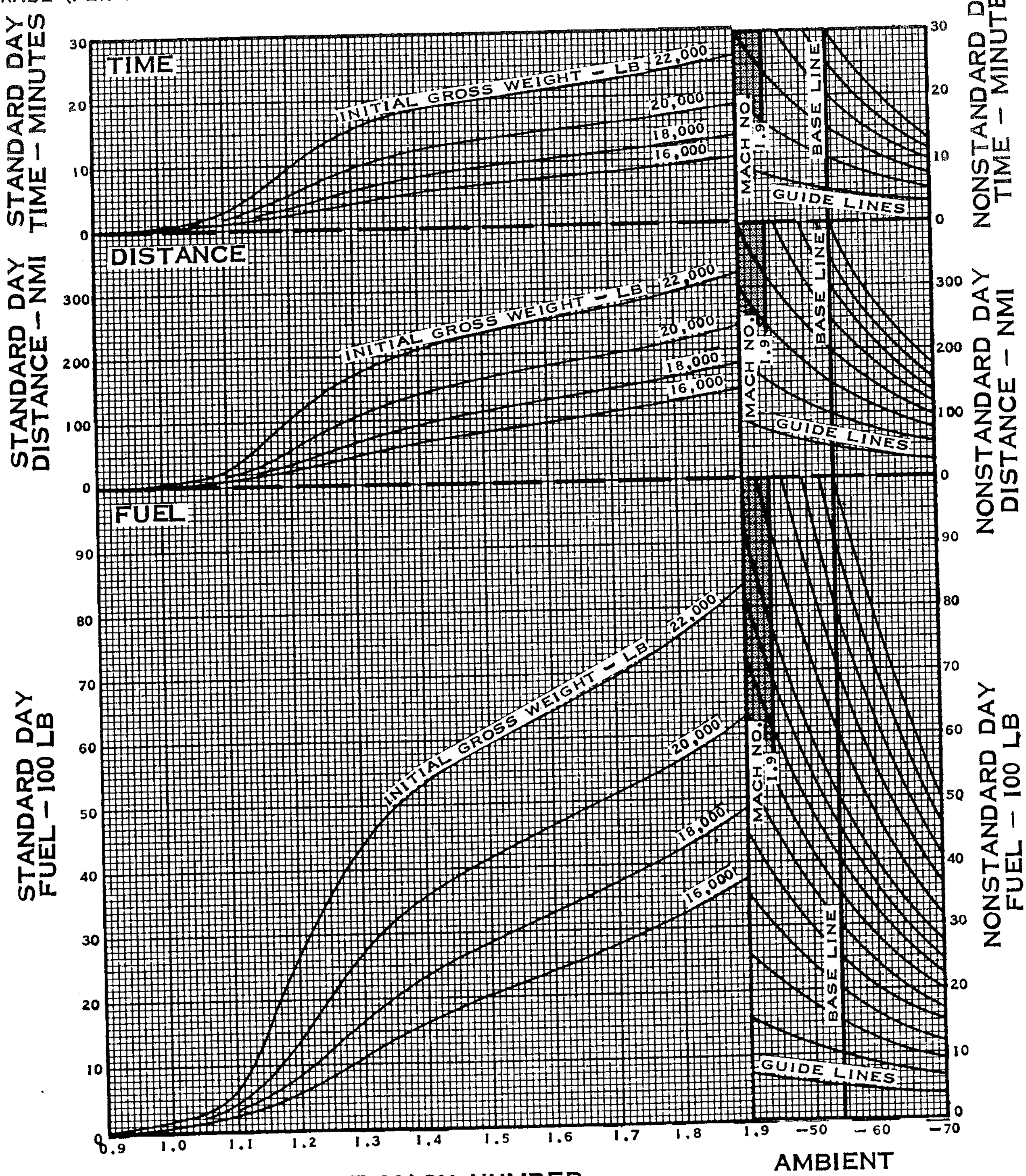
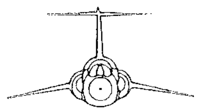


Figure A9-8

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.4

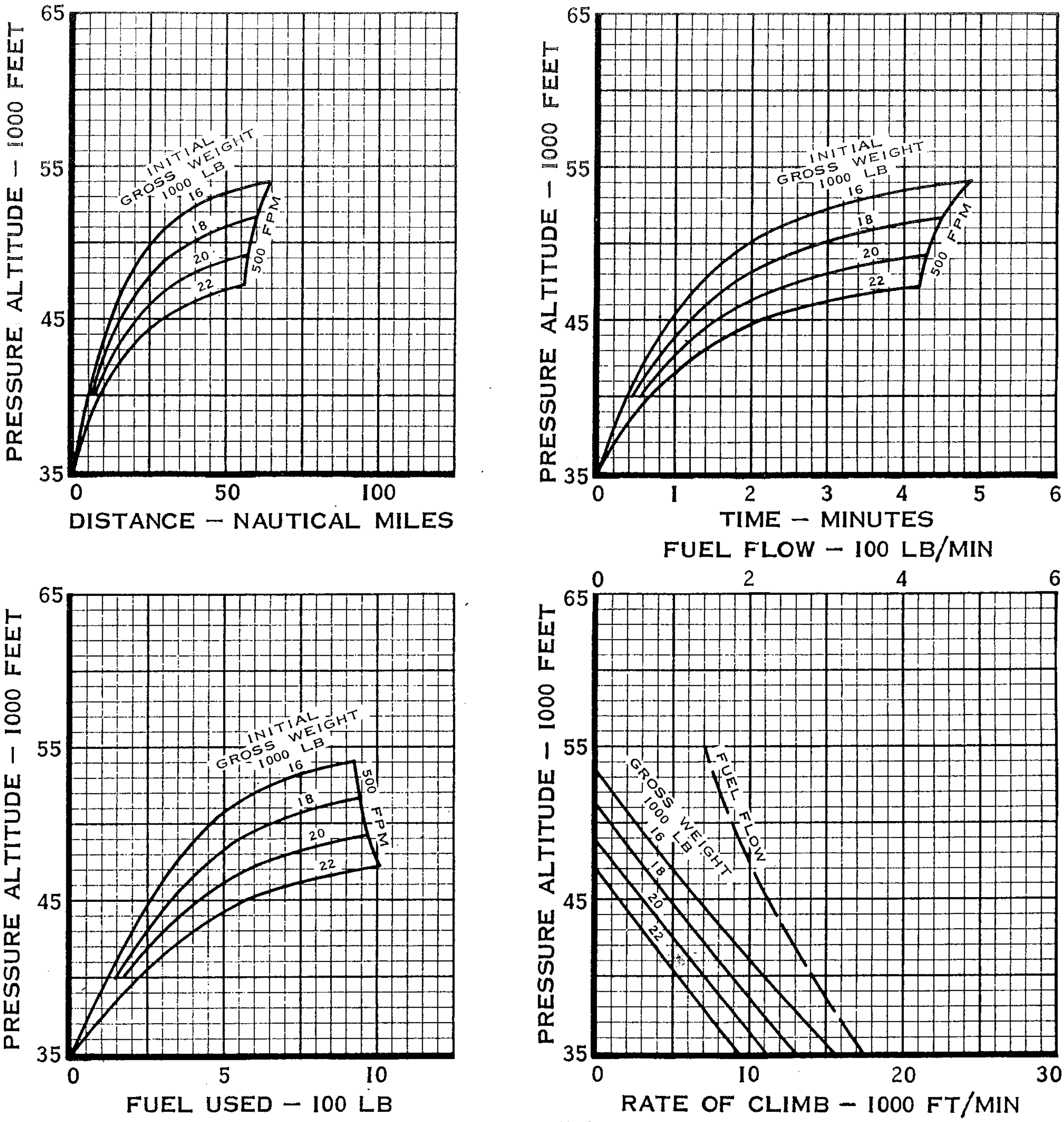


NO EXTERNAL STORES

Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 500 LB PER 10°F BELOW STANDARD.

Figure A9-9

MAXIMUM THRUST CLIMB CONTROL

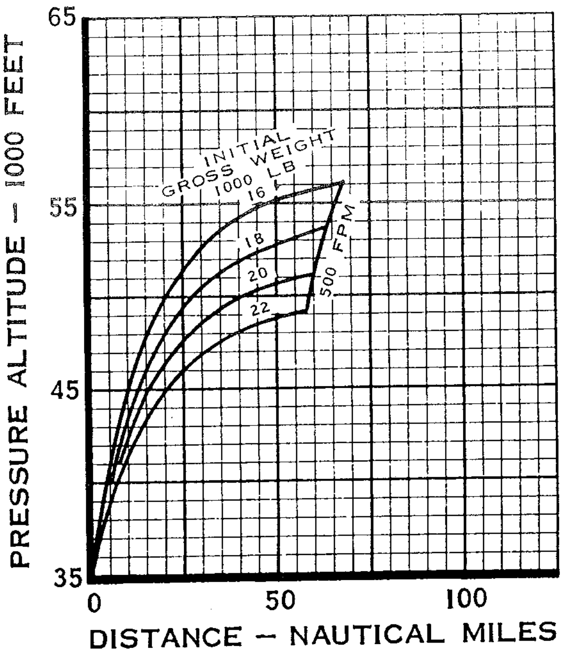
CLIMB SPEED MACH 1.5

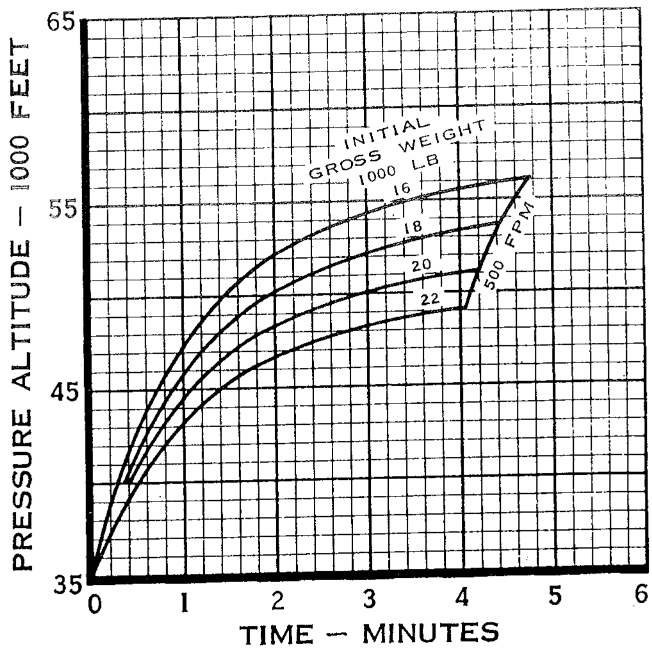
NO EXTERNAL STORES

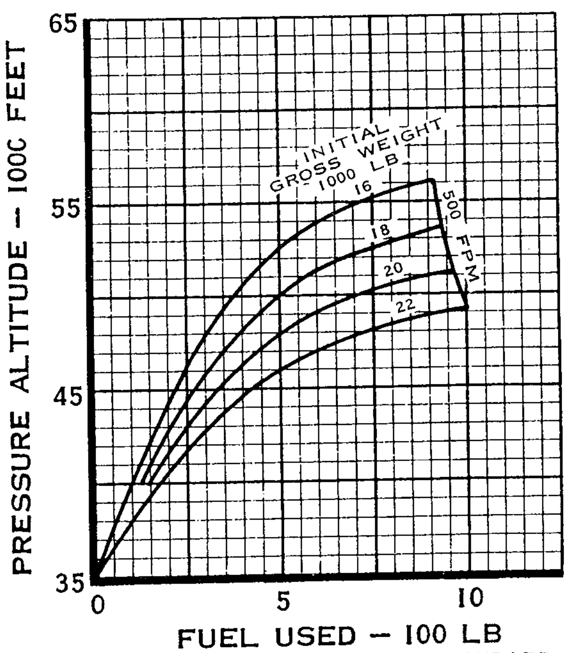
Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

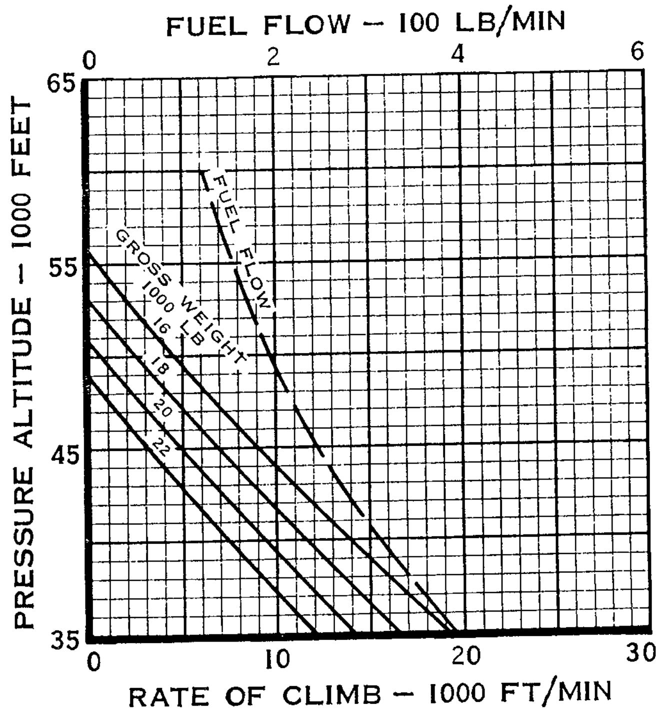
Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY







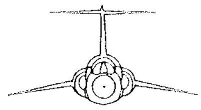


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 500 LB PER 10°F BELOW STANDARD.

Figure A9-10

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.6

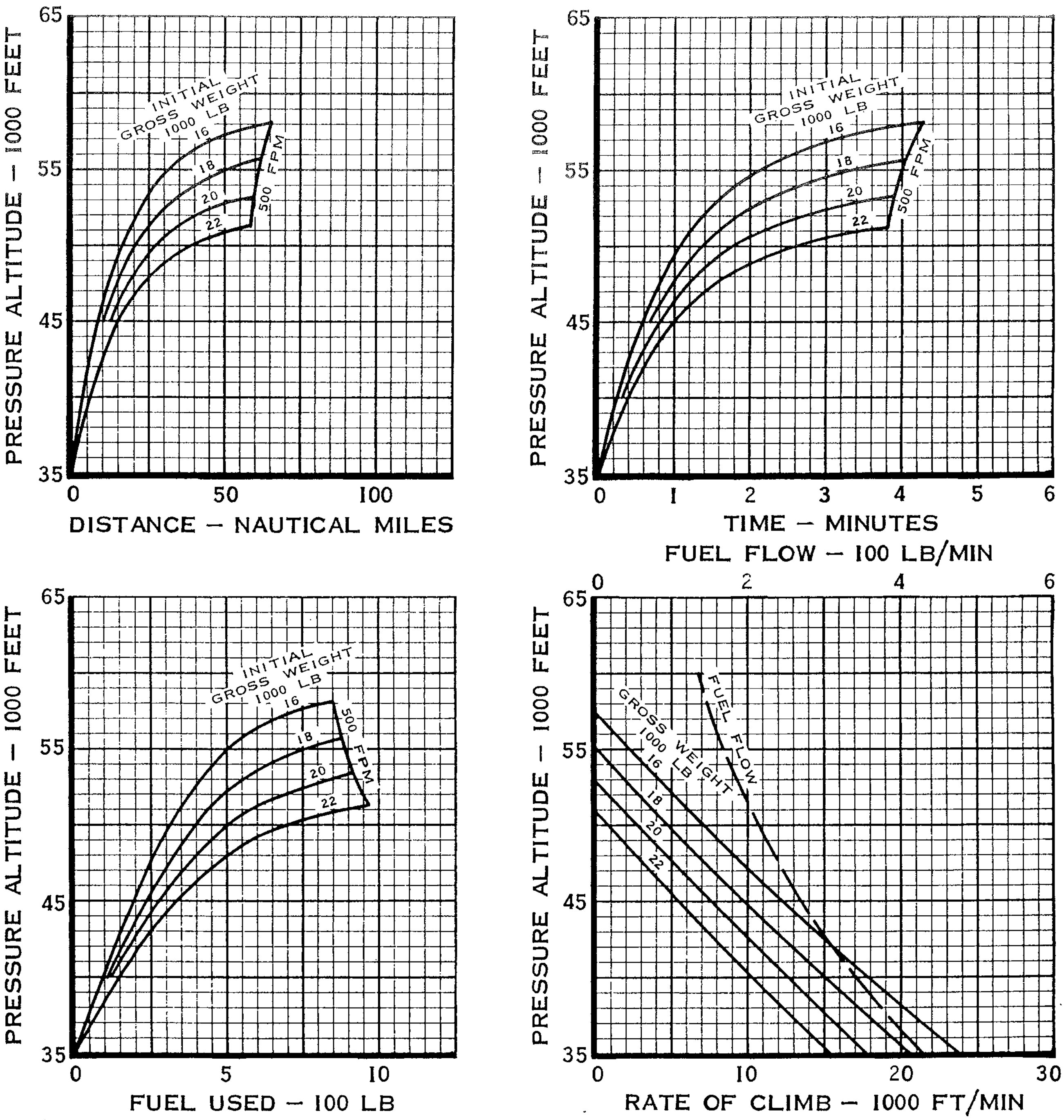


NO EXTERNAL STORES

Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY

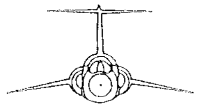


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 1000 LB PER 10°F BELOW STANDARD.

Figure A9-11

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.7

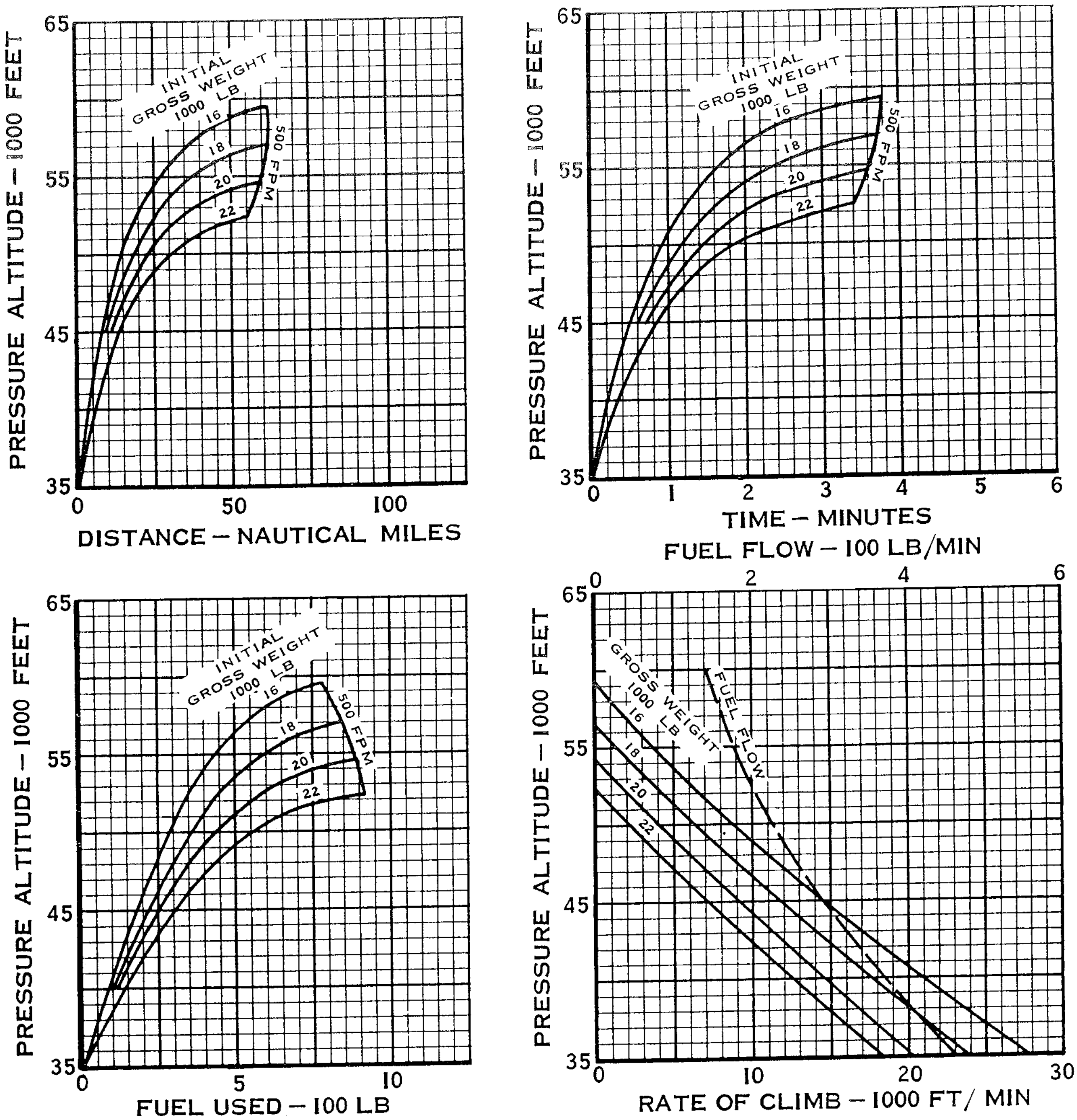


NO EXTERNAL STORES

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



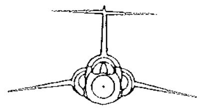
NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10 °F ABOVE STANDARD. SUBTRACT 1000 LB PER 10 °F BELOW STANDARD.

Figure A9-12

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.8

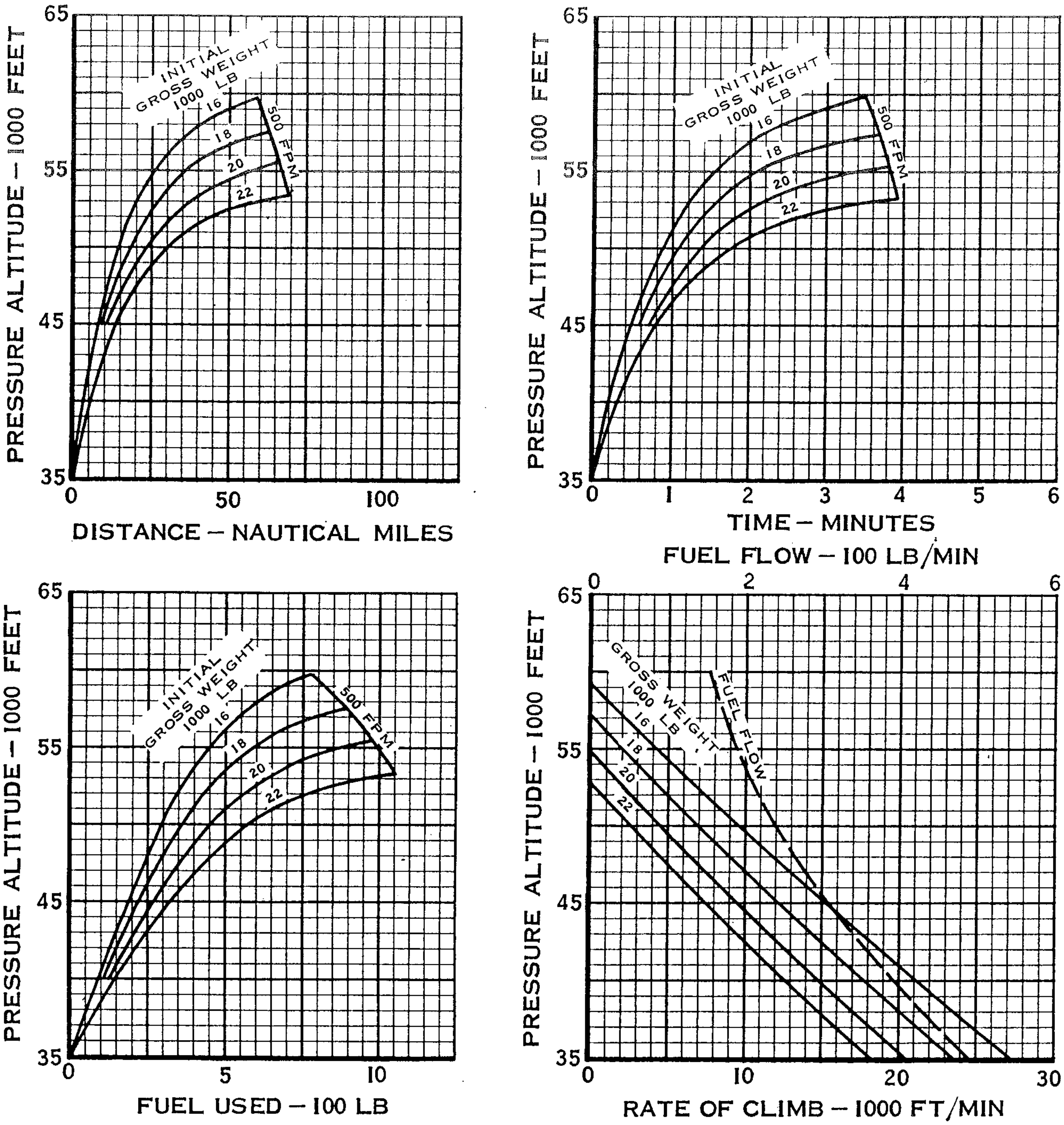
NO EXTERNAL STORES



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10 °F ABOVE STANDARD. SUBTRACT 1000 LB PER 10 °F BELOW STANDARD.

Figure A9-13

MAXIMUM THRUST CLIMB CONTROL

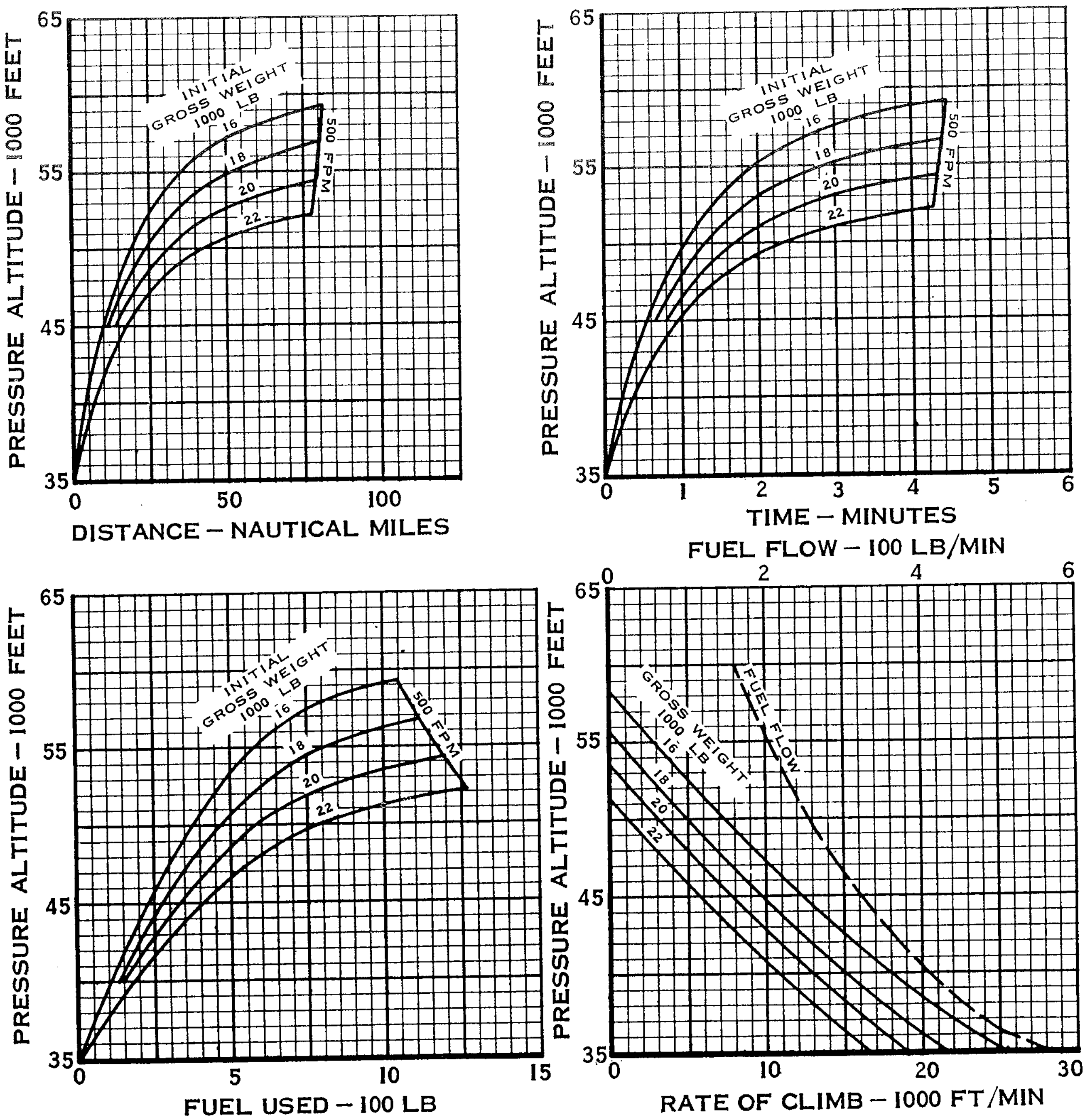
CLIMB SPEED MACH 1.9

NO EXTERNAL STORES

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY

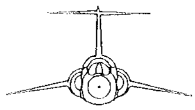


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10 °F ABOVE STANDARD. SUBTRACT 1000 LB PER 10 °F BELOW STANDARD.

Figure A9-14

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 2.0

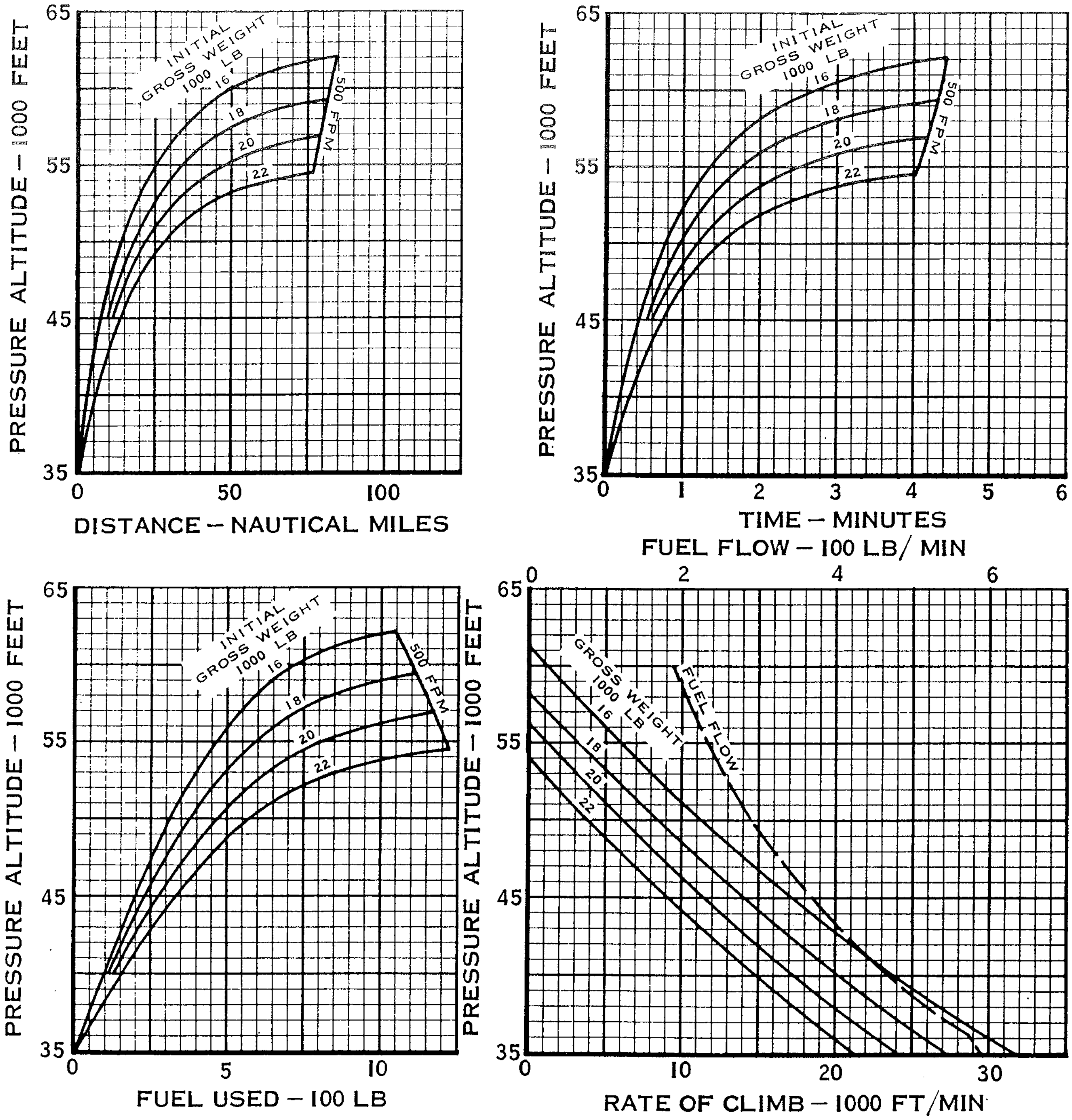


NO EXTERNAL STORES

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



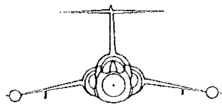
NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10 °F ABOVE STANDARD. SUBTRACT 1000 LB PER 10 °F BELOW STANDARD.

Figure A9-15

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.7

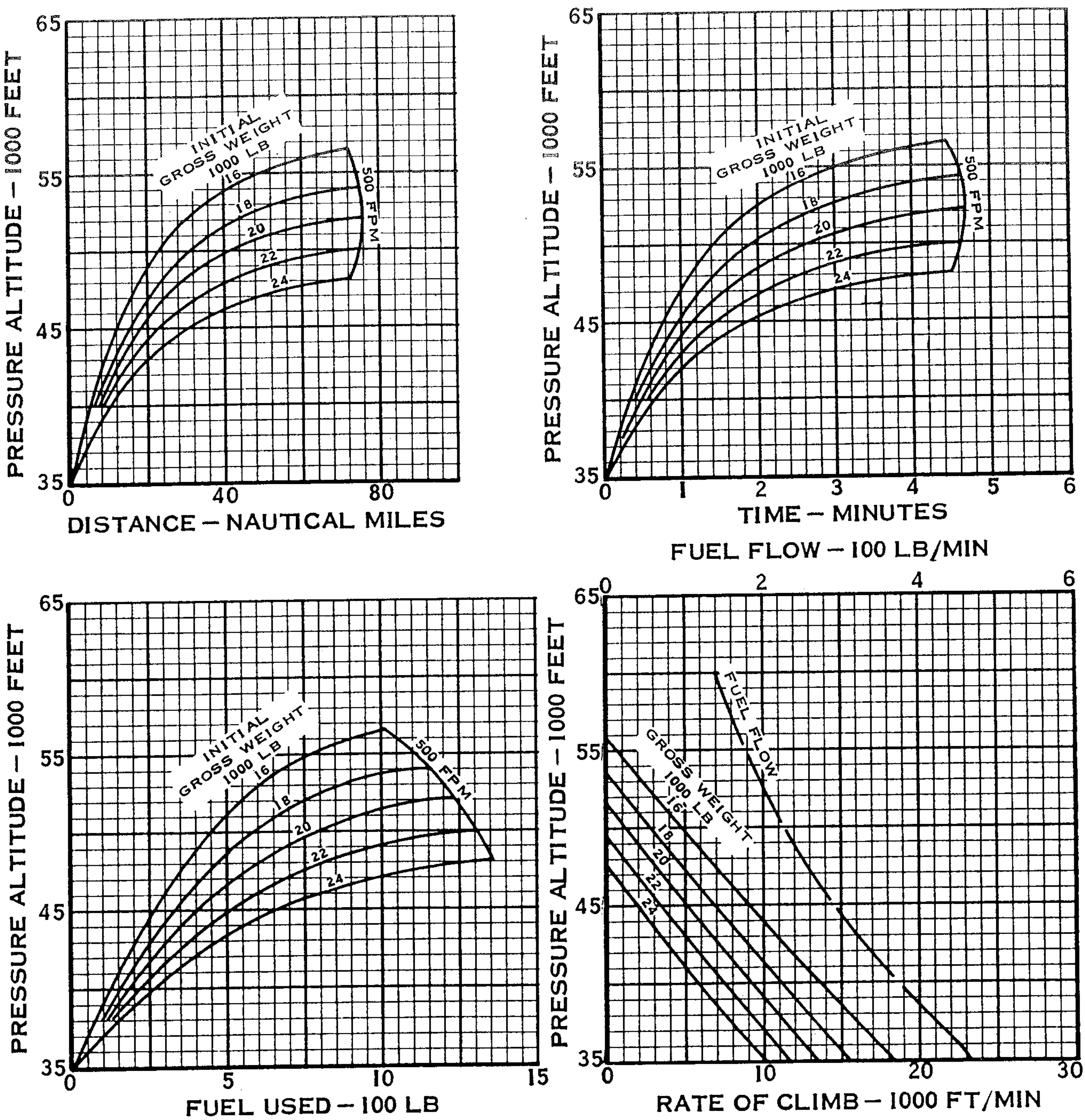
TIP TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



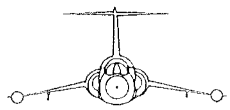
NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1000 LB TO AIRCRAFT GROSS WEIGHT PER 10 °F ABOVE STANDARD. SUBTRACT 1000 LB PER 10 °F BELOW STANDARD.

Figure A9-16

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.8

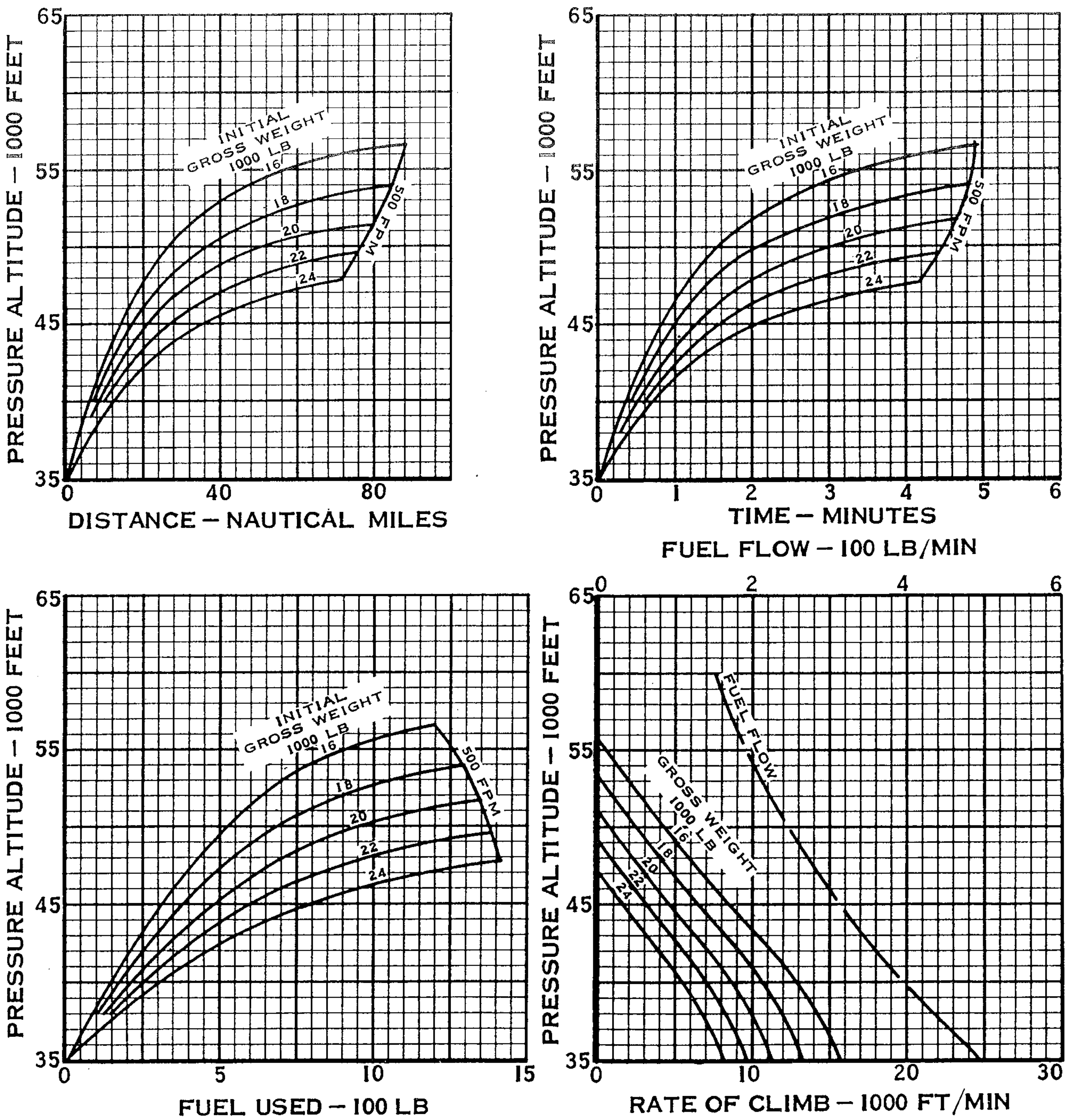
TIP TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 1500 LB PER 10°F BELOW STANDARD.

Figure A9-17

MAXIMUM THRUST CLIMB CONTROL

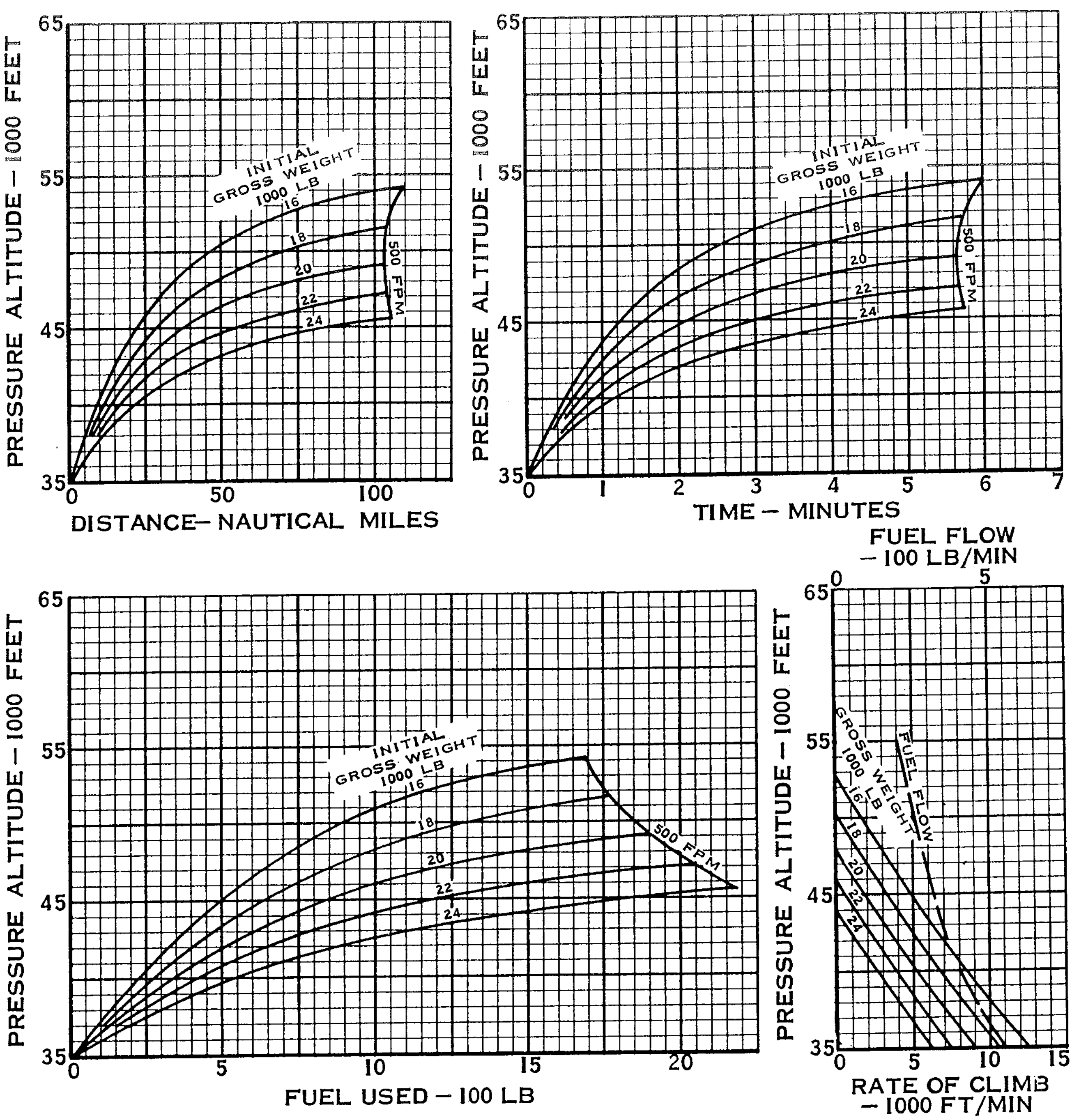
CLIMB SPEED MACH 1.9

TIP TANKS

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY

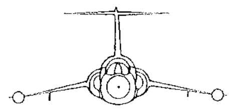


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 1500 LB PER 10°F BELOW STANDARD.

Figure A9-18

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 2.0

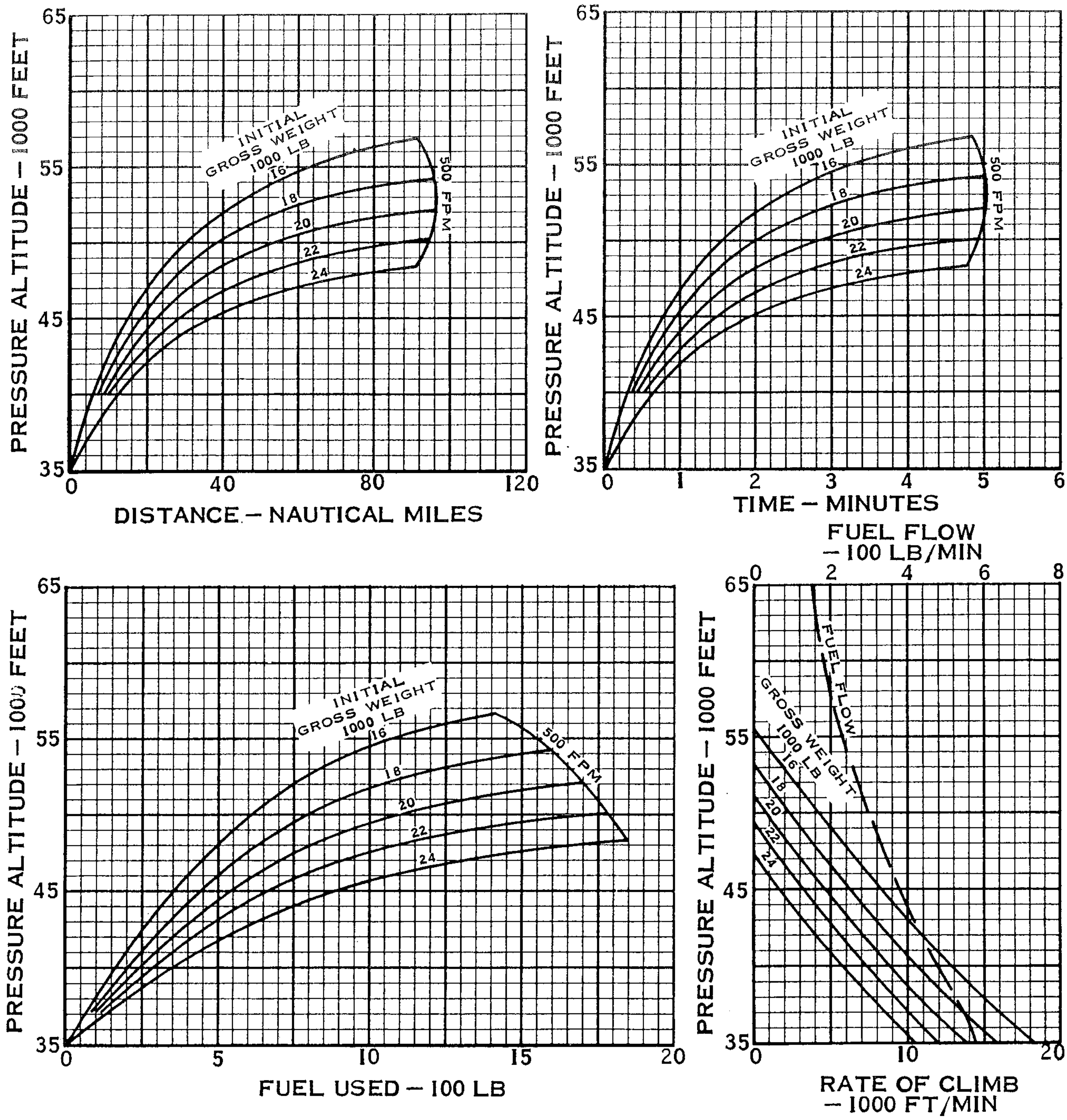


TIP TANKS

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY

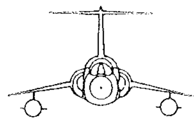


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 1500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 1500 LB PER 10°F BELOW STANDARD.

Figure A9-19

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.6

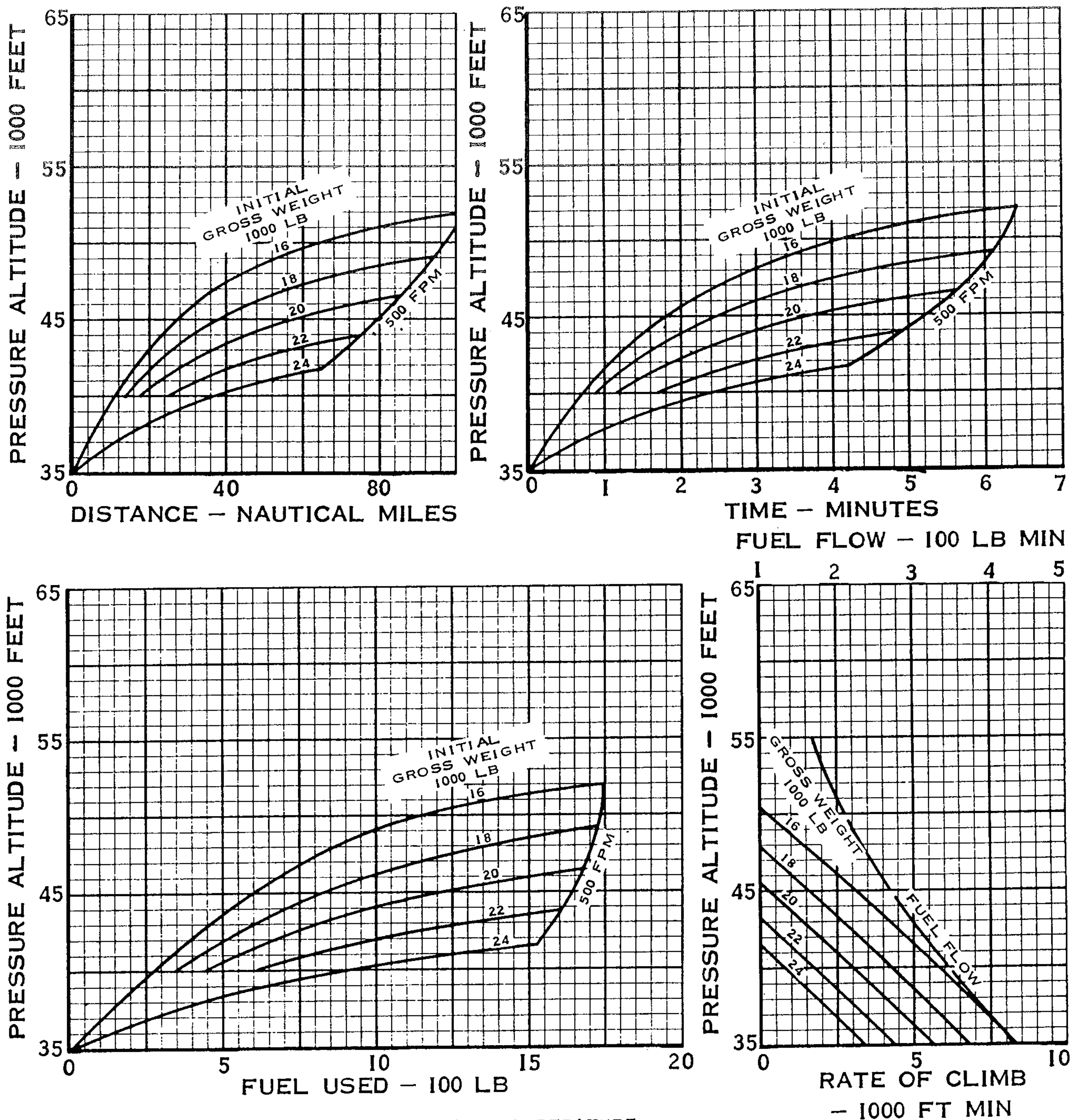


PYLON TANKS

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY

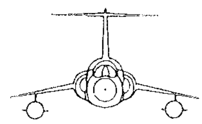


NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 2000 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 2000 LB PER 10°F BELOW STANDARD.

Figure A9-20

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.7

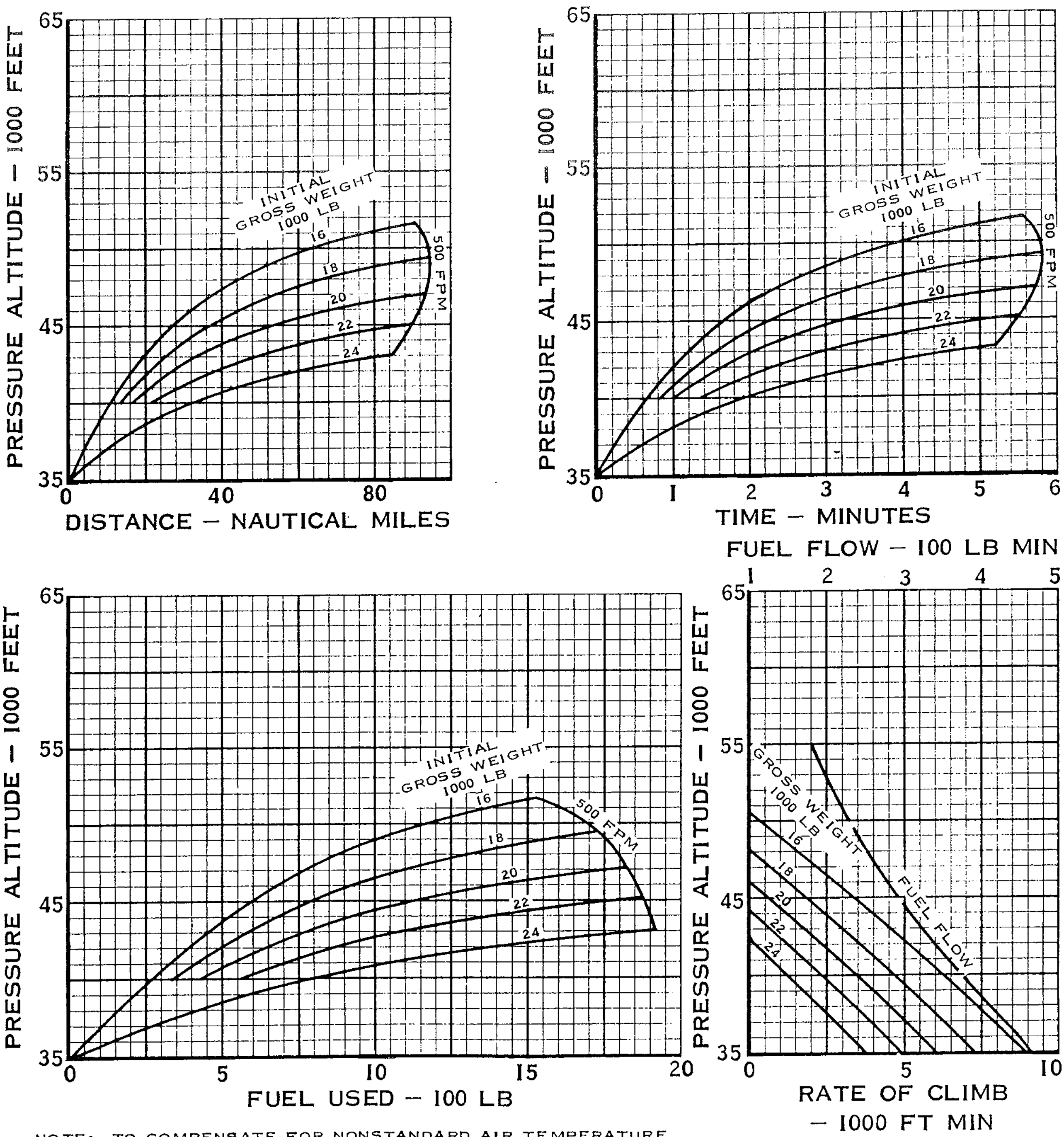


PYLON TANKS

Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



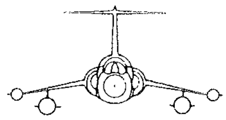
NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 2500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 2500 LB PER 10°F BELOW STANDARD.

Figure A9-21

MAXIMUM THRUST CLIMB CONTROL

CLIMB SPEED MACH 1.6

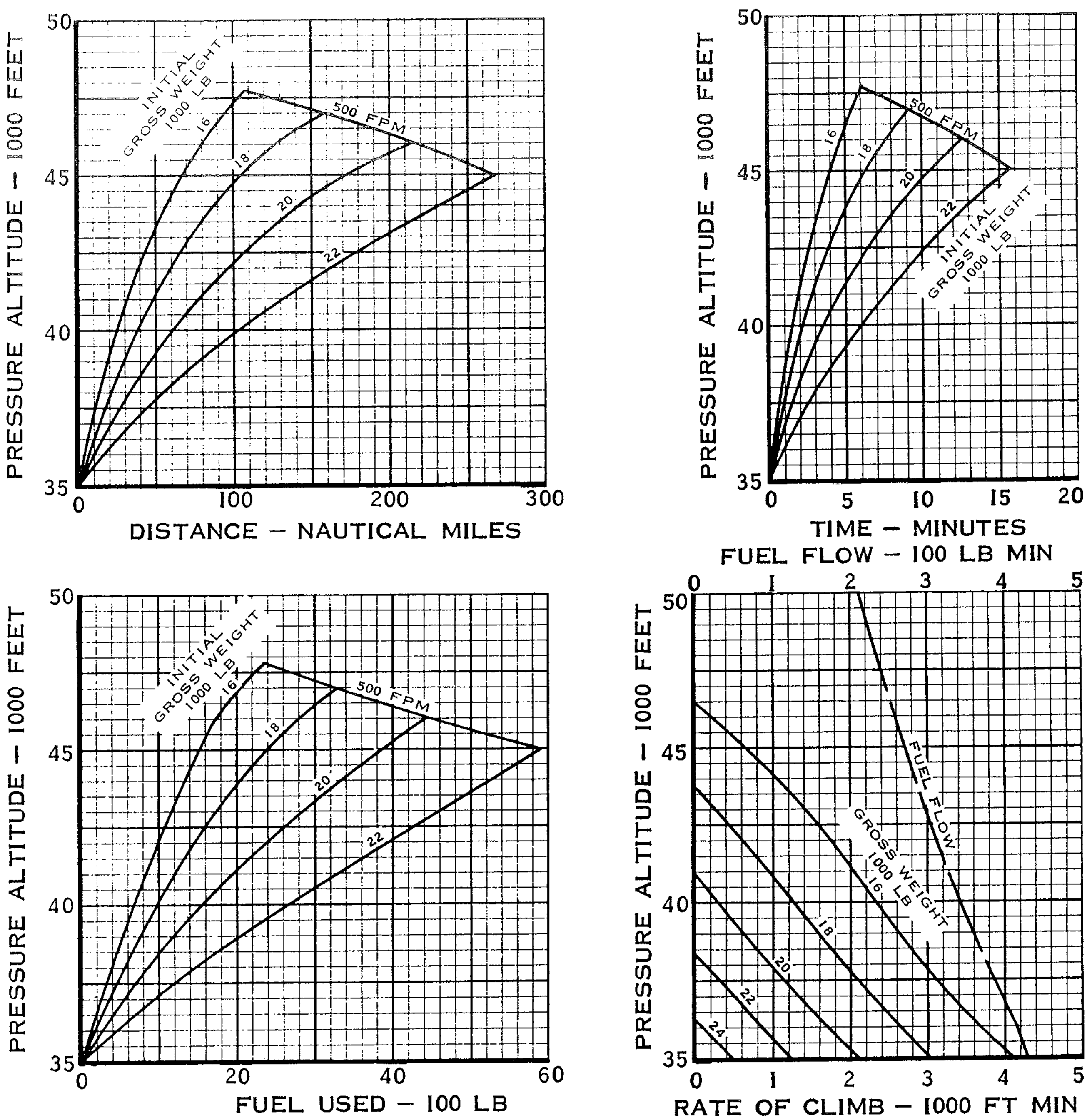
TIP TANKS AND PYLON TANKS



Model: F-104G
Date: 1 February 1963
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

STANDARD DAY



NOTE: TO COMPENSATE FOR NONSTANDARD AIR TEMPERATURE ADD 2500 LB TO AIRCRAFT GROSS WEIGHT PER 10°F ABOVE STANDARD. SUBTRACT 2500 LB PER 10°F BELOW STANDARD.

Figure A9-22

MAXIMUM THRUST TURNING PERFORMANCE
CONSTANT SPEED AND ALTITUDE
NO EXTERNAL STORES

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

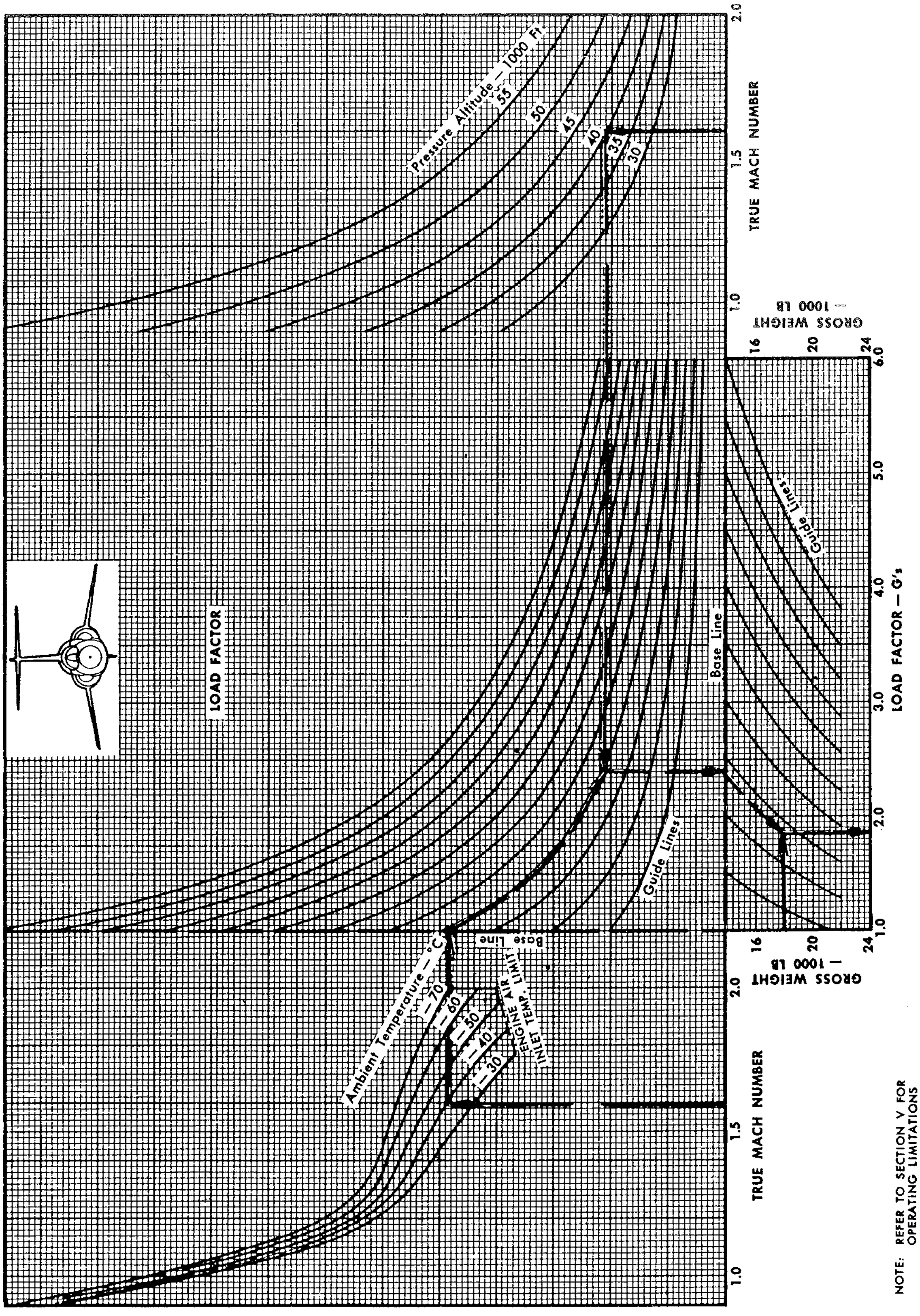


Figure A9-23

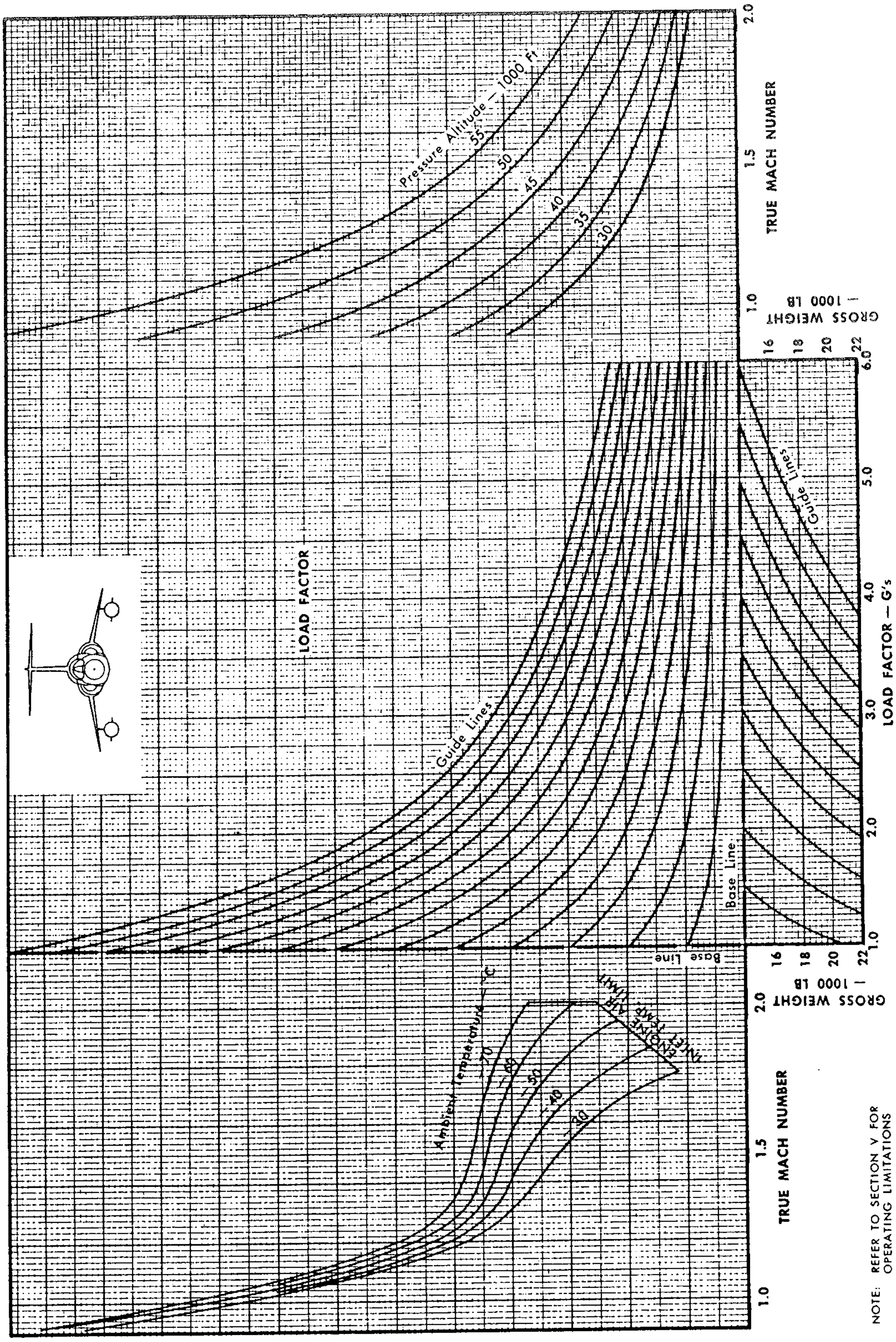
MAXIMUM THRUST TURNING PERFORMANCE

CONSTANT SPEED AND ALTITUDE

PYLON TANKS

Model: F-104G
Date: 1 June 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE: REFER TO SECTION V FOR OPERATING LIMITATIONS

Figure A9-24

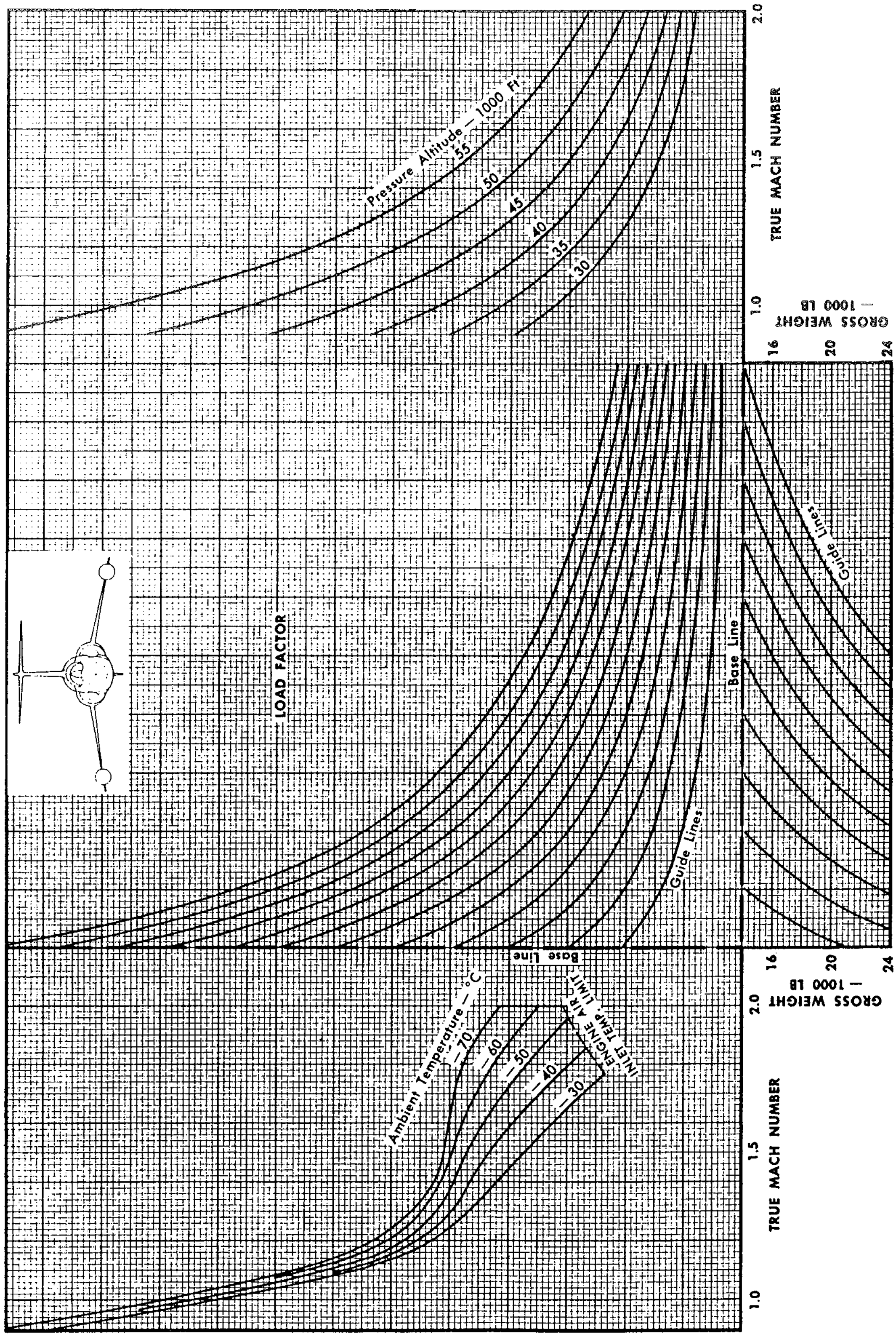
MAXIMUM THRUST TURNING PERFORMANCE

CONSTANT SPEED AND ALTITUDE

TIP TANKS

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE: REFER TO SECTION V FOR OPERATING LIMITATIONS

Figure A9-25

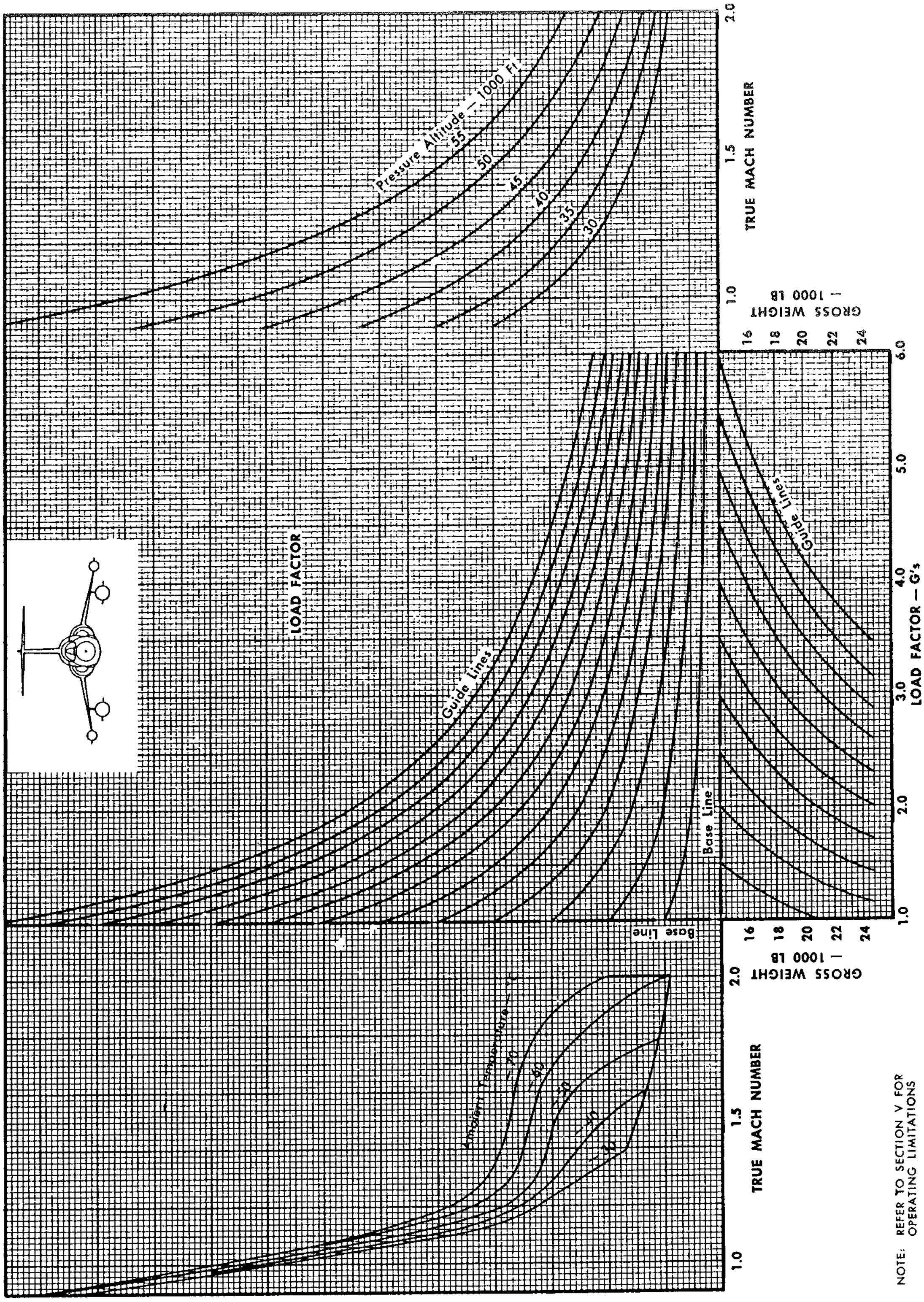
MAXIMUM THRUST TURNING PERFORMANCE

CONSTANT SPEED AND ALTITUDE

TIP TANKS AND PYLON TANKS

Model: F-104G
Date: 1 June 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE: REFER TO SECTION V FOR OPERATING LIMITATIONS

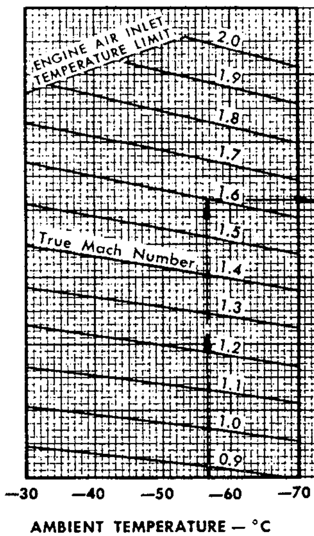
Figure A9-26

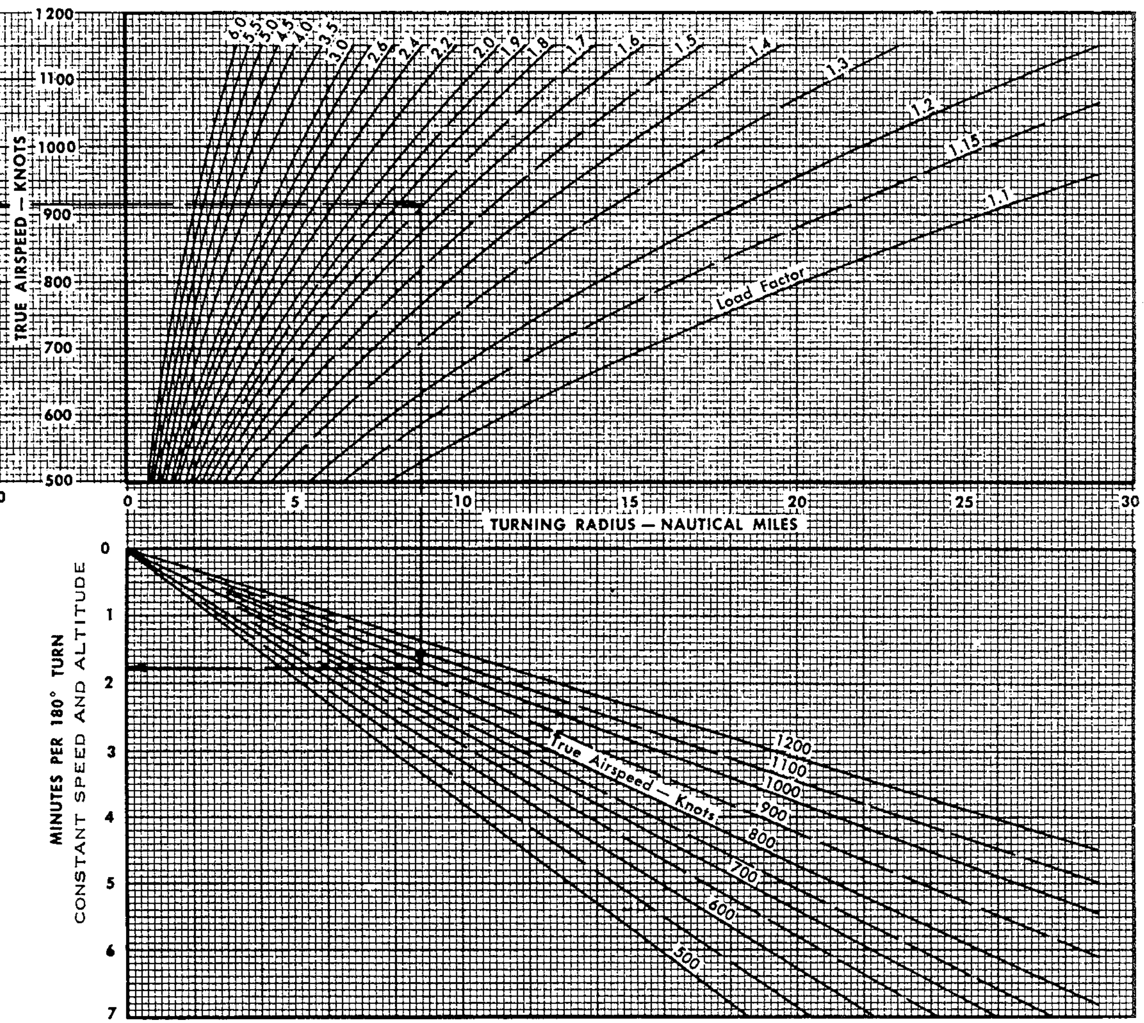
TURNING PERFORMANCE — GENERAL

ALL CONFIGURATIONS

Bank Angle — Minutes per 180° Turn — Turn Radius

CONSTANT SPEED AND ALTITUDE





BANK ANGLE	LOAD FACTOR — G's
20°	1.07
30°	1.16
40°	1.31
50°	1.56
60°	2.00
70°	2.93
80°	5.76

NOTE: REFER TO SECTION V FOR OPERATING LIMITATIONS

Figure A9-27

TURNING PERFORMANCE — GENERAL

LOW ALTITUDE ALL CONFIGURATIONS

Bank Angle Minutes per 180° Turn Turn Radius

CONSTANT SPEED AND ALTITUDE

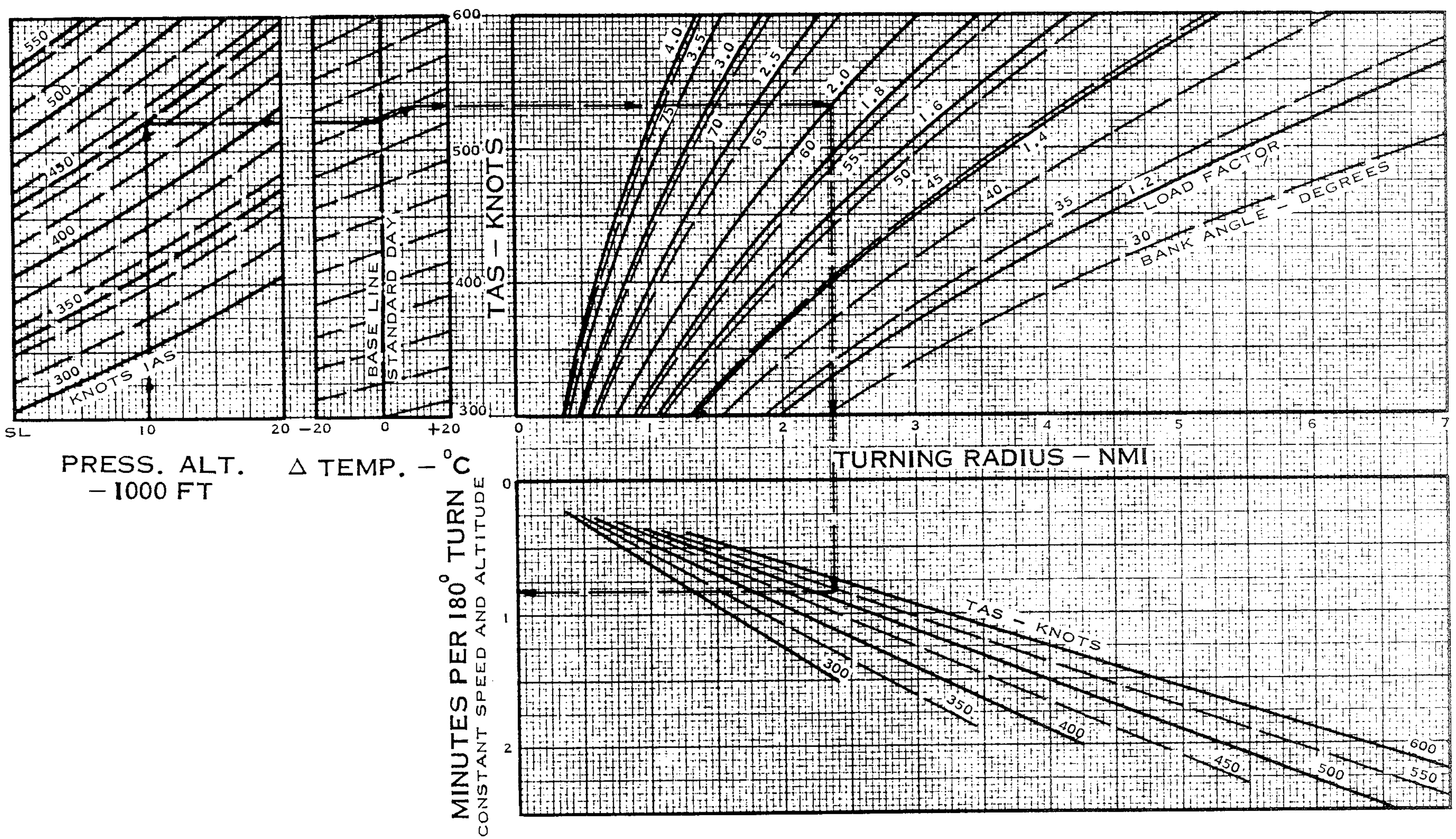


Figure A9-28

NOTE: REFER TO SECTION V FOR OPERATING LIMITATIONS

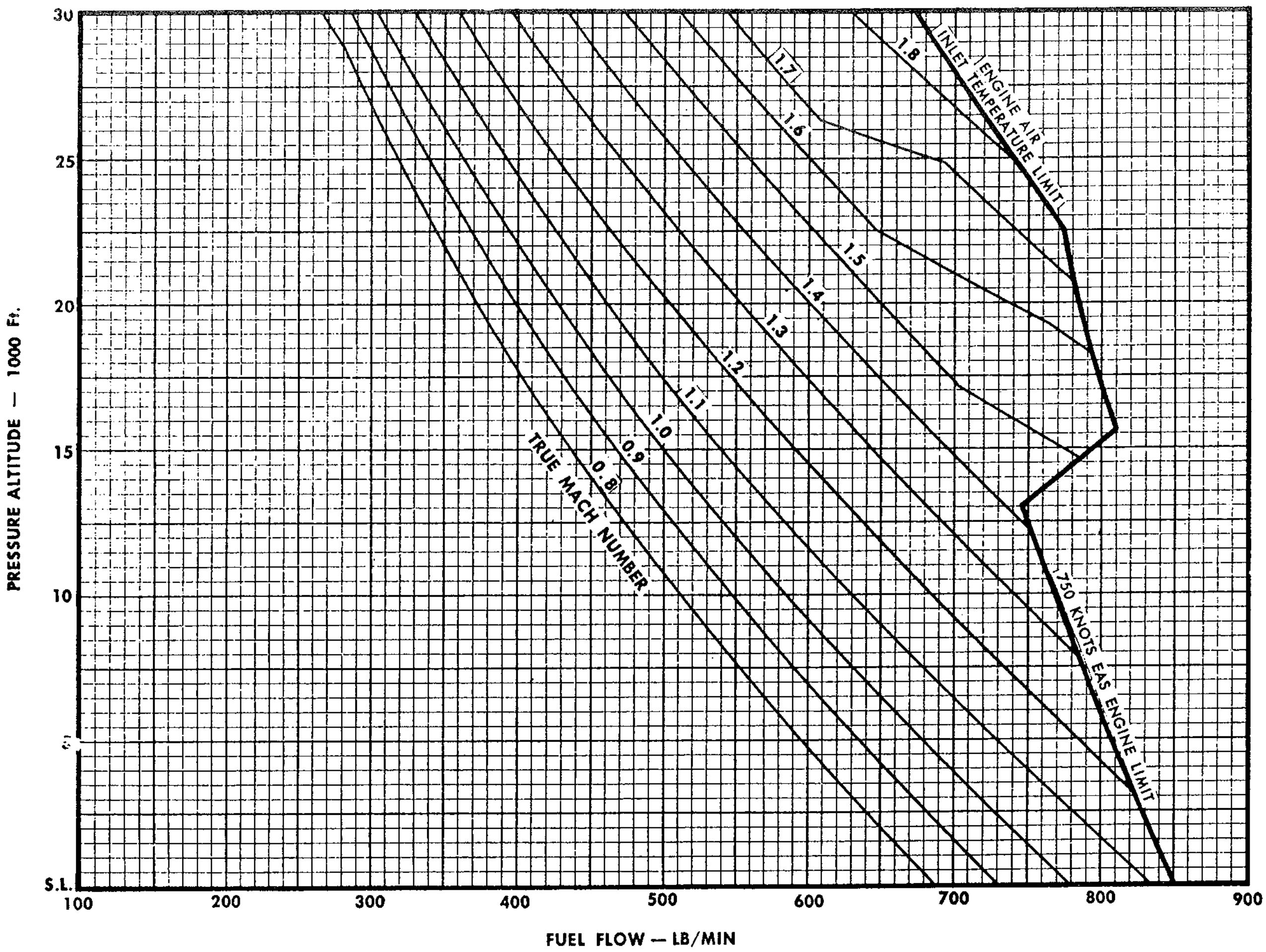
MAXIMUM THRUST FUEL CONSUMPTION

SEA LEVEL TO 30000 FT

Standard Day

Model: F-104G
Date: 1 February 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



NOTE:
1. RATIO OF TOTAL FUEL FLOW RATE TO ENGINE FUEL FLOW RATE: 4 TO 1
2. DATA ARE FOR MAXIMUM THRUST, NOT THROTTLED FOR STRAIGHT AND LEVEL FLIGHT.
3. REFER TO SECTION V FOR OPERATING LIMITATIONS

Figure A9-29

MAXIMUM THRUST FUEL CONSUMPTION

30000 FT TO 60000 FT

Model: F-104G
Date: 1 June 1964
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

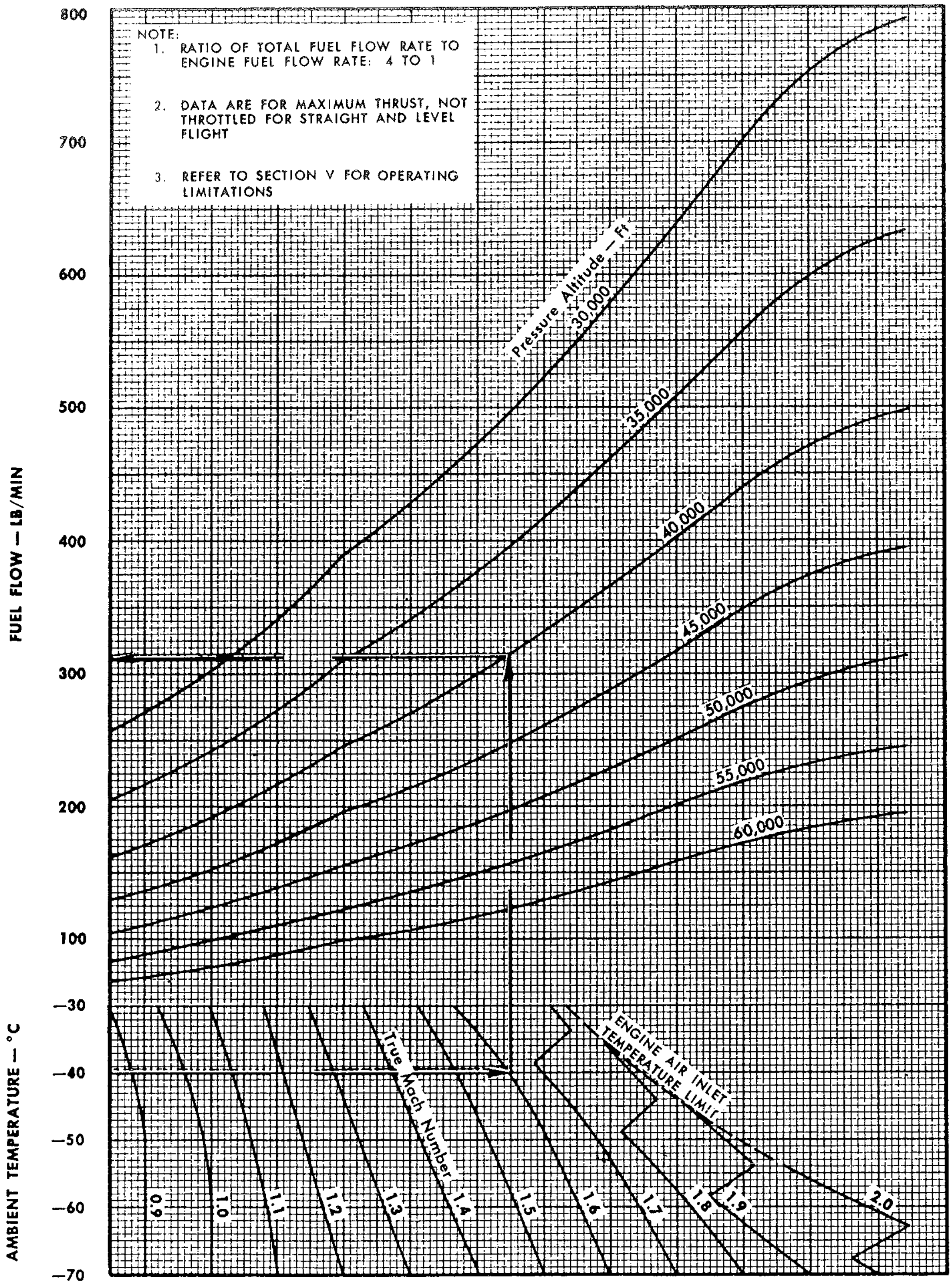


Figure A9-30

MILITARY THRUST FUEL CONSUMPTION

SEA LEVEL TO 35000 FT

Model: F-104G
Date: 1 October 1965
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

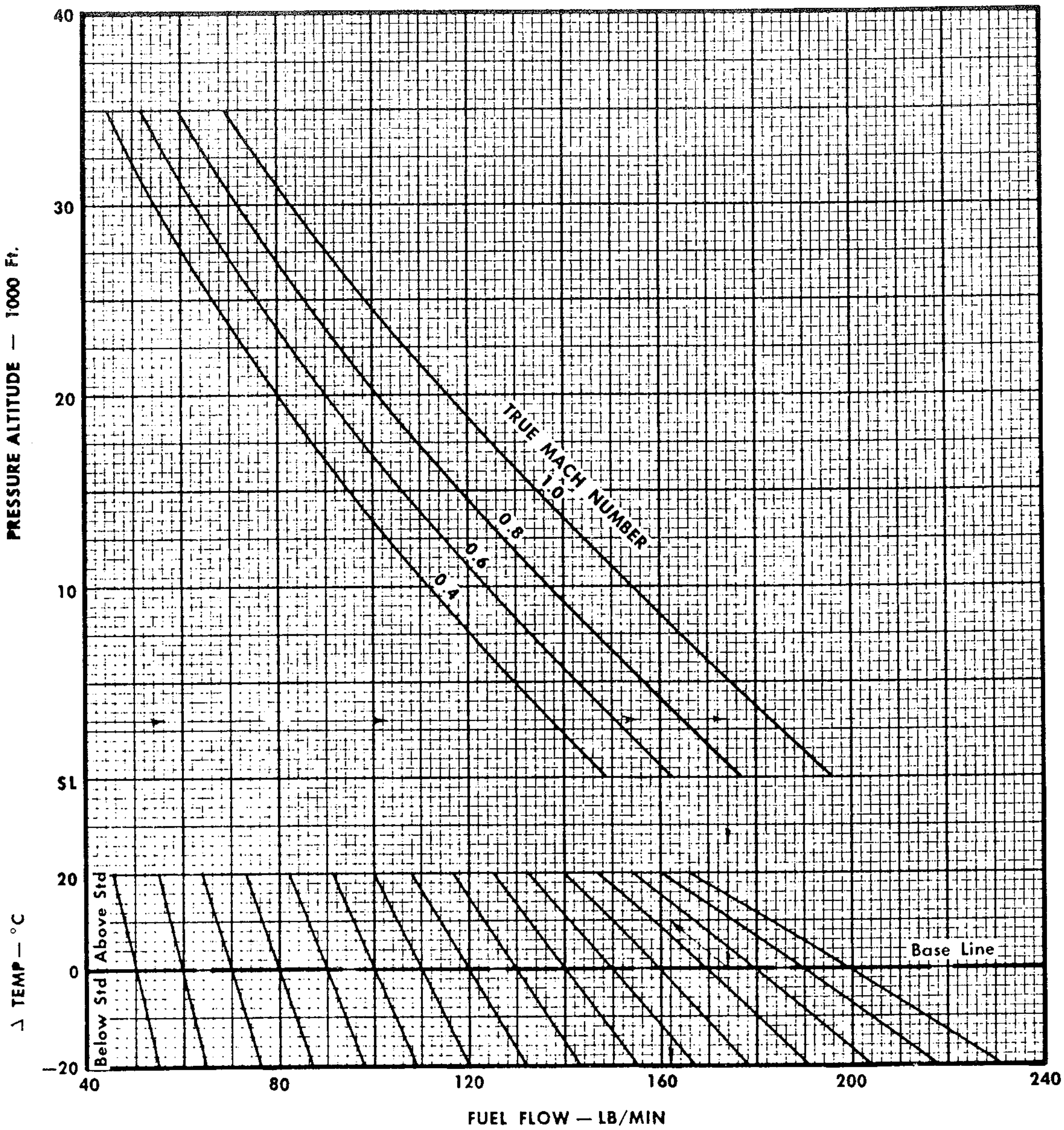
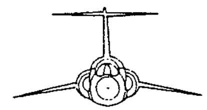


Figure A9-31

LOW ALTITUDE
MILITARY THRUST LEVEL FLIGHT ACCELERATION
FROM 400 KNOTS IAS



Model: F-104G
Date: 15 February 1964
DATA BASIS: ESTIMATED

DI 30

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

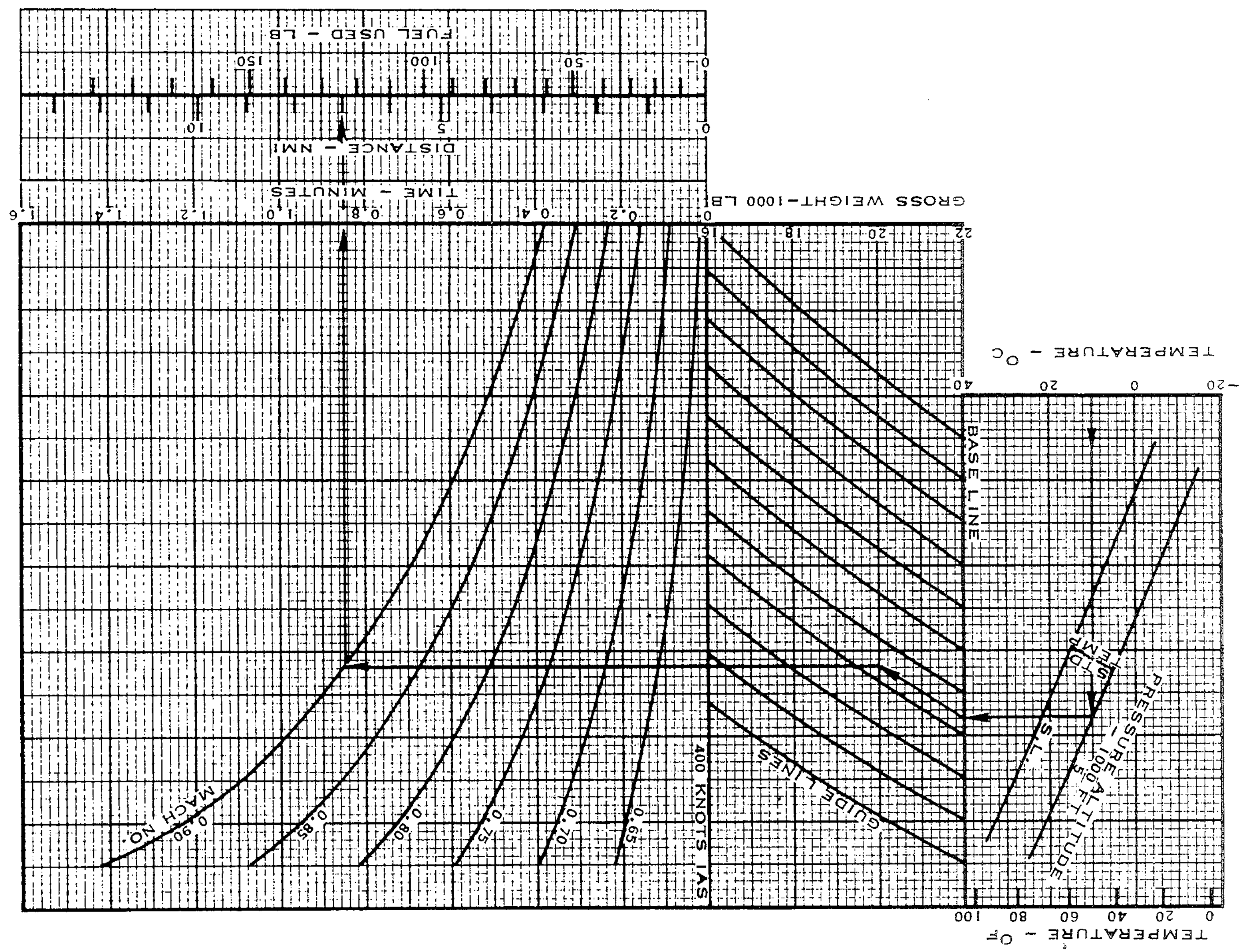


Figure A9-32

LOW ALTITUDE
MILITARY THRUST LEVEL FLIGHT ACCELERATION
FROM 400 KNOTS IAS

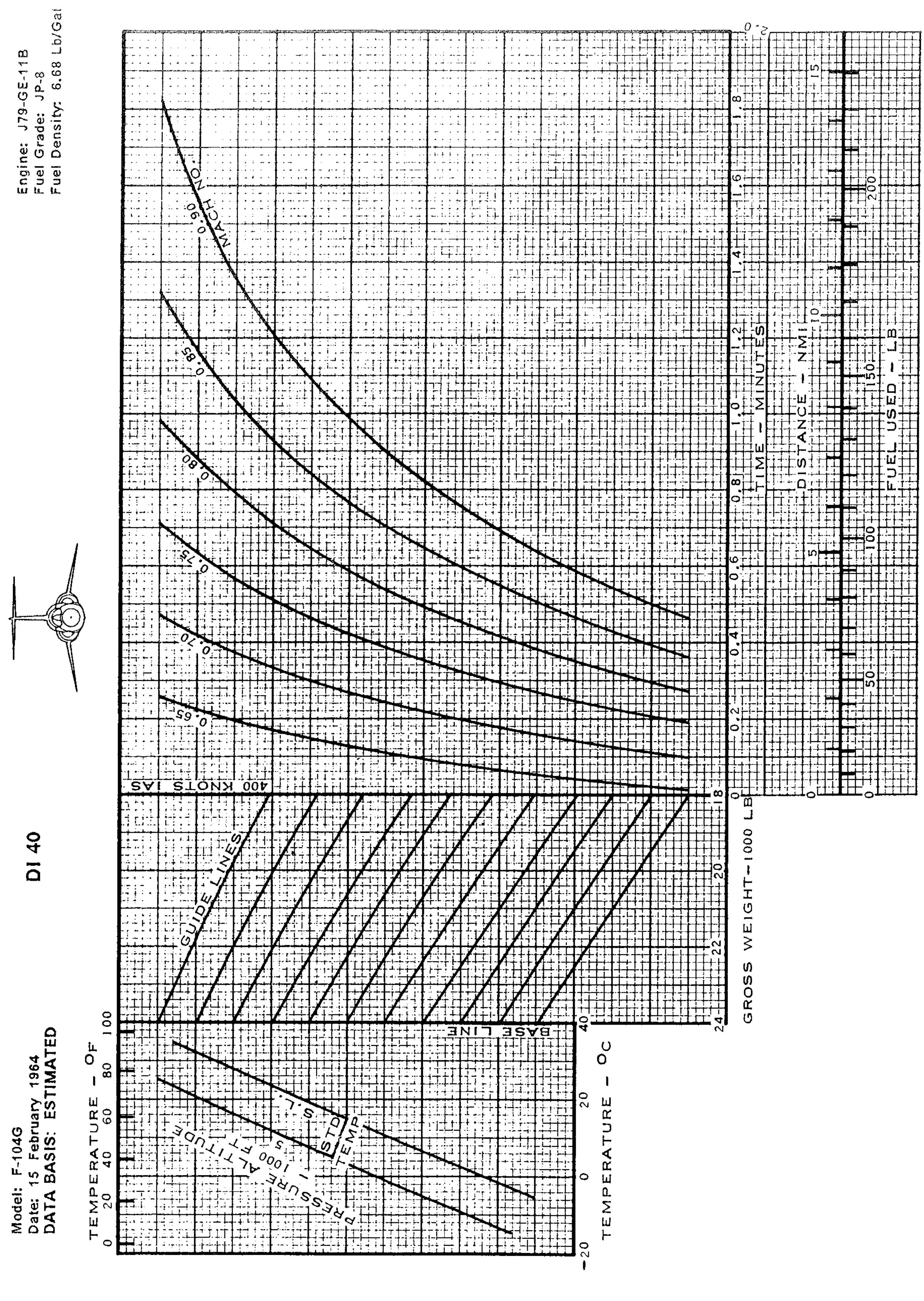
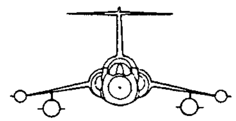


Figure A9-33

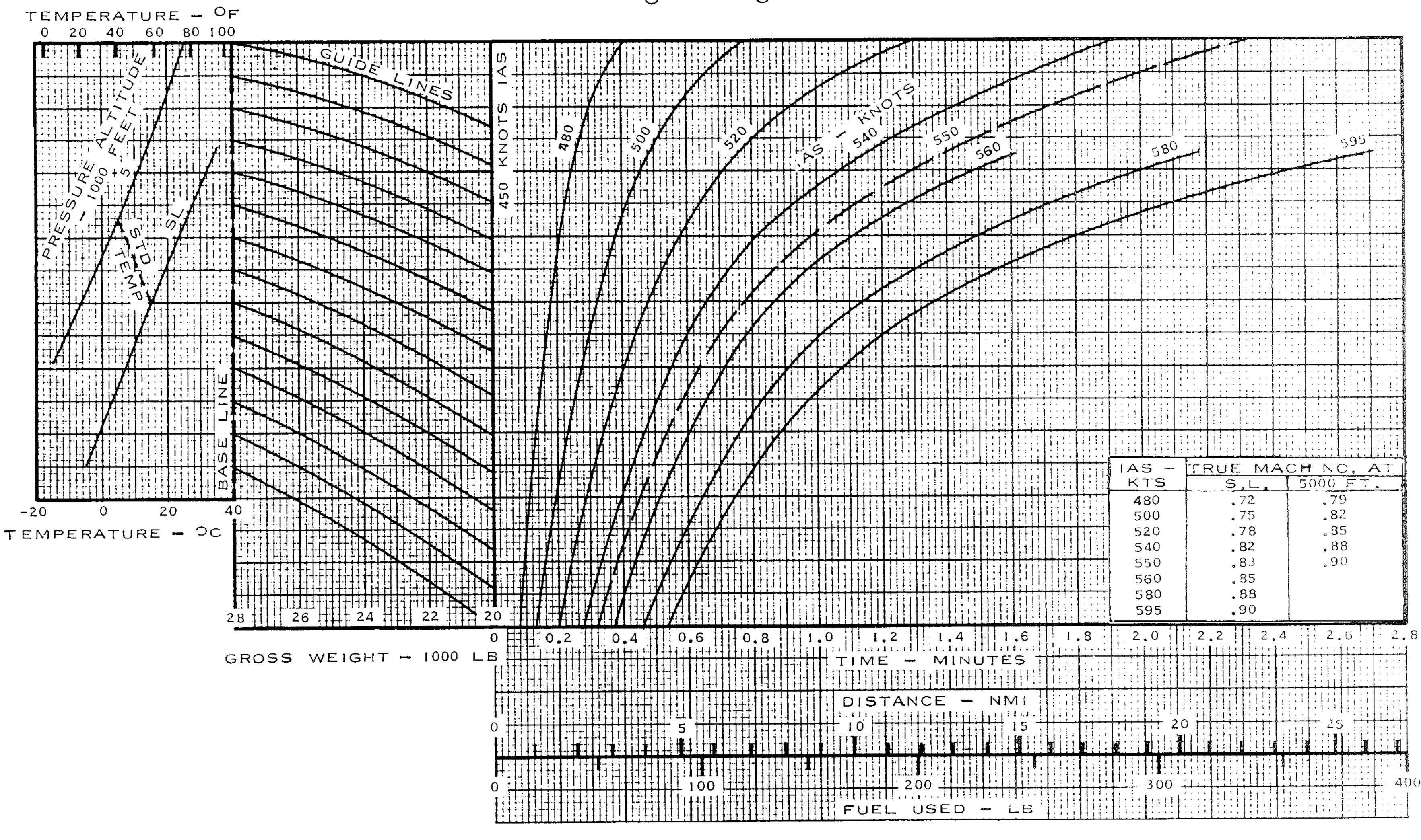
LOW ALTITUDE
MILITARY THRUST LEVEL FLIGHT ACCELERATION
FROM 450 KNOTS IAS

Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

DI 75



Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

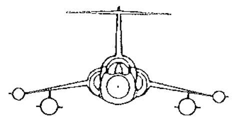


IAS — KTS	TRUE MACH NO. AT S.L.	TRUE MACH NO. AT 5000 FT.
480	.72	.79
500	.75	.82
520	.78	.85
540	.82	.88
550	.83	.90
560	.85	
580	.88	
595	.90	

Figure A9-34

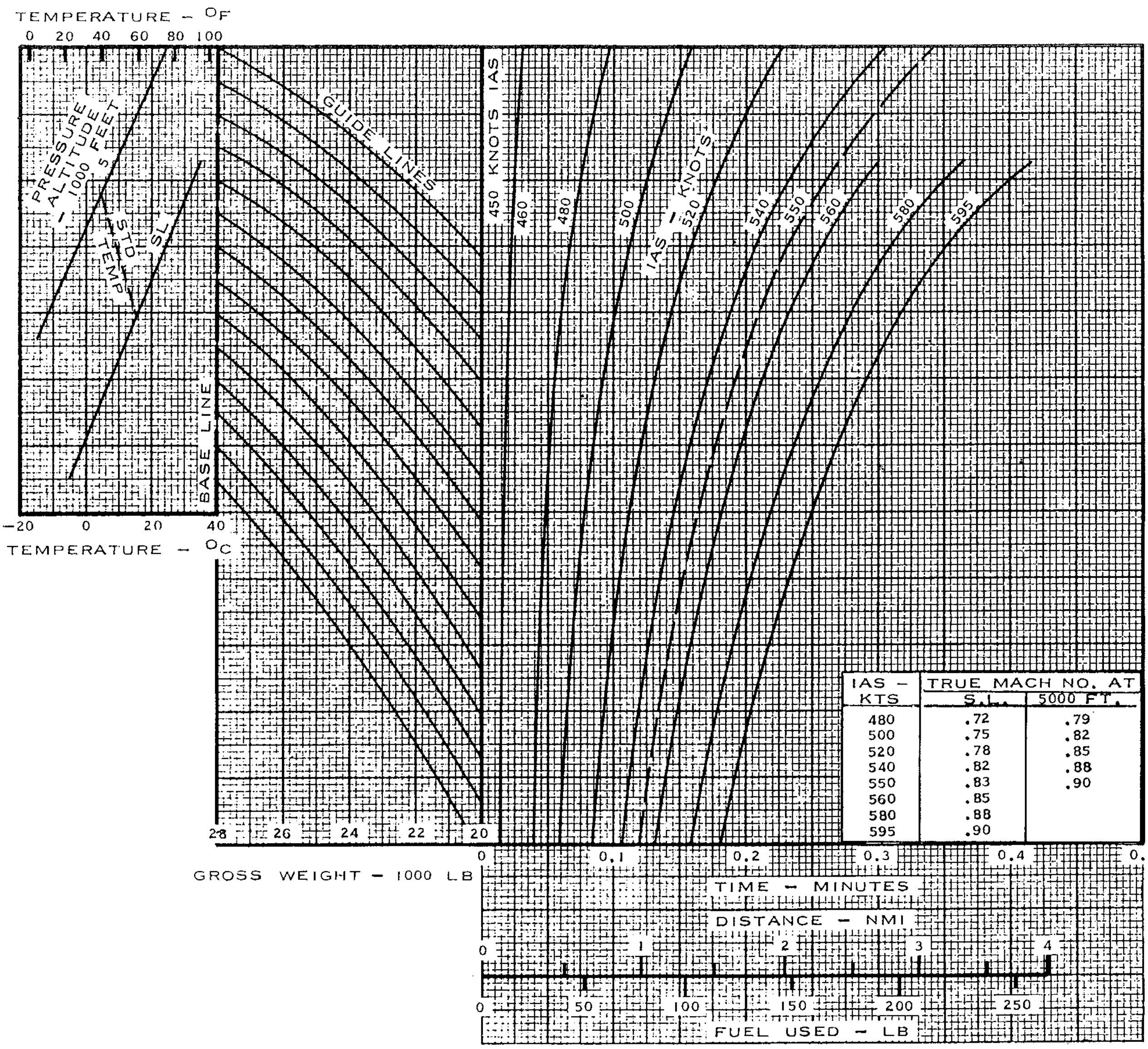
LOW ALTITUDE
MAXIMUM THRUST LEVEL FLIGHT ACCELERATION
FROM 450 KNOTS IAS

DI 75



Model: F-104G
Date: 1 April 1966
DATA BASIS: FLIGHT TEST

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



IAS – KTS	TRUE MACH NO. AT S.L.	TRUE MACH NO. AT 5000 FT.
480	.72	.79
500	.75	.82
520	.78	.85
540	.82	.88
550	.83	.90
560	.85	
580	.88	
595	.90	

Figure A9-35

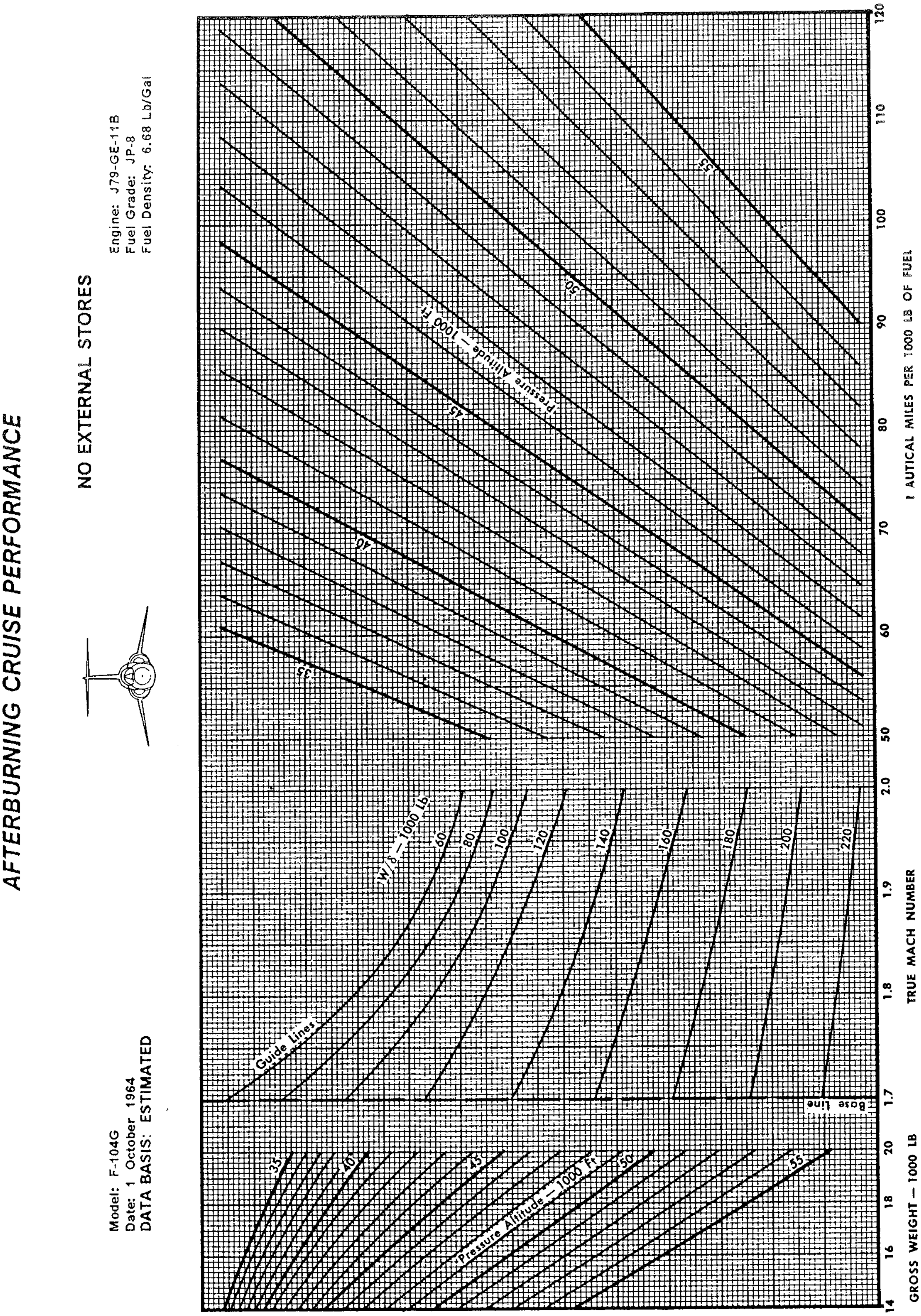
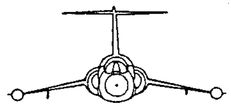

AFTERBURNING CRUISE PERFORMANCE
NO EXTERNAL STORES
Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED
Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal
Pressure Altitude — 1000 Ft
35
40
45
50
55
W/δ — 1000 Lb
60
80
100
120
140
160
180
200
220
Guide Lines
Base Line
GROSS WEIGHT — 1000 LB
14
16
18
20
TRUE MACH NUMBER
1.7
1.8
1.9
2.0
NAUTICAL MILES PER 1000 LB OF FUEL
50
60
70
80
90
100
110
120

Figure A9-36

AFTERBURNING CRUISE PERFORMANCE

TIP TANKS



Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

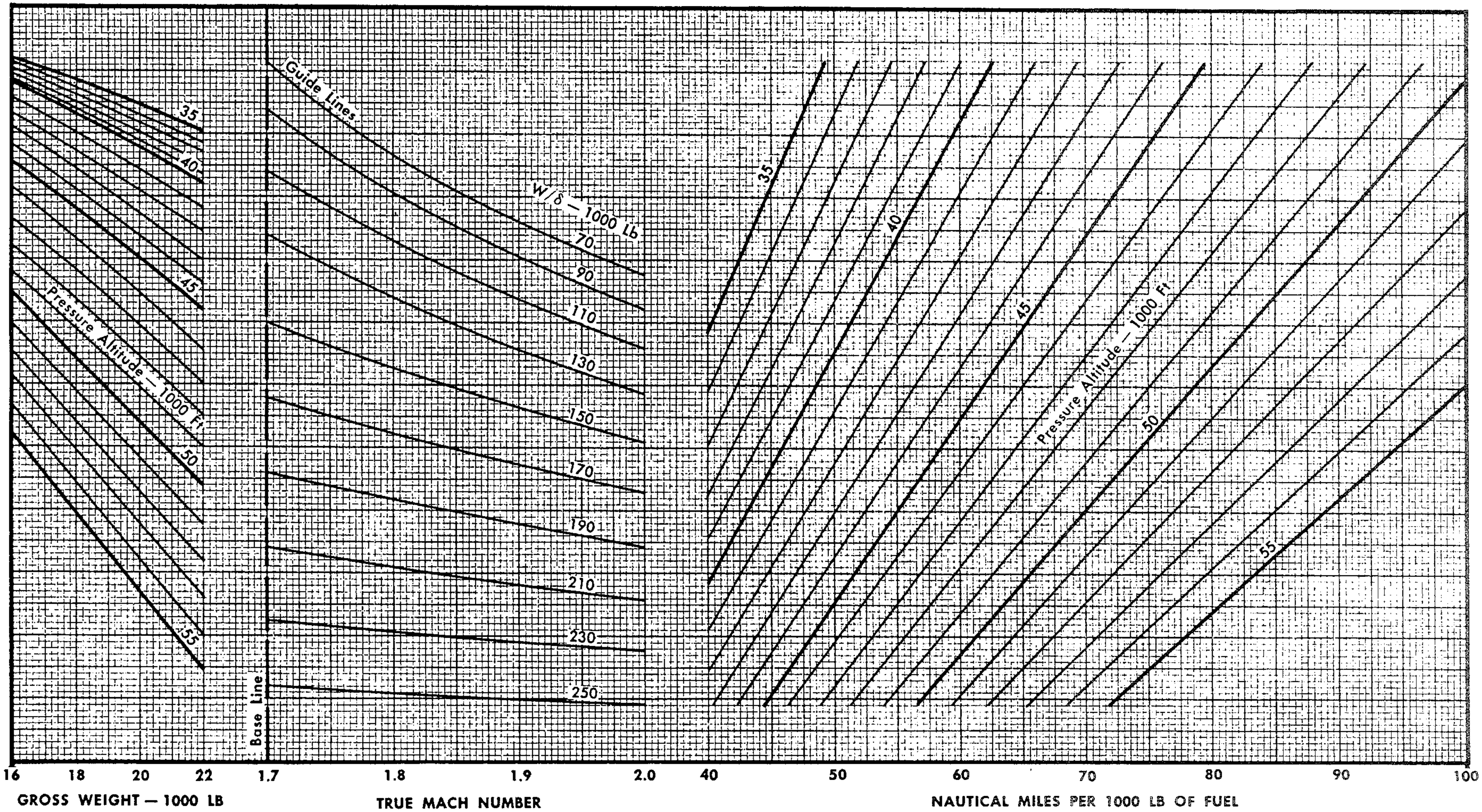
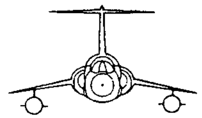


Figure A9-37

AFTERBURNING CRUISE PERFORMANCE

PYLON TANKS

Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED



Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

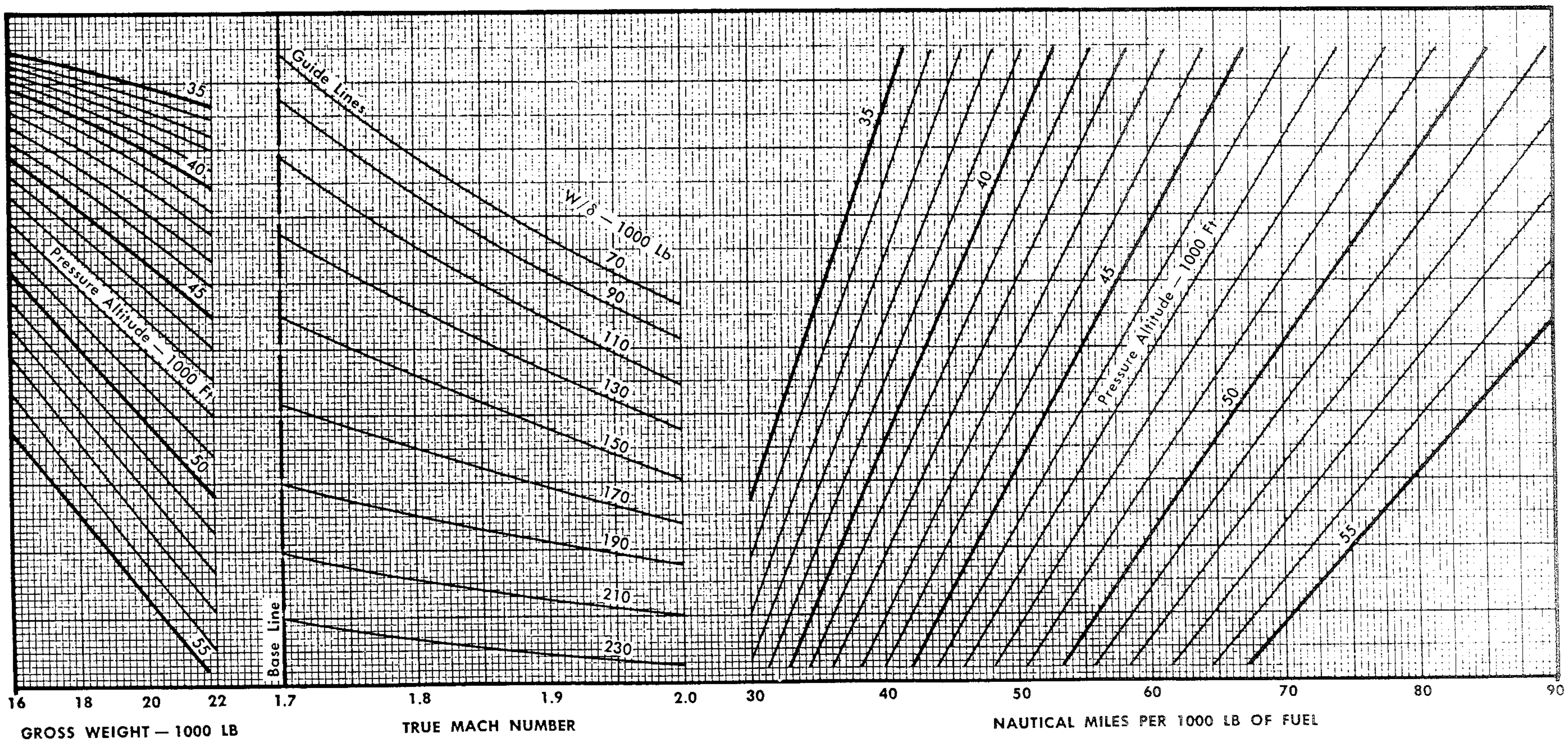
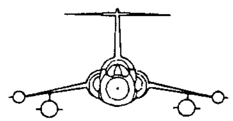


Figure A9-38

AFTERBURNING CRUISE PERFORMANCE

Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED



TIP TANKS AND
PYLON TANKS

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

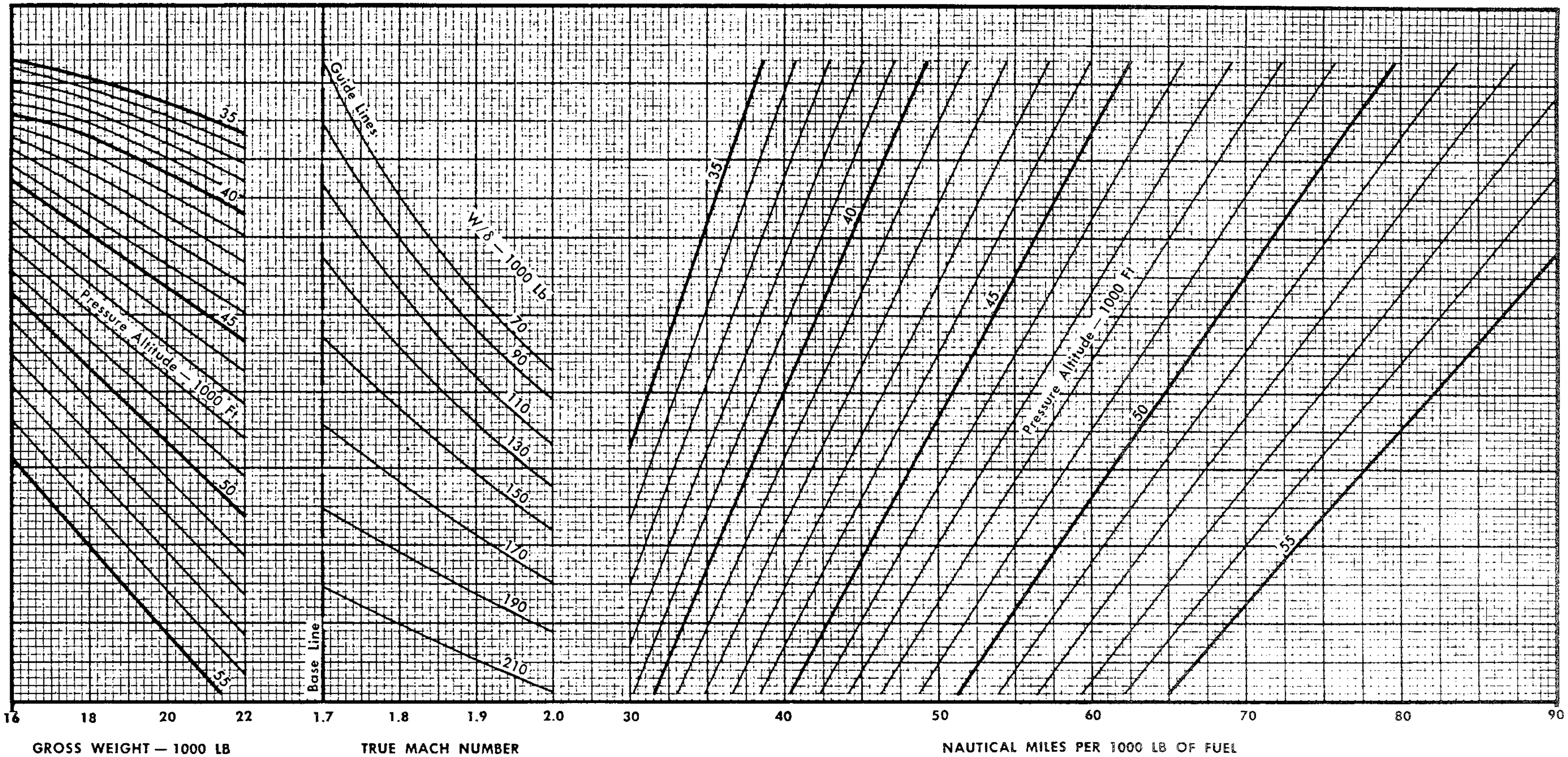
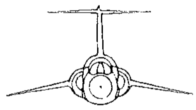


Figure A9-39

MAXIMUM THRUST FLIGHT ENVELOPE

NO EXTERNAL STORES

Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED



Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

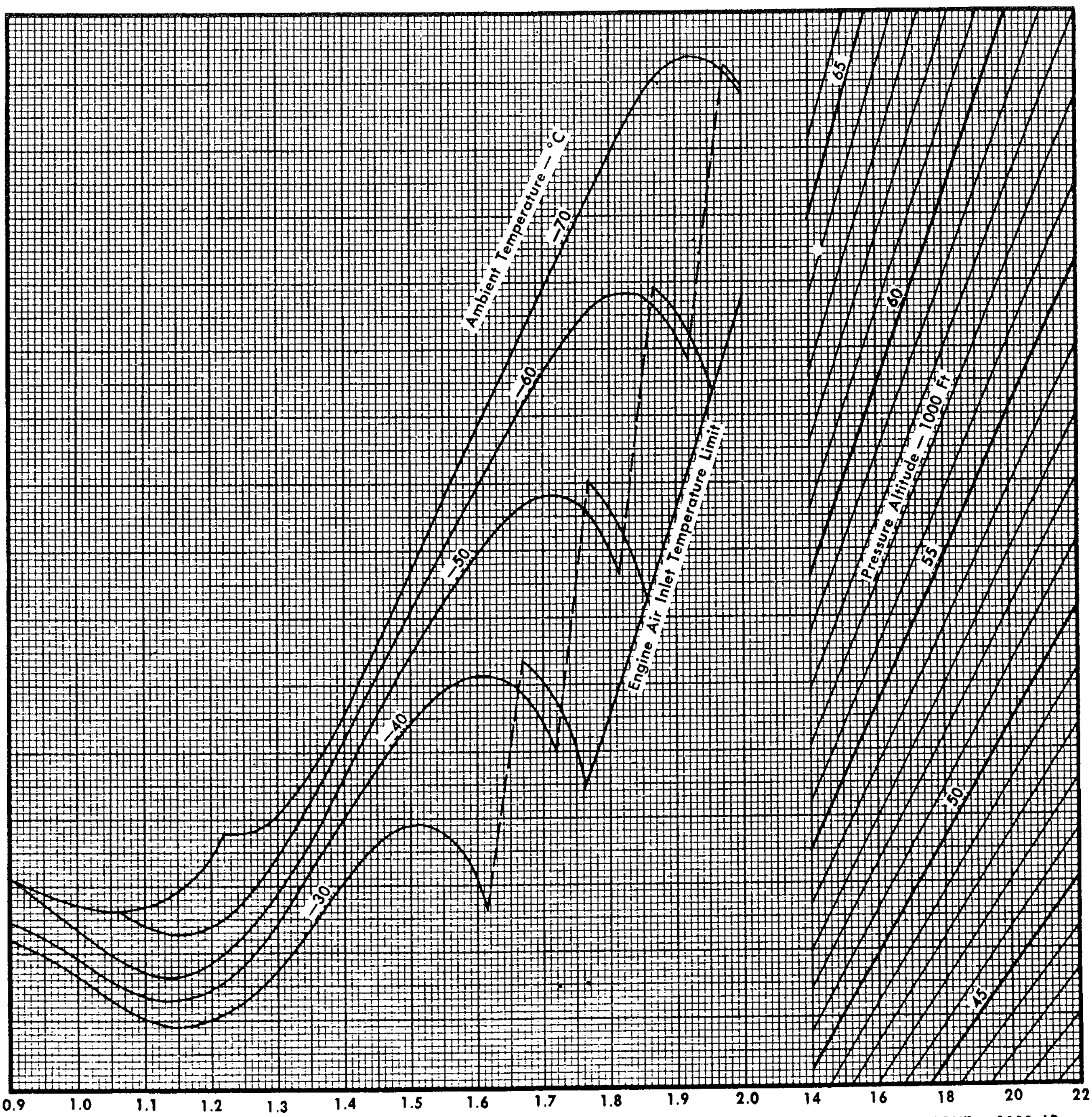


Figure A9-40

MAXIMUM THRUST FLIGHT ENVELOPE

TIP TANKS

Model: F-104G
Date: 1 September 1964
DATA BASIS: ESTIMATED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

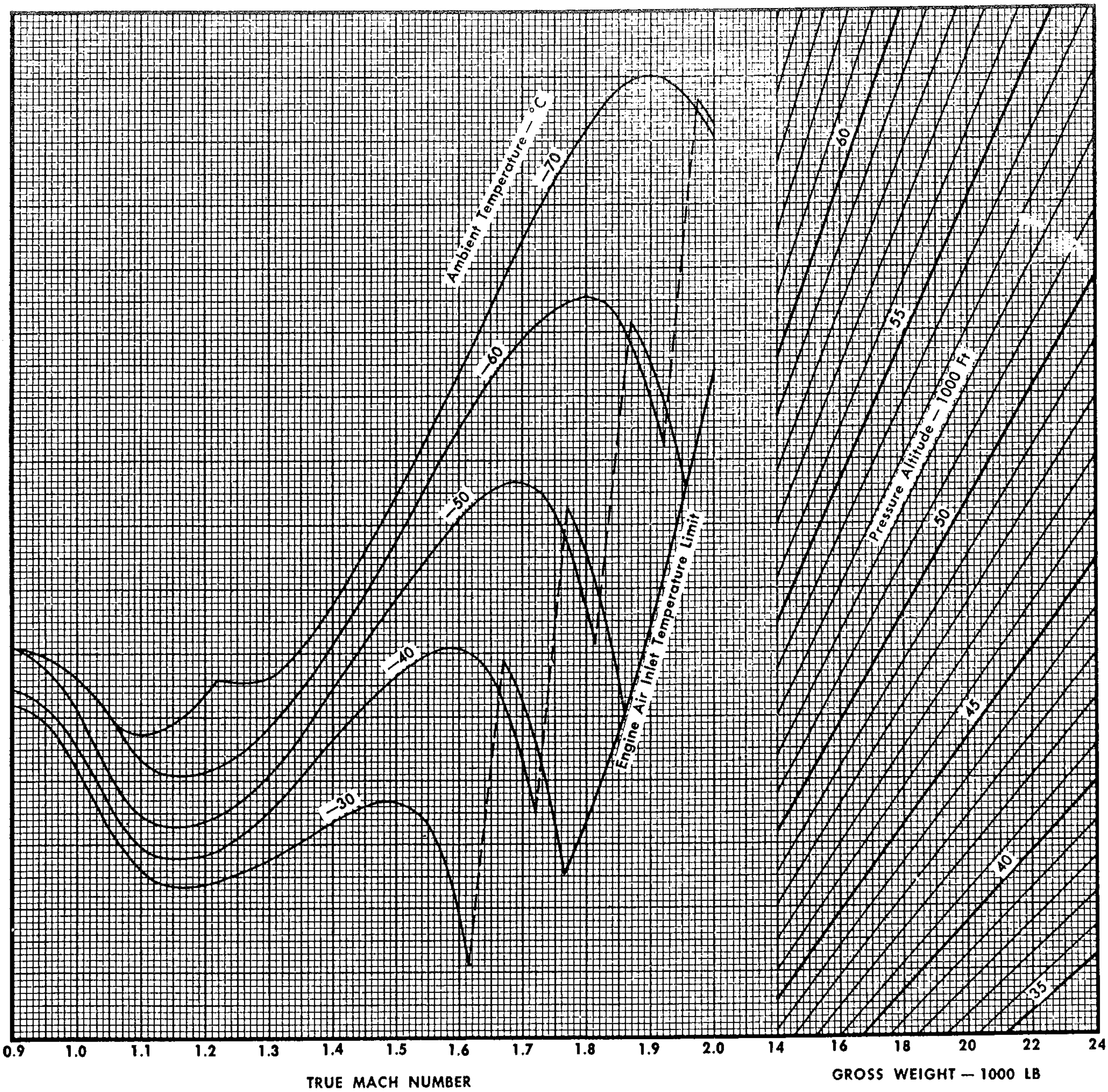


Figure A9-41

MAXIMUM THRUST FLIGHT ENVELOPE

PYLON TANKS

Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal

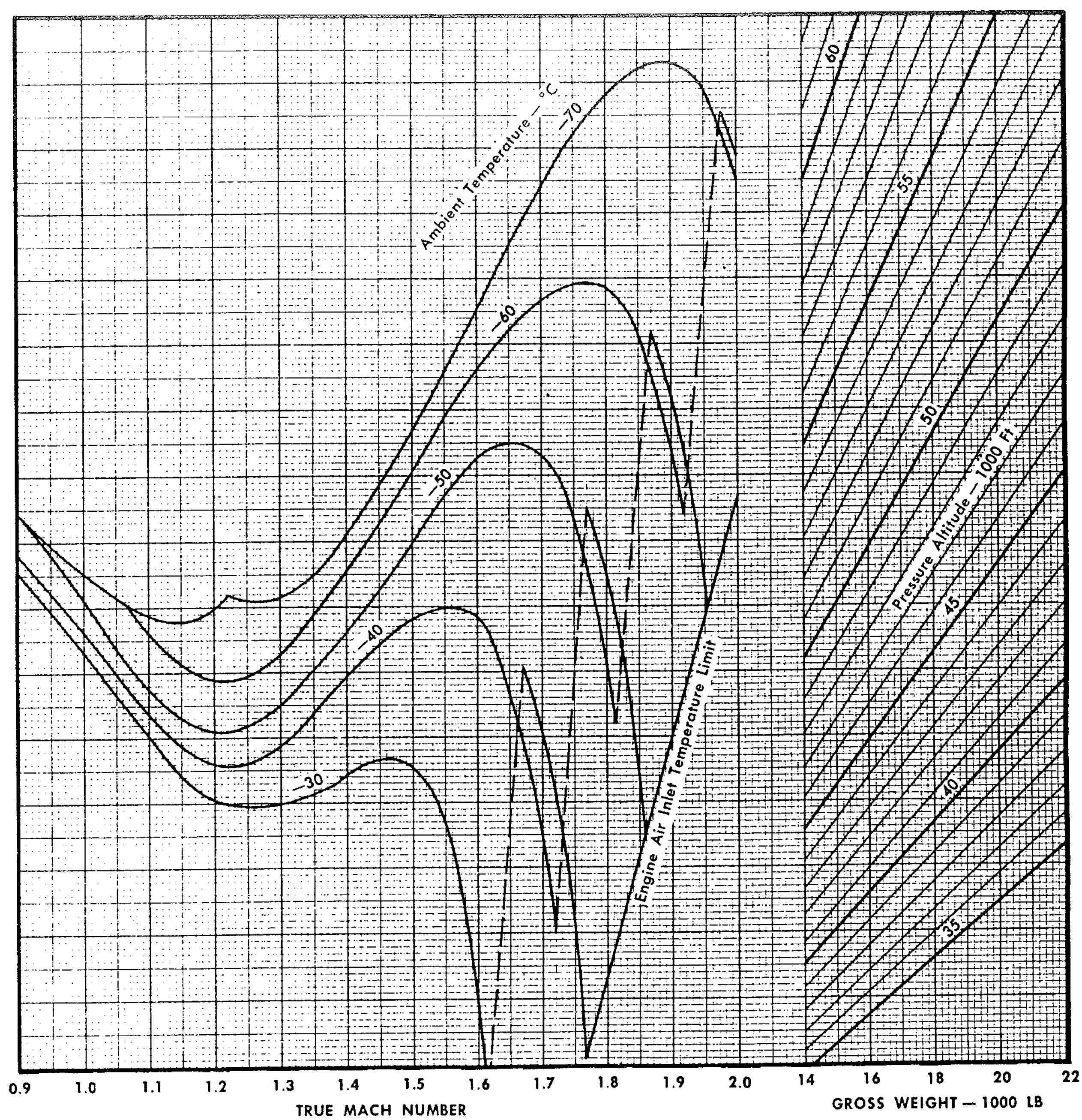


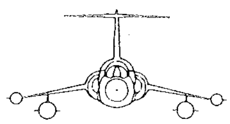
Figure A9-42

MAXIMUM THRUST FLIGHT ENVELOPE

TIP TANKS AND
PYLON TANKS

Model: F-104G
Date: 1 October 1964
DATA BASIS: ESTIMATED

Engine: J79-GE-11B
Fuel Grade: JP-8
Fuel Density: 6.68 Lb/Gal



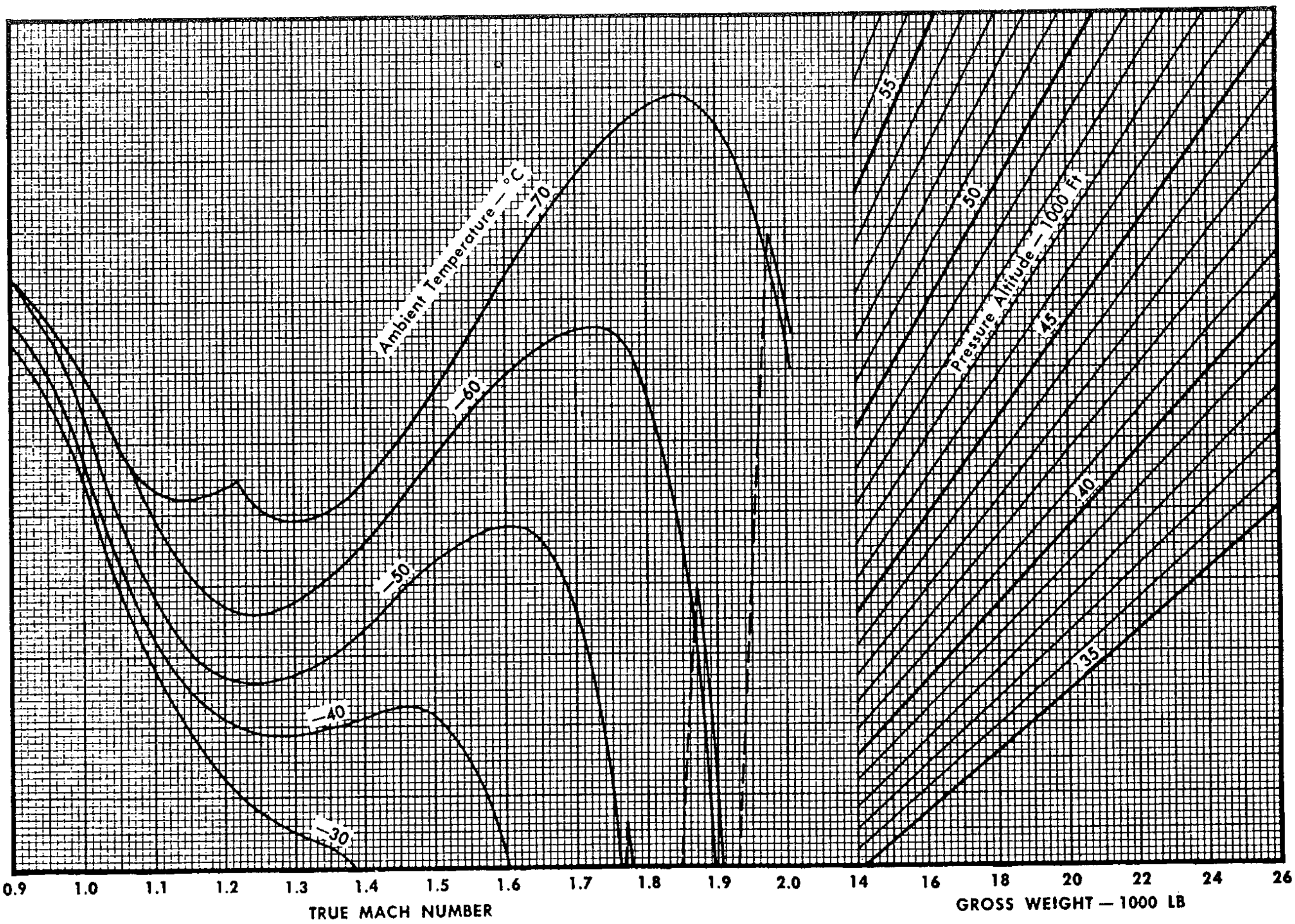


Figure A9-43

PART 10

MISSION PLANNING

TABLE OF CONTENTS

Index items in bold face characters denote illustrations.

MISSION PLANNING

Optimum use of the aircraft to obtain maximum performance at a minimum rate of fuel consumption requires careful preplanning for the mission. One of the most important phase of mission planning is the determination of the maximum radius from base which will allow an adequate return and reserve fuel allowance.
To find the maximum radius, a combat plan shall be formulated in advance, and a "key point" established from which the combat phase of the mission is begun. The key point is the point in flight where outbound cruise ends and the attack maneuver begins.
Depending on the type of mission to be flown, a high-thrust run on the target may be necessary, or supersonic maneuvers might be required, including accelerations, dashes, climbs and turns. The rate of fuel consumption is high during these maneuvers and any delay in breaking off the contact may seriously deplete the planned fuel reserve.

MISSION PLANNING EXAMPLE

The following problem is an exercise in use of the performance charts. The problem is not intended to reflect actual or proposed tactical missions employing this aircraft. The charts presented in parts 1 through 9 provide the performance information necessary to plan many different types of missions.

FERRY MISSION

The ferry mission is a transfer from one base to another at the optimum altitude and speed of minimum fuel specific consumption.

Configuration

A. Aircraft with two 170 gal external fuel tanks on wing tip and two 195 gal external fuel tanks on BL75.

Mission Profile

A. The assumed ambient conditions are no winds and OAT of 15°C (ISA) at the airfield altitude of 0 ft (sea level).

B. The FERRY MISSION can be subdivided into 6 phases:

1. Engine start-up and taxi
2. Take-off and acceleration to climb speed
3. Climb to optimum en-route cruise altitude
4. Optimum en-route cruise
5. Descent to sea level
6. Landing with a fuel reserve of 20 minutes sea level loiter

Configuration Drag Index and Masses

A. The value of the external loads drag index (D.I.) are as follows:

- Two 170 gal external tank on wing tip 16.0
- Two 195 gal external tank on BL75 .... 52.0

TOTAL DRAG INDEX OF THE CONFIGURATION 68.0

B. Masses:

- Operating Mass Empty .... 15056 lb
- Main internal fuel .... 4676 lb
- Two 170 gal external tank on wing tip (fuel only) .... 2271 lb
- Two 170 gal external tank (empty) .... 460 lb
- Two 195 gal external tank on BL75 (fuel only) .... 2605 lb
- Two 195 gal external tank (empty) .... 354 lb
- Wing pylons BL75 equipped for fuel tank .... 271 lb

TAKE-OFF WEIGHT .... 25693 lb

USABLE FUEL .... 9552 lb

Phase 1 – Engine Start-Up and Taxi

The fuel necessary for phase 1 is 150 lb for 7 minutes in ground maneuver (Figure A3-1).

Therefore: M_1 = 25543 lb (25693 − 150)

Phase 2 – Take-off and Acceleration to Climb Speed

The fuel required, the distance covered and the elapsed time to perform the take-off and accelerate to climb speed (Mach 0.85, as "MILITARY THRUST CLIMB CONTROL", Figure A3-10) are found with the use of Figure A3-1.
With a mass of 25543 lb, D.I. = 69 and at sea level, the fuel used, the distance covered and the elapsed time (Figure A3-1, Sheet 1) are:

Fuel .... 630 lb
Distance .... 5 nm
Time .... 1.17 min

Therefore: M_2 = 24913 lb (25543 − 630)

Phase 3 – Climb Cruise Altitude of 20000 ft

With an initial mass of 24913 lb and D.I. = 68, use the "MILITARY THRUST CLIMB CONTROL – CLIMB SPEED MACH 0.85" (Figure A3-10) chart to determine the fuel consumed, the distance traveled and the time to climb from sea level to 20000 ft at constant Mach = 0.85.

Fuel .... 600 lb
Distance .... 35 nm
Time .... 4 min

Therefore: M_3 = 24313 lb (24913 − 600)

Phase 6 – Landing

The landing mass of the aircraft is 16141 lb (with the external tanks empty and internal fuel consumed).
Reserves: 20 minutes LOITER, sea level and D.I. = 68
From Figure A5-1:

Speed .... = 0.42 Mach (275 KTAS)
Fuel flow .... 3200 lb/hr (53.3 lb/minute)
Fuel .... 1067 lb (53.3 lb/min x 20 min)

Therefore: M_6 = 17208 lb (16141 + 1067)

Phase 5 – Descent from 20000 ft

The chosen descent is called "85% RPM DESCENT PERFORMANCE – 300 KIAS" (Figure A7-3) with flaps and gear up.
The descent starts at 20000 ft and ends at sea level.

Speed .... 300 KIAS
Fuel .... 315 lb
Distance .... 55 nm
Time .... 9.5 min

Therefore: M_5 = 17523 lb (17208 + 315)

Phase 4 – Optimum Cruise at 20000 ft

The mean mass for optimum cruise is obtained by dividing by two the sum of the masses at the phase 3 (M_3) and phase 5 (M_5).

$$M_4 = \frac{M_3 + M_5}{2} = \frac{24313 + 17523}{2} = 20918 \text{ lb}$$

D.I. = 68
Altitude: 20000 ft

From Figure A4-1 "MAXIMUM RANGE — CONSTANT ALTITUDE CRUISE"

Optimum Cruise Speed = 0.78 MACH = 520 KTAS

Nautical miles per 1000 lb fuel = 127

The fuel available for the cruise phase is given by subtracting the fuel used during the passive phases and the reserve from the total fuel carried out:

Fuel for cruise =
9552 − (150 + 630 + 600 + 315 + 1067) = 6790 lb

$$\text{Cruise range} = \frac{6790 \text{x} 127}{1000} = 862 \text{ nm}$$

$$\text{Time} = \frac{862 \text{ nm}}{520 \text{ KTAS}} \text{ x } 60 = 99 \text{ min}$$

DISTANCE COVERED IN THE FERRY MISSION

Distance = 862 + 5 + 35 + 55 = 957 nm

TOTAL TIME OF THE FERRY MISSION

Time = 7 + 1.17 + 4 + 9.5 + 99 = 120.7 min = 2 h
Figure A10-1 contains a schematic representation of the mission profile and a table summarizing the specific phases.

EXAMPLE OF FERRY MISSION

MISSION: FERRY		TEMPERATURE: ISA				ENGINE: J79-GE-11B				
CONFIGURATION: 2 x 170 gal tanks on wing tip + 2 x 195 gal fuel tanks on BL75										
TAKEOFF WEIGHT: 25693 lbs						FUEL RESERVE (lb): 1067				
LANDING WEIGHT: 17208 lbs						RANGE (nm): 957				
INTERNAL FUEL: 4676 lbs						MISSION TOTAL TIME (h/min): 2 h 00 min				
EXTERNAL FUEL: 4876 lbs										
MISSION PHASE		ALT (ft)	DRAG INDEX	INITIAL MASS (kg)	FUEL USED		DISTANCE COVERED (nm)	TIMES		AVERAGE SPEED (KCAS/M)
					PART. (kg)	TOT. (kg)		PART. (min)	TOT. (min)	
1	ENGINE START AND TAXI	S.L.	68	25693	150	150	–	7.0	7.0	–
2	TAKEOFF AND ACCELERATION	S.L.	68	25543	630	780	5	1.17	8.17	–
3	CLIMB	20000	68	24913	600	1380	35	4.0	12.17	0.85 M
4	OPTIMUM CRUISE	20000	68	24313	6790	8170	862	99.0	111.17	0.78 M
5	DESCENT	S.L.	68	17523	315	8485	55	9.5	120.67	300 KIAS
6	LANDING AND RESERVE	S.L.	68	17208	1067	9552	–	–	–	–

MISSION PROFILE

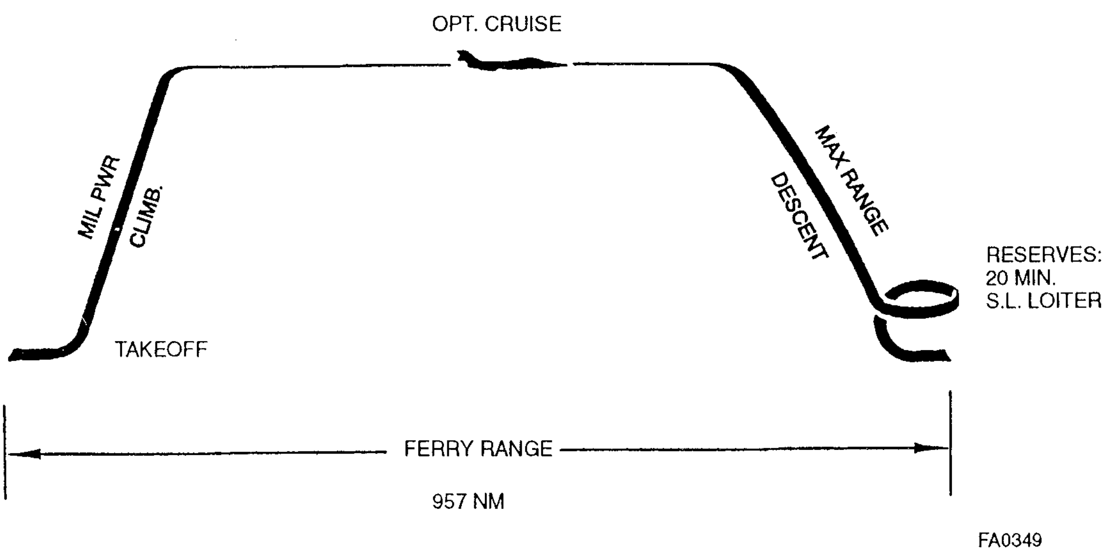


Figure A10-1

ALPHABETICAL INDEX

A

B

F

G

N

O

P

T

U

V

W

Z

GENERAL ARRANGEMENT

1 PITOT–STATIC BOOM
2 RADAR
3 FORWARD ELECTRONICS COMPARTMENT
4 UPPER IFF, TACAN, AND UHF ANTENNA
5 AFT ELECTRONICS COMPARTMENT
6 AUXILIARY FUEL CELL AND FILLER WELL
7 AIR CONDITIONING PACKAGE
8 FWD MAIN FUEL CELL
9 AFT MAIN FUEL CELL
10 FILLER WELL
11 LEFT MAIN FUEL CELL
12 J79–GE–11B ENGINE WITH AFTERBURNER
13 HORIZONTAL STABILIZER POWER CONTROL ASSEMBLY
14 CONTROLLABLE HORIZONTAL STABILIZER
15 RUDDER POWER CONTROL ASSEMBLY
16 NAVIGATION LIGHTS
17 DRAG CHUTE STOWAGE COMPARTMENT
18 SPEED BRAKE
19 ARRESTING HOOK
20 TRAILING EDGE FLAPS
21 AILERON
22 TIP TANK
23 AILERON POWER CONTROL ASSEMBLY
24 LEADING EDGE FLAP
25 PYLON TANK
26 ENGINE AIR INTAKE DUCT
27 FWD AND AFT AUXILIARY FUEL TANKS
28 EJECTION SEATS
29 LOWER UHF ANTENNA
30 TACAN ANTENNA

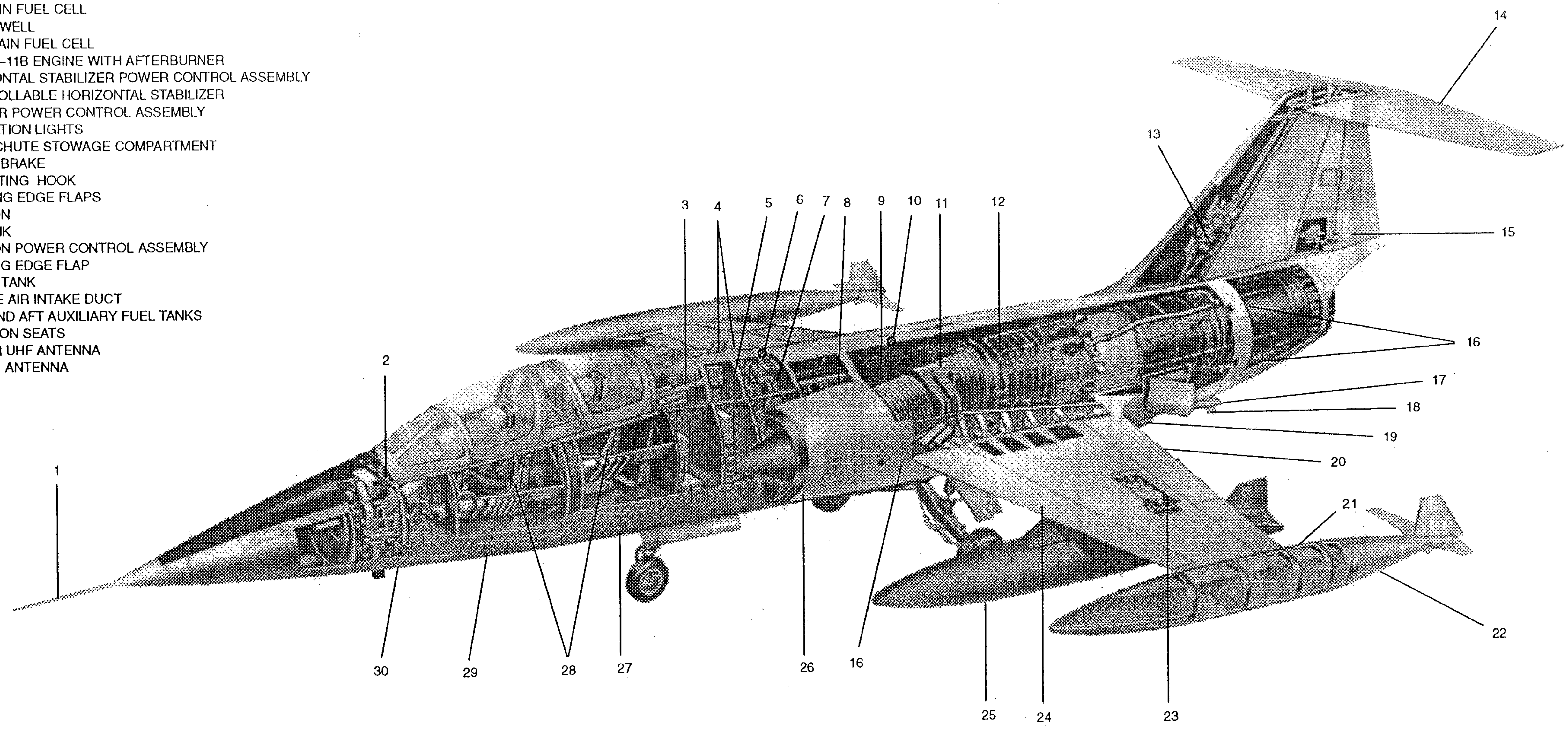


Figure FO-1

FA0198

ENGINE

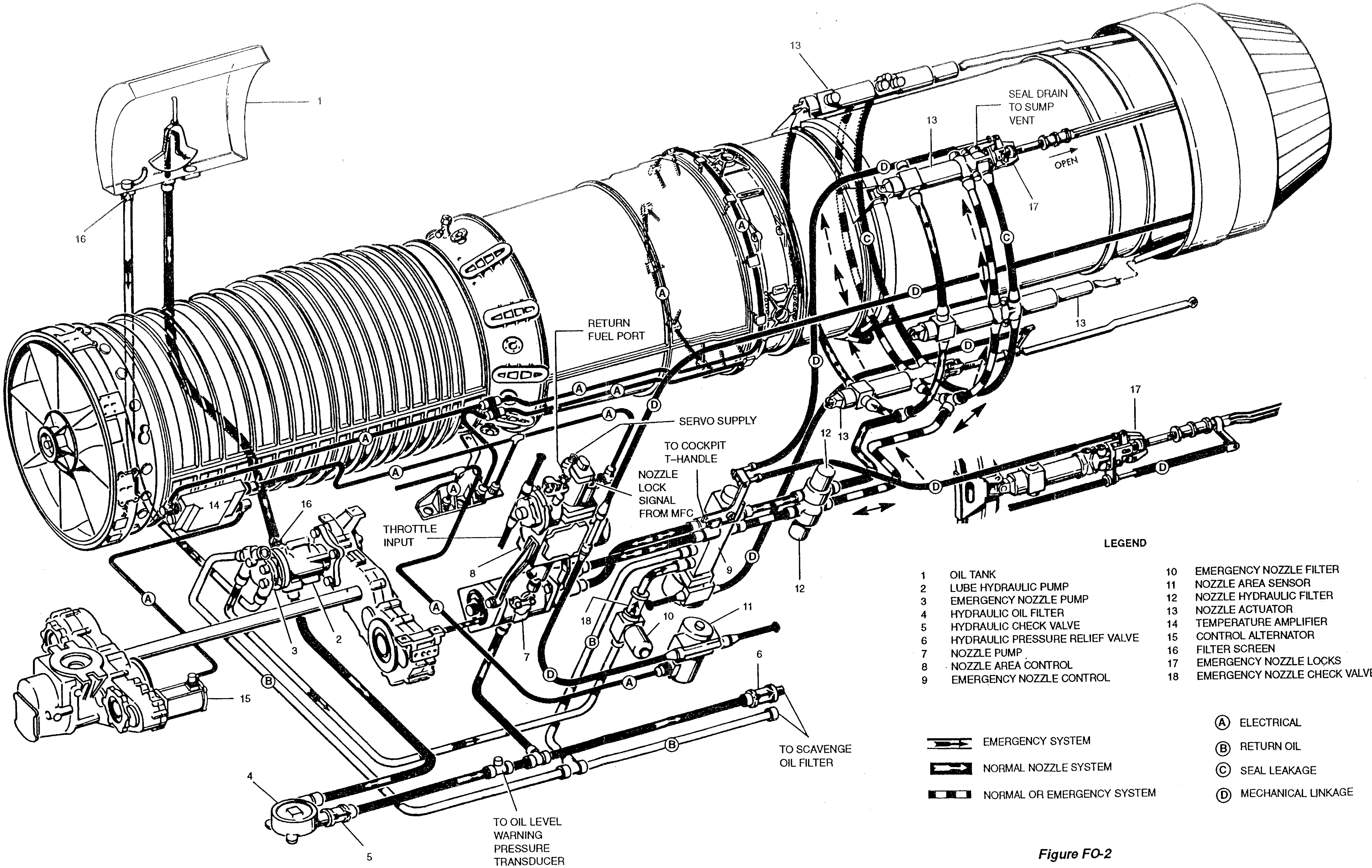


Figure FO-2

ENGINE FUEL SYSTEM

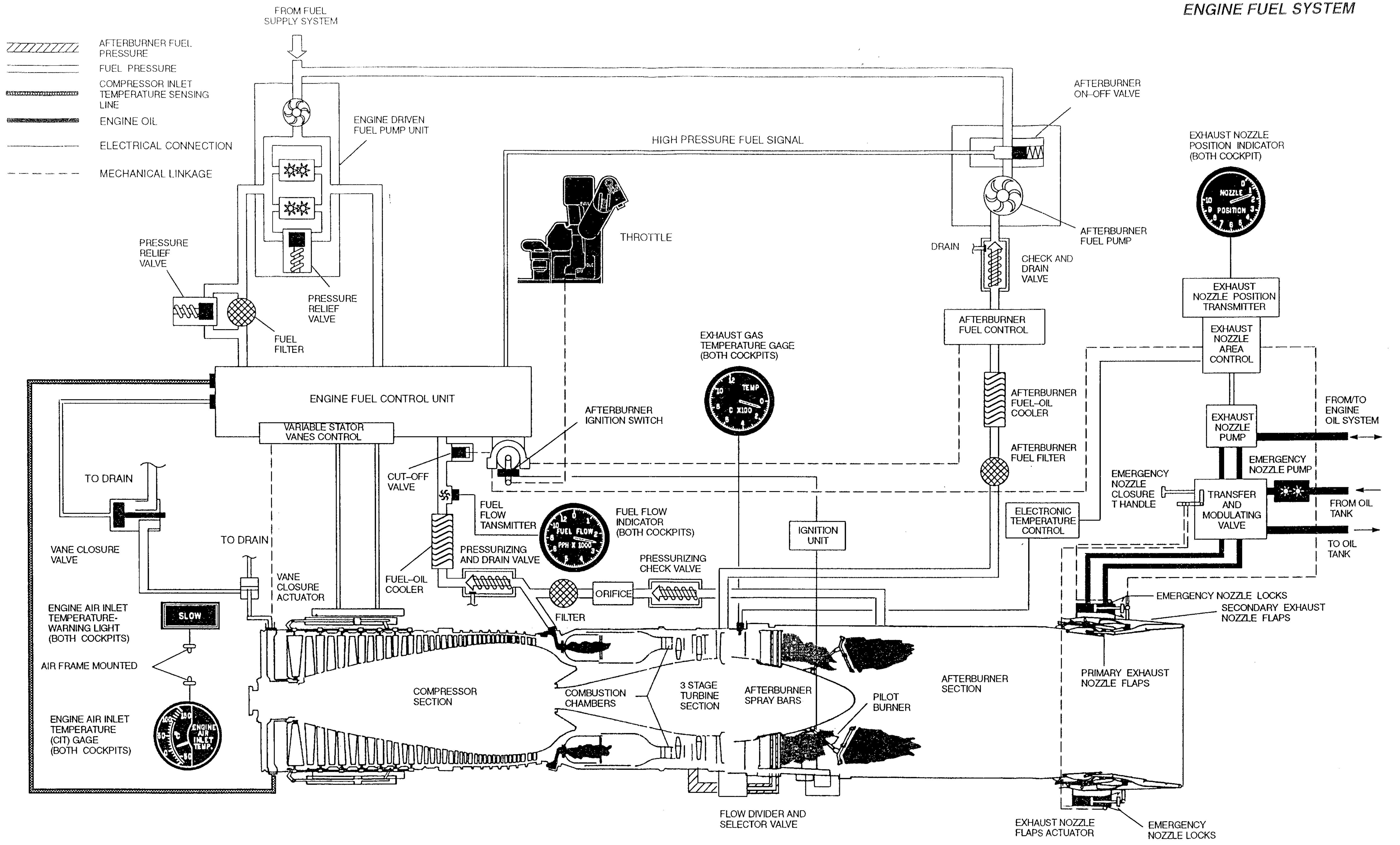


Figure FO-3

FA0302

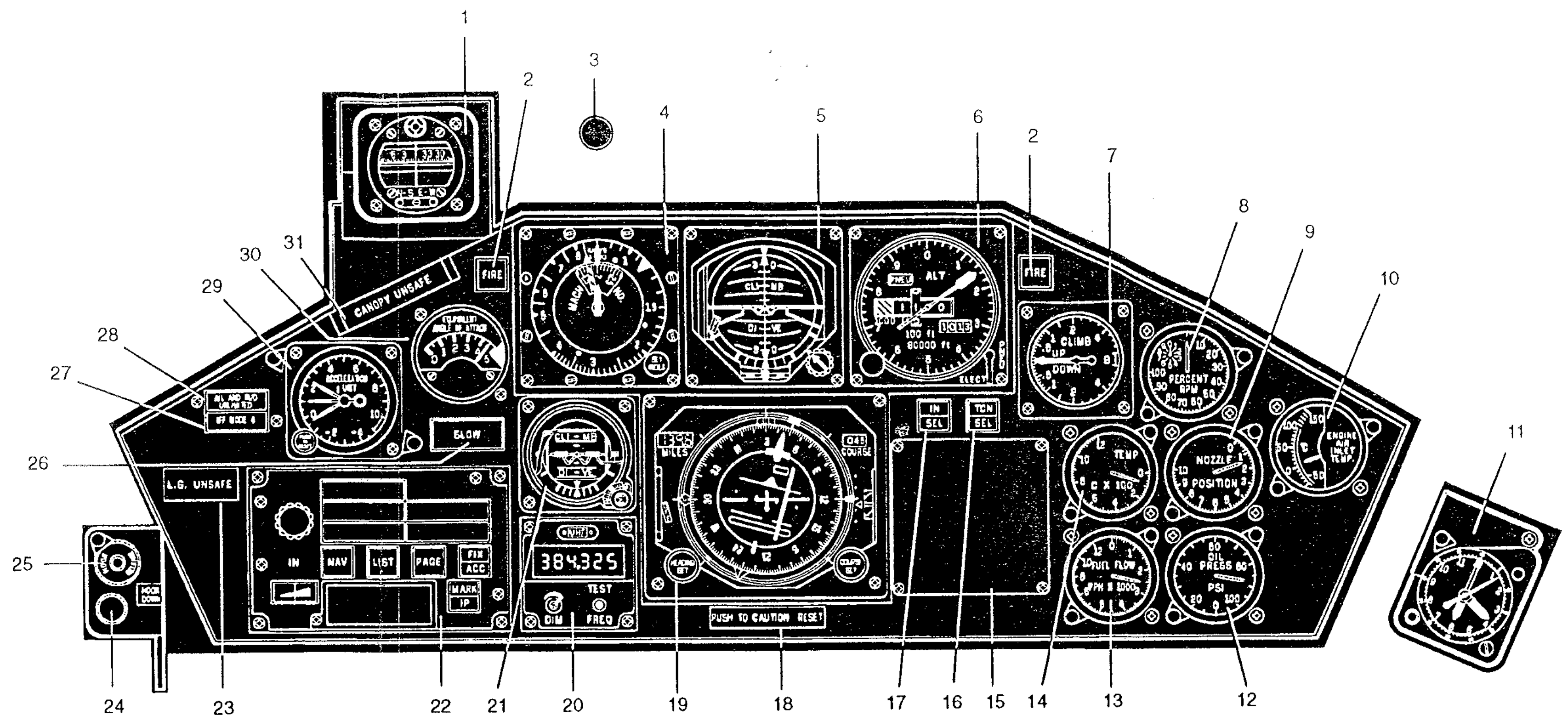
FRONT COCKPIT INSTRUMENT PANEL

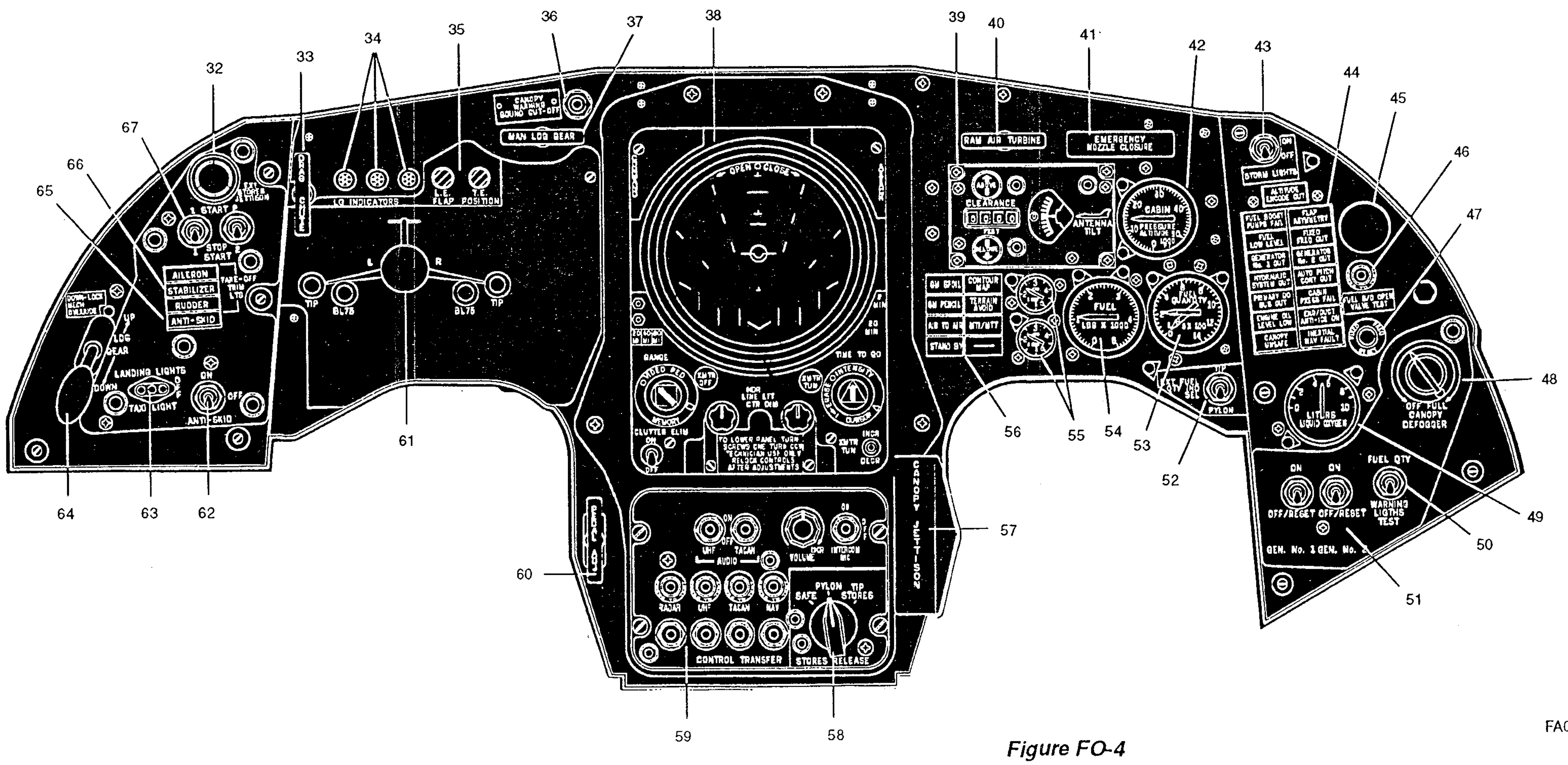
UPPER INSTRUMENT PANEL

1 STANDBY COMPASS
2 FIRE WARNING LIGHT
3 RADAR LOCK–ON LIGHT
4 AIRSPEED AND MACH NUMBER INDICATOR
5 ATTITUDE INDICATOR
6 ALTIMETER
7 VERTICAL VELOCITY INDICATOR
8 TACHOMETER
9 EXHAUST NOZZLE POSITION INDICATOR
10 CIT GAGE
11 CLOCK
12 OIL PRESSURE GAGE
13 FUEL FLOW INDICATOR
14 EGT GAGE
15 BLANK
16 TCN SELECTOR PUSHBUTTON
17 IN SELECTOR PUSHBUTTON
18 CAUTION LIGHT
19 HORIZONTAL SITUATION INDICATOR
20 CHANNEL FREQUENCY INDICATOR
21 STANDBY ATTITUDE INDICATOR
22 MASTER IN CDU
23 LANDING GEAR UNSAFE WARNING LIGHT
24 ARRESTING HOOK INDICATOR LIGHT
25 ARRESTING HOOK RELEASE BUTTON
26 SLOW WARNING LIGHT
27 IFF MODE 4 WARNING LIGHT (INOPERATIVE)
28 AIL AND RUD UNLIMITED WARNING LIGHT
29 ACCELEROMETER
30 APC INDICATOR
31 CANOPY UNSAFE WARNING LIGHT

LOWER INSTRUMENT PANEL

32 EXTERNAL STORES JETTISON BUTTON
33 DRAG CHUTE HANDLE
34 LANDING GEAR POSITION INDICATORS
35 WING FLAP POSITION INDICATORS
36 CANOPY WARNING SOUND CUT–OFF SWITCH
37 MANUAL LANDING GEAR RELEASE HANDLE
38 RADAR INDICATOR
39 CLEARANCE PLANE AND ANTENNA TILT INDICATOR
40 RAT HANDLE
41 EMERGENCY NOZZLE CLOSURE HANDLE
42 CABIN ALTIMETER
43 STORM LIGHTS SWITCH
44 WARNING LIGHTS PANEL
45 BLANK
46 FUEL SHUTOFF OPEN VALVE TEST BUTTON
47 FIXED FREQUENCY RESET BUTTON
48 CANOPY DEFOGGER KNOB
49 OXYGEN QUANTITY INDICATOR
50 FUEL QUANTITY AND WARNING LIGHTS TEST SWITCH
51 GENERATOR SWITCHES
52 EXTERNAL FUEL QUANTITY INDICATOR SELECTOR SWITCH
53 EXTERNAL FUEL QUANTITY INDICATOR
54 INTERNAL FUEL QUANTITY INDICATOR
55 HYDRAULIC PRESSURE GAGES
56 RADAR MODE INDICATOR LIGHTS
57 CANOPY JETTISON HANDLE
58 EXTERNAL STORES SELECTOR SWITCH
59 CONTROL TRANSFER PANEL
60 RUDDER PEDAL ADJUSTMENT HANDLE
61 EXTERNAL STORES SELECTOR BUTTONS AND INDICATORS
62 ANTI–SKID SWITCH
63 LANDING AND TAXI LIGHTS SWITCH
64 LANDING GEAR LEVER
65 ANTI–SKID LIGHT
66 TAKE–OFF TRIM LIGHTS
67 START SWITCHES

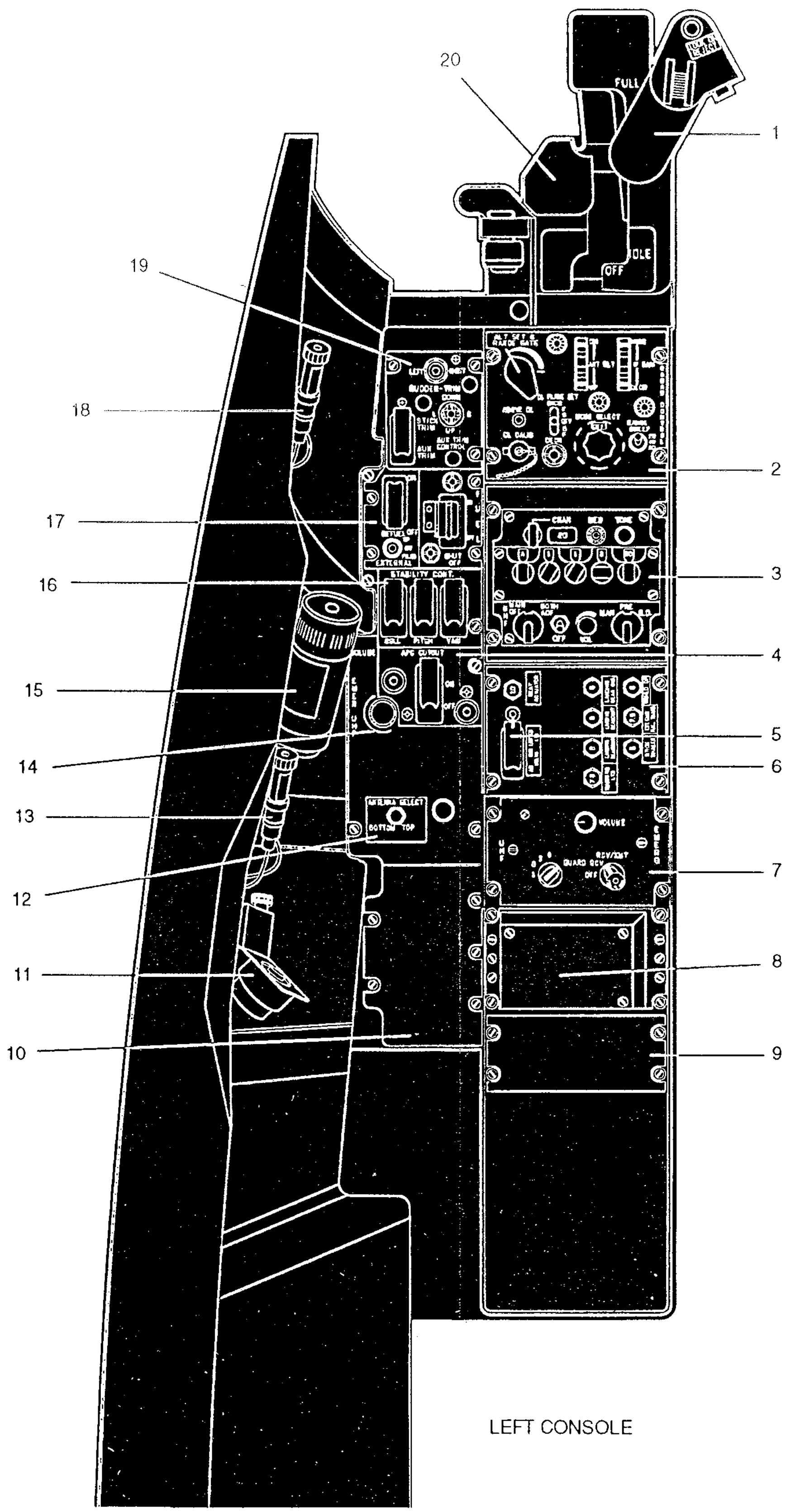


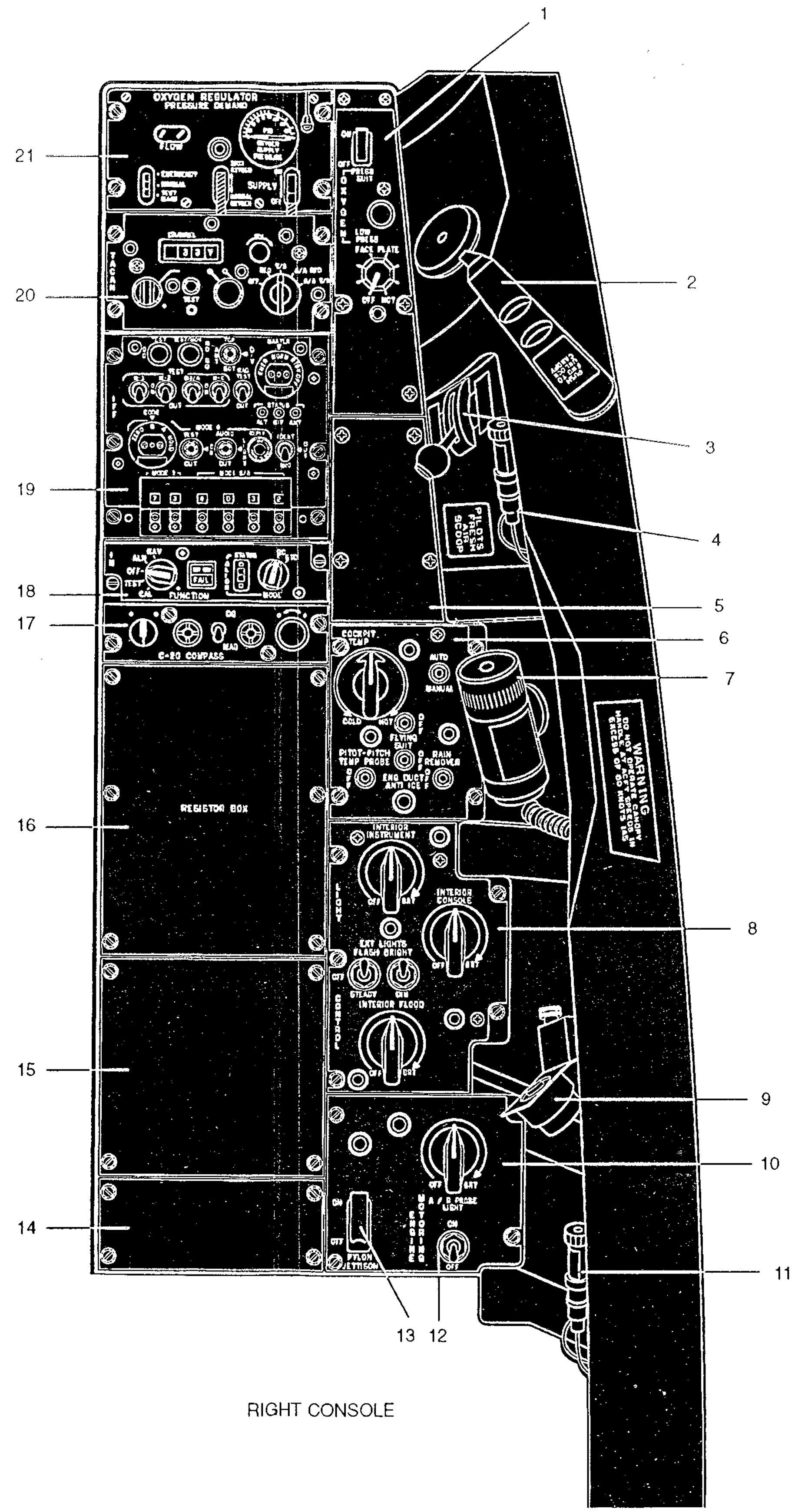


FA0232

Figure FO-4

FRONT COCKPIT
LEFT AND RIGHT CONSOLES





LEFT CONSOLE

1 ENGINE THROTTLE
2 RADAR CONTROL PANEL
3 UHF RADIO CONTROL PANEL
4 APC CUTOUT SWITCH
5 AIL AND RUD LIMITER SWITCH
6 CIRCUIT BREAKER CONTROL PANEL
7 EMERGENCY UHF CONTROL PANEL
8 BLANK
9 BLANK
10 BLANK
11 STORM LIGHT
12 ANTENNA SELECTOR SWITCH
13 FLOOD LIGHT
14 EMERGENCY UHF VOLUME CONTROL KNOB
15 SPOT LIGHT
16 STABILITY CONTROL SWITCHES
17 FUEL CONTROL PANEL
18 FLOOD LIGHT
19 TRIM CONTROL PANEL
20 FLAP LEVER

RIGHT CONSOLE

1 OXYGEN LOW PRESSURE WARNING LIGHT
2 CANOPY LOCKING LEVER
3 FRESH AIR SCOOP LEVER
4 FLOOD LIGHT
5 BLANK
6 HEATING CONTROL PANEL
7 SPOT LIGHT
8 LIGHT CONTROL PANEL
9 STORM LIGHT
10 PROBE LIGHT CONTROL KNOB (INOPERATIVE)
11 FLOOD LIGHT
12 ENGINE MOTORING SWITCH
13 PYLON JETTISON SWITCH
14 BLANK
15 BLANK
16 BLANK
17 C–2G COMPASS CONTROL PANEL
18 IN CONTROL PANEL
19 IFF CONTROL PANEL
20 TACAN CONTROL PANEL
21 OXYGEN CONTROL PANEL

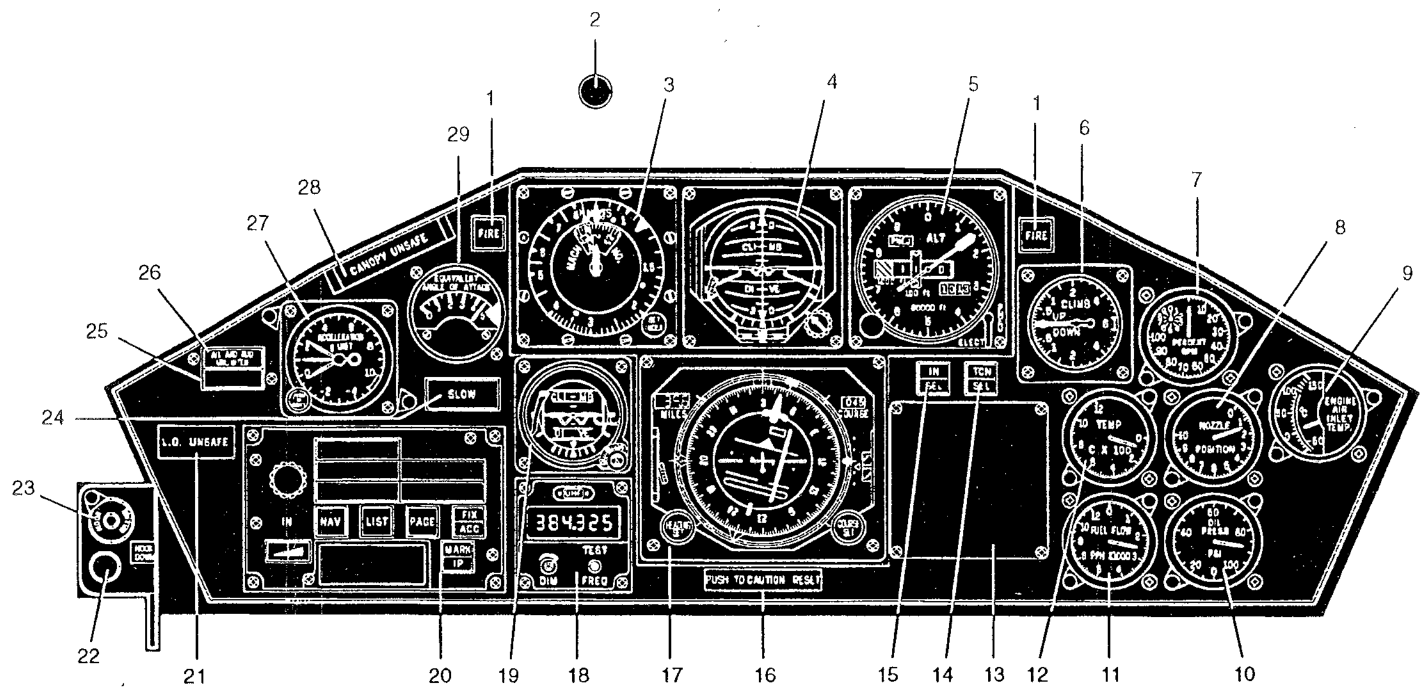
Figure FO-5

FA0257

REAR COCKPIT INSTRUMENT PANEL

UPPER INSTRUMENT PANEL

1 FIRE WARNING LIGHT
2 RADAR LOCK–ON LIGHT
3 AIRSPEED AND MACH NUMBER INDICATOR
4 ATTITUDE INDICATOR
5 ALTIMETER
6 VERTICAL VELOCITY INDICATOR
7 TACHOMETER
8 EXHAUST NOZZLE POSITION INDICATOR
9 CIT GAGE
10 OIL PRESSURE GAGE
11 FUEL FLOW INDICATOR
12 EGT GAGE
13 BLANK
14 TCN SELECTOR PUSHBUTTON
15 IN SELECTOR PUSHBUTTON
16 CAUTION LIGHT
17 HORIZONTAL SITUATION INDICATOR
18 CHANNEL FREQUENCY INDICATOR
19 STANDBY ATTITUDE INDICATOR
20 SLAVE IN CDU
21 LANDING GEAR UNSAFE WARNING LIGHT
22 ARRESTING HOOK INDICATOR LIGHT
23 ARRESTING HOOK RELEASE BUTTON
24 SLOW WARNING LIGHT
25 BLANK
26 AIL AND RUD UNLIMITED WARNING LIGHT
27 ACCELEROMETER
28 CANOPY UNSAFE WARNING LIGHT
29 APC INDICATOR



LOWER INSTRUMENT PANEL

29 EXTERNAL STORES JETTISON BUTTON
30 DRAG CHUTE HANDLE
31 CANOPY WARNING SOUND CUT–OFF SWITCH
32 LANDING GEAR POSITION INDICATORS
33 WING FLAP POSITION INDICATORS
34 MANUAL LANDING GEAR RELEASE HANDLE
35 RADAR INDICATOR
36 CLEARANCE PLANE AND ANTENNA TILT INDICATOR
37 RAT HANDLE
38 EMERGENCY NOZZLE CLOSURE HANDLE
39 CABIN ALTIMETER
40 STORM LIGHTS SWITCH
41 WARNING LIGHTS PANEL
42 OXYGEN QUANTITY INDICATOR
43 FUEL SHUTOFF OPEN VALVE TEST BUTTON
44 FUEL QUANTITY AND WARNING LIGHTS TEST SWITCH
47 FIXED FREQUENCY RESET BUTTON
45 GENERATOR SWITCHES
46 EXTERNAL FUEL QUANTITY INDICATOR
47 INTERNAL FUEL QUANTITY INDICATOR
48 HYDRAULIC PRESSURE GAGES
49 RADAR MODE INDICATOR LIGHTS
50 CANOPY JETTISON HANDLE
51 CONTROL TRANSFER PANEL
52 RUDDER PEDAL ADJUSTMENT HANDLE
53 EMERGENCY STORES CUT–OUT SWITCH
54 EXTERNAL STORES POSITION INDICATORS
55 ANTI–SKID SWITCH
56 ANTI–SKID LIGHT
57 LANDING AND TAXI LIGHTS SWITCH
58 LANDING GEAR LEVER
59 TRIM LIGHTS
60 START SWITCHES

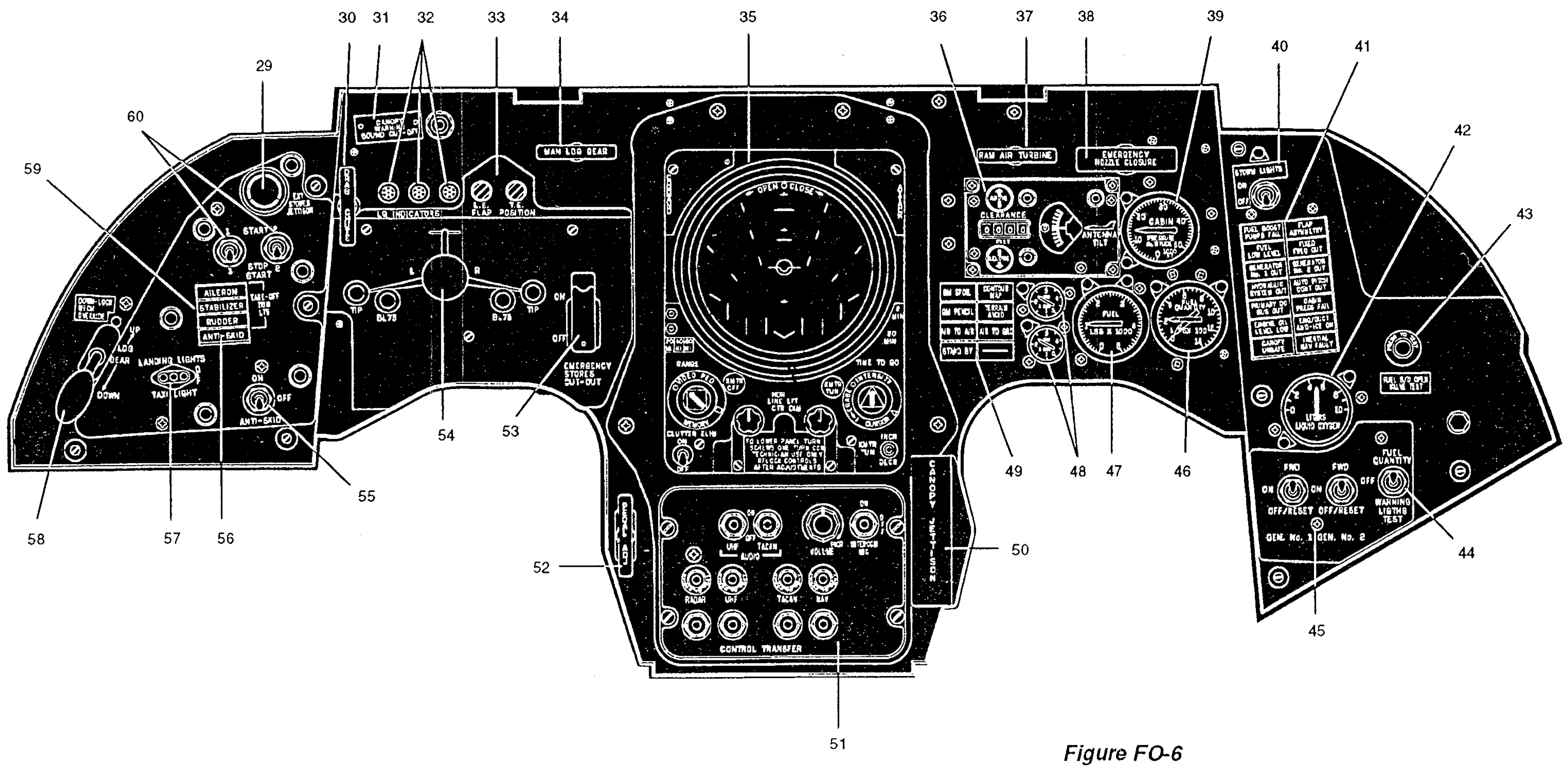
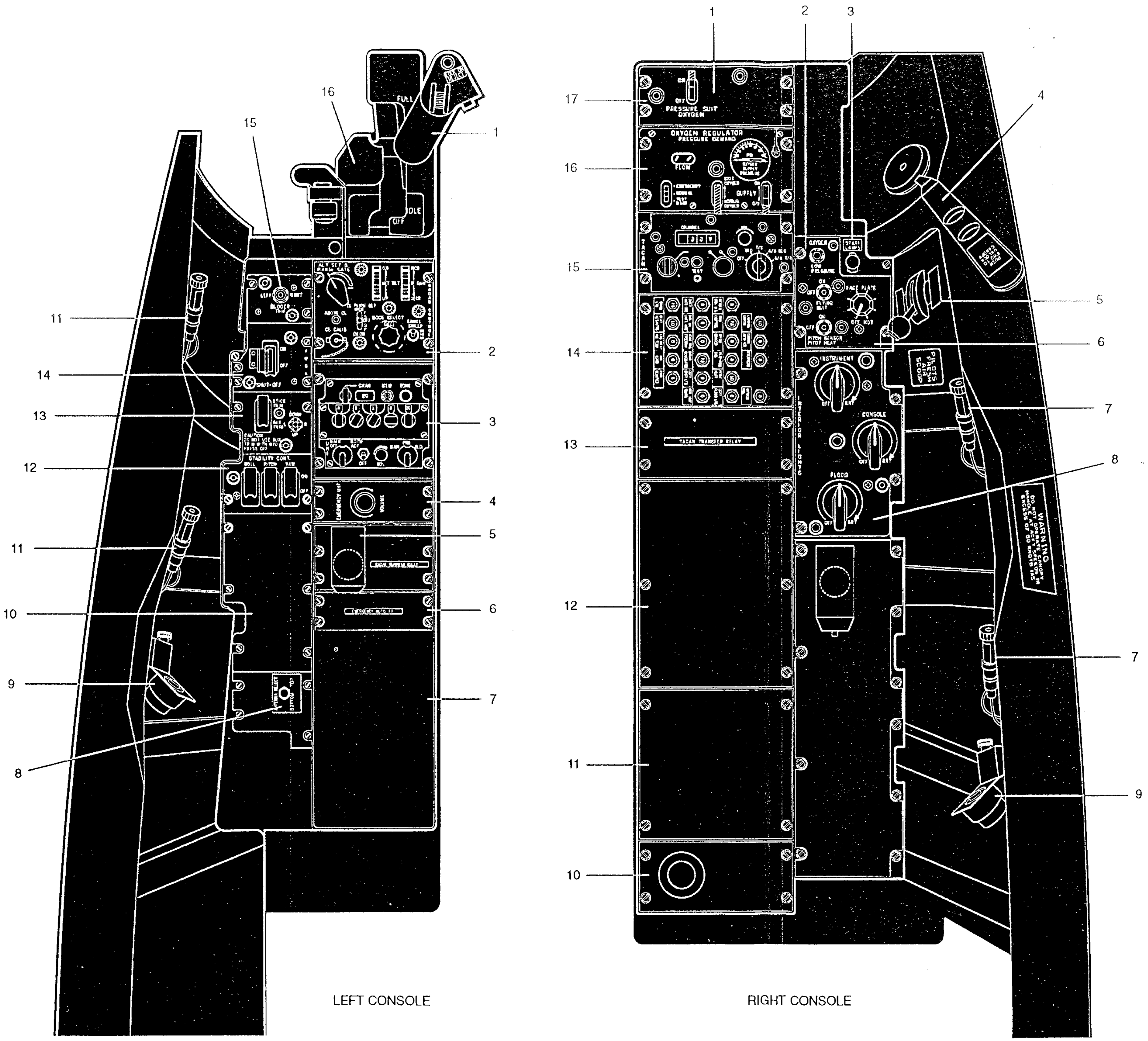


Figure FO-6

FA0258

REAR COCKPIT
LEFT AND RIGHT CONSOLES



LEFT CONSOLE

1 ENGINE THROTTLE
2 RADAR CONTROL PANEL
3 UHF RADIO CONTROL PANEL
4 EMERGENCY UHF VOLUME CONTROL KNOB
5 SPOT LIGHT
6 BLANK
7 BLANK
8 ANTENNA SELECTOR SWITCH
9 STORM LIGHT
10 BLANK
11 FLOOD LIGHT
12 STABILITY CONTROL SWITCHES
13 TRIM CONTROL PANEL
14 FUEL CONTROL PANEL
15 RUDDER TRIM SWITCH
16 FLAP LEVER

RIGHT CONSOLE

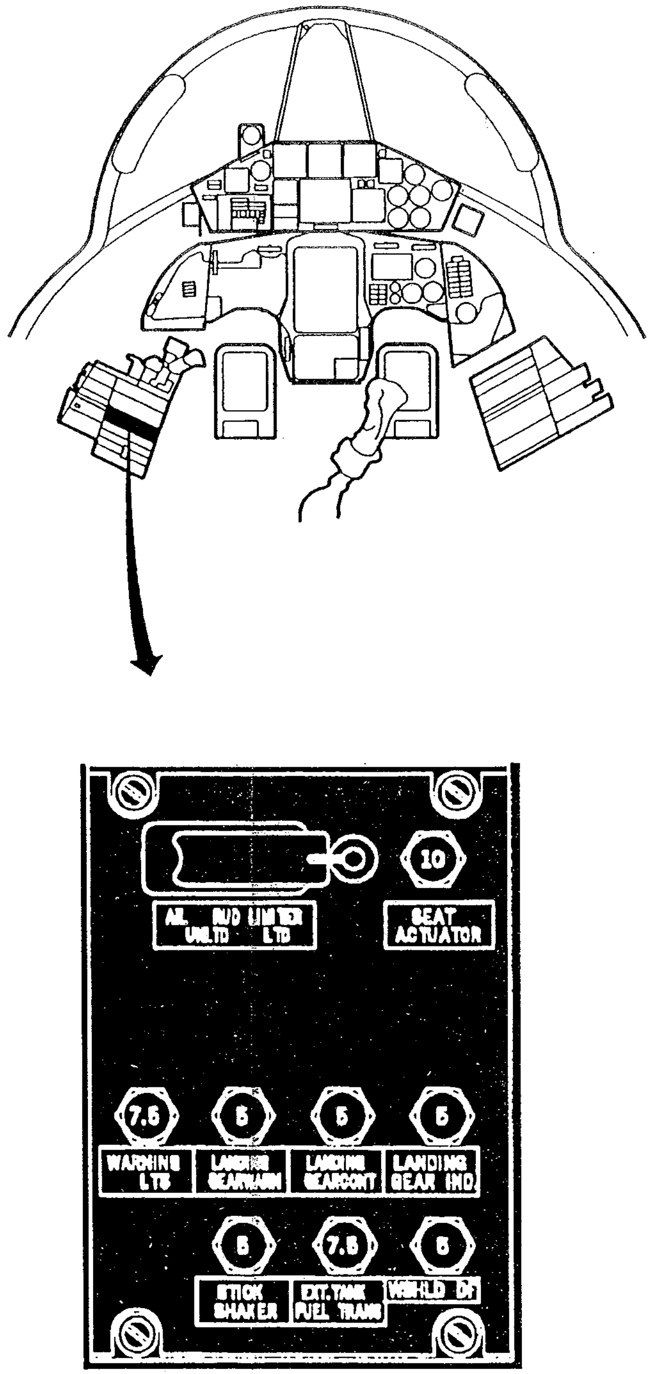
1 PRESSURE SUIT OXYGEN SWITCH (INOPERATIVE)
2 OXYGEN LOW PRESSURE LIGHT
3 SPARE LAMPS
4 CANOPY LOCKING LEVER
5 FRESH AIR SCOOP LEVER
6 PITOT PITCH TEMP PROBE SWITCH
7 FLOOD LIGHT
8 INTERIOR LIGHT CONTROL PANEL
9 STORM LIGHT
10 SPOT LIGHT
11 BLANK
12 BLANK
13 BLANK
14 BLANK
15 CIRCUIT BREAKER PANEL
16 TACAN CONTROL PANEL
17 OXYGEN CONTROL PANEL

Figure FO-7

FA0266

CIRCUIT BREAKERS PANELS

FRONT COCKPIT

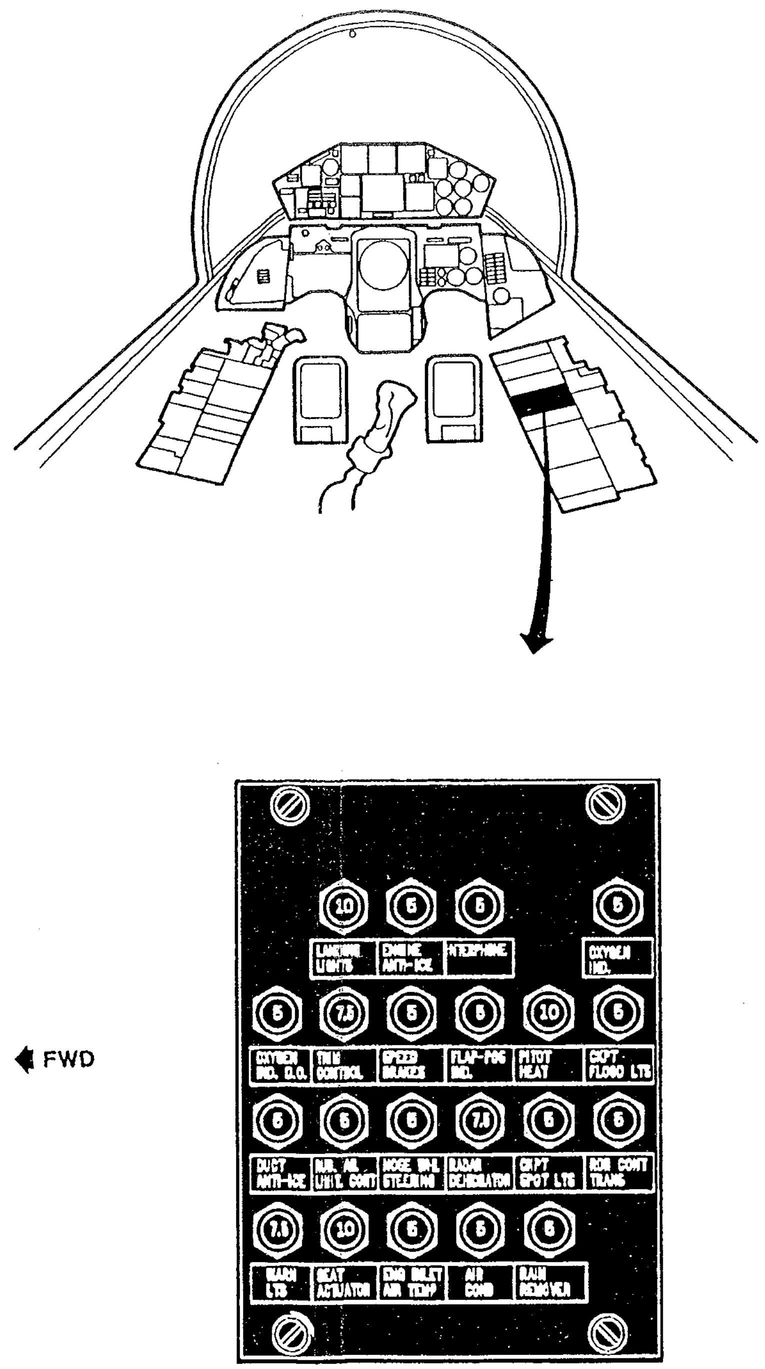


LEFT CONSOLE

FWD

REAR COCKPIT

FWD



RIGHT CONSOLE

Figure FO-8

FA0256

FUEL SUPPLY SYSTEM

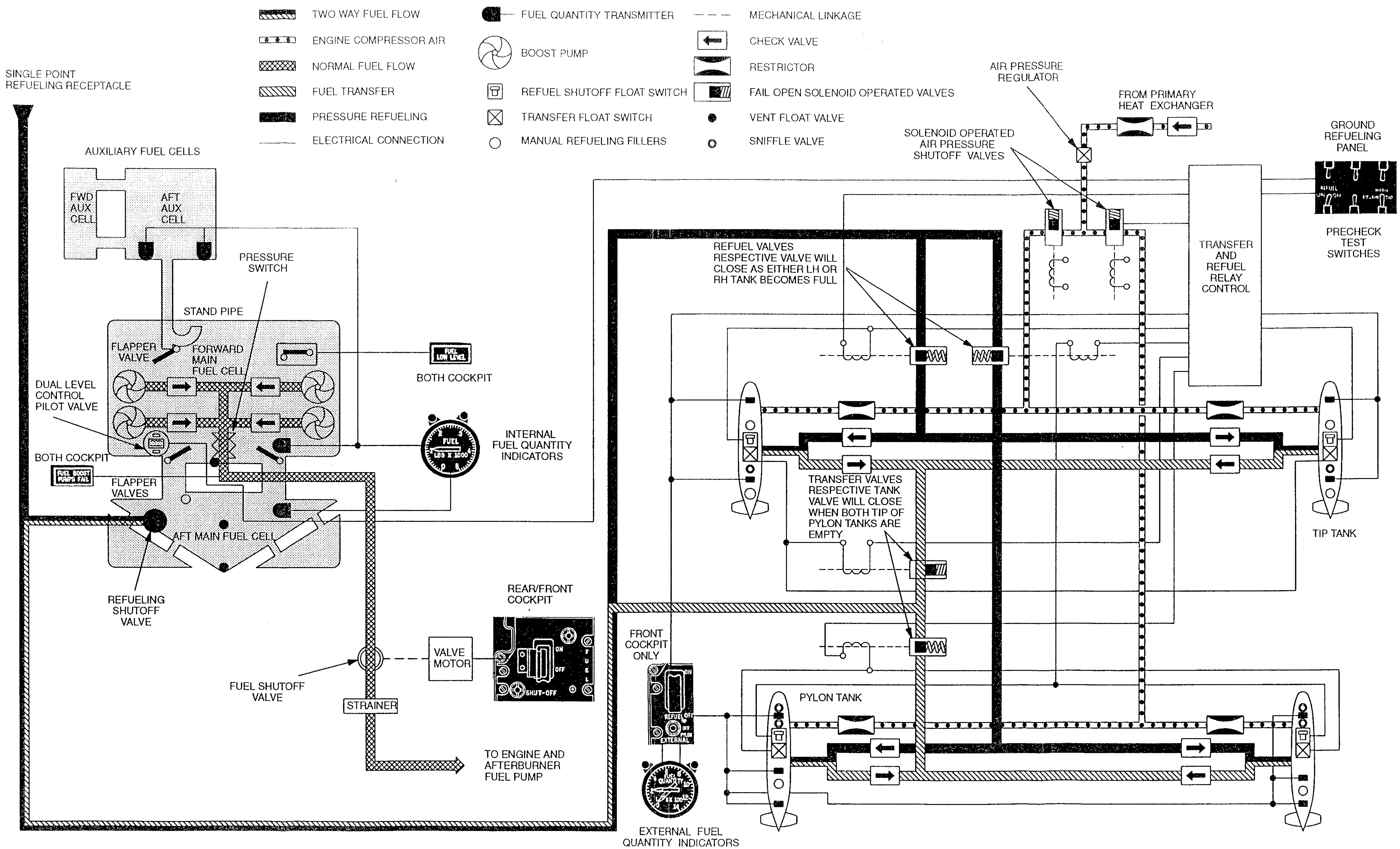
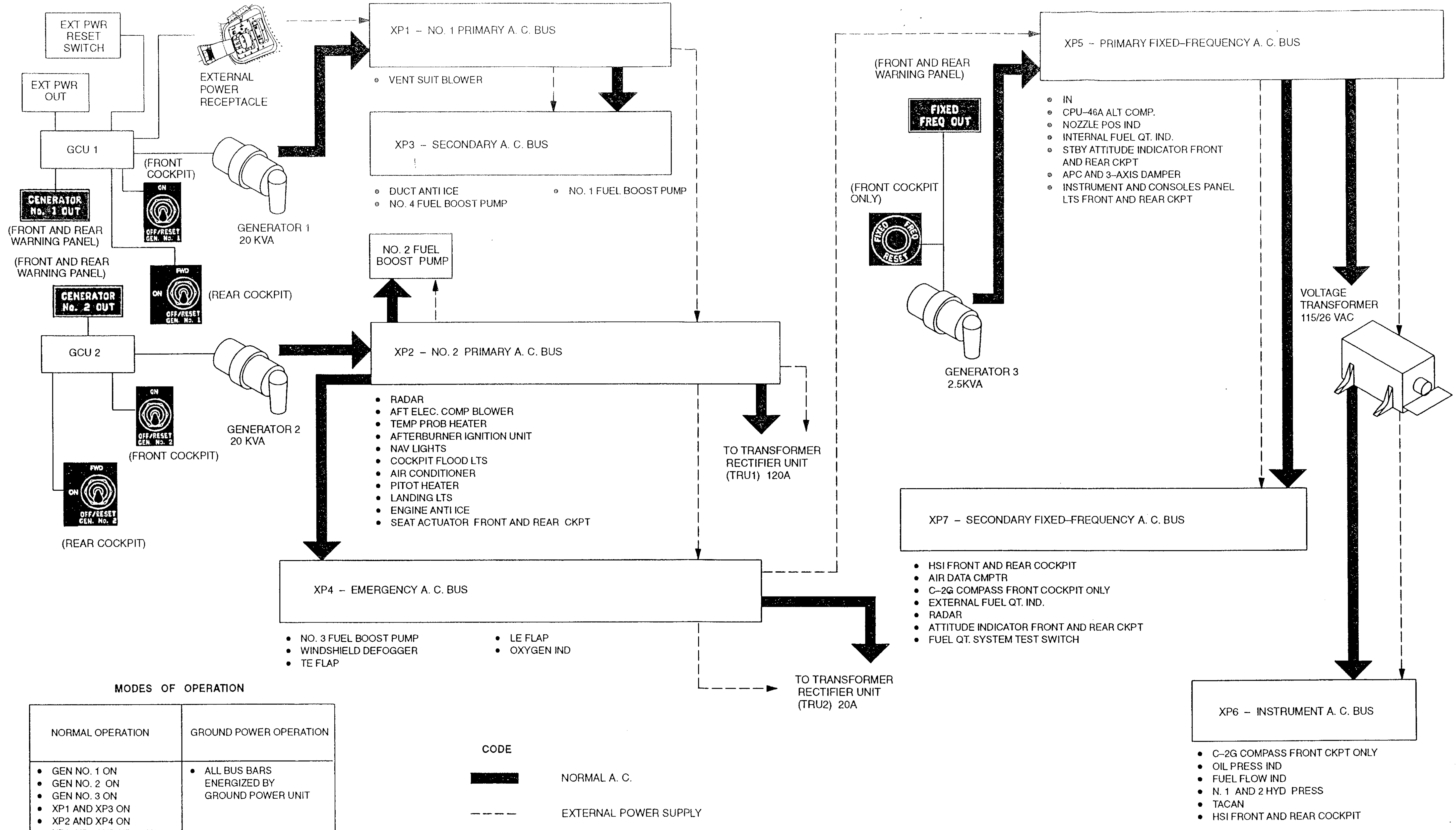


Figure FO-9

FA0070

AC ELECTRICAL POWER SUPPLY SYSTEM NORMAL DISTRIBUTION



MODES OF OPERATION

NORMAL OPERATION	GROUND POWER OPERATION
• GEN NO. 1 ON • GEN NO. 2 ON • GEN NO. 3 ON • XP1 AND XP3 ON • XP2 AND XP4 ON • XP5, XP6 AND XP7 ON • RAT INOPERATIVE	• ALL BUS BARS ENERGIZED BY GROUND POWER UNIT

Figure FO-10 (Sheet 1 of 2)

FA0018

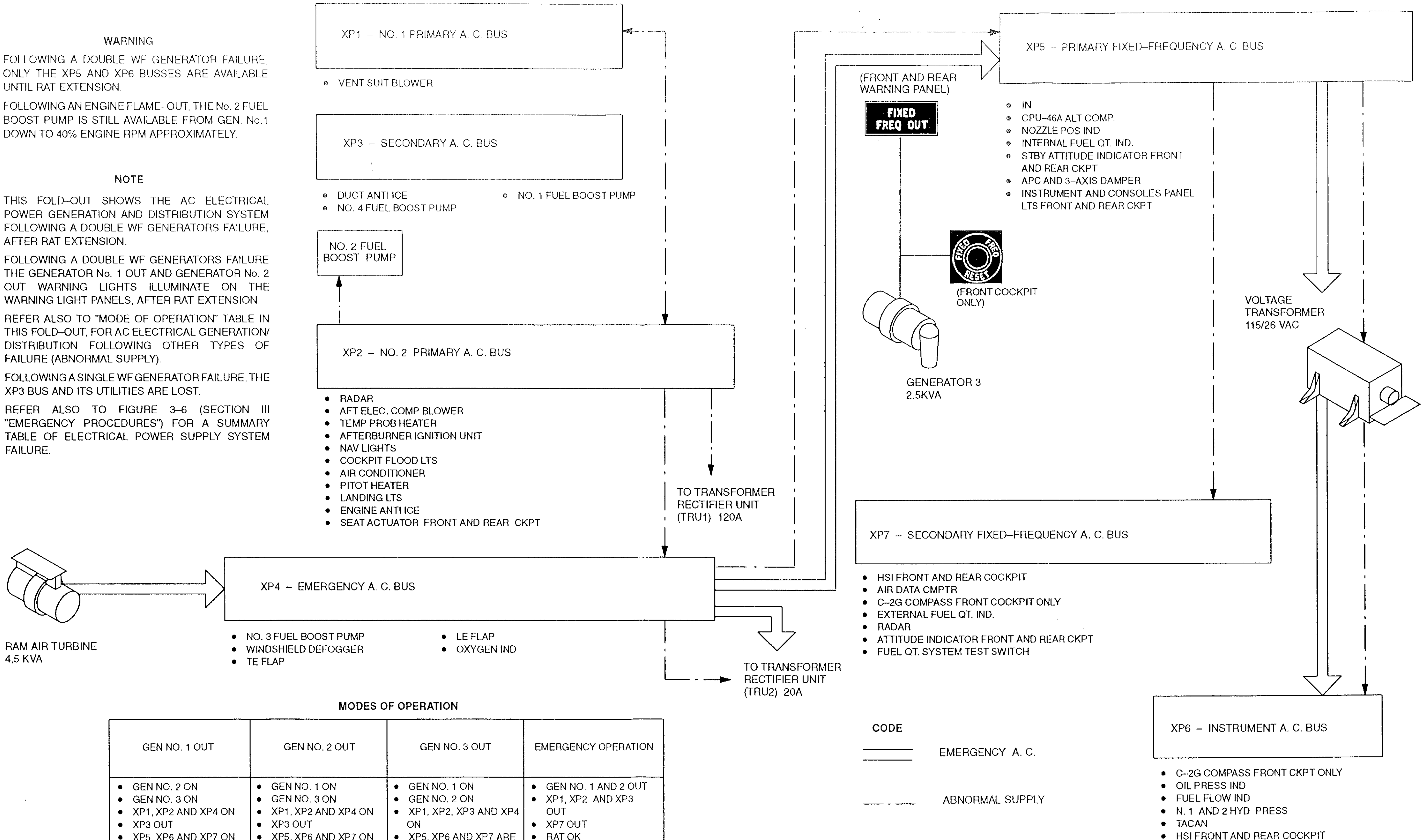
AC ELECTRICAL POWER SUPPLY SYSTEM EMERGENCY DISTRIBUTION

WARNING

- FOLLOWING A DOUBLE WF GENERATOR FAILURE, ONLY THE XP5 AND XP6 BUSSES ARE AVAILABLE UNTIL RAT EXTENSION.
- FOLLOWING AN ENGINE FLAME-OUT, THE No. 2 FUEL BOOST PUMP IS STILL AVAILABLE FROM GEN. No.1 DOWN TO 40% ENGINE RPM APPROXIMATELY.

NOTE

- THIS FOLD-OUT SHOWS THE AC ELECTRICAL POWER GENERATION AND DISTRIBUTION SYSTEM FOLLOWING A DOUBLE WF GENERATORS FAILURE, AFTER RAT EXTENSION.
- FOLLOWING A DOUBLE WF GENERATORS FAILURE THE GENERATOR No. 1 OUT AND GENERATOR No. 2 OUT WARNING LIGHTS ILLUMINATE ON THE WARNING LIGHT PANELS, AFTER RAT EXTENSION.
- REFER ALSO TO "MODE OF OPERATION" TABLE IN THIS FOLD-OUT, FOR AC ELECTRICAL GENERATION/ DISTRIBUTION FOLLOWING OTHER TYPES OF FAILURE (ABNORMAL SUPPLY).
- FOLLOWING A SINGLE WF GENERATOR FAILURE, THE XP3 BUS AND ITS UTILITIES ARE LOST.
- REFER ALSO TO FIGURE 3-6 (SECTION III "EMERGENCY PROCEDURES") FOR A SUMMARY TABLE OF ELECTRICAL POWER SUPPLY SYSTEM FAILURE.



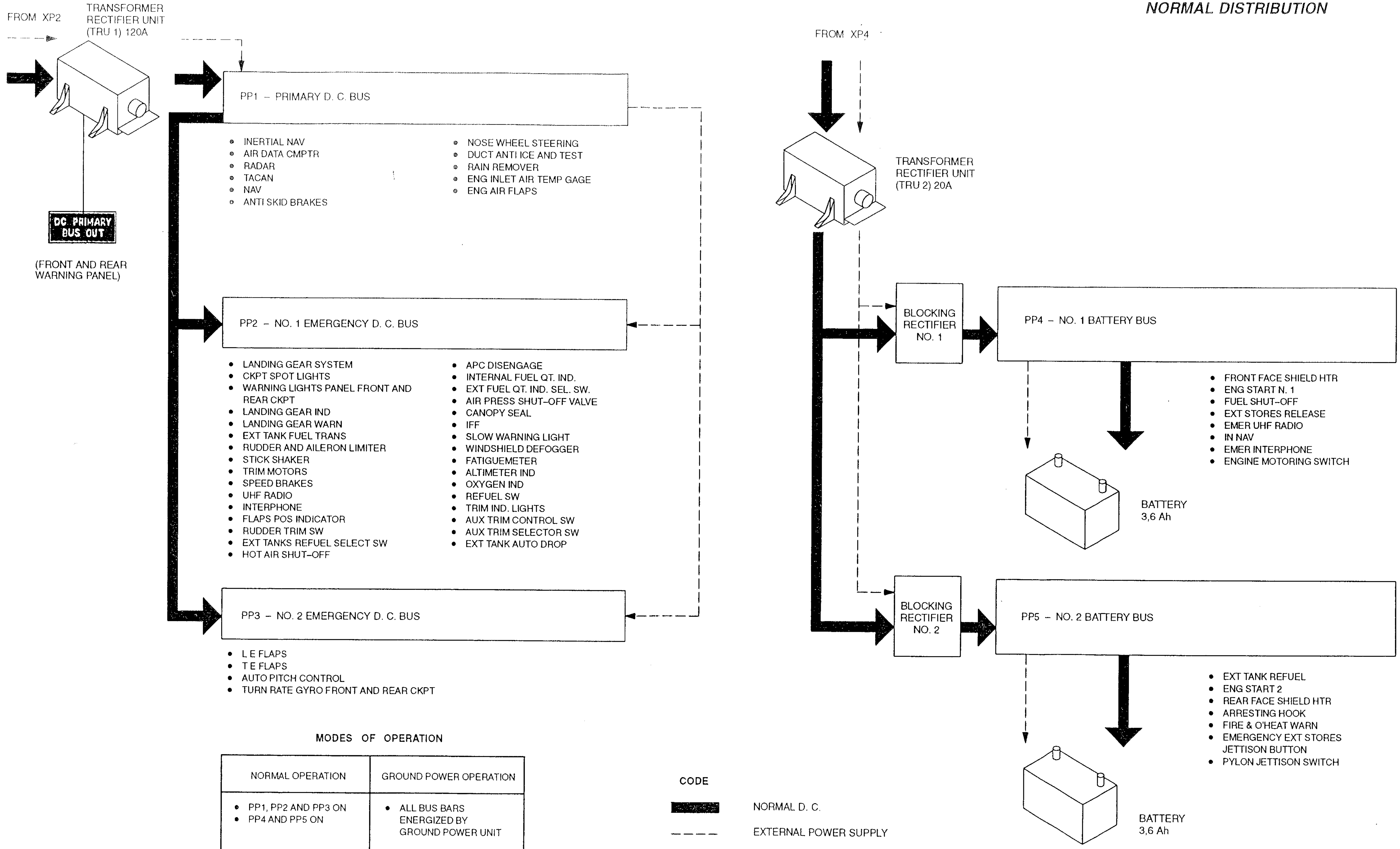
MODES OF OPERATION

GEN NO. 1 OUT	GEN NO. 2 OUT	GEN NO. 3 OUT	EMERGENCY OPERATION
• GEN NO. 2 ON • GEN NO. 3 ON • XP1, XP2 AND XP4 ON • XP3 OUT • XP5, XP6 AND XP7 ON • RAT INOPERATIVE	• GEN NO. 1 ON • GEN NO. 3 ON • XP1, XP2 AND XP4 ON • XP3 OUT • XP5, XP6 AND XP7 ON • RAT INOPERATIVE	• GEN NO. 1 ON • GEN NO. 2 ON • XP1, XP2, XP3 AND XP4 ON • XP5, XP6 AND XP7 ARE ENERGIZED BY XP4	• GEN NO. 1 AND 2 OUT • XP1, XP2 AND XP3 OUT • XP7 OUT • RAT OK • XP4, XP5 AND XP6 ON

Figure FO-10 (Sheet 2 of 2)

FA0306

DC ELECTRICAL POWER SUPPLY SYSTEM NORMAL DISTRIBUTION



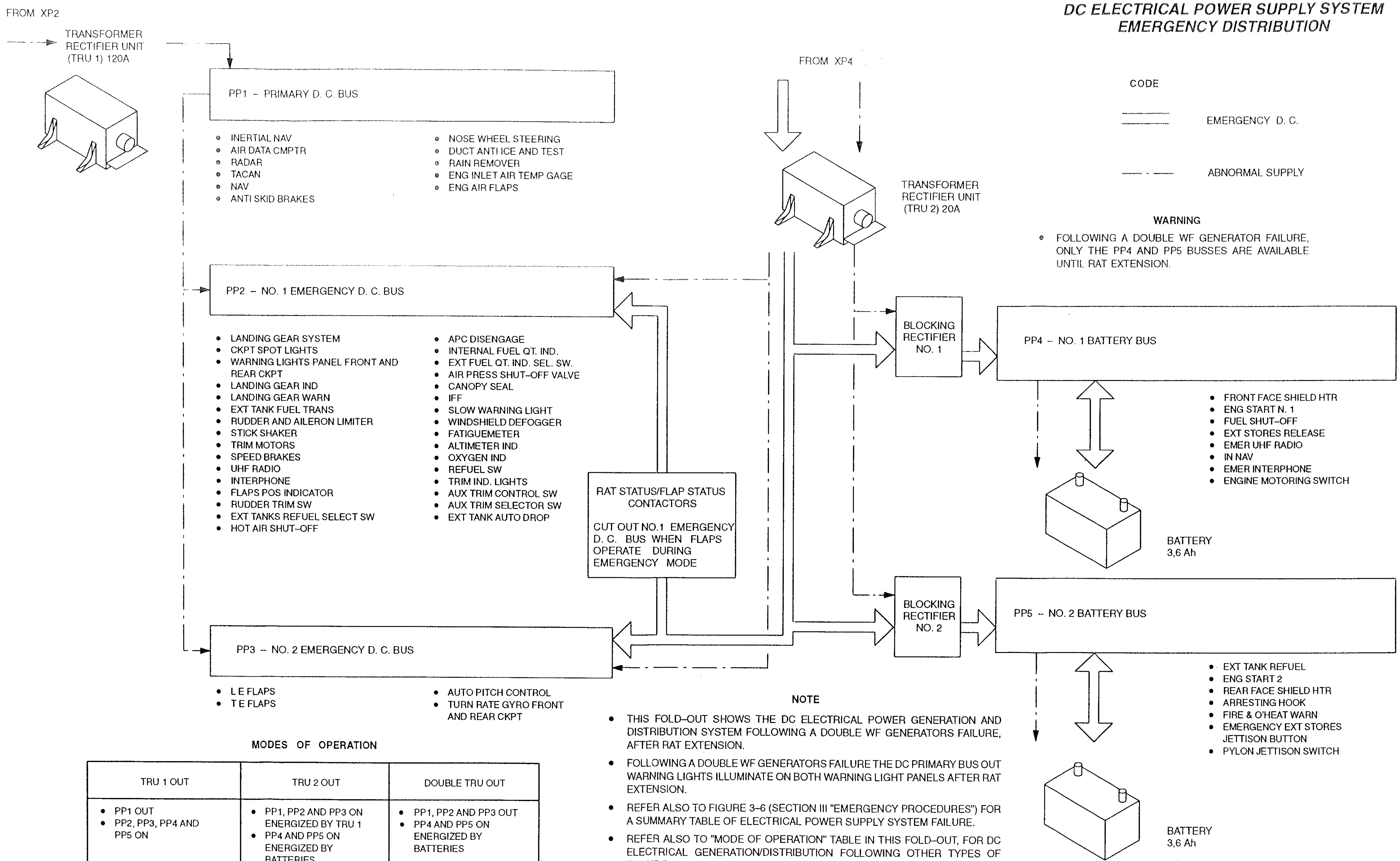
MODES OF OPERATION

NORMAL OPERATION	GROUND POWER OPERATION
• PP1, PP2 AND PP3 ON • PP4 AND PP5 ON	• ALL BUS BARS ENERGIZED BY GROUND POWER UNIT

Figure FO-11 (Sheet 1 of 2)

FA0019

DC ELECTRICAL POWER SUPPLY SYSTEM EMERGENCY DISTRIBUTION



WARNING

- FOLLOWING A DOUBLE WF GENERATOR FAILURE, ONLY THE PP4 AND PP5 BUSSES ARE AVAILABLE UNTIL RAT EXTENSION.

MODES OF OPERATION

TRU 1 OUT	TRU 2 OUT	DOUBLE TRU OUT
• PP1 OUT • PP2, PP3, PP4 AND PP5 ON	• PP1, PP2 AND PP3 ON ENERGIZED BY TRU 1 • PP4 AND PP5 ON ENERGIZED BY BATTERIES	• PP1, PP2 AND PP3 OUT • PP4 AND PP5 ON ENERGIZED BY BATTERIES

NOTE

- THIS FOLD-OUT SHOWS THE DC ELECTRICAL POWER GENERATION AND DISTRIBUTION SYSTEM FOLLOWING A DOUBLE WF GENERATORS FAILURE, AFTER RAT EXTENSION.
- FOLLOWING A DOUBLE WF GENERATORS FAILURE THE DC PRIMARY BUS OUT WARNING LIGHTS ILLUMINATE ON BOTH WARNING LIGHT PANELS AFTER RAT EXTENSION.
- REFER ALSO TO FIGURE 3-6 (SECTION III "EMERGENCY PROCEDURES") FOR A SUMMARY TABLE OF ELECTRICAL POWER SUPPLY SYSTEM FAILURE.
- REFER ALSO TO "MODE OF OPERATION" TABLE IN THIS FOLD-OUT, FOR DC ELECTRICAL GENERATION/DISTRIBUTION FOLLOWING OTHER TYPES OF FAILURE (ABNORMAL SUPPLY).

Figure FO-11 (Sheet 2 of 2)

FA0307

HYDRAULIC POWER SUPPLY SYSTEMS

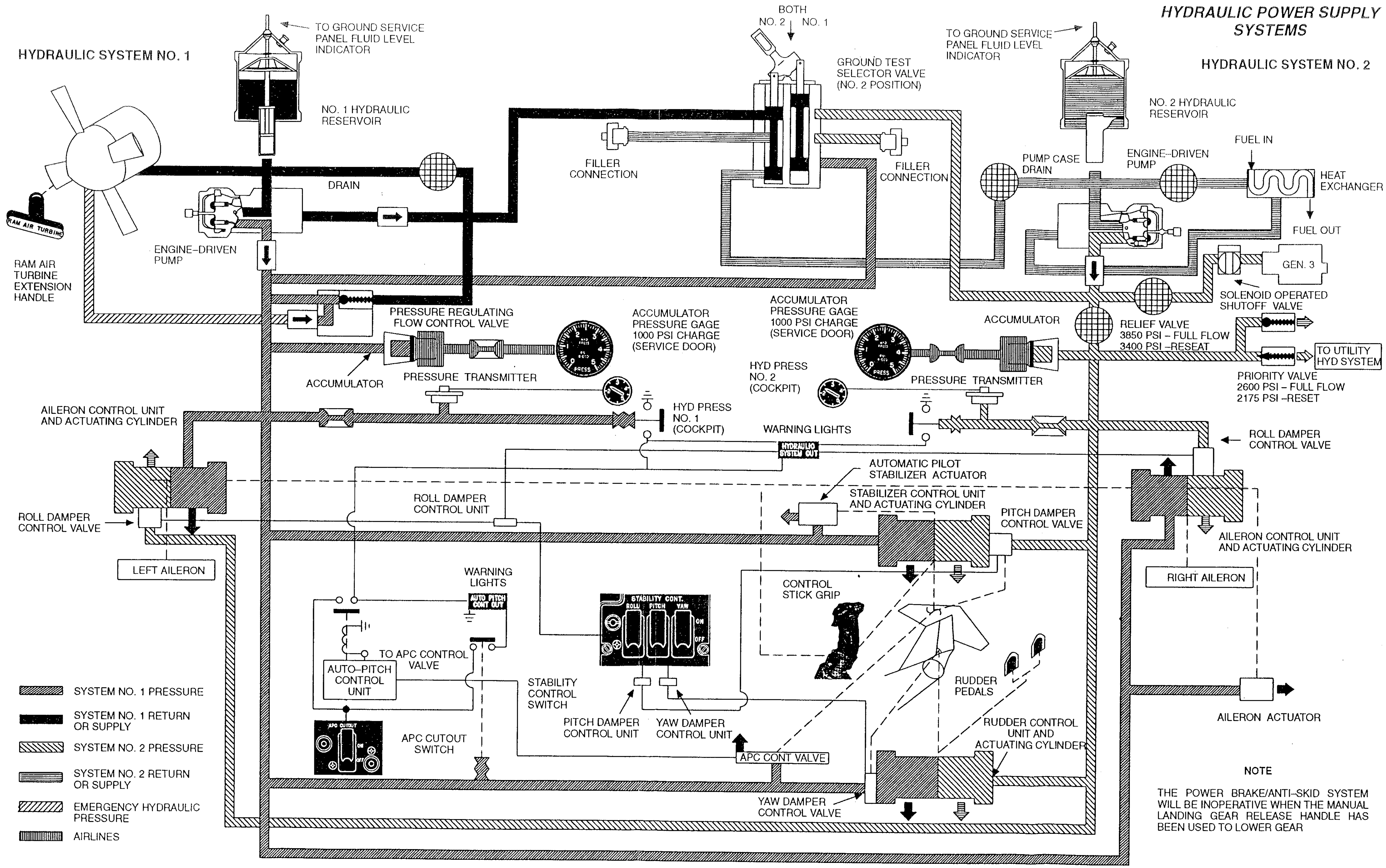


Figure FO-12

FA0020

UTILITY HYDRAULIC SYSTEM

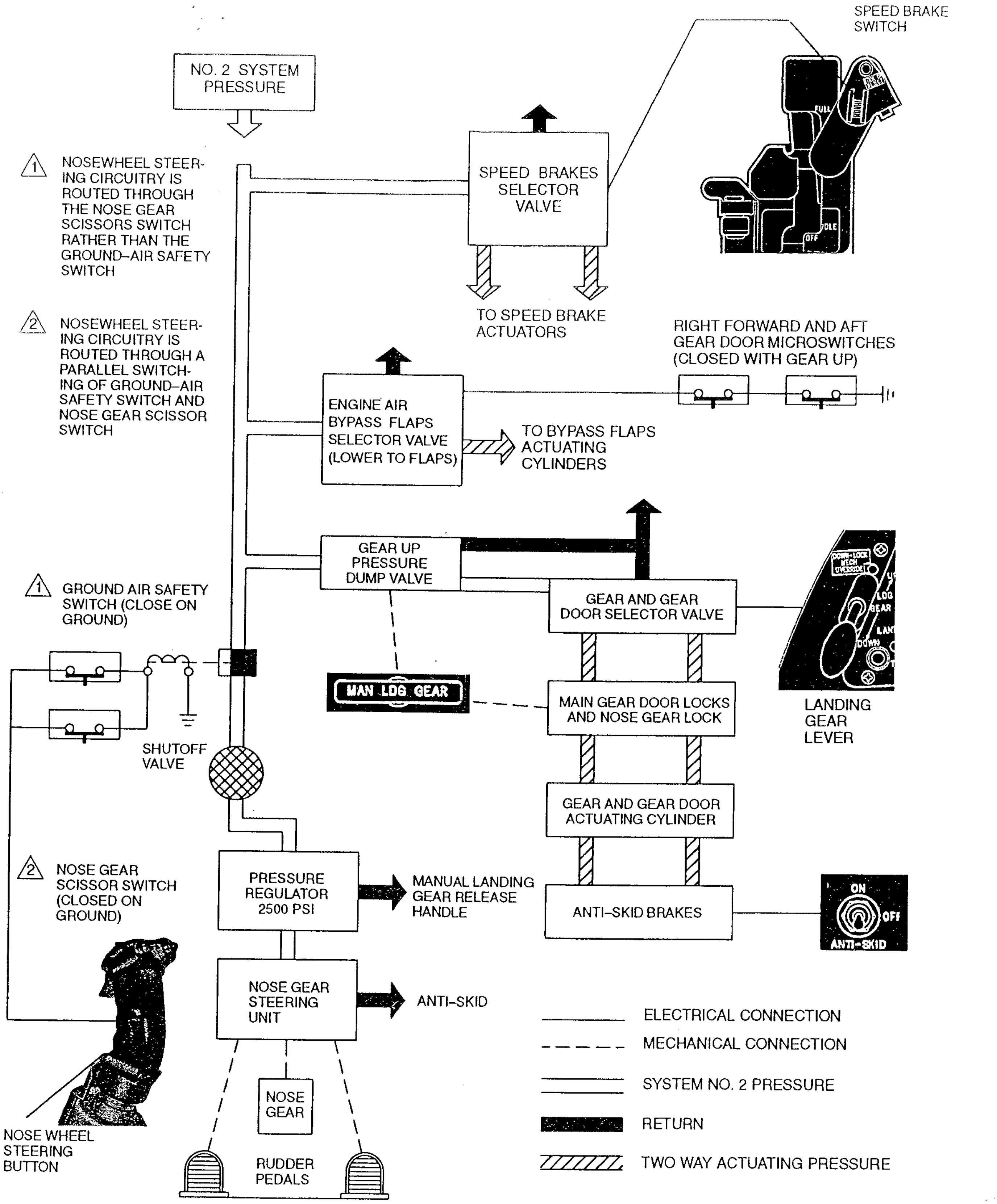
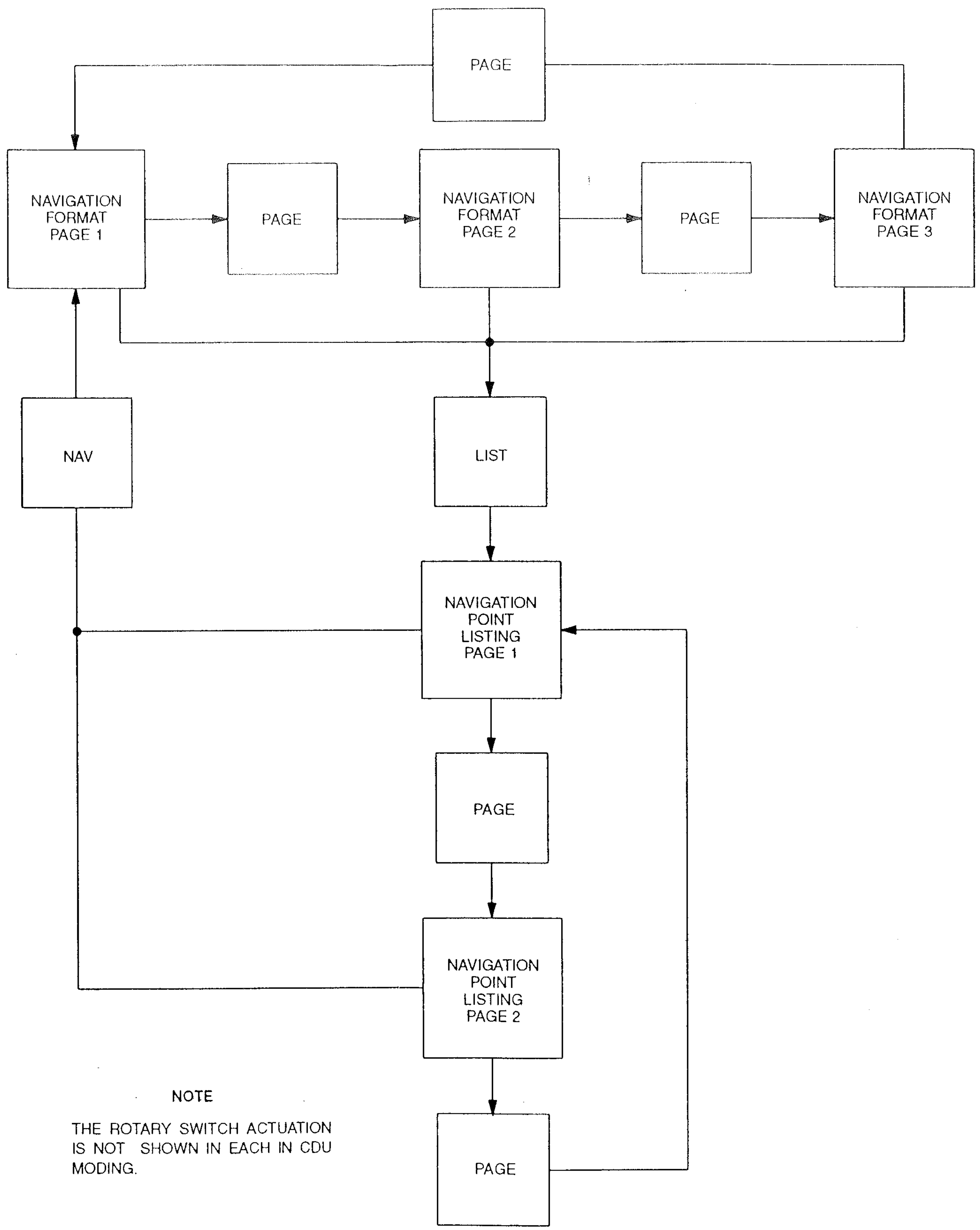


Figure FO-13

FA0078

IN/CDU NAVIGATION MODING BLOCK DIAGRAM

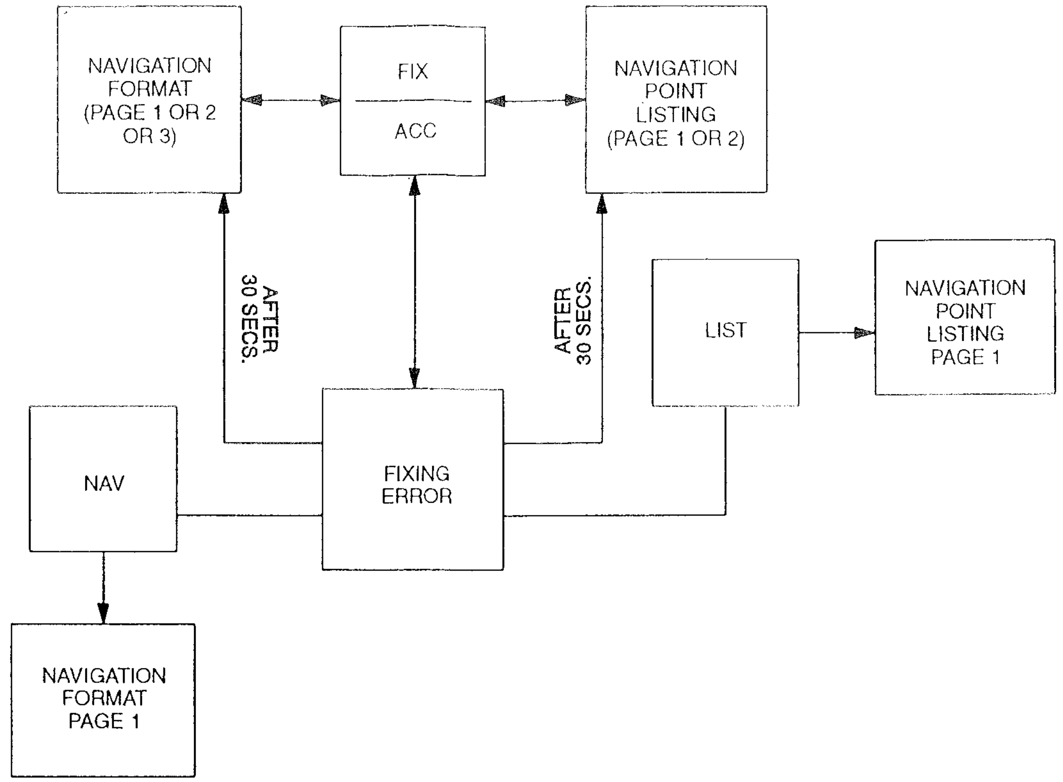
IN/CDU NAVIGATION/LIST MODING



NOTE

THE ROTARY SWITCH ACTUATION IS NOT SHOWN IN EACH IN CDU MODING.

IN/CDU ON TOP FIXING



IN/CDU MARK POINT DATA ACQUISITION

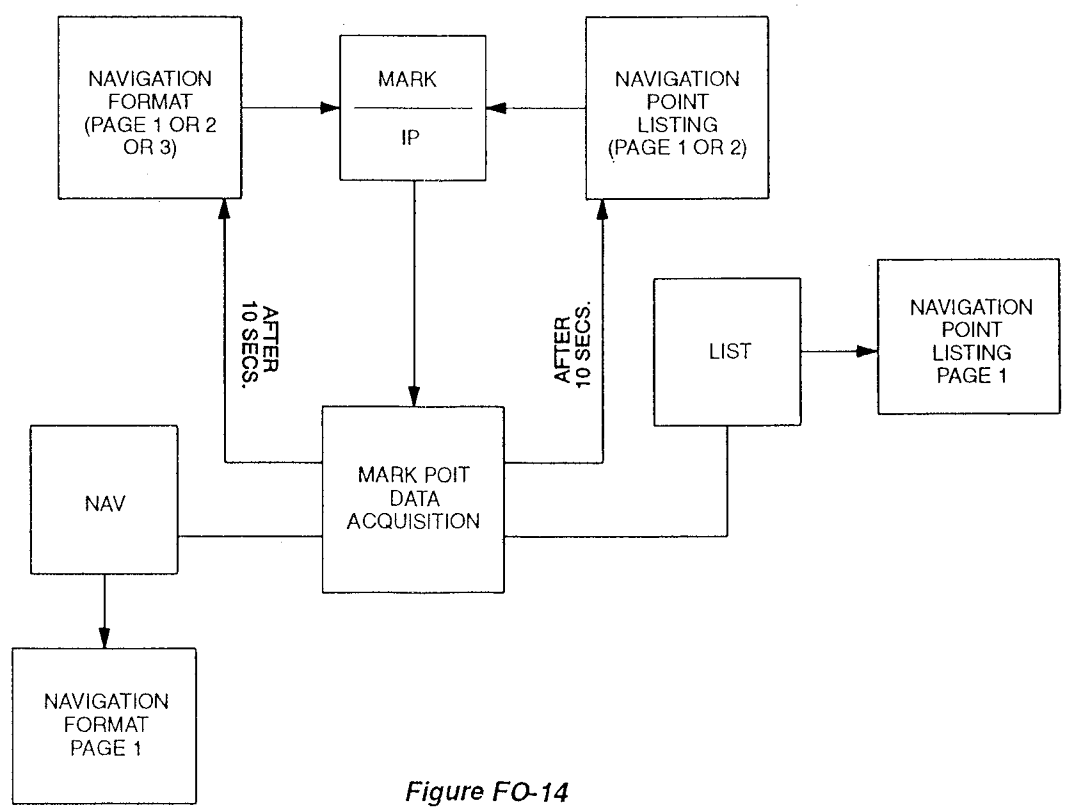


Figure FO-14

FA0265