COVER PAGE



F-104A 56-0734 AT NASA Edwards AFB 16 NOV 1960

LOCKHEED F-104A FLIGHT MANUAL

This manual was used in the NASA Edwards AFB Facility pilot's office by the pilots flying F-104 aircraft for research and X-15 support

This manual was donated by William H. Dana to the flight manual collection of Thomas R. Woodford in 1994 who created this scan in December 2024

NOTES:

Manual regraded as UNCLASSIFIED in accordance with Wright Air Development Division message dated 16 Mar 60 WWZ SPH-3-115-E. Markings not changed due to the historical value of this document

The binder contains the flight manual and confidential supplement along with a note from Lockheed inserted by Neil Armstrong concerning survival kits and time history charts of four "JF-104C" 55-2965 [which was a JF-104A] climb performance test runs

The safety of flight supplements bear the initials of the following legendary pilots who used this manual:

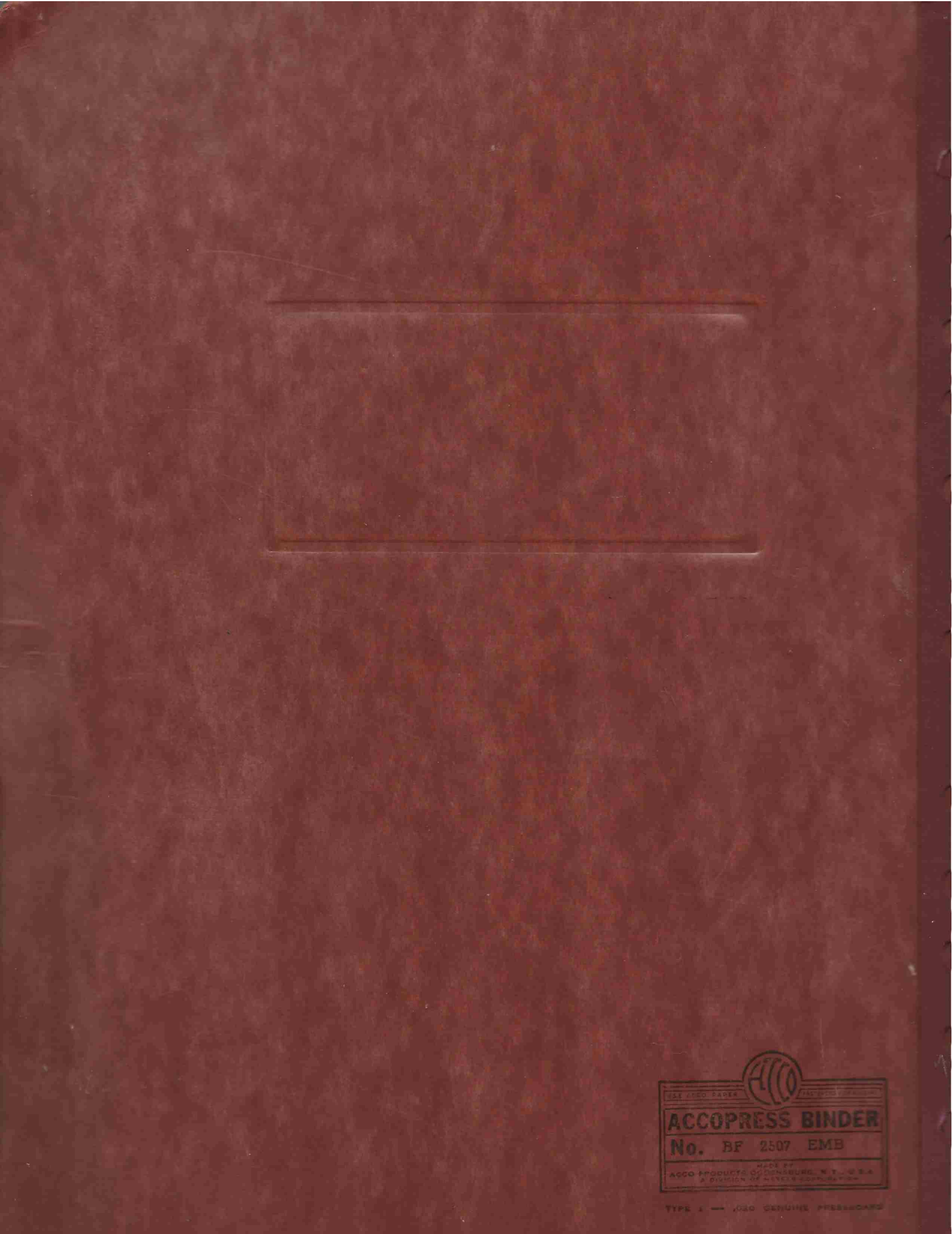
NAA – Neil A. Armstrong
JAW – Joseph A. Walker
FSP – Forrest S. Petersen, USN
JBM – John B. McKay
SPB – Stanley P. Butchart
MOT – Milton O. Thompson

THIS COVER PAGE ALLOWS VIEWING THE PDF DOCUMENT WITH THE PROPER FACING PAGES WHEN IN TWO-PAGE VIEW

COVER PAGE

PILOTS HANDBOOK

F-104A


ACCO
ACCOPRESS BINDER
No. BF 2507 EMB

FS FS FS FS FS FS FS FS FS FS FS FS FS FS FS FS

T.O. 1F-104A-(SF)1-2

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

NOTE COMMANDING OFFICERS ARE RESPONSIBLE FOR BRINGING THIS SUPPLEMENT TO THE ATTENTION OF ALL AF PERSONNEL CLEARED FOR OPERATION OF SUBJECT AIRCRAFT.

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2 SEPTEMBER 1960

NOTE

Effective 15 July 1960, Safety of Flight Supplement technical order numbers contain an (SF) and use consecutive suffix numbers (-1, -2, -3, etc.) instead of letters. Existing supplements will not be reissued for renumbering purposes only. See T.O. 0-1-1A for details.

1. PURPOSE.

To amplify existing start instructions so that starter destruction may be prevented.

2. GENERAL.

Several starters have disintegrated during attempted manual starts resulting in subsequent fire and aircraft loss. Repeated exposure to overspeed conditions (above 40% engine RPM) will cause starter fatigue and subsequent disintegration of the starter. Therefore, to avoid destruction of the starter, observe the following instructions.

FS FS FS FS FS SAFETY OF FLIGHT FS FS FS FS FS

3. INSTRUCTIONS.

a. Manual and battery starts, without auto-start control cable connected, are prohibited except when mission requirements dictate otherwise.

b. If a manual or battery start without auto-start control cable connected is required, it is imperative for the pilot and ground crewman to coordinate their actions to prevent overspeed of the starter. The pilot must signal the ground crewman at 40% engine RPM to disconnect external air immediately.

END

N. ARMSTRONG LOCKHEED INFO.

Subject: PRODUCTION AND INTERIM SURVIVAL KIT - F104

FS/217125-C..Two models of the survival kit are now installed in F104 aircraft using the C or C-1 seat. The first type, know as the "Interim Survival Kit", is for use on F104A aircraft 1001 through 1035. The second type, known as the "Production Survival Kit" is for use on F104A aircraft 1036 and subsequent, also F104B aircraft 5001 and subsequent.

The following information covers operation and maintenance of both types of survival kits. Those items which do not apply to the Interim kit are marked "N/A Interim".

I. Preflight and Post-flight Checks.

A. Pilot's Preflight Check List.

1. Security of quick disconnect, which provides ship's oxygen to regulator mounted in survival kit. This disconnect is mounted onto a "T" bracket on the rear of the seat.

a. Attachment of lanyard at airplane structure.

b. Manual disconnect know must be in locked position.

(N/A Interim)

2. Attachment of breakaway fitting to right leg restraint. During bailout this automatically turns on emergency oxygen supply.

3. Oxygen cylinder pressure in kit at 1800 psi. Gauge is visible through small round window in parachute support section of survival kit.

4. Security of let-out hose at kit.

(N/A Interim)

5. Security of rocket jet fittings on kit.

6. Security of hoses (tinnerman clamps) from survival kit disconnect to suit and mask.

7. Check between sides of bucket and kit for tools or foreign matter.

8. Connect and lock kit to parachute.

9. Connect and lock mask and radio.

10. If wearing MC-4 flight suit and pressure helmet, before flight, while seated on survival kit, first hookup survival kit to "D" ring of parachute harness, then connect hoses to face plate (for breathing oxygen) and suit (capstan pressure).

If wearing MC-1 flight suit with A-13A mask, after hooking up survival kit to parachute harness connect mask adaptor with conversion leaf for intercommunication.

B. Pilot's Post-flight Check List

1. Recap the kit hose outlets.

2. Manually disconnect the parachute from the kit at the "D" rings.

C. Crew's Preflight Check List

1. Let out the pelvic restraint straps and lay forward.

2. Place the kit in the bucket.

3. Mate and lock the quick disconnect to the ship's supply, that extends from the aft left corner of the ship, turn full left 90° for locked and 90° to right for unlocked.

(N/A Interim)

4. Connect and lock the breakaway fitting to the right leg brace.

5. Check the security of the disconnect lanyard at the point of attach to the ship and the disconnect.

6. Radio Checkout

D. Caution for Flight Line Crews
(N/A Interim)

1. Do not pull the kit release or harness release handles.

2. Do not open the kit except to remove the adapter, located in rear compartment.

3. Do not pull the green apple.

4. Do not remove caps from the hoses, (let the pilot do this).

5. Do not pull on the hose.

6. Do not put tools, etc., between the kit and the sides of bucket.

7. Do not remove the kit without first disconnecting the quick disconnect and the breakaway connection.

8. Do not leave the panel oxygen valve in the ON position.

II. Duration of Oxygen

A. Ship's Oxygen

The duration of oxygen supply with the standard 1800 psi high pressure system when full is:

Cabin Altitude	Duration of Oxygen
5,000 Feet	0.8 Hours
10,000	0.9
15,000	1.2
20,000	1.5
25,000	1.9
30,000	2.5
35,000	3.2
40,000 and up	3.2

When the pressure drops below 400 psi consider it an emergency and descend to an altitude that does not require oxygen.

B. Emergency Supply of Oxygen

The duration of oxygen supply when used from the emergency bottles stowed in the survival kit will depend on the altitude when used. With an MC-4 partial pressure suit and an MA-2 helmet the following duration may be expected:

ALTITUDE	Duration
10,000 Feet	4.1 Minutes
20,000	6.2
30,000	11.0
40,000	18.5

III. Instructions on Pressure Breathing

A. With the MC-3 or MC-4 Partial Pressure Suit and MA-2 Helmet

The activation of pressure in the helmet with counter pressure in the suit and breathing bladders is entirely automatic, and requires no effort on the part of the pilot. When breathing under the inflated condition at altitude no special technique is required.

The regulator will automatically provide the necessary increase in pressure over ambient regardless of altitude. It is recommended however, that all personnel using any of the pressure suits have been given pressure suit indoctrination at an Air Force Physiological Training Unit.

B. If the aircraft is operated with an A-13-A mask and adapter, pressure breathing in the usual sense will result when flying over 40,000 feet cabin pressure. When pressure breathing in this instance, forced expiration is required with passive inspiration. Rate of breathing must be controlled so that excessive depletion of carbon dioxide will not resutl.

CONFIDENTIAL

FIGURE NO.

ZOOM CLIMB PERFORMANCE

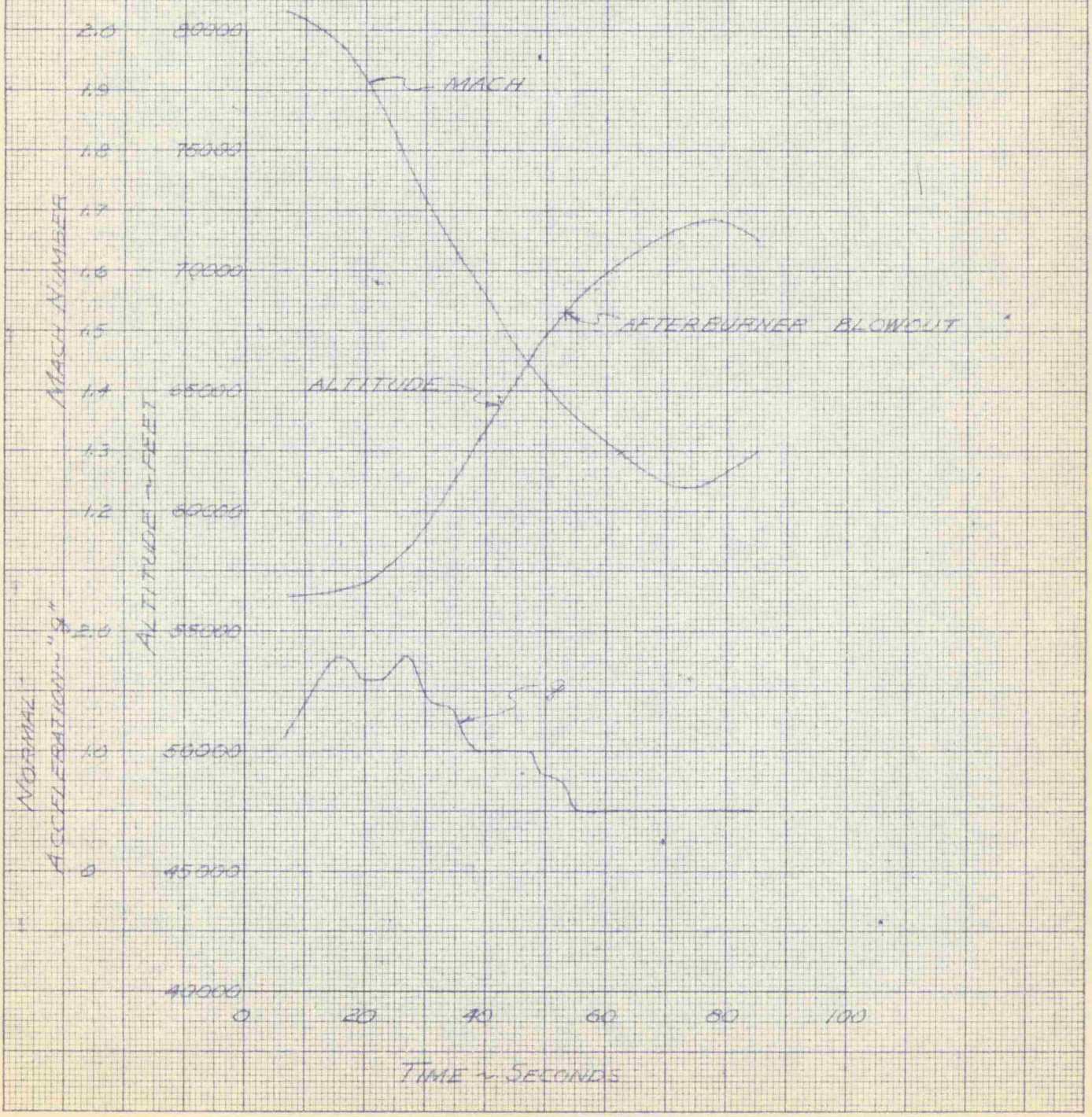
JF-104C USAF NO. 55-2965

YJ-79-GE-7 ENGINE

MAXIMUM POWER

SIDEWINDER MISSILES

GROSS WEIGHT AT PULL UP = 15590 LBS.



10 X 10 TO THE CM. 359T-14G
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE

CONFIDENTIAL

CONFIDENTIAL

FIGURE NO.

ZOOM CLIMB PERFORMANCE

JA-104C USAF NO. 55-2965

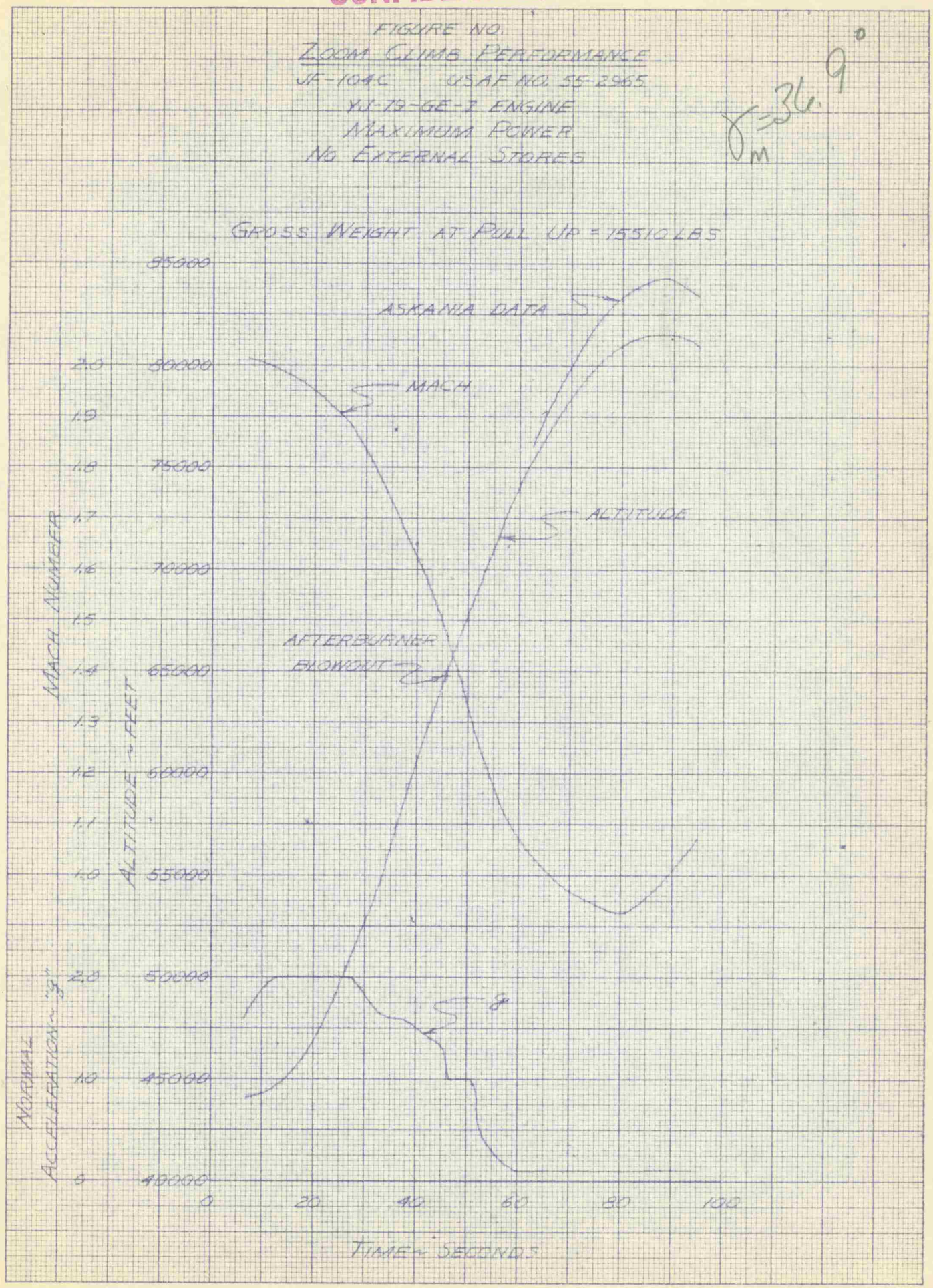
XJ-79-GE-3 ENGINE

MAXIMUM POWER

NO EXTERNAL STORES

$\gamma_m = 36.9°$

GROSS WEIGHT AT PULL UP = 15510 LBS



K+E 10 X 10 TO THE CM. 359T-14G KEUFFEL & ESSER CO. MADE IN U.S.A. ALBANENE (R)

CONFIDENTIAL

CONFIDENTIAL

FIGURE NO.

ZOOM CLIMB PERFORMANCE

F-101C USAF NO. 56-2965
YJ-79-GE-7 ENGINE
MAXIMUM POWER
NO EXTERNAL STORES

ASKANIA DATA

MACH

ALTITUDE

AFTERBURNER BLOWOUT

GROSS WEIGHT AT
FULL CAP = 15470 LBS

MACH NUMBER

ALTITUDE - FEET

NORMAL ACCELERATION - "g"

TIME - SECONDS

2.0 1.9 1.8 1.7 1.6 1.5 1.4 1.3 1.2 1.1 1.0

50000 55000 60000 65000 70000 75000 80000 85000 90000

2.0 1.0 0

50000 45000 40000

0 20 40 60 80 100

K&E 10 X 10 TO THE CM. 359T-14G KEUFFEL & ESSER CO. ALBANENE MADE IN U.S.A.

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FIGURE NO.

ZOOM CLIMB PERFORMANCE

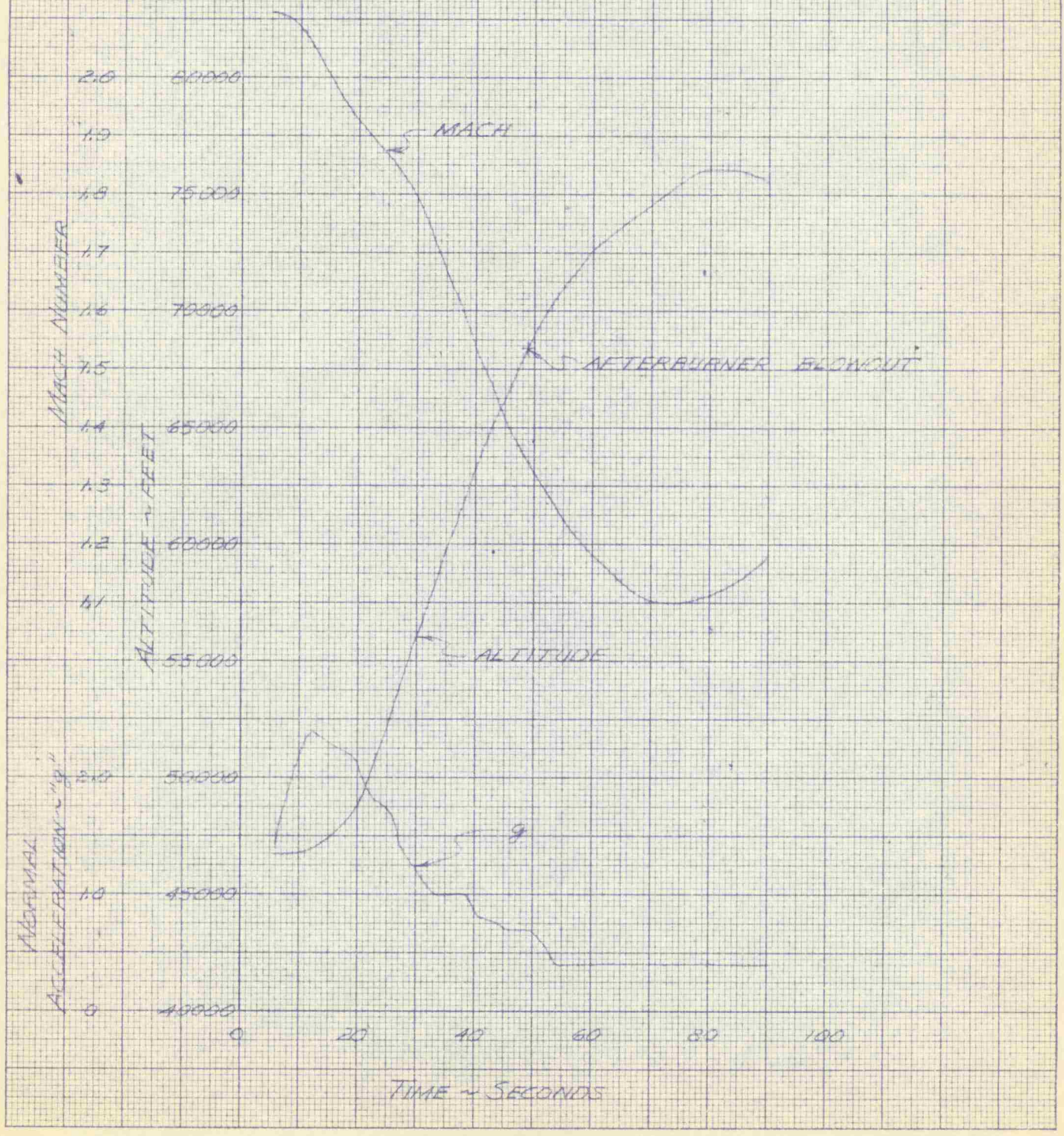
JF-104C USAF NO. 55-2965

XJ-79-GE-7 ENGINE

MAXIMUM POWER

SIDEWINDER MISSILES

GROSS WEIGHT AT PULL UP = 15450 LBS.



CONFIDENTIAL

K&E 10 X 10 TO THE CM. KEUFFEL & ESSER CO. ALBANY, N.Y. MADE IN U.S.A. 359T-14G

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T.O. 1F-104A-(SF)1-1

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T. O. 1F-104A-1. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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★

25 JULY 1960

1. PURPOSE.

To impose temporary airspeed restriction.

2. GENERAL.

In-flight failure of the first stage nozzle diaphragm (J79 engine) occurred during recent flight testing. Subsequent analysis and static load testing confirmed that the diaphragm assembly is subjected to stresses beyond its capability at high Mach number and low altitudes. Until corrective action can be accomplished, the following instructions shall be observed.

3. INSTRUCTIONS.

Do not exceed Mach 1.0 below 10,000 feet.

END

AF SMAMA Apr 60, 50M

AF SMAMA Jul 60, 1800

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

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18 MARCH 1960

1. PURPOSE.

To provide for IFF check when airplane is to operate within a radar advisory area, and to establish correct instructions for resetting of altimeter when approaching 24,000 feet.

2. GENERAL.

a. To insure safety of flight, all aircraft operating within a radar advisory area on a VFR, DVFR, or VFR on top clearance, are required by Air Route Traffic Control to have a functional IFF. In the event the IFF is inoperative, an aircraft may proceed through a radar advisory area only if ARTC has granted approval.

b. When a radar facility is available, the IFF should be checked for normal operation twice during each flight. The first check must be performed after take-off and prior to entering radar advisory areas. The second check should be within one hour prior to landing.

c. The check consists of ascertaining that the IFF transmitter is functioning normally and being properly received by the ground interrogator.

SAFETY OF FLIGHT

d. Present regulations require usage of an altimeter setting of 29.92 when flying at or above 24,000 feet while within the ZI. To aid in assuring that this is accomplished and proper traffic control is maintained, correct instructions concerning resetting the altimeter shall be included in the checklist.

3. INSTRUCTIONS.

a. Add the following item to the "Before Taxiing" checklist:

IFF – STANDBY.

b. Revise the "After Take-Off – Climb" checklist to include the following:

IFF – Check.

If positive operation of the normal mode of IFF has not been established during departure with an air traffic control facility, a check should be made with such a facility, as soon after take-off as flight conditions permit. This check must be made prior to entering a radar advisory area. If IFF is inoperative, consult the appropriate navigation publications.

Altimeter – Reset to 29.92 (or as required) when climbing through 23,500 feet.

c. Revise the "Descent" checklist to include the following:

IFF – Check.

Check IFF for positive operation within one hour prior to the estimated time of landing.

Altimeter – Reset to the altimeter setting at point of descent when descending through flight level 240.

END

Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs Fs

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

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27 NOVEMBER 1959

1. PURPOSE.

To provide information and instructions concerning possible engine flameout under conditions of low airspeed, high altitude, and turbulent air.

2. GENERAL.

A recent accident occurred as a result of an afterburner blowout and subsequent engine flameout during turbulent air penetration at high altitude and low airspeed. In turbulent air, inlet duct flow disturbances occur which momentarily change the engine inlet pressure. At high altitude and low airspeed, engine operation is more susceptible to the pressure surges caused by the air flow disturbances; therefore, the possibility of afterburner blowout and engine flameout is greatly increased. Afterburner operation in sector burning increases the possibility even further. In addition to the present instructions in Section IX under "Turbulence and Thunderstorms", the following instruction shall be observed.

3. INSTRUCTIONS.

a. Maintain a minimum of 275 KIAS during turbulent air conditions above 40,000 feet.

Fs Fs Fs Fs Fs SAFETY OF FLIGHT Fs Fs Fs Fs Fs

WARNING

Experience indicates that at high altitude critical engine operation is rapidly approached as airspeed drops below 275 KIAS in turbulent air because of the increased effect of inlet airflow distortion. Therefore, the minimum acceptable speed for operation in turbulent air is 275 KIAS.

b. If afterburner thrust is necessary, maintain full afterburner.

WARNING

Do not attempt turbulent air penetration using sector burning because of the high probability of afterburner blowout and possible subsequent engine flameout.

END

T.O. 1F-104A-1FN

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

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2 OCTOBER 1959

1. PURPOSE.

To provide instructions for pre-flight check of the MM-2 attitude indicator.

2. GENERAL.

As a result of numerous in-flight malfunctions of the MM-2/K4B attitude gyro system, a pre-flight check of the system should be accomplished to ascertain if the MM-2 attitude indicator is reliable for flight. Therefore, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Warning "OFF" flap retracted within 2 1/2 minutes.

b. Horizon line for proper attitude and freedom from oscillation (energized).

c. Horizon line for proper response to trim knob.

WARNING

If the warning "OFF" flag requires longer than 2 1/2 minutes to retract, or if any oscillations are noted on the indicator after the "OFF" flag has retracted, a possible malfunction exists. Either of the above is cause for rejection of the indicator and should be entered in the Form 781.

END

SAFETY OF FLIGHT

T.O. 1F-104A - 1FM

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL
USAF SERIES

F-104A

AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T. O. 1F-104A-1 AND REPLACES FORMAL SAFETY OF FLIGHT SUPPLEMENT T. O. 1F-104A-1FK, DATED 18 DECEMBER 1959, BECAUSE OF TYPOGRAPHICAL ERROR IN DATE OF ISSUANCE. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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10 SEPTEMBER 1959

1. PURPOSE.

To provide additional instructions for pre-flight check of the boundary layer control (BLC) system.

2. GENERAL.

To assist in detecting BLC system malfunction (crushed ducts, stuck valve, etc.) prior to flight, the following check shall be performed with the assistance of the ground crew during the AUTO-PITCH, STICK SHAKER, AND STABILITY AUGMENTATION CONTROL SYSTEM CHECK.

3. INSTRUCTIONS.

a. Wing flap lever - LAND.

With the flaps in the landing setting, have ground crew check BLC for proper operation and for possible warpage of the BLC ducts. During normal BLC operation at idle rpm, an EGT rise of approximately 40°C should occur.

b. Wing flap lever - TAKEOFF.

After the flaps have reached the take-off setting, have ground crew verify flap position and check for absence of BLC airflow and that ducts have not been crushed.

NOTE
If any malfunction is detected, the flight should be aborted.

END

AF SMAMA Oct 59, 2,300

A.F. SMAMA 4/16/59—100M

SAFETY OF FLIGHT

T.O. 1F-104A-1FL

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

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18 SEPTEMBER 1959

1. PURPOSE.

To provide instructions in the event of main gear tire failure during take-off.

2. GENERAL.

Main gear tire failure on take-off may present more problems than tire failure on landing. Directional control is more difficult and braking efficiency is greatly reduced at higher gross weights with failure of one or both main gear tires. Therefore, under certain conditions the take-off should be continued rather than aborted. If main gear tire failure occurs during take-off, the following instructions shall be observed.

3. INSTRUCTIONS.

a. If nose wheel lift-off speed has been attained, external load should be retained and take-off continued.

b. If speed is greater than 150 KIAS but below nose wheel lift-off speed, jettison external load and continue take-off.

c. If speed is less than 150 KIAS, abort take-off.

(See page 2 for WARNING)

SAFETY OF FLIGHT

WARNING

If take-off is continued, the landing gear should not be retracted if tire has failed or suspected to have failed, until tire has been visually checked for fire by a report from another airplane or the tower. After the tire is checked and if the gear is to be retracted, the wheel brakes should be applied to stop wheel rotation before retraction to prevent tire fragments from damaging equipment in the wheel well. Landing should be accomplished in accordance with the instructions for "Main Gear Tire Failure on Landing" presently contained in the Flight Manual.

END

T.O. 1F-104A-1FJ

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1 AND REPLACES FORMAL SAFETY OF FLIGHT SUPPLEMENT T.O. 1F-104A-1FG, DATED 29 JULY 1959, WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

NOTE COMMANDING OFFICERS ARE RESPONSIBLE FOR BRINGING THIS SUPPLEMENT TO THE ATTENTION OF ALL AF PERSONNEL CLEARED FOR OPERATION OF SUBJECT AIRCRAFT.

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12 AUGUST 1959

1. PURPOSE.

To revise the instructions for lowering flaps to the landing setting prior to landing.

2. GENERAL.

Several incidents have occurred recently in which the boundary layer control (BLC) duct was crushed as the flaps extended from the takeoff setting to landing setting. The resulting asymmetrical BLC airflow constitutes a serious accident potential because of the strong rolling moment at low altitude. Therefore, to minimize this dangerous situation and to provide a means of taking immediate corrective action, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Under the DOWNWIND portion of the BEFORE LANDING checklist, add the following:

Wing flap lever – LAND below 240 KIAS and above 210. Keep hand on wing flap lever until it is determined that BLC and flaps are functioning normally.

A.F. SMAMA 6/16/59—100M—(OFF)

A.F. SMAMA 8/12/59—2,500

WARNING

If BLC malfunction occurs, as detected by a strong rolling moment, immediately return wing flap lever to TAKEOFF position and follow existing instructions in the Flight Manual.

Under various conditions of heavy gross weight or high ambient temperatures, with flaps in the landing setting, sufficient thrust may not be available at full Military thrust to maintain proper rate of descent and airspeed during turn from downwind to final. In the event this condition develops, immediately return wing flap lever to TAKEOFF and if conditions do not appear favorable for continuation of the approach, initiate go-around and plan subsequent approach for a "takeoff flaps" landing. For a takeoff flaps landing fly final approach at not less than 190 KIAS. When this condition is suspected before landing, flaps should be retained at the takeoff setting and a "takeoff flaps" landing made.

b. Under the BASE LEG portion of the BEFORE LANDING checklist, delete the following step:

Wing flap lever – LAND below 240 KIAS.

END

T.O. 1F-104A-1FD

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

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13 May 1959

1. PURPOSE.

To provide instructions on use of oil low level warning light installed with priority system.

2. GENERAL.

The priority oil system consists of a modification to the oil tank pendulum, which provides a reserve oil supply in the event an oil leak occurs in the engine hydraulic system. Concurrently with the installation of the priority oil system, a low level warning light was installed in the cockpit. The low level light will come on when the oil level is down to 0.8-gallon remaining. This amount of oil is sufficient to lubricate the engine for approximately two hours, assuming a maximum oil consumption of two pounds per hour. If an oil leak occurs in the lubrication system, no reserve oil will be available. By using both the light indication and oil pressure indication, the pilot can determine in what system the oil leak occurred and what to expect with respect to engine operation. For example:

a. Low level light ON, oil pressure remains normal - the leak is in the hydraulic system and the pilot can expect the nozzle to fail to open. Under this condition, the pilot should use the existing procedures for an open nozzle failure.

b. Low level light ON, and oil pressure subsequently dropping - the leak is in the lube system and bearing failure probably will occur in approximately 4 - 6 minutes, depending upon bearing condition prior to loss of oil and operating conditions. Under this condition, the pilot should use the existing procedure for an oil system failure.

In view of the above, the following procedures apply in the event the low level light comes on.

A.F. SMAMA 5/14/59—2,400
A.F. SMAMA 4/16/59—100M

SAFETY OF FLIGHT

3. INSTRUCTIONS.

a. If light comes on prior to take-off - abort the flight.

b. If light comes on during take-off and sufficient runway remains - abort the take-off.

c. If the light comes on after committed to the take-off or while in flight, the pilot should proceed to the nearest suitable field and land as soon as possible. Dependent upon the oil pressure indication, use either the oil system failure or the nozzle open failure procedure, whichever applies.

END

T.O. 1F-104A-1FC

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

USAF SERIES

F-104A

AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1 AND REPLACES INTERIM SAFETY OF FLIGHT SUPPLEMENT T.O. 1F-104A-1EW, DATED 23 MARCH 1959, WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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5 MAY 1959

1. PURPOSE.

To modify instructions relative to flame-out landings and to provide basic criteria to aid the pilot in determining the best course of action - ejection, or forced landing.

2. GENERAL.

Normally, ejection is the best course of action with a windmilling or frozen engine. Due to the excessive high rates of descent and airspeeds required to maintain control, it is NOT advisable to attempt a dead stick landing. If an air start cannot be effected, it is RECOMMENDED that the pilot eject. Because of the many variables encountered, the final decision to attempt a flame-out landing, or to eject, must remain with the pilot. It is impossible to establish a pre-determined set of rules and instructions which would provide a ready made decision applicable to all emergencies of this nature. The basic conditions listed below, combined with the pilot's analysis of the condition of the airplane, type of emergency, and his proficiency are of prime importance in determining whether to attempt a flame-out landing, or to eject. These variables make a quick and accurate decision difficult. If the decision is made to eject, prior to

A.F. SMAMA 5/4/59—2,550
A.F. SMAMA 2/19/59—100M

ejection, if possible, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or property on the ground or water is least likely to occur. Before a decision is made to attempt a flame-out landing, the following basic conditions should exist.

3. INSTRUCTIONS.

a. Flame-out landings should only be attempted by pilots who have satisfactorily completed simulated flame-out approaches in this airplane.

b. Flame-out landings should only be attempted on a prepared or designated suitable surface.

c. Approaches to the runway should be clear and should not present a problem during a flame-out approach.

NOTE

No attempt should be made to land a flamed-out airplane at any field whose approaches are over heavily populated areas, if a suitable area is available to abandon the airplane. If possible, prior to ejection, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or property on the ground or water is least likely to occur.

d. Weather and terrain conditions must be favorable. Cloud cover, ceiling, visibility, turbulence, surface wind, etc., must not impede in any manner the establishment of a proper flame-out landing pattern.

NOTE

Night flame-out landings, or flame-out landings under poor lighting conditions such as at dusk or dawn, should not be contemplated regardless of weather or field lighting.

e. Flame-out landings should only be attempted when either a satisfactory "High Key" or "Low Key" position can be achieved.

f. If at any time during the flame-out approach, conditions do not appear ideal for successful completion of the landing, ejection should be accomplished. EJECT no later than the "Low Key" altitude.

END

SAFETY OF FLIGHT SUPPLEMENT

FLIGHT MANUAL

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AIRCRAFT

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28 APRIL 1959

1. PURPOSE.

To warn of possible misset altimeters.

2. GENERAL.

It is possible to set a majority of the standard altimeters in USAF use in error by 10,000 feet. This happens when the barometric set knob is continuously rotated after the baro scale is out of view. The knob can be rotated until eventually the numbers will reappear in the Kollsman window from the opposite side. If the correct altimeter setting is then established, the altimeter will read approximately 10,000 feet in error.

3. INSTRUCTIONS.

As a preflight check, special attention should be given to the altimeter to assure that the 10,000 foot pointer is reading correctly. See figure 1, page 2, for example of correctly set and misset altimeters.

A.F. SMAMA 4/27/59—2,500
A.F. SMAMA 2/19/59—100M

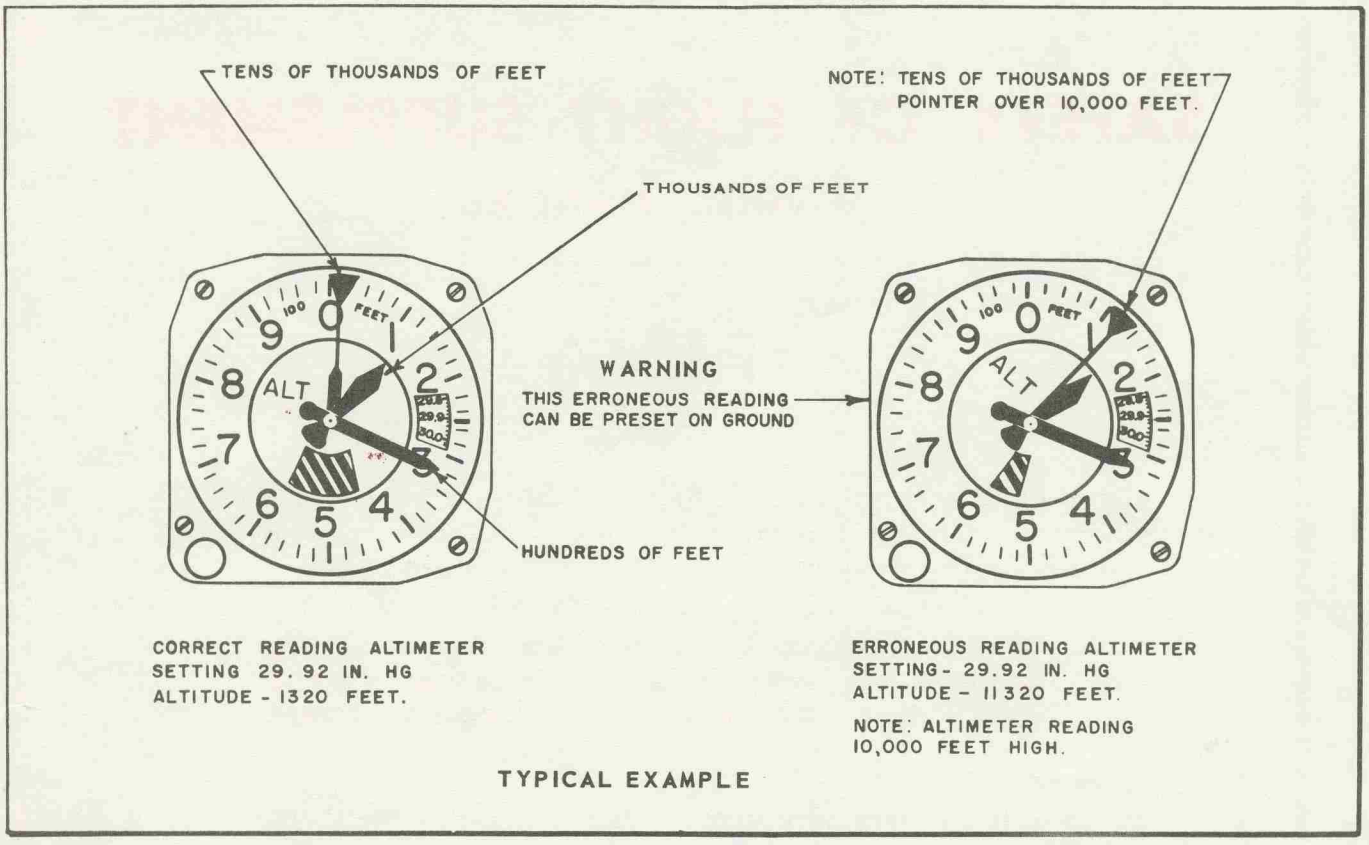

TENS OF THOUSANDS OF FEET
THOUSANDS OF FEET
HUNDREDS OF FEET
NOTE: TENS OF THOUSANDS OF FEET POINTER OVER 10,000 FEET.
WARNING
THIS ERRONEOUS READING CAN BE PRESET ON GROUND
CORRECT READING ALTIMETER
SETTING 29.92 IN. HG
ALTITUDE - 1320 FEET.
ERRONEOUS READING ALTIMETER
SETTING - 29.92 IN. HG
ALTITUDE - 11320 FEET.
NOTE: ALTIMETER READING 10,000 FEET HIGH.
TYPICAL EXAMPLE

Figure 1

END

FS FS FS FS FS FS FS FS FS FS FS FS FS FS FS FS

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8 APRIL 1959

1. PURPOSE.

To warn of possible defective switch guards on the external stores release selector switch.

2. GENERAL.

Investigation of a recent incident involving inadvertent loss of tip tanks has shown that it is possible for the guard on the external stores release selector switch to be in the closed position and the switch not be completely in the OFF position. This situation can lead to dropping the external stores if the external stores release button is pressed.

3. INSTRUCTIONS.

All guarded switches should be positively checked for proper position by raising the guard to determine the switch position.

CAUTION

Switch position should not be assumed as correct because switch guard is in the proper position.

END

A.F. SMAMA 4/7/59—2,550
A.F. SMAMA 2/19/59—100M

FS FS FS FS FS SAFETY OF FLIGHT FS FS FS FS FS

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SAFETY OF FLIGHT SUPPLEMENT

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THIS PUBLICATION SUPPLEMENTS T. O. 1F-104A-1 AND REPLACES INTERIM SAFETY OF FLIGHT SUPPLEMENT T. O. 1F-104A-1EQ, DATED 3 MARCH 1959, WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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7 APRIL 1959

1. PURPOSE.

To warn against attaching personal equipment leads to metal clips previously used to hold the arm restraining net and to warn against allowing elbows to move outward when pulling ejection ring.

2. GENERAL.

Maintenance instructions recently issued required removal of the arm restraining net from the ejection seat. Since removal of the arm net, some pilots have been attaching the anti-G suit lead to metal clips at the rear of the seat which were previously used to hold the arm net. Such action may delay separation from the seat. Care should be taken to insure that none of the pilot's personal equipment is entangled with these clips. The following instructions should be observed.

3. INSTRUCTIONS.

Do not attach any personal equipment leads to the metal clips previously used to hold the arm restraining net.

WARNING

With the arm restraining net removed, care must be taken when pulling the ejection ring to insure that elbows stay close to the body and within the ejection envelope. Grasping the ejection ring with palms upward will aid in keeping elbows close to the body.

END

A.F. SMAMA 4/7/59—2,550

A.F. SMAMA 2/19/59—25M

SAFETY OF FLIGHT

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T.O. 1F-104A-1EV

SAFETY OF FLIGHT SUPPLEMENT

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AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1 AND REPLACES FORMAL SAFETY OF FLIGHT SUPPLEMENTS, T.O. 1F-104A-1DF, DATED 16 MAY 1958, AND T.O. 1F-104A-1ED, DATED 22 OCTOBER 1958, WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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27 MARCH 1959

1. PURPOSE.

To define the emergency procedures to be followed in the event of an exhaust nozzle area control system failure and to provide instructions in the use of the nozzle area control switch on the subject airplanes.

2. GENERAL.

There is a possibility that the exhaust nozzle area control system can malfunction, allowing the exhaust nozzle to open fully, or to close to the mechanically scheduled area, or to fluctuate during afterburning or non-afterburning operation. Experience has shown that most of the difficulties encountered with the nozzle area control have been caused by a malfunction in the electrical portion of the nozzle area control system. For this reason, a cockpit switch has been incorporated to allow the pilot to override this electrical portion and manually close the exhaust nozzle to the mechanical schedule. With this switch in manual, there is no automatic temperature modulation. In addition, the mechanical schedule has been redesigned to increase the minimum nozzle area in the afterburner range in order to prevent excessive RPM drop, and/or engine stall during a nozzle failure to the closed position during maximum afterburner operation. A failure of the hydro-mechanical portion of the nozzle area control system is also a possibility, and in this case, the pilot will probably have no control of the exhaust nozzle. Each type of problem is treated separately below:

A.F. SMAMA 3/26/59—2,550
A.F. SMAMA 2/19/59—25M

Fs Fs Fs Fs Fs SAFETY OF FLIGHT Fs Fs Fs Fs Fs

a. Failure to the Mechanically Scheduled Area – Non-afterburning: The effect of this failure is apparent only in the military thrust setting since the mechanical schedule is the normal position of the nozzle at lower power settings. Under military thrust conditions, this shift to the mechanically scheduled area produces only a slight change. The nozzle area decreases slightly and the exhaust gas temperature increases with a slight increase in thrust. An attempt to light the afterburner will result in an over-temperature condition.

b. Failure to the Wide Open Area – Non-afterburning: The apparent effects of this failure are a significant decrease in thrust (equal to approximately 85% RPM), plus a reduction in exhaust gas temperature as well as an increase in nozzle area to the full open indicator position. Insufficient thrust is available to maintain level flight at any speed if the gear is extended. Minimum rate of sink is experienced at approximately 165 knots IAS for the landing flap and gear configuration, and 190 knots IAS for the take-off flap and gear configuration. With take-off flaps extended but gear up, sustained level flight is possible near sea level altitude with an airspeed of approximately 190 knots IAS. With gear and flaps retracted, level flight can be maintained up to 18,000 feet altitude at an airspeed of 300 knots IAS. Flight at higher or lower airspeeds results in a decrease in level flight altitude. If below 18,000 feet, a climb to this altitude is possible using a climb speed of 300 knots IAS. This performance information is based on a standard day with 2000 lbs. of fuel remaining.

c. Failure to the Mechanically Scheduled Area – Afterburning: A nozzle system failure under this condition will result in an over-temperature condition. In sector afterburning, excessive RPM drop and/or engine stall may result.

d. Failure to the Wide Open Area – Afterburning: This failure is indicated by a slight reduction in thrust, accompanied by a slight decrease in exhaust gas temperature and an increase in nozzle area and will probably not be detected by the pilot. Flight tests have shown that the afterburner will continue to burn with a wide open nozzle as long as the throttle is not retarded to the sector range. As long as the afterburner is maintained at this setting, there is no corrective action required. Failure to the wide open area – non-afterburning has been determined to be a critical condition. Therefore, a nozzle area control switch has been incorporated in the cockpit to allow the pilot to select the closed position. In view of the preceding, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Failure to the Mechanically Scheduled Area – Non-afterburning.

(1) Reduce throttle to maintain 585° C maximum.

(2) Actuate nozzle switch – CLOSE. This will insure nozzle position remaining on the Mechanical Schedule in case of subsequent electrical failure.

b. Failure to the Wide Open Area – Non-afterburning.

(1) Nozzle failure open during military thrust take-off run: Procedure is the same as for engine failure.

(2) Nozzle failure open immediately following military thrust take-off.

(a) Actuate nozzle switch.

(b) Gear up.

(c) Zoom to safe ejection altitude.

(d) If nozzle closed, monitor EGT 585° C maximum, burn out fuel and land.

(e) If nozzle remained open and more power is required, advance throttle rapidly to maximum afterburning without hesitating in sector position.

NOTE

An afterburner light with a wide open nozzle at low indicated airspeed cannot be guaranteed, but it is possible. The best technique is to advance the throttle rapidly to full afterburner position. A 3 to 5 second delay may be experienced before afterburner light is obtained. If an afterburner light is not obtained, leaving the throttle in the afterburning range will not adversely affect engine operation and will reduce the fuel load.

(f) If an afterburner light is obtained, deactivate the nozzle switch and obtain desired altitude and airspeed. Reduce fuel load and land. Since level flight in military thrust with an open nozzle can be sustained only with gear up, do not lower the gear until on the final approach and landing is assured.

(g) If afterburner fails to light, jettison external tanks and follow engine failure procedure.

(3) Nozzle failure open during flight.

(a) Advance throttle to military power.

(b) Actuate nozzle switch.

(c) Check nozzle position and monitor exhaust gas temperature to maintain 585° C maximum. Military thrust will be available.

NOTE

If the nozzle switch fails to close nozzle, maintain 300 KIAS, jettison external tanks and reduce fuel load and land when practical. Do not lower the gear until on the final approach and landing is assured.

(4) Nozzle failure open in landing pattern.

(a) Advance throttle to military thrust; speed brakes in.

(b) Actuate nozzle switch.

(c) Gear up.

(d) Flaps to take-off position.

(e) Jettison external tanks.

(f) Check the nozzle position and monitor exhaust gas temperature.

(g) When landing is assured, extend landing gear.

c. Failure to the Mechanically Scheduled Area – Afterburning.

(1) Rapidly retard the throttle to military thrust and monitor exhaust gas temperature.

(2) Actuate nozzle switch – Close.

d. Failure to the Wide Open Area – Afterburning.

(1) When retarding out of afterburning, retard the throttle slowly and check to see if the nozzle is closing. If nozzle is not closing, maintain afterburner thrust until desired altitude and airspeed are obtained.

(2) If nozzle is not closing normally, momentarily actuate nozzle switch to close position and return to normal. Check for sharp rise in EGT to insure proper functioning of switch prior to coming out of afterburner. Then proceed with paragraph 3.b.(3) of this technical order.

CAUTION

If nozzle switch is left in the close position while afterburner is lit, a severe over-temperature condition will result.

e. Severe nozzle fluctuations when at military or maximum thrust. Speed and altitude permitting, reduce thrust just below military to mechanical schedule. If nozzle does not stabilize, actuate the nozzle switch. Monitor exhaust gas temperature.

NOTE

Executing landings under any of the nozzle emergency conditions shall be accomplished with the flaps in the take-off position.

END

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SAFETY OF FLIGHT SUPPLEMENT

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24 MARCH 1959

1. PURPOSE.

To prohibit formation take-off and to provide information on flight characteristics when a single GAR-8 missile is carried.

2. GENERAL.

During recent flight tests of the F-104/GAR-8 combination in the one-missile configuration a moderate rolling moment was noted, particularly at subsonic speeds. In all cases the wing on which the single missile was installed rolled up during aircraft longitudinal and/or positive vertical accelerations and rolled down during aircraft longitudinal deceleration and/or negative vertical acceleration. It was possible to trim the aircraft laterally throughout the subsonic range; however, roll-off was experienced during any displacement from the trimmed condition. The wing-up rolling tendency was most bothersome immediately after becoming airborne, and the wing-down rolling tendency was most critical during low airspeed/negative G conditions. Setting rudder or aileron trim before take-off to counteract the rolling tendency was investigated and it was found that this procedure is not satisfactory as a corrective measure. Furthermore, it was found that the rolling tendency on take-off is controllable.

A.F. SMAMA 3/23/59—2,550

SAFETY OF FLIGHT

3. INSTRUCTIONS.

a. Formation take-off involving any aircraft in the one-missile configuration is prohibited.

b. When flight in the one-missile configuration is anticipated and/or required, caution should be exercised during take-off and during low airspeed/negative G maneuvers.

END

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AIRCRAFT

THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1 AND REPLACES FORMAL SAFETY OF FLIGHT SUPPLEMENT T.O. 1F-104A-1EE, DATED 3 NOVEMBER 1958, WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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20 MARCH 1959

1. PURPOSE.

To provide procedures in event of oil pressure fluctuations on the J79 engine installed in the subject aircraft and to correct the engine oil pressure record card variation limits stated in the Flight Manual.

2. GENERAL.

a. As a result of the engine oil pressure indicating system design and installation features, oil pressure indications may be affected by changes in altitude, Mach number, and maximum engine speed.

b. The normal engine oil pressure for each engine-airplane combination is stated on the engine oil pressure record card installed in the cockpit. If, during pre-flight engine check at Military thrust, the indicated oil pressure varies within limits (± 5 psi) from the normal pressure stated on the record card, use the indicated pressure observed during the Military thrust check as the normal oil pressure in lieu of the pressure stated on the record card.

CAUTION

If the indicated oil pressure varies more than ± 5 psi from that listed on the record card, the flight should be aborted and an engine inspection performed.

A.F. SMAMA 3/20/59—2,550

A.F. SMAMA 2/19/59—25M

c. An engine may exhibit the following oil pressure indication changes from the normal oil pressure as defined above:

(1) An oil pressure increase of 2 to 3 psi above normal during a climb from approximately 20,000 to 28,000 feet.

(2) An oil pressure increase of approximately 3 psi above normal at high Mach numbers.

(3) An oil pressure increase of approximately 6 psi above normal with full T2 reset (maximum engine speed).

(4) The maximum oil pressure increase could be approximately 12 psi, however this is not probable. Normally, the pressure rise will not exceed 6 or 7 psi.

d. A decrease in oil pressure is an indication of oil loss to the engine, leading eventually to a complete loss of oil to the bearings. The engine will operate for a period of one minute at Military thrust with a complete interruption of oil to the engine system without detrimental effects to the bearings. Limited experience has indicated the engine should operate for a period of approximately four to five minutes at 80% to 90% rpm, before complete failure occurs.

e. If the maximum oil pressure increase of 12 psi above normal is exceeded, or in event of low oil pressure, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Reduce thrust. Reduce thrust to that necessary to maintain safe altitude and level flight (86% to 89% rpm for a clean airplane).

b. External stores – Jettison (if necessary). Jettison external stores if necessary to maintain safe altitude and level flight. Retain external tanks if safe level flight can be maintained and external fuel is needed.

c. Avoid abrupt maneuvers causing high "G" forces.

d. Avoid rapid and large throttle movements.

e. Land as soon as possible using a pattern to insure a safe landing in the event of engine failure.

WARNING

Increasing vibration is an indication of a bearing failure. Extreme vibration, usually accompanied by a rise in E.G.T., indicates engine seizure will occur within a few seconds. The throttle should be cut off to prevent excessive destruction to the engine, resulting in possible damage to the aircraft structure.

NOTE

With a complete loss of oil, the nozzle may go to the full OPEN position which would require a higher thrust setting than normal. A throttle setting of 86% to 87%, with a normal engine, represents the thrust that would be obtained with a wide open nozzle at 100 to 101.5% engine speed.

NOTE

The J79-3 engine utilizes oil for positioning the compressor stator vanes. A complete loss of oil would cause the stator vanes to remain in the position at which they were set at the time of oil loss. If a throttle setting is at or near 100% rpm, the same power setting should be maintained and other means of regulating aircraft speed be used.

END

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20 MARCH 1959

1. PURPOSE.

To revise Engine Fire Procedures.

2. GENERAL.

Recent incidents involving cases of possible engine fire have highlighted some changes required to the basic engine fire procedures. After throttle has been stopcocked there is no electrical power available to actuate the warning light test switch, unless the ram air turbine is extended. This fact, coupled with the possibility of the fire extinguishing itself after the fuel is shut off, requires that some provisions be made to check the warning light test circuit.

3. INSTRUCTIONS.

a. Engine Fire After Lift-Off.

If fire warning lights come on after airplane is airborne, and sufficient runway and overrun are not available to land and stop the aircraft successfully, the following is recommended if altitude is too low for safe ejection.

(1) Throttle - Military.

(2) External tanks - Jettison.

(3) Zoom to safe ejection altitude.

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SAFETY OF FLIGHT

(4) Throttle - Adjust to minimum to maintain safe ejection altitude.

(5) Check for fire - Confirm fire by any possible means such as report from ground, other aircraft, engine instruments, smoke in cockpit or visible smoke trail behind aircraft.

(6) If fire exists - EJECT.

(7) If fire cannot be confirmed, make decision to land or eject.

b. Engine Fire During Flight.

If fire warning lights come ON during flight, proceed as follows:

(1) Throttle - Idle or minimum practical power.

(2) Head toward nearest emergency field - 275 KIAS.

(3) Check for fire - Confirm fire by any possible means such as report from other aircraft, engine instruments, smoke in cockpit or visible smoke trail behind aircraft.

(4) If fire is confirmed and lights remain ON:

(a) Stopcock throttle.

(b) If lights remain ON - EJECT.

(c) If lights go out - Extend RAT.

(d) Check circuit - If circuit does not check OK, EJECT (if circumstances permit).

(e) If circuit is OK, make decision to eject or attempt a flame-out landing.

(5) If fire cannot be confirmed and lights go out after retarding throttle:

(a) Actuate test circuit.

(b) If circuit is OK, and no other indications of fire, land as soon as possible at reduced power.

(c) If circuit does not check OK, EJECT (if circumstances permit).

(6) If there are no other indications of fire and lights remain ON after retarding throttle:

(a) Land as soon as possible, continually checking for fire.

END

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AIRCRAFT

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18 MARCH 1959

1. PURPOSE.

To change engine air start procedures and engine stall clearing procedures.

2. GENERAL.

a. Flight experience has indicated that some changes to the air start procedures are desirable to aid in obtaining a quicker air start and restoration of thrust.

b. High altitude stalls resulting from maneuvers or rapid throttle movements will normally clear themselves when descending below 30,000 feet. These stalls can sometimes be cleared by throttle movement and by establishing a straight and level flight condition.

c. Stalls resulting from malfunction of the inlet guide vane system can be cleared by using the inlet guide vane emergency reset switch. Use of this switch will set the IGVs in a fixed temperature rescheduled position. In this reset position, throttle modulation is possible above 97% rpm. Once the throttle is reduced below 97% rpm, it may not be possible to accelerate the engine to a higher rpm than that to which the reduction had been made. Care should be taken in planning the approach for landing after operating the IGV emergency reset switch to avoid the necessity of advancing the throttle once it has been retarded below 97%.

3. INSTRUCTIONS.

a. Engine Air Start Procedures.

(1) Both start switches - START. (Hold momentarily to insure contact of switches.)

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Monitor engine instruments for immediate relight if engine rpm is still high.

(2) Throttle - Positively OFF and immediately return to Military.

(3) Glide speed - 275 KIAS. While establishing best glide speed, the aircraft should be headed toward nearest suitable landing field.

(4) Both start switches - START. If no air start occurs within 30 to 40 seconds, actuate start switches again. Do not move throttle to OFF before this second actuation of the start switches.

(5) Ram air turbine - EXTEND. If no air start is made after the second attempt, extend the RAT and check for fuel flow indication.

(6) Start switches - START. (If fuel flow is available.)

(7) Throttle - Adjust as required. Allow engine instruments to stabilize and adjust throttle to settings necessary for flight. Engine instruments will give the most reliable indication of a relight.

b. Prior to take-off, advance throttle to 100% rpm, then reduce power slowly to 80% rpm. If compressor stall is encountered, abort flight. Make appropriate entry in Form 781A.

c. Engine Stall Clearing Procedures.

NOTE

- Any of the following procedures may clear the engine stall and all procedures need not be accomplished in the event stall clears.
- If landing roll distance is critical, deactivate the inlet guide vane switch after touchdown. Idle thrust is slightly higher when the IGV switch is in the manual position.
- Do not use the afterburner with the IGV switch in manual.
- Entries must be made in Form 829A each time this emergency device is actuated to clear a compressor stall. If five emergency actuations are made on any one engine, this information should be forwarded to SMAMA for disposition.

Below 15,000 Feet

(1) IGV switch - Manual.

(2) Air start procedure.

(3) Maintain 97% rpm, until landing is assured. Power may be reduced below 97% rpm, if landing conditions are such as to require a reduced thrust. Care must be exercised not to reduce thrust below that necessary to complete the landing because it may not be possible to regain higher thrust, once it is reduced. Speed brakes should be used to keep airspeed consistent with pattern and landing conditions.

Above 15,000 Feet

(1) Throttle - Military.

(2) Throttle - Idle. Check EGT for normal reading and rpm for possible hang-up.

(3) Air start procedure. If stall is not cleared, descend to below 30,000 feet and attempt another air start procedure.

(4) Check for IGV stall, before descending to land, by retarding throttle slowly to 80% rpm. This check may not indicate an open IGV condition stall above 20,000 feet. If stall reoccurs, at this time or during descent, use the procedure for the below 15,000 feet procedure.

END

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2 FEBRUARY 1959

1. PURPOSE.

To emphasize the correct instructions for attaching the oxygen mask connector tie-down strap to the parachute chest strap.

2. GENERAL.

Proper attachment of the oxygen mask connector tie-down strap is extremely important to assure that:

a. The oxygen hose does not become accidentally disconnected during flight, resulting in loss of oxygen supply to the pilot.

b. The oxygen hose does not prevent quick separation from the seat during ejection.

c. The oxygen hose does not flail during ejection, causing pilot injury.

d. The tie down strap does not accidentally open the parachute chest strap snap during ejection.

SAFETY OF FLIGHT

3. INSTRUCTIONS.

a. Attach the oxygen mask hose (male connector) to parachute chest strap by wrapping the mask connector tie-down strap underneath and up behind the chest strap twice, as close to the chest strap snap as possible, and then snapping it.

WARNING

Failure to double loop the connector tie-down strap around the parachute chest strap may permit the tie-down strap to slip into and open the chest strap snap during ejection.

Do not wrap the tie-down strap around the chest strap snap.

b. Connect the mask-to-regulator tubing female disconnect to the mask male connector. Listen for the click, and visually check that sealing gasket is only half exposed.

c. Attach the alligator clip to the tie-down strap as close to the chest strap snap as possible.

WARNING

Do not attach alligator clip to the parachute harness as this may prevent quick separation from the seat during ejection. The force required to pull the clip loose from the parachute harness is considerably greater than from the tie-down strap.

END

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T.O. 1F-104A-1EN

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THIS PUBLICATION SUPPLEMENTS T. O. 1F-104A-1 AND REPLACES INTERIM SAFETY OF FLIGHT SUPPLEMENT T. O. 1F-104A-1EJ WITH CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic publication by personnel responsible for maintaining the publication in current status.

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14 JANUARY 1959

1. PURPOSE.

To provide additional information concerning low altitude ejection.

2. GENERAL.

Investigation of a recent aircraft accident has revealed that a low altitude ejection was unsuccessful for reasons other than ejection system malfunction. Two possible reasons were indicated for failure of parachute to open:

a. The pilot continued to hold the ejection ring too long to allow separation from the seat prior to ground contact.

b. Stability of the ejection seat was such that it did not assist in separating the pilot from the seat. Additionally, it was found that the parachute D-ring lanyard was not hooked up. Therefore, the following instructions shall apply for all low altitude (below 2,000 feet) ejections.

3. INSTRUCTIONS.

When ejection is necessary below 2,000 feet altitude, release ejection ring immediately upon clearing aircraft and kick free from the seat with a positive action.

WARNING

Proper hook up of parachute D-ring lanyard is mandatory to obtain a parachute opening with least possible delay. If not hooked, a one-second delay will occur before parachute opening sequence begins.

A.F. SMAMA 1/16/59—2,550

END

A.F. SMAMA 10/31/58—100M

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9 JANUARY 1959

1. PURPOSE.

To define the conditions under which the ram-air turbine (RAT) should be extended and the areas where it can be extended without affecting normal engine operation.

2. GENERAL.

The ram-air turbine is available for emergency electrical and hydraulic power when the engine-driven power sources are lost. Extension of the RAT with the engine running can, under certain conditions, adversely affect engine operation. Because several inadvertent RAT extensions have been experienced, the contractor has investigated the operating envelope with the engine running and the RAT extended. These investigations have shown:

a. The RAT can be extended in level flight without affecting engine operation at any speed up to 550 knots IAS, except as follows:

At 40,000 feet and above - 350 knots IAS minimum.

At 35,000 feet - 325 knots IAS minimum.

At 30,000 feet and below - No minimum limit.

b. Normal airstarts with the RAT extended can be made at all altitudes up to 35,000 feet.

A.F. SMAMA 10/31/58—100M

A.F. SMAMA 1/22/59—2,550

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c. Maneuverability is satisfactory with the No. 1 and No. 2 hydraulic pumps inoperative with the RAT supplying hydraulic pump power up to 500 knots IAS.

d. Spiral climbs and descents can be made without affecting normal engine operation or airplane maneuverability.

e. Best range with the RAT extended and 3,000 lbs. of fuel remaining is realized by cruising at 0.82 Mach number at 27,000 feet. Range will be approximately 170 nautical miles per 1,000 lbs. 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 30,000 feet. Below 30,000 feet, 45 degree banks do not affect engine operation.

3. INSTRUCTIONS.

a. Deploy the ram-air turbine only for:

(1) Double hydraulic failure.

(2) Double electrical failure.

(3) Flameout landing.

(4) Seized engine.

(5) Dead engine descent in weather.

b. Observe the following minimum airspeeds when extending the ram-air turbine with the engine operating.

40,000 feet and above - 350 knots IAS minimum.

35,000 feet - 325 knots IAS minimum.

30,000 feet and below - No minimum limit.

NOTE

If a flameout or engine stall occurs when the RAT is extended, accomplish normal airstart or stall clearing procedures.

c. Avoid abrupt or uncoordinated maneuvers.

d. Move throttle slowly and only when necessary.

e. Accomplish a precautionary landing as soon as practical and use thrust as required.

f. Do not attempt afterburner lights, unless absolutely necessary.

END

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8 JANUARY 1959

1. PURPOSE.

To provide instructions to be followed when engine anti-icing system is reactivated and warning light installed in the subject aircraft.

2. GENERAL.

a. The engine anti-icing system consists of an anti-icing valve which, when opened, allows hot air to circulate through the engine compressor front frame and inlet guide vanes. An indicator light is installed in the cockpit to enable the pilot to determine whether the valve is open (light on), or closed (light out). There is no provision to combat airframe icing.

b. It has been determined that icing will occur on the inlet ducts and engine compressor front frame at subsonic speeds only. Ram-air temperature rise at supersonic speeds will be sufficient to preclude icing at these speeds. If the engine is operated above 82%, the anti-icing air temperature is sufficient to prevent rapid ice build-up on the engine front frame and inlet guide vanes. Flight tests have shown that the engine can safely ingest aircraft inlet duct ice at engine rpm less than 88%; however, at higher rpm's, the inlet guide vanes may sustain some damage. Engine operation is still possible with limited inlet guide vane damage.

c. The requirement for the engine anti-icing is a direct function of the indicated compressor inlet temperature (CIT). Operation of the anti-icing system above a CIT of 10 degrees Centigrade, compromises the service life expectancy of the magnesium front frame. Therefore, if weather conditions indicate a need for anti-icing; the anti-icing system should be necessary only at a CIT indication of 10 degrees Centigrade, or below. Furthermore, the cockpit indicator light (valve open) will stay on for 1 1/2-minutes after the anti-icing switch has been deactivated. This condition exists because the

A.F. SMAMA 5/28/58 —200M
A.F. SMAMA 1/8/59—2,450

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temperature sensing device is operated in the de-icing air duct, and there is no means of dumping the hot air immediately after the anti-icing valve is closed. Light illumination for more than 1 1/2-minutes, indicates the valve is either stuck fully or partially open. In view of the above information, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Ground Check (to be performed on the first flight of the day only):

(1) Activate the anti-icing switch, after the engine comes up to idle speed.

(2) Advance throttle. At approximately 80%, the indicator light should illuminate.

(3) Prior to taxiing, deactivate switch and monitor indicator light. If light remains on more than 1 1/2-minutes, abort flight and make appropriate entry in Form 871.

b. In-Flight Procedure:

(1) If flight is anticipated through known or suspected icing conditions, activate the anti-icing system when at subsonic speed and when the indicated CIT is at 10 degrees Centigrade, or below. When the anti-icing valve open, a maximum speed of 350 KIAS or Mach 1.0, whichever is lower, should not be exceeded.

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

(3) Should it be necessary to fly in known icing conditions at low altitude and low power settings (80 to 86% rpm), the engine power should be increased to 100% rpm every five minutes to assure that an adequate anti-icing air circulation is available at the engine compressor front frame. This power increase should be maintained for approximately thirty seconds.

END

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2 JANUARY 1959

1. PURPOSE.

To provide warning against high speed taxi turns.

2. GENERAL.

Information has been received of broken main landing gear wheel assemblies occurring while taxiing. It is suspected that one of the contributing factors may be side loads imposed on the wheel during high speed taxi turns.

3. INSTRUCTIONS.

All pilots are cautioned against high speed taxi turns to reduce the side loads imposed on the main landing gear wheel assemblies.

END

A.F. SMAMA 5/28/58 —200M

A.F. SMAMA 1/5/59—2,450

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1 DECEMBER 1958

ROUTED TO F-104 CREWS 11 JAN 59

1. PURPOSE.

To provide standardized hand signals for use during starting engine.

2. GENERAL.

Whenever an electrical connection between the ground start unit and the aircraft is not available to control the flow of starting air, the pilot must depend upon ground personnel to perform this function. Coordination must be effected, between the pilot and ground crew, to insure timely stoppage of air flow in order to preclude overspeed of the starter air turbine and subsequent failure. Furthermore, even with the electrical connection available, there is a possibility that this system may fail and result in overspeeding the starter air turbine. In view of the above, and regardless of whether the electrical connection is available or not, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Prior to entering the cockpit, the pilot will brief the appropriate ground personnel on the hand signals to be used when starting air is required and when starting air should be discontinued.

b. The hand signals to be used are as follows:

(1) Pilot will raise right hand above head with index finger pointing up. Twirling the right hand in a rotary motion will signify to the ground crew to turn on starting air.

(2) Upon first indication of engine RPM, the pilot will lower right arm.

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(3) Between 40 and 43% RPM, the pilot will raise right hand above head, palm downward, and wave to the right. This will signify to the ground crew to manually turn off starting air.

NOTE

- The open right hand, palm down, waved to the right, will always signify to the ground crew to manually turn off starting air. This may be necessary at any time during the starting cycle.

- The above listed hand signals will be incorporated, at the earliest possible time, into AFR 62-10 (Aircraft Operation and Movement on the Ground or Water). At such time that the hand signals are incorporated into AFR 62-10, this Safety of Flight Supplement will be rescinded. Neither the description nor illustrations of the hand signals will be incorporated into the subject Flight Manual.

END

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1 DECEMBER 1958

1. PURPOSE.

To prevent inadvertent loss of oxygen supply.

2. GENERAL.

If the supply lever is left ON when the pressure demand regulator is not being used, the regulator can automatically go to positive pressure should cockpit altitude exceed 28,000 feet. This has occurred on one occasion during flight at high altitude using a pressure suit which deleted the requirement for the pressure demand regulator. The pressure demand regulator supply lever was left ON and faulty cabin pressurization allowed the cockpit altitude to exceed 28,000 feet. The pressure demand regulator automatically went to positive pressure and after approximately six minutes, under these conditions, the oxygen supply was depleted. To preclude this possibility, the following instructions shall be observed.

3. INSTRUCTIONS.

a. The supply lever on the MD-1 pressure demand oxygen regulator panel must be in the OFF position when using a pressure suit.

b. The supply lever on the ME-2 control panel must be in the OFF position when using the A-13A oxygen mask.

END

A.F. SMAMA 12/1/58—2,450
A.F. SMAMA 10/31/58—100M

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19 NOVEMBER 1958

1. PURPOSE.

To provide additional instructions for pre-flight check of the auto-pitch control (APC) system.

2. GENERAL.

Recent aircraft incident reports have been received concerning in-flight APC malfunctions caused by failure to retract of the piston on the APC actuator, after the APC in-flight check. To insure that the APC system can be overpowered in flight in case of a malfunction, and until final corrective action can be determined, the following instructions shall be observed.

3. INSTRUCTIONS.

NOTE

The following check shall be performed immediately after step 2 of the AUTO-PITCH, STICK SHAKER, AND STABILITY AUGMENTATION CONTROL SYSTEM CHECK presently outlined in Section II of the Flight Manual.

A.F. SMAMA 11/20/58—2,450

A.F. SMAMA 5/28/58 —200M.

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With APC operating, apply an aft force to the control stick to insure the APC system can be manually overpowered. The force required to overpower the APC system should be approximately 35 pounds at the neutral stick position.

WARNING

If the APC system cannot be manually overpowered or excessive aft stick force is required, abort mission and have system checked and cause of malfunction eliminated.

END

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THIS PUBLICATION SUPPLEMENTS T.O. 1F-104A-1 AND REPLACES INTERIM SAFETY OF FLIGHT SUPPLEMENT T.O. 1F-104A-1DW, DATED 2 SEPTEMBER 1958, AND INCLUDES CHANGES TO THE TEXT. Reference to this supplement will be made on the title page of the basic manual by personnel responsible for maintaining the publication in current status.

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22 OCTOBER 1958

1. PURPOSE.

To define the emergency procedures to be followed in the event of an exhaust nozzle area control system failure and to provide instructions in the use of the newly installed nozzle area control switch on the subject airplanes.

2. GENERAL.

There is a possibility that the exhaust nozzle area control system can malfunction, allowing the exhaust nozzle to open fully, or to close to the mechanically scheduled area, or to fluctuate during afterburning or non-afterburning operation. Experience has shown that most of the difficulties encountered with the nozzle area control have been caused by a malfunction in the electrical portion of the nozzle area control system. For this reason, a cockpit switch has been incorporated to allow the pilot to override this electrical portion and manually close the exhaust nozzle to the mechanical schedule. With this switch in manual, there is no automatic temperature modulation. In addition, the mechanical schedule has been redesigned to increase the minimum nozzle area in the afterburner range in order to prevent excessive RPM drop, and/or engine stall during a nozzle failure to the closed position during maximum afterburner operation. A failure of the hydro-mechanical portion of the nozzle area control system is also a possibility, and in this case, the pilot will probably have no control of the exhaust nozzle. Each type of problem is treated separately below:

A.F. SMAMA 10/23/58—2,550
A.F. SMAMA 10/7/58—50M

a. Failure to the Mechanically Scheduled Area – Non-afterburning: The effect of this failure is apparent only in the military thrust setting since the mechanical schedule is the normal position of the nozzle at lower power settings. Under military thrust conditions, this shift to the mechanically scheduled area produces only a slight change. The nozzle area decreases slightly and the exhaust gas temperature increases with a slight increase in thrust. An attempt to light the afterburner will result in an over-temperature condition.

b. Failure to the Wide Open Area – Non-afterburning: The apparent effects of this failure are a significant decrease in thrust plus a reduction in exhaust gas temperature as well as an increase in nozzle area to the full open indicator position. Insufficient thrust is available to maintain level flight with any configuration or airspeed at takeoff gross weights. At normal landing gross weight, level flight cannot be maintained at any speed if the gear and/or landing flaps are extended or external stores are carried. Level flight can be maintained up to 3500 feet in the clean configuration at 300 knots IAS with normal landing gross weights, providing ambient temperatures are standard or cooler. With takeoff flaps extended, level flight can be sustained up to 5500 feet at an airspeed of 190 knots IAS at normal landing gross weight, providing ambient temperatures are at standard conditions or cooler. The capability to sustain level flight at landing gross weight in either the clean configuration or with takeoff flaps extended is highly improbable on days with ambient temperatures hotter than standard.

c. Failure to the Mechanically Scheduled Area – Afterburning: A nozzle system failure under this condition will result in an over-temperature condition. In sector afterburning, excessive RPM drop and/or engine stall may result.

d. Failure to the Wide Open Area – Afterburning: This failure is indicated by a slight reduction in thrust, accompanied by a slight decrease in exhaust gas temperature and an increase in nozzle area and will probably not be detected by the pilot. Flight tests have shown that the afterburner will continue to burn with a wide open nozzle as long as the throttle is not retarded to the sector range. As long as the afterburner is maintained at this setting, there is no corrective action required. Failure to the wide open area – non-afterburning has been determined to be a critical condition. Therefore, a nozzle area control switch has been incorporated in the cockpit to allow the pilot to select the closed position. In view of the preceding, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Failure to the Mechanically Scheduled Area – Non-afterburning.

(1) Reduce throttle just below military thrust.

(2) Monitor exhaust gas temperature.

b. Failure to the Wide Open Area – Non-afterburning.

(1) Nozzle failure open during military thrust takeoff run: Procedure is the same as for engine failure.

(2) Nozzle failure open immediately following military thrust takeoff.

(a) Actuate nozzle switch.

(b) Gear up.

(c) Jettison external stores.

(d) Check nozzle position and monitor exhaust gas temperature.

(e) If nozzle failed to close, advance throttle to maximum thrust.

NOTE

An afterburner light with a wide open nozzle at low indicated airspeed can not be guaranteed, but it is possible. The best technique is to advance the throttle to the mechanically scheduled A/B area, delay momentarily and then slowly advance to full A/B regardless of when A/B lightoff occurs. If an A/B light is not obtained, leaving the throttle in the afterburning range will not adversely affect engine operation and will reduce the fuel load.

(f) If an afterburner light is obtained, deactivate the nozzle switch and obtain desired altitude and airspeed before coming out of A/B. Reduce fuel load and land. Since level flight in military thrust with an open nozzle can be sustained only with gear up, do not lower the gear until on the final approach and landing is assured.

(3) Nozzle failure open during flight.

(a) Advance throttle to military power.

(b) Actuate nozzle switch.

(c) Check nozzle position and monitor exhaust gas temperature. Military thrust will be available.

NOTE

If the nozzle switch fails to close nozzle, maintain 300 KIAS, jettison external stores and reduce fuel load and land when practical. Since level flight can be sustained with gear up at extremely low altitudes only, emergency landing procedures should be followed. Do not lower the gear until on the final approach and landing is assured.

(4) Nozzle failure open in landing pattern.

(a) Advance throttle to military thrust; speed brakes in.

(b) Actuate nozzle switch.

(c) Gear up.

(d) Flaps to takeoff position.

(e) Jettison external stores.

(f) Check the nozzle position and monitor exhaust gas temperature.

(g) It is recommended that a go around be completed since landing gear has been retracted.

(h) If the nozzle failed to close and more power is required, advance throttle directly to maximum afterburning without hesitating in sector position.

(i) If afterburner light is obtained, deactivate the nozzle switch and obtain the desired altitude and airspeed before coming out of A/B. Since level flight at the military thrust throttle position with open nozzle can be sustained with gear up at extremely low altitudes only, emergency landing procedures should be followed. Do not lower the gear until on the final approach and landing is assured.

(j) If an A/B light is not obtained, leaving the throttle in the afterburning range will not adversely affect engine operation, and will reduce the fuel load. If unable to reach landing area, and altitude permits, abandon aircraft.

c. Failure to the Mechanically Scheduled Area – Afterburning.

(1) Retard the throttle to military thrust and monitor exhaust gas temperature.

d. Failure to the Wide Open Area – Afterburning.

(1) When retarding out of afterburning, retard the throttle slowly and check to see if the nozzle is closing. If nozzle is not closing, maintain afterburner thrust until desired altitude and airspeed are obtained. Then proceed with paragraph 3.b.(3) of this technical order.

e. Severe nozzle fluctuations when at military or maximum thrust. Speed and altitude permitting, reduce thrust just below military to mechanical schedule. If nozzle does not stabilize, actuate the nozzle switch. Monitor exhaust gas temperature.

NOTE

Executing landings under any of the nozzle emergency conditions shall be accomplished with the flaps in the takeoff position.

END

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THIS PUBLICATION SUPPLEMENTS T. O. 1F-104A-1 AND REPLACES SAFETY OF FLIGHT SUPPLEMENT T. O. 1F-104A-1CX, DATED 17 APRIL 1958. Reference to this supplement will be made on the title page of the basic publication by personnel responsible for maintaining the publication in current status.

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16 MAY 1958

1. PURPOSE.

To define the emergency procedures to be followed in the event of an exhaust nozzle area control system failure on the subject airplane.

2. GENERAL.

There is a possibility that the exhaust nozzle area control system can malfunction, allowing the exhaust nozzle to open fully or to close to the mechanically scheduled area during afterburning or non-afterburning operation. Each case is treated separately below.

a. Failure to the Mechanically Scheduled Area - Non-afterburning: The effect of this failure is apparent only in the military power setting since the mechanical schedule is the normal position of the nozzle at lower power settings. Under military power conditions, this shift to the mechanically scheduled area produces only a slight change. The nozzle area decreases slightly and the exhaust gas temperature increases with a slight increase in thrust.

b. Failure to the Wide Open Area - Non-afterburning: The apparent effects of this failure are a significant decrease in thrust plus a reduction in exhaust gas temperature as well as an increase in nozzle area to the full open indicator position. Insufficient thrust is available to maintain level flight at any speed if the gear is extended. Minimum rate of sink is experienced at approximately 165 knots IAS for the landing flap and gear configuration, and 190 knots IAS for the takeoff flap and gear configuration. With takeoff flaps extended but gear up, sustained level flight is possible near sea level altitude with an airspeed of approximately 190 knots IAS. With gear and flaps retracted, level flight can be maintained up to 18,000 feet altitude at an airspeed of 300 knots IAS. Flight at higher or lower airspeeds results in a decrease in level flight altitude. If below 18,000 feet, a climb to this altitude is possible using a climb speed of 300 knots IAS.

A.F. SMAMA 4/10/58—200M
A.F. SMAMA 5/16/58—2,950

c. Failure to the Mechanically Scheduled Area - Afterburning: A nozzle system failure under these conditions can produce one of three effects.

(1) In some cases the engine speed will decrease to less than 84 percent rpm, causing the afterburner fuel flow to be stopped. The engine will then accelerate up to an rpm above 96 percent, at which time the afterburner will re-light. If the throttle position is maintained in afterburner range, this cycle will be repeated. The time for one cycle, as described, is approximately three to five seconds and will be readily recognized because of the change in noise level.

(2) The second possibility is that main engine rpm will decrease to a point just above the afterburner cutoff (84 - 86 percent rpm) and stabilize; exhaust gas temperature will rise to approximately 750°C, and stabilize.

(3) The third possibility is that, as the engine rpm is decreasing through the 90 - 87 percent region, there will be several sharp reports (bangs) from the compressor. This condition is accompanied by a distinct and bright torching of the afterburner flame.

NOTE

During all three cases a distinct change in noise level will be noted.

d. Failure to the Wide Open Area - Afterburning: This failure is indicated by a slight reduction in thrust, accompanied by a slight decrease in exhaust gas temperature and an increase in nozzle area and will probably not be detected by the pilot. Flight tests have shown that the afterburner will continue to burn with a wide open nozzle as long as the throttle is not retarded to the sector range. As long as the afterburner is maintained at this setting, there is no corrective action required.

In view of the preceding, the following instructions shall be observed.

3. INSTRUCTIONS.

a. Failure to the Mechanically Scheduled Area - Non-afterburning.

(1) Throttle back, as required to observe exhaust gas temperature limits.

(2) Do not light the afterburner.

b. Failure to the Wide Open Area - Non-afterburning.

(1) Nozzle failure open during military power takeoff run procedure is the same as for engine failure.

(2) Nozzle failure open following military power takeoff.

(a) Gear up.

(b) Advance throttle to maximum power.

NOTE

An afterburner light with a wide open nozzle at low indicated airspeed can not be guaranteed, but it is possible. The best technique is to advance the throttle to the mid A/B area, delay momentarily, and then slowly advance to full A/B (regardless of when A/B lightoff occurs). If an A/B light is not obtained, leaving the throttle in the afterburning range will reduce the fuel load and improve climb-out performance and will not adversely affect engine operation.

(c) Jettison external stores.

(d) Airspeed 190 knots IAS, if the takeoff flaps are extended and cannot be safely retracted; 300 knots IAS, if the flaps are up.

(e) Reduce the fuel load and land when practical.

NOTE

Since level flight can be sustained only with the gear up, do not lower the gear until on the final approach and landing is assured.

(3) Nozzle failure open during flight.

(a) Throttle military power.

(b) Gear up.

(c) Flaps up.

(d) Speed brakes in.

(e) Airspeed 300 knots IAS.

(f) Jettison external stores.

(g) Reduce the fuel load and land when practical.

NOTE

Since level flight can be sustained only with the gear up, do not lower the gear until on the final approach and landing is assured.

c. Failure to the Mechanically Scheduled Area - Afterburning.

(1) Retard the throttle rapidly and sharply to the approximately 1/2-throttle position, and then advance to military power.

(2) Follow procedures outlined in paragraph 3.a., above.

d. Failure to the Wide Open Area - Afterburning.

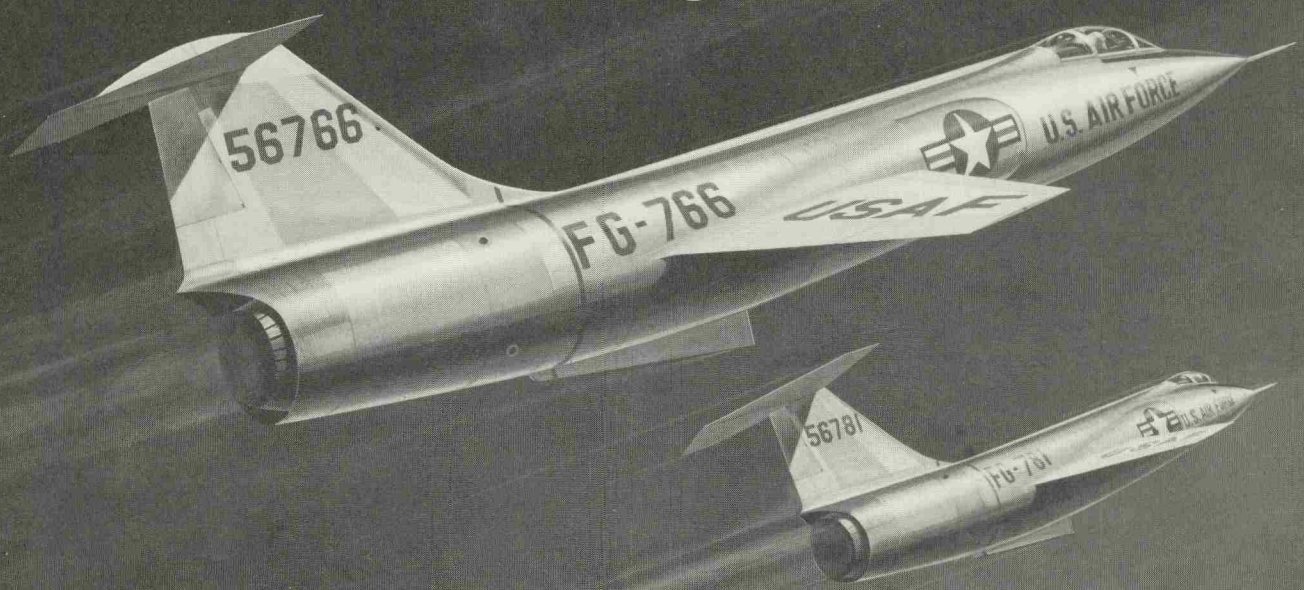
(1) When retarding out of afterburning, retard the throttle slowly and check to see if the nozzle is closing.

(2) Follow procedures outlined in paragraph 3.b., above, after moving throttle to military power position.

END

FLIGHT MANUAL

USAF SERIES F-104A AIRCRAFT



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

Published under authority of the Secretary of the Air Force.

This change replaces Safety of Flight Supplements through -1EE, except -1DF and -1ED. See Basic Index, T.O. 0-1-1, and Weekly Index, T.O. 0-1-1A, for current status of Safety Flight Supplements.

This manual is incomplete without Confidential Supplement, T.O. 1F-104-1A.

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★ REFER TO CONFIDENTIAL SUPPLEMENT, T.O. 1F-104A-1A



This is Your Life...

SCOPE. This manual contains all the information necessary for safe and efficient operation of the F-104A. These instructions do not teach basic flight principles, but are designed to provide you with a general knowledge of the airplane, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

SOUND JUDGMENT. The instructions of this manual are designed to provide for the needs of a crew inexperienced in the operation of this aircraft. This book provides the best possible operating instructions under most circumstances, but it is a poor substitute for sound judgment. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures contained herein.

PERMISSIBLE OPERATIONS. The Flight Manual takes a "positive approach" and normally tells you only what you can do. Any unusual operation or configuration (such as asymmetrical loading) is prohibited unless specifically covered in the Flight Manual. Clearance must be obtained from ARDC before any questionable operation is attempted which is not specifically covered in the Flight Manual.

STANDARDIZATION. Once you have learned to use one Flight Manual, you will know how to use them all—closely guarded standardization assures that the scope and arrangement of all Flight Manuals are identical.

ARRANGEMENT. The manual has been divided into 10 fairly independent sections, each with its own table of contents. The objective of this subdivision is to make it easy both to read the book straight through when it is first received and thereafter to use it as a reference manual. The independence of these sections also makes it possible for the user to rearrange the book to satisfy his personal taste and requirements. The first 3 sections cover the minimum information required to safely get the airplane into the air and back down again. Before flying any new aircraft these 3 sections must be read thoroughly and fully understood. Section IV covers all equipment not essential to flight but which permits the aircraft to perform special functions. Sections V and VI are obvious. Section VII covers lengthy discussions on any technique or theory of operation which may be applicable to the particular aircraft in question. The experienced pilot will probably be aware of the information in this section but he should check it for any possible new information. The contents of the remaining sections are fairly obvious.

YOUR RESPONSIBILITY. These Flight Manuals are constantly maintained current through an extremely active revision program. Frequent conferences with operating personnel and constant review of UR's, accident reports, flight test reports, etc., assure inclusion of the latest data in these manuals. In this regard, it is essential that you do your part! If you find anything you don't like about the book, let us know right away. We cannot correct an error whose existence is unknown to us.

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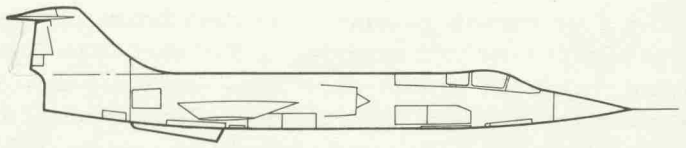
WARNINGS, CAUTIONS, AND NOTES. For your information, the following definitions apply to the "Warnings," "Cautions," and "Notes" found throughout the manual:

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

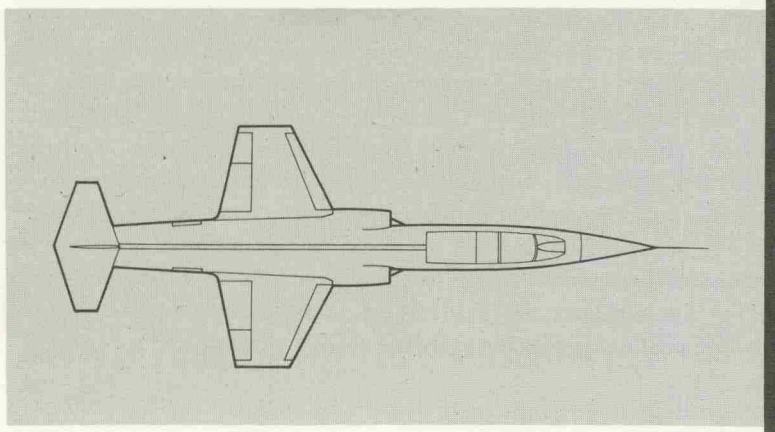
CAUTION — Operating procedures, practices, etc, which if not strictly observed will result in damage to equipment.

Note — An operating procedure, condition, etc, which it is essential to emphasize.

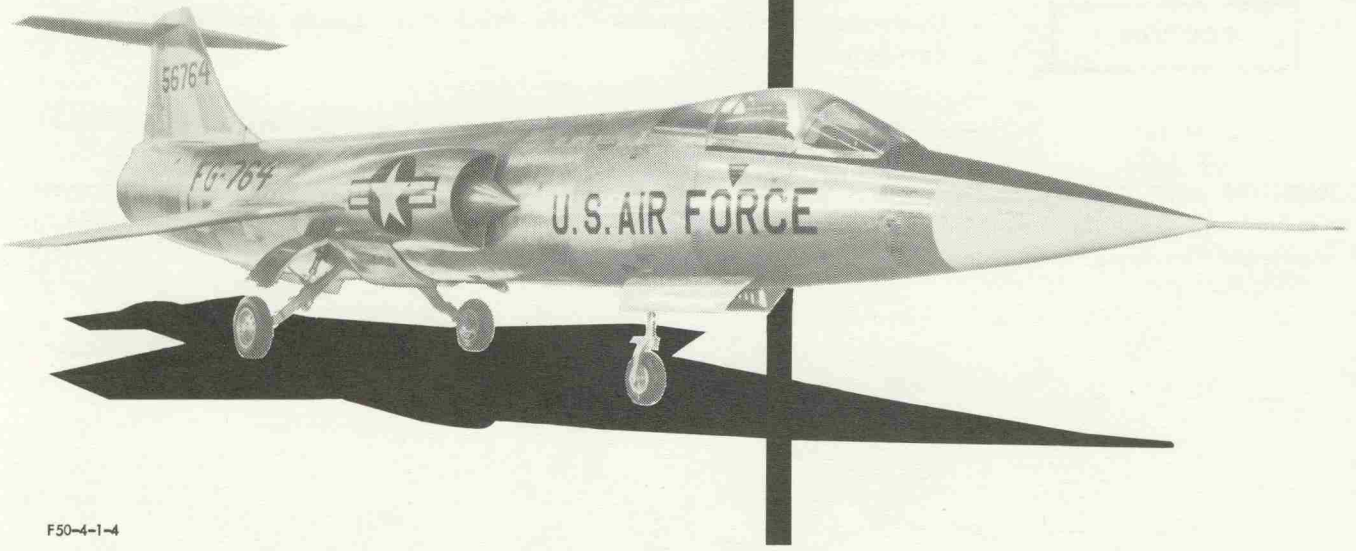
COMMENTS AND QUESTIONS. Comments and questions regarding any phase of the Flight Manual program are invited and should be forwarded through your Command Headquarters to Commander, Detachment #1, Hq Air Research and Development Command, Wright-Patterson AFB, Ohio, ATTN: RDZSPH.



F-104A







F50-4-1-4

Figure 1-1



F50-B-1-5
AM2829

description

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THE AIRPLANE.

The F-104A is a single-place, high performance, day and night fighter powered by an axial flow, turbojet engine with afterburner. The airplane is built by Lockheed Aircraft Corporation, and is designed for high subsonic cruise and high supersonic combat. Some notable features of the airplane are extremely thin flight surfaces, short straight wings with negative dihedral, irreversible hydraulically-powered ailerons, and a controllable horizontal stabilizer mounted at the top of the vertical stabilizer. 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. Air escape is accomplished with a downward ejection system. A drag chute is installed to reduce landing roll. The forward and aft main fuel cells are filled from a single point refueling well. The external fuel tanks and the internal auxiliary cell are filled from individual ports.

AIRPLANE DIMENSIONS.

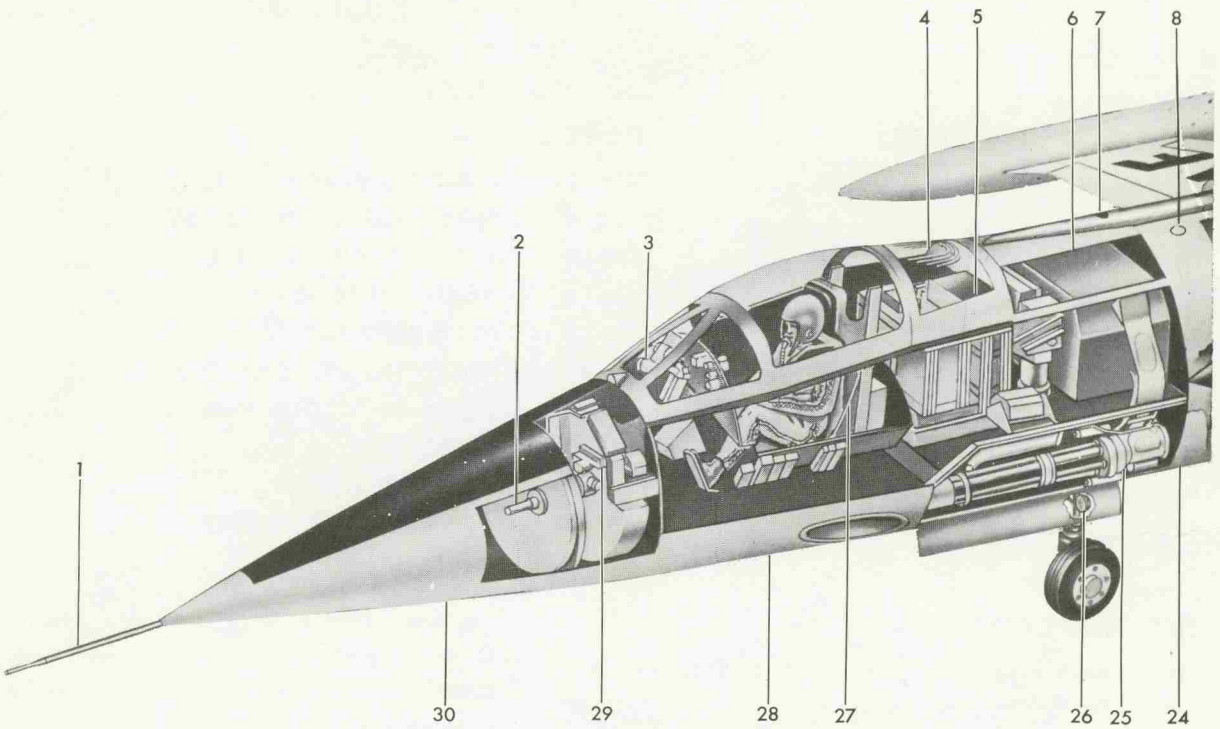
Overall dimensions of the airplane are as follows:

Wing span	21.94 feet
Length	54.77 feet
Height (to top of vertical stabilizer)	13.49 feet
Tread	8.79 feet

Refer to Figure 2-4 for minimum turning radius and ground clearances.

GENERAL ARRANGEMENT

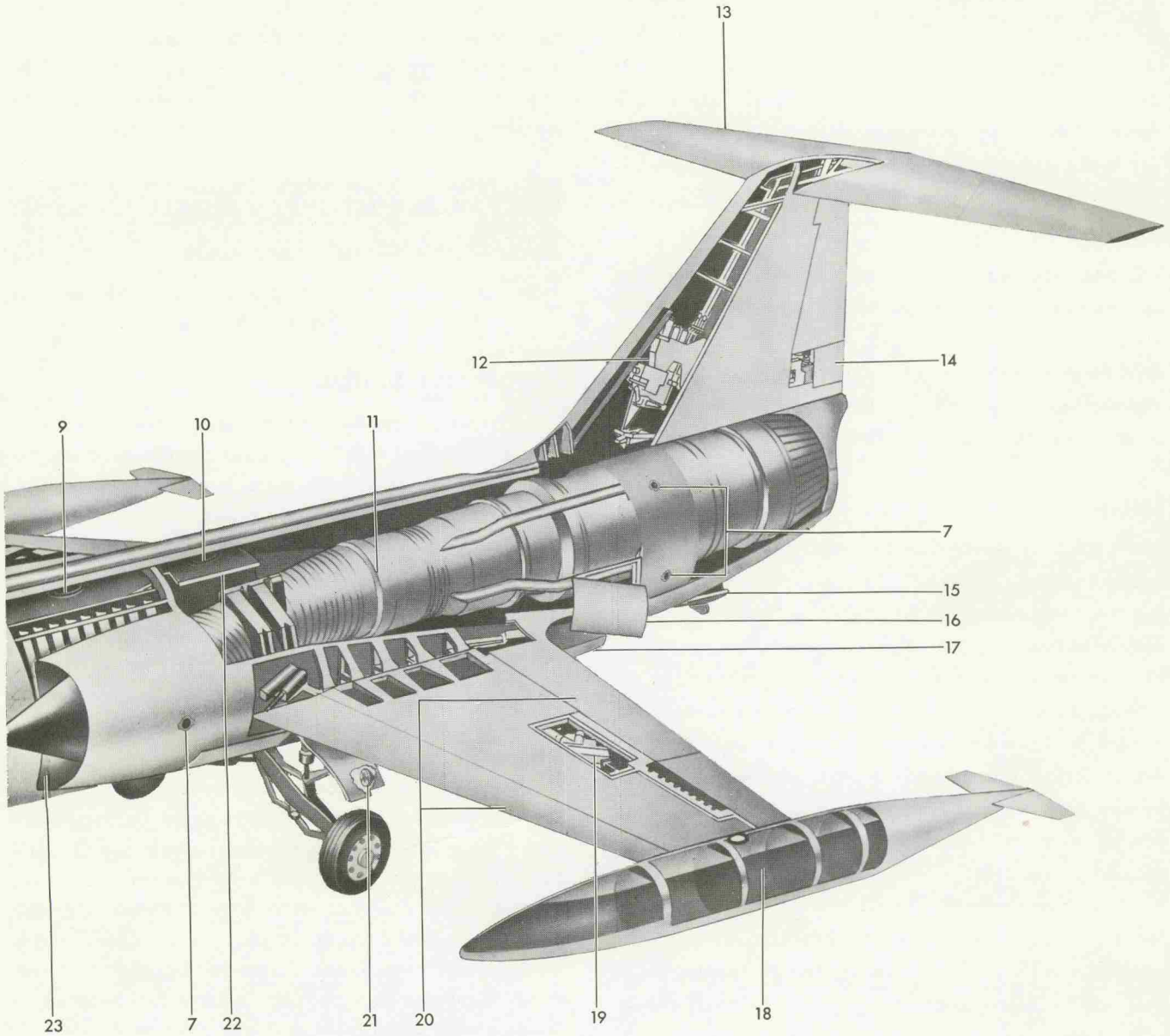
1 PITOT STATIC BOOM
2 AN/ASG-14 RADAR ANTENNA
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12 HORIZONTAL STABILIZER POWER CONTROL ASSEMBLY
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14 YAW DAMPER
15 DRAG CHUTE STOWAGE COMPARTMENT
16 SPEED BRAKE
17 VENTRAL FIN
18 TIP TANK
19 LEFT AILERON POWER CONTROL ASSEMBLY
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AM2129
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Figure 1-2



21 LEFT LANDING LIGHT
22 REFUELING WELL (FORWARD MAIN AND AFT CELLS)
23 ENGINE AIR INTAKE DUCT
24 UHF ANTENNA
25 20MM GUN
26 TAXI LIGHT
27 DOWNWARD EJECTION SEAT
28 IFF ANTENNA
29 AN/ASG-14 RADAR
30 GLIDE SLOPE ANTENNA

F50-3-1-6-2

AIRPLANE GROSS WEIGHT.

The approximate take-off gross weights of the airplane (including full internal fuel and pilot) are as follows:

No external load	19,502 pounds
Sidewinders	19,936 pounds
Tip tanks	22,007 pounds
Sidewinders and pylon tanks	22,986 pounds
Tip tanks and pylon tanks	25,057 pounds

ARMAMENT.

The basic armament consists of one 20 mm electrically operated gun located in the lower left side of the forward fuselage. Its ammunition supply of 750 rounds is fired at an average rate of 4000 rounds per minute. In addition, one Sidewinder air-to-air guided missile may be carried at each wing tip in place of tip tanks.

ENGINE.

The airplane is powered by a General Electric J79-GE-3A turbojet engine (figure 1-3). Its sea level static thrust rating is approximately 15,000 pounds at Maximum Power (full afterburning) and approximately 9,600 pounds at Military Power (maximum power non-afterburning). A 17-stage axial flow compressor, driven by a three stage turbine produces a compressor pressure ratio of 12 to 1. An anti-icing system and aerodynamic, variable area exhaust nozzles are provided. The inlet guide vanes and first six stages of stator blades are variable and are positioned by engine fuel pressure, automatically controlled by the engine fuel control unit as a function of engine speed and engine air inlet temperature. By varying vane angle, the volume of air passing through the engine is controlled. This reduces the possibility of compressor stall. During operation, air is drawn or rammed into the air inlet at the front of the engine where it is directed into the compressor by the variable inlet vanes located in the compressor front frame. The compressor compresses the air and forces it rearward. The compressed air is channeled through the compressor rear frame, which supports the fuel nozzles, and into the combustion liners. The fuel nozzles project into the combustion liners and eject a spray of fuel that is mixed with the compressed air and burned. The resulting expanded and heated gases flow at high velocities through the transition liner and into the turbine section. The three turbine wheels are bolted together and move as a unit on one common shaft which is directly splined to the compressor rotor. The exhaust gases, after passing through the turbine section, pass into the afterburner. At this point, additional fuel may be injected into the hot exhaust gases and burned, thus producing considerable thrust augmentation. The exhaust gases then pass through the variable area exhaust nozzles and are dissipated into the atmosphere.

J79 TURBOJET ENGINE WITH AFTERBURNER

Refer to Confidential Supplement. T.O. 1F-104A-1A Figure 1-3.

ENGINE FUEL SYSTEM.

The system (figure 1-4) incorporates a fuel control unit which is supplied with high pressure fuel from an engine-driven, high pressure fuel pump. Fuel is supplied to this pump at aircraft boost pump pressure. Metered fuel from the engine fuel control unit is piped through an oil cooler and then enters the pressurizing and drain valve. The outlet from the pressurizing valve is connected directly to the main fuel manifold. An afterburner fuel system is provided also. Refer to Afterburner System in this Section.

Engine Fuel Pump. An engine driven pump having a centrifugal boost element and two gear type elements supplies the high fuel pressure required by the engine fuel system. The centrifugal boost element supplements fuel cell boost pump pressure. Failure of the centrifugal boost element does not affect engine operation except during high fuel demands required at low altitude-high Mach number conditions. Either of the gear type elements is capable of supplying sufficient fuel pressure to the engine fuel control unit if one element should fail.

Engine Fuel Control Unit. This control is a hydro-mechanical device which uses engine fuel as the hydraulic controlling medium. Fuel flow to the engine is controlled by the throttle and is delivered and regulated by the engine fuel control unit. Fuel from the engine pump enters the fuel control through an inlet strainer. The strainer screen is spring-loaded and will by-pass fuel to the fuel control if the screen becomes clogged. Major basic control elements consist of a metering valve and a by-pass valve. The by-pass valve maintains a constant pressure head across the metering valve by by-passing excess fuel back to the engine fuel pump inlet.

MAIN DIFFERENCE TABLE

	F-104A(1) (AF552955 THRU AF56-763)	F-104A (AF56-764 & SUBS.)	F-104C	F-104B	F-104D
CREW	ONE	ONE	ONE	TWO	TWO
ENGINE	J79-GE-3 OR J79-GE-3A	J79-GE-3A	J79-GE-7	J79-GE-3A	J79-GE-7
NOSE WHEEL STEERING	TRAVEL UNRESTRICTED BY WING FLAP LEVER POSITION	TRAVEL UNRESTRICTED BY WING FLAP LEVER POSITION	TRAVEL UNRESTRICTED BY WING FLAP LEVER POSITION	TRAVEL RESTRICTED WITH WING FLAP LEVER IN THE UP POSITION	TRAVEL RESTRICTED WITH WING FLAP LEVER IN THE UP POSITION
NOSE GEAR	RETRACTS FORWARD	RETRACTS FORWARD	RETRACTS FORWARD	RETRACTS AFT	RETRACTS AFT
REFUELING PROVISIONS	GRAVITY FILLING	GRAVITY FILLING	PRESSURE TYPE SINGLE-POINT AND AIR REFUELING	GRAVITY FILLING	GRAVITY FILLING
OXYGEN SYSTEM	HIGH PRESSURE GASEOUS WITH SURVIVAL KIT CONTROL PANEL AND MB-2 OR MD-2 PRESSURE DEMAND REGULATOR	HIGH PRESSURE GASEOUS OR 5 LITER LIQUID WITH SURVIVAL KIT CONTROL PANEL AND MD-1 OR MD-2 PRESSURE DEMAND REGULATOR	5 LITER LIQUID WITH SURVIVAL KIT CONTROL PANEL AND MD-1 PRESSURE DEMAND REGULATOR	10 LITER LIQUID WITH SURVIVAL KIT CONTROL PANEL AND MD-1 PRESSURE DEMAND REGULATOR	10 LITER LIQUID WITH SURVIVAL KIT CONTROL PANEL AND MD-1 PRESSURE DEMAND REGULATOR
EJECTION SEAT	B-2, C, OR C-1	C-1	C-1	C-1	C-1

F50-5-1-110

Figure 1-3A

The metering valve is positioned in response to various internal operating signals, and meters fuel to the engine as a function of these integrated signals. Interconnected with the fuel control are the nozzle area control and the electronic temperature control systems. The nozzle area control system forces the nozzle area to follow a predetermined schedule which is linked to the throttle lever position. The electronic temperature control translates exhaust gas temperature into a command signal that causes the nozzle control to open the exhaust gas exit area if exhaust gas temperature exceeds safe limits.

Cut-Off Valve. A cut-off valve, located near the fuel outlet port of the engine fuel control unit, will shut off the fuel supply to the engine burners when the pilot retards the throttle to OFF.

Linkage System. Mechanical linkage, enclosed in rigid conduit, transmits mechanical motion from the throttle to those accessories requiring mechanical coordination to obtain the degree of thrust desired. Movement of the throttle is transmitted by mechanical linkage to the engine fuel control unit. Teleflex cable from the control box links the nozzle area control, and the afterburner fuel control. A separate teleflex transmits a push-pull motion from the nozzle area control to the variable nozzle hydraulic pump. Responding to this motion the nozzle pump regulates its hydraulic output to position the exhaust nozzle. The exhaust nozzle originates a mechanical response, transmitted via teleflex cable, which signals a nozzle position status. This nozzle position status signal is commonly known as "feedback." Another feedback signal is originated by the inlet guide vanes and variable stator vanes to transmit a position status signal, via teleflex, to the engine fuel control unit. These feedback signals indicate, to the initiating accessory, that the required response has been attained.

Inlet Guide Vanes and Variable Stator Vanes Control. The inlet guide vanes and variable stator vanes are hydraulically positioned as a function of engine air inlet temperature and engine speed to control air flow through the initial stages of compression. The two actuators of this system are supplied with engine fuel pressure from the engine fuel control unit. The mechanical feedback linkage of this system prevents overtravel or vane angle error by transmitting a signal to the servo piston in the engine fuel control unit. When engine speed is reduced, reverse pressure is applied to the actuators and the inlet guide vanes and variable stator vanes begin to close.

INLET GUIDE VANES EMERGENCY RESET SWITCH.

An inlet guide vane emergency reset switch (34A, figure 1-6) is installed on the lower left portion of the instrument panel. The switch is labeled IGV CONT and has two positions, MAN and AUTO. The switch is placed in the AUTO position for normal ground and in-flight operation. If the compressor inlet temperature sensor fails, allowing the inlet guide vanes to move off schedule toward the open position, the pilot can place the switch in the MAN position. This will provide an average temperature setting to the main fuel control and allow limited engine operation with the inlet guide vanes in a fixed temperature rescheduled position. The system is energized through the battery bus.

Fuel-Oil Cooler. The purpose of the fuel-oil cooler is to remove heat from the engine oil system. This is accomplished by the main fuel flow acting as a coolant to control the temperature of the oil. The assembly consists of a fuel-oil cooler, 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 by-pass valve system is designed so that if the oil is cool it will not go through the cooler; also, it will allow the fuel to by-pass the cooler if the fuel flow becomes high or it will by-pass oil if the fuel becomes too hot.

Pressurizing and Drain Valve. The pressurizing and drain valve maintains back pressure to the engine fuel control system in order to provide an acceptable fuel pressure level for servo operation. The valve opens to allow fuel flow to the engine when discharge pressure exceeds a preset value. The drain valve in the unit drains the engine fuel manifold whenever the engine is shut down.

THROTTLE.

The throttle quadrant (figure 1-5) is labeled OFF, IDLE, and FULL. The throttle is spring-loaded in the inboard direction so that forward motion from OFF will cause it to drop into IDLE without forcing. Full travel from IDLE to the point where Military Power is reached is by a straight, forward movement. A potentiometer with a built-in switch for gunsight manual range control is installed in the grip assembly of the throttle. The throttle also incorporates the speed brake switch and two push-button switches, one for the microphone and one for gunsight cage. A throttle actuated switch for the landing gear warning signal circuit is installed in the throttle quadrant. During engine ground starts the throttle is moved from OFF to IDLE position. This opens the fuel cut-off valve. After the engine has started, throttle

ENGINE FUEL SYSTEM

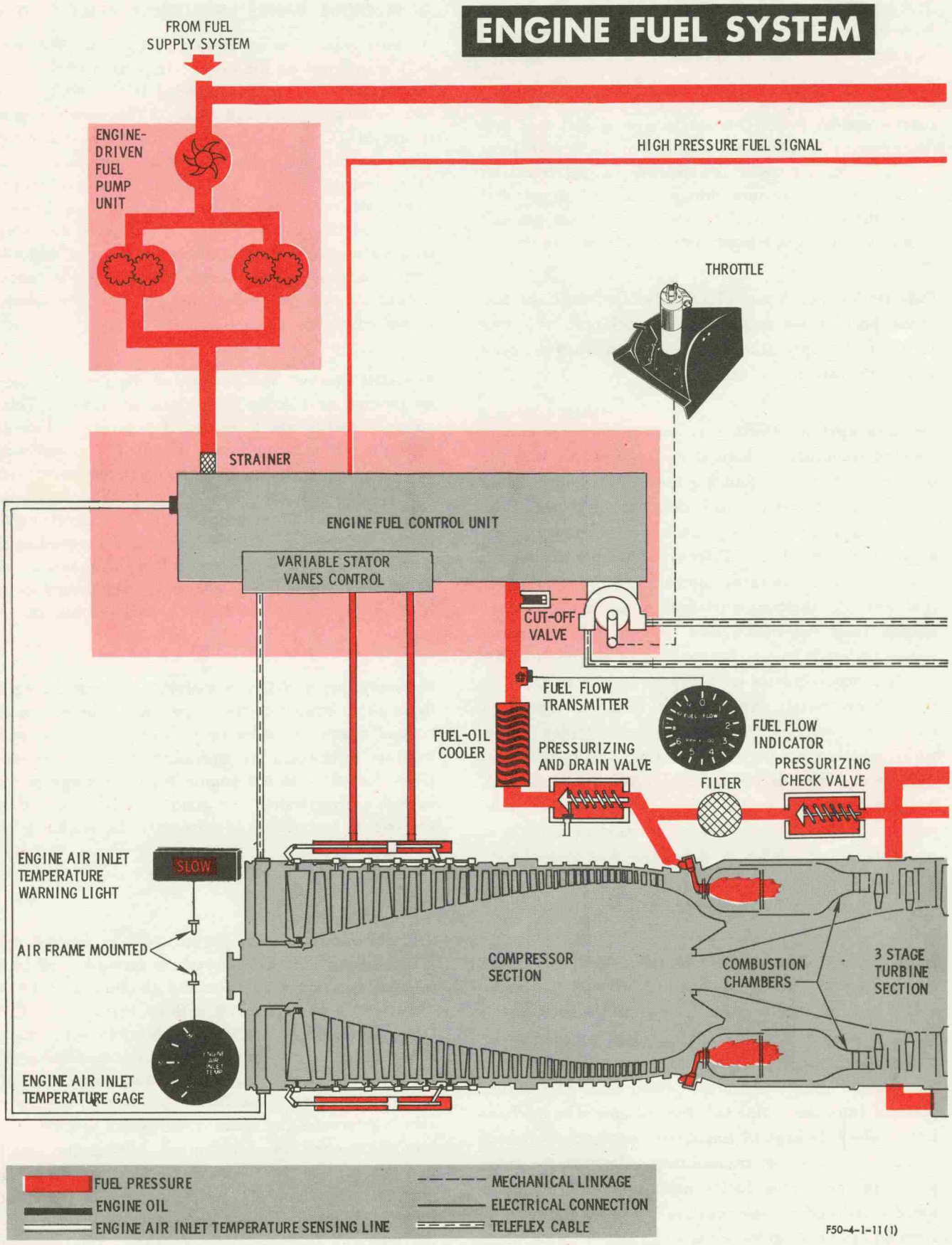
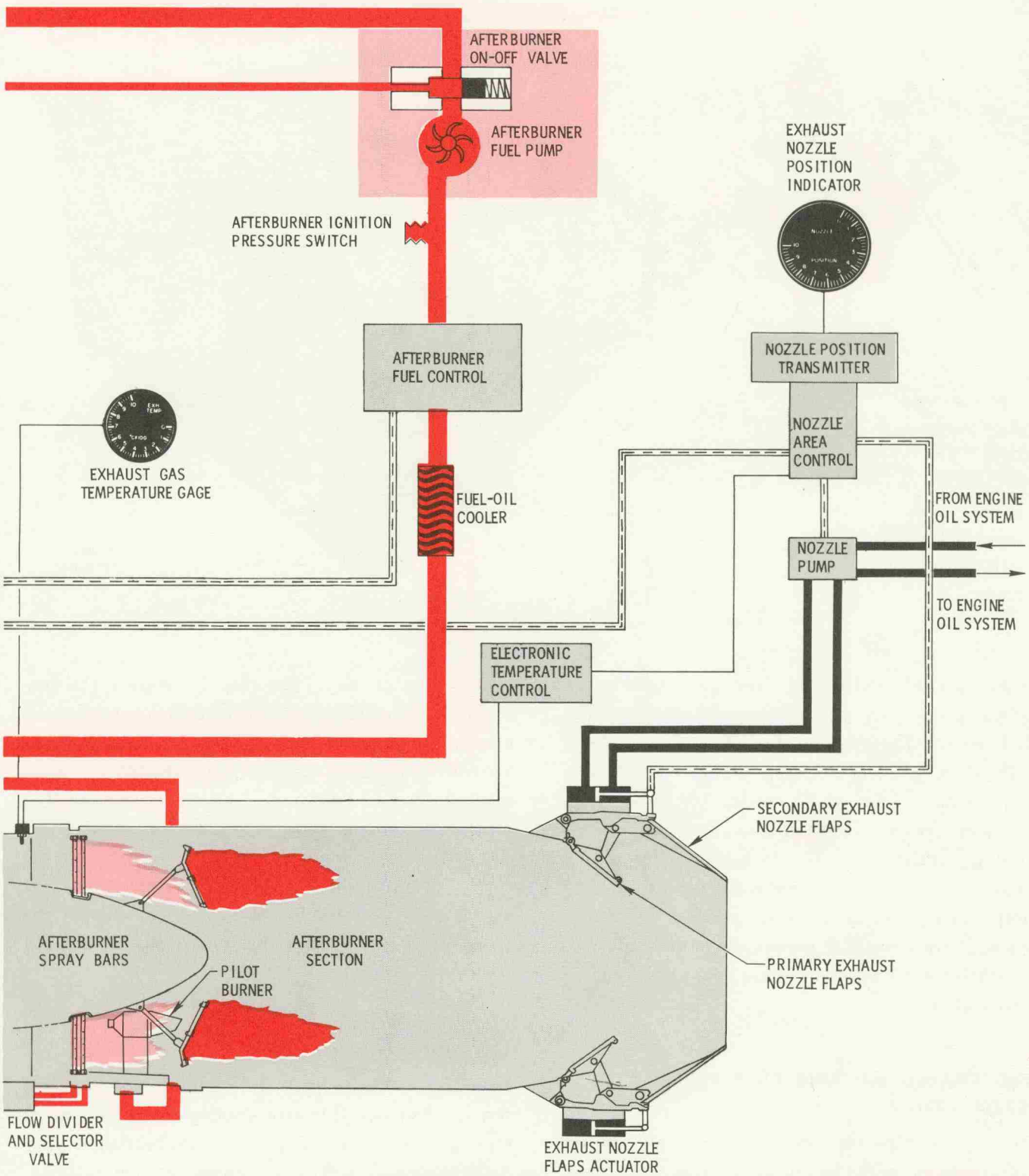
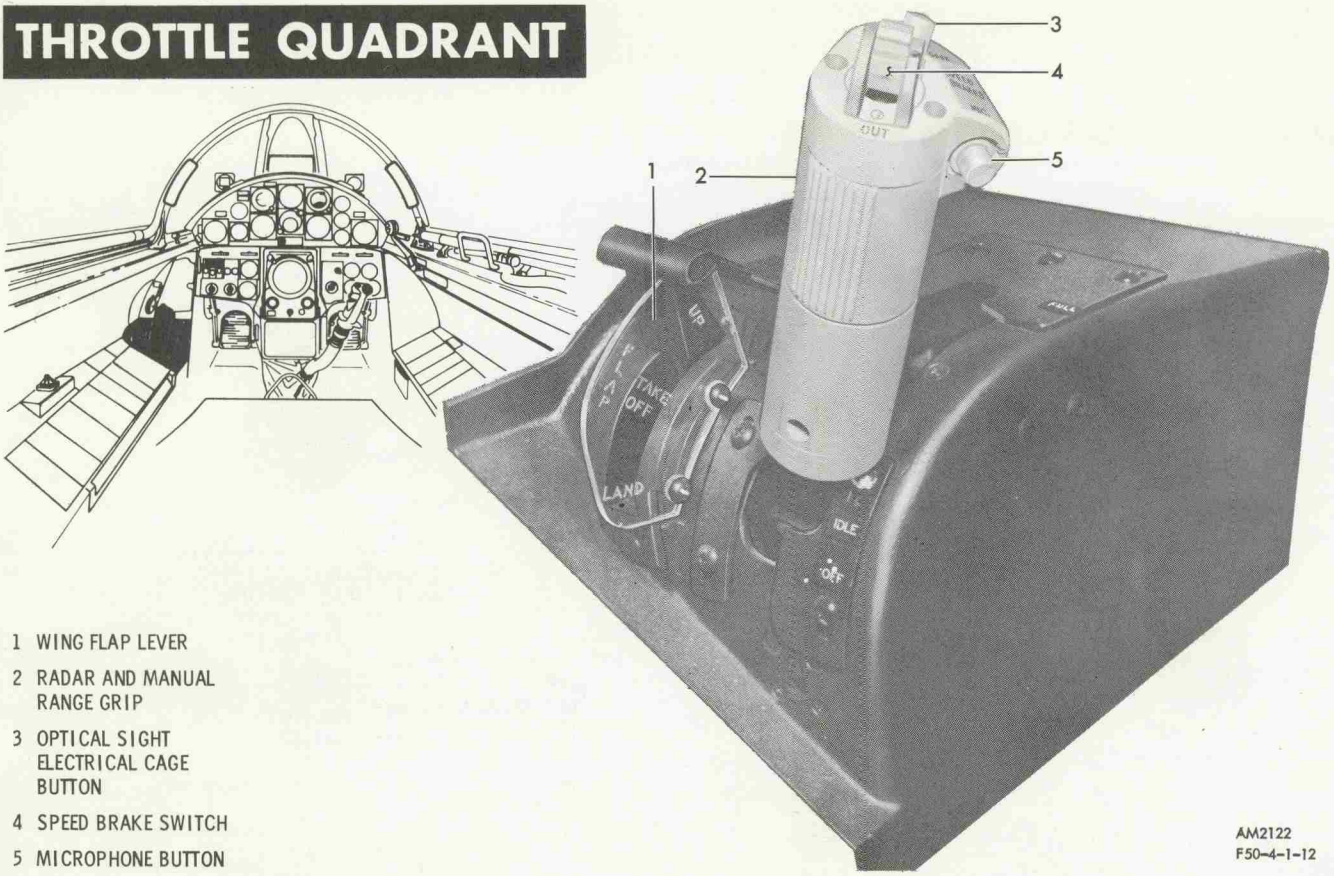


Figure 1-4


AFTERBURNER ON-OFF VALVE
AFTERBURNER FUEL PUMP
EXHAUST NOZZLE POSITION INDICATOR
AFTERBURNER IGNITION PRESSURE SWITCH
AFTERBURNER FUEL CONTROL
NOZZLE POSITION TRANSMITTER
NOZZLE AREA CONTROL
EXHAUST GAS TEMPERATURE GAGE
FUEL-OIL COOLER
FROM ENGINE OIL SYSTEM
NOZZLE PUMP
TO ENGINE OIL SYSTEM
ELECTRONIC TEMPERATURE CONTROL
SECONDARY EXHAUST NOZZLE FLAPS
AFTERBURNER SPRAY BARS
AFTERBURNER SECTION
PILOT BURNER
PRIMARY EXHAUST NOZZLE FLAPS
FLOW DIVIDER AND SELECTOR VALVE
EXHAUST NOZZLE FLAPS ACTUATOR

F50-4-1-11(2)

THROTTLE QUADRANT



1 WING FLAP LEVER
2 RADAR AND MANUAL RANGE GRIP
3 OPTICAL SIGHT ELECTRICAL CAGE BUTTON
4 SPEED BRAKE SWITCH
5 MICROPHONE BUTTON

Figure 1-5

advancement increases engine speed until 100% rpm is reached. At this point, maximum engine speed is attained and is governed at this rpm throughout the remainder of throttle lever travel including maximum afterburner. Afterburning is initiated by moving the throttle outboard and forward into the afterburner slot. A substantial thrust variation in the afterburner range can be attained by movement of the throttle in this slot. The throttle linkage is designed to provide the correct amount of friction and to prevent the lever from creeping; therefore, no throttle friction control is provided in the cockpit.

AERODYNAMIC VARIABLE AREA EXHAUST NOZZLE SYSTEM.

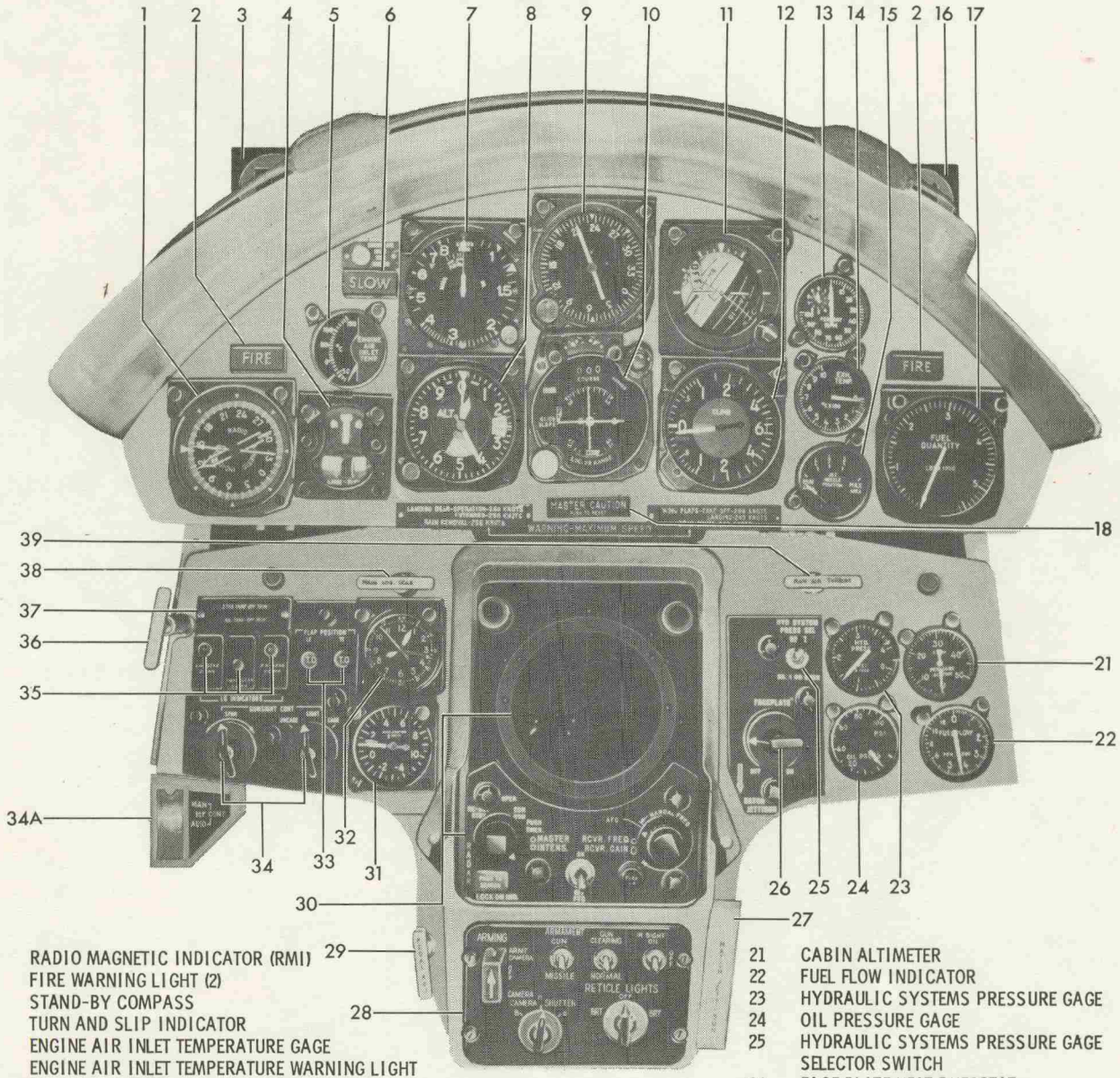
Two sets of flaps, operating together, provide the variable exhaust area. The primary exhaust nozzle flaps, hinged to the aft end of the tail pipe, control the convergent portion of the nozzle while the secondary exhaust nozzle flaps, hinged to a support ring, control the divergent portion of the nozzle. (See figure 1-9.) The two sets of flaps are linked together and maintain a scheduled area-and-spacing ratio which is infinitely variable between extremes. Movement of the flaps is accomplished automatically by four synchronized hydraulic 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 Supply.

Secondary airflow is supplied for engine ground operation through four sets of inward opening ("suck-in") doors (spring loaded to closed) just aft of the firewall. During flight, secondary airflow is supplied through automatically operated engine air by-pass flaps installed at the joint between the main duct and the engine. Electrically operated valves, actuated by the main landing gear door

INSTRUMENT PANEL

(TYPICAL)

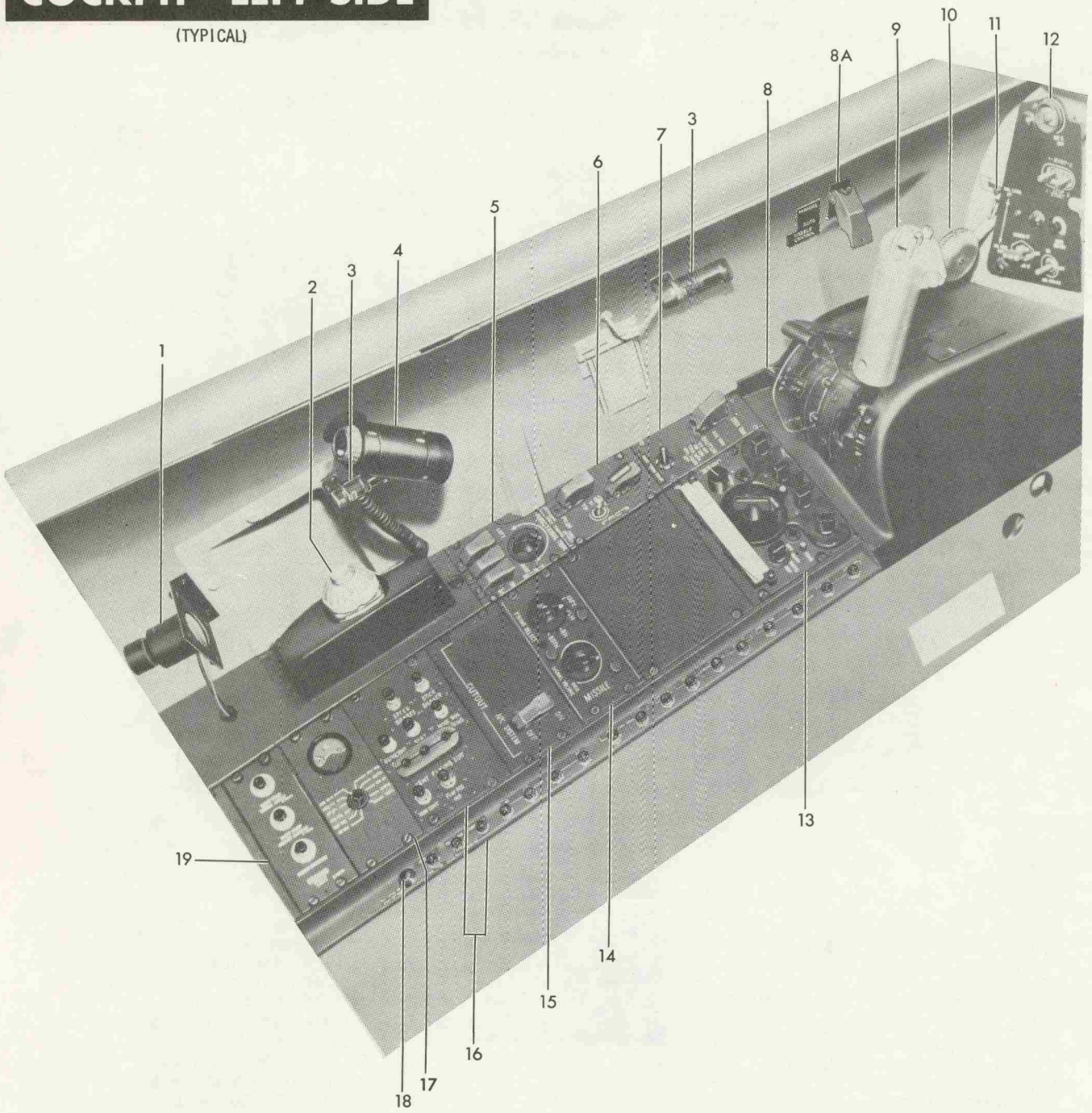


1 RADIO MAGNETIC INDICATOR (RMI)
2 FIRE WARNING LIGHT (2)
3 STAND-BY COMPASS
4 TURN AND SLIP INDICATOR
5 ENGINE AIR INLET TEMPERATURE GAGE
6 ENGINE AIR INLET TEMPERATURE WARNING LIGHT
7 AIRSPEED AND MACH NUMBER INDICATOR
8 ALTIMETER
9 DIRECTIONAL INDICATOR
10 COURSE INDICATOR
11 ATTITUDE INDICATOR
12 VERTICAL VELOCITY INDICATOR
13 TACHOMETER
14 EXHAUST GAS TEMPERATURE GAGE
15 EXHAUST NOZZLE POSITION INDICATOR
16 AUTO-PITCH CONTROL INDICATOR
17 FUEL QUANTITY INDICATOR
18 MASTER CAUTION LIGHT AND RESET BAR
19 DELETED
20 DELETED
21 CABIN ALTIMETER
22 FUEL FLOW INDICATOR
23 HYDRAULIC SYSTEMS PRESSURE GAGE
24 OIL PRESSURE GAGE
25 HYDRAULIC SYSTEMS PRESSURE GAGE SELECTOR SWITCH
26 FACE PLATE HEAT RHEOSTAT
27 ESCAPE HATCH JETTISON HANDLE
28 ARMAMENT CONTROL PANEL
29 RUDDER PEDAL ADJUSTMENT HANDLE
30 RADAR SCOPE AND CONTROL PANEL
31 ACCELEROMETER
32 CLOCK
33 WING FLAP POSITION INDICATORS
34 GUNSIGHT CONTROL SWITCHES
34A INLET GUIDE VANES SWITCH
35 LANDING GEAR POSITION INDICATOR LIGHTS
36 DRAG CHUTE HANDLE
37 STABILIZER AND AILERON TAKE-OFF TRIM INDICATOR LIGHTS
38 MANUAL LANDING GEAR RELEASE HANDLE
39 RAM AIR TURBINE EXTENSION HANDLE

Figure 1-6

COCKPIT—LEFT SIDE

(TYPICAL)



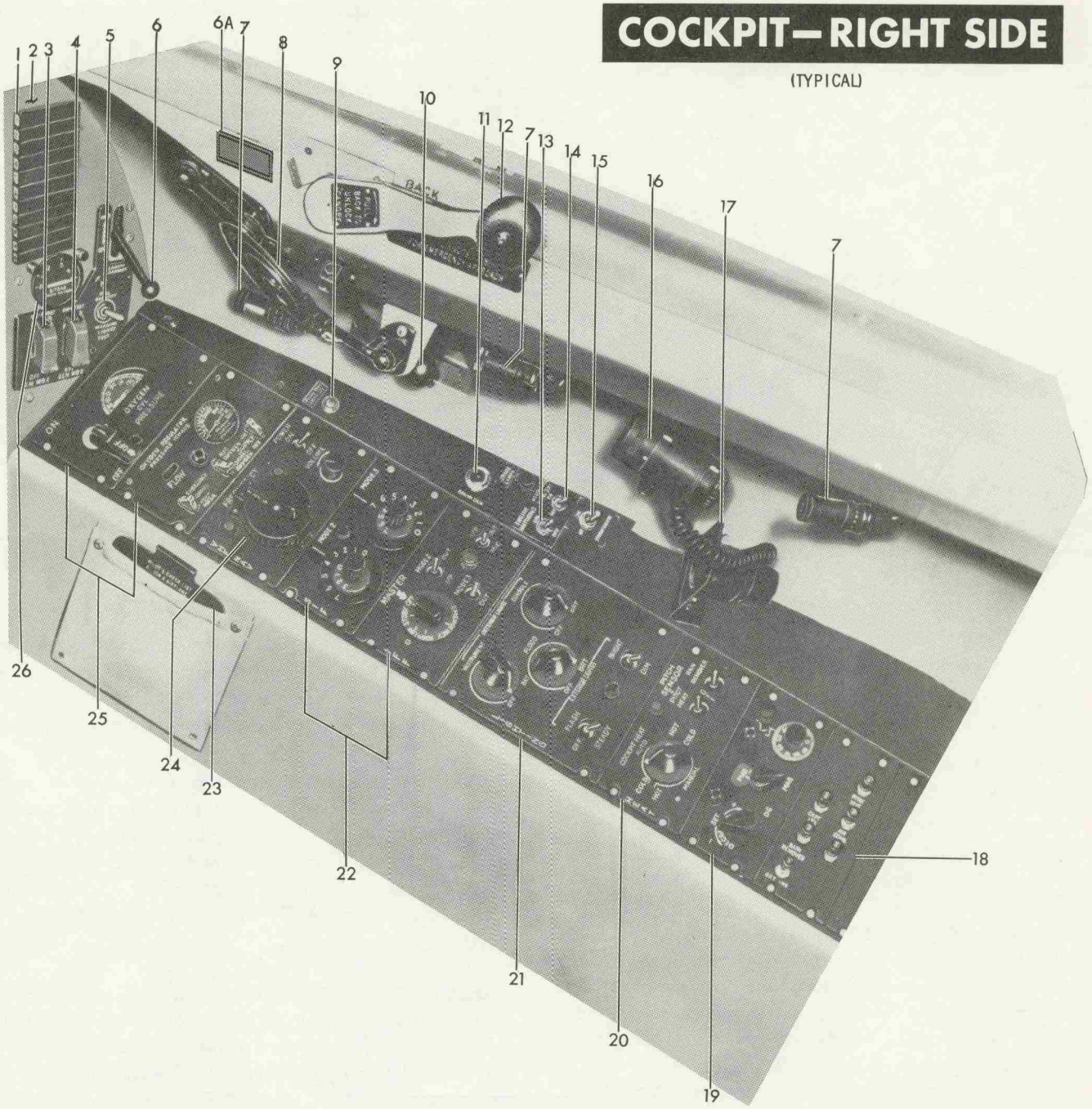
1 THUNDERSTORM LIGHT
2 ANTI-G SUIT VALVE
3 FLOODLIGHT (2)
4 SPOTLIGHT
5 TRIM AND STABILITY CONTROL PANEL
6 FUEL CONTROL PANEL
7 AUXILIARY TRIM CONTROL PANEL
8 WING FLAP LEVER
8A EXHAUST NOZZLE SWITCH
9 THROTTLE
10 LANDING GEAR LEVER
11 LANDING GEAR LEVER OVERRIDE BUTTON
12 LEFT FORWARD PANEL
13 UHF COMMAND CONTROL PANEL (AN/ARC-66)
14 MISSILE CONTROL PANEL
15 AUTO-PITCH CUTOUT PANEL
16 LEFT CIRCUIT BREAKER PANELS
17 RADAR TEST PANEL
18 FUEL PRESSURE AIR SHUT-OFF TEST SWITCH
19 ARMAMENT GROUND TEST PANEL

F50-4-1-8

Figure 1-7

COCKPIT—RIGHT SIDE

(TYPICAL)



1 WARNING LIGHT PANEL
2 RIGHT FORWARD PANEL
3 NO. 1 GENERATOR SWITCH
4 NO. 2 GENERATOR SWITCH
5 FUEL QUANTITY AND WARNING LIGHTS SYSTEMS TEST SWITCH
6 CANOPY DEFROSTING LEVER
6A ENGINE OIL PRESSURE RECORD CARD
7 FLOOD LIGHT (3)
8 COMPUTER
9 PYLON TANKS EMPTY LIGHT
10 RAM AIR SCOOP LEVER
11 GROUND CREW INTERPHONE BUTTON
12 CANOPY INTERNAL LOCKING LEVER
13 ENGINE MOTORING SWITCH
14 VENTILATED SUIT BLOWER SWITCH
15 THUNDERSTORM LIGHTS SWITCH
16 SPOTLIGHT
17 THUNDERSTORM LIGHT
18 RIGHT CIRCUIT BREAKER PANEL
19 J-4 DIRECTIONAL INDICATOR CONTROL PANEL
20 HEATING CONTROL PANEL
21 LIGHTING CONTROL PANEL
22 IFF CONTROL PANELS
23 CHECK LIST
24 VHF NAVIGATION CONTROL PANEL (AN/ARN-56)
25 OXYGEN CONTROL PANELS
26 LIQUID OXYGEN QUANTITY GAGE (AF SERIAL 56-789 AND SUBSEQUENT)

F-50-4-1-9

Figure 1-8

EXHAUST NOZZLE

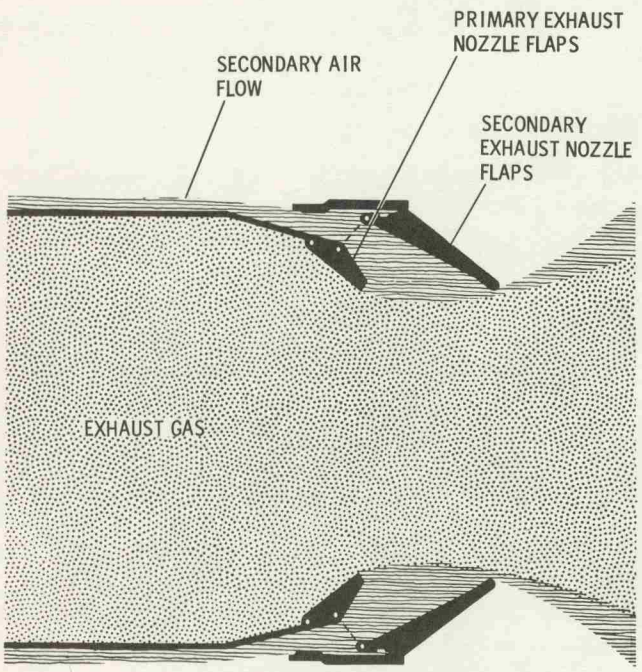


Figure 1-9

uplock switches, allow No. 2 hydraulic pressure to open the two lower by-pass flaps. When the landing gear is extended, the two lower flaps are closed. Electrical power for operation of the by-pass flap valves is derived from the d.c. monitored bus.

Emergency Blow-Out Panel. An emergency blow-out panel is installed on the bottom of the airplane forward of the hydraulic panel to relieve excessively high pressures which could build up in the aft fuselage under certain circumstances; for example in the event of afterburner flame-out, or failure in any part of the by-pass flaps control or exhaust nozzle systems which might cause the flaps to stay open while at high supersonic speeds at lower altitudes.

Nozzle Area Control System.

The nozzle area control system incorporates an exhaust gas temperature sensing control, nozzle area control, and an engine driven control alternator. These units function to position the engine exhaust nozzle to conform with engine operating requirements. This feature is effective throughout the entire engine operating range but is primarily important in the Military Power and afterburning regions. There is a mechanical schedule of the nozzle area control which starts closing the exhaust nozzle immediately following the IDLE range of throttle position and continues to the Military Power position where the nozzle is at its smallest opening. If this mechanical schedule were maintained during Military Power and afterburner operation, excessive turbine outlet temperature would be reached long before the full afterburner schedule was completed. An electronic temperature control which is powered by an engine-driven control alternator will prevent this condition. The electronic temperature control will override the mechanical schedule of the nozzle area control, opening the nozzles as required to keep temperatures within operating limits. (Refer to figure 1-13 for nozzle area at various engine speeds.)

Electronic Temperature Control. The purpose of the electronic temperature control is to compare engine exhaust gas temperature with a reference temperature and then send signals to open the nozzle if necessary. It is cooled by fuel circulating through the underside of the unit. An engine driven alternator, independent of aircraft electrical power, supplies power for the electronic temperature control.

Nozzle Pump. The nozzle hydraulic pump is an engine-driven, variable pressure, variable displacement pump. The amount and direction of flow are determined by a teleflex push-pull signal from the nozzle area control.

Exhaust Nozzle Flap Actuators. There are four nozzle hydraulic actuators which are supplied with high pressure engine oil from the nozzle hydraulic pump. The four actuators are equally spaced and are located on the tailpipe. The nozzle actuators mechanically open and close the nozzles automatically through a series of rods and levers.

Exhaust Nozzle Control Switch. An exhaust nozzle control switch (8A, figure 1-7) is located on the left side of the cockpit just forward of the throttle quadrant. The switch is labeled NOZZLE CONTROL and has two positions, MANUAL and AUTO. The electronic temperature control of the nozzle area control system is energized with the switch in the AUTO position. If the electronic temperature control system should malfunction, the pilot may override the automatic feature and close the exhaust nozzle to the mechanical schedule by placing the switch in MANUAL. With the switch in MANUAL, there is no automatic temperature modulation. The switch is powered from the d.c. emergency bus.

Exhaust Nozzle Position Indicator. An indicator (15, figure 1-6) located on the right side of the instrument panel shows the exit area of the exhaust nozzle. The instrument is placarded "JET NOZZLE POSITION"

and is calibrated from MIN. AREA in ¼th increments (or 0 through 10 on AF Serial 56-826 and subsequent) to MAX AREA. Power for the instrument is derived from the d.c. monitored bus.

ENGINE STARTER AND IGNITION SYSTEMS.

Engine Starter System.

The engine is equipped with an air driven starter which requires air from a ground turbine compressor. The receptacle for connecting the air supply line is located in the right main wheel well. A special electrical receptacle located adjacent to this air connector may or may not be connected by the ground crew with a control valve on the ground turbine compressor. If this electrical connection is made, the pilot has control of the start; starting air to the engine starter is provided through electrical operation of the valve by the pilot. A centrifugal switch closes the air valve to disengage the starter automatically between 42 and 47% engine rpm. If this electrical connection is not made, the ground crew manually positions the valve open and close on signal from the pilot.

Engine Ignition System.

Dual ignition systems are provided for air start reliability. Ignition power is supplied by the battery bus and is used during engine start only. When energized, the ignition circuit selected will supply power to its respective spark plug for 45 seconds or until manually discontinued. The spark plug will ignite the fuel air mixture in its vicinity. Combustion is propagated through crossfire tubes.

Start Switches.

Two start switches (figure 1-10) are located on the left forward panel and are labeled No. 1 and No. 2. The switches have a START, STOP-START and a center NEUTRAL position. The switches are spring-loaded from the START and STOP-START positions to NEUTRAL. By momentarily placing either start switch to START, battery bus power is supplied to energize the ignition circuit and begin the 45 second ignition cycle. Placing the start switch in the STOP-START position discontinues ignition. Both switches may be used simultaneously to energize the ignition systems during air starts. This provides dual airstart reliability. When the special electrical connection is made between the airplane and the control valve, the start switch also controls starting air.

Engine Motoring Switch.

The engine motoring switch (13, figure 1-8), located on the right console, is spring loaded from ON to OFF. It is provided to energize the ground turbine compressor air valve which directs compressor air to the engine starter for motoring the engine without ignition. The engine motoring switch is electrically powered from the battery bus. (Refer to Engine Starter System in this Section for information on the ground turbine compressor and engine starter.)

Fuel Flow Indicator.

The fuel flow indicator (22, figure 1-6) located on the lower right instrument panel is operated by a flowmeter installed in the engine fuel line. This instrument indicates fuel consumption rate in pounds per hour and is calibrated to 12,000 pounds. The system receives 26-volt a.c. power from the instrument a.c. bus through the instrument auto-transformer and fuses on the electronic compartment circuit breaker panel. The instrument does not indicate afterburner fuel flow.

Engine Air Inlet Temperature Gage.

A temperature gage (5, figure 1-6), located on the upper left side of the instrument panel, measures engine air inlet temperature. The temperature detector is located in the right 20 KVA, a.c. generator blast tube which carries engine inlet air from the right intake duct. The instrument is calibrated from −50°C to +150°C and receives power from the 28-volt d.c. monitored bus.

Engine Air Inlet Temperature Warning Light.

A placard type warning light (6, figure 1-6) located on the upper left side of the main instrument panel is powered from the d.c. emergency bus. This light will be energized when engine air inlet temperature exceeds the limitation. When energized the word "SLOW" will flash on and off and alert the pilot to slow the airplane to avoid engine damage.

Exhaust Gas Temperature Gage.

An exhaust gas temperature gage (14, figure 1-6) located on the right side of the main instrument panel is calibrated from 0°C to 1000°C. The unit is operated electrically by self-generating thermocouples and provides visual indications of engine operating temperature.

Tachometer.

The tachometer (13, figure 1-6) mounted on the right side of the main instrument panel, registers engine speed in percentage of maximum rated rpm (7460). The instrument is powered by a tachometer generator which generates a frequency proportional to engine speed and therefore does not require power from the airplane electrical system.

LEFT FORWARD PANEL

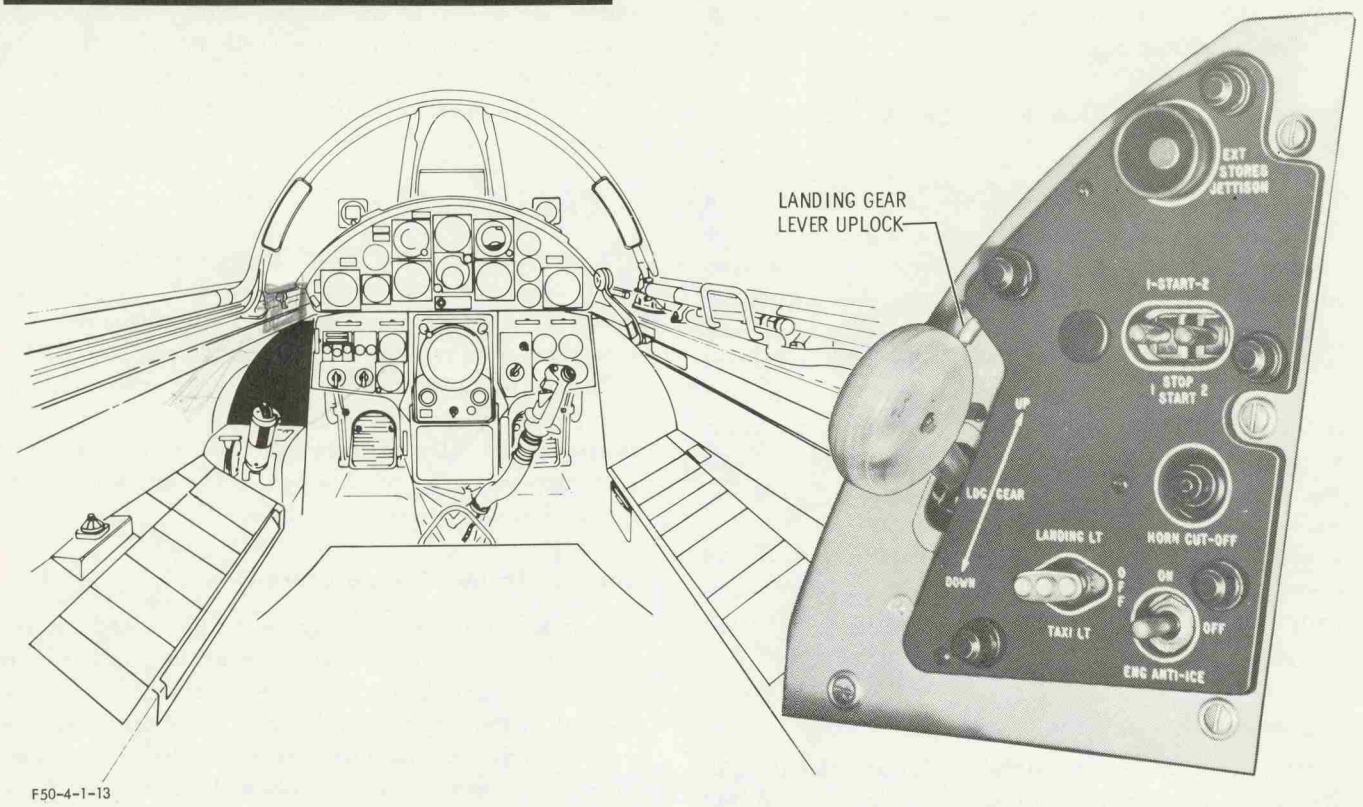


Figure 1-10

Oil Pressure Gage.

An oil pressure gage (24, figure 1-6) is mounted on the right side of the lower instrument panel. The gage registers oil pressure in pounds per square inch. The gage receives power from the instrument a.c. bus through the 26 volt auto-transformer.

AFTERBURNER SYSTEM.

The afterburner section is located just aft of the turbine section and is comprised of the tail pipe, aerodynamic variable area exhaust nozzle, pilot burner, spray bars and manifold. The purpose of the afterburner section is to provide thrust augmentation by injecting additional fuel into the exhaust gases and igniting the mixture.

AFTERBURNER FUEL SYSTEM.

Fuel from the aircraft fuel tanks is admitted to the afterburner on-off valve, and when the afterburner system is operating the fuel will pass through this valve and into the afterburner fuel pump (see figure 1-4). From the afterburner fuel pump the fuel passes to the afterburner fuel control and then to the afterburner fuel-oil cooler. From the cooler the fuel passes to the flow divider and selector valve and on to the spray bars. Fuel flow is taken immediately downstream of the pressurizing and drain valve in the engine fuel system and is used for ignition purposes. Afterburner fuel pump discharge pressure is sensed by an afterburner fuel pressure switch which controls the operation of the afterburner ignition spark plug. The ignition system receives power from the No. 2 a.c. bus.

Afterburner On-Off Valve.

The afterburner on-off valve is an integral part of the afterburner fuel pump. It is located on the inlet of the pump and allows fuel to pass into the afterburner fuel pump on a high pressure fuel signal from the engine fuel control unit. This signal is controlled first by throttle position and second by engine speed. If the throttle is advanced into the afterburner region, the high pressure fuel signal will be applied to the engine speed control valve in the engine fuel control unit. If engine speed is high enough the signal will pass through this valve and into the afterburner on-off valve, thereby actuating it. If there is no signal the valve will remain closed and no fuel will enter the afterburner pump.

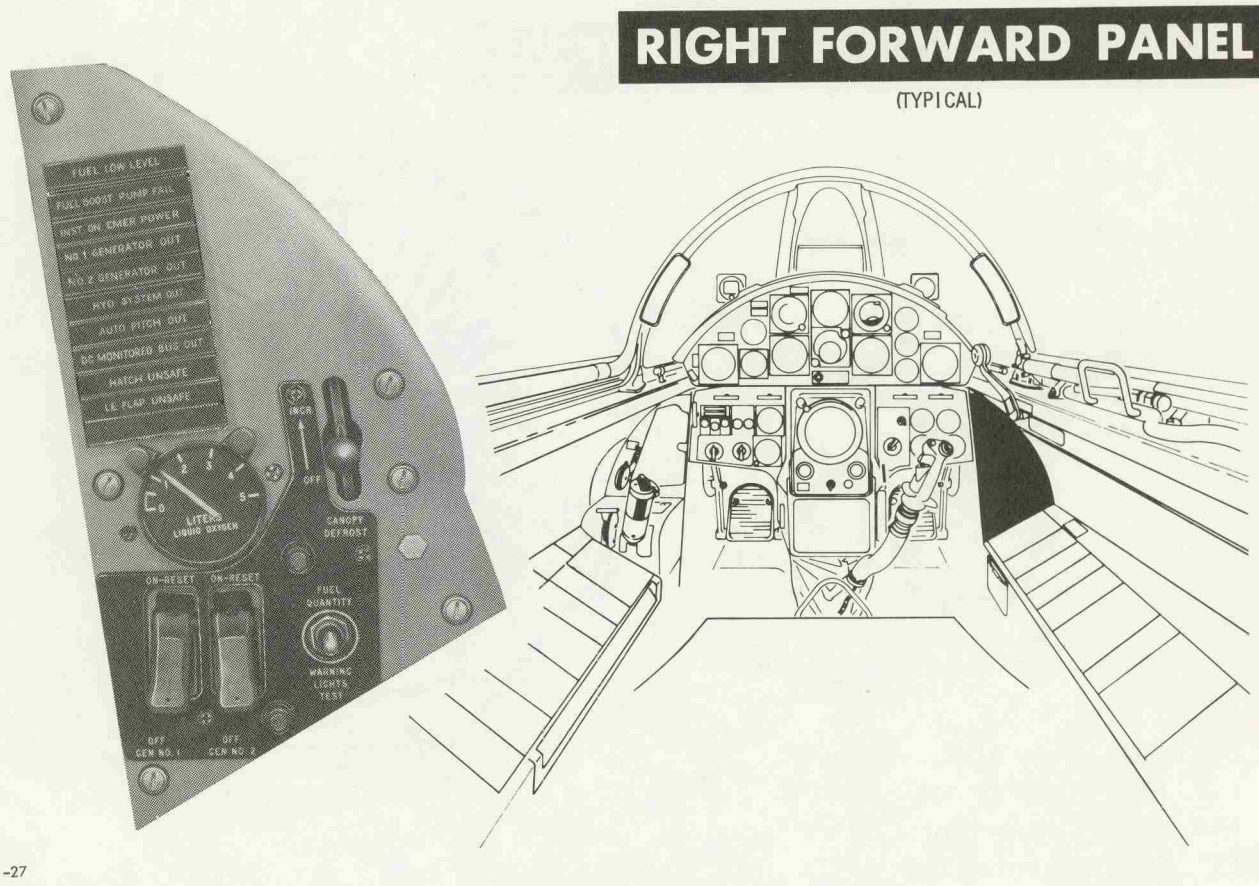


Figure 1-11

Afterburner Fuel Pump.

The afterburner fuel pump is an engine-driven centrifugal pump. It 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 through the use of teleflex cabling. Any movement of the throttle moves the engine fuel control teleflex and also moves the teleflex to the afterburner fuel control. Fuel entering the afterburner fuel control is metered by the fuel control in response to throttle movement and changes in compressor discharge pressure (optimum fuel air ratio). The afterburner fuel control is made to hold a constant pressure drop across an orifice while the area of that orifice is varied in accordance with throttle position and compressor discharge pressure.

Afterburner Fuel-Oil Cooler.

Fuel from the afterburner fuel control passes through the fuel-oil cooler which removes heat from the engine oil in much the same manner as the fuel-oil cooler in the engine fuel system.

Flow Divider and Selector Valve.

Metered fuel from the afterburner fuel control passes through the fuel oil-cooler to the inlet of the flow divider and selector valve. Here fuel is metered to the various sectors of the afterburner to obtain the best pattern for the condition of afterburning required.

Afterburner Sector Light Up.

The flow divider and selector valve assembly distributes the fuel to the spray bars in sequence (see figure 1-12).

There are four stages of fuel flow:

1. Primary sector.
2. Secondary sector.

AFTERBURNER FUEL MANIFOLD

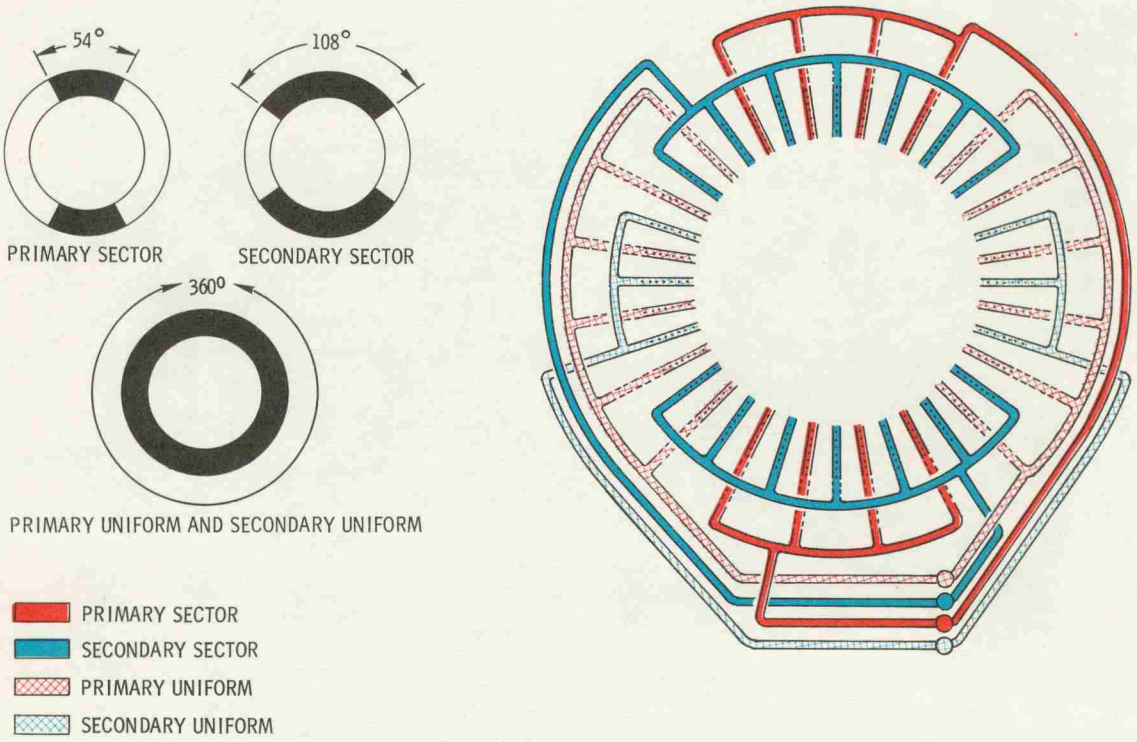


Figure 1-12

3. Primary uniform.
4. Secondary uniform.

When the throttle is first advanced into the afterburner position, the primary sector will light up. Further advancement will cause the secondary sector to light up. When the throttle is advanced still further a distinct increase in thrust occurs. As the throttle is advanced to the maximum afterburner position, the final (secondary uniform) manifold flows fuel. This is full uniform burning.

Afterburner Ignition System.

The afterburner ignition switch is activated by afterburner pump discharge pressure. The afterburner ignition unit receives power from the No. 2 a.c. bus when afterburner pump discharge pressure is above a preset value. The spark plug is located within the pilot burner. During afterburning, operation of the spark plug is continuous to assure positive ignition of the pilot burner.

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

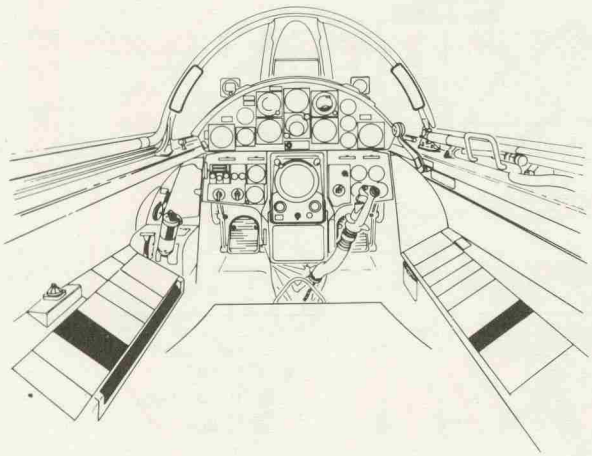
Refer to Confidential Supplement T.O. 1F-104A-1A Figure 1-13.

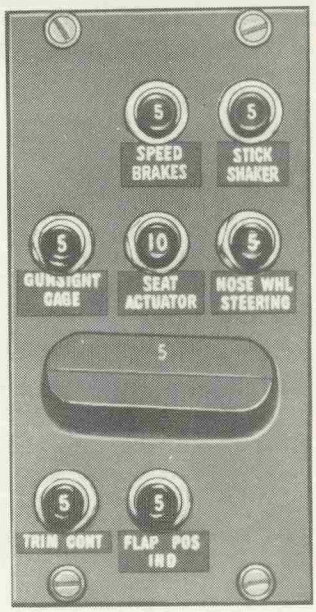
OIL SUPPLY SYSTEM.

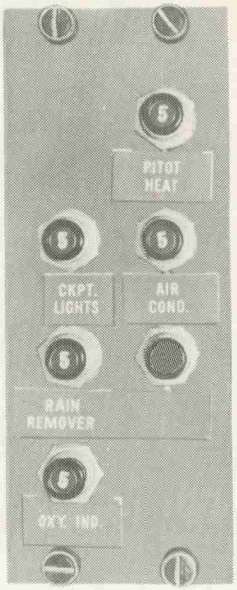
The engine oil system is automatically operated. (Refer to figure 1-30 for oil grade and specification.) The system uses 4 US gallons of oil in a 5 US gallon tank (1 gallon expansion space). Necessary pressure and scavenge pumps and supply lines to those areas requiring lubrication are provided. Engine oil also is used to actuate the exhaust nozzles. The flow of engine oil for the exhaust nozzle actuators is automatically controlled by the nozzle area control system. On modified aircraft, access to the oil quantity dip stick is provided on the top surface of the fuselage directly over the wing.

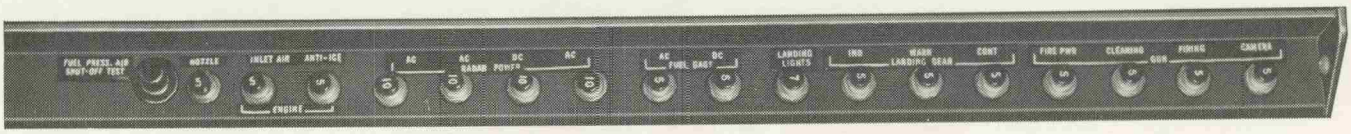
CIRCUIT BREAKER PANELS

(TYPICAL)









F50-3-1-86

Figure 1-14

ENGINE OIL LOW LEVEL WARNING LIGHT (Modified Aircraft).

A warning light is installed on the warning light panel and it illuminates whenever the engine oil supply level decreases to a critical level (approximately one gallon). Illumination of this light also illuminates the master caution light. Both lights are powered from the 28 volt d.c. emergency bus.

ENGINE OIL PRESSURE RECORD CARD.

An engine oil pressure record card (6A, figure 1-8) is provided on the forward right side of the cockpit. This card lists the normal engine oil pressure at Military Power for each engine-airplane combination.

FUEL SUPPLY SYSTEM.

The aircraft fuel system (figure 1-15) consists of one main fuel tank comprised of four separate interconnected, bladder-type non-self-sealing cells, four tank-mounted submerged boost pumps, a shut-off valve, a strainer, and the necessary plumbing and electrical circuits. Flapper valves are installed between the forward main and aft center cells and permit fuel flow by gravity from the aft cells to the forward main cell. A filler well in the aft center cell is provided for servicing all four internal cells. All the necessary plumbing and electrical circuits are provided for the installation of tip and pylon tanks. An auxiliary fuel cell located forward of the main fuel cell transfers fuel by means of a transfer pump and transfer float valve to the forward main cell. A separate filler is used for the auxiliary fuel cell. For fuel specification and grade, refer to figure 1-30. Refer to figure 1-17 for fuel tank capacities.

FORWARD MAIN CELL.

All of the fuel that goes to the engine is fed from the forward main cell. Fuel from the aft center cell enters the forward main cell through two flapper-type check valves. Fuel from the external tanks enters the forward main cell through a transfer float valve. Fuel from the auxiliary fuel cell also enters the forward main cell through a transfer float valve. A low level warning switch, quantity transmitter, four boost pumps, plumbing connections to "boost pump fail" warning switches, two vent valves, and fuel manifold are located inside the cell. Drain valves are located in the four corners of the cell and are accessible from outside the airplane.

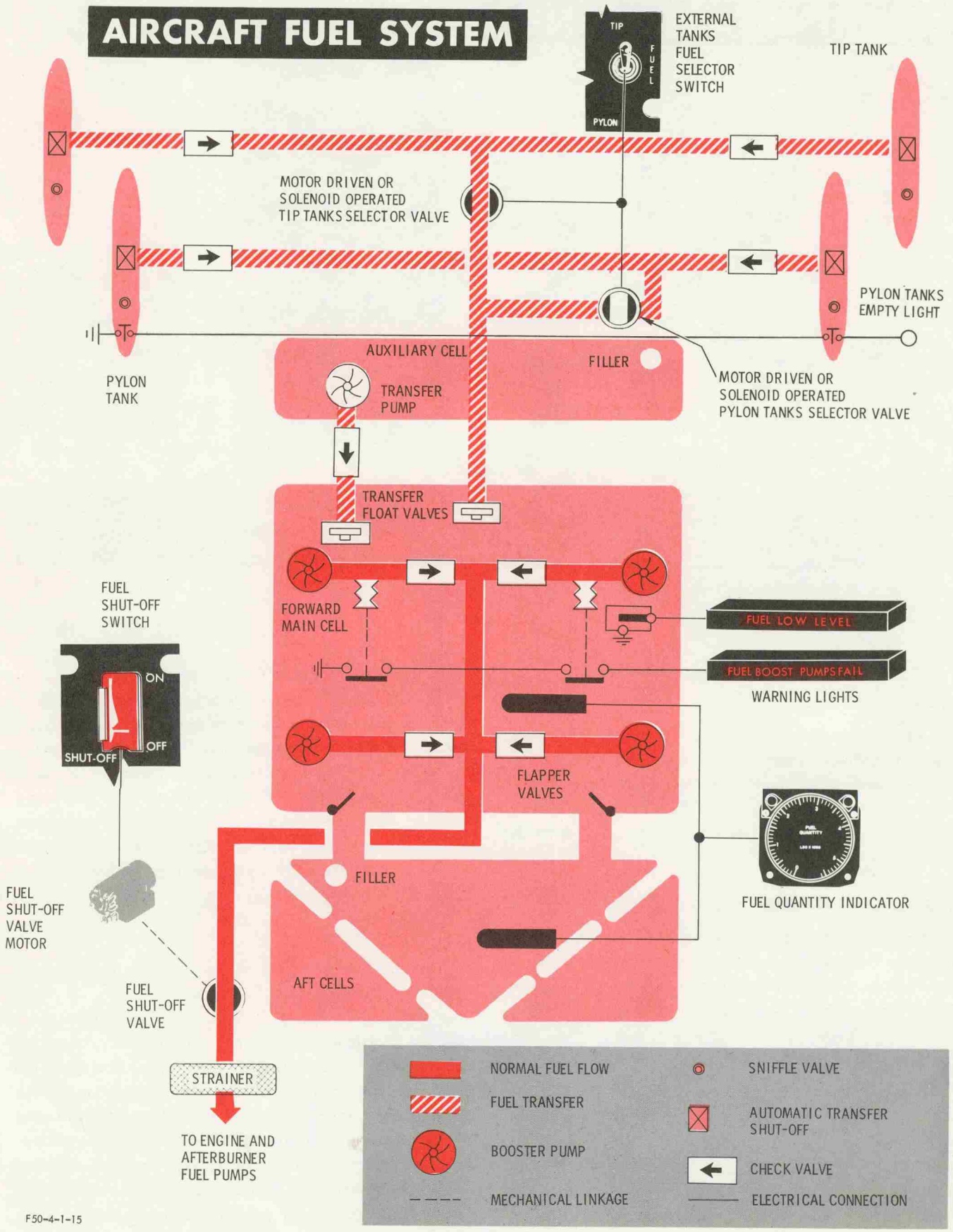

AIRCRAFT FUEL SYSTEM
TIP
FUEL
PYLON
EXTERNAL TANKS FUEL SELECTOR SWITCH
TIP TANK
MOTOR DRIVEN OR SOLENOID OPERATED TIP TANKS SELECTOR VALVE
PYLON TANKS EMPTY LIGHT
AUXILIARY CELL
FILLER
PYLON TANK
TRANSFER PUMP
MOTOR DRIVEN OR SOLENOID OPERATED PYLON TANKS SELECTOR VALVE
TRANSFER FLOAT VALVES
FUEL SHUT-OFF SWITCH
FORWARD MAIN CELL
FUEL LOW LEVEL
FUEL BOOST PUMPS FAIL
WARNING LIGHTS
ON
OFF
SHUT-OFF
FLAPPER VALVES
FILLER
FUEL QUANTITY INDICATOR
FUEL SHUT-OFF VALVE MOTOR
AFT CELLS
FUEL SHUT-OFF VALVE
STRAINER
TO ENGINE AND AFTERBURNER FUEL PUMPS
NORMAL FUEL FLOW
FUEL TRANSFER
BOOSTER PUMP
MECHANICAL LINKAGE
SNIFFLE VALVE
AUTOMATIC TRANSFER SHUT-OFF
CHECK VALVE
ELECTRICAL CONNECTION
F50-4-1-15

Figure 1-15

AFT CENTER CELL.

The aft center cell is located between the engine inlet ducts. Two vent valves, a quantity transmitter, and two drain valves are in the cell. The drain valves are at the forward bottom end of the cell and are accessible from the forward end of the wheel well.

AFT RIGHT AND LEFT CELLS.

The aft right and aft left cells fit around the engine air inlet ducts outboard of the aft center cell. Each cell is connected by tubes, at the bottom and top, to the aft center cell. Fuel flows to the aft center cell through the bottom tube. The top tube serves as a vent connection. These cells and the aft center cell will hereafter be considered and referred to as the aft main tank.

AUXILIARY FUEL CELL.

A 145 gallon auxiliary fuel cell is located forward of the main fuel cell. A separate filler is provided. Fuel is automatically transferred by means of a transfer pump and a transfer float valve to the main fuel cell. The auxiliary fuel cell also contains a vent float valve and two drain valves. The drain valves are accessible from beneath the airplane. The transfer pump is powered by the No. 2 a.c. bus when the airplane's electrical system is energized and shuts off automatically when the tank is empty.

FUEL BOOST PUMPS.

Four boost pumps, operated by 3-phase a.c. motors, are installed, one in each corner, in the forward main fuel cell. Power for operation of the No. 1 boost pump is supplied from the emergency a.c. bus. The No. 2 boost pump is on the No. 2 a.c. bus. No. 3 and No. 4 boost pump power is supplied from the No. 1 a.c. bus. The pumps are manifolded together through check valves into the main fuel supply line to the engine. The main fuel supply line is routed aft from the forward cell to the shut-off valve at the firewall. A line connects the shut-off valve to the fuel strainer aft of the firewall. A drain valve and an overboard drain line is plumbed from the strainer sump. The fuel from the strainer is routed through a flexible hose to the engine and afterburner fuel pumps. The boost pump circuits are energized any time the airplane electrical system is energized and the circuit breakers are in. The pumps supply fuel at 35 psi. The boost pump failure warning system is connected to the two forward pumps. No fuel pressure indicating system is provided.

FUEL TANK PRESSURIZATION AND VENT SYSTEM.

The vent system vents the internal fuel cells, provides self-pressurization of the cells in a climb to prevent loss of fuel, and provides controlled pressurization of the cells in a dive.

Vent Float Valves.

Five vent float valves are installed in the system. Two are in the forward main cell, two in the aft center cell and one in the auxiliary fuel cell. The valves are float-actuated and close the fuel tank vent in all attitudes of flight when the fuel reaches a predetermined level at the valve. This prevents fuel from flowing out the vent.

Altitude Vent Valve (Unmodified Aircraft Only).

An altitude-closing vent valve is provided. Its purpose is to provide self-pressurization of the cells in a climb, reduce loss of fuel at high altitudes, provide unpressurized tanks on the ground, and permit some measure of cell pressurization during a dive. On modified aircraft the bellows within the valve is de-activated and the valve assumes the function of a vent sniffle valve.

Auxiliary Relief Valve.

An auxiliary relief valve which works in conjunction with the altitude vent valve provides a pressure and vacuum relief if the altitude valve fails.

Pressure Regulator.

A dual air pressure regulator is installed to maintain an air pressure differential between the fuel cell cavity and the inside of the fuel cell. Either side of the dual regulator is capable, by itself, of fulfilling the entire requirement. The regulator senses both pressure within the cells and pressure in the cell cavity and closes when the pressure within the cells begins to exceed the cell cavity pressure by a preset value.

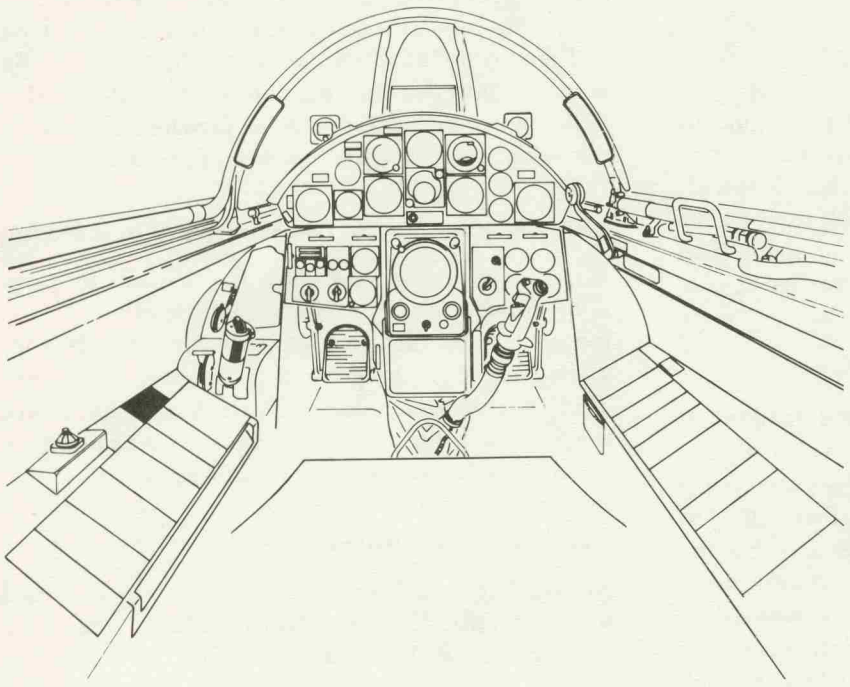
FUEL PRESSURE AIR SHUT-OFF TEST SWITCH (Unmodified Aircraft).

A press-to-test switch (18, Figure 1-7), located on the left console, is provided to ground test the fuel pressure air system for correct operation. The system receives power from the d.c. emergency bus. Pressing the test switch opens the air shut-off valve which normally is closed on the ground. This allows internal fuel cell pressurization. Pressurized air will then flow out of the vents located at each wing tip and may be verified by ground personnel. On modified aircraft this switch is de-activated.

EXTERNAL TANKS.

165 gallon tip tanks are provided. Provisions are included for carrying pylons and 200 gallon pylon tanks under

FUEL CONTROL PANEL



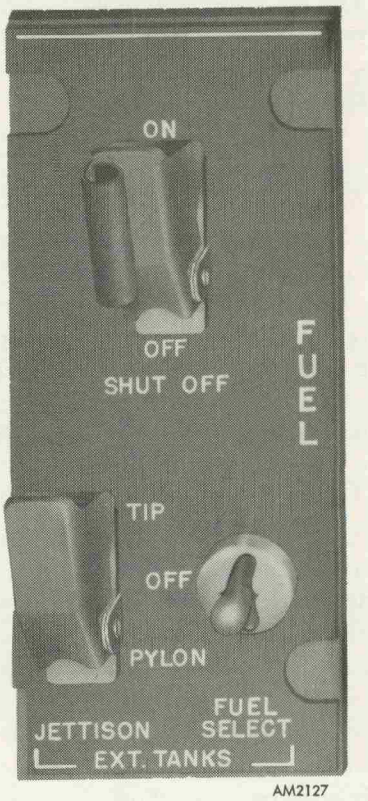


Figure 1-16

each wing. An auto drop system as well as means for electrically jettisoning the tip, pylons and pylon tanks are provided.

Automatic Transfer Shut-Off.

Fuel in the tip tanks moves from an aft compartment to a forward compartment through a tube. A door at the forward end of the tube permits flow of fuel only in the forward direction. The fuel outlet to the transfer line and the fuselage tank is in the forward compartment. Each tank contains a low level float switch which shuts off the transfer valve and pressurizing air when the tanks become empty. The shut-off prevents engine bleed air from entering the fuel transfer lines and internal cells when the tip tanks are empty.

External Fuel Transfer System.

The fuel transfer system provides a means of transferring fuel from the tip and pylon tanks to the forward main fuel cell. The fuel is transferred by using air pressure supplied by the engine compressor. Air is cooled by the primary heat exchanger and is controlled by a pressure regulator set to maintain a constant psi. The air passes through motor-driven air shut-off valves. The shut-off valves are controlled by low level float switches and are open when the tanks are installed and have fuel in them. Plumbing is arranged so that it is symmetrical about the airplane centerline to provide equal flow from both tanks. Sniffle valves are located in the tip tanks and in the pylons which support the pylon tanks. They act as relief valves in case of a malfunctioning regulator. A failure in the regulator will cause it to fail to the open position. The unit is vented to the atmosphere by a line on the left side of the fuselage.

Air Shut-Off Valves. Two air shut-off 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 motor operated and are controlled by low level float switches in the external tanks.

FUEL QUANTITY DATA

DATA BASIS:
GROUND TEST, STANDARD DAY CONDITIONS WITH CONVERSION FACTOR 6.5 LBS/GAL

REMARKS:
LEVEL FLIGHT ATTITUDE — TOP OF FUSELAGE 3° NOSE UP, STATIC ATTITUDE — TOP OF FUSELAGE 0°.

	USABLE FUEL IN LEVEL FLIGHT ATTITUDE		FULLY SERVICED IN STATIC ATTITUDE	
	U. S. GALS	LBS.	U. S. GALS	LBS.
INTERNAL FUEL	763	4959.5	772	5018
AUXILIARY TANK (ESTIMATED)	143	929.5	150	975
TIP TANKS (EACH)	164	1066	165	1073
PYLON TANKS (EACH)	192	1248	199	1294

TOTAL USABLE FUEL IN LEVEL FLIGHT ATTITUDE

INTERNAL AND AUXILIARY FUEL . . . . . 906 GALS. — 5889 LBS.
WITH TIP TANKS . . . . . 1234 GALS. — 7721 LBS.
WITH PYLON TANKS . . . . . 1290 GALS. — 8385 LBS.
WITH PYLON AND TIP TANKS . . . . . 1618 GALS. — 10,517 LBS.

F50-5-1-17

Figure 1-17

Fuel Transfer Float Valve. A fuel transfer float valve is installed in the forward main cell. The valve controls fuel transfer from the external tanks. The level of fuel in the forward main cell is maintained by movement of a float inside the valve that opens and closes the valve by its movement until external fuel is exhausted.

External Stores Auto Drop System.

A tip and pylon auto drop system is provided. If either a pylon or tip tank should become disengaged accidentally, the system will automatically jettison both the disengaged tank and the corresponding tank on the opposite wing, provided the airplane electrical system is in operation. The system is powered from the d.c. emergency bus. To prevent the auto drop systems from operating on the ground when one tank is intentionally removed, safety pins are provided for the tip and pylon tanks to disarm the system. The pins are inserted under each wing for the tip tanks and in the pylons for the pylon tanks.

Note

The auto-drop system is in operation when missiles are on the wing tips. Only in event of a missile launcher becoming disengaged will the auto-drop system function as described. Missile firing will not activate the system.

FUEL SHUT-OFF SWITCH.

The fuel shut-off switch (figure 1-16) is located on the left console and is guarded to the ON position. The switch (battery bus powered) may be used to actuate electrically the fuel shut-off valve, located just aft of the main fuel cell. The valve is motor-driven. The motor is connected through the fuel shut-off switch and through a circuit breaker in the electronic compartment to the battery bus. The valve provides a means of shutting off fuel to the engine in case of fire, crash landing or for ground maintenance.

EXTERNAL TANK FUEL SELECTOR SWITCH.

Motor-operated external fuel selector valves permit fuel to be transferred from both tip tanks or from both pylon tanks. The selector valves are connected to the d.c. monitored bus through the air shut-off valve relays, the fuel selector switch (figure 1-16) on the fuel control panel and a circuit breaker on the electronic compartment panel.

Note

- Should failure of the d.c. monitored bus occur, the fuel selector valves will fail to the position they are in at the time of failure.
- AF Serial 56-878 and subsequent, and modified aircraft, incorporate solenoid operated external fuel selector valves which fail to the open position upon loss of power from the d.c. monitored bus.

The two-position fuel selector switch, labeled PYLON, and TIP, controls valve operation as long as external tanks are installed. A transfer float valve in the forward main cell allows fuel transfer from the selected external tanks when the fuel level in the forward main cell is lowered.

Note

If both tip and pylon tanks are installed, after the selected tanks become empty, fuel will automatically transfer from the remaining tanks. If only one set of tanks is installed or if one set of tanks has been jettisoned, fuel will automatically transfer from the installed tanks, regardless of selector switch position.

EXTERNAL STORES RELEASE SELECTOR SWITCH AND EXTERNAL STORES RELEASE BUTTON.

External tanks are dropped by pressing the external stores release button (2, figure 1-22) on the control stick grip. The external stores release selector switch (figure 1-16) determines which external stores will be dropped when the release button is pressed. The selector switch is labeled PYLON, OFF, TIP. The system is powered from the battery bus. Pressing the release button closes whichever circuit is selected by the selector switch causing the ejectors to jettison the stores. In order to jettison the pylons after the pylon stores are jettisoned, the external stores release button must be released and pressed again.

EXTERNAL STORES JETTISON BUTTON.

A button (figure 1-10) located on the left forward panel is provided to jettison all external stores in an emergency. The circuit is connected directly to the battery bus through a circuit breaker in the electronic compartment and is "hot" any time the circuit breaker is pushed in and a battery installed. By pushing the button both tip and pylon stores may be jettisoned. However, the external stores release button is the only means of jettisoning the pylons.

PYLON TANK EMPTY LIGHT.

AF Serial 56-772 and subsequent, and earlier modified aircraft, incorporate an amber light (9, figure 1-8) on the right console which illuminates when the pylon tanks are empty. The light is powered from the d.c. emergency bus.

FUEL QUANTITY INDICATOR.

The fuel quantity indicating system indicates in pounds the fuel quantity remaining in the main internal fuel cells.

Note

The fuel quantity indicator does not indicate external or auxiliary fuel remaining.

The system consists of an indicator (17, figure 1-6) on the right side of the main instrument panel and two fuel cell transmitters; one in the forward cell and one in the aft cell. The transmitter in the forward cell is combined with a fuel density compensating unit. The system receives power from the No. 2 a.c. bus and the d.c. emergency bus through circuit breakers on the left circuit breaker panel.

Fuel Quantity System Test Switch.

A test switch (5, figure 1-8) is located on the right forward panel. When the airplane's electrical system is energized, placing the switch in the FUEL QUANTITY (up) position will ground the system power supply,

which will cause the fuel quantity gage indicating needle to go toward zero if the system is functioning properly. This will not activate the low level warning indication as the systems are independent. (The switch is also used to check the warning lights circuits in the WARNING LIGHTS TEST (down) position.

FUEL BOOST PUMP FAILURE WARNING LIGHT.

Two pressure switches are installed in the forward fuel cell. One switch is connected to the pressure feed line from the left forward pump (No. 1) and one into the line from the right forward pump (No. 2). The pressure switches are connected in series to the 28-volt d.c. emergency bus through the warning light circuit breaker on the electronic compartment panel. The switches are normally closed; they open when the forward pumps are in normal operation. The switches close if the pumps fail and the "BOOST PUMP FAIL" warning light will illuminate on the warning light panel (figure 1-11). The master caution light will go on also. Pressure from both forward pumps must decrease to approximately 10 psi to energize the warning circuits. The warning system is connected to the two front pumps because these are the critical pumps in the approach or nose down attitude of the airplane.

Note

At AF Serial 56-878 and subsequent the BOOST PUMP FAIL warning light is removed. The operation of each boost pump is checked by ground maintenance.

FUEL LOW LEVEL WARNING LIGHT.

A low-level warning system is installed to indicate to the pilot that the fuel level in the forward main cell has decreased to a critical level. The system includes a float-actuated switch installed in the forward main cell, and a light on the warning panel. When the fuel level falls to 195 gallons (approximately 1270 pounds), the switch closes the circuit and energizes the master caution light and the "FUEL LOW LEVEL" warning light becomes illuminated on the warning light panel (figure 1-11). The system receives electrical power from the d.c. emergency bus.

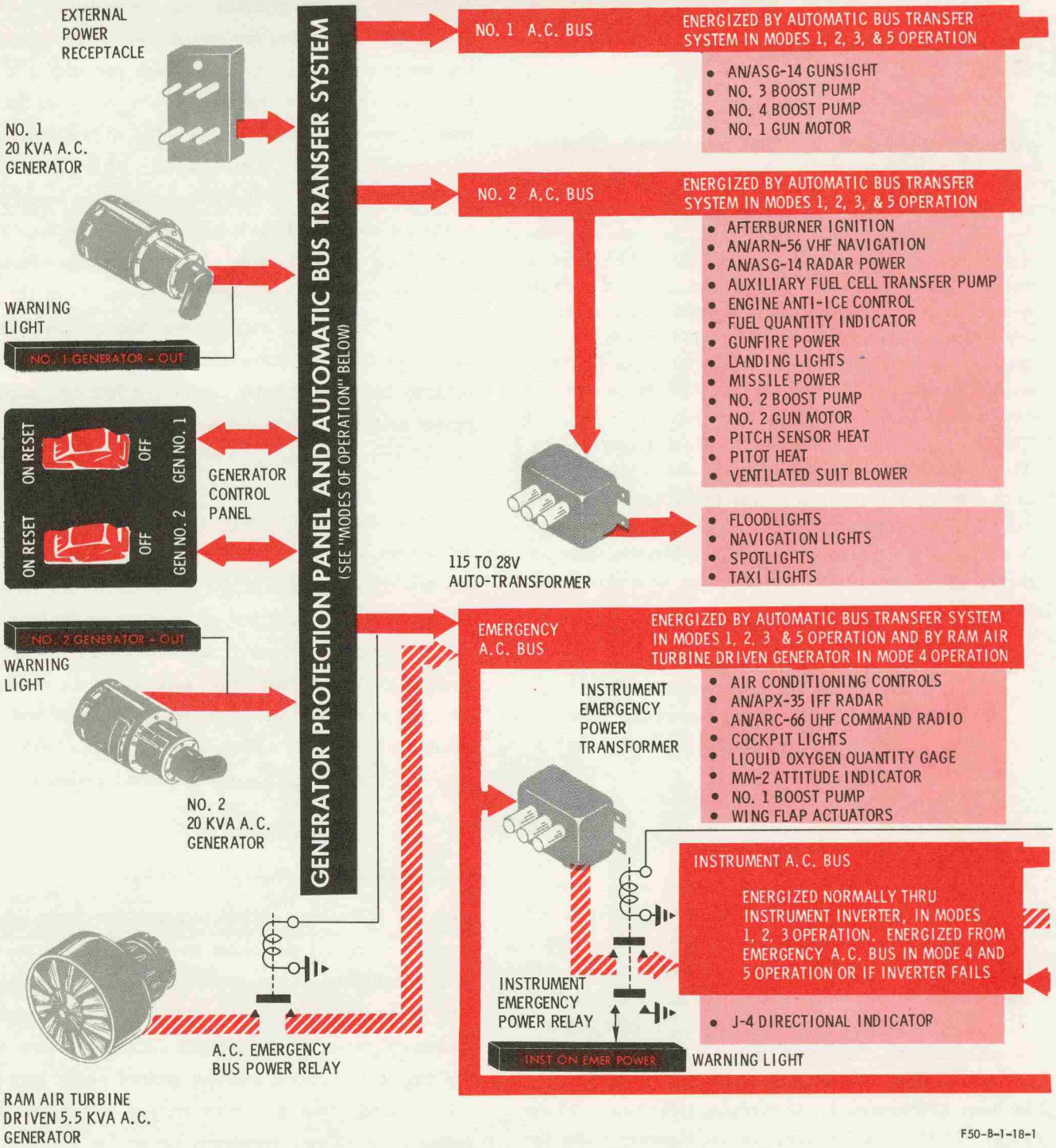
ELECTRICAL POWER SUPPLY SYSTEMS.

The aircraft electrical systems obtain power primarily from two engine driven alternating current generators. This power is utilized by the main a.c. electrical system, a d.c. electrical system and an instrument inverter system. Emergency power is supplied by a ram air turbine-driven a.c. generator and a small, 4 amp/hr. battery. For ground operation, the external power receptacle on the lower right side of the fuselage provides a means for connecting an external a.c. power source to the aircraft. Electrical power is distributed to the individual systems through an a.c. bus for each of the generators, an emergency a.c. bus, and a d.c. monitored bus. Power to the d.c. monitored bus is made available by the conversion of a.c. power in a 100 amp. transformer rectifier. The instrument inverter receives power from the d.c. monitored bus during normal operation and supplies a.c. power to the instrument a.c. bus and the three axes control damper. The d.c. monitored bus also supplies power to a d.c. essential bus and a d.c. emergency bus during normal operation. The battery bus receives power from the 20 ampere transformer rectifier through a blocking rectifier. During emergency operation the instrument a.c. bus receives power from the emergency a.c. bus through an emergency instrument transformer, while the d.c. essential bus, d.c. emergency bus and battery bus receive their power from the emergency a.c. bus through a 20 amp. transformer rectifier. In the event that even the emergency power supply is lost, the 4 amp./hr. battery will furnish sufficient power to operate those items on the battery bus for a limited period of time.

A.C. ELECTRICAL POWER SUPPLY.

Two 20 KVA engine-driven generators serve as the primary source of a.c. electrical power for the aircraft. They are located on the accessory section of the engine. The generators supply 200/115 volt 3-phase variable frequency power to the aircraft electrical system when the engine is running and the ground power supply is disconnected. This generator output is controlled by means of a voltage regulator, protection panel, relays for automatic transfer of the two a.c. busses from one generator to the other, and a control switch for each generator. Normally, the No. 1 generator output goes to the No. 1 a.c. bus and the No. 2 generator output to the No. 2 a.c. bus. In addition to this, the No. 2 generator normally provides power for the emergency

ELECTRICAL POWER DISTRIBUTION

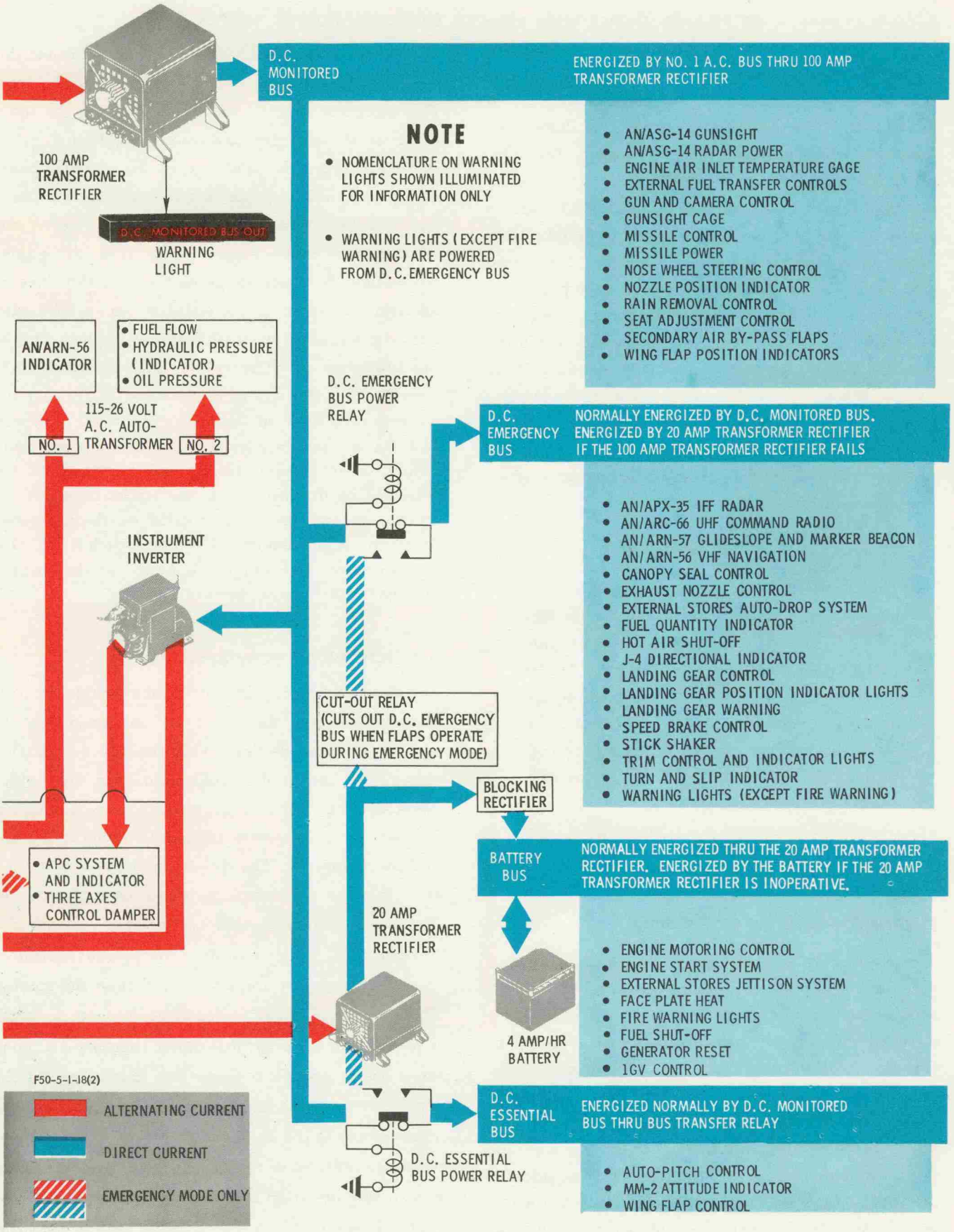


F50-B-1-18-1

MODES OF OPERATION

MODE 1 - NORMAL OPERATION	MODE 2 - NO. 1 GENERATOR OUT	MODE 3 - NO. 2 GENERATOR OUT	MODE 4 - EMERGENCY OPERATION	MODE 5 - GROUND POWER OPERATION
• NO. 1 GENERATOR ON NO. 1 A.C. BUS • NO. 2 GENERATOR ON NO. 2 A.C. BUS AND EMERGENCY BUS	• NO. 2 GENERATOR ON ALL BUSSES	• NO. 1 GENERATOR ON ALL BUSSES	• NO. 1 & 2 BUSSES OUT • NO. 1 & 2 GENERATORS OUT • EMERGENCY GENERATOR ON EMERGENCY A.C. BUS	• ALL BUSSES ENERGIZED BY EXTERNAL POWER SOURCE

Figure 1-18


D.C. MONITORED BUS
ENERGIZED BY NO. 1 A.C. BUS THRU 100 AMP TRANSFORMER RECTIFIER
100 AMP TRANSFORMER RECTIFIER
D.C. MONITORED BUS OUT
WARNING LIGHT
NOTE
• NOMENCLATURE ON WARNING LIGHTS SHOWN ILLUMINATED FOR INFORMATION ONLY
• WARNING LIGHTS (EXCEPT FIRE WARNING) ARE POWERED FROM D.C. EMERGENCY BUS
• AN/ASG-14 GUNSIGHT
• AN/ASG-14 RADAR POWER
• ENGINE AIR INLET TEMPERATURE GAGE
• EXTERNAL FUEL TRANSFER CONTROLS
• GUN AND CAMERA CONTROL
• GUNSIGHT CAGE
• MISSILE CONTROL
• MISSILE POWER
• NOSE WHEEL STEERING CONTROL
• NOZZLE POSITION INDICATOR
• RAIN REMOVAL CONTROL
• SEAT ADJUSTMENT CONTROL
• SECONDARY AIR BY-PASS FLAPS
• WING FLAP POSITION INDICATORS
AN/ARN-56 INDICATOR
• FUEL FLOW
• HYDRAULIC PRESSURE (INDICATOR)
• OIL PRESSURE
115-26 VOLT A.C. AUTO-TRANSFORMER
NO. 1
NO. 2
D.C. EMERGENCY BUS POWER RELAY
D.C. EMERGENCY BUS
NORMALLY ENERGIZED BY D.C. MONITORED BUS. ENERGIZED BY 20 AMP TRANSFORMER RECTIFIER IF THE 100 AMP TRANSFORMER RECTIFIER FAILS
• AN/APX-35 IFF RADAR
• AN/ARC-66 UHF COMMAND RADIO
• AN/ARN-57 GLIDESLOPE AND MARKER BEACON
• AN/ARN-56 VHF NAVIGATION
• CANOPY SEAL CONTROL
• EXHAUST NOZZLE CONTROL
• EXTERNAL STORES AUTO-DROP SYSTEM
• FUEL QUANTITY INDICATOR
• HOT AIR SHUT-OFF
• J-4 DIRECTIONAL INDICATOR
• LANDING GEAR CONTROL
• LANDING GEAR POSITION INDICATOR LIGHTS
• LANDING GEAR WARNING
• SPEED BRAKE CONTROL
• STICK SHAKER
• TRIM CONTROL AND INDICATOR LIGHTS
• TURN AND SLIP INDICATOR
• WARNING LIGHTS (EXCEPT FIRE WARNING)
INSTRUMENT INVERTER
CUT-OUT RELAY (CUTS OUT D.C. EMERGENCY BUS WHEN FLAPS OPERATE DURING EMERGENCY MODE)
BLOCKING RECTIFIER
BATTERY BUS
NORMALLY ENERGIZED THRU THE 20 AMP TRANSFORMER RECTIFIER. ENERGIZED BY THE BATTERY IF THE 20 AMP TRANSFORMER RECTIFIER IS INOPERATIVE.
• APC SYSTEM AND INDICATOR
• THREE AXES CONTROL DAMPER
20 AMP TRANSFORMER RECTIFIER
4 AMP/HR BATTERY
• ENGINE MOTORING CONTROL
• ENGINE START SYSTEM
• EXTERNAL STORES JETTISON SYSTEM
• FACE PLATE HEAT
• FIRE WARNING LIGHTS
• FUEL SHUT-OFF
• GENERATOR RESET
• IGV CONTROL
F50-5-1-18(2)
ALTERNATING CURRENT
DIRECT CURRENT
EMERGENCY MODE ONLY
D.C. ESSENTIAL BUS
ENERGIZED NORMALLY BY D.C. MONITORED BUS THRU BUS TRANSFER RELAY
D.C. ESSENTIAL BUS POWER RELAY
• AUTO-PITCH CONTROL
• MM-2 ATTITUDE INDICATOR
• WING FLAP CONTROL

a.c. bus. In cases where an under voltage or an over-voltage condition exists for either generator, that generator is automatically removed from its bus; the bus is automatically transferred to the other generator, and the warning light panel in the cockpit is illuminated indicating which generator is not operating. The automatic bus transfer system provides for five possible modes of operation as indicated on figure 1-18. Various electrically operated units (see figure 1-18) take their power directly from the No. 1 and No. 2 a.c. bus. The No. 1 a.c. bus also directs generator output to the 100 amp. transformer rectifier where the a.c. is changed to 28-volt d.c. before being sent to the d.c. monitored bus. The No. 2 a.c. generator normally furnishes power for the emergency a.c. bus. If the instrument inverter should fail for any reason, the emergency a.c. bus will assume the load automatically by feeding power to the instrument a.c. bus through the instrument emergency power transformer.

Note

The electrical supply system is equipped with under frequency relays which will cut the two 20 KVA generators off the busses when engine rpm drops below approximately 65%. Under this condition, all electrically operated equipment except one aircraft boost pump and the battery bus will be inoperative. The boost pump will continue to operate at lower engine rpm (down to approximately 40%). This feature insures sufficient boost pump pressure for high altitude air starts.

EMERGENCY A.C. POWER SUPPLY.

The airplane is equipped with a normally concealed ram air turbine which supplies emergency electrical and hydraulic power as required when extended. Once extended the ram air turbine can not be retracted in the air. If both the No. 1 and No. 2 generators fail, the ram air turbine-driven a.c. generator (5.5 KVA) will supply power for the emergency bus. It will also furnish power for the instrument a.c. bus through the instrument emergency power transformer. In addition to this, the emergency a.c. bus will direct power to the d.c. essential bus, d.c. emergency bus and battery bus through a 20 ampere transformer rectifier.

EXTERNAL POWER SUPPLY.

The aircraft is equipped with a receptacle for connecting an external a.c. power source to the electrical system. This receptacle (figure 1-30) is located on the lower right side of the fuselage and is accessible through a door above the hydraulic panel. When the external power supply is applied to the aircraft, the generators are automatically disconnected from their respective busses, and all three a.c. busses receive power from the ground power unit. In order to prevent unnecessary inverter operation, a system of protective relays operates automatically whenever external power is applied. One of the functions of these relays is to disconnect the inverter from the d.c. monitored bus. Instrument bus power is then supplied automatically through the emergency a.c. bus and the emergency instrument power transformer. It is possible for ground personnel to test operation of the instrument inverter by means of the inverter ground test switch located on the electronics compartment junction box. When the switch is held ON, the instrument bus will be connected to the inverter through the instrument power relay.

D.C. ELECTRICAL POWER SUPPLY.

The direct current requirements of the aircraft normally are supplied from the No. 1 a.c. bus through a 100 amp. transformer rectifier. This changes the 200/115 volt a.c. to 28-volt d.c. which is directed to the d.c. monitored bus. Power is drawn directly from this bus to operate various units (figure 1-18) including the instrument inverter. The d.c. essential bus and the d.c. emergency bus are also connected to the d.c. monitored bus during normal operation. The d.c. essential and d.c. emergency busses furnish power to units which are considered necessary for safe operation of the aircraft. Therefore, an alternate source of power to these busses is provided in the event that power from the d.c. monitored bus is disrupted. Under this condition, the d.c. emergency and d.c. essential busses will be connected automatically to the 20 ampere transformer rectifier unit which is connected to the a.c. emergency bus. When the ram-air turbine-driven a.c. generator is operative (Emergency Mode), it is important that the load on the emergency a.c. bus be minimized when using the aircraft

leading and trailing edge flaps since they are powered directly from the emergency a.c. bus. To reduce loads and insure maximum flap effectiveness, the d.c. emergency bus is automatically disconnected from the 20 ampere transformer rectifier as long as the flaps are in operation, and those units which are powered from this bus, including UHF command radio, will be inoperative during the period of flap operation.

EMERGENCY D.C. POWER SUPPLY.

If all three generators fail, the battery will furnish a supply of direct current to the battery bus. The battery is the only independent source of direct current in the aircraft electrical system. Normally, the battery and battery bus are paralleled with the 20 ampere transformer rectifier, and the battery is thereby maintained in a fully charged condition. In emergency operation, battery output is prevented from discharging to the 20 ampere transformer rectifier by a blocking rectifier in order to conserve its limited power supply for those units connected directly to the battery bus. There is no battery control switch in the cockpit, and operation of the battery system is entirely automatic.

Note

The battery is a serviceable unit and is kept in a charged condition from the 20 ampere transformer rectifier.

INSTRUMENT POWER SUPPLY.

Alternating current necessary for the operation of various flight instruments is furnished normally by an instrument inverter. The inverter is located in the electronics compartment of the aircraft. The inverter functions to convert 28-volt d.c., from the d.c. monitored bus, to 115-volt, 400 cycle, a.c. power for the instrument a.c. bus. Failure of the instrument inverter will actuate the instrument emergency power relay which automatically connects the instrument a.c. bus to the emergency a.c. bus through the instrument emergency power transformer so that there will be no interruption of instrument operation. The inverter is disconnected from the d.c. monitored bus when an external power source is connected to the aircraft electrical system. However, it may be tested for proper operation by ground personnel by means of the inverter ground test switch in the electronics compartment. During external power operation, alternating current for the instruments is supplied by the ground power unit through the emergency a.c. bus and instrument emergency power transformer.

CIRCUIT BREAKERS.

The circuit breaker panels (16, figure 1-7) and (18, figure 1-8) on the left and right consoles in the cockpit contain push-to-reset, pull-out type breakers for certain a.c. and d.c. circuits. All of 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 flight, are located in the electronics 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 effects and results. Pulling circuit breakers can eliminate from the system some related warning system, interlocking circuit, or canceling signal, which could result in an undesirable reaction.

GENERATOR SWITCHES.

A generator switch is provided for each of the 20 KVA generator systems. These switches (3, 4, figure 1-8) are identical and are located on the right forward panel. The switches are powered from the battery bus. Each switch has three positions: ON RESET (up), OFF (down), and a center NEUTRAL position which is the normal position of the switch when covered by the guard. The switches are spring loaded to return to the NEUTRAL position. Placing the switch up to the ON-RESET position will return the generator to normal operation if it has been removed from the line for any reason other than complete generator failure. When placed down to the OFF position, the switch will energize the generator control relay which will remove the generator from its associated bus.

RAM AIR TURBINE EXTENSION HANDLE.

Emergency a.c. power is made available by extending the ram air turbine into the aircraft slip stream. This can be accomplished by pulling the yellow ram air turbine extension handle (39, figure 1-6) located on the lower right side of the instrument panel. The handle requires a firm pull of about 4 inches to the stop to extend the ram air turbine.

Note

There is no means of retracting the ram air turbine in the air.

GENERATOR OUT WARNING LIGHTS.

Two generator out warning lights (figure 1-11), one for each of the 20 KVA generators, are located on the

HYDRAULIC SYSTEM

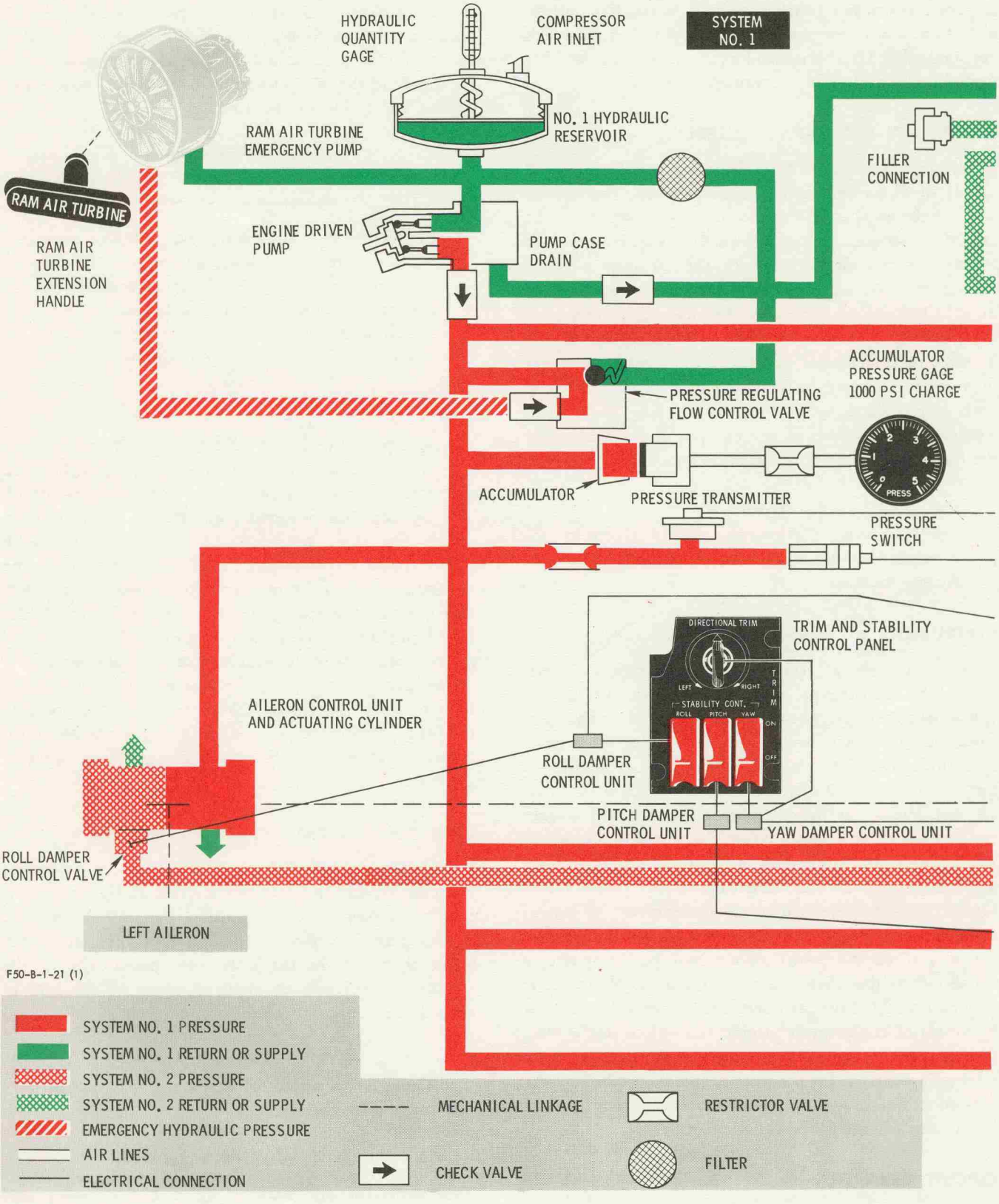
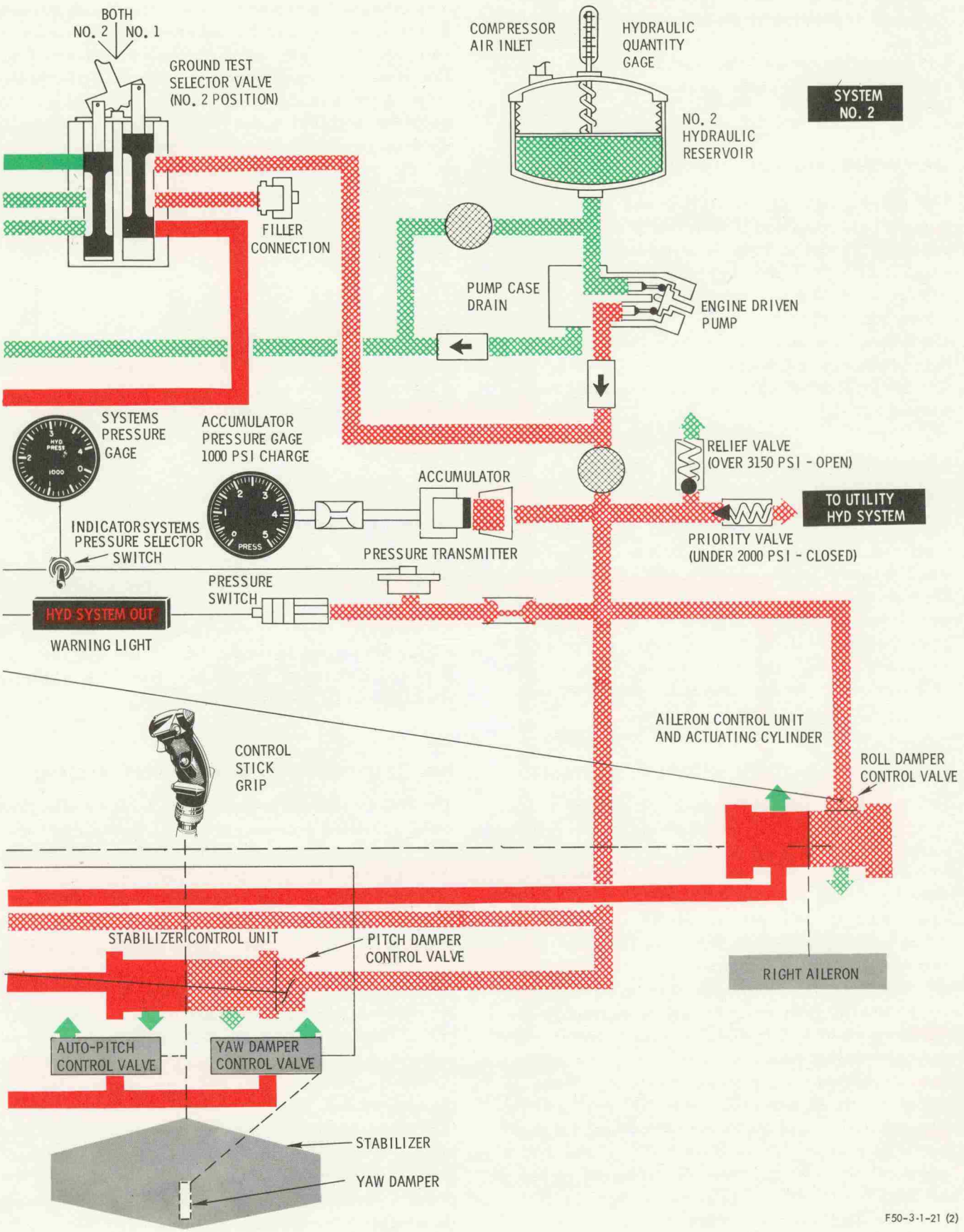


Figure 1-19


BOTH
NO. 2
NO. 1
GROUND TEST SELECTOR VALVE (NO. 2 POSITION)
FILLER CONNECTION
COMPRESSOR AIR INLET
HYDRAULIC QUANTITY GAGE
NO. 2 HYDRAULIC RESERVOIR
SYSTEM NO. 2
PUMP CASE DRAIN
ENGINE DRIVEN PUMP
SYSTEMS PRESSURE GAGE
ACCUMULATOR PRESSURE GAGE 1000 PSI CHARGE
ACCUMULATOR
RELIEF VALVE (OVER 3150 PSI - OPEN)
TO UTILITY HYD SYSTEM
INDICATOR SYSTEMS PRESSURE SELECTOR SWITCH
PRESSURE TRANSMITTER
PRIORITY VALVE (UNDER 2000 PSI - CLOSED)
PRESSURE SWITCH
HYD SYSTEM OUT
WARNING LIGHT
AILERON CONTROL UNIT AND ACTUATING CYLINDER
ROLL DAMPER CONTROL VALVE
CONTROL STICK GRIP
STABILIZER CONTROL UNIT
PITCH DAMPER CONTROL VALVE
RIGHT AILERON
AUTO-PITCH CONTROL VALVE
YAW DAMPER CONTROL VALVE
STABILIZER
YAW DAMPER

F50-3-1-21 (2)

warning light panel. The lights are placarded NO. 1 GENERATOR OUT and NO. 2 GENERATOR OUT. Each light is powered by the 28-volt d.c. emergency bus and will illuminate whenever its respective generator is not generating voltage. The master caution light will also illuminate when either generator-out warning light illuminates.

D.C. MONITORED BUS OUT WARNING LIGHT.

This warning light (figure 1-11) is also located on the warning light panel and is energized by the 28-volt d.c. emergency bus. The light is placarded D.C. MONITORED BUS OUT. The light will be illuminated any time power to the d.c. monitored bus is discontinued during normal operation. The master caution light will also illuminate when the d.c. monitored bus-out warning light illuminates. (Refer to figure 1-18 for units which will be inoperative when the d.c. monitored bus-out warning light is on.)

INSTRUMENTS ON EMERGENCY POWER WARNING LIGHT.

This light (figure 1-11) is located on the warning light panel and is energized by the 28-volt d.c. emergency bus. The light is placarded INST. ON EMER. POWER. The light will illuminate any time power from the instrument inverter is discontinued and the instrument emergency power transformer is supplying power to the instrument a.c. bus. The master caution light will also illuminate when the instruments on emergency power warning light is illuminated.

HYDRAULIC POWER SUPPLY SYSTEMS.

Two completely independent hydraulic systems and an emergency system provide power to the various hydraulically actuated units in the aircraft. See figures 1-19 and 1-20. The No. 1 and No. 2 systems are in simultaneous operation during all normal operating conditions and supply fluid at 3000 psi pressure to their respective hydraulically actuated units. Both No. 1 and No. 2 systems are provided with separate reservoirs which differ only in size and location, the No. 2 reservoir having the larger capacity. Both reservoirs are pressurized to prevent pump cavitation. In addition, 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 or No. 2 system, the remaining system will maintain fluid pressure for flight control at a reduced rate. If both No. 1 and No. 2 system fail, the emergency ram air turbine-driven pump will furnish enough fluid pressure through the No. 1 system for flight control at a reduced rate.

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' function is to store a supply of high pressure fluid and also to 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 (figure 1-30) 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 accumulator only when hydraulic pressure is zero.

Hydraulic Panel.

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

Ground Test Selector Valve.

The manually controlled ground test selector valve on the hydraulic panel is the only link between No. 1 and No. 2 system. A three position lever extends from the top of the valve. Mechanical linkage from this lever to a fixed bracket inside the fuselage assures that the lever is placed and locked in the No. 2 position whenever the engine access door is closed.

NO. 1 HYDRAULIC POWER SUPPLY SYSTEM.

The No. 1 hydraulic system (figure 1-19) supplies fluid under regulated pressure exclusively to the flight controls. Power is supplied to the stabilizer aft cylinder, the five inboard cylinders for each aileron, the auto-pitch actuator, and the yaw damper (including directional trim). 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 by the reservoir which is pressurized to prevent pump cavitation. Fluid from the pump is supplied, under 3000 psi pressure, directly to the flight control components. The pressure-regulating flow-control valve is connected into the pressure line. This valve contains a relief valve which relieves system pressure to the return line, in case of a pressure surge. When the emergency pump is in use, the pressure-regulating flow-control valve also regulates the pressure, maintaining near constant flow of fluid from the emergency pump to the No. 1 system.

UTILITY HYDRAULIC SYSTEM

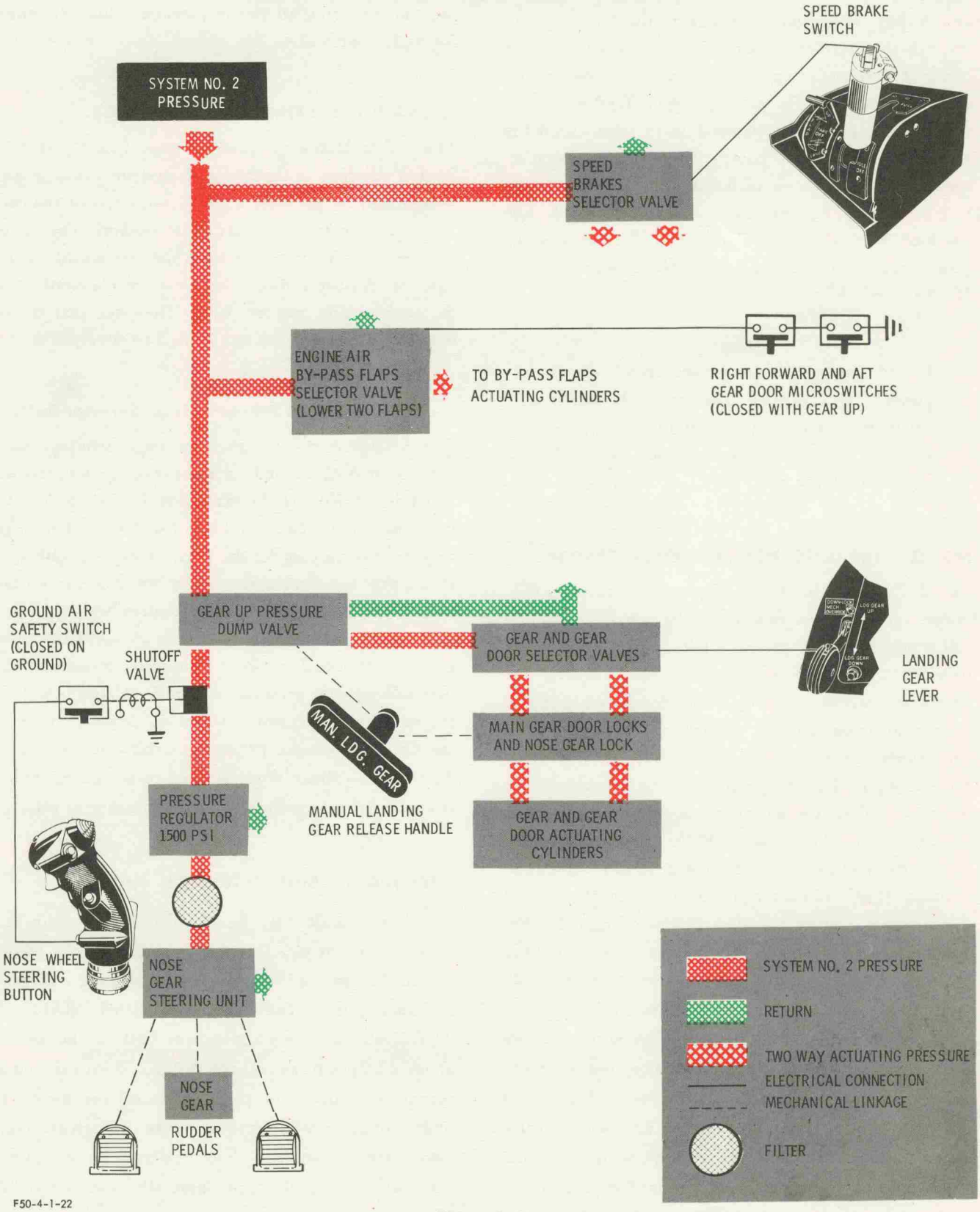


Figure 1-20

EMERGENCY HYDRAULIC POWER SUPPLY SYSTEM.

The emergency hydraulic system (figure 1-19) consists of a pump that is supplied by the No. 1 system reservoir and delivers fluid under pressure to the No. 1 system through the pressure-regulating flow-control valve. The pump is a constant volume, piston type, powered by the ram air turbine. The pressure-regulating flow-control valve diverts emergency pump fluid to return until the ram air turbine (and pump) has reached operating speed. Thus, a hydraulic load cannot be imposed on the turbine before it has reached a speed sufficient to handle the load. 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 ram air turbine will furnish emergency electrical power if necessary. Once extended, the ram air turbine cannot be retracted in the air.

NO. 2 HYDRAULIC POWER SUPPLY SYSTEM.

The No. 2 hydraulic system (figure 1-19) supplies fluid under regulated pressure to the flight controls, pitch and roll damper control valves, landing gear, nose wheel steering, engine air by-pass 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 two filters. Fluid is supplied to the pump from the reservoir which is pressurized to prevent pump cavitation. Fluid from the pump is supplied under 3000 psi pressure, through a filter directly to the accumulator and flight control components, (five outboard cylinders for each aileron and the stabilizer forward cylinder). 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 should a pressure surge occur in the system. The priority valve opens to full flow at 2350 psi. A pressure line from the priority valve outlet port carries fluid to the utility hydraulic system (figure 1-20) which includes the engine-air bypass flaps, landing gear system, nose wheel steering system and speed brake selector valve. The priority valve reseats to zero flow when system pressure drops to 2000 psi, thus retaining all system pressure below this range, for flight controls.

HYDRAULIC SYSTEMS PRESSURE GAGE.

The hydraulic systems pressure gage (23, figure 1-6), located adjacent to the hydraulic systems pressure gage selector switch, provides a visual indication of the pressure available from the hydraulic systems. The system receives 26-volt a.c. power from the instrument a.c. bus auto-transformer through fuses on the electronic compartment circuit breaker panel. The gage dial is calibrated in increments of 100, from 0 to 4000 psi.

Hydraulic Systems Pressure Gage Selector Switch.

The hydraulic systems pressure gage selector switch (25, figure 1-6) located on the lower right instrument panel is labeled "HYD. SYSTEM PRESS. SEL." The two positions are No. 2 and No. 1 or EMER. The switch may be used to connect the pressure gage to either the No. 1 pressure transmitter or the No. 2 pressure transmitter. The No. 1 transmitter measures No. 1 hydraulic system pressure when the system is on normal or emergency operation. The No. 2 transmitter measures No. 2 hydraulic system pressure. A signal voltage is induced in the transmitter which varies in proportion to the amount of hydraulic pressure available in the system. This signal voltage is transmitted to the gage where it is converted to a scale reading of pressure in psi.

RAM AIR TURBINE EXTENSION HANDLE.

A yellow handle (39, figure 1-6) located below the main instrument panel on the right side may be used to extend the ram air turbine which powers the emergency hydraulic pump. The handle is labeled "RAM AIR TURBINE" and requires a firm pull to the stop of about 4 inches to extend the turbine. With the ram air turbine extended into the air stream, the emergency hydraulic pump will supply pressure through the No. 1 system for operation at about one-sixth normal rate of the various hydraulic units normally operated by the No. 1 system.

HYDRAULIC SYSTEM OUT WARNING LIGHT.

The "HYD. SYSTEM OUT" light located on the warning light panel (figure 1-11) illuminates when either the No. 1 or No. 2 hydraulic system decreases to approximately 1250 psi. The hydraulic pressure indicating system can be used to determine which system is out. The warning light is powered from the emergency d.c. bus through a circuit breaker in the electronics compartment. The master caution light also illuminates when the hydraulic system out warning light illuminates.

FLIGHT CONTROL SYSTEM.

Flight controls are comprised of conventional cable and push rod systems, mechanical and electro-hydraulic servo systems, 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, a rudder, and a yaw damper. The rudder is the only primary control surface that is directly controlled by pilot effort, through a cable system. Rudder travel is limited to 25 degrees on either side of neutral position.

FULL POWER IRREVERSIBLE CONTROL SYSTEM.

The ailerons and horizontal stabilizer are dependent upon a complete hydraulic power control system for operation. Movement of the control stick in any direction, even to the slightest degree, immediately affects a servo mechanism. This servo mechanism immediately responds and directs hydraulic pressure to the control surface cylinders which move in the required direction. As soon as the control surface begins to move, a follow-up linkage begins to cancel the original control signal called for by control stick motion, to stop the control surfaces at the required deflection. When the required deflection of the control surface is reached it stops and is hydraulically locked in that position by the actuating cylinders and cannot be moved by external forces reacting upon it. An irreversible feature is also built into the rudder in the form of a centering spring which aids in returning the rudder to neutral and a lock that maintains the rudder in neutral. The lock is provided to cancel aerodynamic forces on the rudder induced by action of the yaw damper surface. The rudder cannot be moved by external forces acting upon it, but can be moved out of the neutral position by operation of the rudder pedals.

STABILITY AUGMENTATION CONTROL SYSTEM.

Roll, Pitch and Yaw.

A three axes stability augmentation control system is provided to compensate and correct for small and rapid changes-of-rate in fundamental airplane stability due to speed or attitude change. Normal electrical power is supplied from the d.c. monitored bus through the instrument inverter. Emergency electrical power is supplied from the a.c. emergency bus through the instrument emergency power transformer and instrument a.c. bus. The system measures rate-of-change and generates an electrically amplified signal. This signal moves a system of valves which in turn directs hydraulic flow and pressure to the actuating cylinders to move the yaw damper, stabilizer or ailerons to a position relative to the amount of correction necessary. This operation does not move the normal surface control linkage or have any effect upon the control stick. While the ailerons move for correction about the roll axis and the stabilizer moves for corrections about the pitch axis a somewhat different system is used for corrections about the yaw axis. This system is a yaw damper (14, figure 1-2) which is installed directly below the rudder and is powered normally from the instrument inverter. The total area of the yaw damper is approximately one square foot and is operated through plus and minus 20° by a hydraulic actuator controlled in the same manner as the actuators used in the pitch and roll system. The stability augmentation system also contains a "washout" circuit. The "washout" circuit is provided to allow 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 small rapid disturbances. The "washout" circuit cancels these signals in favor of pilot initiated signals. In order to decrease the possibility of excessive pitch rate changes with resultant high negative G forces, there is no wash out circuit incorporated in the pitch axis.

AUTO-PITCH CONTROL (APC) SYSTEM.

An auto-pitch actuator is installed. Its function is to prevent inadvertent stalls by moving the control stick forward to neutral when either the pitching velocity, angle of attack, or a combination of both reach a critical value. Signals to effect this come from a rate gyro and/or a vane actuated angle-of-attack detecting system. These signals are amplified and when a critical value is reached they trigger a solenoid valve and operate a small hydraulic cylinder which in turn causes the

TRIM AND STABILITY CONTROL PANELS

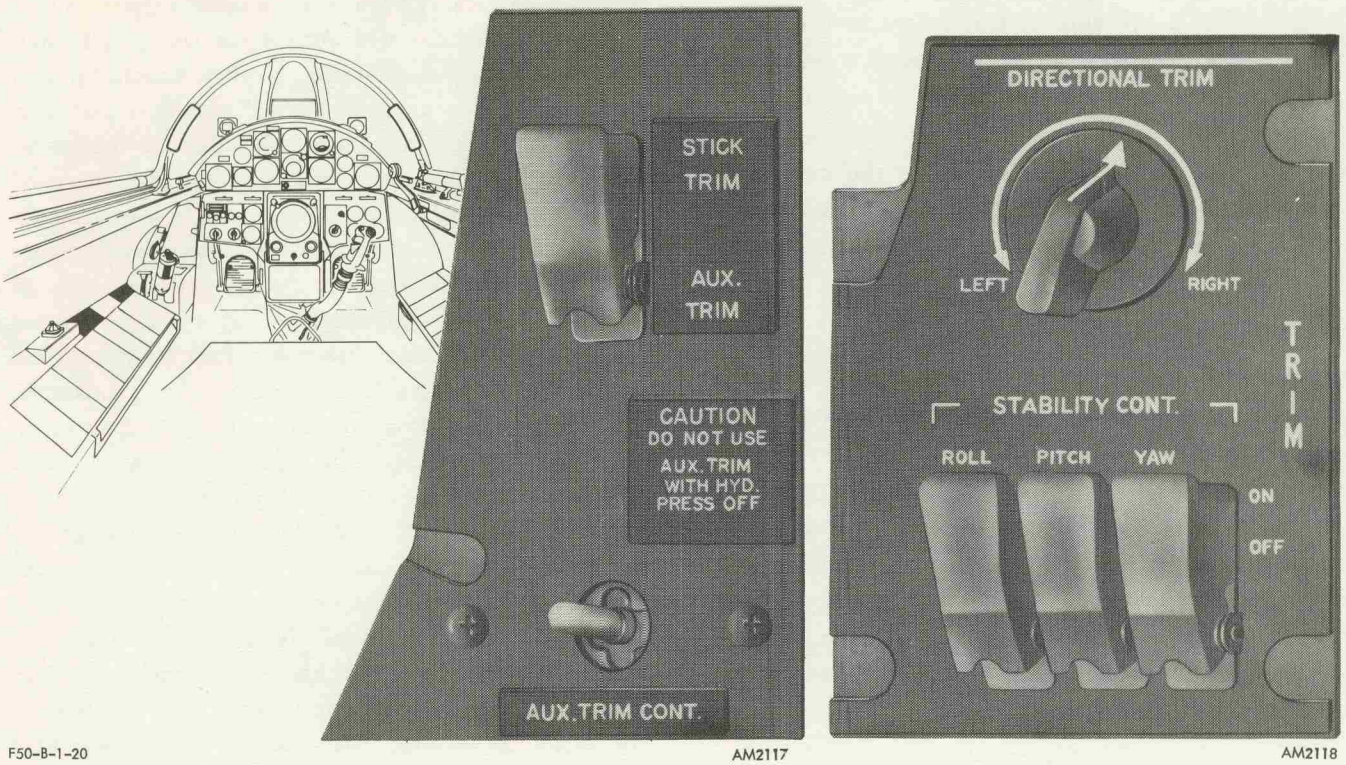


Figure 1-21

stabilizer to assume an airplane-nose-down deflection and the control stick to move forward. This is the only control surface deflection that is ever 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 approximately 32 pounds applied on the stick can override the automatic pitch system. The auto-pitch control vane is located on the right side of the fuselage just forward of the canopy. Electrical power for the auto-pitch control system is supplied by the instrument inverter through the three axes control damper. Hydraulic power is supplied from the No. 1 system. Refer to Section VI for additional information on this system.

Note

The auto-pitch control system is de-energized when the wing flaps are positioned at TAKE-OFF or LAND to preclude an airplane nose-down deflection at low altitudes during take-off or landing.

AUTO-PITCH CONTROL CUT-OUT SWITCH.
An auto-pitch control cut-out switch (15, figure 1-7) is located on the left console and is guarded and safetied to the QN position. The OFF position may be used to de-energize the auto-pitch control system if necessary. Power for the switch is received from the instrument inverter through the three axes control damper. The switch energizes the "AUTO-PITCH OUT" warning light when turned to the OFF position.

WARNING

- The auto-pitch control cut-out switch is provided to de-energize the system because of malfunction or for ground maintenance.
- If the safety wire is broken, do not fly the airplane until cleared by maintenance personnel.

AUTO-PITCH CONTROL INDICATOR.

An APC indicator (16, figure 1-6) located on the right side of the glare shield and powered by the instrument inverter will give an indication of equivalent angle of attack. The indicator may also be used throughout the flight to ascertain operation of the auto-pitch control system as well as inform the pilot of the airplane's relation to the stall condition. The indicator dial is graduated in increments from 0 to 5 with a red area at 5 which is the point where the auto-pitch control becomes effective. Since the stick shaker will be energized prior to auto-pitch control, experience with the indicator will aid the pilot in obtaining maximum performance from the aircraft.

AUTO-PITCH OUT WARNING LIGHT.

A d.c. emergency bus powered, "AUTO-PITCH OUT" warning light on the warning panel (figure 1-11) and the master caution light will become illuminated should the auto-pitch control or the No. 1 hydraulic system become inoperative. This will warn the pilot that more attention will be required in preventing a stall by avoiding high accelerations and low airspeed.

STICK SHAKER SYSTEM.

A control stick shaker, stall warning system has been incorporated into the flight control system. The system will be energized when pitching velocity, angle of attack, or a combination of both reach a value which is less than the auto-pitch control activating system requires. When energized, electrical power from the d.c. emergency bus activates an eccentric motor on the control stick which agitates it in a forward and aft motion. This shaking is a pre-warning to a stall and will commence before the auto-pitch control is actuated. Refer to Section VI for additional information on this system.

Note

No switch is provided to de-activate the stick shaker; however, a circuit breaker (figure 1-14) on the left console can be used to de-activate the system if necessary.

AILERON TRAVEL LIMITER.

A mechanical device is installed in the control stick mechanism which limits aileron travel to about 65 percent normal when the wing flap lever is in the up position. This installation reduces the roll rate of the airplane, and thus reduces the chance of structural damage due to a too rapid roll in normal flight, yet maintains full aileron deflection on landing and take-off. There is no cockpit control for this device other than the wing flaps lever. Refer to Section VI for additional information on this system.



Figure 1-22

TRIM CONTROL SYSTEM.

Aileron and Stabilizer.

The trim actuators are mechanically connected to the trim motors by flexible drive shafts and provide electrical trim of the control surfaces by movement of the servo valve assembly input linkage arms. This operation causes deflection of the control surfaces to a trimmed position but does not move the control stick. The trim motors are d.c. emergency bus powered and contain cam-actuated up-and-down limit switches and a take-off trim indicator light switch.

Yaw Damper.

The trim system for the yaw damper is provided to allow electrical directional trim control of the airplane in the yaw axis. Trim control on the yaw damper has an authority of plus or minus 10°. The system for trim of the yaw damper reacts in the same manner as for augmentation control except that the electrical signals originate from a trim potentiometer located on the trim and stability control panel (figure 1-21) and controlled by the pilot instead of from the yaw damper rate gyro.

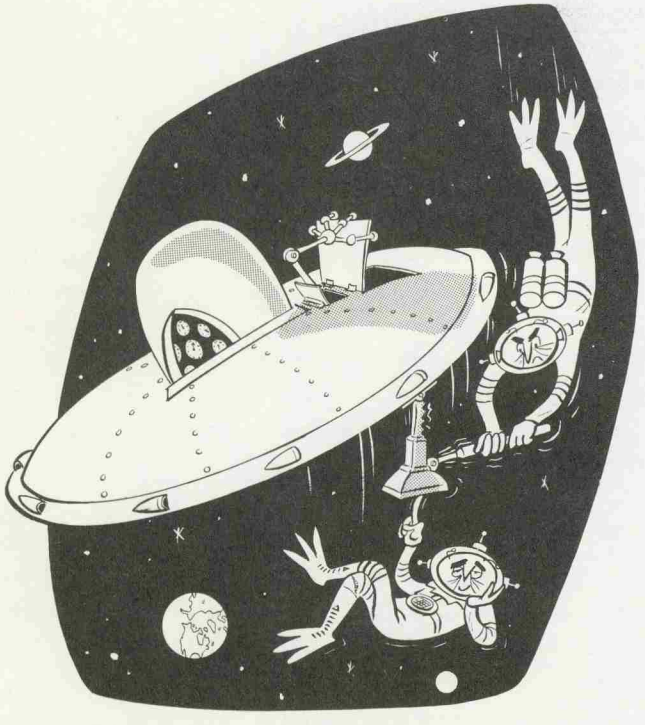
ARTIFICIAL FEEL SYSTEM.

The use of a full power, irreversible control system for actuation of the flight controls prevents air loads and resulting "feel" from reaching the control stick. Therefore, an artificial feel system is installed to provide a sense of control feel under all flight conditions. Normal stick 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 or proportionally to the number of "g's."

CONTROL STICK.

The control stick is mechanically connected (by means of control cables and push rods) to hydraulic control valves at the aileron and horizontal stabilizer hydraulic actuators. Movement of the stick positions these control valves so that power from the flight control hydraulic systems 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 obtained. The control stick grip (figure 1-22) incorporates the primary aileron and horizontal stabilizer trim switch, nose wheel steering button, camera-armament trigger switch, external stores release button and radar track action button.

caution!



THE AUXILIARY TRIM CONTROL SWITCH SHOULD NOT BE OPERATED WITHOUT HYDRAULIC PRESSURE AVAILABLE AS THIS MAY DAMAGE THE TRIM MOTOR.

RUDDER PEDALS.

Primary control for the rudder consists of conventional hanging type rudder pedals, which are connected to cables running aft through the fuselage to a rudder lock assembly. Movement of the pedals causes the cables to operate the rudder lock and rudder. The rudder lock maintains the rudder in neutral and a centering spring aids in restoring the rudder to a neutral position. The rudder pedals can be adjusted fore and aft by use of a gray handle (29, figure 1-6) labelled "PEDAL ADJ." located to the left of the center control panel. The wheel brakes are applied conventionally by toe action on the rudder pedals. Rudder pedal movement also controls nose wheel steering. (Refer to NOSE WHEEL STEERING SYSTEM in this Section.)

AILERON AND HORIZONTAL STABILIZER TRIM SWITCH.

Lateral and longitudinal trim control is provided by a four-position spring-loaded, thumb-actuated switch with a center off position, located on top of the control stick grip (1, figure 1-22). The switch is used to control d.c. emergency bus powered trim motors. Lateral movement of the switch to the left causes a left aileron up, right aileron down operation of the trim motor and trim actuators. Lateral movement to the right causes the reverse. Forward movement of the switch causes a stabilizer leading edge up (aircraft nose down) operation of the trim motor and actuator. Aft movement causes the reverse.

WARNING

- An improperly installed or defective trim switch is subject to sticking in any or all of the actuated positions, resulting in application of extreme trim. If this condition occurs during preflight check and the switch does not return automatically to the center OFF position, enter this fact in the DD Form 781 with red cross and do not fly the airplane.

- If the trim switch sticks in any actuated position during flight, manually return to the center OFF position and use the auxiliary trim control system.

Note

A pressure switch in the No. 1 hydraulic system prevents operation of the aileron and horizontal stabilizer trim switch without hydraulic pressure. Failure of the No. 1 hydraulic system in flight will necessitate the use of the auxiliary trim system to trim the ailerons and stabilizer.

AUXILIARY TRIM SELECTOR SWITCH.

A two position guarded trim selector switch (figure 1-21) is located on the left console and labeled STICK TRIM and AUX. TRIM. The switch is powered from the d.c. emergency bus and may be used to override the control stick trim switch. In the event of failure of the control stick trim switch, the selector switch allows selection of the auxiliary trim switch for control of the stabilizer and aileron trim circuits.

AUXILIARY TRIM CONTROL SWITCH.

A four-position, spring-loaded switch (figure 1-21), with a center OFF position is located just aft of the auxiliary trim selector switch on the left console. This switch (d.c. emergency bus powered) produces the same effects as the control stick trim switch, provided the auxiliary trim selector switch is in AUX. TRIM position. The switch is labeled "AUX. TRIM CONT." and is provided as an auxiliary or standby switch, should the control stick trim switch fail.

DIRECTIONAL TRIM RHEOSTAT.

A rheostat (figure 1-21) placarded "DIRECTIONAL TRIM" located on the trim and stability control panel (left console), is provided to allow electrical directional trim control of the airplane on the yaw axis. Electrical signals from the rheostat are amplified and transmitted to the yaw damper servo valve assembly. The yaw damper mechanism reacts in the same manner as it does for stability augmentation control except that the rheostat adjusts yaw signal output from the three axes control damper amplifier. Power for the three axes control damper amplifier originates from the a.c. instrument bus.

ROLL, PITCH AND YAW DAMPER SWITCHES.

Three guarded switches (figure 1-21) placarded "STABILITY CONTROL," "YAW," "PITCH AND ROLL" are located on the left console. These switches are guarded to the ON position but may be used to disconnect the stability augmentation control system in any or all three axes at any time it may be required by placing switches in the OFF position. Any one or two of the systems may be disconnected without adversely affecting stability augmentation control of the remaining system. All three systems can be disconnected without seriously disturbing flight control; however, the systems are especially desirable for aircraft stability during use of the fire control system. A slight change in trim will occur when the dampers are disconnected.

STABILIZER AND AILERON TAKE-OFF TRIM INDICATOR LIGHTS.

Two d.c. emergency bus powered takeoff trim indicator lights (37, figure 1-6) are located on the left side of the lower instrument panel. With the airplane on the ground, the electrical system energized and hydraulic systems under pressure, the trim indicator lights can be energized either by use of the control stick trim switch, the auxiliary trim control switch, or by the warning light system test switch (5, figure 1-8). The lights are provided to indicate aileron and stabilizer take-off trim position. When the lights are energized the words "STAB. TAKE-OFF TRIM" will be illuminated on the upper light and "AIL. TAKE-OFF TRIM" will be illuminated on the lower light. The lights illuminate whenever the trim motors are run through the take-off trim position of the aileron or stabilizer by action of the trim switches. The lights will not remain on after the trim switch is released. On unmodified aircraft the lights cannot operate once the airplane is airborne, due to a landing gear-actuated ground-air safety switch. On modified aircraft the aileron trim light will illuminate when airborne whenever the trim switch is pressed and the aileron trim is in neutral.

HORIZONTAL STABILIZER TRIM MARKER.

A black "T" painted on the right side of the vertical stabilizer is used as a take-off trim index. When trim is set for take-off, the leading edge of the horizontal stabilizer should be aligned with the index. This enables the ground personnel to assure the pilot that the stabilizer is within the correct tolerance for take-off.

WING FLAP SYSTEM.

The wing flap system consists of trailing edge flaps and leading edge flaps. A boundary layer control system is provided for use with the trailing edge flaps and is automatically operated with the wing flaps in the landing configuration. Both sets of flaps are used for take-off and landing. Each set is electrically interconnected by a control circuit and mechanically interconnected by flexible drive shafts. Although the trailing and leading edge flaps are electrically interconnected by the control circuit, they are not mechanically interconnected. Both sets of flaps are controlled by a single lever.

BOUNDARY LAYER CONTROL DUCT SECTION

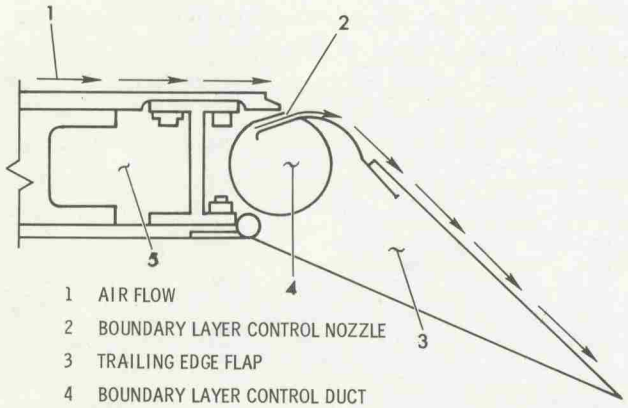


Figure 1-23

Wing Flap Sequencing System.

An automatic wing flap sequencing system is provided. The system operates automatically when the ram air turbine has been extended to obtain electrical power, but only if the wing flap lever is placed to TAKE-OFF position. On unmodified aircraft, when the ram air turbine is extended for electrical power and TAKE-OFF position is selected, the system prevents electrical power from going to the trailing edge flaps until the leading edge flaps have moved to take-off position. On modified aircraft, the flap sequencing is reversed, that is, electrical power is prevented from going to the leading edge flaps until the trailing edge flaps have moved to the take-off position. This reduces the electrical load which can be applied to the ram air turbine-driven generator under these conditions.

WARNING

- The wing flaps are sequenced only to take-off position with the ram air turbine-driven generator supplying electrical power. Therefore, the flap lever never should be placed in the LAND position or returned to the UP position under these circumstances. This action can stall the ram air turbine-driven generator and all electrical power except that furnished by the battery will be lost.
- Also, when using the ram air turbine for emergency electrical power, airspeed should be in excess of 200 knots IAS until the flaps have reached take-off position. This will furnish sufficient air flow for generator operation.
- These precautions are not necessary when the ram air turbine has been extended to obtain emergency hydraulic power only.

TRAILING EDGE FLAP SYSTEM.

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. The flaps are hinged at the forward, lower edge. Movement of the flaps is controlled by two a.c. powered actuators, either of which will provide operation of both right and left trailing edge flaps should one actuator fail. The control circuit for the flaps is from the 28-volt d.c. essential bus and the power circuit is 200/115-volt 3-phase a.c. from the emergency bus.

LEADING EDGE FLAP SYSTEM.

The leading edge flaps form the leading edge of each wing between the fuselage and the wing tip fairings. The flaps are hinged at the aft lower edge. Flap movement is controlled by two a.c. powered actuators. The system is so designed that both right and left leading edge flaps will operate from a single actuator should one or the other fail. The control circuit for the flaps is from the 28-volt d.c. essential bus and the power circuit is 200/115-volt 3-phase a.c. from the emergency bus.

Leading Edge Flap Lock System.

A leading edge flap lock system is provided to lock the flaps in the up position. Each flap is provided with a locking assembly and lock actuator. The left flap lock actuator only has a d.c. emergency bus powered motor. The right flap lock actuator is driven by the left lock actuator through an interconnecting flexible drive shaft. A warning system is installed to indicate when the flaps are in an unsafe position.

WING FLAP LEVER.

The wing flap lever (8, figure 1-7) is located immediately to the left of the throttle. The positions are UP, TAKE-OFF, and LAND. The lever is powered from the d.c. essential bus. A spring loaded guard prevents G loads from pulling the wing flap lever from the UP position. Both leading and trailing edge flaps are controlled by the flap lever. Selection of the TAKE-OFF position will extend the leading edge flaps 15 degrees from the faired position and trailing edge flaps 15 degrees from the faired position. In the LAND position, the leading edge flaps extend 30 degrees from the faired position and the trailing edge flaps extend 45 degrees from the faired position. On modified aircraft, when moving the flap lever from the LAND to the UP position, the lever will latch at the TAKE-OFF position during its travel. In order to release the latch, the lever must be pulled back (toward LAND) approximately ¼ inch. The lever can then be moved forward to the UP position. In the UP position both sets of flaps will retract to the UP (faired) position; however, the "L. E. FLAP UNSAFE" warning light on the warning light panel will be energized until the leading edge flaps are fully retracted and locked.

WING FLAP POSITION INDICATORS.

Position indicators (33, figure 1-6) for the trailing and leading edge flaps are located on the left side of the lower instrument panel. Two windows are provided and 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. A cross-hatched indication will be given when the flaps are in any position other than that selected or when the electrical system is not energized. The indicators are powered by the 28-volt d.c. monitored bus.

LEADING EDGE FLAP UNSAFE WARNING LIGHT.

A warning light (figure 1-11) on the warning light panel is provided to warn the pilot of an unlocked condition of the leading edge flaps. When energized, the words "L.E. FLAP UNSAFE" will be illuminated. The lights will be energized only when the flap lever is in the UP position and the leading edge flap locks are not engaged. The light is powered from the 28-volt d.c. emergency bus. The master caution light does not illuminate when the leading edge flap unsafe warning light illuminates.

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 1-23). 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 entrains the adjacent layer of air and causes it to bend through the flap deflection angle and thus flow separation is prevented. This results in a reduced landing speed. The system is completely automatic in its operation.

Boundary Layer Control Valve.

Since boundary layer control is used only with a 45° flaps setting there is no air flow for flap angles of 15° or less. This is accomplished by a valve which is mechanically driven by the flap actuator, thus the valve's position will always correspond to that of the flaps. The valve butterfly remains closed from 0° to 15° flap angle. For angles greater than 15° the valve modulates to full open at 45°

SPEED BRAKE SYSTEM.

The speed brakes consist 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 60 degrees from the retracted (faired) position. The speed brakes are electrically controlled from the d.c. emergency 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 2000 psi. No position indication is provided for the speed brakes.

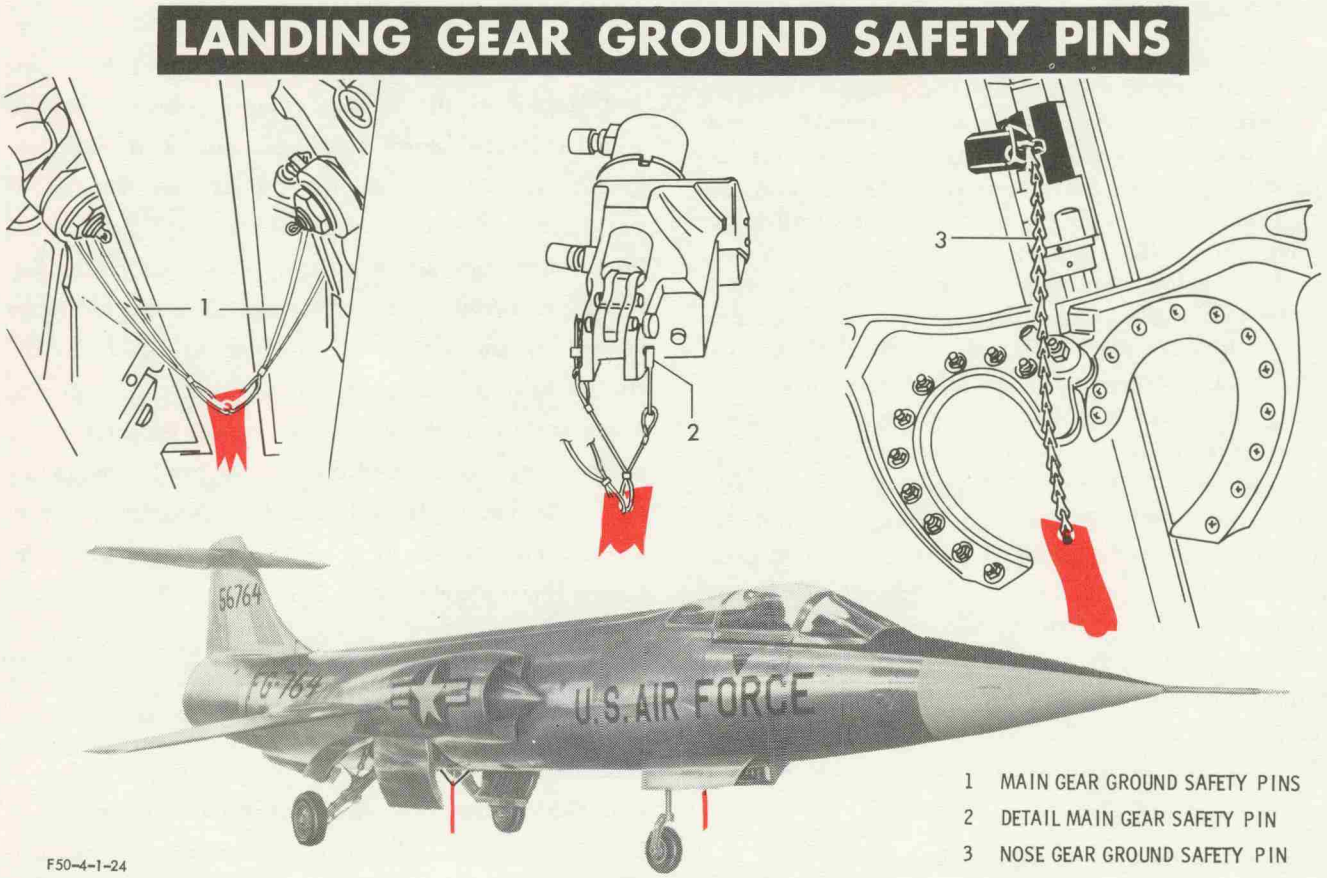


Figure 1-24

Speed Brake Switch.

The speed brake switch (4, figure 1-5) is a thumb actuated, slide type switch located in the top of the throttle lever and powered from the d.c. emergency 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 electrically de-energizes the solenoid valves and hydraulically locks the speed brakes in any position between out and in.

Note

Due to leakage in some speed brake control valves, the speed brakes may gradually extend when the switch is left in the NEUTRAL position. Inadvertent extension may be prevented by positioning the switch to the IN position whenever the speed brakes are not being used.

LANDING GEAR SYSTEM.

The aircraft is equipped with two main gear and a nose landing gear. Normally the landing gear system is hydraulically operated and electrically controlled by the d.c. emergency bus. Normal extension or retraction time of the landing gear is 4 to 5 seconds. A manual release system is provided for emergency extension of the gear which also requires 4 to 5 seconds. There is no means of retracting the landing gear after an emergency extension.

Main Landing Gear.

The main landing gear retracts forward and inward into the fuselage wheel wells. A linkage causes the wheels to rotate 90 degrees 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. The aft door is mechanically linked to the gear and travels up and down with the gear. 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 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 four 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 (2, figure 1-24) of each gear.

Nose Landing Gear.

The nose gear retracts forward 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 hook engages a lug on the nose wheel fork to lock the gear up. A static wire on the nose wheel fork electrically grounds the airplane when the wheel touches the ground. A ground safety pin is provided for manual installation on the downlock stop cartridge (3, figure 1-24). The nose gear is steerable through use of the rudder pedals. (Refer to NOSE WHEEL STEERING SYSTEM, of this section).

LANDING GEAR LEVER.

The landing gear lever (figure 1-10) is located on the left forward panel. The lever has two positions: LDG. GEAR DOWN and LDG. GEAR UP. The lever electrically controls the landing gear and landing gear door hydraulic selector valves. When the lever is moved to the UP position, 28-volt d.c. emergency bus power is directed to selector valves which are electrically sequenced to direct hydraulic pressure to open the main gear forward doors, retract the nose gear, the 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 placed in the DOWN position the same electrical power actuates the selector valves which 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. When the airplane touches down and its weight rests on the main gear, a ground-air safety switch causes the landing gear lever downlock solenoid to lock the lever in the DOWN position, thereby preventing the gear from inadvertently being raised on the ground.

LANDING GEAR LEVER UPLOCK.

A landing gear lever uplock mechanism is provided. A trigger (figure 1-10), which extends upward from the top of the lever is used to release the lever uplock mechanism. The lock is provided to prevent the gear lever from being lowered inadvertently.

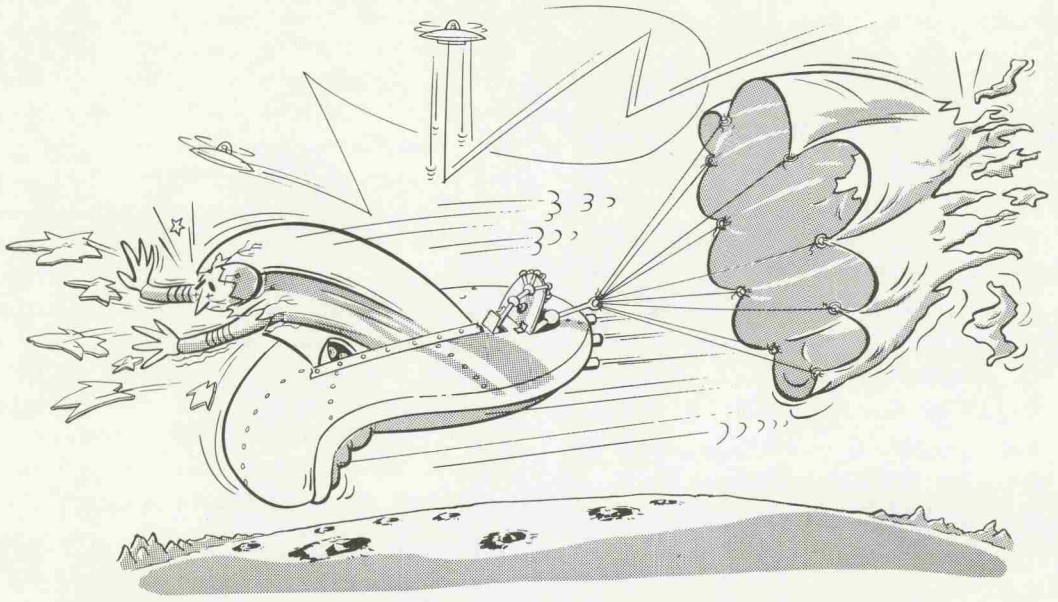
LANDING GEAR LEVER OVERRIDE BUTTON.

A landing gear lever override button (figure 1-10) is provided just above the landing gear lever. The button may be used in an emergency to override the lever downlock should it become necessary to raise the gear when the weight of the aircraft is on the landing gear. When the airplane 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 down-lock bypass, operated by the push-button. When the weight of the aircraft is off the landing gear, 28-volt d.c. emergency 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 (38, figure 1-6) located below the main instrument panel on the left side is labeled "MAN. LDG. GEAR." The handle is provided to release the main landing gear door uplocks and the nose gear uplocks which allows the gear to lower by gravity and air load forces. The gear is then locked by spring-loaded downlocks. Approximately a 10 inch pull to the stop is required to release the gear. The landing gear cannot be retracted after being lowered by means of the manual landing gear release handle. If the manual landing gear release handle is used, a notation should be made in the DD Form 781 so that the system valves will be repositioned prior to the next flight.

caution!



DUE TO THE LOCATION OF THE DRAG CHUTE, IT CAUSES AN AIRCRAFT NOSE-DOWN PITCHING MOMENT WHEN DEPLOYED; THEREFORE, IT SHOULD NOT BE DEPLOYED UNTIL THE NOSE WHEEL IS ON THE GROUND AND SPEED IS BELOW THE SHEAR LINK STRUCTURAL LIMIT.

LANDING GEAR POSITION INDICATORS.

Three green lights are installed on the left side of the lower instrument panel (35, figure 1-6). When the lights are illuminated they indicate that the landing gear is down-and-locked. The lights are labeled "LH GEAR DOWN," "NOSE GEAR DOWN" and "RH GEAR DOWN." As each gear reaches the down-and-locked position, 28-volt d.c. power from the emergency bus is directed through the warning light dimming circuit to illuminate the indicator lights. The lights go off any time the gear is not down and locked except when energized by the warning light test switch.

LANDING GEAR SYSTEM MALFUNCTION LIGHT.

A red warning light is installed in the transparent knob on the landing gear lever (10, figure 1-7). This light provides the pilot with a visual signal whenever the landing gear is not up and locked or down and locked. The light receives power from the 28-volt d.c. emergency bus.

LANDING GEAR WARNING SIGNAL.

A warning signal is produced in the pilot's earphones through the interphone system when the landing gear is not in the down-and-locked position.

Unmodified Aircraft.

The warning is actuated by the throttle and mechanically-actuated gear and gear door switches. The landing gear warning signal will be heard by the pilot if the throttle is moved toward the IDLE position far enough to operate the landing gear warning, throttle operated switch, and if the nose gear and main gear are not down and locked. This causes 28-volt, d.c. power to be delivered from the emergency bus to the signal system.

WARNING

At the present time the landing gear warning signal will be heard by the pilot whenever the throttle is retarded below approximately 83% and the gear is not down and locked. The high power requirements necessary throughout the landing pattern make this warning system unsuitable. The landing gear position indicators (3 green lights) should be checked prior to landing to insure proper landing gear operation.

Modified Aircraft.

An engine speed and pitot static operated landing gear warning signal has been incorporated in modified aircraft. When the throttle is retarded below 100%, the altitude is below 10,000 ±1500 feet and the airspeed is below approximately 220 Knots IAS, the warning signal will be energized.

Landing Gear Warning Signal Cut-Off Button. (Unmodified Aircraft)

A warning signal cut-off button (figure 1-10) is provided on the left forward panel. The button is labeled "HORN CUT-OFF" and may be used to silence the warning signal produced in the earphone by the warning system. The signal may be silenced by pushing the button; however, if the throttle is moved forward and then retarded again, it will be necessary to push the button again to silence this signal. (Refer to LANDING GEAR WARNING SIGNAL in this Section for additional information.)

NOSE WHEEL STEERING SYSTEM.

The steering system provides power steering for the nose wheel when the airplane is on the ground. The nose wheel is steerable 25 degrees either side of center. Steering is accomplished by a steer-damper unit that is hydraulically powered and is controlled through a cable system by the rudder pedals. No. 2 hydraulic system pressure from the landing gear-down line is routed to the steering system through a solenoid shut-off valve and a pressure reducing valve which reduces system pressure from 3000 psi to 1500 psi. The solenoid shut-off valve is controlled by a switch on the control stick grip. The system is irreversible in that forces on the nose wheel cannot be transmitted back to the rudder pedals. Upon retraction, the nose wheel will automatically center itself.

Note

- The ground-air safety switch and relay are connected into the shut-off valve circuit in a way which insures that the steering system is inoperative unless the weight of the aircraft is on the main landing gear.
- If the manual landing gear release handle is used to lower the gear, hydraulic pressure to the nose wheel steering system is lost and nose wheel steering is inoperative.

Steer-Damper Unit.

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

Note

The clutch will engage only when the nose wheel and rudder pedals are in the same relative position.

NOSE WHEEL STEERING BUTTON.

A push-button (5, figure 1-22) mounted on the control stick grip, engages the nose wheel steering system. When the button is pressed and held, d.c. monitored bus power is directed to a shut-off valve which directs No. 2 hydraulic system pressure to the nose wheel steering unit. A clutch is then engaged hydraulically to link the rudder cable with the steering unit when the rudder pedals and nose wheel are aligned.

Note

The nose wheel steering button is operable only if d.c. monitored bus power is available and the weight of the aircraft is on the main landing gear.

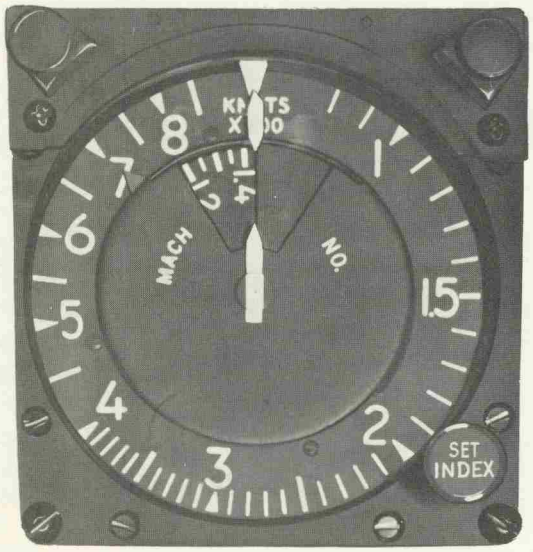
WHEEL BRAKE SYSTEM.

Each main gear incorporates a hydraulic brake assembly. The brakes are self-adjusting, segmented rotor-type. The brake hydraulic system is independent of aircraft hydraulic system. Each brake is operated conventionally by toe action on the rudder pedals which are linked to master brake cylinders. Fluid reservoirs are incorporated as part of each brake valve. A fluid quantity sight gage (figure 1-30) is mounted on the front of the pressure bulkhead and is accessible by sliding the nose section forward. The gage is connected by tubing to both master brake cylinders and is used to fill the system. No parking brakes are provided.

DRAG CHUTE SYSTEM.

A 16-foot ribbon-type drag chute is provided to reduce landing distances. The chute, packed in a deployment bag, is stowed in a compartment located in the lower surface of the aft fuselage and is mechanically controlled from the cockpit. A shear link is provided which will shear and release the drag chute if deployed at airspeeds in excess of the shear link structural limit.

AIRSPEED AND MACH NUMBER INDICATOR



AM2123
F-50-8-1-26

Figure 1-25

DRAG CHUTE HANDLE.

The drag chute handle (36, figure 1-6) is located at the left of the lower instrument panel. When pulled straight aft (about 2 inches) to the stop (without turning the handle) the spring loaded drag chute door will open and a pilot chute will be deployed, the pilot chute in turn, deploys the drag chute. The drag chute can then be released at any time by turning the handle 90 degrees clockwise and pulling to the next stop (about 4 inches). The handle is under spring tension during the final pull to jettison the chute and when the handle is released it will retract to the first stop.

INSTRUMENTS.

Most of the instruments are powered by the a.c. and/or d.c. electrical systems.

Note

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

PITOT PRESSURE AND STATIC SYSTEMS.

The pitot pressure and static systems operate the airspeed indicator, altimeter, and vertical velocity indicator. The system is also connected to the gunsight pressure transmitters. The pitot-static head is mounted on a boom extending forward from the nose radome. The head is electrically heated. The heating element in the head is controlled by a switch on the right console (figure 4-3). (Refer to Defrosting and Anti-Icing Systems, Section IV.)

AIRSPEED AND MACH NUMBER INDICATOR.

The indicator (7, figure 1-6) consists of a pitot-static operated indicated airspeed mechanism which drives a pointer to indicate airspeed 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 80,000 feet. The Mach number range is from 0.50 to 2.2. 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 positioned as a function of equivalent airspeed, and indicates in terms of indicated airspeed the specific value of equivalent airspeed which has been preset into the instrument.

Airspeed Setting Index.

An airspeed setting index is provided to assist the pilot in setting a speed reference marker. The setting of the index is controlled by a knob in the lower right corner. The index is adjustable over a range of 100 to 700 knots indicated airspeed.

ALTIMETER.

In addition to the standard 1000 and 100-foot pointers, the altimeter (8, figure 1-6) uses a new 10,000 foot pointer (segmented disk with an extension pointer) which serves as a warning indicator. The warning indicator is a striped section which appears through the segmented disk at altitudes below 16,000 feet. This altimeter offers improved readability and gives warning when an altitude of less than 16,000 feet is entered.

ACCELEROMETER.

A three-pointer accelerometer (31, figure 1-6) indicates positive and negative G-loads. In addition to the conventional 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 (1G) position, press the knob on the lower left corner of the instrument.

STANDBY COMPASS.

A standby magnetic compass (3, figure 1-6) is provided for navigation if the electrical system fails or to check the directional indicator. The compass is located on the left side above the instrument panel glare-shield, and is hinged to fold forward when not in use. The light within the compass case is controlled by a rheostat on the right console interior lights panel (figure 4-5).

J-4 DIRECTIONAL INDICATOR.

Refer to NAVIGATION EQUIPMENT in Section IV.

MM-2 ATTITUDE INDICATOR.

An MM-2 attitude indicator (11, figure 1-6) provides the pilot with a remote reference pictorial presentation of aircraft attitudes. The installation consists of an indicator assembly mounted on the instrument panel and a remotely located K-4B control assembly which incorporates a vertical gyro, rate of turn gyro, pitch and bank erection system, and associated components. Any angular motion of the aircraft with respect to the vertical reference established by the gyro is detected by the control assembly and electrically transmitted to the universally mounted sphere on the indicator assembly. The gyro is mounted so that aircraft attitude is shown accurately through 360° of roll and plus or minus 85° of pitch. The pitch and bank erection system reduces turning errors to a minimum and prevents the accumulation of pitch errors during multiple acrobatic maneuvers. Acceleration and deceleration will cause slight errors in pitch indications which will be most noticeable on take-off. Indications of aircraft attitude are presented by movement of the universally mounted sphere which serves as a background for a miniature reference airplane. The horizon is represented by a fluorescent white line on the sphere and separates sky and earth presentations which are light gray and black, respectively. Both the sky and earth areas are marked with horizontal lines in five degree increments, enabling the pilot to determine accurately his pitch attitude up to 85 degrees of climb or dive. The scale is expanded to amplify pitch displacement, providing quick readability within one degree of pitch attitude. Bank angles are indicated by the position of the pointer at the top of the sphere in relation to the semicircular bank scale around the upper half of the instrument. An adjustment knob located on the lower right side of the instrument permits alignment of the horizontal lines on the sphere with the miniature reference airplane. Turning the knob causes the sphere to be electrically rotated so that a relationship between reference lines and the miniature airplane may be established to provide greater ease in maintaining a desired pitch attitude. The OFF flag in the upper left corner of the instrument will be visible whenever d.c. or a.c. power to the instrument is disrupted. The system starts operating as soon as 28-volt d.c. essential bus power and 115-volt a.c. from the emergency bus are available, but the OFF flag will not retract until a warmup period of approximately 2½ minutes has elapsed. The "OFF" flag will appear in case of a complete a.c. or d.c. power failure. However, a slight reduction in a.c. or d.c. power, or failure of certain electrical components within the system *will not* cause the "OFF" flag to appear, even though the system is not functioning properly.

WARNING

- It is possible that a malfunction of the attitude gyro can be detected only by checking it with the slaved gyro magnetic compass and the turn and slip indicator.

- A slight amount of pitch error in the indication of the Type MM-2 attitude indicator will result from accelerations or decelerations. It will appear as a slight climb indication after a forward acceleration and as a slight dive indication after deceleration when the airplane is flying straight and level. This error will be most noticeable at the time the airplane breaks ground during the take-off run. At this time, a climb indication error of about 1½ bar widths will normally be noticed; however, the exact amount of error will depend upon the acceleration and elapsed time of each individual take-off. The erection system will automatically remove the error after the acceleration ceases.

Gyro caging is accomplished automatically by the pitch and bank erection system thereby eliminating the necessity for manual caging provisions. If the pitch limit of 85° is exceeded, the sphere will roll 180 degrees, reversing the normal position of the sky and earth reference area, but still providing an accurate indication of aircraft attitude. As soon as the aircraft returns below the 85 degree limit, the sphere will roll back to its normal position without introducing any errors into the system.

TURN AND SLIP INDICATOR.

A conventional turn and slip indicator (4, figure 1-6) is mounted on the main instrument panel. The instrument is electrically driven by 28-volt d.c. power from the d.c. emergency bus.

VERTICAL VELOCITY INDICATOR.

The vertical velocity indicator (12, figure 1-6) is mounted on the main instrument panel. The indicator registers the rate of climb or descent in feet per minute and is operated by the static air system.

WARNING LIGHT SYSTEM.

WARNING PANEL LIGHTS SYSTEM.

The warning panel lights system functions to give the pilot visual indication of failure of certain critical power equipment and in case of failure or unsafe conditions in critical areas of the aircraft. The system consists of a warning light panel (figure 1-11), a master caution light and reset bar (18, figure 1-6), 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. Should a failure occur in one of the systems, the warning light for that particular system will remain on until the failure is corrected. The warning panel lights system is powered from the emergency d.c. bus through a circuit breaker in the electronics compartment. (Refer to the particular system associated with each warning light in this Section.)

Master Caution Light.

The "MASTER CAUTION" (18, figure 1-6) light goes on when any of the warning panel lights (except the "L.E. FLAP UNSAFE" light) are energized. A reset bar on which the master caution light is mounted, permits the pilot to push and de-energize the master caution light even though a fault persist and the individual warning panel light stays on. This permits the master caution light to indicate a second fault if one should occur while the first fault is still present.

Warning Panel Lights.

The following placard type warning lights are located on the warning panel: "FUEL LOW LEVEL," "FUEL BOOST PUMP FAIL.," "INST. ON EMER. POWER;" "NO. 1 GENERATOR OUT," "NO. 2 GENERATOR OUT," "HYD. SYSTEM OUT," "AUTO PITCH OUT," "DC MONITORED BUS OUT," "HATCH UNSAFE," "L.E. FLAP UNSAFE." "ENGINE ANTI-ICING ON" and "ENGINE OIL LOW LEVEL."

WARNING LIGHTS DIMMING SYSTEM.

The warning lights dimming circuit provides a means of reducing the brilliance of all warning lights with a single rheostat. The warning light dimming relay coil is connected to the emergency d.c. bus through the "WARN LT." circuit breakers and the instrument light switch.

Warning Lights Dimming Rheostat (Instrument Light Dimming Rheostat).

A rheostat on the right console lighting panel (figure 4-5) labeled "INSTRUMENT," OFF and BRT. may be used to dim the warning lights. When the rheostat is in the OFF position, as 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 a reduced brilliance. Once the airplane's electrical system has been de-energized 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

PAGES 1-47 THRU 1-52 ARE MISSING

EMERGENCY EQUIPMENT
CANOPY
EJECTION SEAT

PAGES 1-47 THRU 1-52 ARE MISSING

EMERGENCY EQUIPMENT
CANOPY
EJECTION SEAT

Note

The rigid seat style survival and oxygen kit container is contractor furnished equipment. No parachute support block is required because the aft section of the kit has a parachute support. If aircraft are furnished without kit containers, either the MC-2 seat cushion or the MD-1 contoured seat style survival kit container should be used. When the MD-1 container is used, the parachute support block is required.

AUTOMATIC SEAT BELT.

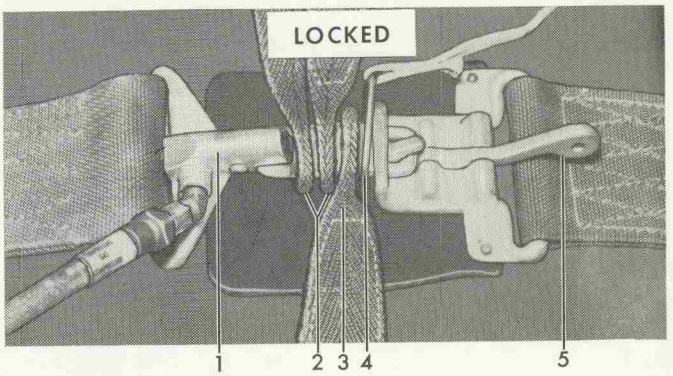
The ejection seat is equipped with a modified MA-6 automatic-opening seat belt (figure 1-29) which facilitates pilot separation from the seat following ejection.

warning!

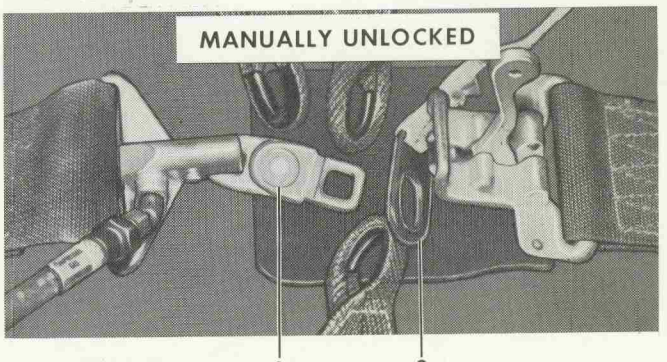


THE INITIATORS AND THE CATAPULT CONTAIN EXPLOSIVE CHARGES AND SHOULD BE HANDLED BY QUALIFIED MAINTENANCE PERSONNEL ONLY.

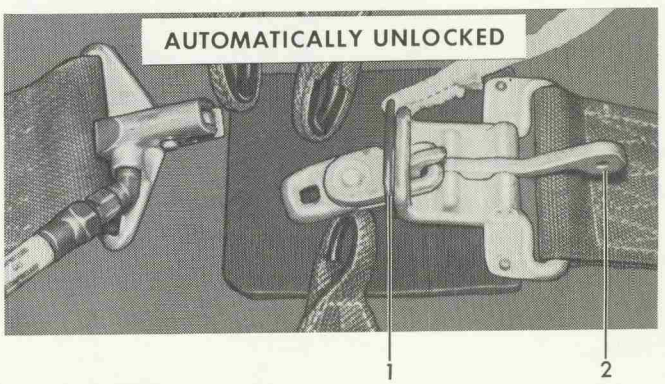
AUTOMATIC SEAT BELT



1. AUTOMATIC RELEASE
2. SHOULDER HARNESS
3. TIE-DOWN STRAP
4. PARACHUTE LANYARD ANCHOR
5. MANUAL RELEASE



1. SWIVEL LINK
2. PARACHUTE LANYARD ANCHOR FREE

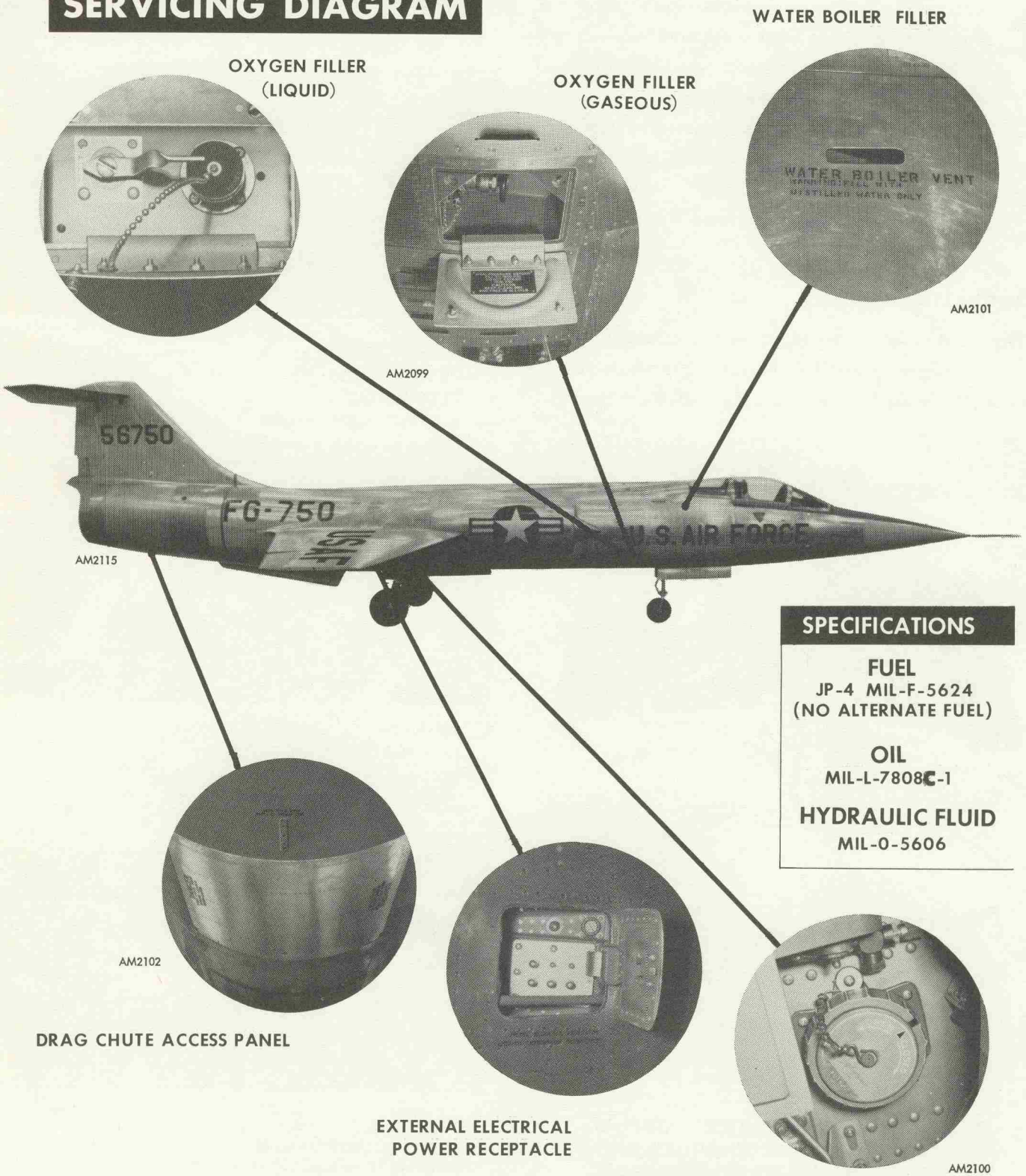


1. PARACHUTE LANYARD ANCHOR RETAINED BY SHOULDER ON SWIVEL LINK
2. MANUAL RELEASE LEVER LOCKED

F50-4-1-28

Figure 1-29

SERVICING DIAGRAM



SPECIFICATIONS

FUEL
JP-4 MIL-F-5624
(NO ALTERNATE FUEL)

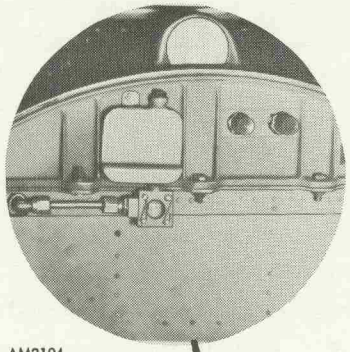
OIL
MIL-L-7808C-1

HYDRAULIC FLUID
MIL-O-5606

GROUND TURBINE COMPRESSOR AND SPECIAL ELECTRICAL RECEPTACLES

Figure 1-30

HYDRAULIC BRAKE
SIGHT GAGE AND FILLER

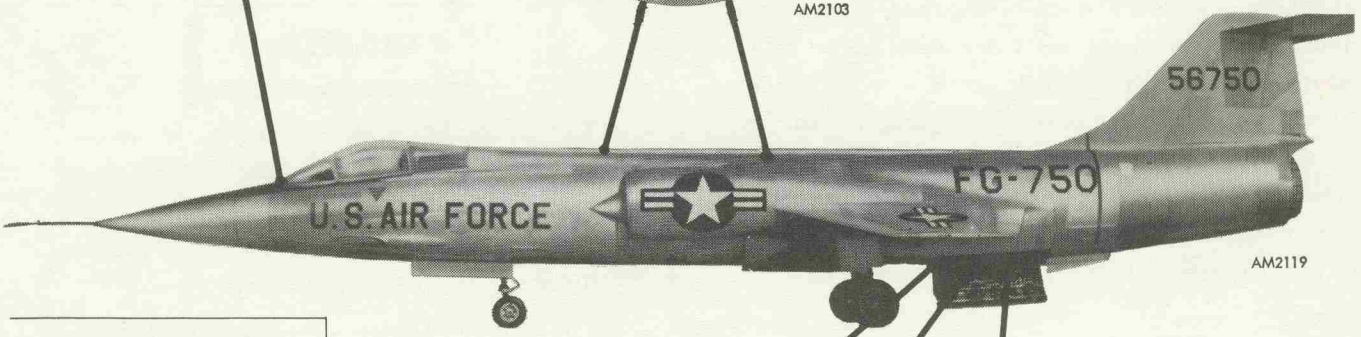


AM2104

REFUELING WELLS



AM2103

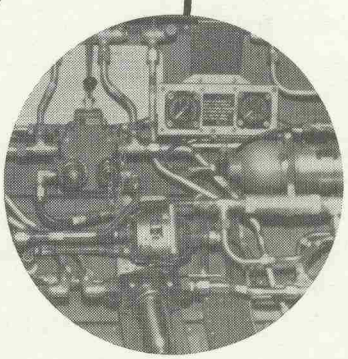


AM2119

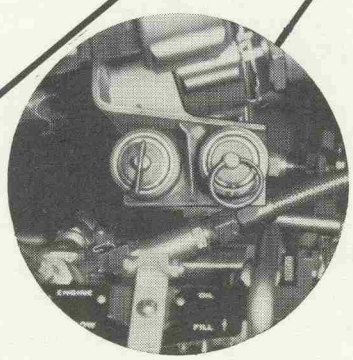
OXYGEN
GASEOUS BB-O-925 TYPE I
LIQUID BB-O-925 TYPE II

NITROGEN
MIL-N-6011 GRADE A
TYPE I (WATER PUMPED)

WATER
USE DISTILLED WATER

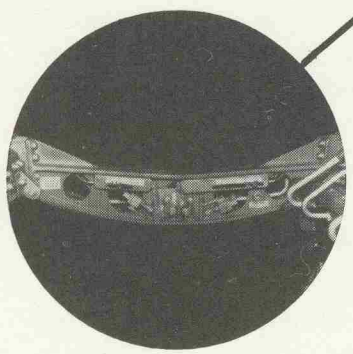


HYDRAULIC SYSTEM FILLERS
ACCUMULATOR AND
PRESSURE GAGES



AM2126

OIL FILLER



HYDRAULIC SYSTEM QUANTITY
GAGES AND BLEED LINES

F50-5-1-30(2)

Belt opening is accomplished as part of the ejection sequence and requires no additional effort on the part of the pilot. As the seat travels down the rails during ejection a mechanical trip bar fires the seat belt initiator. This is a one second delay initiator which upon firing, produces gas pressure which is directed to the belt buckle through a flexible hose. Gas pressure opens the buckle which releases the seat belt, shoulder harness and tie down strap.

Seat Belt—Parachute Attachments.

If the pilot is wearing an automatic-opening aneroid type parachute, the parachute lanyard anchor from the chute-opening device must be attached to the swivel link. As the pilot separates from the seat, the lanyard which is anchored to the belt, serves as a static line to arm the chute-opening device. The parachute will then open at the preset time lapse or altitude. The automatic-opening seat belt is unlocked in the same manner as a conventional seat belt, and can be opened manually if the automatic opening fails.

SEAT VERTICAL ADJUSTMENT SWITCH.

The seat is adjusted vertically by means of an electric actuator mounted on the lower end of the catapult. The switch (15, figure 1-28) is located on the right side of the seat bucket. Power for seat adjustment is derived from the 28-volt d.c. monitored bus.

SHOULDER HARNESS INERTIA REEL LOCK LEVER.

The shoulder harness inertia reel lock lever, on the left side of the seat bucket, is a conventionally operated manual device for locking and unlocking the shoulder harness. The lever has two positions, LOCKED and UNLOCKED. Each position is spring loaded to hold the lever in the selected position. An inertia reel, located on the back of the seat, will maintain a constant tension on the shoulder straps to keep them from becoming slack upon backward movement. This inertia reel also incorporates a locking mechanism which will lock the shoulder harness when a 2 to 3 G force has been exerted in a forward direction. When the reel is locked in this manner, it will remain locked until the lock lever is moved to the LOCKED position and then returned to the UNLOCKED position.

Note

It is recommended that the inertia reel lock lever be left in the LOCKED position since all cockpit controls are easily accessible with the inertia reel locked.

EJECTION RING.

An ejection ring (8, figure 1-28) located on the front of the seat bucket is the primary control for ejection. The one and only ejection system safety pin (10, figure 1-28) is installed in the ejection ring housing bracket and safeties both ejection ring and primary initiator.

WARNING

- The ground safety pin inserted in the ejection ring housing prevents the catapult from being fired accidentally while the aircraft is on the ground. This safety pin must be removed before flight and replaced after landing.
- The catapult system is armed when the initiator ground safety pin is removed.

FOOT SPURS.

Foot spurs (9, figure 1-28), attached to the pilot's feet, are attached to the ejection seat by cables. Normal foot movement is in no way restricted as the cables are under a slight spring tension and reel in and out freely. When the ejection ring is pulled and the knee guards rotate from their stowed position, the cables to the foot spurs are reeled in, pulling the feet into the foot rests. As the seat moves down the rails, a mechanical trip bar fires an initiator which activates the automatic seat belt and one side of the cable cutters. This will release the seat belt and both feet. A secondary cable cutter system is also activated by a separate trip bar which fires another initiator operating the other side of the cable cutters, thus insuring foot release. Either side of the cable cutters, when actuated, will completely release both feet.

MANUAL CABLE CUTTER RING.

All modified ejection seats incorporate an emergency means for cutting the foot retractor cables. A yellow D ring (16, figure 1-28) located to the right of the seat

headrest has been connected to an initiator which operates the cable cutters. This provides a manual means for activating the cable cutters initiator should the automatic cable cutter system fail or if a rapid abandonment from the aircraft is required on the ground.

ESCAPE HATCH JETTISON HANDLE.

A yellow handle (27, figure 1-6) located on the right of the center control stand may be used to jettison the escape hatch in case the automatic ejection system fails. Manual release of the hatch and seat is accomplished by first pulling the hatch escape handle to the stop (about 7 inches). This action manually jettisons the hatch and pulls the pin on the catapult free-fall disconnect. The ejection ring must then be repulled. This action will cause the free-fall disconnect to revolve and the initiator-operated thruster to be by-passed. The catapult initiator will then fire the catapult and eject the seat or the seat will free fall clear of the aircraft. When this secondary means of escape is necessary, the automatic foot retractors, knee guards, tie-down strap tightening, inertia reel lock, and control stick stowage do not operate. However, if the feet are pulled back, they will be retained in that position and released in the normal manner of ejection.

HATCH UNSAFE WARNING LIGHT.

If the hatch is not properly locked, a mechanical indicator plus a microswitch will illuminate the "HATCH UNSAFE" warning light on the warning panel (figure 1-11). Power for the warning light is derived from the emergency d.c. bus.

PRE-EJECTION SEQUENCE.

When the ejection ring is pulled, firing the initiator, the emitted gas pressure operates the thruster which causes the following: Raises the knee guards and deploys the arm nets, pulls the pilot's feet back into the foot rests, tightens the tie-down strap, locks the inertia reel, stows the control stick in the forward position and dumps the cabin pressure. The emergency oxygen supply is automatically actuated by movement of the right knee guard. The ejection ring is connected to the initiator by a linkage having three inches of slack.

Note

- The slack assures that the initiator will not fire if the ejection ring is inadvertently knocked or pulled. Also, the slack in the system places the hands and arms in the most advantageous position for a safe ejection.
- The ejection ring cable is restrained by a spring-cartridge designed to absorb the sudden shock of the airloads which could cause the pilot to lose his grip on the ring.

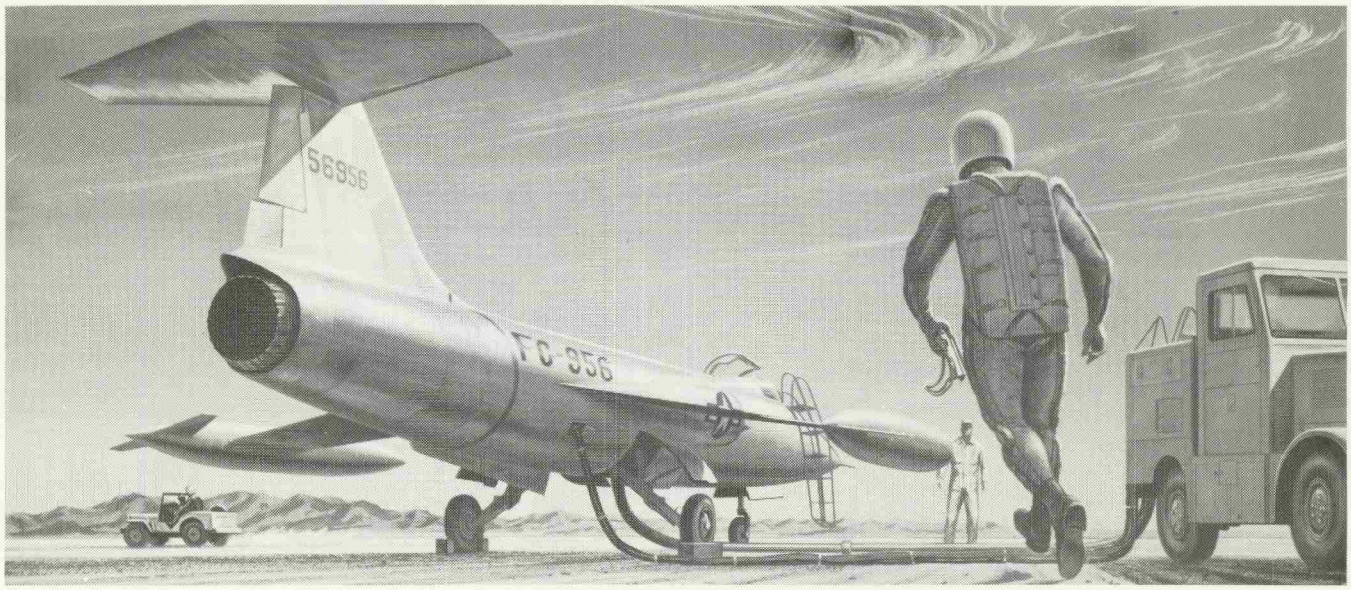
FINAL EJECTION SEQUENCE.

The thruster fires the catapult initiator which, in turn fires the seat catapult. One inch of seat travel releases and ejects the escape hatch. As the seat continues downward, mechanical trip bars fire the initiators which open the seat belt and cut the foot retractor cables.

AUXILIARY EQUIPMENT.

Information concerning the following auxiliary equipment is supplied in Section IV, Air Conditioning and Pressurization System, Defrosting and Rain Removal System, Anti-icing Systems, Communications and Associated Electronic Equipment, Lighting Equipment, Oxygen System, Navigation Equipment, Armament Equipment, and Miscellaneous Equipment.





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AM-2828

normal procedures SECTION II

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PREPARATION FOR FLIGHT

FLIGHT RESTRICTIONS.

Refer to Section V for detailed airplane and engine limitations.

FLIGHT PLANNING.

Refer to Appendix I of Confidential Supplement, T.O. 1F-104A-1A, to determine the fuel quantity, engine settings and airspeeds that are required to complete the mission.

TAKE-OFF AND LANDING DATA CARDS.

Refer to Appendix I of Confidential Supplement, T.O. 1F-104A-1A, for the information necessary to fill out the take-off and landing data cards before each flight.

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ENTERING COCKPIT

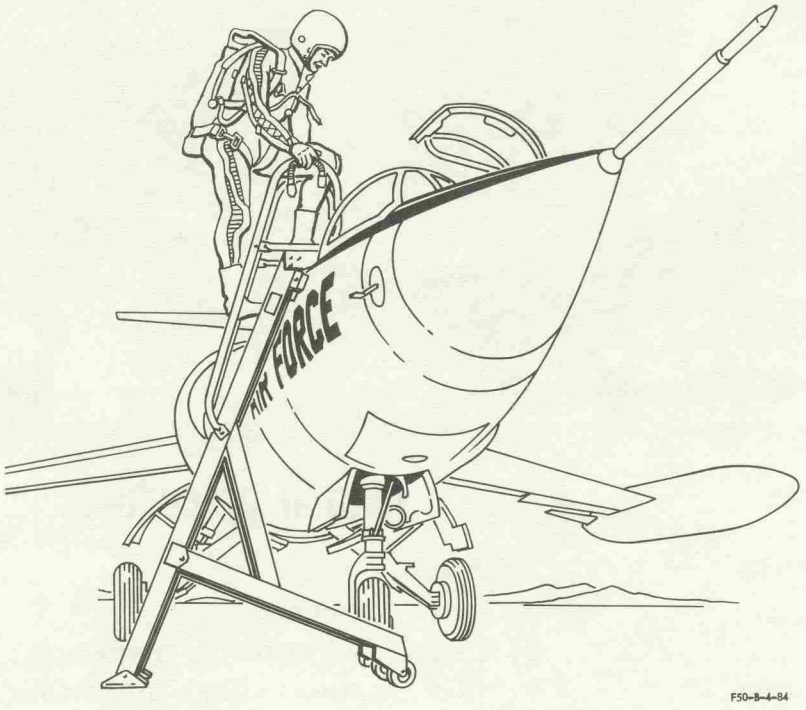


Figure 2-1

PREPARATION FOR FLIGHT CONTINUED

WEIGHT AND BALANCE.

Refer to Section V for weight limitations. For loading information, refer to Handbook of Weight and Balance Data, T.O. 1-1B-40. Before each flight, check the following:

1. Take-off and expected landing gross weight and balance—Check.
2. Weight and balance clearance DD Form 365F.—Check.
3. Make sure total amount of fuel, ammunition, oil, oxygen and special equipment is sufficient for mission to be performed. For servicing points, see Figure 1-30.

ENTRANCE.

The cockpit is entered from the right side of the airplane via a ladder. See Figure 2-1.

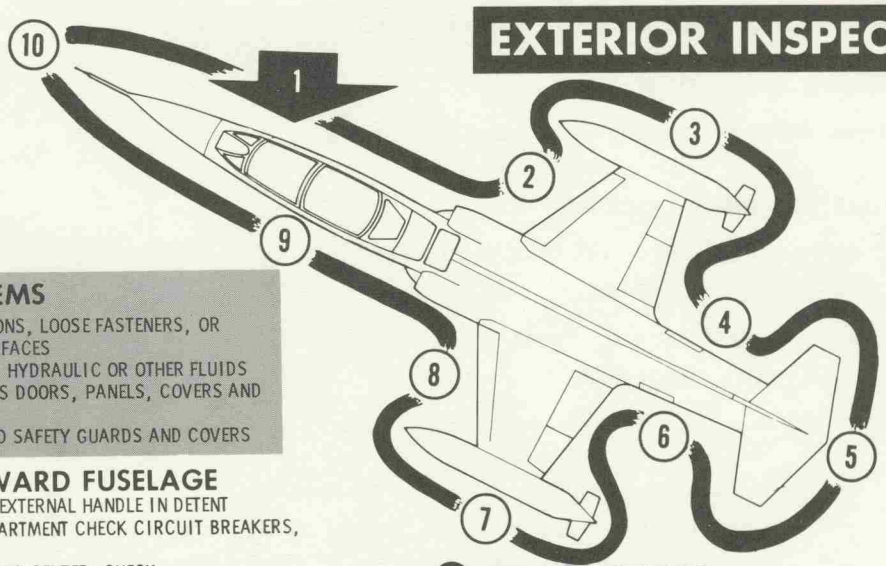
PREFLIGHT CHECK

BEFORE EXTERIOR INSPECTION.

Check DD Form 781 for engineering status, and make sure aircraft has been properly serviced.

CONTINUED ON NEXT PAGE

EXTERIOR INSPECTION



GENERAL ITEMS

- 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

1 RIGHT FORWARD FUSELAGE

- CANOPY UNLOCKED, EXTERNAL HANDLE IN DETENT
- ELECTRONICS COMPARTMENT CHECK CIRCUIT BREAKERS, COVER SECURED
- OPEN ELECTRICAL LOAD CENTER, CHECK CIRCUIT BREAKERS AND SECURE PANEL
- AIR CONDITIONING RAM AIR SCOOP, NO DEBRIS
- RAM AIR TURBINE SECURE, NO LEAKAGE
- ENGINE INTAKE DUCT UNOBSTRUCTED NAVIGATION LIGHTS UNDAMAGED

2 MAIN LANDING GEAR

- GROUND SAFETY PINS REMOVED
- DOWNLOCKS IN PLACE (KNOB ON DRAG STRUT DOWN)
- SHOCK STRUTS EXTENDED (1 1/2" TO 1 3/4")
- LANDING LIGHTS SECURE AND UNDAMAGED
- GROUND-AIR SAFETY SWITCH (LEFT GEAR ONLY) CLEAN, UNDAMAGED
- WHEEL BRAKE LINES SECURE, NO LEAKAGE, SELF ADJUSTER EXPOSED NOT LESS THAN 1/4"
- TIRES-INFLATION, CONDITION AND SLIPPAGE
- WHEEL CHOCKS IN PLACE

3 RIGHT WING

- LEADING EDGE FLAP AND TIP CONDITION, NO CRACKS OR DISTORTIONS
- ATTACHMENT OF EXTERNAL STORES SECURE, GENERAL CONDITION
- AILERON AND TRAILING EDGE FLAP CONDITION, NO CRACKS OR DISTORTIONS
- SURFACE CONDITION, NO CRACKS OR DISTORTIONS, PANELS SECURE
- VISUALLY CHECK RIGHT TIP AND PYLON TANKS, IF INSTALLED FOR FUEL LEVEL AND AMOUNT. IF QUANTITY IS QUESTIONABLE, USE DIP STICK TO DETERMINE THE AMOUNT

4 RIGHT AFT FUSELAGE

- SPEED BRAKE CONDITION, CONNECTIONS, NO LEAKAGE
- NAVIGATION LIGHTS UNDAMAGED
- DRAG CHUTE INSTALLED
- BLOW OUT PANEL SECURE
- VENTRAL FIN FOR ALIGNMENT

5 EMPENNAGE

- VERTICAL AND HORIZONTAL STABILIZERS AND YAW DAMPER SECURE, NO CRACKS OR DISTORTIONS
- EXHAUST NOZZLE FLAP LINKAGES AND SEGMENTS SECURE, NO CRACKS OR DISTORTIONS
- AFTERBURNER SPRAY BARS, FLAMEHOLDER AND LINERS CONDITION, NO CRACKS OR DISTORTIONS

6 LEFT AFT FUSELAGE

- NAVIGATION LIGHTS UNDAMAGED
- SPEED BRAKE CONDITION, CONNECTIONS, NO LEAKAGE
- HYDRAULIC SYSTEM ACCUMULATOR PRESSURES 1000 ± 25 PSI
- HYDRAULIC SYSTEM MANUAL SELECTOR VALVE SAFETIED IN NUMBER 2 POSITION
- HYDRAULIC SYSTEM QUANTITY GAGES AT PROPER LEVEL
- HYDRAULIC ACCESS DOOR SECURE
- ENGINE OIL DIPSTICK (MODIFIED AIRCRAFT) - COVER SECURE, FLAG DOWN

7 LEFT WING

- AILERON AND TRAILING EDGE FLAP CONDITION, NO CRACKS OR DISTORTIONS
- ATTACHMENT OF EXTERNAL STORES SECURE, GENERAL CONDITION
- LEADING EDGE FLAP AND TIP CONDITION, NO CRACKS OR DISTORTIONS
- SURFACE CONDITION, NO CRACKS OR DISTORTIONS PANELS SECURE
- VISUALLY CHECK LEFT TIP AND PYLON TANKS, IF INSTALLED FOR FUEL LEVEL AND AMOUNT. IF QUANTITY IS QUESTIONABLE, USE DIP STICK TO DETERMINE THE AMOUNT

8 LEFT FORWARD FUSELAGE

- FUEL FILLER CAP SECURE
- NAVIGATION LIGHT UNDAMAGED
- ENGINE INTAKE DUCT UNOBSTRUCTED

9 NOSE GEAR

- GROUND SAFETY PIN REMOVED
- SCISSORS PROPERLY CONNECTED
- DOWNLOCK FULLY ENGAGED IN SLOT
- TAXI LIGHT SECURE AND UNBROKEN
- SHOCK STRUT EXTENDED 2"
- TIRE-INFLATION, CONDITION, AND SLIPPAGE
- SEAT EJECTION HATCH PROPERLY SECURED

10 NOSE SECTION

- GUN PORT AND BLAST TUBES GENERAL CONDITION
- INFRA-RED SIGHT WINDOW UNDAMAGED
- RADOME LATCHES SECURE
- PITOT HEAD - SECURE, COVER REMOVED, OPENINGS CLEAN
- PITCH SENSOR VANES FREE TO MOVE, GUARDS REMOVED

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NOTE

DO NOT CONCLUDE THAT TIP TANKS ARE FULL IF FUEL OVERFLOWS WHEN THE OUTER CAP IS REMOVED AND THE INNER CAP IS UNSEATED. IT IS POSSIBLE THAT THE FORWARD COMPARTMENT HAS TRAPPED AIR WHICH CAN BE RELEASED BY HOLDING THE INNER CAP DEPRESSED. THEREFORE, CONTINUE TO HOLD THE INNER CAP DEPRESSED UNTIL IT IS DETERMINED THAT THERE IS NO TRAPPED AIR AND THE TANK IS FULL OR THAT THE TANK REQUIRES TOPPING OFF.

Figure 2-2

PREFLIGHT CHECK CONTINUED

EXTERIOR INSPECTION.

Perform exterior inspection as outlined in Figure 2-2.

BEFORE ENTERING COCKPIT.

1. Canopy—Check for cracks, cleanliness and distortion.
2. Seat free fall linkage—Secure.
3. Manual cable cutter ring—Secure.
4. Ejection hatch release mechanism—Secure.
5. Ejection ring in position and safety pin installed—Check.

Note

Care should be taken to insure that the ejection ring is in the proper position and the pin is properly installed in the ejection ring housing bracket.

6. Shoulder harness straps over upper tubular crossmember of seat—Check.
7. Ejection seat and seat belt initiator hoses—Inspect general condition.

INTERIOR CHECK.

WARNING

Stowage of items such as maps, navigation kits, flying clothes, etc., under the survival kit cushion and under and around the ejection seat can interfere with the normal operation of the escape system. This foreign material, in the event ejection becomes necessary, could restrict separation of the man and kit from the seat, or preclude proper ejection operation. In view of the above, do not stow any foreign equipment in and around ejection seat.

Note

A brief pilot's check list (23, figure 1-8) is located in a pocket on the right side of the cockpit.

1. Landing gear lever—DOWN.
2. Foot retractors—Attach.
3. Seat and rudder pedals—Adjust.
4. Seat belt, shoulder harness, tie-down strap and parachute lanyard anchor—Fasten.

WARNING

Failure to attach the straps in the following proper sequence may prevent separation from the ejection seat after ejection.

a. Place the right and left shoulder harness loops over the manual release end of the swivel link.
b. Place the tie-down strap loop over the manual release end of the swivel link.

CONTINUED ON NEXT PAGE

PREFLIGHT CHECK CONTINUED

c. Place the automatic parachute lanyard anchor over the manual release end of the swivel link.

d. Fasten the seat belt by locking the manual release lever.

5. Parachute D ring lanyard—Hook.

6. Oxygen hoses to face plate and pressure suit, other personal leads—Connect.

WARNING

The survival kits installed in the aircraft have an integrated oxygen hose lead mounted on the right rear of the kit. The hose lead is partly enclosed in a fabric sock. Attached to the fabric sock is a lanyard, approximately six inches in length, which is to be used for fastening the integrated hose leads to the parachute harness during flight. The purpose of the lanyard is to keep the hose leads in place and to prevent flailing during ejection. It has been found that some aircraft have the lanyard fastened around the webbing of the ejection seat. This condition could disconnect the pilot's emergency oxygen during bailout and would create a possibility of hanging up in the seat during seat separation. In view of the above, do not, at any time, attach the integrated oxygen hose stocking lanyard to any thing other than the pilot's parachute harness.

Note

- Connection of personal equipment should be done with the assistance of a personal equipment man.
- If the pressure suit is not used, the personal leads located on the right rear of the survival kit should be removed and stored in the kit.

7. If pressure suit is not used, head set, oxygen mask, G-suit—Connect.

8. MA-2 or MA-1A external power unit—Connected and ON.

Note

"MASTER CAUTION" "INST. ON EMER POWER," "HYD. SYSTEM OUT," "AUTO PITCH OUT" and "No. 1 and No. 2 GENERATOR OUT" lights will illuminate until engine is started and external power is disconnected.

9. Left console circuit breakers—IN.

Note

Gun firing circuit breaker may be positioned as required.

10. Anti-G suit valve—As desired.

11. Auto-pitch control cut-out switch—ON and safetied.

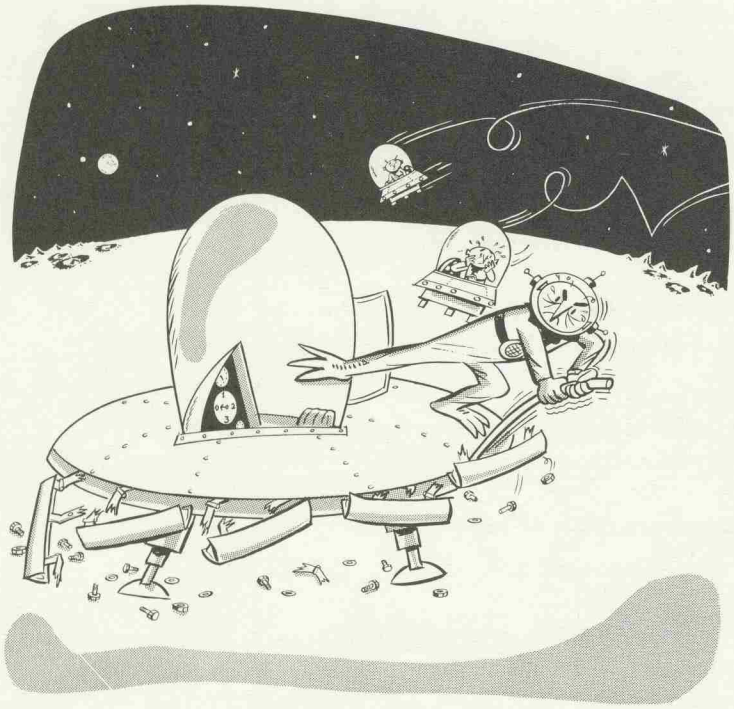
WARNING

If safety wire is broken do not fly aircraft until cleared by maintenance personnel.

CONTINUED ON NEXT PAGE

PREFLIGHT CHECK CONTINUED

12. Stability control switches (roll, pitch, and yaw)—ON.
13. Directional trim rheostat—Centered.
14. Missile firing selector switch—SAFE.
15. Fuel shut-off switch—ON.
16. External stores release selector switch—OFF.
17. External tank fuel selector switch—As required.
18. Auxiliary trim control switch—NEUTRAL.



caution!

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

19. Auxiliary trim selector switch—STICK TRIM.
20. UHF command radio—As required.
21. Wing flap lever—UP (check indicator).
22. Throttle OFF—Check.

CAUTION

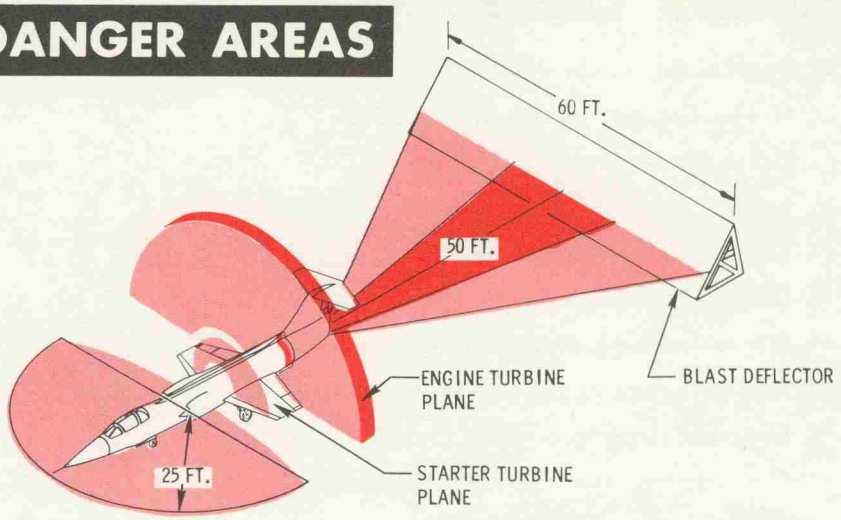
To prevent possible damage to the nozzle control unit do not move throttle when engine is not operating.

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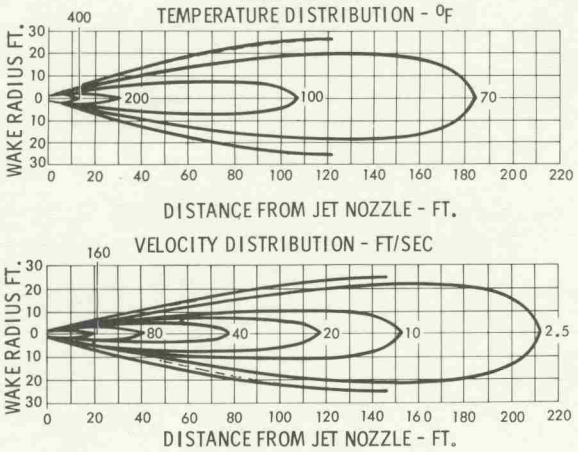
DANGER AREAS

WARNING

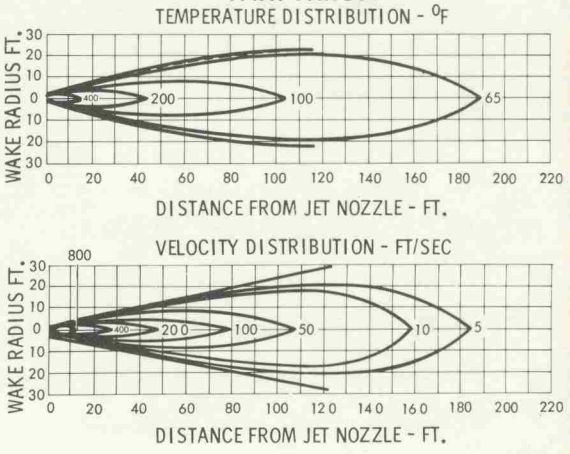
- THE AREA NEAR THE INTAKE DUCTS AND THE EXHAUST IS VERY DANGEROUS - KEEP CLEAR.
- DURING START AND RUNUP AVOID PLANE OF STARTER TURBINE AND ENGINE TURBINE WHEELS.
- DURING RUNUP, ENGINE NOISE CAN CAUSE PERMANENT DAMAGE TO EARS. WITHIN 100 FEET USE EAR PLUGS. WITHIN 50 FEET USE EAR PLUGS AND PROTECTIVE COVERS.
- IF BLAST DEFLECTOR IS NOT AVAILABLE, CLEAR AREA FOR 250 FEET.



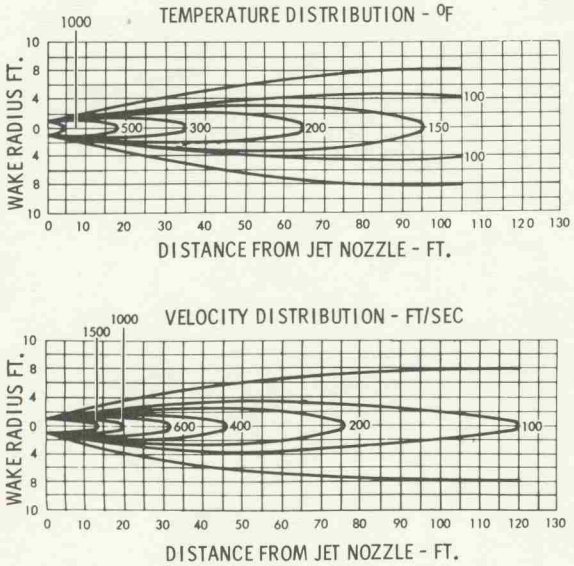
IDLE THRUST



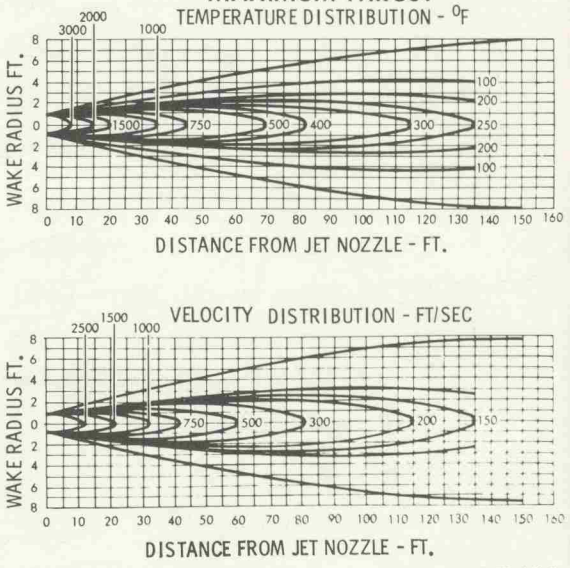
TAXI THRUST



MILITARY THRUST



MAXIMUM THRUST



F-50-4-2-3

Figure 2-3

PREFLIGHT CHECK CONTINUED

23. Speed brake switch—NEUTRAL.
24. Red landing gear warning light—OFF.
25. Green landing gear indicator lights—ON.
26. Landing and taxi lights switch—OFF.
27. Engine anti-ice switch—OFF.
28. Drag chute handle—Stowed.
29. Manual landing gear release handle—Stowed.
30. Ram air turbine extension handle—Stowed.
31. Gun sight target span selector switch—As desired.
32. Gun sight mechanical cage switch—CAGE.
33. Accelerometer—Set.
34. Clock—Check.
35. Airspeed setting index—Set.
36. Altimeter—Set.
37. Radar—STANDBY.
38. Arming switch—OFF.
39. Weapon selector switch—As desired.
40. Gun clearing switch—NORMAL.
41. IR sight switch—OFF.
42. Reticle lights rheostat—Check both lights.
43. Camera shutter selector switch—As desired.
44. Hatch jettison handle—Stowed.
45. Face plate heat rheostat—As required.
46. Hydraulic systems pressure gage selector switch—No. 2.
47. No. 1 and No. 2 generator switches—NEUTRAL (Guards down).
48. Canopy defrosting lever—As required.
49. Fuel quantity and fuel indicating system—Check.
50. Warning light system test switch—WARNING LIGHTS TEST.
51. Oxygen pressure gage—1800 ± 50 psi, approximately 70 psi, if liquid oxygen.
52. Liquid oxygen quantity gage—4 liters minimum.
53. Oxygen system—Check.
    Refer to Oxygen System Preflight Check in Section IV.
54. VHF navigation radio—OFF.
55. S.I.F. panel—As required.
56. IFF master switch—OFF.
57. Ram air scoop lever—CLOSED (lever in last aft detent).

CONTINUED ON NEXT PAGE

PREFLIGHT CHECK CONTINUED

CAUTION

It is recommended that the ram air scoop lever be placed to CLOSED during the preflight check and be kept at CLOSED for all ground operation. This will provide sufficient cooling air for the electronic equipment. If the ram air scoop is opened on the ground, the supply of cooling air to the electronics compartment is shut-off and the electronic equipment may reach overtemperature limits.

58. Engine motoring switch—OFF.
59. Thunderstorm lights switch—OFF.
60. Ventilated suit blower switch—As desired.
61. Interior lights rheostat—As desired.
62. Exterior lights switches—As desired.
63. Cockpit heat rheostat—AUTO (position as desired).
64. Pitch sensor and pitot heat switch—OFF.
65. Rain removal switch—OFF.
66. Directional indicator function selector switch—MAG.
67. Right console circuit breakers—IN.
68. Flashlight, maps, reference books and personal gear—Check.

BEFORE STARTING ENGINE

Before starting engine, make sure that main wheels are securely chocked and that danger areas (figure 2-3) fore and aft of the airplane 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 suction when the engine is started which may be strong enough to draw articles of clothing or loose equipment into the engine. The rotational planes of the turbine wheels (indicated by a red warning stripe) should be clear of personnel. When practicable, start and run up engine on paved surface to minimize the possibility of dirt and foreign objects being drawn into the compressor and damaging the engine. Start engine with airplane heading into the wind whenever possible. When available, an external electrical power source should be connected when starting the engine. Also the special electrical connection between the airplane and the starting air control valve must be made so that the start switches control starting air.

CAUTION

- If the special electrical connection is not made, the pilot has no control over starting air in the event of starter overspeed. Repeated exposure to overspeed conditions (above 45% rpm) will cause starter fatigue and subsequent disintegration of the starter. This can result in serious damage to the airplane.
- The starter is limited to 1 minute of continuous operation, after which 2 minutes must be allowed for cooling before using the starter again.

STARTING ENGINE

Start engine as follows:

1. Ground turbine compressor and special electrical connection—Connected and ON.
   MA-2 or MA-1A ground turbine compressor connected and delivering proper air volume.
2. No. 1 Start switch—START and release.

Note

Successive engine starts should be alternated between ignition systems. This procedure will serve as a check on system operation. Make the first engine start on the No. 1 ignition system and the second start on the No. 2 ignition system.

3. Throttle—IDLE.
   At first indication of engine rotation, advance throttle beyond IDLE detent and then retard to IDLE detent.
4. Fuel flow, 500-800 pounds per hour—Check.

CAUTION

- If fuel flow exceeds 800 pounds per hour, a hot start may result. If fuel flow is less than 500 pounds per hour for ground starts, it may be too low at altitude to accomplish an air start. Therefore the aircraft should be cleared by maintenance personnel before flight.
- Combustion should occur within 10 seconds after fuel flow is established. If no combustion occurs in 10 seconds after fuel flow indication, the engine fails to accelerate to normal idle rpm, or exhaust gas temperature exceeds starting limits, proceed as indicated in Clearing Engine procedure.

5. No. 1 and No. 2 start switches—STOP-START at 45% rpm.
   At 45% rpm, simultaneously move the No. 1 and No. 2 start switches to the STOP-START position and signal ground crew to stop air flow.
6. Engine instruments for proper indications—Check.
7. External electrical power and ground turbine compressor—Disconnect.

CLEARING ENGINE.

To clear the engine of any trapped fuel vapors during ground operation, proceed as follows:

1. Ground turbine compressor and special electrical connection—Connected and ON.
2. Throttle—OFF.
3. No. 1 and No. 2 start switches—STOP-START.
   Simultaneously move No. 1 and No. 2 start switches to the STOP-START position and signal crew to stop air flow.

CONTINUED ON NEXT PAGE

STARTING ENGINE CONTINUED

4. Wait for engine to stop rotating.
5. Engine motoring switch—Hold ON until rpm reaches 15%.

   Simultaneously hold engine motoring switch at the ON position and signal ground crew to deliver air until engine speed reaches 15% rpm.

CAUTION

The starter is limited to 1 minute of continuous operation, after which 2 minutes must be allowed for cooling before using the starter again.

GROUND OPERATION

No engine warm-up is necessary; however, approximately 30 seconds should be allowed for electronic temperature control warm-up.

CAUTION

- If throttle is unintentionally retarded to OFF, a flame-out occurs immediately. Do not reopen throttle, as relight is impossible and resultant flow of unburned fuel into the engine will create a fire hazard.
- Do not fly with partially filled tip tanks until it has been determined that they are pressurized. To check tip tanks pressurization, have sniffle valves depressed while engine is running. Air will escape from the tanks, if they are pressurized.

GENERATOR RESET CHECK.

To insure operation of the generator circuits:

1. No. 1 generator—OFF, check warning light and RESET.
2. No. 2 generator—OFF, check warning light and RESET.

HYDRAULIC SYSTEMS CHECK.

To insure that the hydraulic systems are operating properly, perform the following checks:

1. Throttle—IDLE.
2. Hydraulic systems pressure gage selector switch—No. 2.
3. Speed brakes and flight controls—Check.
   a. Operate speed brakes through a complete cycle. Pressure should drop quickly to approximately 2150 psi then rise momentarily to approximately 3300 psi and return to normal.
   b. Move stabilizer only. Pressure should drop to approximately 2700 psi then rise to 3300 psi maximum and return to normal.
   c. Move ailerons only. Pressure should drop to approximately 2600 psi then rise to 3300 psi maximum and return to normal.
4. Hydraulic systems pressure gage selector switch No. 1 or EMER.

CONTINUED ON NEXT PAGE

GROUND OPERATION CONTINUED

CAUTION

5. Flight controls—Check.
   a. Repeat tests 3b and 3c.
6. Hydraulic systems pressure gage selector switch—No. 2.

AUTO-PITCH, STICK SHAKER, AND STABILITY AUGMENTATION CONTROL SYSTEM CHECK.

With the assistance of ground personnel, proceed as follows:

1. Rotate right vane clockwise until stick shaker operates.
2. Continue clockwise rotation until APC operates at start of red band on APC indicator.
3. Overpower APC system—Check.

   With APC operating, apply an aft force to the control stick to insure the APC system can be manually overpowered. The force required to overpower the APC system should be approximately 35 pounds at the neutral stick position.

WARNING

If the APC system cannot be manually overpowered or excessive aft stick force is required, abort mission and have system checked and cause of malfunction eliminated.

4. Set right vane approximately one degree below the position where the APC operates and sharply lift nose of aircraft until APC operates.
5. As the nose of the aircraft is being "bounced," check operation of the pitch damper by observing the stabilizer for proper movement.
6. Yaw the aircraft by pushing sideways on the nose and observe the yaw damper for proper operation.
7. Wing flap lever—TAKE-OFF.
8. Rotate right vane clockwise until stick shaker operates.
9. Continue clockwise rotation of vane to its stop to check that APC does not operate with flaps extended.
10. Rotate left vane counter-clockwise until stick shaker operates.
11. Have the aircraft rocked at each wing and observe the movement of the ailerons for proper damper operation. (Each wing must be checked separately because of separate damper circuits.)

TRIM SYSTEM CHECK.

CAUTION

It is possible to damage the trim mechanism by operating trim controls with the control stick in a full throw position. To preclude this possibility, make all trim system checks with the control stick in NEUTRAL.

Make the following checks and have ground crew assure you that control surfaces respond correctly:

1. Directional trim rheostat—Operate through full travel and return to neutral.
2. Auxiliary trim selector switch—AUX. TRIM.
3. Auxiliary trim control switch—Test (all four positions).

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MINIMUM TURNING RADIUS AND GROUND CLEARANCES

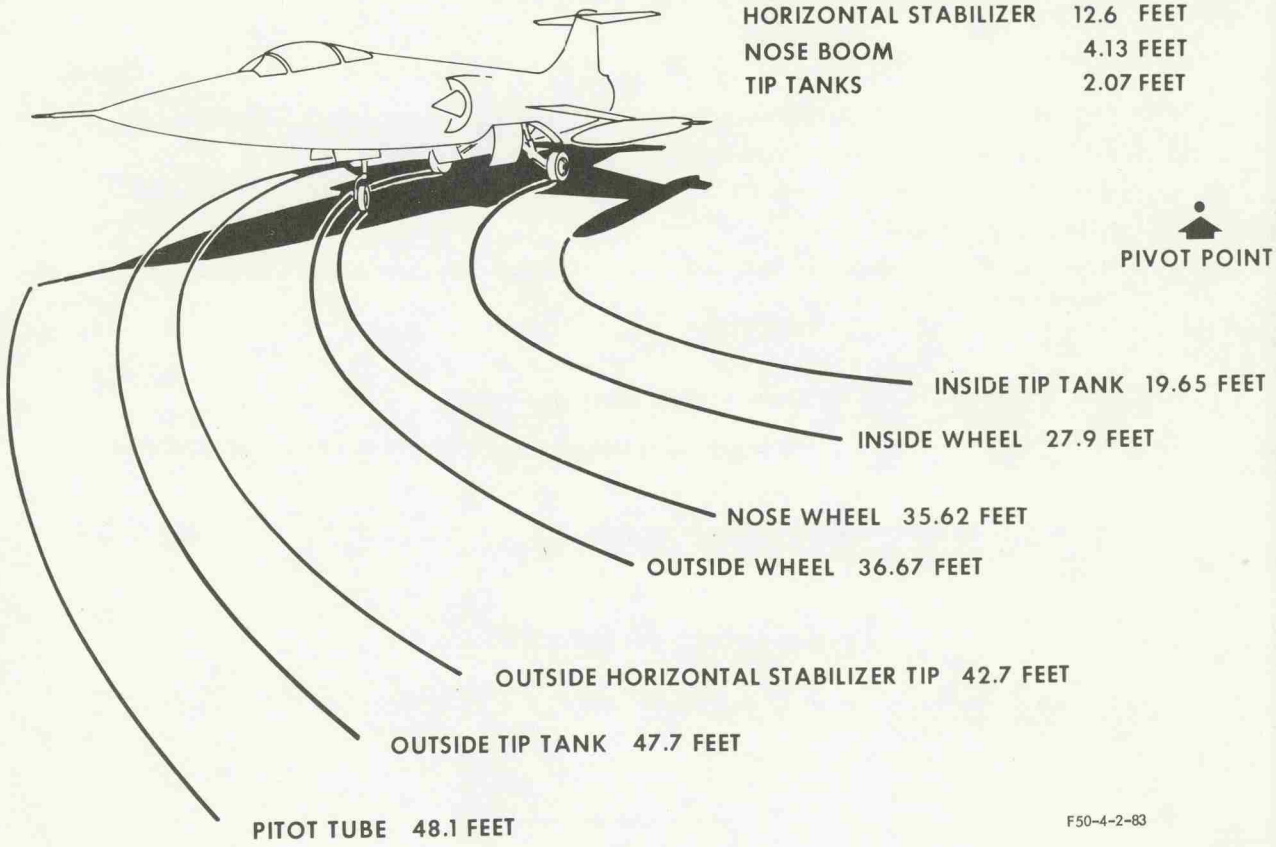


Figure 2-4

GROUND OPERATION CONTINUED

4. Auxiliary trim selector switch—STICK TRIM.
5. Aileron and horizontal stabilizer trim switch—Test (all four positions).

WARNING

An improperly installed or defective trim switch is subject to sticking in any or all of the actuated positions, resulting in application of extreme trim. If this condition occurs during preflight check and the switch does not return automatically to the center OFF position, enter this fact in the AFTO Form 781 with red cross and do not fly the airplane.

CONTINUED ON NEXT PAGE

GROUND OPERATION CONTINUED

Note

Aileron and stabilizer take-off trim indicator lights should momentarily illuminate as the trim motors pass through the take-off setting.

6. Trim—Set for take-off and verified by ground personnel.

Note

Leading edge of horizontal stabilizer should be aligned with black "T" index painted on the vertical stabilizer.

7. Flight controls—Check.
   Operate ailerons, stabilizer and rudder through full travel and check for full freedom of movement.

FUEL AIR PRESSURE CHECK. (Unmodified Aircraft Only)

To ascertain proper operation of the internal fuel air pressure, have ground personnel stationed at one wing tip.

1. Fuel pressure air shut-off test switch—Push and hold.

Note

On modified aircraft this switch is de-activated.

2. When air passage from wing tip port is reported by the ground personnel, release fuel pressure air shut-off test switch.

CAUTION

- If pressurized air fails to emit from the wing tip port, abort mission and have system checked.
- Fuel pressure air shut-off test switch should not be held for more than 15 seconds.

BEFORE TAXIING

Observe the following instructions:

1. IFF master switch—STANDBY.
2. Wing flap lever—TAKE-OFF, check indicator.
3. External stores auto drop system safety pins—Remove.
4. Wheel chocks—Remove.

TAXIING

Note

Engine rpm to initiate taxiing will be approximately 80%.

See figure 2-4 for turning radius.

1. Nose wheel steering—Engage.

   Maintain directional control through nose wheel steering.

Note

The nose wheel and rudder pedals must be in the same relative position before the steering mechanism can be engaged.

2. Maintain a safe distance behind other aircraft.

CAUTION

Objects blown into the intake ducts by other aircraft may cause damage to the engine.

3. Flight indicators—Check.

   Perform operational check of all flight indicators.

BEFORE TAKE-OFF

PRE-FLIGHT AIRPLANE CHECK.

After taxiing to take-off area, complete the following checks:

1. Canopy—Closed and LOCKED.
2. Seat belt—Tightened.
3. Parachute D ring lanyard hooked—CHECK.
4. Inertia reel lock lever—LOCKED.
5. Shoulder harness—Tightened.
6. Oxygen and anti-G suit hoses—Properly connected.
7. Ejection seat ground safety pin—Remove and stow.
8. Speed brake switch—IN.
9. Trim—Set for take-off.
10. Wing flap lever—TAKE-OFF.

PRE-FLIGHT ENGINE CHECK.

Refer to Figure 1-13 for exhaust nozzle positions at various throttle settings and to Figure 5-1 for engine limitations. While in the take-off area make the following checks:

1. Throttle—IDLE, check instruments.
   a. Nozzle position—Approximately ¾ or 7.5.
   b. Tachometer—66% minimum.

CONTINUED ON NEXT PAGE

BEFORE TAKE-OFF CONTINUED

c. Exhaust gas temperature—320-420°C.

d. Oil pressure—12 psi minimum.

CAUTION

- Oil pressure should be noted only after it has stabilized.
- Any time there is a steady state deviation of ±10 psi from normal oil pressure for a given engine at any given rpm, the engine should be inspected.

e. Fuel flow—1000-1300 pounds per hour.

2. Brakes—Pump and hold.

3. Throttle—Military, then reduce to 80% rpm.

   If compressor stall is encountered, abort flight. Make appropriate entry in AFTO Form 781A.

4. Throttle—MILITARY. Check instruments.

   Check acceleration, rpm, exhaust gas temperature, nozzle position, fuel flow and oil pressure.

CAUTION

- The temperature and duration of any operation during which any exhaust gas temperature limit is exceeded, should be entered in AFTO Form 781. If any maximum overtemperature limit is exceeded, turbine bucket replacement is required.
- Should engine speed exceed overspeed limits, either with or without overtemperature conditions, the engine must be rejected for complete overhaul.
- After oil pressure has stabilized check indications against oil pressure record card at Military Power. If the indicated oil pressure varies more than ±10 psi from that listed on the card, the flight should be aborted and an engine inspection performed.

5. External fuel supply—Check.

   If external tanks are installed, check that fuel quantity gage does not show a decreased internal fuel supply. This indicates that fuel used for engine start, taxi and preflight checks is being properly replaced from the external tanks.

TAKE-OFF

Note

The procedures set forth below will produce the results shown in the take-off charts in the Appendix.

NORMAL TAKE-OFF. (Refer to figure 2-5)

1. Align aircraft with the runway.
2. Nose wheel—Centered.
3. Throttle—Military Power.
4. Brakes—Release.
5. Throttle—Maximum Power.

Note

Use either Maximum Power or Military Power on all take-offs. Formation take-offs can be made at near Maximum or near Military Power, but overtaking on the runway during A/B take-off should be controlled with speed brakes.

6. Rpm, exhaust gas temperature and nozzle position—Check.
7. Use nose wheel steering as necessary for directional control until rudder becomes effective at approximately 70 knots.

CAUTION

Nose wheel steering should be disengaged prior to nose wheel lift-off to ensure proper steering clutch release.

8. Nose wheel—Lift off at 155 KIAS.
9. Assume take-off attitude.

Note

- It is recommended that a stabilized afterburner light be obtained prior to advancing throttle to Maximum Afterburner.
- Optimum nose wheel lift-off speed using Military Power is 10 knots below take-off speed for the aircraft configuration.
- Optimum nose wheel lift-off speed using Maximum Power is 20 knots below take-off speed for the aircraft configuration.

MINIMUM RUN TAKE-OFF.

Procedure is the same as for normal take-off with afterburner.

OBSTACLE CLEARANCE TAKE-OFF.

Procedure is the same as for normal take-off with afterburner, refer to Appendix I for distances to clear a 50 foot obstacle.

CROSS-WIND TAKE-OFF.

In addition to the procedures used for normal take-off, increase nose wheel lift-off and take-off speed 5-10 knots to compensate for gusts.

AFTER TAKE-OFF—CLIMB

1. Landing gear lever—UP.

When airplane is definitely airborne, retract gear and check red and green landing gear position indicators lights off.

CONTINUED ON NEXT PAGE

TAKE-OFF

(TYPICAL)
BASED ON A NORMAL
TAKE-OFF GROSS WEIGHT
OF 19,500 LBS.

NOTE

- NECESSARY ENGINE INSTRUMENT CHECKS, WHEN USING THE AFTERBURNER, SHOULD BE MADE DURING THE INITIAL PORTION OF TAKE-OFF ROLL BECAUSE THE AIRPLANE CANNOT BE HELD WITH THE BRAKES.
- REFER TO APPENDIX 1 FOR TAKE-OFF DISTANCES AND SPEED FOR OTHER GROSS WEIGHTS

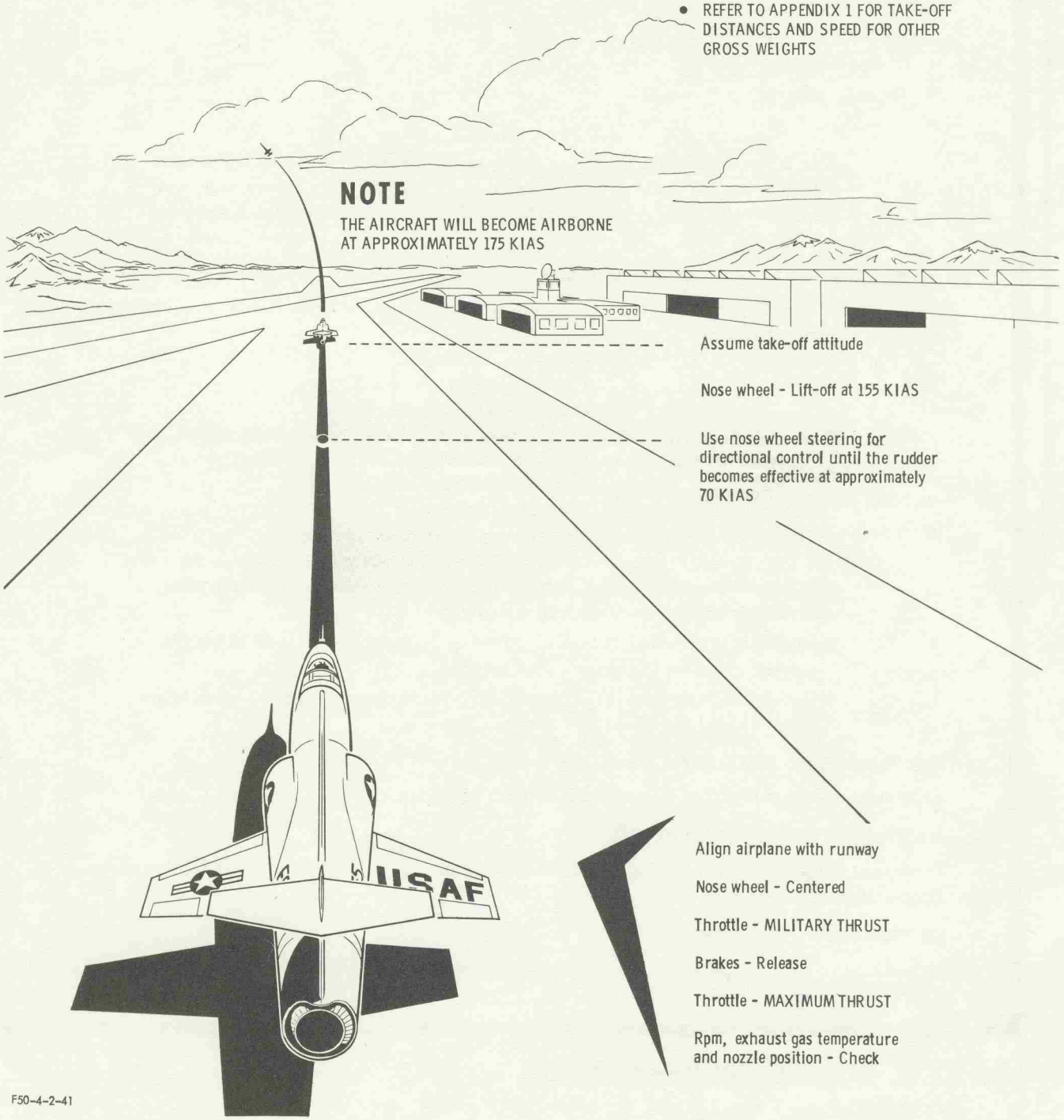


Figure 2-5

AFTER TAKE-OFF—CLIMB CONTINUED

CAUTION

Landing gear and doors should be completely up and locked before landing gear transient limit airspeed is reached; otherwise, excessive airloads may damage the mechanism, or stall gear retraction.

2. Wing flap lever—UP.

   Check indicator.

WARNING

Do not retract wing flaps before reaching 240 KIAS as buffeting will be experienced. Do not exceed wing flap structural airspeed.

Note

- Expect an easily controllable nose up tendency as the flaps retract.
- At transonic speeds, normal airframe stresses could cause the rudder cables and pedals to move. If the pilot's feet tend to restrict the fore and aft motion of the pedals, the effect will be inadvertent tension on the rudder cable. This may cause the rudder lock to become disengaged. Because of this, and because the rudder is not required for coordinated flight at speeds in excess of 300 knots, IAS, it is recommended that the pilot remove his feet from the pedals following take-off and after reaching 300 knots, IAS.

3. Parachute D ring lanyard—Unhook.

   After reaching minimum safe ejection altitude.

4. Airspeed—Best climb.

   Refer to Appendix I for best climb speed. Care should be taken following take-off to anticipate the high forward acceleration. As climb speed is approached assume the proper climb attitude to insure maximum performance.

5. Throttle—As desired.

   As soon as afterburner thrust is no longer needed, shut down the afterburner by moving throttle aft and inboard.

6. Fuel quantity indicator—Check.

   When carrying external tanks, the fuel quantity indicator should be monitored. A low reading indicates that external tanks are empty or are not feeding properly.

7. Auto-pitch and stick shaker system—Check.

   a. 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.

   b. Apply a small rapid stick deflection and note APC indicator 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.

CLIMB

Military Power is recommended for maximum range climb when minimum climbing time to altitude is not important. For a minimum time to altitude such as in a point interception mission, Maximum Power should be used. Refer to climb Charts (Appendix I) for recommended speeds to be used during climb, and for rates of climb and fuel consumption.

CAUTION

The roll stability augmenter should be turned off before reaching 575 knots IAS with wing tip stores installed. With tip stores installed and the roll stability augmenter operating, wing torsional oscillations sufficient to cause structural damage may be experienced at high indicated airspeeds. Missile launchers are not considered as tip stores; therefore, the roll stability augmenter should be left on when carrying bare launchers.

CRUISE

Refer to Appendix I for Cruise Operating Data.

The windshield and canopy defrosting 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 pre-heat because there is insufficient time during rapid descents to heat these areas to temperatures which prevent the formation of frost or fog.

AFTERBURNER OPERATION

Note

- Refer to Figure 1-13 for nozzle positions during afterburner operation.
- Refer to Section V for engine and aircraft limitations.

Before moving the throttle into the afterburner range, check the nozzle position indicator is in its normal range for Military Power. Move the throttle smoothly outboard and forward into the afterburner burning range. Check exhaust gas temperature, rpm and nozzle position.

CAUTION

If an afterburner light is not obtained within 5 seconds at sea level or 15 seconds at altitude, after the throttle is moved into the afterburner range, the throttle should be moved inboard to Military Power and then after 3 to 5 seconds, return to the afterburner range to recycle the system. After the initial light is obtained, move the throttle forward with a positive motion if a maximum thrust is desired.

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AFTERBURNER OPERATION CONTINUED

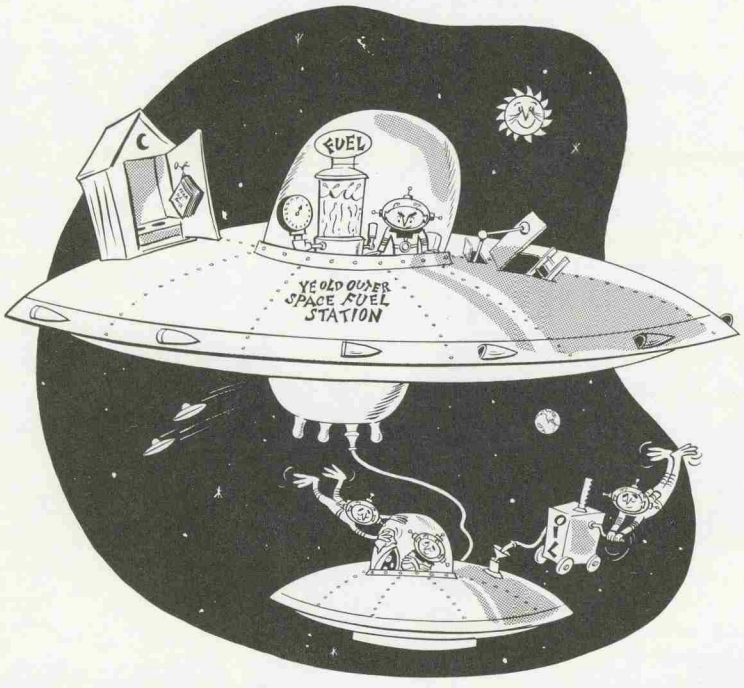
Note

- It is recommended that a stabilized afterburner light be obtained prior to advancing the throttle to Maximum Afterburner.
- The fuel flow indicator does not indicate afterburner fuel flow.

When shutting the afterburner off, retard the throttle aft and inboard to the Military Power position. Monitor the nozzle position indicator to check that the nozzles close normally as the throttle is being retarded from maximum afterburning.

WARNING

At high altitude except in an emergency, maintain an airspeed above 280 KIAS when lighting the afterburner, altering afterburner thrust, or shutting down the afterburner.



caution!

BECAUSE OF THE GREATLY INCREASED FUEL CONSUMPTION, THE AFTERBURNER SHOULD BE USED ONLY WHEN MAXIMUM THRUST IS ESSENTIAL AND SHOULD BE SHUT OFF AS SOON AS THIS NEED NO LONGER EXISTS.

FLIGHT CHARACTERISTICS

Refer to Section VI for information regarding Flight Characteristics.

DESCENT

Refer to Appendix I for recommended descent technique and accomplish the following steps:

1. Throttle—As desired.
2. Speed brake switch—OUT, or as desired.
3. Armament switches—OFF.
4. Gunsight mechanical cage switch—CAGE.
5. Fuel quantity—Check.
6. No. 1 and No. 2 hydraulic system pressures—Check.
7. Seat belt and shoulder harness—Tightened.
8. Shoulder harness inertia reel lock lever—LOCKED.

BEFORE LANDING

Note

- The procedures set forth below will produce the results shown in the landing chart in Appendix I.
- The airspeeds listed herein are based on a landing gross weight of 14,500 pounds.
- The drag induced with LAND flaps and gear down makes it unnecessary to use speed brakes in the landing pattern.

CAUTION

Observe structural limiting airspeeds on wing flaps and landing gear.

INITIAL.

1. Airspeed—300 KIAS.
2. Altitude above runway—1500 feet.
3. Engine speed—Approximately 88% rpm.
4. Parachute D ring lanyard—Hook.

   Before reaching minimum safe ejection altitude.
5. Wing flap lever—TAKE-OFF position at pitch.

DOWNWIND.

1. Landing gear lever—Down below 260 KIAS.
2. Landing gear indicators—Check.

CONTINUED ON NEXT PAGE

BEFORE LANDING CONTINUED

WARNING

On unmodified aircraft the landing gear warning signal will be heard by the pilot whenever the throttle is retarded below approximately 83% and the gear is not down and locked. The high power requirements necessary throughout the landing pattern make this warning system unsuitable. The landing gear position indicators (3 green lights) should be checked prior to landing to insure proper landing gear operation.

3. Altitude above runway—1500 feet.
4. Engine speed—Approximately 88%.

BASE LEG.

1. Wing flap lever—LAND below 240 KIAS.

Note

A mild roll transient may be experienced on some aircraft as flaps move from the TAKE-OFF to the LAND position. This is attributed to asymmetric differences in boundary layer control systems and will vary in intensity and direction with individual aircraft but should not exceed an inch lateral stick displacement. After the flaps are in the full down position some lateral unbalance may persist. Although not objectionable, this unbalance can be trimmed out, if desired.

CAUTION

If a boundary layer control system malfunction is experienced as manifested by a strong rolling moment as the wing flaps travel to the land position, proceed as follows:

a. Immediately return the wing flap lever to TAKE-OFF.

b. Fly final approach at not less than 190 KIAS.

c. Touchdown at 160 KIAS.

2. Engine speed—Approximately 88% rpm.
3. Brakes—Pump.

FINAL.

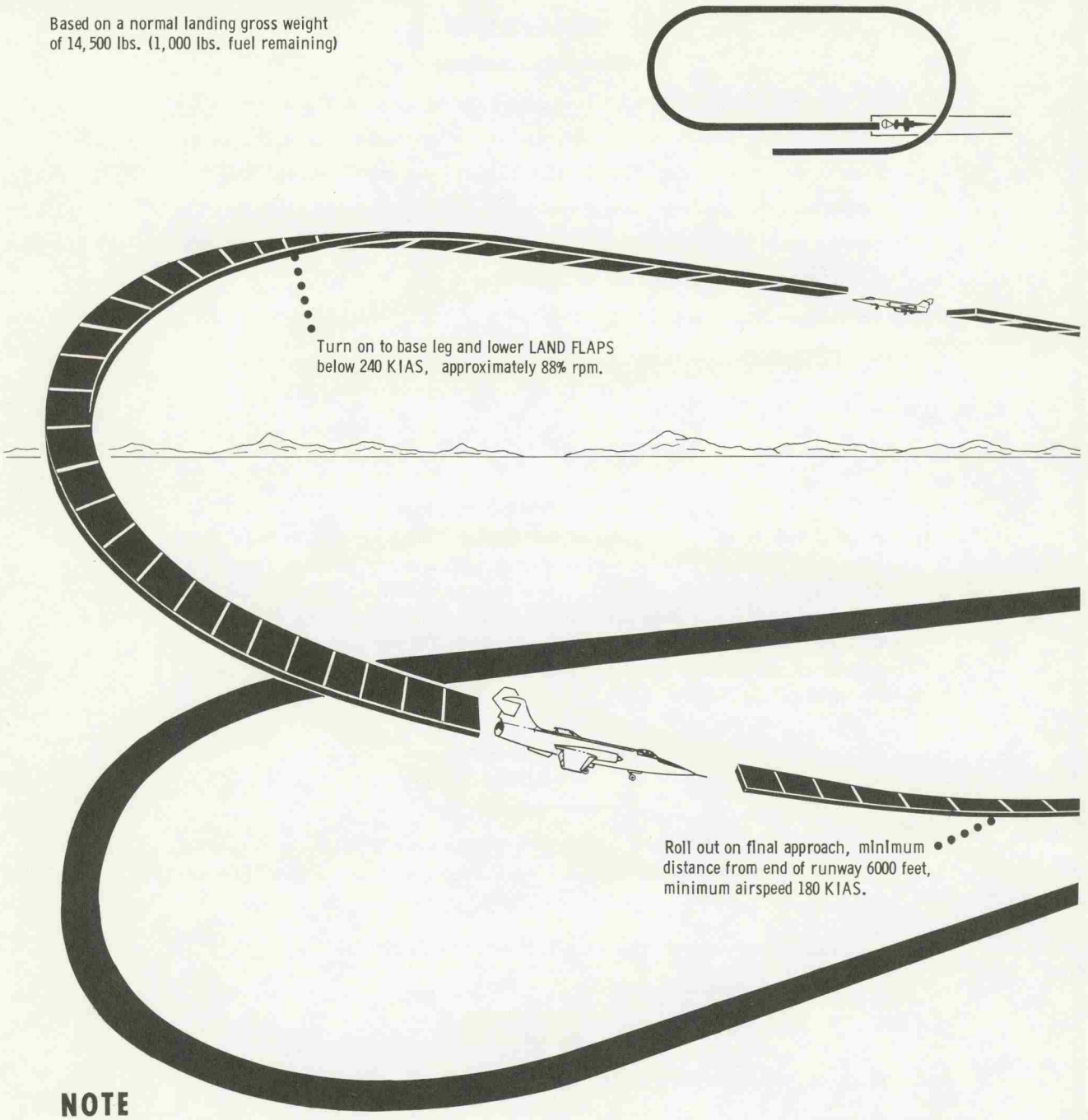
1. Roll out on final approach, minimum distance from end of runway—6000 feet; minimum airspeed—180 KIAS.
2. Engine speed 87-90% rpm.
3. Airspeed—160 KIAS minimum.

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LANDING PATTERN

TYPICAL

Based on a normal landing gross weight of 14,500 lbs. (1,000 lbs. fuel remaining)



NOTE

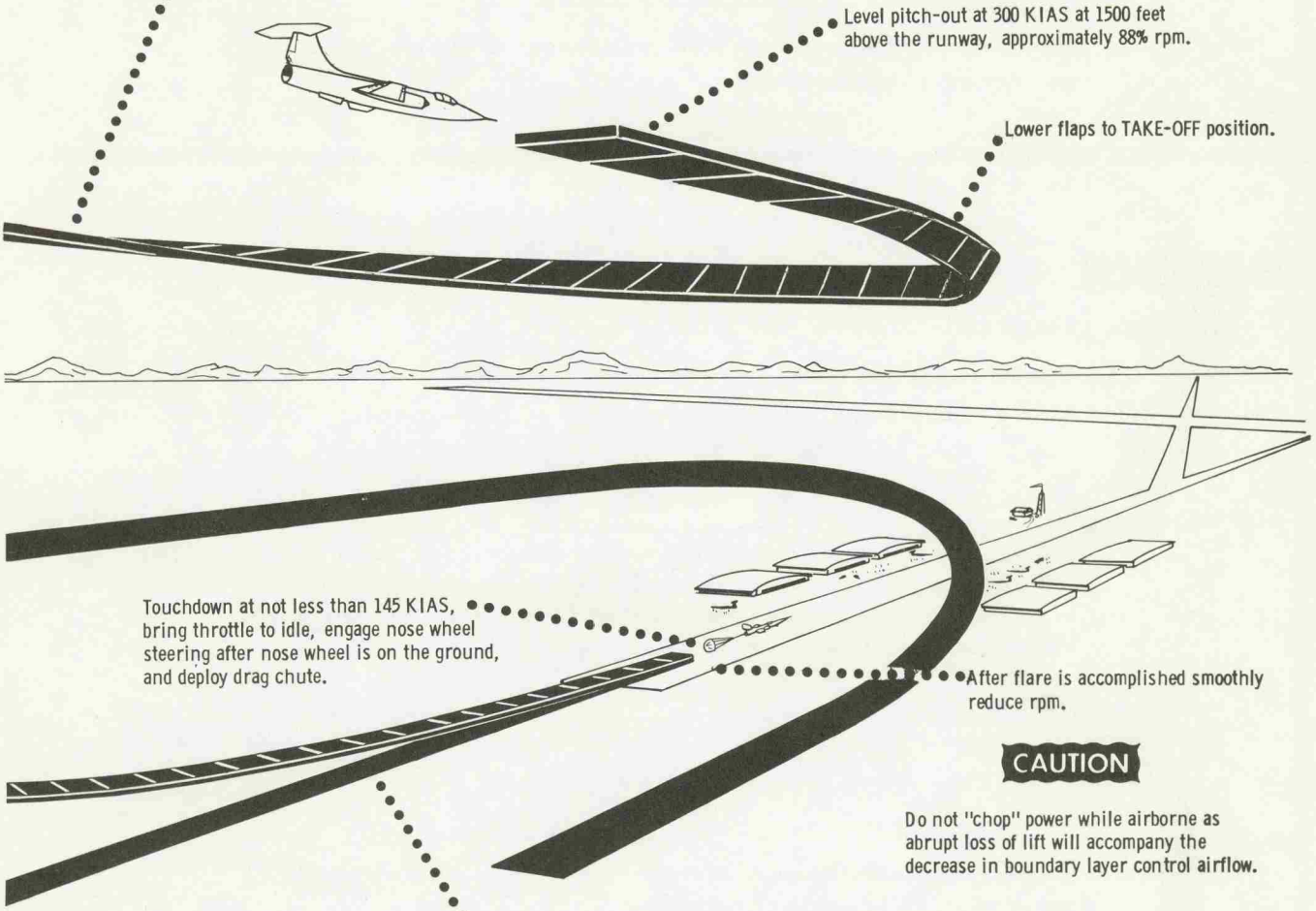
- Increase approach and landing speeds 5 knots for each additional 1,000 lbs. of airplane weight.
- Increase final approach and touchdown speeds 10 knots in turbulent air.
- Refer to landing distance charts in APPENDIX 1 for final approach and touchdown speeds at other gross weights.

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Figure 2-6

NOTE

Avoid an abrupt pitch-out



CAUTION

Do not "chop" power while airborne as abrupt loss of lift will accompany the decrease in boundary layer control airflow.

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/GCA (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.

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BEFORE LANDING CONTINUED

FLARE.

1. After flare is accomplished smoothly, reduce power.

CAUTION

Avoid rapid throttle reductions prior to touchdown as this results in roll-off tendency and a high rate of sink.

LANDING

NORMAL LANDING.

1. Touchdown—At proper airspeed.
   (145 KIAS minimum based on landing gross weight of 14,500 pounds.)

CAUTION

If the aircraft is held off to speeds below those recommended, lateral stability and control will deteriorate and wing drop tendencies will be experienced. In addition, the high pitch angles required for flight at these low speeds will be excessive and can result in tail dragging.

2. Throttle—Retard to IDLE.
3. Nose wheel steering—Engage.
   Engage nose wheel steering as soon as nose wheel is on the ground.

Note

If severe nose wheel shimmy is encountered, release nose wheel steering and attempt to hold weight off nose wheel.

4. Drag chute—Deploy.
   To obtain maximum aerodynamic braking, deploy drag chute as soon as nose wheel is on the ground.

CAUTION

Because the location of the drag chute will cause a nose down pitching moment when deployed, do not deploy the chute until all three gear are on the ground to prevent damage to the aircraft.

Note

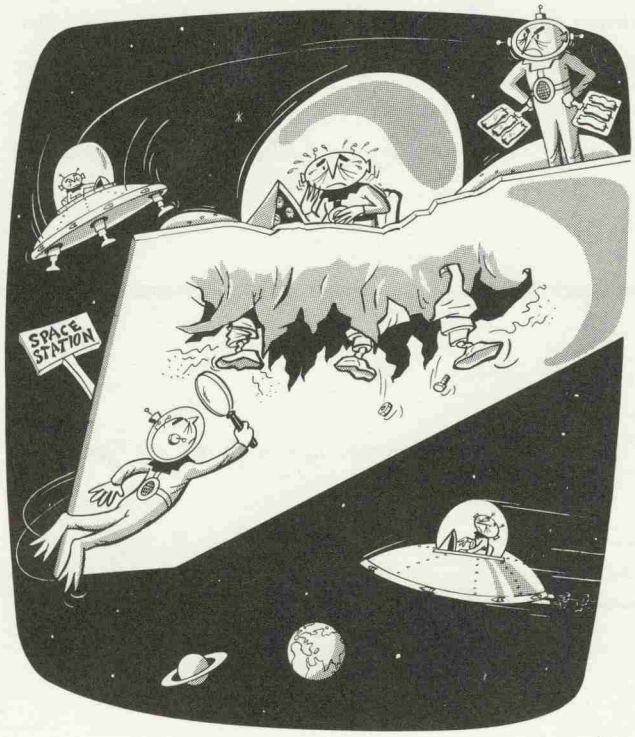
It is recommended that the drag chute be used for all landings. However, it should be standard practice to land so that the aircraft can be stopped on the runway without the chute.

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LANDING CONTINUED

CROSS-WIND LANDING.

Winddrift may be compensated for by "crabbing" or the "wing down" method, or a combination of both. In strong crosswinds the "wing down" or a combination of the two methods is more suitable. The most important things to remember are the following: lower the nose immediately after touchdown, engage the nose wheel steering before deploying the drag chute. The drag chute may be deployed in 90 degree crosswinds of 20 knots, or 45 degree crosswinds of 30 knots provided the nose wheel steering is engaged. The airplane will tend to weather vane, but directional control can be maintained with nose wheel steering. After landing, some difficulty may be encountered releasing the drag chute; however, turning the airplane directly into the wind should solve the problem.



caution!

IF A HARD LANDING IS MADE, THE AIRPLANE SHOULD BE INSPECTED FOR SIGNS OF STRUCTURAL DAMAGE BEFORE THE NEXT FLIGHT.

HEAVY WEIGHT LANDING.

The same technique used for normal landing applies for heavy weight landing (landing made with airplane near maximum take-off gross weight) except for necessary increases in engine power settings. The approach and landing speeds should be increased 5 knots for each 1,000 pounds of airplane weight above normal landing gross weight.

MINIMUM RUN LANDING.

For a minimum run landing, use normal pattern speeds and techniques. Touchdown at 140 KIAS as near the end of the runway as possible and extend the speed brakes. As nose wheel touches, deploy the drag chute and apply heavy braking, but do not skid the tires. Hold heavy braking action until the aircraft stops. Increase approach and landing speeds 5 knots for each 1000 pounds of airplane weight above normal landing gross weight.

CONTINUED ON NEXT PAGE

LANDING CONTINUED

Note

- Stick shaker action can be experienced as the airspeed drops to 140 KIAS.
- Heavy braking is accomplished by applying high brake pedal force as soon as the drag chute deploys. Hold brake pedal force with no pumping action until the aircraft stops. This procedure causes the brakes to heat and is not recommended as normal practice.

LANDING ON SLIPPERY RUNWAYS.

Normal landing can be made on wet or icy runways. Landing rolls will differ with each condition encountered. Use minimum run landing touchdown speed as dictated by the gross weight of aircraft, and land as near to end of the runway as possible. Extend speed brakes on touchdown. Engage nose wheel steering as nose wheel touches down and deploy drag chute. (Refer to Cross-wind Landing in this section for drag chute operation in crosswinds.) Leave wing flaps at LAND during the landing roll and use light to moderate braking as dictated by runway condition.

GO-AROUND

Make decision to go-around as soon as possible and observe the procedures as shown on figure 2-7.

CAUTION

The use of excessive nose-up trim during final approach will appreciably reduce the effect of forward stick. Therefore, as power is advanced, the aircraft should be trimmed toward neutral.

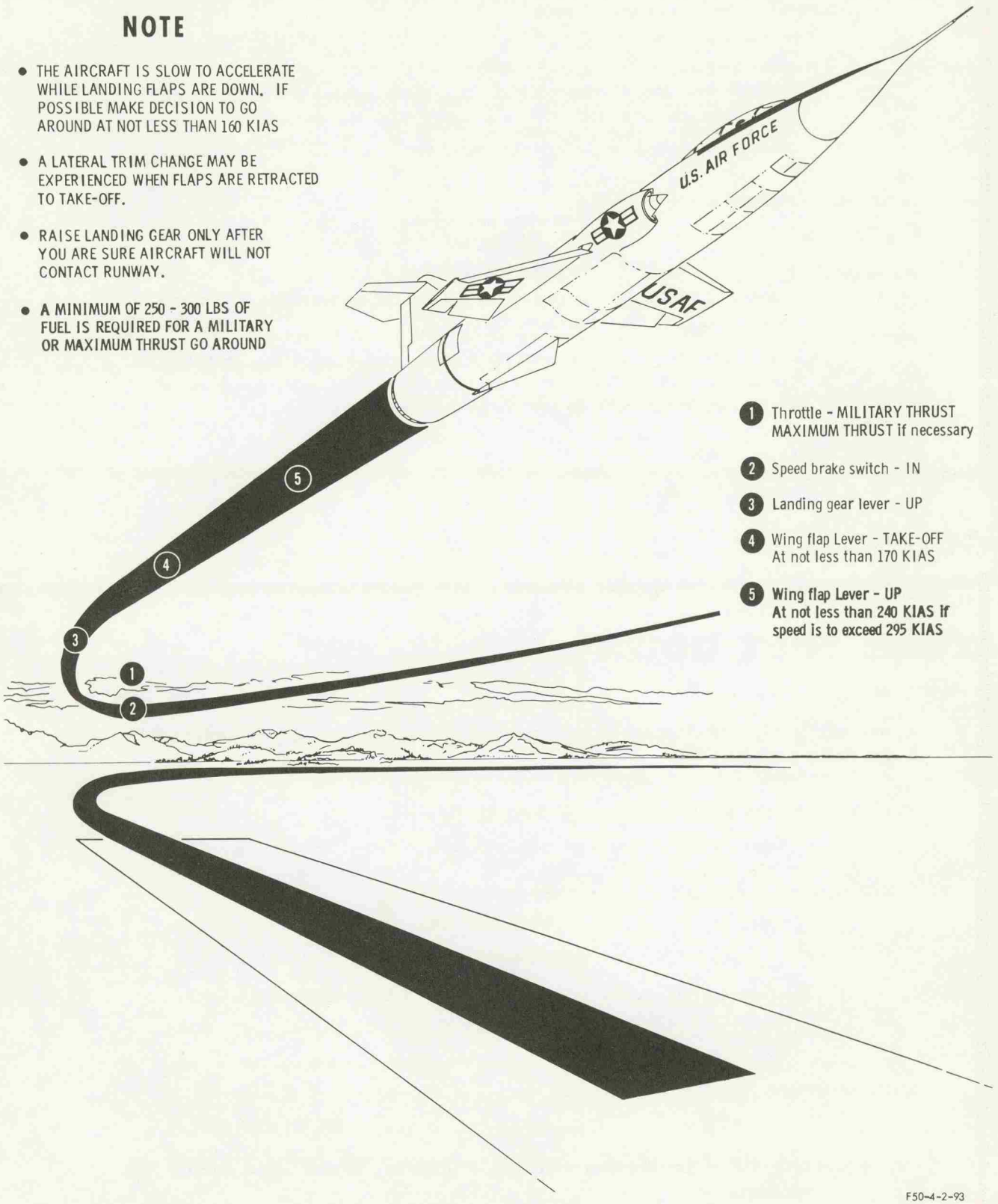
WARNING

On un-modified aircraft the wing flap lever can be moved inadvertently through the TAKE-OFF position to the UP position, if the lever is depressed throughout the range of movement. In the event of a go-around at low altitude or a touch-and-go landing, the results could be hazardous. Be extremely careful to positively position the wing flap lever to the TAKE-OFF position whenever a go-around is necessary or touch-and-go landing are being practiced.

GO AROUND

NOTE

- THE AIRCRAFT IS SLOW TO ACCELERATE WHILE LANDING FLAPS ARE DOWN. IF POSSIBLE MAKE DECISION TO GO AROUND AT NOT LESS THAN 160 KIAS
- A LATERAL TRIM CHANGE MAY BE EXPERIENCED WHEN FLAPS ARE RETRACTED TO TAKE-OFF.
- RAISE LANDING GEAR ONLY AFTER YOU ARE SURE AIRCRAFT WILL NOT CONTACT RUNWAY.
- A MINIMUM OF 250 - 300 LBS OF FUEL IS REQUIRED FOR A MILITARY OR MAXIMUM THRUST GO AROUND



1. Throttle - MILITARY THRUST MAXIMUM THRUST if necessary
2. Speed brake switch - IN
3. Landing gear lever - UP
4. Wing flap Lever - TAKE-OFF At not less than 170 KIAS
5. **Wing flap Lever - UP At not less than 240 KIAS if speed is to exceed 295 KIAS**

Figure 2-7

AFTER LANDING

Maintain directional control with nose wheel steering and brakes and proceed as follows:

1. Drag chute—Jettison in appropriate area.

Note

To obtain best drag chute service life, it is recommended that the chute be jettisoned at the lowest possible taxi speed with the chute still inflated, immediately upon reaching an appropriate area.

CAUTION

- To avoid collapsing and damaging the drag chute, do not make sharp turns while taxiing with the chute deployed.
- Jettison the drag chute before taxiing downwind in winds exceeding approximately 15 knots because of the possibility of collapsing and burning the chute by contact with the hot area of the exhaust nozzle.

2. Speed brake switch—IN.
3. Wing flap lever—TAKE-OFF.
4. Trim—Take-off.

ENGINE SHUT DOWN

1. Wing flap lever—UP.
2. Run engine for three minutes at lowest exhaust gas temperature (approximately 82% rpm) for proper engine cooling.

Note

Operation during taxiing can be considered as part of this time.

3. Rain removal switch—ON for 30 seconds, then OFF.

Note

This procedure is necessary to free the lines of condensation in order to protect the rain removal shut-off valve from corrosion.

4. Throttle—OFF.

Note

Check that engine decelerates freely. Listen for any excessive noise during shut down.

BEFORE LEAVING AIRPLANE

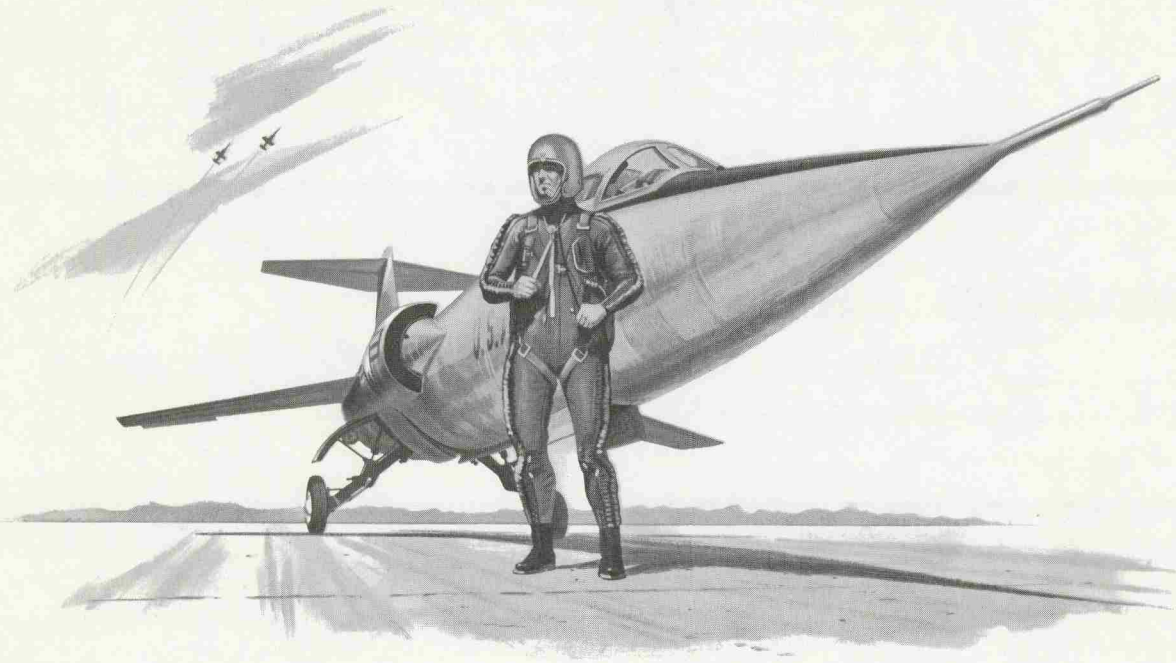
1. Oxygen supply valve lever—OFF.
2. All electrical switches—OFF (except generator switches).
3. Ejection seat ground safety pin—Installed.
4. Wheels—Chocked.
5. Landing gear ground safety pins—Installed.
6. External stores auto drop system safety pins—Installed.
7. AFTO Form 781—Complete.

CAUTION

Make appropriate entries in AFTO Form 781 covering any limits in the Flight Manual that have been exceeded during flight. Entries must also be made when in the pilot's judgment the aircraft has been exposed to unusual or excessive operations such as hard landings; excessive braking action during aborted take-offs; long and fast landings and long taxi runs at high speeds, etc.

CONDENSED CHECK LIST

Refer to pages 2-33 through 2-49 for the Condensed Check List.





F50-8-3-59
AH2831

emergency procedures

SECTION III

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ENGINE FAILURE.

ENGINE FAILURE DURING TAKE-OFF RUN.

If engine failure occurs before airplane leaves ground, and insufficient runway remains for a normal stop, accomplish as much of the following as time permits:

1. Throttle—OFF.

Note

Nose wheel steering is inoperative when the engine is not operating.

2. Drag chute—Deploy.
3. External stores—Jettison.
4. Brakes—Apply.

   Use wheel brakes hard but do not skid.
5. Canopy internal locking lever—UNLOCK.
6. Fuel shut-off switch—OFF.

ENGINE FAILURE DURING TAKE-OFF (AIRPLANE AIRBORNE).

Abandon the aircraft if altitude permits rather than land on an unprepared surface. However, if a safe ejection altitude cannot be attained, accomplish as much of the following as time permits:

1. Throttle—Retard to position consistent with rpm.
2. Start switches—START.
3. External stores—Jettison.
4. Wing flap lever—TAKE-OFF.
5. Ram air turbine extension handle—Pull.

6. If unable to remain airborne, manual landing gear release handle—Pull.

WARNING

Less injury and less aircraft damage results when the landings are made with the gear extended because of the absorption of the initial shock by the extended landing gear.

7. Throttle—OFF.
8. Canopy internal locking lever—UNLOCK.
9. Fuel shut-off switch—OFF.

Note

Land straight ahead, changing course only enough to miss obstacles.

10. Drag chute—Deploy (after touchdown).

ENGINE FAILURE DURING FLIGHT.

If a complete power failure occurs, follow this procedure:

1. Throttle—OFF.
2. Glide speed—275 KIAS.
   Refer to figure 3-1.
3. Air start—Attempt.
   Refer to Engine Air Start in this Section.

WARNING

Extension of the ram air turbine can result in a distortion of engine air flow which will hinder air starts. On engine flame-out, extension of the ram air turbine is not recommended except under extenuating circumstances such as immediate requirements for radio, lighting, or flight instruments. If an air start is made with the ram air turbine extended, engine operation should be monitored closely due to the possibility of flame-out, especially above 25,000 feet.

ENGINE AIR START.

1. Throttle—OFF.
2. Glide speed—275 KIAS.

Note

Percent windmilling rpm at 275 KIAS will be approximately equal to altitude in thousands. For example, at 40,000 feet, windmilling rpm will be approximately 40%.

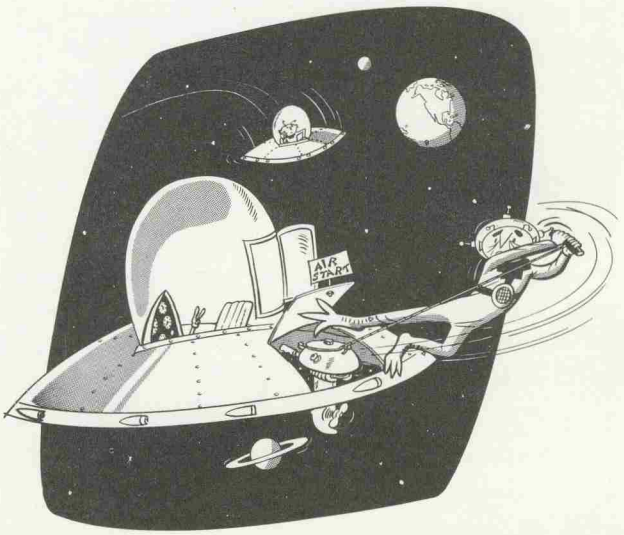
3. Start switches—START.
4. Throttle—IDLE.
   Advance throttle momentarily beyond and then back to IDLE. (Engine should start within 30 to 40 seconds.)
5. If no combustion occurs within 30 to 40 seconds—Repeat steps 1 through 4.
6. If no start is obtained after several attempts—Extend the ram air turbine and repeat steps 1 through 4.

WARNING

If an air start is obtained with the ram air turbine extended, follow the procedure for Precautionary Landing.

7. Rpm and exhaust gas temperature—Stabilize.
8. Throttle—As desired.

warning!



IF ENGINE WILL NOT ACCELERATE ABOVE 70% RPM AFTER AN AIR START, RECYCLE IGNITION, MOVE THROTTLE TO OFF MOMENTARILY, AND RE-ADVANCE TO IDLE.

MAXIMUM GLIDE.

Windmilling Glide.

Maximum glide distance is obtained with gear up, speed brakes in, wing flaps up and maintaining best glide speed. Best glide speeds and distances are shown in Figure 3-1.

Note

- Unless the engine is damaged, the windmilling engine speed will produce sufficient hydraulic pressure to operate the flight control system.
- Extension of the ram air turbine is not recommended except under extenuating circumstances such as immediate requirements for radio, lighting, or flight instruments.

FORCED LANDINGS.

Windmilling Or Frozen Engine.

Because of the many variables encountered, the final decision to attempt a flame-out landing or eject must remain with the pilot. These variables make a quick and accurate decision difficult. Furthermore, it is impossible to establish a pre-determined set of rules and instructions which would provide a ready made decision applicable to all emergencies of this nature because unique circumstances will be associated with each such emergency. However, certain basic conditions, as listed below, must exist before attempting a flame-out landing. Otherwise, ejection is the best course of action.

a. Flame-out landings should only be attempted by pilots who have satisfactorily completed simulated flame-out approaches in this aircraft.

b. Flame-out landings should only be attempted on prepared surfaces that provide at least twice the landing distance normally required.

c. Approaches to the landing field must be unrestricted (clear area of approximately three to five thousand feet in length).

Note

No attempt should be made to land a flamed-out aircraft at any field whose approaches are over heavily populated areas.

d. Weather and terrain conditions must be favorable. Cloud cover, ceiling, visibility, turbulence, surface wind, etc., must not impede in any manner the establishment of a proper flame-out landing pattern.

Note

Night flame-out landings or flame-out landings under poor lighting conditions such as at dusk or dawn should not be contemplated regardless of weather or field lighting.

e. Flame-out landings should only be attempted when either the "High Key" or "Low Key" positions can be achieved.

Note

In any event, the pilot should eject at the "Low Key" position if the approach pattern has, in any way, been unsatisfactory, and the pilot doubts that the landing can be carried to a successful completion.

All of the above basic conditions combined with the pilot's analysis of the condition of the aircraft, type of emergency, and his proficiency are of prime importance in determining whether to attempt a flame-out landing or to eject. If the decision is made to eject, the pilot should attempt to turn the aircraft toward an unpopulated area, if possible prior to ejection. The recommended procedures for making a forced landing are set forth in figure 3-2.

Simulated Forced Landing.

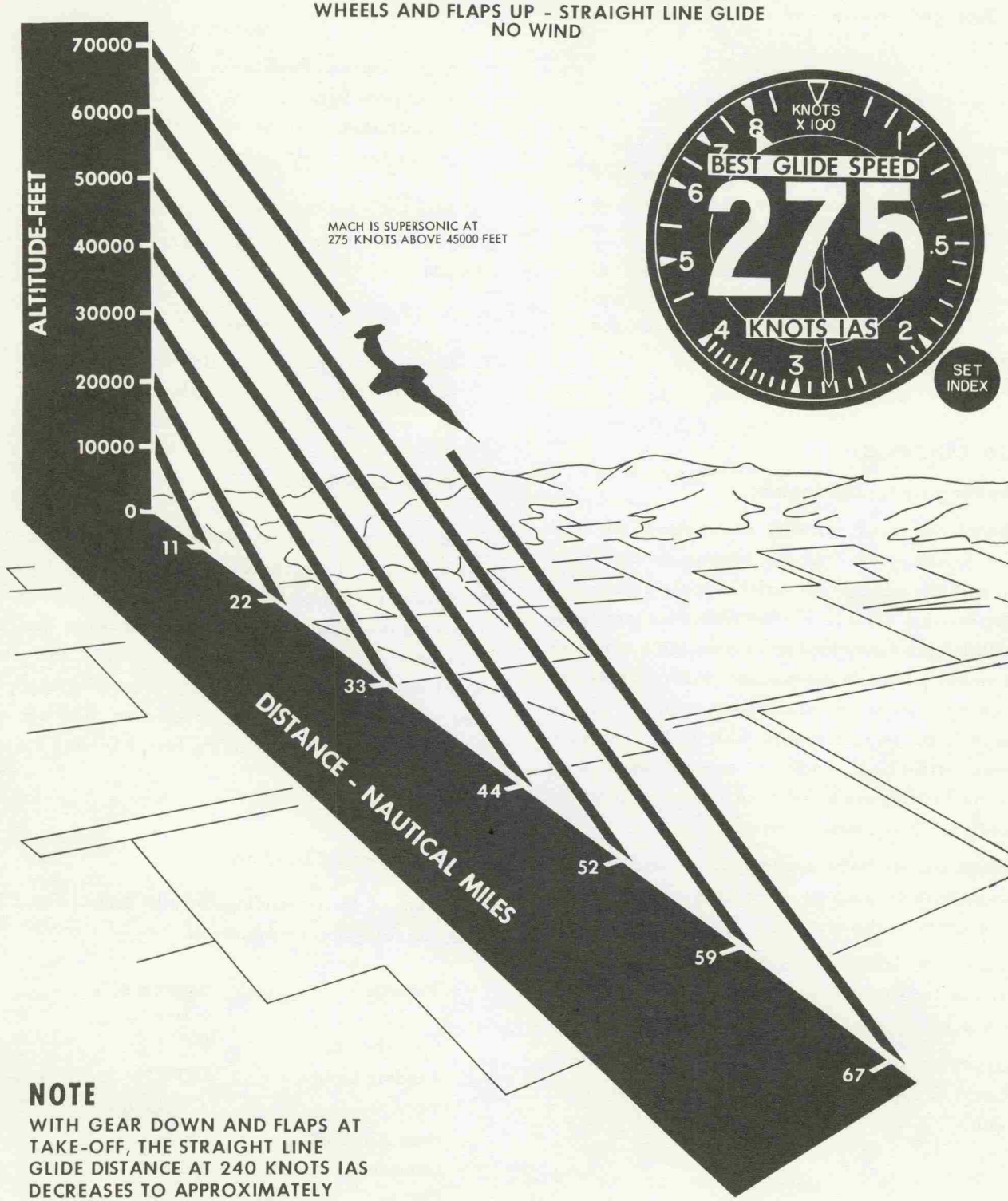
Simulation of forced landings may be accomplished by using the following configuration.

Throttle	Approx. 80% rpm
Speed brakes	IN
Landing Gear	DOWN
Flaps	TAKE-OFF
Ram Air Turbine	IN
Indicated Airspeed	240 Kts.

It is recommended that practice dead engine descents and landings be made using the above configuration so that the additional safety of having the landing gear extended

MAXIMUM GLIDE DISTANCE

NO EXTERNAL STORES - ENGINE WINDMILLING
SPEED BRAKES IN - RAM AIR TURBINE OUT -
WHEELS AND FLAPS UP - STRAIGHT LINE GLIDE
NO WIND



NOTE

WITH GEAR DOWN AND FLAPS AT TAKE-OFF, THE STRAIGHT LINE GLIDE DISTANCE AT 240 KNOTS IAS DECREASES TO APPROXIMATELY 3 MILES FOR EACH 5000 FEET OF ALTITUDE

F50-4-2-40

Figure 3-1

during the landing flare can be realized. Simulation of extension of the landing gear during the flare may be made by opening the speed brakes.

Precautionary Landing With Defective Engine Or With Ram Air Turbine Extended.

Whenever the pilot believes that the engine is defective or may flame out as a result of extending the ram air turbine, the safest landing procedure is as follows:

1. Wing flap lever—TAKE-OFF.
2. Throttle—IDLE.
3. Proceed with Forced Landing Procedure (refer to figure 3-2) except that the landing gear should be extended using the normal system.
4. Ram air turbine—Extend if the engine should fail in this pattern.
5. Continue with Forced Landing procedures.

FIRE.

ENGINE FIRE DURING START.

If the fire warning lights come on or there are other indications of fire during engine start, follow this procedure:

1. Throttle—OFF.
2. Start switches—STOP-START.
3. If fire warning lights go out or fire indications cease —Clear engine.

   Refer to Clearing Engine in Section II.
4. Leave airplane.

ENGINE FIRE DURING TAKE-OFF.

Illumination of the fire warning lights during take-off requires immediate action. The exact procedure to follow varies with each set of circumstances and depends upon altitude, airspeed, length of runway and overrun clearing remaining, location of populated areas, etc. To help you make a decision, the following procedures are presented.

Engine Fire Before Lift-Off. If the fire warning lights come on during ground roll, and sufficient runway or overrun area is available to abort the take-off:

1. Throttle—OFF.
2. Drag chute—Deploy.
3. External stores—Jettison.
4. Wheel brakes—Apply.
5. Fuel shut-off switch—OFF.
6. Canopy—Unlocked.
7. Leave airplane as soon as it stops.

Engine Fire After Lift-Off. If fire warning lights come on after the airplane is airborne, and sufficient runway and overrun are not available to land and stop the aircraft successfully, the following is recommended if altitude is too low for safe ejection:

1. Maintain power.
2. External stores—Jettison.
3. Maximum climb.

   Zoom to a safe ejection altitude.
4. Throttle—Adjust to maintain safe ejection altitude.

   See figures 3-3 and 3-5 for minimum safe ejection altitudes.
5. Check for fire.

   Confirm fire by abnormal instrument readings, smoke in cockpit, report from ground or other aircraft.
6. If fire exists—Eject.
7. If fire cannot be confirmed, make decision to land or eject.

ENGINE FIRE DURING FLIGHT.

If fire warning lights come on during flight, proceed as follows:

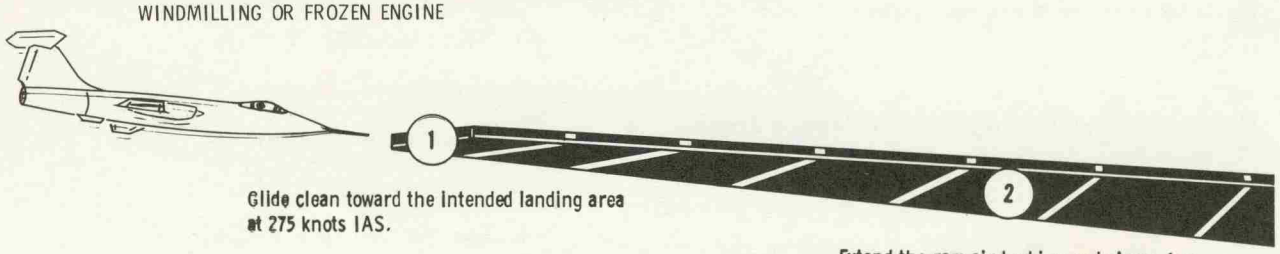
1. Throttle—IDLE, immediately.
2. If lights go out, activate test switch.

   Place switch to WARNING LIGHT TEST position to determine whether the circuit has burned through.
3. If the lights illuminate when tested, land immediately at reduced power.
4. If lights do not illuminate when tested—Eject.

   If lights do not illuminate when tested, it indicates that the fire warning circuit has burned through.

FORCED LANDING PATTERN

WINDMILLING OR FROZEN ENGINE



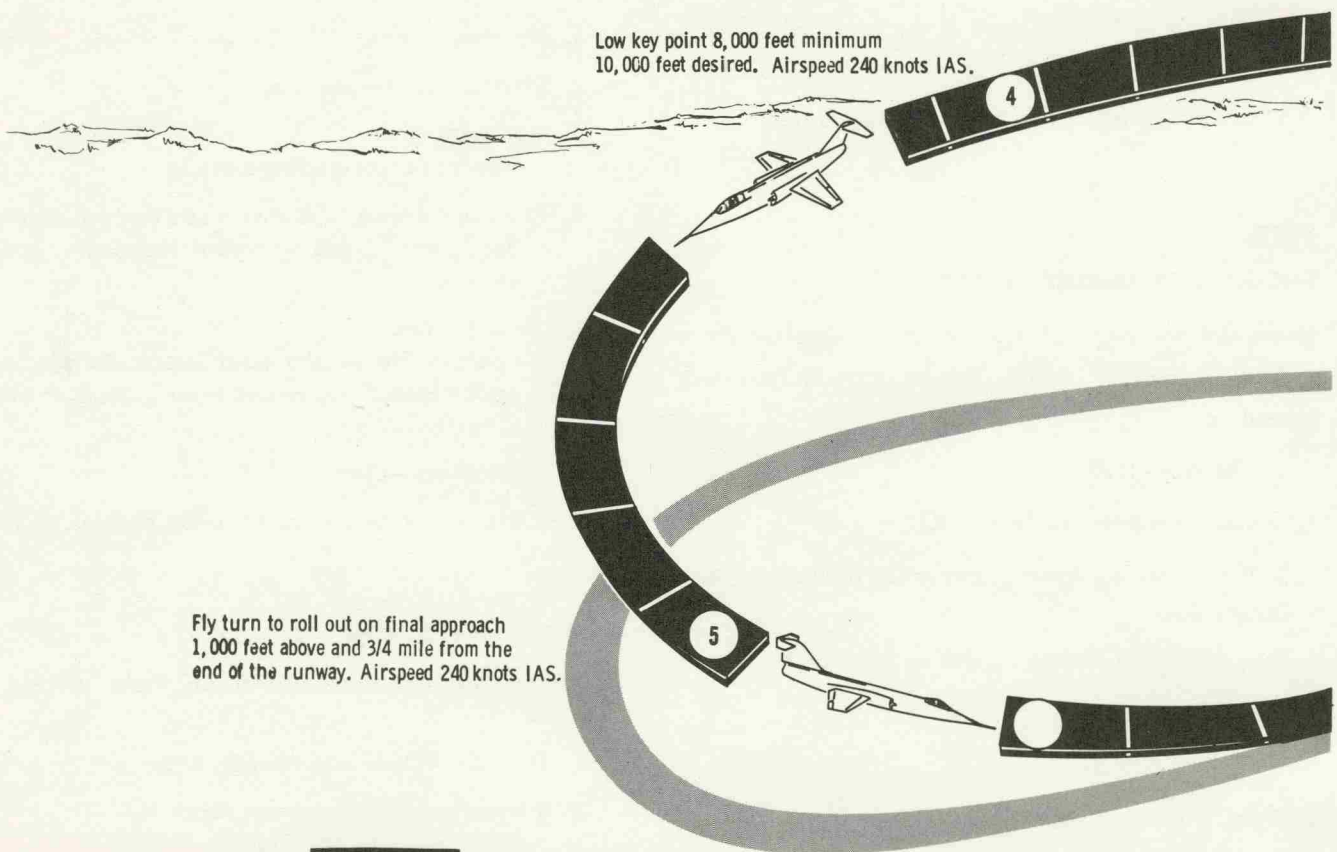
Glide clean toward the intended landing area at 275 knots IAS.

Extend the ram air turbine and place wing flap lever to TAKE-OFF. Reduce speed to 240 knots IAS.

WARNING

Place wing flap lever in TAKE-OFF position only. Do not select LAND or return lever to UP position as this may stall the ram air turbine.

Low key point 8,000 feet minimum
10,000 feet desired. Airspeed 240 knots IAS.



Fly turn to roll out on final approach 1,000 feet above and 3/4 mile from the end of the runway. Airspeed 240 knots IAS.

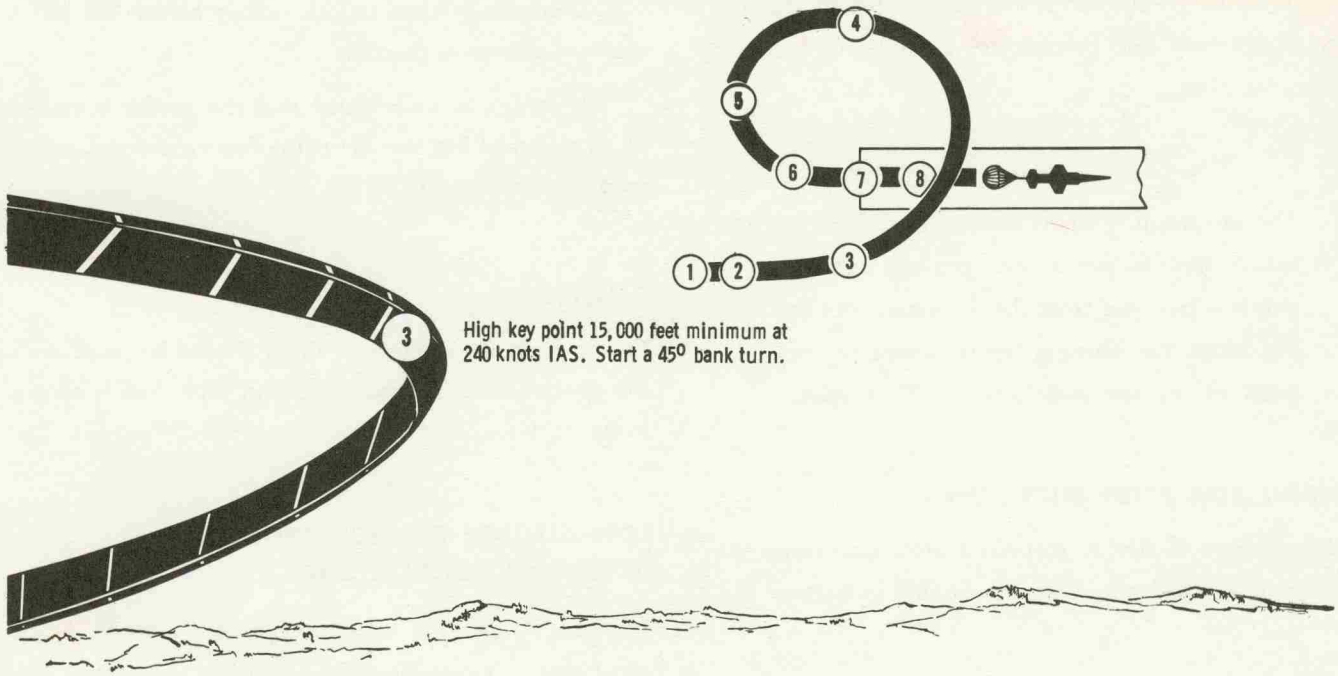
Start flare 500-700 feet above ground. When flare is assured, pull the manual landing gear release handle.

WARNING

It is imperative that the landing gear remain retracted until after the flare has been started otherwise the rate of descent will be increased from approximately 7,000 feet to approximately 11,000 feet per minute and the altitude and speed loss required for flare will be increased to the point that a safe dead-stick landing will be extremely difficult to perform.

It has been found that after leaving the low key point the airspeed should be monitored closely and held as close as possible to the recommended 240 knots. In any case the airspeed should not be allowed to drop below 220 knots. Attempts to regain speed will cause the rate of descent to increase to the point where the flare cannot be accomplished.

Figure 3-2



High key point 15,000 feet minimum at 240 knots IAS. Start a 45° bank turn.

After touchdown, lower the nose wheel, deploy the drag chute and use brakes as necessary to stop.

Complete flare at 190-200 knots IAS. Airplane may be set down when gear indicates down and locked or held off if necessary to 165 knots, the minimum recommended touchdown speed for dead engine condition.

NOTE

Gear will extend in 4 - 5 seconds and the airplane can be held off the ground for approximately 10 seconds following release of gear.

5. If lights do not go out, throttle—OFF.

6. If lights go out and it can be determined that the fire is definitely out, prepare for a flame-out landing; otherwise, eject.

Note

The fire warning lights are powered from the battery bus. However, the warning light test switch is powered from the d.c. emergency bus. Therefore, the warning lights cannot be energized by the test switch below IDLE rpm.

ENGINE FIRE AFTER SHUT-DOWN.

Clear engine if fire is suspected after shut-down on the ground. (Refer to Clearing Engine in Section II.)

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. Generator switches—OFF.

2. All electrical accessory switches—OFF.

3. Operate only those units necessary for safe flight and landing, by placing that unit on and resetting the generators.

4. Return generator switches to OFF position when operation is completed.

5. Land as soon as possible.

ELIMINATION OF SMOKE OR FUMES.

Should smoke or fumes enter the cockpit, proceed as follows:

1. If using A-13A oxygen mask, diluter lever—100%.

2. Emergency lever—EMERGENCY.

3. Descend to 25,000 feet, or lower, if circumstances permit.

4. Ram air scoop lever—OPEN.

5. Generator switches—OFF.

6. If smoke persists, unlock canopy below 260 KIAS and land as soon as possible.

7. If it can be determined that the smoke is caused by an electrical fire, use Electrical Fire procedures, otherwise, reset generators.

EJECTION.

Escape from the aircraft in flight should be made with the ejection seat. The basic ejection procedure is shown in figure 3-3.

LOW-ALTITUDE EJECTION WITH THE "ONE AND ZERO" SYSTEM.

In order to provide an improved low altitude escape capability, a system incorporating a one-second seat belt delay and a zero-second parachute delay ("one and zero" system) is provided for ejection seat escape systems. This system makes use of a detachable lanyard that connects the parachute timer knob to the parachute D-ring. At very low altitudes and airspeeds this D-ring lanyard must be connected, thus providing parachute actuation immediately after separation of the pilot from the ejection seat. At other altitudes and airspeeds, the lanyard MUST BE DISCONNECTED from the D-ring, thus allowing the parachute timer to actuate the parachute below the critical parachute opening speed and below the parachute timer altitude setting. A ring attached to the parachute harness is provided for stowage of the lanyard hook when it is not connected to the parachute D-ring. THIS "HOOK UP" and "UNHOOK" OF THE LANYARD MUST BE DONE MANUALLY BY THE PILOT. An illustration of the hook (figure 3-4) depicting the *hooked* and *unhooked* conditions is provided for information purposes only. The lanyard configuration is one of several which will be in service use. Although each type lanyard differs in appearance, the hook and attaching positions are similar. *The following procedure is mandatory for use with the one-and-zero escape system:*

a. Before take-off the lanyard must be hooked up to the D-ring. A check should be made as part of the pilot's cockpit check sequence prior to take-off, to insure that the lanyard is connected.

b. After take-off, the lanyard must be unhooked and stowed by the pilot, after passing through the minimum safe ejection altitude for his particular escape system. This minimum safe ejection altitude must be determined by the pilot from the chart shown in figure 3-5.

WARNING

The lanyard must be disconnected whenever operating at high altitudes or airspeeds in order that the safety-delay provided by the parachute timer-aneroid will not be overridden.

c. Before landing, the pilot must hook up the lanyard prior to reaching the minimum safe ejection altitude for his particular escape system.

d. When the pilot's flight duties necessitate hooking and unhooking of the lanyard at altitudes higher than those specified in the preceding paragraphs, it should be noted that with the one-and-zero system the safe ejection speeds for the various parachute configurations are as shown in figure 3-6.

e. After landing it is not necessary to disconnect the D-ring lanyard, since it connects the D-ring to the timer knob and is not attached to the seat belt. The parachute may be removed from the aircraft with the lanyard in the hooked up condition.

EJECTION ATTITUDES.

On ejection, the seat and the pilot will have a component of thrust provided by the aircraft. Therefore, if seat ejection is initiated when the aircraft is in a climb, the ejected pilot will reach a higher altitude than if he were ejected from an aircraft in level flight—the opposite will be true if ejection is accomplished when the aircraft is in a nose down attitude. Accordingly, in marginal altitude ejections, the aircraft should be "zoomed" and ejection initiated while the aircraft is in a climbing attitude.

WARNING

In most cases no advantage will be gained by rolling the aircraft to an inverted attitude for ejection. If airspeed is low, the altitude lost during the roll may exceed any altitude advantage intended to be gained due to the direction of ejection.

EJECTION ALTITUDES.

For low altitude ejections (below 2000 feet) the technique which results in the highest possible parachute deployment is to pull the aircraft nose above the horizon (zoom-up maneuver) before ejection resulting in ejection seat trajectory closer to the horizontal. The "zoom-up maneuver" will result in an increase in altitude and time available for separation from the seat and parachute deployment. Figure 3-5 indicates the minimum altitude for successful ejection with different combinations of automatic ejection equipment. These figures are applicable for LEVEL FLIGHT attitude. The data is conservative for climb attitudes and optimistic for dive attitudes.

WARNING

Minimum safe ejection altitudes quoted herein were determined through an extensive series of flight tests and are based on distance above terrain on initiation of seat ejection (initiation of seat ejection is defined as the time the seat is fired). However, human error and equipment malfunctions were not considered in the determination of these altitudes. Therefore, whenever possible, ejection should be initiated at altitudes higher than the minimums shown in figure 3-5.

The "one-and-zero" system has been successfully flight tested at speeds between 120 KIAS and the maximum safe parachute opening speeds. Therefore, data for this system is applicable between 120 KIAS and the maximum safe ejection speeds for the "one-and-zero" system as shown in figure 3-6. The following equipment combinations are applicable:

a. M12 (1 second) automatic seat belt delay initiator.

b. M4 seat catapult.

c. B-4 or B-5 pack/C-9 canopy or B-5 pack/C-11 canopy automatic parachute with a 1 second F-1B timer or a 2 second F-1A timer.

Note

- Style of automatic parachute and type of canopy, pack and automatic release is defined in T.O. 14D1-1-1.
- The type of automatic seat belt used does not affect the determination of the minimum altitude.

CAUTION

For non-automatic parachutes used with automatic seat belts, lanyard part No. 57C6200, will be used. The minimum safe escape altitudes specified for the 1-second seat belt and 0-second parachute setting apply when the lanyard is attached to the rip-cord and seat belt.

Once the equipment is known, the minimum altitude for safe ejection can be found on figure 3-5. For example, if the B-5 pack/C-11 canopy with the 2 second F-1A timer is used, the sequence is a 1 and 2 system and the minimum ejection altitude under level flight conditions is 700 feet. With the D-ring lanyard attached, the minimum altitude is 450 feet. The altitudes given in figure 3-5 should be used only as a guide. Once a minimum altitude has been determined for a particular configuration of equipment, the decision as to when to eject

EJECTION PROCEDURE

BEFORE EJECTION (IF TIME AND CONDITIONS PERMIT)

1. REDUCE SPEED AS MUCH AS POSSIBLE
2. HEAD AIRCRAFT TOWARD UNPOPULATED AREA
3. GIVE LOCATION AND INTENTIONS TO NEAREST RADIO FACILITY AND TURN IFF TO EMERGENCY
4. STOW ALL LOOSE EQUIPMENT
5. ACTUATE BAIL OUT BOTTLE IF USING A-13A MASK

EJECTION

1. BRING HEAD BACK TO HEADREST
2. PULL EJECTION RING WITH BOTH HANDS



NOTE

- AT HIGH AIRSPEEDS AND ALTITUDES HOLD EJECTION RING UNTIL SPEED DECREASES TO KEEP ARMS FROM FLAILING
- AT LOW ALTITUDES RELEASE EJECTION RING IMMEDIATELY AFTER CLEARING THE AIRCRAFT AND KICK CLEAR OF SEAT

MINIMUM SAFE EJECTION ALTITUDE - LEVEL FLIGHT

MANUALLY ACTUATED PARACHUTE . . . . . 1300 FEET

AUTOMATIC PARACHUTE ACTUATED BY SEAT BELT . . . SEE FIGURE 3-5

AFTER EJECTION

IN NORMAL OPERATION, THE SEAT BELT WILL OPEN 1 SECOND AFTER EJECTION ALLOWING SEPARATION FROM THE SEAT. WITH THE PARACHUTE LANYARD ANCHOR ATTACHED TO THE BELT, THE PARACHUTE WILL OPEN AUTOMATICALLY AT A PRESET ALTITUDE, OR IF BELOW THAT ALTITUDE, THE PARACHUTE WILL OPEN 1 OR 2 SECONDS AFTER SEPARATION FROM SEAT (DEPENDING ON PARACHUTE TIME DELAY).

NOTE

- IN ALL EJECTIONS BELOW 14,000 FEET MANUALLY PULL THE PARACHUTE "D" RING IMMEDIATELY FOLLOWING SEPARATION FROM THE EJECTION SEAT TO INSURE PARACHUTE OPENING
- IF WEARING AUTOMATIC OPENING PARACHUTE WITHOUT LANYARD ANCHOR ATTACHED TO SEAT BELT BUCKLE, PULL PARACHUTE LANYARD ANCHOR
- IF PARACHUTE AUTOMATIC OPENING DEVICE HAS OBVIOUSLY FAILED, OR IF WEARING A MANUALLY OPERATED PARACHUTE, PULL "D" RING AT ALTITUDE WHERE SAFE BREATHING IS POSSIBLE

WARNING

- EJECTION SHOULD NOT BE DELAYED WHEN THE AIRCRAFT IS IN A DESCENDING ATTITUDE AND CANNOT BE LEVELED OUT. THE CHANCES OF SUCCESSFUL EJECTION AT LOW ALTITUDES UNDER THIS CONDITION ARE GREATLY REDUCED
- IF THE SEAT BELT FAILS TO OPEN AUTOMATICALLY AFTER 1 SECOND, PULL MANUAL CABLE CUTTER RING BEFORE THE SEAT BELT IS OPENED MANUALLY TO INSURE SEPARATION OF THE FEET FROM THE SEAT. KICK FREE OF SEAT AND PULL PARACHUTE LANYARD ANCHOR IF ABOVE 14,000 FEET OR THE PARACHUTE "D" RING IF BELOW 14,000 FEET.

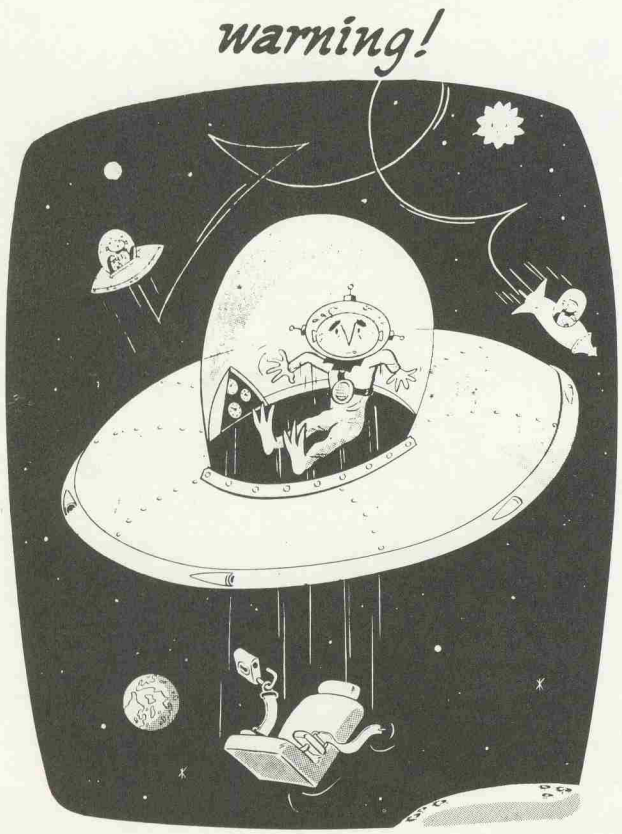
Figure 3-3

or not to eject in an emergency should not be rigidly determined by the fact that the aircraft is above or below the minimum altitude as determined from these figures. Every emergency will have its particular set of circumstances involving such factors as aircraft speed, attitude and control, as well as altitude. Based on the figures and the escape configuration available, a decision should be made before take-off concerning action to be taken in the event of a low-altitude emergency.

EJECTION SPEEDS.

Analysis of escape techniques pertaining to possible injury from air blast and deceleration has revealed that ejection accomplished at airspeeds ranging from stall speed to 525 KIAS result in relatively minor forces being exerted on the body, thus reducing the injury hazard. The pilot will undergo appreciable forces on the body when ejection is performed at airspeeds between 525-600 KIAS, and escape is more hazardous than at lower airspeeds. Above 600 KIAS, ejection is extremely hazardous because of the excessive forces to which the body is subjected.

Figure 3-6 is a plot of three parameters, namely: altitude, speed, and sequence time of the parachute-automatic seat belt combination. The graph shows safe ejection speeds with regard to parachute capability and body injury because of parachute opening shock. The sequence lines (slanting lines) indicate the limits above which the parachute will probably be damaged on opening or the crew member will probably suffer body injury resulting from parachute opening shock.



Failure of Seat to Eject.

If the seat does not eject when the ejection ring is pulled, continue holding ejection ring with one hand and proceed as follows:

1. Pull feet back into stirrups.
2. Escape hatch jettison handle—Pull.

Note

This action releases and ejects the escape hatch and removes the free fall disconnect safety pin.

3. Manual emergency oxygen supply actuator—Pull.
4. Place head back on headrest.
5. Ejection ring—Re-pull through maximum travel.

Note

- This by-passes the thruster initiator, locks the foot retractor cables in the retracted position, fires the catapult and ejects the seat, or the seat will gravity fall.
- With the thruster by-passed, the automatic features of the knee guards, tie-down strap tightening, inertia reel lock and control stick stowage, will not operate. The foot retractor cables will be cut by the action of seat movement.

LANDING EMERGENCIES.

All landing emergencies involving landing on prepared or unprepared surfaces should be made with the landing gear extended. The extended gear, even on reasonably rough terrain, provides an absorption of the initial shock resulting in less injury to the pilot and damage to the aircraft.

Note

- These instructions also apply in those cases in which the runway is overshot or undershot and touchdown cannot be avoided.
- The helmet visor is considered a protective device and should be lowered prior to ditching or making a crash landing.

BELLY LANDING.

Successful belly landings have been made on prepared surfaces. Abandon the aircraft rather than attempt a

PARACHUTE D-RING LANYARD

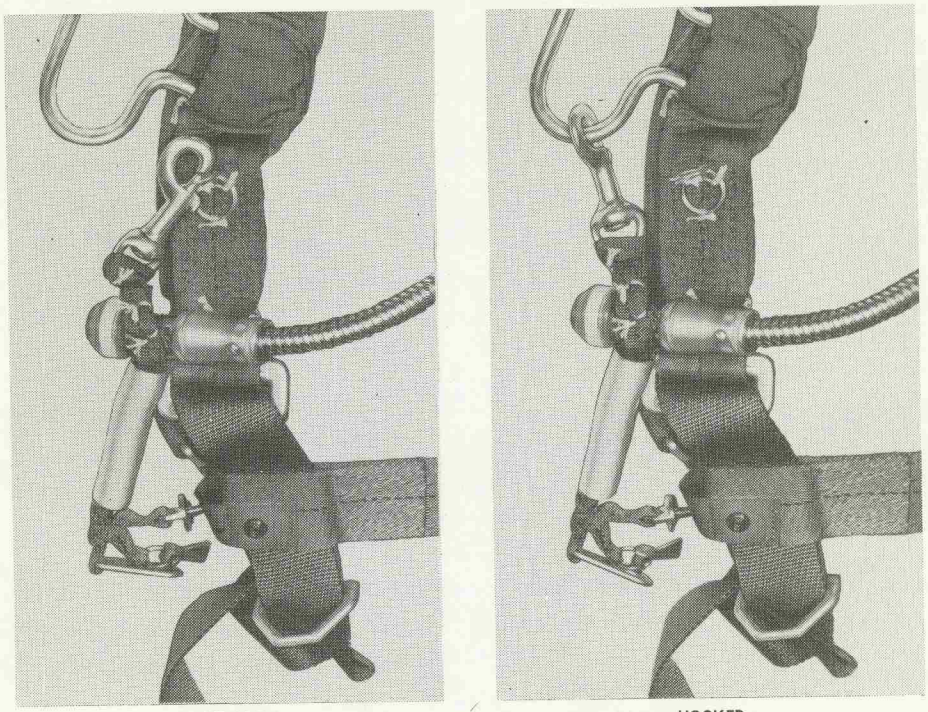


Figure 3-4

belly landing on an unprepared surface. If a gear up landing is unavoidable proceed as follows:

1. External stores—Jettison.
   Jettison external stores in appropriate area; retain tip and pylon tanks if they are empty.
2. Make normal pattern.
   Make normal pattern with landing gear lever UP and landing flaps extended.
3. Canopy internal locking lever—UNLOCKED.
4. Make flat approach.
5. Throttle—OFF at touchdown.
6. Drag chute—Deploy.
7. Fuel shut-off switch—OFF.
8. When stopped—Leave aircraft.

NOSE GEAR FLAT TIRE LANDING.

If it is necessary, to land with a flat tire, proceed as follows:

1. Nose gear—Hold off.
   Hold the nose wheel off as long as practical and then lower gently to runway.
2. Drag chute—Deploy.
   Do not deploy the drag chute until nose wheel is on the ground because of the nose down pitching moment which occurs when the drag chute inflates.

MAIN GEAR FLAT TIRE LANDING.

1. Touch down on good tire.
   Touch down on the side of the runway away from the flat tire.
2. Lower nose wheel.
3. Nose wheel steering—Engage.
4. Drag chute—Deploy.

ASYMMETRIC TIP TANK FUEL LOAD LANDING.

Adequate control is available for landing with one tip tank full and one tank empty under smooth air conditions; however, consideration should be given to the added aileron requirements under strong or gusty crosswind conditions before attempting a landing with an

MINIMUM SAFE EJECTION ALTITUDES

	2 SECOND PARACHUTE (F-1A Timer)		2 SECOND PARACHUTE (F-1A Timer)		1 SECOND PARACHUTE (F-1B Timer)		1 SECOND PARACHUTE (F-1B Timer)		0 SECOND PARACHUTE (Lanyard to "D" Ring)		0 SECOND PARACHUTE (Lanyard to "D" Ring)	
	B-4 or B-5 Pack	C-9 Canopy	B-5 Pack	C-11 Canopy	B-4 or B-5 Pack	C-9 Canopy	B-5 Pack	C-11 Canopy	B-4 or B-5 Pack	C-9 Canopy	B-5 Pack	C-11 Canopy
1 SECOND AUTOMATIC SEAT BELT M12 Initiator	650		700		500		550		400		450	

NOTE THESE ARE EMERGENCY MINIMUMS. EJECTION SHOULD BE STARTED ABOVE 2000 FEET, IF POSSIBLE.

F50-4-3-107

Figure 3-5

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.

FULL OR PARTIALLY FULL PYLON TANK LANDING.

With fuel in the pylon tanks, the pylon strength may be exceeded during a landing; therefore, if a landing must be made immediately and the pylon tanks contain fuel, they should be jettisoned prior to landing. If they cannot be jettisoned, adhere to the recommended landing speeds and exercise close control over sink rate at touchdown.

PARTIAL GEAR LANDING.

Will be added when available.

NO FLAP LANDING.

Because of the high approach and touchdown speeds required to accomplish a no-flap landing, it is recommended that the following instructions be observed.

Note

All of the following speeds are based on landing gross weight of 14,500 lbs. which includes 1000 lbs. fuel remaining. Increase approach and touchdown speeds 5 knots for each additional 1000 lbs. of airplane weight.

Trailing Edge or Leading and Trailing Edge Flap Failure.

Note

Since the primary purpose of the leading edge flaps is to delay buffeting at high angles of attack, little increase in lift is obtained with leading edge flaps only. Therefore, the same procedure is used with trailing edge flap failure as with both leading and trailing edge flap failure.

1. Minimum runway length—10,000 feet.

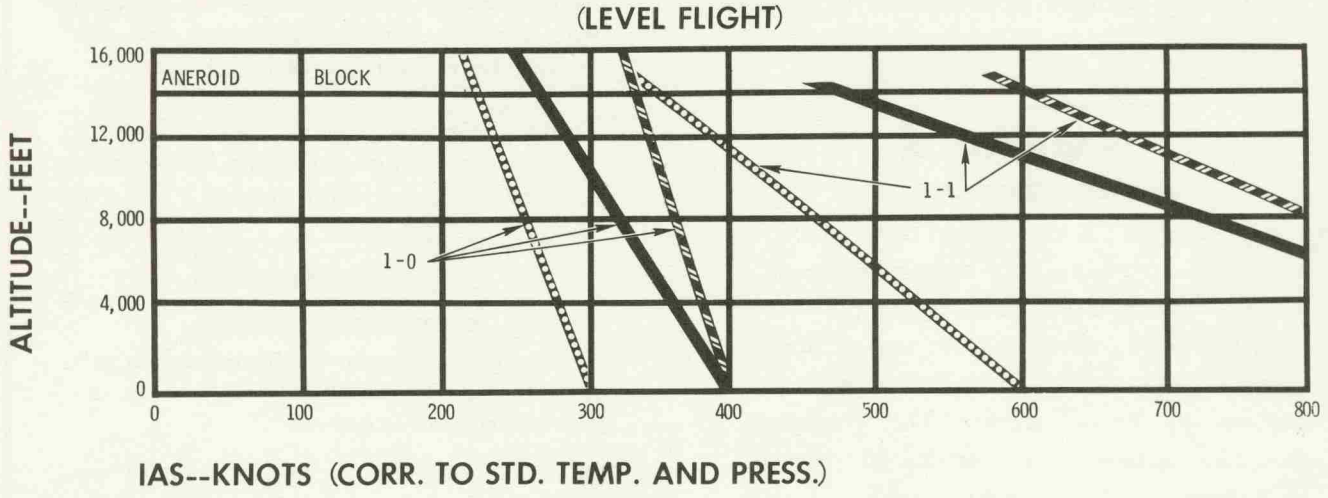
Note

A minimum length runway of 10,000 feet is recommended as an added safety precaution in event of brake or drag chute failure.

2. Pattern speed—40 KIAS above normal.
3. Final approach speed—225 KIAS minimum.
4. Make long flat approach.

   A long flat approach will decrease required rotation for flare.

SAFE EJECTION SPEEDS—ALTITUDE VS. SEQUENCE



TYPE C-9, 28 FT. FLAT CANOPY, TYPE B-4 PACK
TYPE C-9, 28 FT. FLAT CANOPY, TYPE B-5 PACK WITH 1/4 BAG
TYPE C-11, 30 FT. GUIDE CANOPY, TYPE B-5 PACK
DUMMY WEIGHT - 230 TO 250 POUNDS TOTAL

F50-4-3-108

NOTE

THIS GRAPH DEPICTS SAFE EJECTION SPEEDS FOR PARACHUTE STRUCTURAL CAPABILITY UNDER IDEAL LEVEL FLIGHT AND AVERAGE PARACHUTE PERFORMANCE CONDITIONS ONLY; OTHER EJECTION ATTITUDES, TUMBLING, SEPARATION DELAYS, VARIATIONS IN PARACHUTE OPENING TIME, ETC., ARE NOT INCLUDED.

Figure 3-6

Note

Airframe buffeting will be experienced as speed is reduced and/or G-load is increased.

5. Touchdown speed—190 KIAS minimum.

Leading Edge Flap Failure Only.

1. If only the leading edge flaps fail and the trailing edge flaps can be lowered to the LAND position, thereby making boundary layer control available, normal pattern and touchdown speeds can be used.

2. If trailing edge flaps can be extended only to the take-off position, fly final approach at not less than 190 KIAS and touchdown at not less than 160 KIAS.

Note

Buffeting may be experienced at higher than normal speeds without leading edge flaps.

RUNWAY OVERRUN BARRIER.

Tests using the MA-1A runway overrun barrier in various configurations indicate that the minimum speed for successful engagement is 40 knots ground speed. The highest speed tested was 120 knots. Test results indicate that successful engagement is possible at higher speeds. However, speeds above 100 knots accentuate the inherent limitations of the MA-1A overrun barrier increasing the possibility of unsuccessful or one gear engagements. Pylons alone have no effect on barrier engagements. Wing tip stores and wing flap position have no effect on barrier engagement. It was determined that successful engagements cannot be made with pylon tanks installed. With the landing gear lowered by means of the manual landing gear release system (forward main doors fully open), successful barrier engagements were made up to 100 knots ground speed. However, the reliability of the engagements is considerably reduced above this speed. After landing (or aborted take-off), if the aircraft cannot be slowed to a safe turn-off speed before reaching the end of the runway, the steps listed below should be accomplished if a runway overrun barrier is available:

1. Throttle—IDLE.

2. Pylon tanks (if installed)—Jettison.
   The pilot should jettison pylon tanks immediately after deciding to effect a barrier engagement.

WARNING

- For speeds up to 100 knots, pylon tanks should be jettisoned a minimum of 400 feet prior to barrier engagement (dry runway) to prevent the tanks from overrunning the aircraft and possible interference with barrier engagement, gear collapse or aircraft fire. This distance should be increased when operating on wet or icy runway, or at higher ground speeds.
- Pylon tank fuel may ignite by heat generated from the tank skidding on the runway, therefore, in case of emergency, jettison pylon tanks before landing.

Note

- If time permits, drop only the pylon tanks by placing the external stores jettison selector switch to the PYLON position and pushing the stick external stores jettison button. If sufficient time is not available, drop all external stores by pushing the external stores jettison button.
- If the pylon tanks are jettisoned on the runway, the tanks will strike the adjacent main landing gear but do not alter the course of the aircraft.

3. Ground speed—Reduce.
   Decrease ground speed below 100 knots if possible by deploying the drag chute and utilizing maximum wheel braking.

CAUTION

Avoid excessive braking just prior to and during barrier engagement to prevent blowing tires and possible loss of control.

4. Canopy internal locking lever—Unlock.
5. Aim for center portion of barrier.
   Contact barrier as close to a 90° angle as possible.

Note

Nose wheel steering is inoperative when the manual landing gear release system has been used. Therefore, differential braking and rudder must be used for directional control.

6. Throttle—OFF.
   After barrier engagement and it has been determined that nose wheel steering is not required.

WARNING

- If a one gear engagement is made and aircraft becomes uncontrollable, stopcock throttle and turn fuel shut-off switch off.
- Do not unfasten the seat belt or shoulder harness until the aircraft has come to rest.

EMERGENCY ENTRANCE.

The procedure to be used by rescue personnel in assisting the pilot from the airplane following a crash landing is shown in figure 3-7.

DITCHING.

Note

Inspect emergency equipment, parachute, life vest, and life raft before each over-water flight.

Ditch only as a last resort as the nose section of the fuselage may break off. Also, the airplane will probably sink quite rapidly. All emergency survival equipment is carried by the pilot; consequently, there is no advantage in riding the airplane down. However, if altitude is not sufficient for ejection and ditching is unavoidable, accomplish as much of the following as possible:

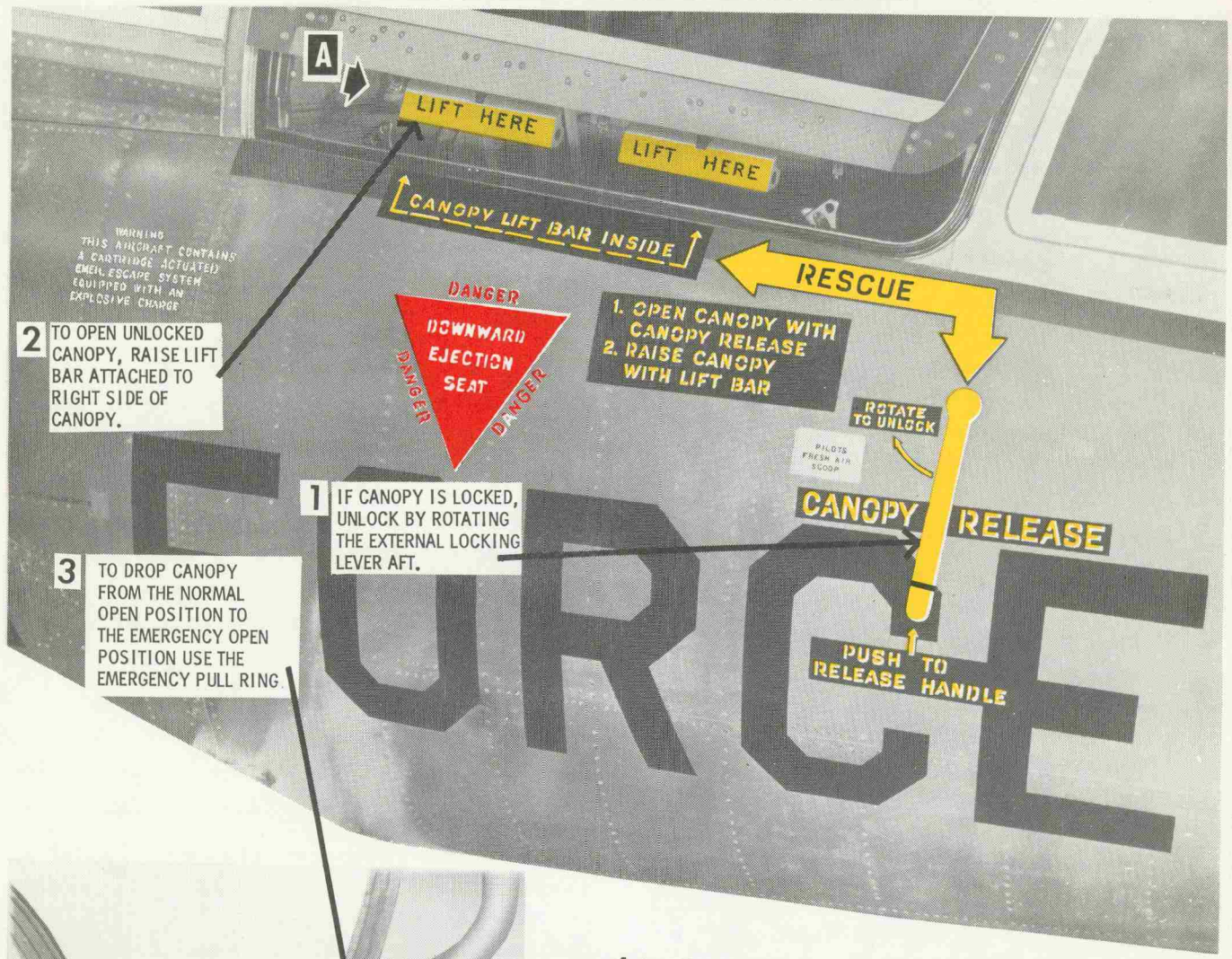
1. Radio—Distress procedure.
2. External stores—Jettison.

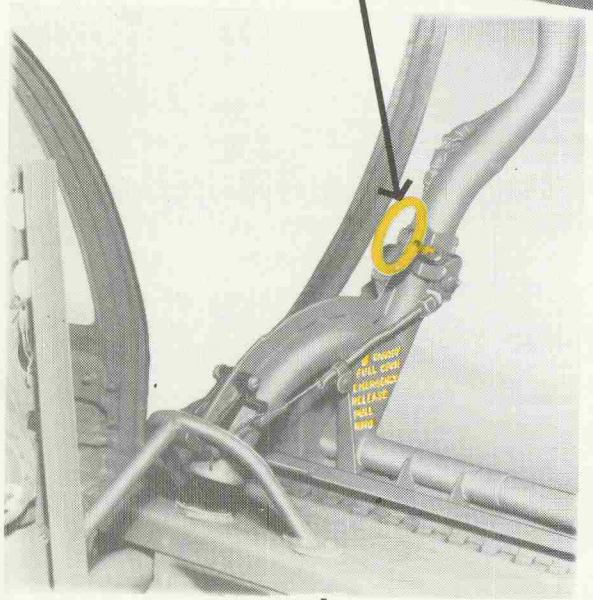
Note

Do not jettison empty tip tanks as the buoyancy obtained from the empty tanks will aid in flotation.

3. Personal equipment—Check.
   See that no personal equipment will foul when you leave the cockpit.

EMERGENCY ENTRANCE





DETAIL A

F50-3-3-43

4 SHUT OFF OXYGEN SUPPLY AT OXYGEN CONTROL PANEL.

5 INSURE THAT SEAT WILL NOT FIRE ACCIDENTALLY - SAFETY THE EJECTION RING.

6 RELEASE SEAT BELT AND HARNESSES.

7 OPEN HELMET FACE PLATE BEFORE DISCONNECTING OXYGEN LINE TO AVOID POSSIBLE RUPTURING OF PILOT'S LUNGS IN EVENT THE PRESSURE SUIT IS PRESSURIZED.

8 DISCONNECT PILOT'S PERSONAL LEADS.

9 PULL MANUAL CABLE CUTTER RING TO FREE PILOT'S FEET FROM THE SEAT.

10 REMOVE PILOT AS GENTLY AS POSSIBLE TO AVOID AGGRAVATING POSSIBLE INTERNAL INJURIES.

NOTE

- THE PILOT MUST BE FREED FROM THE 20 FEET LANYARD ATTACHING HIM TO THE SURVIVAL KIT. THIS MAY BE DONE BY UNFASTENING THE SURVIVAL KIT PARACHUTE ATTACHMENTS.
- IF CANOPY CANNOT BE OPENED, BREAK CANOPY.

Figure 3-7

4. Personal leads—Disconnect.
5. Landing gear lever—UP.
6. Speed brakes—IN.
7. Seat belt—Tighten.
8. Canopy internal locking lever—UNLOCKED.
9. Throttle—OFF.
10. Fuel shut-off switch—OFF.
11. Generator switches—OFF.
12. Normal approach—Keep nose high.
    Make normal approach and flare out to normal landing attitude, being careful to keep the nose high.

Note

Unless wind is high or sea is rough, plan approach heading parallel to any uniform swell pattern and try to touchdown along wave crest or just after crest passes. If wind is as high as 25 knots or surface is irregular, the best procedure is to approach into the wind and touchdown on the falling side of a wave.

13. When stopped—Leave airplane.

EXTERNAL STORES EMERGENCY JETTISON.

To jettison external stores during an emergency proceed as follows:

1. External stores jettison button—Press.
   Pressing the external stores jettison button jettisons pylon and tip stores.
2. If stores fail to release, external stores selector switch—As required.
3. External stores release button—Press.

Note

Refer to External Stores Jettisoning Speed in Section V for jettisoning airspeed limitations.

AFTERBURNER FAILURE.

LOSS OF AFTERBURNER DURING TAKE-OFF.

If the afterburner fails during take-off, a considerable loss of thrust occurs and take-off should be aborted if speed and distance to stop permit. However, if committed to take-off, proceed as follows:

1. Throttle—Military Power.
   Move throttle out of the afterburner range to insure shutting off afterburner fuel flow.
2. Continue take-off at Military Power.
   Do not attempt an afterburner relight until attaining safe airspeed and altitude.

AFTERBURNER SURGE.

Afterburner surge can easily be detected by instability feel, exhaust gas temperature gage and nozzle position indicator oscillation. If afterburner surging occurs proceed as follows:

1. Throttle—Military Power.
   Move the throttle out of the afterburner range as soon as possible, preferably before more than 2 cycles of surge. This action is necessary to prevent compressor stall and possible engine flame-out.
2. Throttle—Afterburner range.
   After 3 to 5 seconds, move throttle smoothly outboard and forward into afterburner range.
3. If afterburner surging continues, throttle—Military Power.
   Avoid using afterburner.

COMPRESSOR DECELERATION STALLS IN FLIGHT.

Above 5,000 Feet.

1. Throttle—IDLE.
2. If stall does not clear—Actuate air start switches.
3. Throttle—OFF.
4. When rpm drops below 70%—Move throttle rapidly to Military.
5. If stall does not clear, throttle—OFF.
6. Inlet guide vane emergency reset switch—MAN.
7. Air start switches—START.
8. Throttle—Advance rapidly to Military.

Note

- Once stall is broken do not modulate the throttle below 97% rpm until landing is assured.
- If landing roll distance is critical, deactivate the inlet guide vanes switch after touchdown. Idle thrust is slightly higher when the reset switch is activated.

Below 5,000 Feet.

1. Throttle—OFF.
2. Inlet guide vanes emergency reset switch—MAN.
3. Air start switches—START.
4. Throttle—Advance rapidly to Military.

Note

- Do not modulate the throttle below 97% rpm until landing is assured.
- If landing roll distance is critical, deactivate the inlet guide vanes switch after touchdown. Idle thrust is slightly higher when the reset switch is activated.
- Do not use the afterburner with the inlet guide vanes reset switch actuated.

OIL SYSTEM FAILURE.

In general, if an oil system malfunction (as evidenced by high or low oil pressure) has caused prolonged oil starvation of engine bearings, the result will be a progressive bearing failure and subsequent engine seizure. The time interval from the moment of oil starvation to complete failure depends on such factors as: condition of the bearings prior to oil starvation, operating temperatures of bearings, and bearing loads. Bearing failure due to oil starvation is generally characterized by a rapidly increasing vibration; when the vibration becomes moderate to heavy, complete failure is only seconds away and in most instances the pilot will increase his chances for a successful ejection or power-off landing by shutting down the engine.

Note

A decrease in oil pressure is an indication of oil loss to the engine, leading eventually to a complete loss of oil to the bearings.

The engine will operate for a period of one minute at Military Power, with a complete interruption of oil to the engine system, without detrimental effects to the bearings. Limited experience has indicated the engine should operate for a period of approximately four to five minutes at 80% to 90% rpm before a complete failure occurs.

In view of the above, the following operating procedures shall be observed for oil system pressure changes:

1. Thrust—Reduce to maintain safe level flight (86% to 89% rpm with a clean aircraft).

WARNING

Complete loss of engine oil will also result in loss of exhaust nozzle control. The resulting loss of thrust with the nozzles in a trail position may be as much as 70% at 100% rpm.

High thrust settings should be avoided if at all possible in order to keep temperature and bearing loads at a minimum. Upon detection of an oil system malfunction (as evidenced by the oil pressure gage) a minimum thrust setting should be established depending on aircraft configuration, gross weight and altitude. This setting should be sufficient to maintain level flight and allow for safe approach maneuvers (subsequent throttle movement should be avoided if possible). However, if the malfunction has gone unnoticed and has progressed to the point where bearing failure has started, as evidenced by vibration, the throttle should not be retarded. If the throttle is retarded, the resistance to rotation offered by one or more failing bearings may cause further deceleration and complete engine seizure in a very short time.

WARNING

If moderate to heavy vibration occurs, the chance of a successful ejection or power-off landing will be improved by shutting down the engine.

2. External stores not required—Jettison.
3. G-forces—Minimize.

   Avoid rapid changes in attitude and "G" forces.

4. Hold throttle changes to a minimum.

5. Land as soon as possible, using Precautionary Landing Procedure.

WARNING

Increasing vibration is an indication of bearing failure. Extreme vibration, usually accompanied by a rise in EGT, indicates engine seizure will occur within a few seconds. The throttle should be cut off to prevent excessive destruction to the engine, resulting in possible damage to the aircraft structure.

Note

With a complete loss of oil, the nozzle may go to the full open position which would require a higher thrust setting than normal. A throttle setting of 86% to 87%, with a normal engine, represents the thrust that would be obtained with a wide open nozzle at 100 to 101.5% engine speed.

ENGINE OIL LOW LEVEL WARNING LIGHT.

Illumination of the engine low level warning light indicates that the oil quantity has decreased to approximately one gallon. If the light illuminates proceed as follows:

1. Thrust—Reduce to minimum required.
2. Land as soon as possible.
   If altitude permits it is recommended that a precautionary landing pattern be established.

FUEL SUPPLY SYSTEM FAILURE.

AIRCRAFT FUEL SYSTEM.

If the fuel boost pump failure warning light illuminates, avoid steep nose down attitudes which may starve the aft pumps of fuel. Abort the mission and make a normal landing.

Note

Any one of the boost pumps will furnish sufficient fuel pressure for engine operation as long as the pump is submerged in fuel.

ELECTRICAL POWER SYSTEM FAILURES.

COMPLETE ELECTRICAL SYSTEM FAILURE.

If a complete electrical failure occurs, (which includes failure of both engine driven generators and ram air turbine driven generator) only those items powered from the battery bus will be operable. (Refer to figure 1-18). Flight under these conditions is limited, and the following precautions should be observed:

1. Land as soon as possible.

2. Use manual landing gear release handle to lower landing gear.

3. Use No Flap Landing procedure.
   It will be impossible to lower the wing flaps. Refer to No Flap Landing in this section.

Note

Under the above conditions of complete electrical failure, all of the instruments except the tachometer and exhaust gas temperature gage, the airspeed and mach number indicator, accelerometer, clock, altimeter, cabin altimeter, and vertical velocity indicator, will be inoperative. The indicating pointer on electrically operated instruments will have the following reaction with a complete electrical failure:

a. Directional indicator and fuel quantity indicator—Pointer will not move.
   Avoid all abrupt maneuvers causing high "G" forces.
b. Hydraulic pressure gage, oil pressure gage, and fuel flow indicator—Pointer will move to zero.
c. Turn and slip indicator—Vertical needle will remain in center position, ball will continue to operate.
d. Engine air inlet temperature gage—Pointer will move to bottom of dial.
e. Radio magnetic indicator—Compass card and pointers will not move.
f. Course indicator—OFF flags will appear, pointers will move out of view.
g. Attitude indicator—OFF flag will appear, gyros will tumble eventually.
h. Nozzle position indicator—Pointer will move to full left side of dial.

Refer to figure 1-18 for instruments inoperative with ram air turbine extended.

GENERATOR FAILURE.

No. 1 or No. 2 Generator Out.

If No. 1 or No. 2 generator out warning light illuminates:

1. Move corresponding generator switch to OFF then to ON RESET and release.

   This will restore generator to service if failure was caused by momentary over-voltage.

No. 1 and No. 2 Generator Out.

If both generators fail as evidenced by failure of all electrical equipment except that powered by the battery, proceed as follows:

1. No. 1 generator switch—OFF, then ON RESET position.

2. No. 2 generator switch—OFF, then ON RESET position.

3. If generator operation is not restored—Land as soon as possible.

WARNING

- Extension of the ram air turbine is not recommended except under extenuating circumstances such as immediate requirements for radio, lighting or flight instruments. If the ram air turbine is extended, engine operation should be monitored closely due to the possibility of flame-out, especially above 25,000 feet.

- If the ram air turbine has not been extended and a no flap landing is undesirable, follow Precautionary Landing procedure extending the ram air turbine at "High Key" point.
- The leading and trailing edge flaps are sequenced to extend separately to the TAKE-OFF position only, when using ram air turbine driven generator for electrical power. Therefore, the LAND or UP position should never be selected under this condition as it may stall the generator. Keep airspeed above 200 knots IAS until flaps have reached TAKE-OFF position. (The wing flap position indicator will be inoperative.)

D.C. MONITORED BUS FAILURE.

If the d.c. monitored bus warning light illuminates, the flight should be aborted. Refer to figure 1-18 for systems rendered inoperative with the d.c. monitored bus out.

AUTOMATIC BUS TRANSFER FAILURE.

Should the automatic bus transfer fail to operate, as indicated by electrical equipment failure, place the No. 2 generator switch in the OFF position. If the electrical equipment failure is not remedied, reset the No. 2 generator and place the No. 1 generator switch in the OFF position. Leaving one of the two generators switches in the OFF position should activate the bus transfer system and allow the other generator to assume the entire load. If bus transfer fails to take place, reset both generators.

Note

Illumination of the "D.C. MONITORED BUS OUT" and the "INST. ON EMER. POWER" warning lights may also indicate automatic bus transfer failure. In this case the No. 1 generator switch should be placed in the OFF position to activate the bus transfer system and allow the No. 2 generator to assume the entire load.

HYDRAULIC SYSTEM FAILURE.

NO. 1 OR NO. 2 SYSTEM OUT.

1. Hydraulic pressure—Check both systems.
   Check both systems pressure to determine which system is out.
2. Land as soon as possible.
3. Use Precautionary Landing procedure.

BOTH NO. 1 AND NO. 2 SYSTEMS OUT.

1. Ram air turbine—Extend immediately.
2. No. 1 system pressure—Check for increase.
   Check to see that No. 1 system pressure increases and that controls operate satisfactorily.

Note

Maximum hydraulic flow available under these conditions is reduced; however, it is sufficiently high for safe flight and moderate maneuvers necessary for landing.

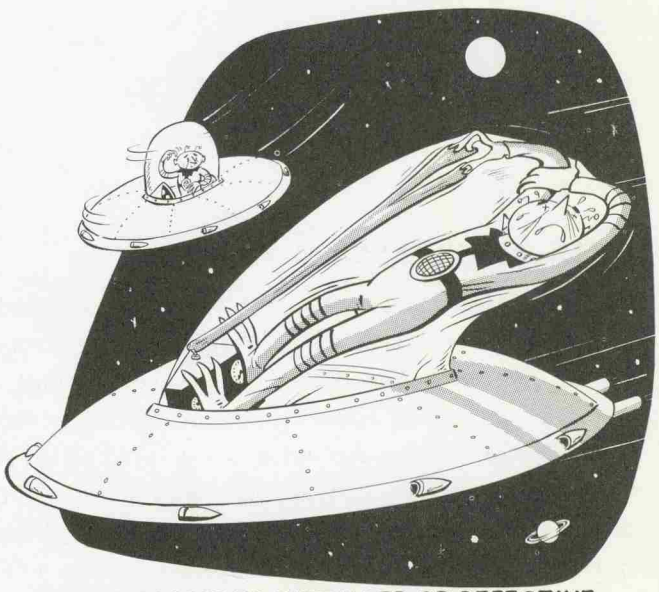
3. Land as soon as possible.
4. Use Precautionary Landing procedure.

TRIM FAILURE.

If trim failure should occur, proceed as follows:

1. Trim selector switch—AUX. TRIM position.
2. Use auxiliary trim control switch as necessary.
3. If trim still malfunctions, trim circuit breaker—Pull.

caution!



AN IMPROPERLY INSTALLED OR DEFECTIVE TRIM SWITCH MAY BE SUBJECT TO OCCASIONAL STICKING IN AN ACTUATED POSITION, RESULTING IN EXTREME TRIM. WHEN THIS CONDITION OCCURS, THE AUXILIARY TRIM SWITCH SHOULD BE USED AND THE FAILURE ENTERED IN DD FORM 781 WITH A RED CROSS.

STABILITY AUGMENTATION CONTROL SYSTEM FAILURE.

Failure in any one of the stability augmentation control system channels (roll, pitch or yaw) may cause control system oscillation. If this occurs, proceed as follows:

1. Switch for the affected channel—OFF.
2. If any further trouble in system occurs, two remaining switches—OFF.
3. Before entering traffic pattern, all three switches—OFF.

All three switches should be turned off and remain off for landing.

Note

Refer to Section VI for the effect of an inoperative stability augmentation control system on flight characteristics.

AUTO-PITCH CONTROL SYSTEM FAILURE.

If the auto-pitch control system fails as indicated by illumination of the "AUTO-PITCH OUT" warning light, proceed as follows:

1. Avoid high acceleration and low airspeed.

WING FLAP SYSTEM FAILURE.

Refer to No Flap Landing in this Section.

SPEED BRAKE SYSTEM FAILURE.

There is no emergency means for operating the speed brakes. The speed brakes are electrically controlled from the d.c. emergency bus and cannot be operated unless the bus is energized by either the normal electrical system or the ram air turbine. Actuation of the speed brakes is by means of 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 is lost or drops below 2000 psi. If hydraulic pressure is lost with the speed brakes in the full open position, the speed brakes will not retract due to airload distribution. Because of the decreased range capabilities of the aircraft in this configuration a decision should be made to land as soon as possible. If an engine flame-out is experienced with the speed brakes extended, electrical power for speed brake retraction is lost. At best glide speed with the speed brakes full out, the rate of descent increases to approximately 11,000 feet per minute. If an engine air start cannot be obtained after several attempts the ram air turbine should be lowered and engine air starts re-attempted. The ram air turbine will furnish electrical power for speed brake operation and the windmilling engine at best glide speed will supply sufficient hydraulic pressure to the No. 2 hydraulic system to retract the speed brakes. This will decrease the rate of descent and make possible a forced landing should the engine fail to restart.

WARNING

If the flame-out is due to a frozen engine the speed brakes cannot be retracted. Bailout is recommended because the high rate of descent would preclude a safe forced landing.

LANDING GEAR EMERGENCY OPERATION.

LANDING GEAR LEVER MALFUNCTION.

1. Reduce airspeed.

   Keep airspeed below transient landing gear structural limit.
2. Landing gear down-lock mechanical override button—Press.
3. Landing gear lever—UP.

LANDING GEAR RETRACTION FAILURE.

If the landing gear warning light in the landing gear lever remains on after the lever has been placed in the UP position, proceed as follows:

1. Reduce airspeed.

   Keep the airspeed below the transient structural limits; use speed brakes if necessary.
2. Recycle gear at lowest practical airspeed.
3. If warning light remains on, lower gear and land.

LANDING GEAR EMERGENCY EXTENSION.

If the landing gear indicators do not show gear down and locked after lever is placed in the DOWN position, keep speed below the transient landing gear structural limit and proceed as follows:

1. Recycle gear.
2. If gear does not lock down, leave gear lever down and pull manual landing gear release handle.

CAUTION

- The landing gear cannot be retracted in flight after being lowered by means of the manual landing gear release handle.
- Also, nose wheel steering will be inoperative if the gear is lowered by this method.

Note

Pulling G's and yawing the aircraft will help to lock the gear in the down position.

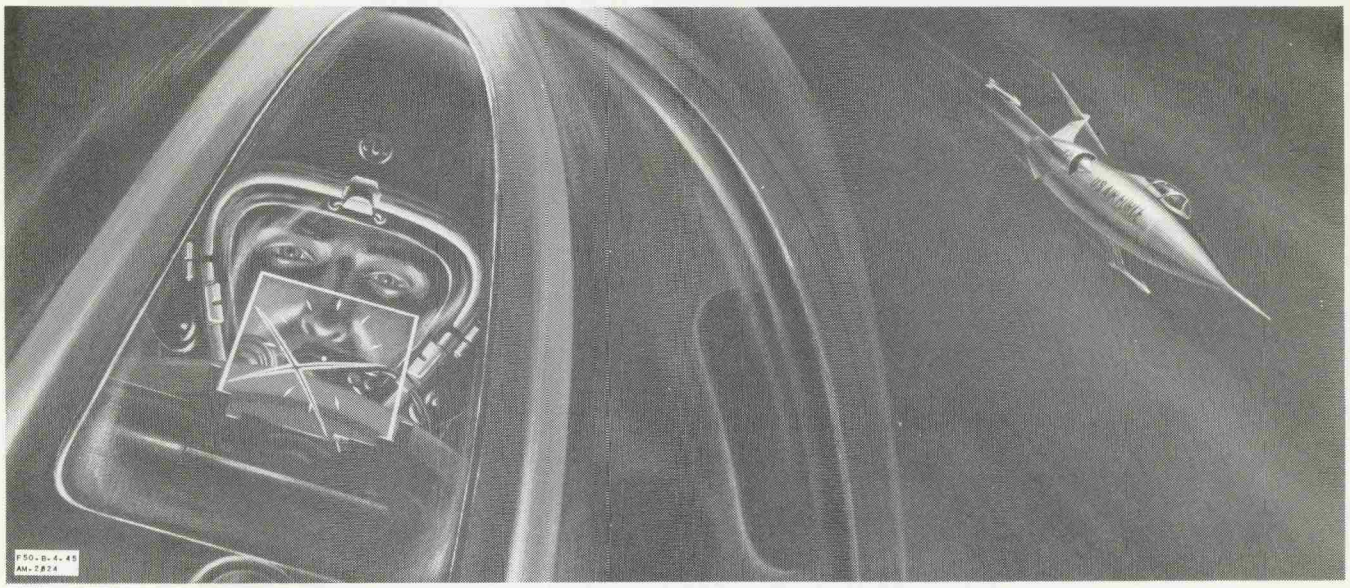
LOSS OF EMERGENCY BLOW-OUT PANEL.

Will be added when available.

CONDENSED CHECK LIST.

Refer to pages 3-23 through 3-34 for the Condensed Check List.





auxiliary equipment SECTION IV

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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 stage of the engine compressor (figure 4-1). The temperature of this air as it leaves this last stage of compression is approximately 900°F and the pressure is 250 psi at maximum engine rpm and maximum airspeed. 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 air passes through a shut-off valve, after which a portion goes to the rain removal duct, canopy and electronic compartment, anti-G suit, and radar pressurization systems. The remainder then passes either through or around a refrigeration unit (figure 4-1), depending upon the positions of the by-pass valves. The compressor air which goes through the by-pass valves is directed to an air mixing chamber where it meets and mixes with the air which has gone through the refrigeration unit. This mixture is directed through a water separator, and enters the cockpit through shoulder outlets and foot warmers. The temperature of the air entering the cockpit depends upon the position of the by-pass valves. For maximum heating, the by-pass valves will be fully opened and most of the air entering the cockpit will by-pass the refrigeration unit. For maximum cooling, the by-pass 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 air temperature is above water boiling temperature, the water will boil and the air will be cooled through this evaporation process. The temperature of the air entering the cockpit can be varied by the pilot's cockpit heat rheostat which controls the position of the by-pass valves. In normal operation, air temperature is maintained between 40°F and 100°F automatically by means of a thermostat control. This senses cockpit temperature, compares this temperature with the cockpit heat rheostat selection, and sends electrical signals to the by-pass valves to change their positions as necessary to make cockpit air temperature correspond with rheostat selection. The pilot can also control the temperature manually by means of the cockpit heat rheostat. When this is done, the thermostat control is cut out of the system, and the by-pass valves are directly positioned in accordance with rheostat selections.

AIR CONDITIONING AND PRESSURIZATION SYSTEM

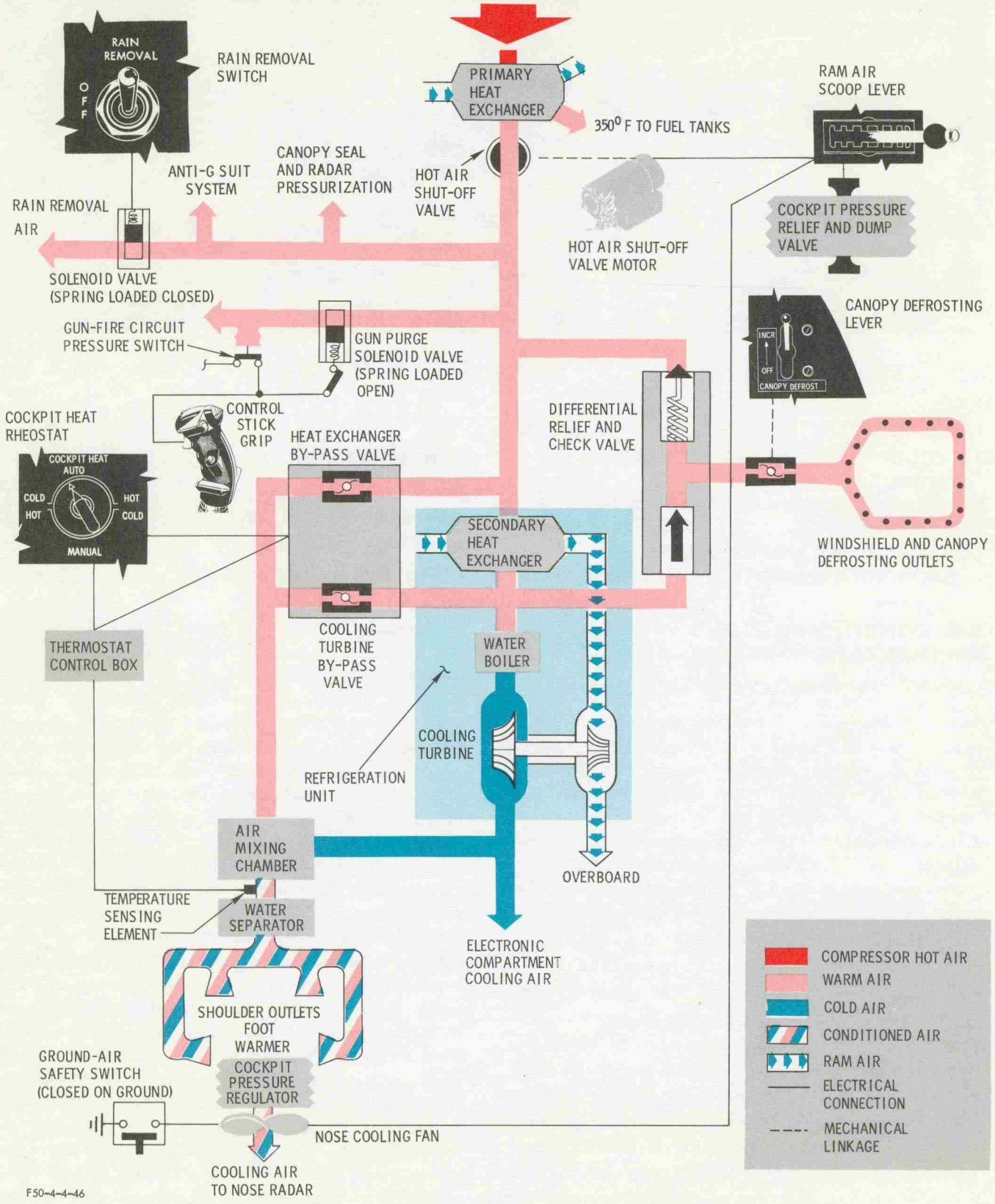


Figure 4-1

COCKPIT PRESSURIZATION.

Cockpit pressurization is maintained at the proper level by an automatic cockpit pressure regulator located in the left forward cockpit area. There is no pressurization in the cockpit below 5,000 feet. Between 5,000 and 18,500 feet, the cockpit altitude will be constant (figure 4-2), while the differential pressure will vary from 0 to 5.0 psi. Above 18,500 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 compartment forward of the cockpit for cooling purposes. The cockpit pressure regulator unit incorporates a cooling fan which will force cockpit air into the radar compartment any time the aircraft is on the ground. This fan is also actuated whenever the ram 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 air entering the cockpit becomes contaminated, the pilot can alleviate this condition by use of the ram air scoop. This will allow outside ram air to enter the cockpit as well as shutting off the flow of compressor air into the cockpit and releasing cabin air overboard through the cockpit pressure relief and dump valve.

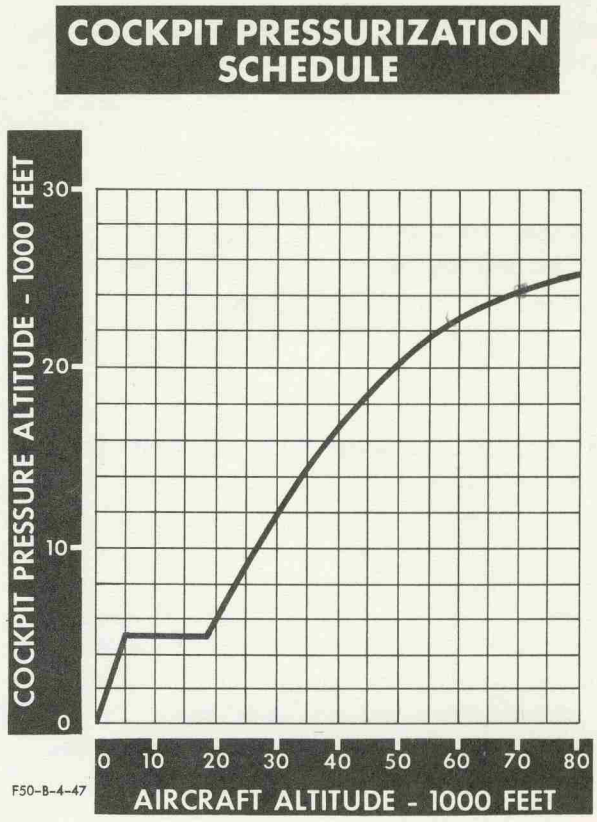


Figure 4-2

COCKPIT HEAT RHEOSTAT.

The cockpit heat rheostat (figure 4-3) located on the right console is used to control cockpit temperature. Under normal conditions the rheostat is used in the AUTO mode of operation and may be set at any point between COLD and HOT (figure 4-3). Cockpit temperature will then be maintained automatically through the thermostat control unit which uses emergency a.c. bus power to control the pneumatically operated by-pass valves. With the rheostat in the MANUAL mode of operation, the thermostat does not function and the by-pass valves are positioned directly in response to movement of the rheostat between the COLD and HOT positions. The cockpit heat rheostat is powered from the emergency a.c. bus.

RAM AIR SCOOP LEVER.

The ram air scoop lever (10, figure 1-8) located outboard of the right console, controls the position of the ram air scoop. Depressing a button on the handle of the lever allows forward movement which opens the scoop. Aft movement of the lever which closes the ram air scoop does not require depressing the button. However, on aircraft serial 56-789 and subsequent the button must be depressed before the lever can be placed in the last aft detent. In all aircraft initial movement of the lever out of the last aft detent actuates a switch which causes emergency d.c. bus power to close the hot air shut-off valve, open the cockpit relief and dump valve, and energize the nose cooling fan. On aircraft serial 56-789 and subsequent, 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 move the scoop to an open position.

CAUTION

- With the engine operating and the ram air scoop opened on the ground, the supply of cooling air to the electronics compartment is shut-off. This can cause the electronic equipment to reach over-temperature limits.
- Make certain that the ram air scoop handle is in the last aft detent so that cooling air for the electronic compartment and cockpit pressurization is available.

CABIN ALTIMETER.

The cabin altimeter (21, figure 1-6), 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 cockpit altitude regardless of actual aircraft altitude.

NORMAL OPERATION OF COCKPIT AIR CONDITIONING AND PRESSURIZATION SYSTEM.

1. Ram air scoop lever—CLOSED.
2. Cockpit heat rheostat—AUTO, and positioned as desired.

EMERGENCY OPERATION OF COCKPIT AIR CONDITIONING AND PRESSURIZATION SYSTEM.

Pressure surging may be encountered when operating on MANUAL. The surging is caused by the thermal switch, which is installed in the system as a high temperature safeguard, shutting off the hot air to the mixing chamber when the air in the duct reaches 210°F. Shutting off this air increases the air flow to the turbine which causes a pressure drop in the air to the cockpit. When the cold air from the turbine hits the thermal switch, it opens and allows hot air to flow again. This cycle may occur at from eight to forty times per minute depending on the temperature of the bleed air and the setting of the control. At slow airplane speeds, where bleed air temperature is below 200°F, no surging will occur even on full HOT. At higher airplane speeds, where bleed air temperature may exceed 400°F, surging will occur at one quarter (¼) MANUAL HOT setting, and if setting is increased the surge cycle will increase to a faster rate.

If cockpit temperature is not maintained at the desired level automatically:

1. Cockpit heat rheostat—MANUAL and position as desired.

Note

- If airflow surging occurs, move cockpit heat rheostat toward COLD.
- If cockpit temperature is excessive and cannot be decreased automatically or manually, open the ram air scoop.

CAUTION

Manual control is provided as a back-up feature only and should not be used except in case of automatic control failure. There is danger of fogging up the cabin on take-off if manual control is used. This will not happen on automatic.

If air in the cockpit becomes contaminated or if depressurization becomes necessary:

1. If using A-13A oxygen mask—diluter lever 100%, emergency lever EMERGENCY.
2. Descend to 25,000 feet, or lower, if circumstances permit.
3. Ram air scoop lever—OPEN.

DEFROSTING AND RAIN REMOVAL SYSTEM.

DEFROSTING SYSTEM.

The defrosting system consists of a number of small air jets directed parallel to the canopy and windshield surfaces. These jets entrain cabin 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 dew point, no fog or frost will be formed. Air for the defrosting system is normally routed from the secondary heat exchanger through a check valve and the defrost flow control and shut-off valve (figure 4-1) to the defrosting outlets located along the inside base of the windshield canopy. This air flow, in itself, is not sufficient to meet all the requirements of the system. To supplement this flow, a bleed line is provided which directs air from a point just downstream of the hot air shut-off valve to a differential relief and check valve. When pressure in the normal flow line drops because of large demands on the system (defrost flow control valve open), the differential relief valve will open and furnish the additional air necessary for effective defrosting under all conditions.

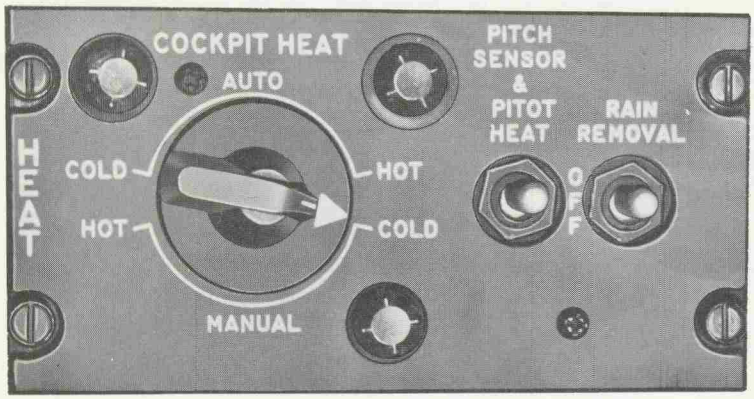
CANOPY DEFROSTING LEVER.

The amount of air directed to the windshield and canopy defrost outlets is determined by the position of the canopy defrosting lever (6, figure 1-8) located on the cockpit right forward panel. Upward movement of the lever increases the amount of defrost air to the outlets by actuating the pneumatically-operated shut-off valve. With the lever in the INCR. (up) position, the valve will be fully opened, while in the OFF (down) position, the valve will be closed and no defrosting air will be available.

FACE PLATE HEAT SYSTEM.

Heating elements are incorporated in the pilot's face plate to prevent or remove any accumulation of moisture on the face plate which would hinder the pilot's vision. This system becomes especially important at high altitudes in the event of any malfunction which may cause rapid decrease in cockpit temperature and pressure. To insure that face plate heating will be available under all operating conditions, except during complete electrical failure, electrical power for the heating elements is taken from the battery bus.

HEATING CONTROL PANEL



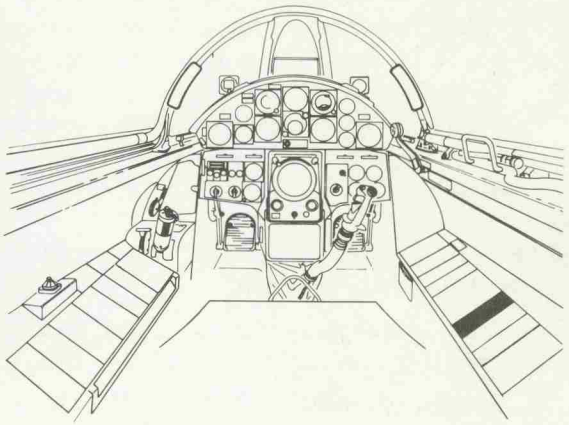


Figure 4-3

FACE PLATE HEAT RHEOSTAT.

D.C. electrical power from the battery bus to the face plate heating elements is controlled by the face plate rheostat (26, figure 1-6) located on the lower right instrument panel. Heat may be applied to the face plate in varying degrees by moving the rheostat clockwise from OFF to any desired position. Heating intensity will be maximum with the rheostat in the ON (extreme clockwise) position.

RAIN REMOVAL SYSTEM.

The rain removal system receives compressor air from the same line which furnishes pressure for the canopy seal, radar system and anti-G suit valve (figure 4-1). Rain removal air passes through a pilot-controlled shut-off valve and is then routed to nozzles 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.

CAUTION

Rain removal 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 ram air scoop should be opened. This will shut off all compressor air to the duct line and direct cold ram air into the cockpit.

RAIN REMOVAL SWITCH.

The rain removal switch (figure 4-3) located on the right console outboard of the cockpit heat rheostat controls the flow of compressor air to the rain removal outlets. Moving the switch forward from OFF to the RAIN REMOVAL position closes a 28-volt d.c. monitored bus circuit to the rain removal shut-off valve, opening the valve and allowing hot, pressurized air to pass through the valve to the outlets.

CAUTION

Do not turn the rain removal system on above its limit air speed, or the rain removal nozzles may be damaged.

NORMAL OPERATION OF DEFROSTING AND RAIN REMOVAL SYSTEM.

If any portion of the windshield or canopy becomes obscured by moisture:

1. Canopy defrosting lever—INCR.
2. Cockpit heat rheostat—AUTO—HOT.
3. Rain removal switch—RAIN REMOVAL.
   If precipitation is obscuring forward visibility.

Note

Canopy defrosting 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 a rapid descent from altitude.

EMERGENCY OPERATION OF DEFROSTING AND RAIN REMOVAL SYSTEMS.

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

1. Canopy defrosting lever—Check (INCR. position).
2. Cockpit heat rheostat—MANUAL—HOT.
3. Rain removal switch—RAIN REMOVAL.
4. 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 take-off, open the ram air scoop.

ANTI-ICING SYSTEMS.

ENGINE ANTI-ICING SYSTEM.

Heated compressed air is bled from the 17th stage of the engine compressor to the engine inlet guide vanes through an electrically-operated, solenoid shut-off valve. The anti-icing system should be utilized whenever icing conditions are encountered or suspected during flight. Inlet guide vane icing will be indicated by a loss of power accompanied by a possible increase in nozzle opening and an increased fuel flow. When icing conditions no longer exist, the anti-icing system should be turned off. The system bleeds air from the engine compressor which reduces available thrust, increases specific fuel consumption, and decreases airspeed.

CAUTION

Do not use the engine anti-icing system on the ground at idle rpm. Air entering the anti-icing outlets will be excessively hot under these conditions and engine damage caused by overheating of inlet guide vanes may result.

Engine Anti-Ice Switch.

The engine anti-ice switch (figure 1-10) located on the left forward panel controls the flow of compressor air to the engine inlet guide vanes. In the OFF position, the circuit to the solenoid operated engine anti-ice shut-off valve will be opened and no air will flow through the valve. Placing the switch up to the ON position allows a.c. power from the No. 2 bus to energize the circuit, which opens the valve, and permits compressor air to be directed to the inlet guide vanes.

CAUTION

On unmodified aircraft the engine anti-icing valve has been found to be unreliable and has shown a tendency to stick in the OPEN position. Since there is no indication in the cockpit to warn the pilot of this condition, it is possible to subject the front frame of the engine to a severe over-temperature condition which can result in an engine failure. A warning light will be installed in the cockpit to inform the pilot when the anti-icing valve is in the OPEN position. Pending the installation of this light-indicator system, the anti-icing valve has been deactivated. In view of this, do not fly through known or suspected icing conditions in aircraft with the engine anti-icing valve deactivated.

"ENGINE ANTI-ICING ON" WARNING LIGHT.

On modified aircraft this light has been added to the annunciator panel to indicate that the anti-icing valve is in the open position regardless of the position of the engine anti-ice switch.

PITCH SENSORS AND PITOT HEAT SWITCH.

The auto-pitch control and stick shaker pitch sensor vanes and pitot-static head are heated electrically by power from the No. 2 a.c. bus. Heating elements within the pitch sensor vanes and the pitot-head receive this power whenever the heating circuit is closed. This is done by placing the pitch sensor and pitot heat switch (figure 4-3) on the cockpit right console from OFF to the PITCH SENSOR and PITOT HEAT position. Heat should be applied any time instrument flying conditions are encountered in order to prevent the formation of ice on these units.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT.

UHF COMMAND RADIO AN/ARC-66.

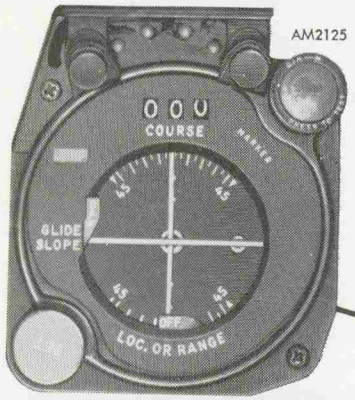
Note

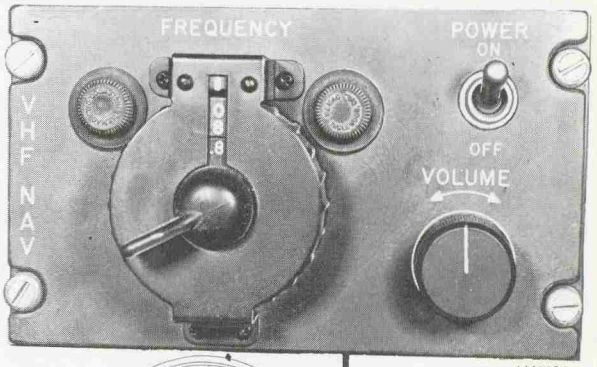
No transmission will be made on emergency (distress) frequency channels except for emergency purposes. For test, demonstration, or drill purposes, the radio equipment will be operated in a shielded room to prevent transmission of messages that could be construed as actual emergency messages.

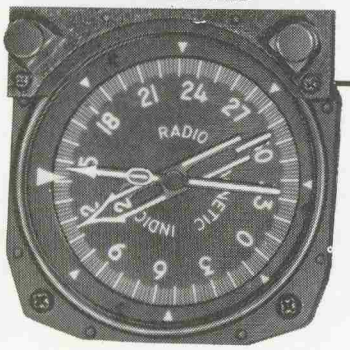
The AN/ARC-66 UHF command radio which is installed in the aircraft provides two-way radio communications (air-to-air and air-to-ground) on 1750 different frequencies in a range extending from 225.0 megacycles through 399.9 megacycles. Any of these frequencies may be selected manually; however, the radio is preset on the ground to

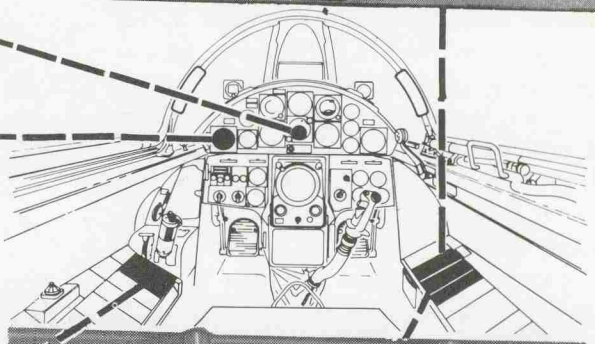
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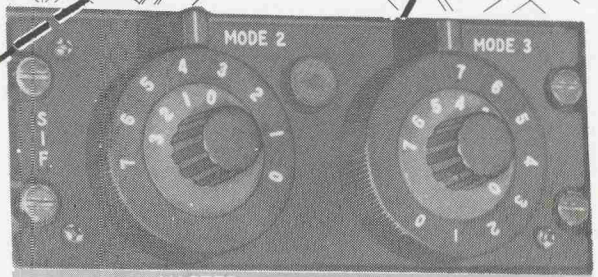
TABLE OF COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

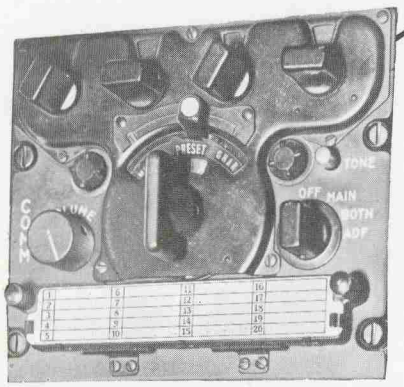














TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION
UHF COMMAND	AN/ARC - 66	TWO-WAY COMMUNICATION AND INTERPHONE	LINE OF SIGHT COCKPIT TO GROUND	LEFT CONSOLE RIGHT CONSOLE
VHF NAVIGATION	AN/ARN - 56	VOR NAVIGATION VOICE RECEPTION LOCALIZER	LINE OF SIGHT	RIGHT CONSOLE AND INSTRUMENT PANEL
	AN/ARN - 57	GLIDE PATH MARKER BEACON	AIRCRAFT TO GROUND	
IFF	AN/APX-35	AIRCRAFT IDENTIFICATION	LINE OF SIGHT	RIGHT CONSOLE

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GB 1144

Figure 4-4

select only the twenty most commonly used frequencies (channels) during normal operation. In addition to the main receiver, the set utilizes a guard receiver which can cover a frequency range between 238.0 and 248.0 megacycles, but which is pre-tuned, prior to installation, to provide an emergency channel which is constantly alerted. The control panel (13, figure 1-7) for the set is located on the left console and includes a function switch, a selector switch, a volume rheostat, a tone button, and four manual tuning knobs (figure 4-4). The OFF position of the function switch disconnects the set from its power source (d.c. emergency bus and a.c. emergency bus). With the switch in the MAIN position, the transmitter and main receiver will be operating on the selected frequncy, but the guard receiver will be inoperative. With the function switch at BOTH, the transmitter and main receiver will operate as indicated above, and the guard receiver will function simultaneously to provide reception on guard channel. The ADF position is inoperative. The selector switch determines frequency selection and has three positions, MANUAL, PRESET, and GUARD. Normally, the PRESET position is used, and permits the pilot to choose any of the 20 preset frequencies or channels by movement of the channel selector knob at the center of the selector switch. A numerical indication of the selected channel is provided in a window directly above the channel selector knob. When the selector switch is in MANUAL, any of the 1750 frequencies in the 225.0 to 399.9 megacycles range may be selected by moving the four manual tuning knobs to the desired positions. An indication of the frequency selected is provided in the windows adjacent to and forward of each tuning knob. The GUARD mode of selector switch operation places the main receiver and transmitter on the fixed guard frequency, permitting the pilot to transmit, as well as receive, on guard channel. Depressing the tone button cuts out radio reception and transmits a continuous tone which aids ground stations in obtaining a bearing on the aircraft during direction finding procedures. The AN/ARC-66 UHF command radio package also contains the interphone system which is powered by the d.c. emergency bus. The interphone system provides communication between the pilot and ground crew during starting and pre-taxi procedures. The interphone is connected to the aircraft through a jack in the external electrical power receptacle. The pilot can receive transmissions from the ground crew whenever the AN/ARC-66 UHF radio is operating and his headset is connected. In order to transmit to the ground crew, the pilot must first depress the interphone button (11, figure 1-8) on the right console. The landing gear warning signal is also connected through the interphone system.

OPERATION OF UHF COMMAND RADIO.

1. Selector switch—PRE-SET.

2. Function switch—BOTH and allow 2 minutes for warm-up.

3. Channel selector knob—As desired.

Note

In order to obtain maximum transmitting and receiving effectiveness, some channel other than the one desired should be selected first. Allow the tuning cycle to be completed and then select the desired channel. Because of an inherent characteristic of the set, full power will not be available unless this procedure is followed.

4. Volume rheostat—As desired.

5. To select a frequency other than one of the pre-set channels:

a. Function switch—MAIN or BOTH.

b. Selector switch—MANUAL.

c. Manual tuning knobs—Position as required to obtain the desired frequency. (A numerical indication of the selected frequency will be presented by the digits appearing in the indicator windows at the top of the control panel.)

6. To transmit as well as receive on guard channel, move the selector switch to GUARD.

Note

The channeling drive motor cuts out automatically after 50 to 125 seconds of continuous operation. If this should occur, place the function switch to OFF, allow 30 seconds for cooling, then place the switch to BOTH and select the desired channel.

VHF NAVIGATION EQUIPMENT AN/ARN-56.

The VHF navigation set can receive transmissions in a frequency range of 108.0 to 135.9 megacycles. Visual and aural information is provided through this equipment to assist the pilot in determining the position of the aircraft in relation to visual omnirange stations on the ground. Transmission of these signals is line-of-sight, so reception range will depend upon altitude and obstructions. At sea level unobstructed range is 3 miles, while at 10,000 feet, it will be approximately 100 miles and at higher altitudes range of reception will be even greater. The receiver components utilize 28-volt d.c. from the emergency bus, and 200/115 volts from the No. 2 a.c. bus, while the indicators operate on

26-volt a.c. from the instrument auto-transformer. A control panel for operation of the equipment is installed on the right console (24, figure 1-8), and two VHF navigation indicators are located on the center of the instrument panel.

AN/ARN-56 CONTROLS.

The VHF navigation control panel (figure 4-4) has a power switch, a frequency selector, and a volume rheostat. The power switch, when moved forward from OFF to POWER ON, puts the set into operation. Frequency selection is made by rotating the concentric knobs until the desired frequency is presented in the vertical indicator window. Starting at the top, the numbers in the window represent hundreds, tens, units, and tenths of megacycles, respectively. Volume can be adjusted as desired by rotating the volume control knob.

AN/ARN-56 INDICATORS.

Radio Magnetic Indicator.

The radio magnetic indicator (1, figure 1-6) located on the left side of the instrument panel includes two needles and a rotating compass card. The compass card is coupled to the J-4 directional indicator so that the aircraft heading will always be indicated under the index mark at the top of the instrument (figure 4-4). Both needles are connected to the AN/ARN-56 receiver and indicate magnetic bearing to the omnirange transmitter whenever VOR signals are being received. The angle between the needles and the index mark at the top of the indicator shows relative bearing to the station.

COURSE INDICATOR.

The course indicator (10, figure 1-6) is mounted on the center instrument panel directly below the directional indicator. There is a set knob in the lower left corner of the course indicator and a marker beacon indicator in the opposite corner (figure 4-4). The course window at the top of the instrument indicates the omni-range radial selected by means of the set knob. The sensing window on the upper left of the indicator tells whether the selected radial leads "TO" or "FROM" the VOR station. The vertical cross-pointer indicates whether the selected radial is to the right or left of the aircraft. The horizontal cross-pointer is a glide slope indicator for use with an ILS receiver. The needle with a circular pointer is a relative heading needle and is coupled to the gyro compass. This needle indicates whether the aircraft is headed to the right or the left of the selected course. Two warning flags marked OFF are incorporated on the face of the indicator to show when course and glide slope signals are weak, unreliable, or non-existent.

VHF NAVIGATION GROUND CHECK.

1. VHF power switch—ON.

2. VHF frequency selector knob—Tune to nearest omni-range station. Adjust volume control to check correct station identification.

3. Radio magnetic indicator—Note that compass card reads the airplane heading and that one needle swings to bearing of selected station.

4. Course indicator—Check that warning "OFF" flag has retracted from vertical bar after equipment has had a 2 minute warmup.

5. Course set knob—Rotate to set bearing to VOR station in course window. Note that vertical bar centers, and that sensing window reads "TO." Note that relative heading needle is displaced to same side of the station as the airplane's heading. Rotate course set knob to set up radials 7 degrees to right and left, and note that vertical bar moves promptly and smoothly to full left and right deflection. Continue rotating course set knob. When difference exceeds 90 degrees note that the vertical bar crosses to the opposite side of the instrument, and sensing window shows "FROM." When reciprocal radial is reached, note that vertical bar comes to center.

6. VHF frequency selector knob—Tune to nearest VAR or localizer transmitter and note that vertical bar makes correct responses.

7. VHF power switch—OFF.

VHF NAVIGATION OPERATION.

1. VHF power switch—ON.

2. VHF frequency selector knob—Rotate to select frequency.

3. VHF volume control knob—Adjust as desired.

4. VHF power switch—OFF, to turn set off.

VHF NAVIGATION—OPERATION WITH VOR.

1. VHF power switch—ON.

2. VHF frequency selector knob—Set for desired VOR station. Allow 2 minutes for warning "OFF" flag to retract from vertical bar.

3. Course set knob—Rotate to center vertical bar. Read radial in course window and identify it as course "TO" or "FROM" the station as indicated in sensing window. Read relative heading needle to determine whether aircraft is headed right or left of course. If reciprocal is desired, rotate course set knob to add or subtract 180 degrees; read course and sensing as now indicated. To fly on a radial other than the one the airplane is on, set up desired radial in course window. Vertical bar will then be deflected toward new radial. Fly toward vertical bar to arrive at desired radial, then turn onto course as bar centers. Adjust heading as necessary to compensate for drift. As long as vertical bar is centered, airplane is tracking along displayed radial, regardless of heading. Relative heading needle will indicate drift angle. When airplane crosses station while tracking along displayed radial, sensing will reverse with no changes in other indications of the instrument. When airplane is not tracking along displayed radial, vertical bar will be off center. In such a case, bar will swing to opposite side when airplane crosses displayed radial. To turn smoothly onto radial, steer to hold point of relative heading needle on vertical bar until both are centered. Sensing will reverse when airplane crosses the radial, that is, at 90 degrees to displayed radial.

VHF NAVIGATION—OPERATION WITH VAR.

1. VHF power switch—ON.

2. VHF frequency selector knob—Set to desired VAR station. Allow 2 minutes for warning "OFF" flag to retract from vertical bar.

3. Note deflection of vertical bar. If bar deflects to left, airplane is in blue sector of range; if bar is to right, airplane is in yellow sector. Consult airways chart to identify sector.

Note

On VAR, the deflection of the vertical bar does not in itself indicate the direction in which to fly to get on course. It indicates merely the color sector in which the airplane is flying.

4. Identify signal in headphones as aural N or A, and consult airways chart to determine whether station is ahead or astern. If aural signals overlap to give a continuous dash, airplane is on aural leg at right angles to visual range.

5. Relative heading needle indicates heading relative to course selected.

Note

Blue and yellow sectors are assigned to opposite sides of the visual range in accordance with the course defined by the airway. At certain terminal airports, VAR is used in the absence of a localizer. In such cases, the sector orientation is the same as for an ILS localizer. That is, the blue sector is charted on the right and the yellow sector is charted on the left when the airplane is inbound on final approach, regardless of the course defined by the beam.

GLIDE PATH AND MARKER BEACON RECEIVER AN/ARN-57.

A glide slope receiver is installed in the airplane for instrument landings. Selection of proper frequencies for an indication on the horizontal cross-pointer of the course indicator (figure 4-4) is automatic and needs no further action, on the part of the pilot, than selection of a frequency authorized for the place of landing on the AN/ARN-56 frequency selector knob. The horizontal cross-pointer will indicate to the pilot whether the airplane is above, below or on glide slope during an ILS approach to landing. The vertical cross-pointer will indicate whether the airplane is to the left, right or on course during an ILS approach to landing. The AN/ARN-56 will be used for localizer information. The marker beacon receiver is automatically turned on when power is supplied to the electrical system of the airplane. A marker beacon indicator light is located on the upper right corner of the course indicator (figure 4-4). The marker beacon receiver is powered through the d.c. emergency bus.

OPERATION WITH THE GLIDE PATH.

1. VHF power switch—ON.

2. VHF frequency selector knob—Set to localizer station. Allow 2 minutes for warning "OFF" flag to retract from horizontal bar.

3. Note deflection of horizontal bar. If the horizontal bar is deflected up, the airplane is below the glide slope; if the bar is deflected down, the airplane is above the glide slope; if the bar is centered, the airplane is on glide slope. Always fly the airplane toward the bar

OPERATION WITH LOCALIZER.

1. VHF power switch—ON.

2. VHF frequency selector knob—Set to localizer station. Allow 2 minutes for warning "OFF" flag to retract from vertical bar.

3. Note deflection of vertical bar. If vertical bar is deflected to left, airplane is in blue sector of localizer range; if bar is to right, airplane is in yellow sector. Blue sector of a localizer is always charted to the right of the inbound course; therefore, a pilot on final approach can center on the beam by flying toward the bar.

4. Relative heading needle indicates required correction angle.

VHF NAVIGATION—OPERATION FOR COMMUNICATIONS.

The receiver can be tuned to the appropriate transmitter to receive weather broadcasts, tower instructions, and general communications.

IDENTIFICATION RADAR AN/APX-35 IFF SYSTEM.

An AN/APX-35 IFF radar set is installed to automatically identify the airplane as friendly whenever it is properly challenged by suitably equipped friendly air or surface forces. Two control panels (22, figure 1-8) are in the cockpit, on the right console. The IFF is powered from the a.c. emergency bus and the d.c. emergency bus. Information regarding the description of the system units will be added when available.

OPERATION OF AN/APX-35 IFF SYSTEM.

1. On IFF control panel (figure 4-4), rotate master switch to STDBY and allow 3 to 4 minutes for equipment to warm up.

2. Select NORM position on master switch.

3. Set mode 2 and mode 3 switches as required.

4. Set I/P—MIC switch as required.

5. On SIF control panel (figure 4-4), set mode 1 and mode 3 switches as required.

6. To turn equipment off, rotate master switch on IFF control panel to OFF.

LIGHTING EQUIPMENT.

EXTERIOR LIGHTING.

Landing and Taxi Lights.

Two landing lights are provided, one on each main gear aft door. The lights will be in position for use any time the landing gear is extended. A taxi light is installed on the nose gear shock strut and will also be available whenever the gear is extended. These lights receive power from the No. 2 a.c. bus and are controlled by a switch (figure 1-10) on the left forward panel. The switch has three positions, LANDING LT, OFF, and TAXI LT. When the switch is placed up to the LANDING LT position, both the landing lights and the taxi light will be energized. Moving the switch down to the TAXI LT position will cause only the taxi light to be energized. On modified aircraft, the lights automatically shut off when the gear is retracted.

Navigation Lights and Switches.

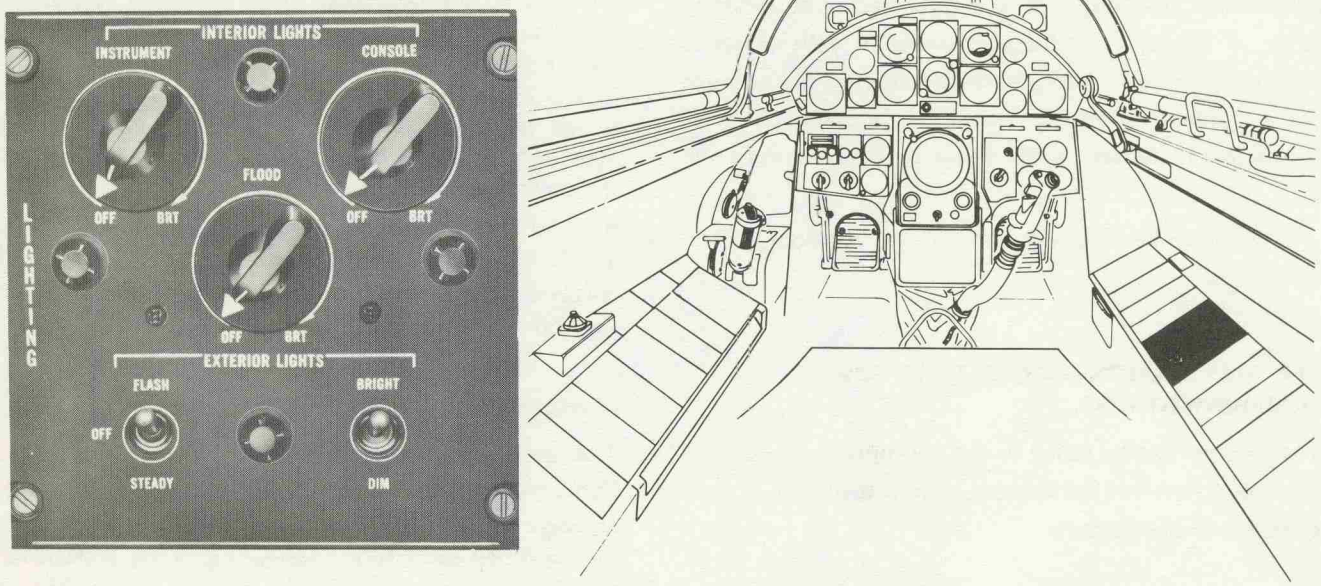
The navigation lights include two yellow upper tail lights, two white lower tail lights, a green and a red fuselage light, and white top and bottom fuselage lights. On modified aircraft formation lights are installed on the missile launchers or on the tip tanks when installed. The lights are controlled by a selector switch and a dimming switch located on the right console (21, figure 1-8). The lights will be energized by 28-volt a.c. from the No. 2 a.c. bus auto-transformer when the selector switch (figure 4-5) is moved from OFF to STEADY or FLASH. Intensity of the navigation lights will depend upon the position (BRIGHT or DIM) of the dimming switch. With the selector switch in the STEADY position, all of the navigation lights will be continuously energized. 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.

Cockpit Lighting System.

The cockpit lighting system includes instrument lights, console panel lights, console floodlights, thunderstorm lights, utility spotlights, and associated wiring, circuit breakers and controls. The spotlights and floodlights receive power from the No. 2 a.c. bus through the auto-transformer while the remainder of the cockpit lighting system is powered directly from the emergency a.c. bus. The instruments are individually illuminated by lights placed in shields above each instrument on the panel. The consoles are lighted by recessed lamps so that all of the controls and decals on each panel

LIGHTING CONTROL PANEL



AM2129
F-50-B-4-50

Figure 4-5

are readily discernible. The console floodlights are provided to light the left and right consoles directly. The thunderstorm lights are located on the left and right consoles and direct light forward onto the instrument panels to overcome the effects of blindness caused by lightning. One C-4A type utility spotlight is mounted on each side of the cockpit above the right and left consoles. These spotlights are detachable and may be moved about the cockpit to take care of special lighting situations. A rheostat on the light itself is used to vary the intensity of the spotlight. In addition to this, a push-button switch enables the pilot to bypass the rheostat and obtain maximum spotlight brilliance instantaneously. The instrument lights, console panel lights and floodlights are controlled by three dimming rheostats (figure 4-5) located on the right console. These rheostats, labeled INSTRUMENT, CONSOLE, and FLOOD, may be rotated clockwise from OFF to BRT, as desired to vary the intensity of the associated lights. Both thunderstorm lights are controlled by a single ON-OFF switch (15, figure 1-8) on the right console.

OXYGEN SYSTEM.

GASEOUS OXYGEN.

(AF Serials 56-764 through 56-788.)

A supply of gaseous oxygen is stored in three cylinders located in the fuselage aft of the nose wheel well. Oxygen is introduced to the cylinders through a filler valve which can be reached from the ground through an access door aft of the nose wheel well. When fully serviced, oxygen pressure in these storage cylinders should be 1800 (±50) psi.

Note

Under conditions of extreme cold, such as would be encountered at high altitudes, the oxygen cylinders will become chilled and the oxygen pressure gage indication will drop, sometimes rather rapidly. This should not cause undue alarm since the normal supply of oxygen will still be available and the pressure will tend to rise again when the aircraft

OXYGEN SUPPLY SYSTEM

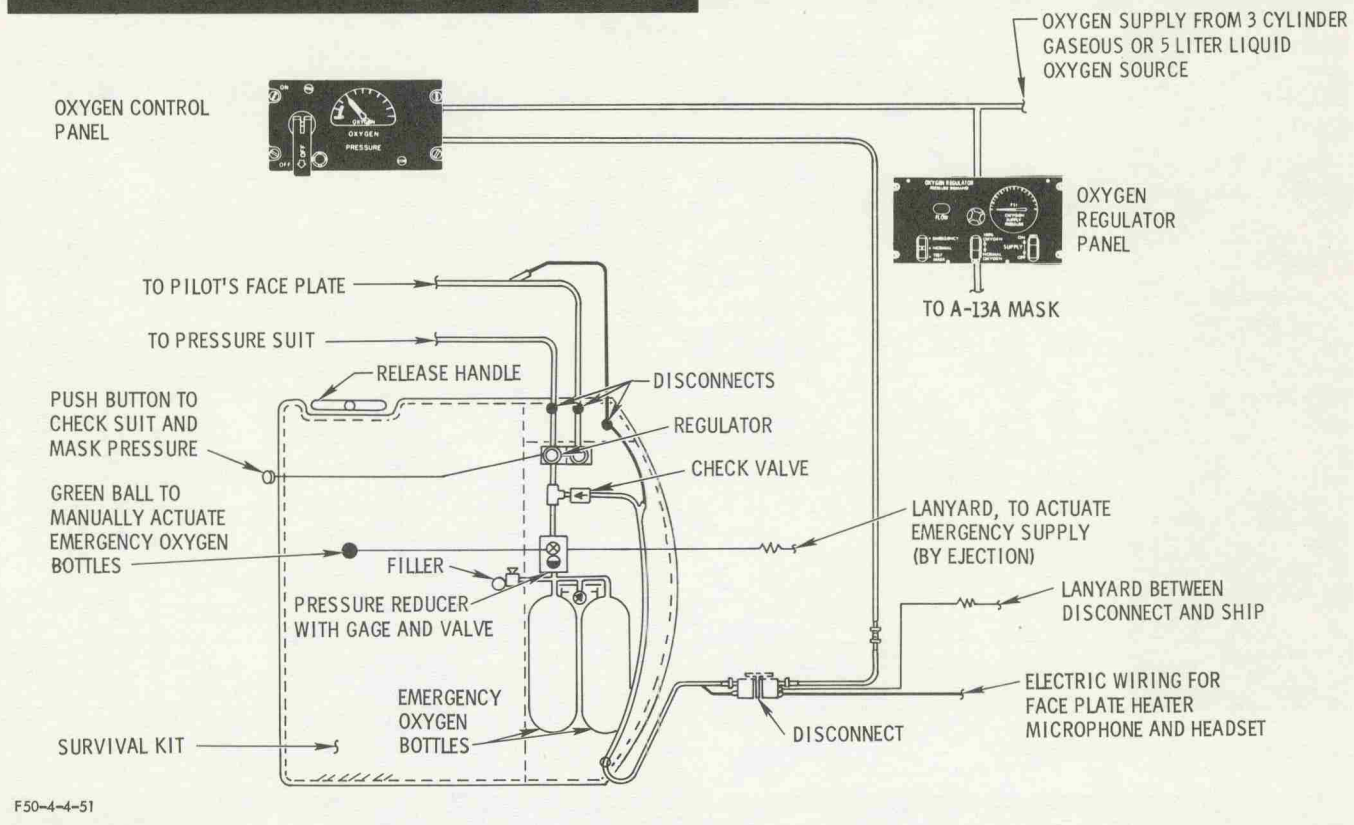


Figure 4-6

descends to warmer altitudes. If oxygen pressure decreases rapidly at low altitudes or during a descent, it is indicative of a leak in the system and a thorough check should be made to determine the cause.

Flow of oxygen is directed to a pressure reducer and control panel on the right console where the pressure is reduced to approximately 70 psi.

LIQUID OXYGEN.

(AF Serials 56-789 and subsequent.)

A supply of liquid oxygen is stored in an insulated converter located aft of the nose wheel well. The system is serviced through a filler valve which can be reached through an access door aft of the nose wheel well. The converter has a capacity of 5 liters (1.3 gallons). Liquid oxygen is converted to gas which then passes through a heat exchanger to make it suitable for breathing. Oxygen is delivered to the oxygen control panel at a pressure of approximately 70 psi.

SURVIVAL KIT CONNECTION.

From the oxygen control panel, oxygen flows through a disconnect on the left side of the pilot's seat to a manifold assembly in the rear section of the survival kit located in the seat bucket (figure 4-6.) This section of the survival kit also contains two emergency oxygen bottles connected to the manifold assembly, and a regulator unit. The manifold assembly includes a pressure gage, relief valve, check valve and filler valve. Either normal aircraft oxygen or emergency oxygen can be directed from the manifold assembly to the regulator unit which sends oxygen pressure to the pilot's pressure

OXYGEN DURATION—HOURS

COCKPIT ALTITUDE (FEET)	GAGE PRESSURE—P.S.I. 1800	1600	1400	1200	1000	800	600	400	BELOW 400
40,000 & UP	3.2	2.8	2.5	2.1	1.8	1.4	1.1	.7	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
35,000	3.2	2.8	2.5	2.1	1.8	1.4	1.1	.7	
30,000	2.5	2.2	1.9	1.6	1.4	1.1	.8	.5	
25,000	1.9	1.7	1.5	1.3	1.0	.8	.6	.4	
20,000	1.5	1.3	1.1	1.0	.8	.6	.5	.3	
15,000	1.2	1.0	.9	.8	.6	.5	.4	.3	
10,000	.9	.8	.7	.6	.5	.4	.3	.2	
5,000	.8	.7	.6	.5	.4	.3	.3	.2	
SEA LEVEL	.6	.5	.5	.4	.3	.3	.2	.1	

3 CYLINDERS : AN 6025 AX147

BREATHING RATES USING SURVIVAL KIT OXYGEN REGULATOR AND PRESSURE SUIT

OXYGEN DURATION—HOURS

COCKPIT ALTITUDE (FEET)	GAGE QUANTITY—LITERS 5.0	4.5	4.0	3.5	3.0	2.5	2.0	1.5	1.0	.5	BELOW .5
35,000 & UP	16.6	14.9	13.3	11.6	10.0	8.3	6.6	5.0	3.3	1.7	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
30,000	12.6	11.3	10.1	8.8	7.6	6.3	5.0	3.8	2.5	1.3	
25,000	9.6	8.7	7.7	6.7	5.8	4.8	3.9	2.9	1.9	1.0	
20,000	7.5	6.7	6.0	5.2	4.5	3.7	3.0	2.3	1.5	.8	
15,000	5.9	5.3	4.7	4.1	3.5	3.0	2.4	1.8	1.2	.6	
10,000	4.7	4.3	3.8	3.3	2.8	2.4	1.9	1.4	.9	.5	
5,000	3.9	3.5	3.1	2.7	2.3	1.9	1.5	1.2	.8	.4	
SEA LEVEL	3.2	2.8	2.5	2.2	1.9	1.6	1.3	1.0	.6	.3	

5 LITER LIQUID OXYGEN CONVERTER

BREATHING RATES USING SURVIVAL KIT OXYGEN REGULATOR AND PRESSURE SUIT

F50-4-4-52

Figure 4-7

OXYGEN CONTROL PANELS

(GASEOUS OXYGEN)

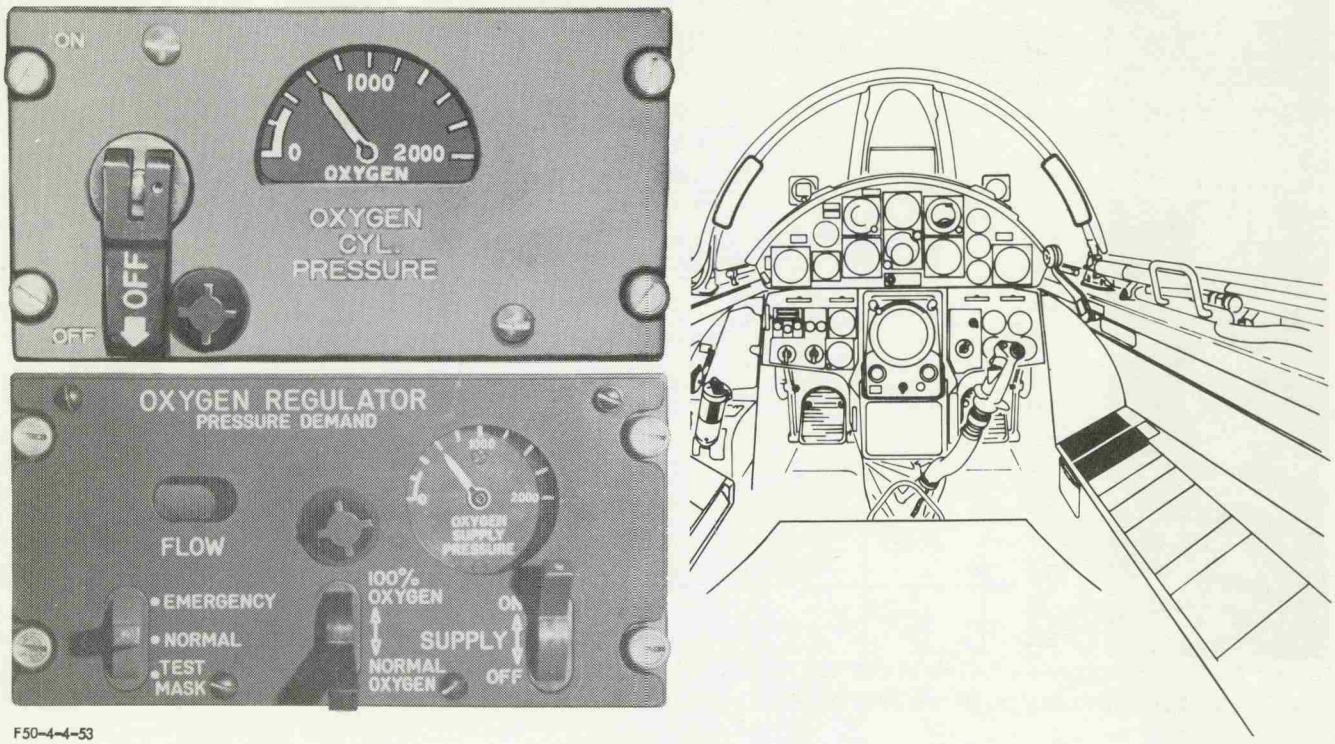


Figure 4-8

suit as well as delivering the correct amount of breathing oxygen to the pilot's face plate hose, from sea level up to 70,000 feet. The ⅜ inch hoses from the regulator to the pilot leave the survival kit at the right rear corner. These hoses are designed for use with type MC-3 and MC-4 pressure suits, and with MA-2 or Modified K-1 helmets.

WARNING

A partial pressure suit will be worn for all flights above 45,000 feet ambient altitudes.

A press-to-test button is located on the right front of the survival kit to test operation of the oxygen supply system. If there is no malfunction, oxygen will flow through the pressure suit and face plate hoses when the button is depressed.

OXYGEN SYSTEM CONTROL PANEL.

(AF Serials 56-764 through 56-788.)

An oxygen pressure gage and supply valve lever are located on the oxygen system control panel (figure 4-8) on the right console. The supply lever has two positions, ON and OFF. Placing the lever in the ON position opens the supply valve and allows oxygen to flow through the pressure reducer and into the system. Moving the lever to OFF closes the supply valve, shutting off the normal oxygen supply. The pressure gage indicates oxygen in the normal supply cylinders and should read 1800 (± 50 psi) when the system is fully serviced before flight.

OXYGEN SYSTEM CONTROL PANEL.

(AF Serial 56-789 and subsequent).

An oxygen pressure gage and supply valve lever are located on the oxygen system control panel (figure 4-10) on the right console. The supply lever has two positions, ON and OFF. Placing the lever in the ON position opens the supply valve and allows oxygen to flow through the

MD-2 OXYGEN DURATION -HOURS

COCKPIT ALTITUDE (FEET)	GAGE PRESSURE–P.S.I. 1800	1600	1400	1200	1000	800	600	400	BELOW 400
40,000 & UP	6.0	5.4	4.7	4.0	3.3	2.7	2.0	1.3	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
	6.0	5.4	4.7	4.0	3.3	2.7	2.0	1.3	
35,000	6.0	5.4	4.7	4.0	3.3	2.7	2.0	1.3	
	6.0	5.4	4.7	4.0	3.3	2.7	2.0	1.3	
30,000	4.4	3.8	3.5	2.9	2.4	2.0	1.5	.9	
	4.5	4.0	3.5	2.9	2.5	2.0	1.5	.9	
25,000	3.5	2.9	2.5	2.2	1.8	1.5	1.1	.7	
	4.2	3.8	3.3	2.9	2.4	1.8	1.5	.9	
20,000	2.5	2.4	2.0	1.6	1.5	1.1	.9	.5	
	4.7	4.2	3.6	3.3	2.7	2.2	1.6	1.1	
15,000	2.0	1.8	1.6	1.5	1.1	.9	.7	.5	
	5.8	5.1	4.5	3.8	3.3	2.5	2.0	1.3	
10,000	1.6	1.5	1.3	1.1	.9	.7	.5	.4	
	5.8	5.1	4.5	3.8	3.3	2.5	2.0	1.3	
5,000	1.3	1.1	1.1	.9	.7	.5	.4	.4	
	5.8	5.1	4.5	3.8	3.3	2.5	2.0	1.3	
SEA LEVEL	1.1	.9	.9	.7	.5	.5	.4	.2	
	5.8	5.1	4.5	3.8	3.3	2.5	2.0	1.3	

UPPER FIGURES INDICATE DILUTER LEVER 100% OXYGEN
LOWER FIGURES INDICATE DILUTER LEVER NORMAL OXYGEN

(BREATHING RATES FOR A-13A PRESSURE DEMAND MASK)

3 CYLINDERS. AN 6025 AX 147

MD-1 OXYGEN DURATION-HOURS

COCKPIT ALTITUDE (FEET)	GAGE QUANTITY—LITERS 5.0	4.5	4.0	3.5	3.0	2.5	2.0	1.5	1.0	.5	BELOW .5
35,000 & UP	31.4	28.3	25.1	22.0	18.9	15.7	12.6	9.4	6.3	3.1	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
	31.4	28.3	25.1	22.0	18.9	15.7	12.6	9.4	6.3	3.1	
30,000	22.6	20.4	18.1	15.9	13.6	11.3	9.1	6.8	4.5	2.3	
	23.3	21.0	18.6	16.3	14.0	11.6	9.3	7.0	4.7	2.3	
25,000	17.5	15.7	14.0	12.2	10.5	8.7	7.0	5.2	3.5	1.8	
	22.0	19.8	17.6	15.4	13.2	11.0	8.8	6.6	4.4	2.2	
20,000	13.3	12.0	10.6	9.3	8.0	6.7	5.3	4.0	2.7	1.3	
	24.9	22.4	19.9	17.4	14.9	12.4	10.0	7.5	5.0	2.5	
15,000	10.7	9.6	8.5	7.5	6.4	5.3	4.3	3.2	2.1	1.1	
	30.2	27.2	24.1	21.1	18.1	15.1	12.1	9.1	6.0	3.0	
10,000	8.6	7.7	6.9	6.0	5.1	4.3	3.4	2.6	1.7	.9	
	30.2	27.2	24.1	21.1	18.1	15.1	12.1	9.1	6.0	3.0	
5,000	6.7	6.3	5.6	4.9	4.2	3.5	2.8	2.1	1.4	.7	
	30.2	27.2	24.1	21.1	18.1	15.1	12.1	9.1	6.0	3.0	
SEA LEVEL	5.7	5.1	4.6	4.0	3.4	2.9	2.3	1.7	1.1	.6	
	30.2	27.2	24.1	21.1	18.1	15.1	12.1	9.1	6.0	3.0	

UPPER FIGURES INDICATE DILUTER LEVER 100% OXYGEN
LOWER FIGURES INDICATE DILUTER LEVER NORMAL OXYGEN

(BREATHING RATES FOR A-13A PRESSURE DEMAND MASK)

5 LITER LIQUID OXYGEN CONVERTER

F50-4-4-100

Figure 4-9

OXYGEN CONTROL PANELS

(LIQUID OXYGEN)

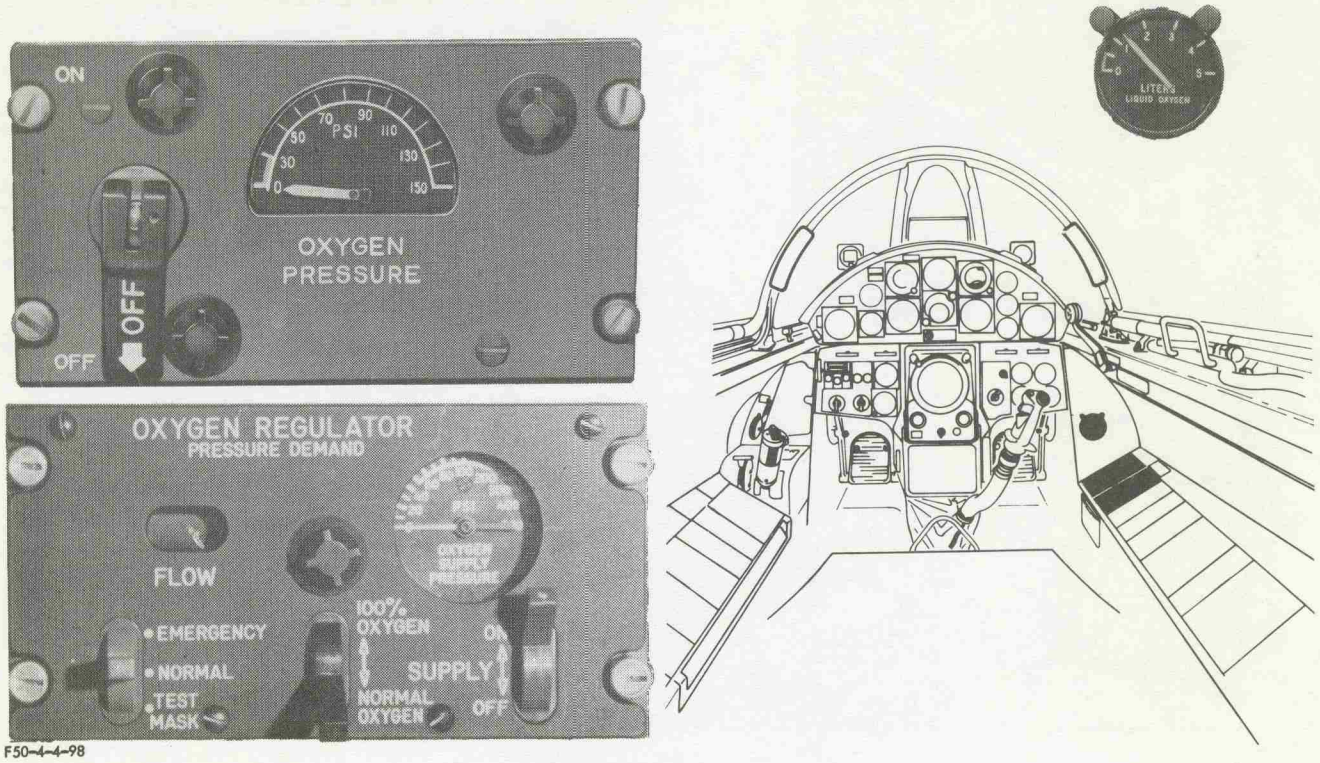


Figure 4-10

panel into the system. Moving the lever to OFF closes the supply valve, shutting off the oxygen supply. The gage indicates gaseous oxygen pressure at the panel and should read approximately 70 psi. The gage is calibrated from 0 to 150 psi.

LIQUID OXYGEN QUANTITY GAGE.

(AF Serial 56-789 and subsequent.)

The liquid oxygen quantity gage is on the right forward panel (figure 4-10). It measures liquid content of the oxygen converter and is calibrated in liters from 0 to 5.

OXYGEN REGULATOR PANEL.

All modified aircraft incorporate a type MD-1 or MD-2 combination pressure-breathing, diluter-demand oxygen regulator (figure 4-8 and 4-10) on the right console. This regulator is provided for use with the A-13A oxygen mask when the pressure suit is not worn. 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. At high altitudes, the regulator supplies oxygen at continuous positive pressure to the pilot. The delivery pressure automatically changes with cockpit altitude. Also on the regulator panel are an oxygen pressure gage and flow indicator.

Diluter Lever. The diluter lever (figure 4-8 and 4-10), painted black, is on the oxygen regulator control panel. The lever should be NORMAL, for normal oxygen use, or at the 100% position for emergency oxygen use.

Emergency Lever. The emergency lever (figure 4-8 and 4-10), on the oxygen regulator panel, is red and should be in the center NORMAL position at all times, unless an unscheduled oxygen pressure increase is desired. Moving the lever to EMERGENCY provides continuous positive pressure to the mask. When the lever is held at TEST, oxygen at positive pressure is provided to test the mask for leaks.

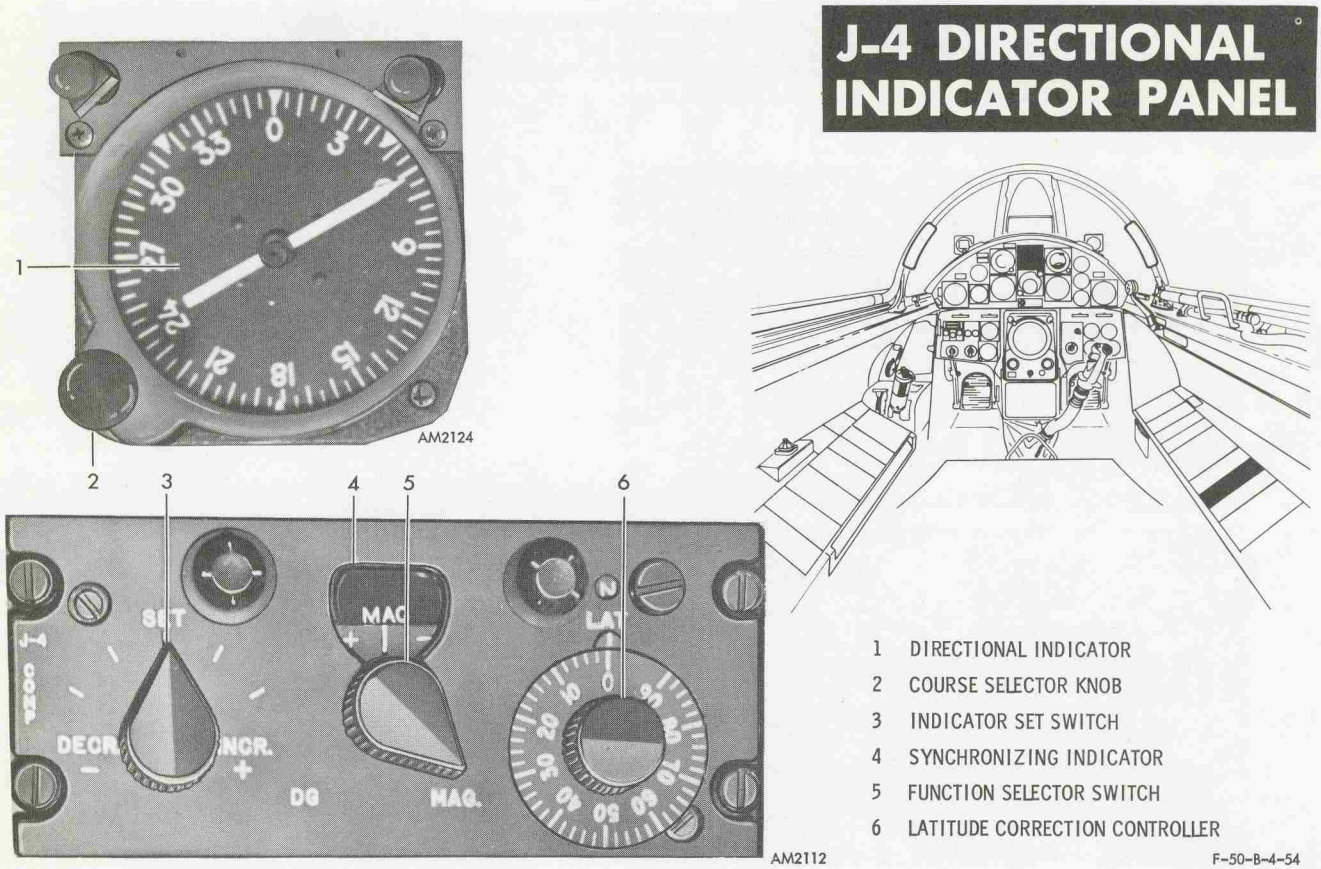


Figure 4-11

CAUTION

When positive pressures are required, it is mandatory that oxygen mask be well fitted to the face. Unless special precautions are taken to insure no leakage, continued use of positive pressure will result in rapid depeletion of the oxygen supply. This rapid depletion could result in extremely cold oxygen flowing to the mask.

Supply Lever. The supply lever (figure 4-8 and 4-10), painted green, is on the oxygen regulator panel. The lever has two positions (ON and OFF) and is used to control the supply of oxygen at the panel.

Pressure Gage and Flow Indicator. The pressure gage and flow indicator (figure 4-8 and 4-10) are on the oxygen regulator control panel. 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 and white alternately during the breathing cycle.

EMERGENCY OXYGEN SUPPLY.

An emergency supply of oxygen, for use in the event of failure of the normal supply system, or for high altitude ejection, is stored in two bottles (cylinders) in the survival kit. The emergency oxygen supply from these bottles goes through the manifold assembly which incorporates a pressure reducer for the emergency system. The emergency supply system is actuated automatically, during ejection, or manually, whenever desired. Upon ejection, the downward movement of the seat creates a force which triggers the disconnect mechanism, separating the seat, survival kit, and associated oxygen equipment from the normal oxygen supply line. Simultaneously, a cable attached to the right knee guard is actuated, opening the emergency oxygen pressure-reducer valve in the manifold assembly. This causes the emergency oxygen supply to flow through this reducer-valve (where pressure is reduced from 1800 psi to 70 psi), and into the regulator. If the emergency bottles are fully serviced (1800 psi), they will provide the pilot with a regulated supply of oxygen for pressure suit, as well as breathing, for a minimum of fifteen minutes. The emergency oxygen supply is

actuated manually by pulling a green ball handle located on the parachute harness. The emergency pressure-reducer valve is then opened by a cable attached to the ball handle. Oxygen pressure in the emergency bottles can be determined by means of a pressure gage attached to the manifold assembly in the survival kit. This is accessible through a hinged door on top of the rear section of the kit.

Note

When using the diluter demand system and A-13A oxygen mask, the only emergency oxygen supply available is from a bail-out bottle attached to the parachute.

OXYGEN SYSTEM PREFLIGHT CHECK.

Note

Normal oxygen pressure is 65-75 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 preflight condition can vary from 70 to 120 psi, depending on the time span 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.

WARNING

If the diluter demand oxygen system is not to be used, the supply lever on the oxygen regulator panel must be placed in the OFF position. If left in the ON position, the regulator will automatically allow positive pressure oxygen flow above 25,000 feet cockpit altitude, which will rapidly deplete the oxygen supply.

Pressure Suit.

1. Oxygen pressure—1800 ±50 psi, if liquid oxygen—Approximately 70 psi.
2. Liquid oxygen quantity gage—4 liters minimum.
3. Quick disconnect on aft part of seat back—Secure, lanyard attached to airplane structure.
4. Manual disconnect knob on forward part of seat back—Locked (pin holes forward).
5. Automatic emergency oxygen actuator cable—Attached to right knee guard.
6. Survival kit emergency oxygen pressure 1800 psi—Check through window in aft section of kit.
7. Check that survival kit cover and straps are connected to lower part of kit.
8. Check security and connection of fittings and hoses at right aft corner of kit.
9. Check between sides of seat and kit for obstructions.
10. Connect and lock kit straps to parachute harness.
11. Attach and lock oxygen and radio connections.
12. Oxygen supply valve lever—ON.
13. Press-to-test button—Depress and note positive pressure to suit and face plate.

Diluter Demand

1. Survival kit oxygen supply valve lever—OFF.
2. Oxygen pressure—1800 ±50 psi; if liquid oxygen—70 psi.
3. Liquid oxygen quantity gage—4 liters minimum.
4. Check oxygen regulator with diluter lever first at NORMAL and then at 100% as follows: Blow gently into end of oxygen regulator hose as during normal exhalation. There should be resistance to blowing. Little or no resistance to blowing indicates a leak or faulty operation.
5. With supply lever ON, oxygen mask connected to regulator, and diluter lever at 100%, breathe normally into mask and conduct following checks:
   a. Observe flow indicator for proper operation.
   b. Deflect emergency lever to EMERGENCY.

A positive pressure should be supplied to the mask. Hold breath to determine whether there is leakage around mask. Return emergency lever to NORMAL position. Positive pressure should cease.

6. Return diluter lever to NORMAL.

Note

All connections and checks should be made with the assistance of a personal equipment man or someone qualified to perform the function.

NORMAL OPERATION OF THE OXYGEN SYSTEM.

Pressure Suit.

1. Oxygen pressure gage—1800 ±50 psi, if liquid oxygen—Approximately 70 psi.

2. Liquid oxygen quantity gage—4 liters minimum.

3. Oxygen supply valve lever—ON.

Diluter Demand.

1. Oxygen pressure gage—1800 ±50 psi; if liquid oxygen—Approximately 70 psi.

2. Liquid oxygen quantity gage—4 liters minimum.

3. Oxygen supply lever—ON.

4. Diluter lever—NORMAL.

5. Emergency lever—NORMAL.

EMERGENCY OPERATION OF THE OXYGEN SYSTEM.

Pressure Suit.

Any time that symptoms of hypoxia are noted, or if it is known that the normal oxygen system has failed, proceed as follows:

1. Manual emergency oxygen supply actuator—Pull.

2. Descend to an altitude where oxygen is not required.

Diluter Demand.

1. Move diluter lever to 100%.

2. Push emergency lever to EMERGENCY.

3. If oxygen regulator becomes inoperative, pull the green ball handle on bail-out bottle to actuate the emergency oxygen supply.

4. Descend to an altitude where oxygen is not required.

NAVIGATION EQUIPMENT.

J-4 DIRECTIONAL INDICATOR SYSTEM.

The J-4 directional indicator system provides the pilot with a directional reference by operating either as a directional gyro or a slaved gyro magnetic compass. The control panel (19, figure 1-8) contains all of the controls necessary for proper operation of the system under all conditions. The system requires 28-volt d.c. from the emergency bus, as well as instrument a.c. bus power, for normal operation.

FUNCTION SELECTOR SWITCH.

A function selector switch (5, figure 4-11) is provided to select the mode of operation of the directional indicator system. When this switch is in the MAG. position, the system will operate as a normal slaved gyro magnetic compass, and the directional indicator will respond accordingly. Placing the function 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 will be very unreliable. If it is deemed necessary, the system can be sent into a fast slaving cycle by moving the function selector switch from MAG to DG and back to MAG. The gyro will then be erected, and any large errors in the system eliminated very rapidly.

INDICATOR SET SWITCH.

The indicator set switch (3, figure 4-11) is used during the magnetic compass mode of operation to effect synchronization of the compass system and the transmitter which feeds heading information to the directional indicator. This control is used during DG operation to position the transmitter to any desired reference.

Synchronizing Indicator.

A synchronizing indicator (4, figure 4-11) is provided to indicate synchronization to within plus or minus one quarter degree between the compass system and the transmitter in the magnetic mode of operation. When the system is not synchronized the synchronizing indicator will show the direction in which the indicator set switch must be turned to achieve the necessary correction. The set switch must be turned clockwise when the indicator is in the (+) region and counterclockwise when a (−) indication appears.

LATITUDE CORRECTION CONTROLLER.

The latitude correction controller (6, figure 4-11) is used in the DG mode of compass operation to correct for the apparent drift of the directional gyro due to the earth's rotation. Latitude corrections are made in a clockwise direction in the Northern Hemisphere and counterclockwise in the Southern Hemisphere.

DIRECTIONAL INDICATOR AND COURSE SELECTOR KNOB.

The directional indicator (1, figure 4-11) on the instrument panel receives heading information from the compass system and gives the pilot visual indication of compass heading by means of an indicator pointer and

a compass card. Any desired course may be placed under the index at the top of the indicator by turning the course selector knob at the lower left corner of the indicator (2, figure 4-11).

ARMAMENT EQUIPMENT.

Refer to Confidential Supplement, T.O. 1F-104A-1A

MISCELLANEOUS EQUIPMENT.

ANTI-G SUIT EQUIPMENT.

The anti-G suit equipment consists of a pressure regulating valve and valve control (2, figure 1-7) on the left console and a pressure hose leading from the valve outlet, through the quick disconnect on the left rear of the seat, to the pilot's anti-G suit tube. When this tube is connected to the pressure hose, air from the engine compressor will flow into the pilot's anti-G suit under pressures which will vary in accordance with valve control setting and aircraft G forces. When the valve control is in the LOW position, the valve will open at 1.7 G, and suit pressurization will increase at a rate of 1.0 psi for each additional G force. With the valve control in the HI position, suit pressurization begins at 1.5 G and increases at a rate of 1.5 psi per G. A button on top of the valve control can be manually depressed to inflate the anti-G suit when desired. This feature may be used to produce a massaging effect which will help lessen fatigue during prolonged flight.

VENTILATED SUIT BLOWER.

The blower supplies cooling air to the pilot's ventilated suit. Flexible ducting carries the cool air from the blower, through a quick disconnect on the left side of the seat, to the suit. The blower derives its electrical power from the No. 2 a.c. bus.

Ventilated Suit Blower Switch.

A two-position switch (14, figure 1-8) located on the right console and powered from the No. 2 a.c. bus may be used to energize the ventilated suit blower. The switch positions are labeled ON and OFF.

CHECK LIST.

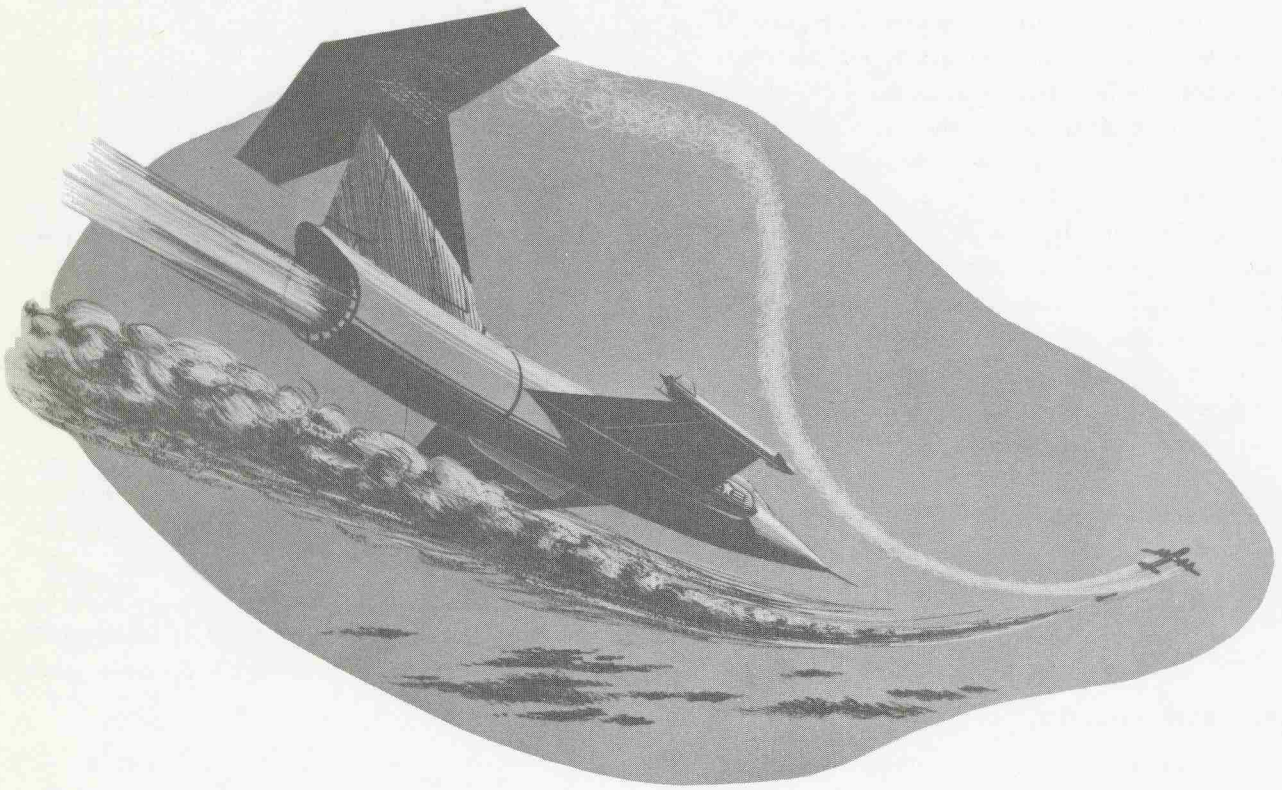
A check list (23, figure 1-8) is located inside a small leather check list case on the lower right side of the cockpit.

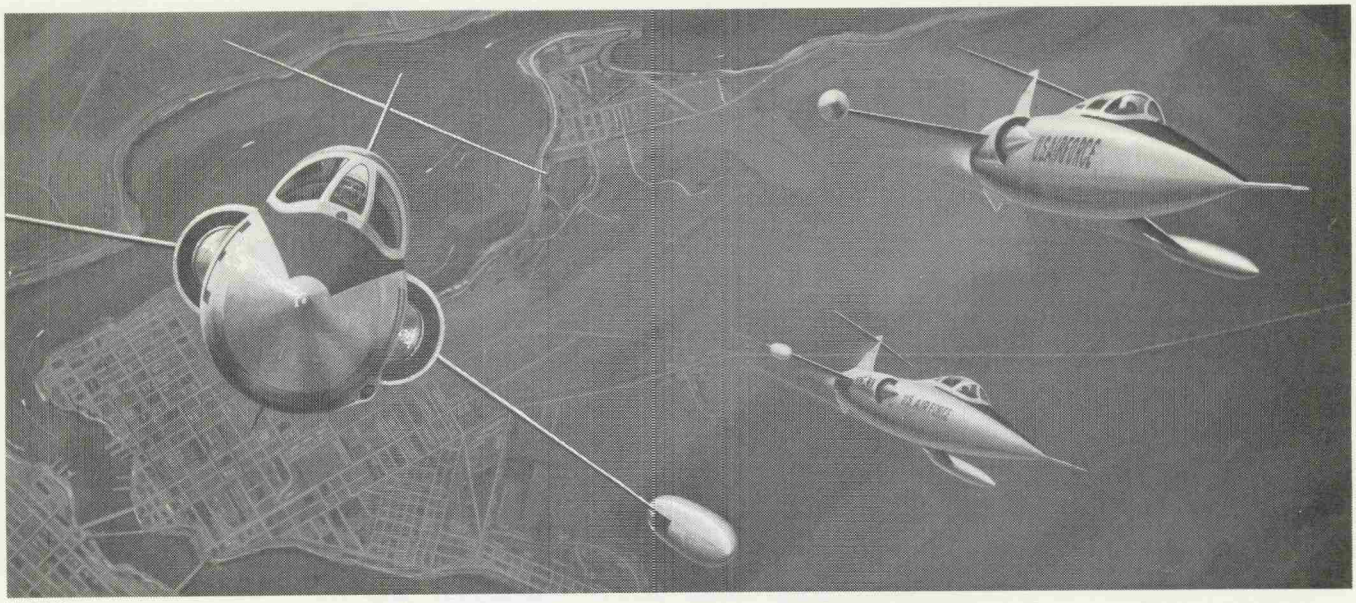
REAR VIEW MIRROR.

Two adjustable rear view mirrors are installed in the cockpit, one on each side of the forward canopy.

COMPUTER.

A manually operated computer (8, figure 1-8) located above the right console is provided to aid the pilot in solving navigational problems. The computer is mounted on a swivel arm so that it may be used whenever necessary and stowed underneath the canopy sill when not in use. Both sides of the computer face may be used to solve problems, and the computer itself is constructed in such a manner as to be completely operable with one hand.



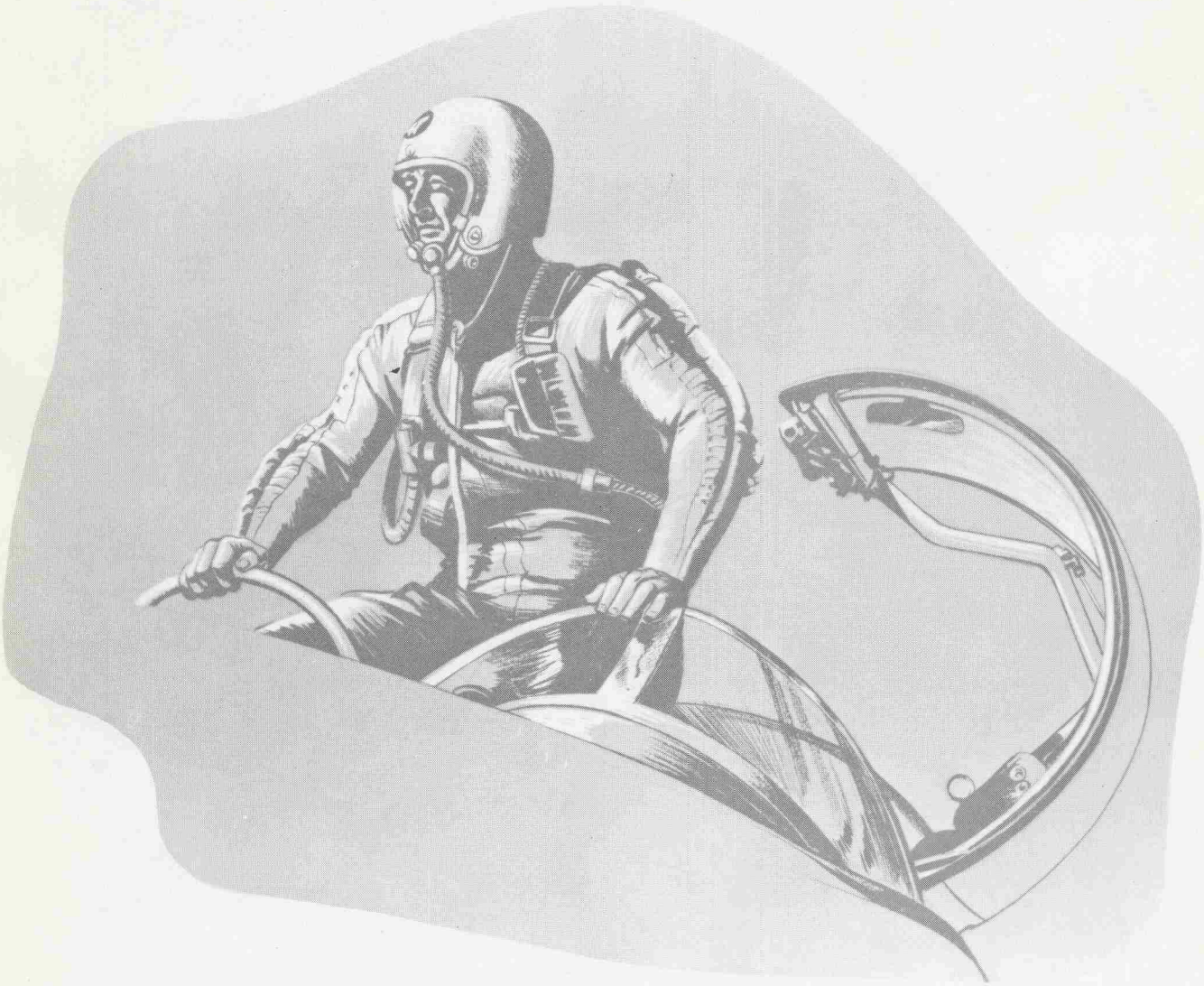


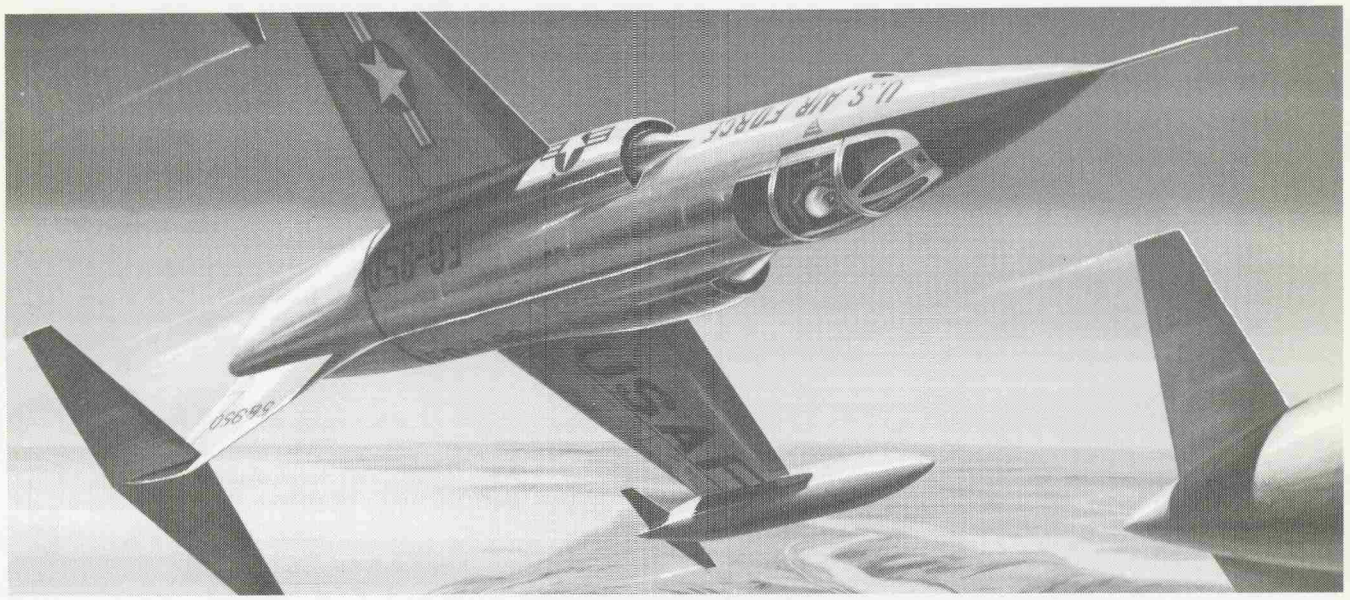
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AM2827

operating limitations

SECTION V

Refer to Confidential Supplement, T.O. 1F-104A-1A





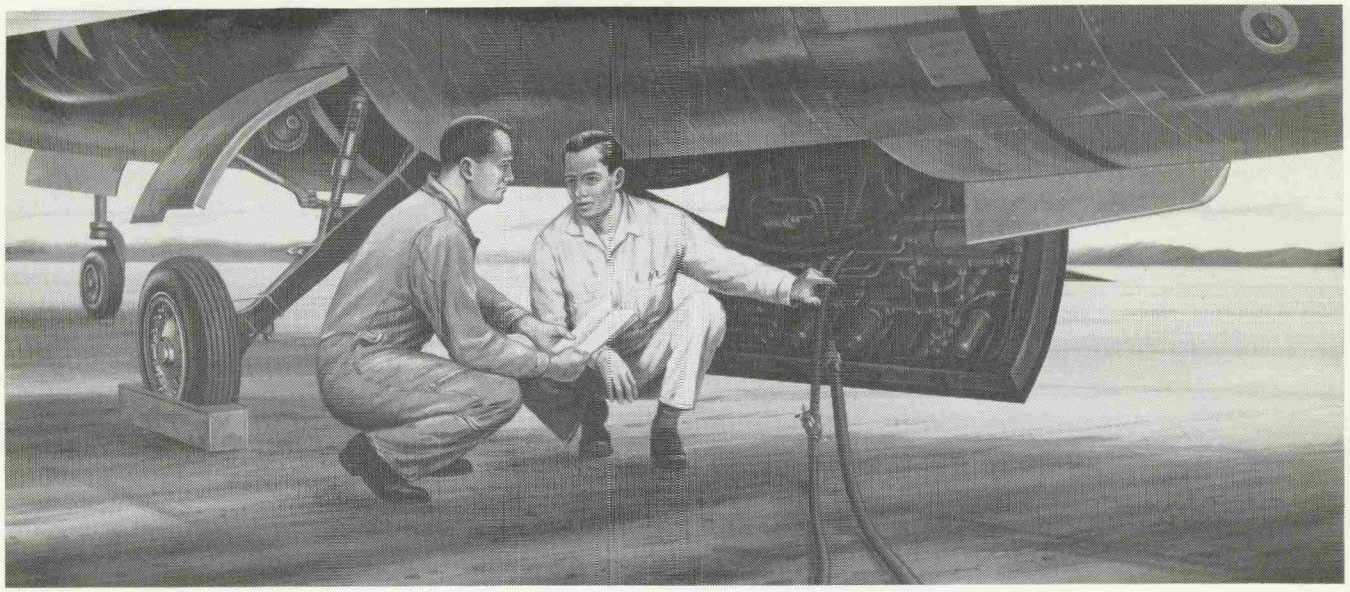
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AM2826

flight characteristics

SECTION VI

Refer to Confidential Supplement, T.O. 1F-104A-1A





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AM2823

systems operation SECTION VII

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COMPRESSOR LOW SPEED STALL.

It is important that operators of the J79-GE-3A engine recognize compressor low-speed or rotating stall so that proper corrective action may be taken. Jet engines in the 10,000 lb. or greater static thrust bracket require high compressor pressure ratios and therefore are subject to low-speed stalls. The compressor passages of the J-79 engine are sized for most efficient operation in the high rpm operating range. Due to this compressor design and lower pressure ratios induced by operating in the low rpm range, the forward stages of the compressor operate in or near the stall range at low rpm while the aft stages operate near the maximum mass flow possible for the pressure ratio existing at the reduced rpm. The stall condition in the first few stages is not complete around the face of the compressor, but instead is localized in several areas over the face. Under some operating conditions these areas have been found to rotate around the compressor face; hence the name "rotating stall." The J-79 engine features variable stators on the inlet stages, enabling them to operate at low angles of attack at low rotor speeds to help avoid low-speed stalls. A normally operating J-79 engine should not exhibit a low-speed compressor stall problem. However, due to fuel schedule errors the problem may be encountered and the pilot should be on the alert for this condition, particularly on initial engine installations. Rotating stall often can be identified by a weak engine rumble and "seat-of-the-pants" vibration accompanied by a slow to moderate rate of rise of exhaust gas temperature. This condition may be encountered in the range between 66-85% rpm. If a throttle burst is made from this condition then a fully-developed compressor stall or surge will result.

ENGINE SPEED CONTROL FEATURES.

In addition to its normal functions, the engine fuel control unit is designed to control engine speed to prevent the occurrence of either high or low corrected engine speed compressor stalls, and to prevent exceeding the maximum allowable compressor discharge pressure.

Note

CIT (compressor inlet temperature) is indicated on the engine air inlet temperature gage located on the instrument panel.

T_2 RESET.

At high CIT, the compressor approaches the low corrected engine speed stall region. The T_2 reset feature of the fuel control attempts to prevent low corrected engine speed compressor stalls from occurring by allowing the indicated engine rpm to increase to a maximum of 104%. Due to T_2 reset (above 85° CIT), the throttle will have no effect on indicated rpm. However, as indicated airspeed is decreased, the CIT will drop below approximately 85°C cutting out the T_2 reset function, thus re-establishing normal throttle control.

RPM CUT-BACK.

The rpm cut-back feature of the fuel control prevents high corrected engine speed compressor stalls by decreasing the maximum allowable indicated engine rpm from 101 to 94% as a function of CIT. Since engine corrected rpm varies inversely with CIT, the high corrected engine speed compressor stall can occur in a situation involving low airspeeds at high rpm with a low CIT. When the CIT decreases enough to allow the corrected engine speed to approach the compressor stall region, the rpm cut-back feature decreases indicated rpm, thus keeping the compressor out of the stall speed area. When in the rpm cut-back region, advancing the throttle will have no effect on indicated rpm until higher CIT is obtained due to increase in indicated airspeed.

CDP LIMITER.

The compressor discharge pressure (CDP) limiter feature of the fuel control prevents exceeding the maximum allowable compressor discharge pressure by decreasing rpm as the maximum limit of pressure is approached. This occurs near the maximum design indicated airspeed of the airplane. The CDP limiter has priority of engine speed control over the T_2 reset. If CDP limiter action is experienced, the pilot should decrease airspeed to avoid entering the low corrected speed stall region.

RECOGNIZING AND CLEARING ENGINE STALLS.

The identification of either high or low corrected speed compressor stalls, or surges, can be made by noting a moderate to fast rise in exhaust gas temperature, accompanied by "bangs" or "chugs" from the engine. As mentioned above, CDP has priority over other control functions in the main fuel control. Since a stall usually involves a sharp drop in CDP, the operating point of the fuel control is changed so that the engine is moved out of the stall condition. Cyclic stalls may occur if the conditions initiating the first stall are unchanged when the engine has recovered. There may be no indicated engine RPM change during a stall. The fact that stalls will occur only during transient aircraft maneuvers or throttle movements (chops and bursts), will not require any pilot action, since the engine will probably clear itself. Normally at altitudes below 30,000 feet, if a stalled condition becomes cyclic, rpm begins to drop, hangs up, or an over-temperature exists, the throttle should be retarded to IDLE. If this action does not eliminate the stall, re-cycle the ignition, retard the throttle to OFF and rapidly advance to IDLE. At higher altitudes if a stall occurs and does not clear itself, normally no pilot action will clear the condition. If no over-temperature exists, a descent should be performed to 30,000 feet. If the engine stall does not clear at this altitude, execute the above clearing procedure.

AFTERBURNER SURGING.

Afterburner surging may be experienced during afterburner initiation, as well as when advancing or retarding through the sector-to-uniform transition range. It can also occur during afterburner flameout and relight, resulting, for instance, from high altitude operation as well as low airspeed intermediate altitude operation. When surging or afterburner blow out occurs it is extremely important to move the throttle out of the afterburner range to MILITARY detent as soon as possible—preferably before more than 1 to 2 cycles of surge have taken place. This action is necessary to prevent stalling the compressor and possibly resulting in engine flameout. There will be no question as to whether or not the afterburner surge is taking place since engine exhaust gas temperature

and nozzle position will oscillate and at the same time the pilot will feel the instability. Afterburner surge can be initiated by a malfunctioning afterburner flow divider and selector valve assembly, by a malfunctioning afterburner pilot burner assembly, or by partial flameholder action induced by high altitude operation.

ENGINE OIL PRESSURE.

When a new engine is first run-up or when a pilot is exposed for the first time to an engine he should take note of the characteristic readings of the various engine operating instruments. This will assist in quickly recognizing a malfunction so that quick remedial action can be taken. Engine oil pressure is one that has several times hinted that something was not right in the engine. The important thing is to notice a difference in the oil pressure reading, for a given rpm, from the value that was considered normal up to this time. If this jump or drop in oil pressure occurs during ground run-up the engine should be shut down immediately and an investigation started.

Because the engine oil pressure indicator measures the differential pressure between the main lube pump and the static pressure outside the aircraft fuselage, oil pressure indications will be affected by altitude and Mach number. This engine may exhibit the following oil pressure indication changes:

a. An increase in oil pressure indication of 2 to 3 psi, during a climb from approximately 20,000 to approximately 28,000 feet.

b. An increase in oil pressure indication of approximately 3 psi can occur at high Mach numbers.

c. An increase of approximately 6 psi can occur at full T2 reset.

d. The maximum pressure increase should not be more than 10 psi. Normally, the maximum pressure rise should be 6 or 7 psi.

ENGINE OIL PRESSURE RECORD CARD.

An engine oil pressure record card is provided on the forward right side of the cockpit. This card lists the normal engine oil pressure at Military Power for each engine-airplane combination. The pilot should check his engine oil pressure indicator at Military Power just prior to brake release and compare this reading with the recorded oil pressure on the card. If the indicated oil pressure varies more than ±10 psi from that listed on the card, the flight should be aborted and an engine inspection performed.

USE OF LANDING WHEEL BRAKES.

The following rules apply in general to all airplanes but become increasingly important at high landing speeds:

1. Avoid unnecessary use of brakes. For example: use all of the runway during landing rolls; don't drag the brakes during taxiing; avoid braking to accelerate turns, utilize nose wheel steering instead.

2. Utilize aerodynamic braking as much as possible.

3. Allow time for brakes to cool; 15 minutes between landings when the landing gear remains extended in the slip stream, or 30 minutes if they are retracted.

4. If maximum braking is necessary, apply the brakes smoothly and with increasing pressure as the speed drops, being careful not to lock the wheels. Maximum braking efficiency occurs when there is approximately 15 to 20 percent slippage on the braking surface, i.e. when the wheels are turning 80 to 85 percent of the speed at which they would turn if they were rolling free.

Note

If one wheel is locked the aircraft will veer noticeably in the opposite direction. Increased pressure on that brake will have no effect. Release *both* brakes, then reapply more smoothly.

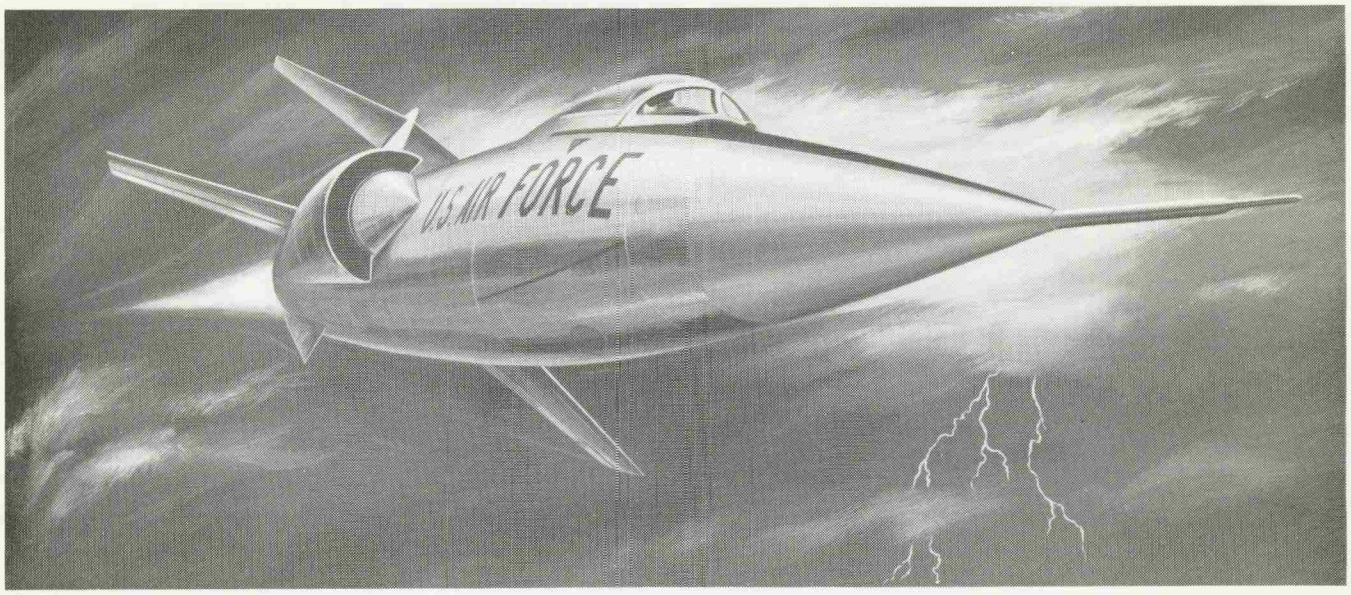
CAUTION

- After the brakes have been used excessively for an emergency stop and are in a heated condition, the aircraft should not be taxied into a crowded parking area. Peak temperature occurs in the wheel and brake assembly from 5 to 15 minutes after a maximum braking operation. To prevent brake fire and possible wheel assembly explosion, the specified procedures for cooling brakes should be followed.
- A wheel once locked before the full weight of the aircraft is on the gear immediately after touchdown, will not become unlocked as the load is increased as long as brake pressure is maintained. Proper braking action cannot be expected until the tires are carrying heavy loads.

Section VIII

CREW DUTIES

Not Applicable To This Airplane



F50-8-9-79
AM2830

all weather operation SECTION IX

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Note

The procedures outlined in this section are preliminary all weather procedures. Changes, additions and deletions will be accomplished as these procedures are improved and expanded. Except for some repetition necessary for emphasis, clarity, or continuity of thought, this section contains only those procedures that differ or are in addition to the normal operating instructions covered in Section II.

INSTRUMENT FLIGHT PROCEDURES

These procedures and techniques pertain primarily to instrument flight conditions and are in addition to normal procedures. Because navigation facilities and terrain features vary at each base, this information is intended to serve as a guide to commanders in establishing instrument flight procedures. This airplane is capable of supersonic speeds during instrument flight conditions, thus demanding a high degree of instrument proficiency. Because of the distance covered in a short period of time and the wide variety of weather conditions which may be

encountered on a single mission, conscientious preflight planning is of paramount importance. Fuel requirements for completion of instrument letdown, approach procedures and possible diversion to alternate bases are much greater than for VFR flights and must have special emphasis in preflight planning. The airplane has immediate and positive control response during any phase of instrument flight. Only essential navigation equipment is installed since ground controlled guidance replaces some of the more conventional navigation aids. The UHF radio is adequate for air-to-air and air-to-ground communications. With existing equipment, three types of instrument approaches may be accomplished: GCA, VOR and ILS.

INSTRUMENT TAKE-OFF.

Instrument take-offs may be made at either Military or Maximum power. Military power take-offs are recommended since the pitch attitudes required in the climb will be lower and turns will be easier to execute than when using Maximum power. Afterburner take-off is recommended to shorten the take-off roll under conditions where the aircraft is heavily loaded.

Note

Be prepared to use a steep pitch attitude after take-off to hold recommended climb schedule. Pitch for afterburner climb may vary from 30° to 45°.

An instrument take-off is essentially the same as a normal VFR take-off. After aligning aircraft with runway center line, set attitude indicator 5° nose down. Pre-set desired track on ID-249 indicator; IFF, to desired position; pitot heat and cockpit lights on, engine anti-icing as required. Check directional indicator indicating runway heading.

CAUTION

To insure nose wheel steering engagement for take-off, keep the nose wheel steering button depressed during engine preflight check.

Bring power to Military while holding brakes then release brakes. Advance throttle to afterburner if required. Maintain heading with nose wheel steering until rudder becomes effective at approximately 70 knots. At the nose wheel lift off speed for aircraft weight and configuration apply back pressure and rotate the airplane to 0° on attitude indicator. Allow the aircraft to fly off in this attitude. (Refer to Appendix for fly off speed.)

Note

During take-off roll, rate of climb indicator will fluctuate slowly between 300 to 700 feet per minute rate of descent.

INSTRUMENT CLIMB.

Raise landing gear when a definite rate of climb is noted and increase pitch to 12° on the attitude indicator. Retract flaps at 240 KIAS. (Expect a mild nose up pitch change when the flaps are retracted.) Increase pitch angle as required when climb schedule is intercepted. (Approximately 33° will be required when using afterburner.)

Note

Be prepared to climb the aircraft at pitch angles steeper than normal in case the landing gear fails to retract fully. This is necessary to maintain airspeed below the transient landing gear structural limit speed.

CAUTION

Exercise caution when attempting turns during afterburner climb. The high pitch attitude diminishes the pilots directional reference and makes precise turns difficult. When a turn is initiated in a steep pitch attitude by banking the aircraft and applying back force to the stick, the nose of the aircraft will inscribe an arc which will not be parallel to the horizon but will be one of increasing pitch attitude. Care must be exercised to avoid a maneuver resembling a chandelle or the top of an off center Immelmann. Turns should be practiced under VFR condition from afterburner straight climbs to familiarize pilots with the reaction of the heading indicator and the airplane attitude before attempting them under actual instrument conditions.

INSTRUMENT CRUISING FLIGHT.

Handling characteristics are such that supersonic instrument cruise flight is possible. Refer to Section VI for level flight characteristics at high speeds. For ease and precision of flight, limit all turns to 30° bank angles.

Note

If your aircraft does not have a shim installed between the turn and slip indicator and the instrument panel to hold the indicator in a vertical plane, expect a momentary indication of a turn opposite to the direction of bank when originating a turn.

LOITERING AND HOLDING.

Based upon minimum fuel consumption, consistent with ease of handling, holding is recommended as follows:

ALT	IAS	% RPM	FUEL FLOW	MAX BANK ANGLES
35,000	305	92±2	2350-2550	30°
20,000	310	86±2	2500-2650	30°

JET PENETRATION.

Several engine and airframe considerations must be made when performing descents. Icing of the inlet guide vanes is most probable at 82% rpm or below, due to inadequate airflow for anti-icing. Therefore, maintain a minimum of 85% rpm when descending in possible icing conditions. The minimum EGT is 170°C. Do not retard the throttle beyond a point that will allow the EGT to fall below 170°C. Airframe buffet and stick shaker actuation will occur at low speeds with the flaps up, therefore, it is recommended that the TAKE-OFF flap position be selected when descending at speeds below 285 KIAS.

WARNING

Steep descents, high indicated Mach numbers, and high angles of bank should be avoided in order to maintain positive control of the airplane at all times. Limit bank angles to 30° at all altitudes and rates of descent.

In consideration of the items listed above there are two instrument descents recommended:

1. Use the following procedures for descents through possible icing:

a. When approaching the radio fix at initial penetration altitude, (slow speed by adjusting power to 85% rpm) turn pitot heat engine anti-ice and canopy defrost on. Over the fix extend the speed brakes, lower wing flaps to TAKE-OFF setting when below 295 KIAS and adjust pitch attitude to hold 275 KIAS (10° to 14° nose down). Retract the speed brakes 1,000 feet above the desired level-off altitude. The average rate of descent will be 4200 feet per minute. With take-off flaps and power at 85% rpm, the aircraft will stabilize at approximately 260 KIAS in level flight inbound to station. Engine airflow should be adequate to provide anti-icing in light to moderate icing conditions. This procedure will require 7 to 8 minutes from 20,000 to return to the radio fix. Distance out from station is about 19 miles and fuel required is approximately 250 pounds. See figure 9-1.

2. Use the following procedure for all other IFR descents:

a. Approaching the radio fix at initial penetration altitude slow speed by adjusting power to 80% rpm or 170°C EGT whichever is higher, pitot heat on, canopy defrost as desired. Over the fix extend the speed brakes and adjust pitch angle to hold 300 KIAS throughout the descent. Pitch will be approximately 18° nose down at the start of descent decreasing in the turn to 4-6° nose down. Retract the speed brakes when starting penetration turn around or at approximately 2000 feet above the desired level off altitude in straight descents. At level off altitude advance power to 85% rpm to maintain 300 KIAS inbound. In this descent at constant IAS the rate of descent will average 8000 feet per minute. This descent will require approximately 5 minutes from 20,000 feet to return to radio fix. The average distance from the station will be 12 miles and 140 pounds of fuel will be consumed. See figure 9-2.

WARNING

On unmodified aircraft, the landing gear audio warning signal will be heard by the pilot whenever the throttle is retarded below approximately 83% rpm and the landing gear is not down and locked.

The high power requirements throughout the penetration and landing pattern make this warning system unsuitable. Be sure to check the landing gear position indicators (3 green lights) and hydraulic pressure prior to landing to insure proper landing gear operation.

RADAR RECOVERY.

Radar recovery procedure to be used following return to base under IFR conditions should be practiced in VFR to develop and improve the team work of the pilot and radar controller. When GCA or ILS approach is required, the descent from the inbound cruising altitude should be initiated at a sufficient distance to permit a straight-in descent at the jet penetration airspeed allowing 5 to 7 miles for deceleration and changing to approach configuration before reaching the turn on point (GATE) to final approach. See figure 9-3, for recommended preliminary procedures.

INSTRUMENT APPROACHES.

The airplane has good handling characteristics during instrument approaches. At low rpm and low indicated airspeeds, response to throttle movement and acceleration are rapid. Therefore, use small power changes in the approach configuration and maintain the recommended approach airspeed. In rain or snow, use the windshield rain removal system to increase forward visibility out of the curved left windshield panel.

VOR APPROACH.

The fuel required to complete the VOR approach is largely determined by the time the aircraft flies outbound before making the procedure turn. This time needs only to be sufficient to permit completion of the cockpit check after the procedure turn and precision tracking to the station at final approach altitude. Procedures are outlined in figure 9-4.

GCA APPROACH.

The GCA approach requires a minimum of power changes and is adaptable to formation as well as single aircraft. This pattern is basic for all types of instrument approaches in this aircraft, therefore, pilots should be very familiar with it. Approach the GCA pick up point in a clean configuration reduce power to 85% rpm and place flaps to TAKE-OFF position. Airspeed will stabilize at approximately 260 KIAS. Limit all bank angles to 30°. Lower the landing gear on the base leg and with 85% rpm airspeed will bleed off to approximately 225 KIAS.

Note

For descents in the pattern use either partial speed brakes or reduce power to 83%.

Turning on to final approach, airspeed will bleed off to 190 KIAS. Place flaps to LAND position at the ten seconds warning prior to entering the glide slope and increase power to 89% rpm. Let airspeed decrease to 170 KIAS and maintain the airspeed on glide slope. Rate of descent will be 900 feet per minute. Use speed brakes on final to adjust rate of descent.

Note

Formation position can be maintained by the wing-man using either partial speed brakes or throttle (±2% rpm) until landing flaps are lowered. After that use speed brakes only.

CAUTION

For take-off flap landing, reduce power to 83% as glide slope is intercepted. Airspeed will be 190 KIAS and rate of descent approximately 950 feet per minute.

WARNING

In heavy precipitation it is difficult for the GCA operators to keep the aircraft visible in the precipitation clutter on the precision scope. This may result in initiating a missed approach.

See figure 9-5 for a typical GCA with missed approach.

ILS APPROACH.

During an ILS approach, maintain aircraft attitude with the basic flight instruments and monitor the ILS indicator for reference to the localizer and glide slope. If localizer interception is from a radio fix, or GCI control, and is at an angle of 90° or less, allow sufficient distance out to perform final cockpit check, slow to approach speed, and establish correct localizer beam heading. On a typical ILS pattern, fly outbound on a reciprocal heading only long enough to complete the above function after the procedure turn inbound. ILS approach procedures are outlined in figure 9-6.

MISSED APPROACHES.

In case of a missed approach, follow the procedure given in Section II for Go-Around. Adjust power to 85% rpm if you are to remain in the pattern; this will yield approximately 260 KIAS. If you intend to exceed 295 KIAS retract the wing flaps fully but not below 240 KIAS.

ICE AND RAIN

Although this airplane does not have anti-icing systems for the wing, empennage and inlet ducts, flight under icing conditions can be made. Defrosting, rain removal, engine guide vane anti-icing, pitch sensor and pitot heat should be turned on prior to entering an area where icing conditions prevail or are suspected. Preliminary results of the icing flight test under simulated icing conditions have indicated that, although engine flame-out did not occur, slight damage to the guide vanes and stator blades, and compressor stalls did occur when heavy ice was ingested at 98% rpm. The compressor stalls resulted in a reduction of power to idle rpm and prevented obtaining higher rpm. Heavy ice was ingested at an engine rpm of 88% and below, with no difficulty and no engine damage. With pitot heat on, ice formed over the static port of the pitot head during heavy icing, resulting in a loss of the airspeed indicator and altimeter. The pitot heat was sufficient to clear this ice off the pitot head within one to two minutes after leaving the icing area. After several flights in heavy rain, sufficient moisture collected in the generator compartment to cause one of the generators to fail.

Note

If reported rain intensity is heavy rain or less, good vision will be obtained through the cleared area. The cleared area will be slightly smaller in heavy rain than in moderate rain. If very heavy rain is reported, some visibility will be retained but it will be substantially impaired.

After several flights in which the windshield rain remover was used, the left windshield cracked. A discoloration of the windshield adjacent to the rain remover was noted previous to the failure. In view of the above, the following instructions should be observed.

1. Whenever possible, avoid flight in conditions conducive to rapid build-up of ice.

2. Have generator compartment dried by hot air after any flight in rain.

3. Inspect windshield for discoloration after any flight in which the rain remover is used, and have windshield replaced if discoloration is noted.

4. After flying in moderate to heavy icing for two minutes or more do not use more than 88% engine rpm and land as soon as practical.

WARNING

Heavy ice accumulations can increase stalling speeds. Therefore, caution must be used to maintain sufficient speed during approach or low altitude flight when icing is present.

When instrument take-offs or approaches are to be made and rain is anticipated, the windshield rain removal system should be in operation to increase forward visibility through the left windshield panel. Flight through rain normally will present no unusual problems. Successive flights through rain at high speed may cause erosion of the radome. Refer to Sections IV and V for procedures and operating limitations of the anti-icing and rain removal equipment.

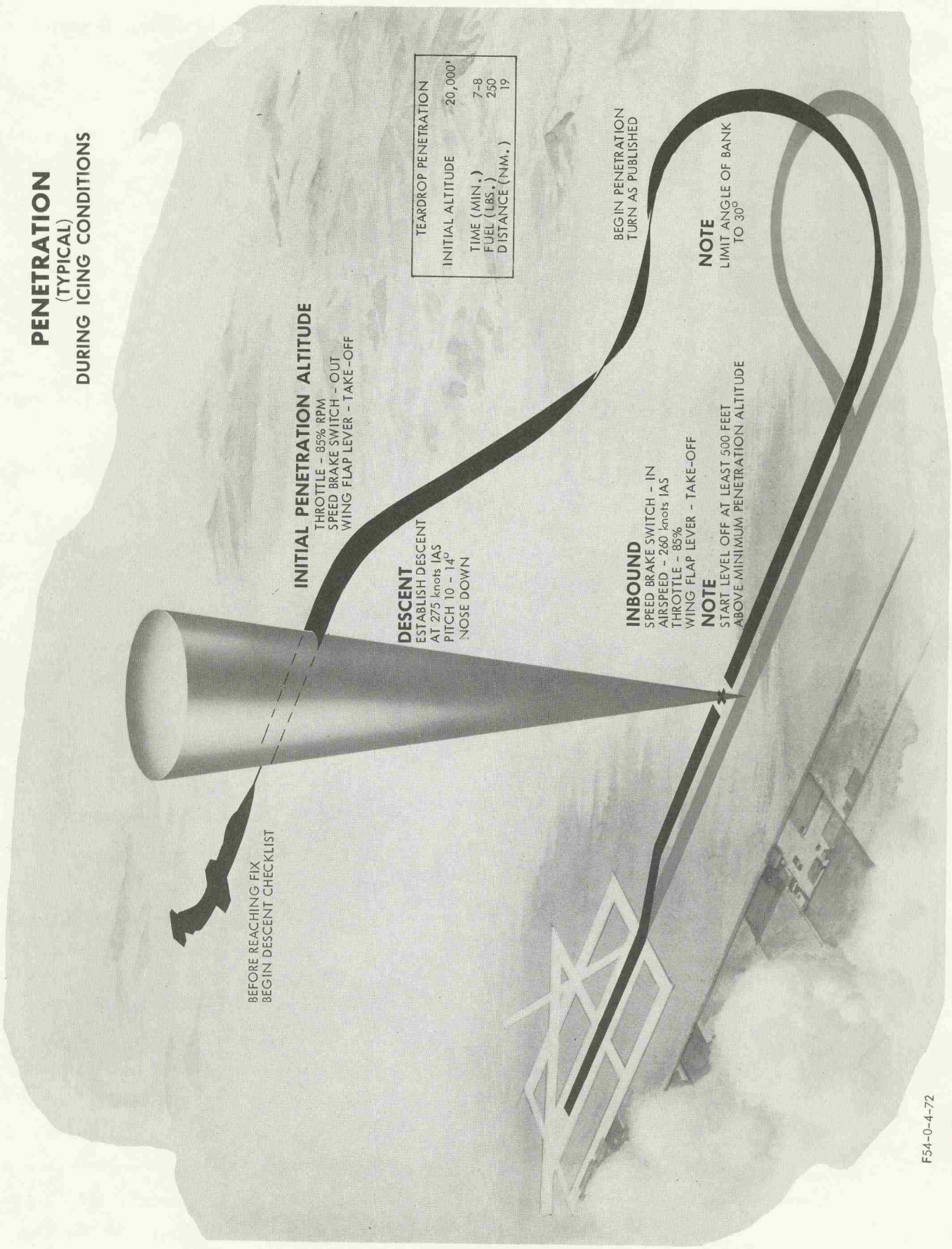

PENETRATION
(TYPICAL)
DURING ICING CONDITIONS
BEFORE REACHING FIX
BEGIN DESCENT CHECKLIST
INITIAL PENETRATION ALTITUDE
THROTTLE - 85% RPM
SPEED BRAKE SWITCH - OUT
WING FLAP LEVER - TAKE-OFF
DESCENT
ESTABLISH DESCENT
AT 275 knots IAS
PITCH 10 - 14°
NOSE DOWN
TEARDROP PENETRATION
INITIAL ALTITUDE 20,000'
TIME (MIN.) 7-8
FUEL (LBS.) 250
DISTANCE (NM.) 19
INBOUND
SPEED BRAKE SWITCH - IN
AIRSPEED - 260 knots IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
NOTE
START LEVEL OFF AT LEAST 500 FEET
ABOVE MINIMUM PENETRATION ALTITUDE
BEGIN PENETRATION
TURN AS PUBLISHED
NOTE
LIMIT ANGLE OF BANK
TO 30°
F54-0-4-72

Figure 9-1

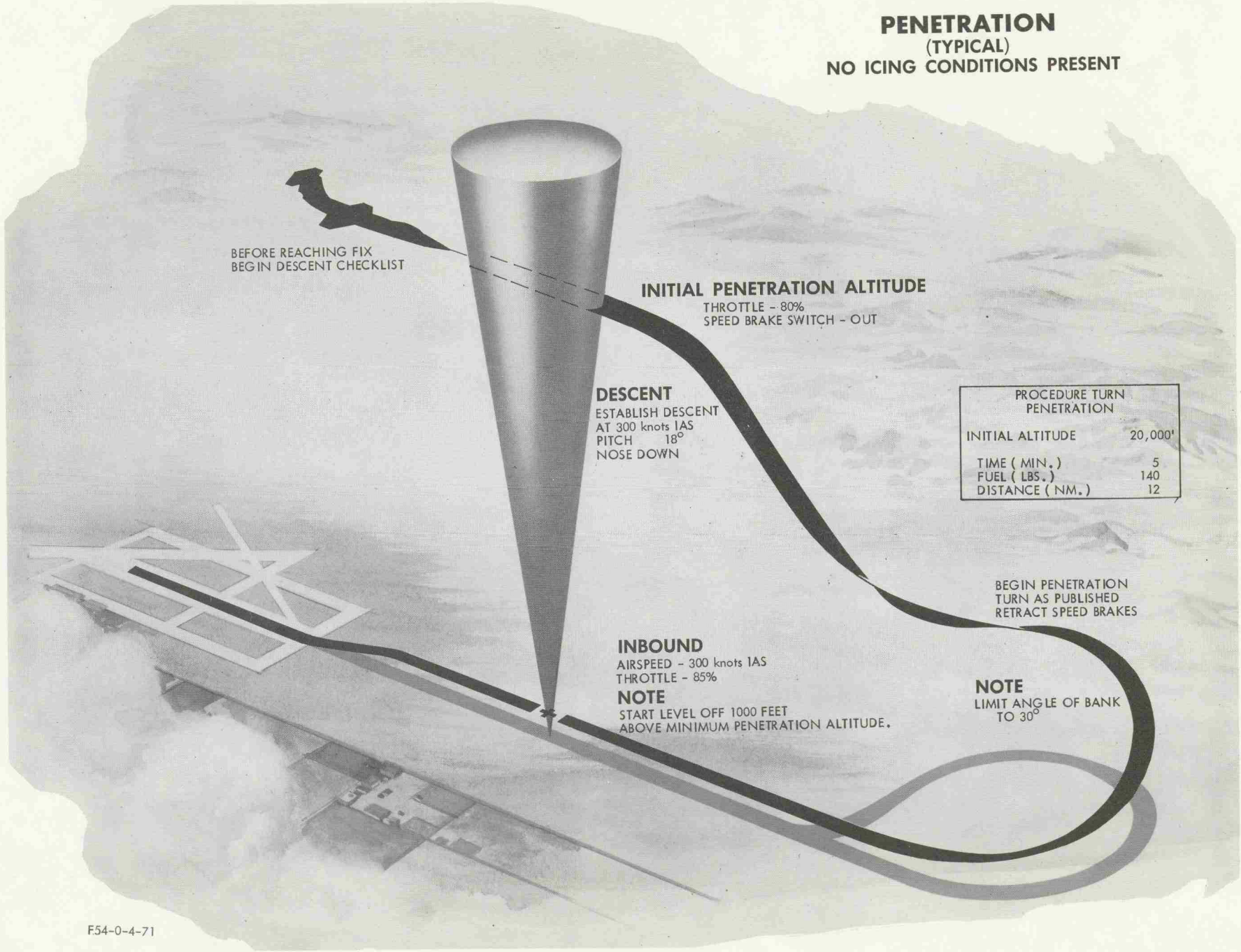

PENETRATION
(TYPICAL)
NO ICING CONDITIONS PRESENT
BEFORE REACHING FIX
BEGIN DESCENT CHECKLIST
INITIAL PENETRATION ALTITUDE
THROTTLE - 80%
SPEED BRAKE SWITCH - OUT
DESCENT
ESTABLISH DESCENT
AT 300 knots IAS
PITCH 18°
NOSE DOWN
PROCEDURE TURN
PENETRATION
INITIAL ALTITUDE 20,000'
TIME (MIN.) 5
FUEL (LBS.) 140
DISTANCE (NM.) 12
BEGIN PENETRATION
TURN AS PUBLISHED
RETRACT SPEED BRAKES
INBOUND
AIRSPEED - 300 knots IAS
THROTTLE - 85%
NOTE
START LEVEL OFF 1000 FEET
ABOVE MINIMUM PENETRATION ALTITUDE.
NOTE
LIMIT ANGLE OF BANK
TO 30°
F54-0-4-71

Figure 9-2

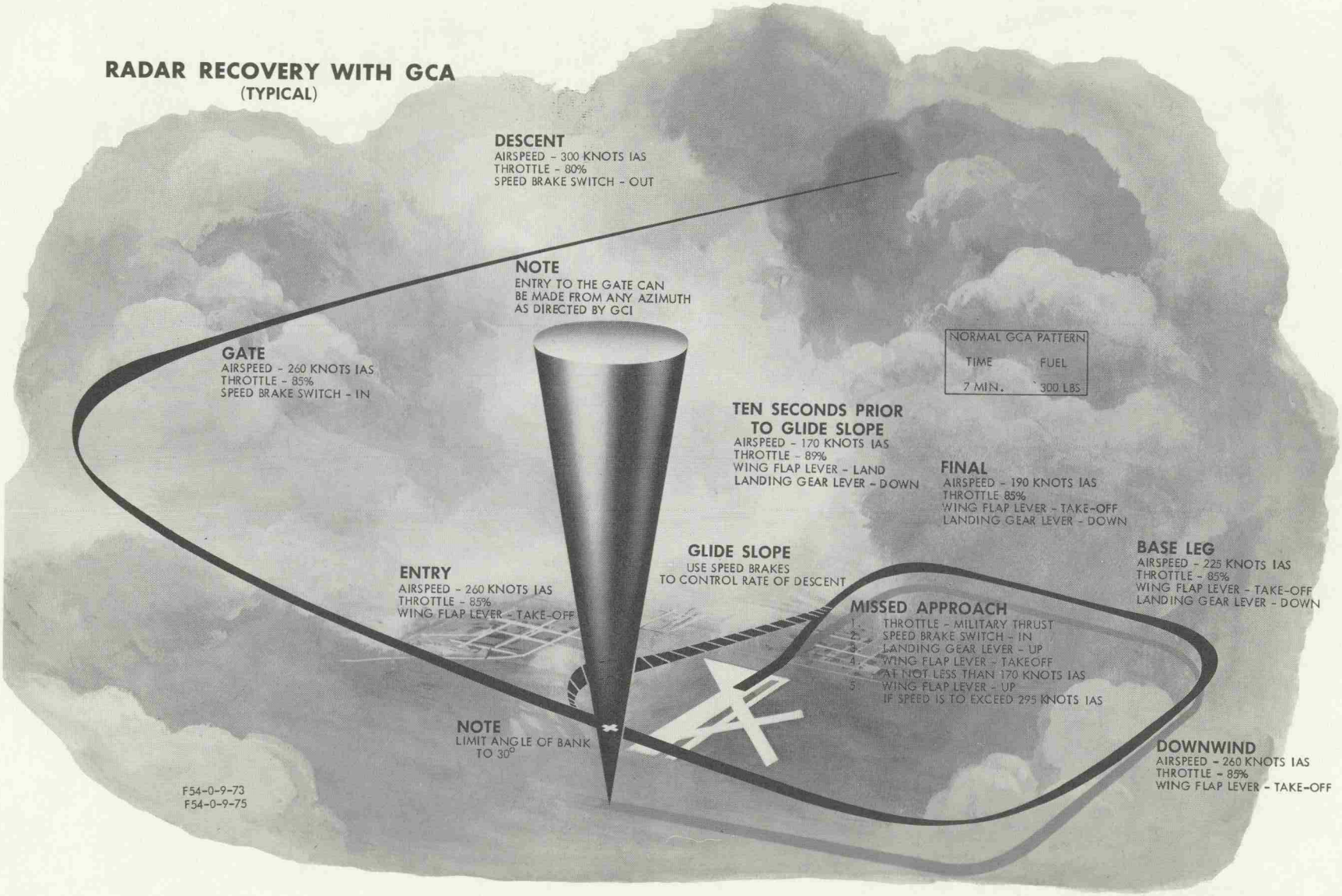

RADAR RECOVERY WITH GCA
(TYPICAL)
DESCENT
AIRSPEED - 300 KNOTS IAS
THROTTLE - 80%
SPEED BRAKE SWITCH - OUT
NOTE
ENTRY TO THE GATE CAN
BE MADE FROM ANY AZIMUTH
AS DIRECTED BY GCI
GATE
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
SPEED BRAKE SWITCH - IN
NORMAL GCA PATTERN
TIME
FUEL
7 MIN.
300 LBS
TEN SECONDS PRIOR
TO GLIDE SLOPE
AIRSPEED - 170 KNOTS IAS
THROTTLE - 89%
WING FLAP LEVER - LAND
LANDING GEAR LEVER - DOWN
FINAL
AIRSPEED - 190 KNOTS IAS
THROTTLE 85%
WING FLAP LEVER - TAKE-OFF
LANDING GEAR LEVER - DOWN
GLIDE SLOPE
USE SPEED BRAKES
TO CONTROL RATE OF DESCENT
BASE LEG
AIRSPEED - 225 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
LANDING GEAR LEVER - DOWN
ENTRY
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
MISSED APPROACH
1. THROTTLE - MILITARY THRUST
2. SPEED BRAKE SWITCH - IN
3. LANDING GEAR LEVER - UP
4. WING FLAP LEVER - TAKEOFF
AT NOT LESS THAN 170 KNOTS IAS
5. WING FLAP LEVER - UP
IF SPEED IS TO EXCEED 295 KNOTS IAS
NOTE
LIMIT ANGLE OF BANK
TO 30°
DOWNWIND
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
F54-0-9-73
F54-0-9-75

Figure 9-3

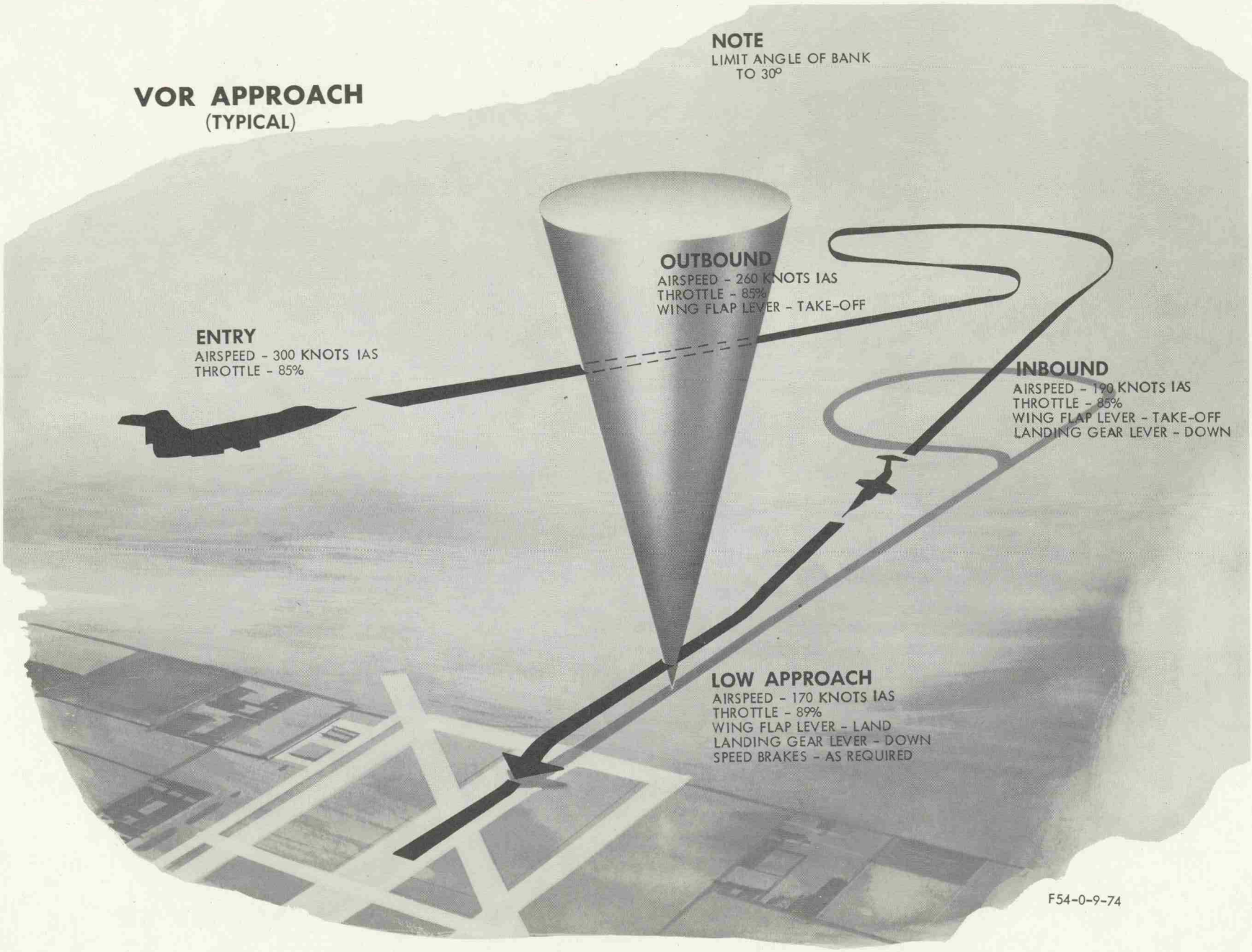

NOTE
LIMIT ANGLE OF BANK
TO 30°
VOR APPROACH
(TYPICAL)
OUTBOUND
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
ENTRY
AIRSPEED - 300 KNOTS IAS
THROTTLE - 85%
INBOUND
AIRSPEED - 190 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
LANDING GEAR LEVER - DOWN
LOW APPROACH
AIRSPEED - 170 KNOTS IAS
THROTTLE - 89%
WING FLAP LEVER - LAND
LANDING GEAR LEVER - DOWN
SPEED BRAKES - AS REQUIRED
F54-0-9-74

Figure 9-4

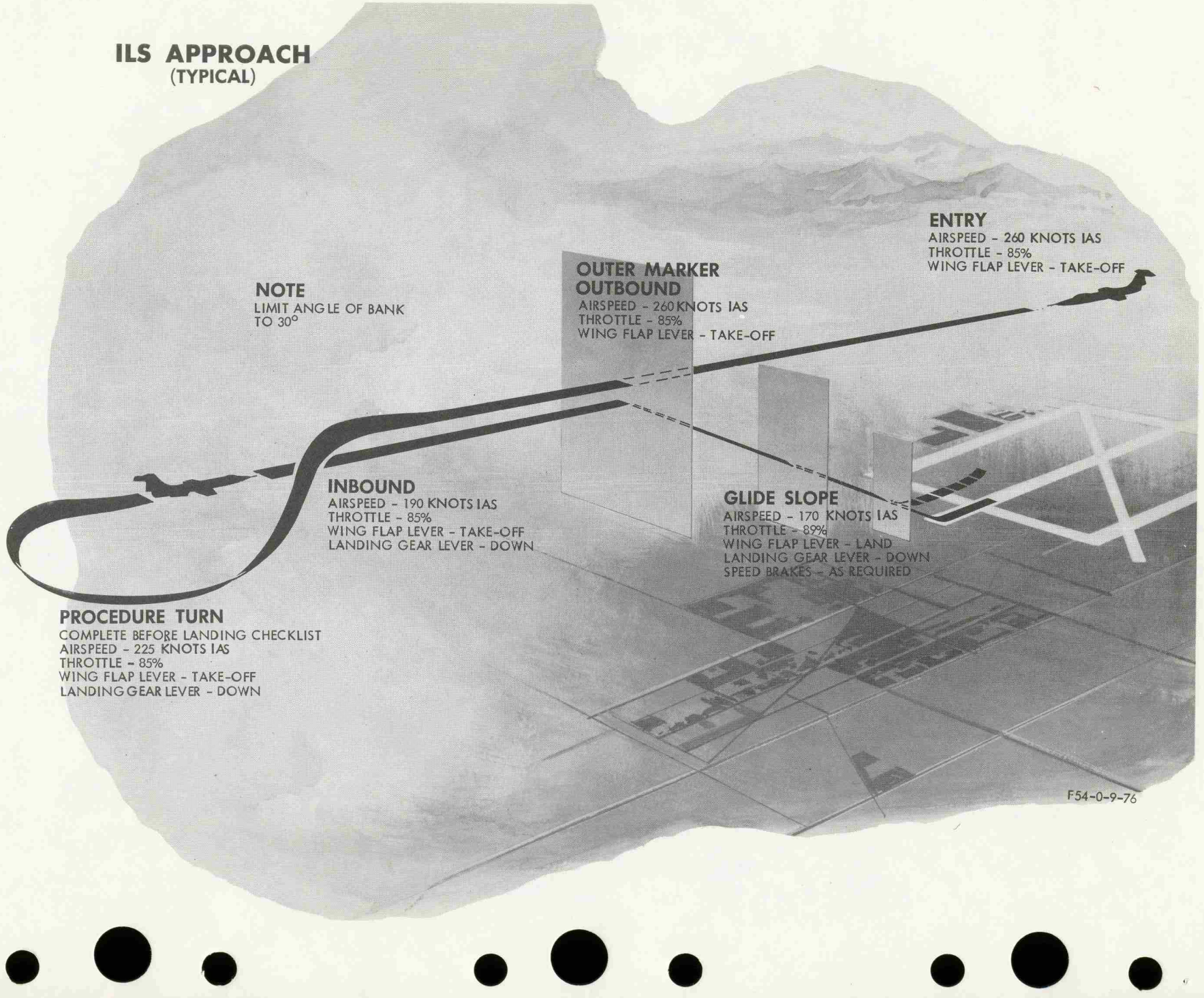

ILS APPROACH
(TYPICAL)
NOTE
LIMIT ANGLE OF BANK
TO 30°
ENTRY
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
OUTER MARKER
OUTBOUND
AIRSPEED - 260 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
INBOUND
AIRSPEED - 190 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
LANDING GEAR LEVER - DOWN
GLIDE SLOPE
AIRSPEED - 170 KNOTS IAS
THROTTLE - 89%
WING FLAP LEVER - LAND
LANDING GEAR LEVER - DOWN
SPEED BRAKES - AS REQUIRED
PROCEDURE TURN
COMPLETE BEFORE LANDING CHECKLIST
AIRSPEED - 225 KNOTS IAS
THROTTLE - 85%
WING FLAP LEVER - TAKE-OFF
LANDING GEAR LEVER - DOWN
F54-0-9-76

Figure 9-5

TURBULENCE AND THUNDERSTORMS

CAUTION

Flight through a thunderstorm should be avoided if at all possible. Hail encountered could cause rapid deterioration of the radome and unpredictable airplane damage. Hail may shatter the infrared sight cover causing cockpit decompression.

If mission requirements or emergencies necessitate thunderstorm penetration use these procedures. Prior to entering the storm establish airspeed of 350 KIAS for a clean aircraft or 275 KIAS when take-off flaps are extended. Turn pitot heat on and adjust thunderstorm and instrument light to minimize blinding effects of lightning. Engine anti-icing ON. Safety belt and shoulder harness should be locked.

CAUTION

The warning panel lights are automatically dimmed when the instrument lights are turned on. Special care should be exercised to detect any warning light illumination.

NIGHT FLYING

Limited flight test experience during night flights indicate that this airplane does not present any special problems on techniques except during landing. During final approach at night, the runway lights are reflected in the windshield panel and disorientation can easily occur. During final approach at night, utilize all possible clues to attitude and flight path ignoring as much as possible the runway light reflection in the windshield.

COLD WEATHER PROCEDURES

The success of low temperature operation depends primarily upon the preparations made during the post-flight inspection, in anticipation of the requirements for operation on the following day. The procedures outlined should be followed during outdoor operation to expedite the pre-flight inspection and to insure satisfactory operation of the aircraft and its systems during the next flight.

BEFORE ENTERING THE AIRCRAFT.

1. Have all protective covers and duct plugs removed.
2. Perform exterior inspection as outlined in Section II.
3. Check the entire aircraft for freedom from snow and ice.

WARNING

- Care should be exercised to insure that water from melted ice is sponged so that it will not drain to some critical area and refreeze.
- Do not attempt to chip or scrape ice from the aircraft surface, as this action may cause damage to the aircraft.
- Dangerous loss of lift and treacherous stalls may result if the aircraft is not adequately cleaned of all snow and ice.

4. Place heater duct in cockpit for 5 minutes when the temperature is below −30°F.

5. Have liquid springs and actuating cylinders checked free from dirt and ice; check shock struts for proper inflation.

6. Have battery voltage checked for 24-26 volt d.c.

WARNING

- Anytime the aircraft is cold soaked overnight at an average temperature of 0°F or below the battery voltage must be checked.
- If battery voltage is low, have battery preheated for 30-45 minutes and simultaneously recharge battery for 45 minutes using external power.

Note

Be sure battery circuit breaker is pushed in.

7. The pressure of both hydraulic accumulators will drop with temperature. A pressure as low as 700 psi can be expected at −65°F.

8. Inspect plastic airspeed and altitude pitot lines for cracking at temperatures of −30°F and below.

CAUTION

Before entering aircraft have crew chief steady cockpit ladder and use caution since the ladder is unsteady and the metal steps become slippery when snow and ice are present.

ON ENTERING THE AIRCRAFT.

1. Make a normal cockpit check for switch and circuit breaker positions as outlined in Section II.

2. Keep the oxygen mask well clear of the face in temperatures below 0°F, to prevent freezing of the inhalation valves.

BEFORE STARTING ENGINE.

1. Make normal checks as outlined in Section II.

2. Preheating of the fuel control will be required to warm the fuel and provide proper atomization spray necessary for initial burning. The following minimum preheat times are recommended:

Fuel temperature 0°F—preheat 10-15 minutes

Fuel temperature −30°F—preheat 30 minutes

Fuel temperature −65°F—preheat 1 hour

STARTING ENGINE.

Make normal start as outlined in Section II.

Note

- Consistent successful starts cannot presently be expected at fuel temperatures of 10°F and below.
- During low ambient temperature engine starts, excessive engine oil pressure for short periods of time is a normal condition and will not cause engine damage. Allow oil pressure to decrease within limits before taxiing.

CAUTION

Snow chocks are not effective on packed glazed snow surface. The aircraft will slide forward at approximately 84-88% rpm with the brakes locked. Make certain the aircraft is clear before advancing the throttle.

WARM-UP AND GROUND CHECK.

1. Close and lock canopy.
2. Close ram air scoop.
3. Set cockpit temperature control to comfortable level. After cockpit has warmed put on oxygen mask.
4. Have crew chief inspect for hydraulic leaks while actuating controls and speed brakes.
5. Crack defroster open.

Note

It will require 2-4 minutes to obtain warm air from the defroster. If immediate defrosting or de-icing is required, turn rain removal ON until the left windscreen and canopy have cleared and then turn rain removal OFF.

6. Check hydraulic pressure, oil pressure, and engine instruments.

7. Check flight instruments.

WARNING

Make sure all instruments have warmed up sufficiently to insure normal operation. Check for sluggish instrument indication during taxiing.

TAXIING INSTRUCTIONS.

1. Taxi at slow speed when taxiing over rough snow packed surfaces.

2. Allow more room than on a cleared surface to bring the aircraft to a stop.

3. On hard packed surfaces a small amount of braking will be required at idle rpm. In loose snow, power will be required to keep the aircraft moving.

4. Successful taxiing can be accomplished in snow up to 6 inches deep.

Note

Only nose wheel steering should be used in deep snow. Braking will cause the snow to melt and moisture to form on the wheels which may later freeze.

BEFORE TAKE-OFF.

Make normal before take-off check as outlined in Section II.

Note

Canopy defrosting air should be operated at the 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.

TAKE-OFF.

1. Line the aircraft up with the runway at idle rpm.

2. Hold brakes and engage nose wheel steering.

3. Slowly advance power until wheels begin to slip (84-88% rpm).

4. Release brakes and check nose wheel steering while advancing power to military.

5. After it has been determined that nose wheel steering is engaged and aircraft is rolling straight down the runway advance the throttle into afterburning.

6. Release the nose wheel steering at between 80-100 KIAS and maintain directional control throughout the remainder of the ground roll by use of the rudder.

CAUTION

Acceleration is very rapid in cold weather. The pilot must make certain that the aircraft is rolling straight down the runway before applying afterburner.

AFTER TAKE-OFF—CLIMB.

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.

WARNING

Until improved engine starting is available, there is no air start capability when the fuel temperature is below 0°F.

LANDING.

Normal landings can be made on wet or icy runways. Landing rolls will differ with each condition encountered. Refer to Landing On Slippery Runways in Section II.

STOPPING THE ENGINE.

The engine is shutdown in the normal manner.

BEFORE LEAVING THE AIRCRAFT.

1. Release brakes after wheels are chocked.
2. Have shock struts and actuating cylinders inspected and wiped clean. It is advisable to keep liquid springs and actuating cylinders exceptionally clean as any scarring of the seals will result in excessive leakage.
3. Have engine oil and hydraulic reservoirs serviced immediately after engine shut down.

Note

Hydraulic reservoirs should be serviced to the high side of FULL to compensate for contraction due to cold soaking. This procedure will preclude the possibility of over servicing during preflight the next day.

4. Whenever possible, leave the aircraft parked with full fuel tanks. Every effort should be made, during servicing, to prevent moisture from entering the fuel system.
5. Have protective covers installed.

HOT WEATHER AND DESERT OPERATIONS

Hot weather and desert procedures differ from normal procedures mainly in that additional precaution must be taken to protect the airplane from damage due to high temperature and dust. Particular care should be taken to prevent the entrance of sand into the various airplane components and systems (engine, fuel system, pitot-static system, etc.). All filters should be checked more frequently than under normal conditions. Units incorporating plastic and rubber parts should be protected as much as possible from wind blown sand and excessive temperatures. Canopy and electronic compartment should be protected from excessive temperatures, due to the sun's radiation, by use of a canopy-electronic compartment sun shade. Tires should be checked frequently for signs of blistering, etc.

BEFORE ENTERING AIRCRAFT.

Note

All metal surfaces exposed to the sun will be extremely hot to the touch, and gloves should be worn during preflight inspections.

1. Inspect tires for evidence of blisters and other deterioration or improper inflation. Inspect wheel brake for evidence of leaking brake seals.
2. Check exposed portions of the liquid spring shock strut pistons for dust and sand, and have them cleaned if necessary.
3. Check engine air intake ducts for accumulations of dust or sand.

4. Make sure all filters have been cleaned and that the airplane has been thoroughly checked for fuel or hydraulic leaks caused by swelling of packing or expanding of fittings.

5. Inspect area behind the airplane to make sure sand or dust will not be blown onto personnel or equipment during starting operations.

ON ENTERING THE AIRCRAFT.

Note

All items in the cockpit that have been exposed to the sun are extremely hot to the touch unless the cockpit has been protected by a canopy sun shade or precooled using a portable air conconditioner.

1. Check the cockpit for excessive accumulation of dust or sand.

2. Complete as much of normal preflight check as possible before starting engine to avoid prolonged ground running.

STARTING ENGINE.

1. Normal starting procedures are used in hot weather. Temperatures will probably be on the high side of the operating ranges.

2. As soon as the engine is started, the temperature control should be placed in the full automatic cold position and the cockpit fresh air scoop closed to provide cooling air to the electronic compartment and cockpit.

TAXIING.

1. Brakes should be used as little as possible to prevent their overheating.

2. The airplane should be taxied with the canopy in the part open position for maximum cooling.

Note

For better cooling while taxiing the canopy full open position can be used but caution must be exercised to prevent damage which might be incurred when gusty or strong crosswinds are present.

BEFORE TAKE-OFF.

Note

Do not attempt take-off in a sandstorm or dust storm. Park airplane cross-wind, shut down the engine, and have crew chief install protective covers.

1. Select an Automatic Cold setting on the air conditioner. Insure that the defroster is OFF to prevent excessive head level temperature.

2. Make sure the runway selected for take-off is of sufficient length as excessively high outside air temperature increases take-off distance.

TAKE-OFF.

CAUTION

It is imperative that take-off be made at recommended speeds. When outside air temperature is high, do not lift from runway too soon. More than usual take-off distance will be required to obtain take-off speed.

DESCENT.

Canopy defrosting air should be operated at the 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.

APPROACH AND LANDING.

1. Maintain the recommended indicated airspeeds for approach and touchdown.

Note

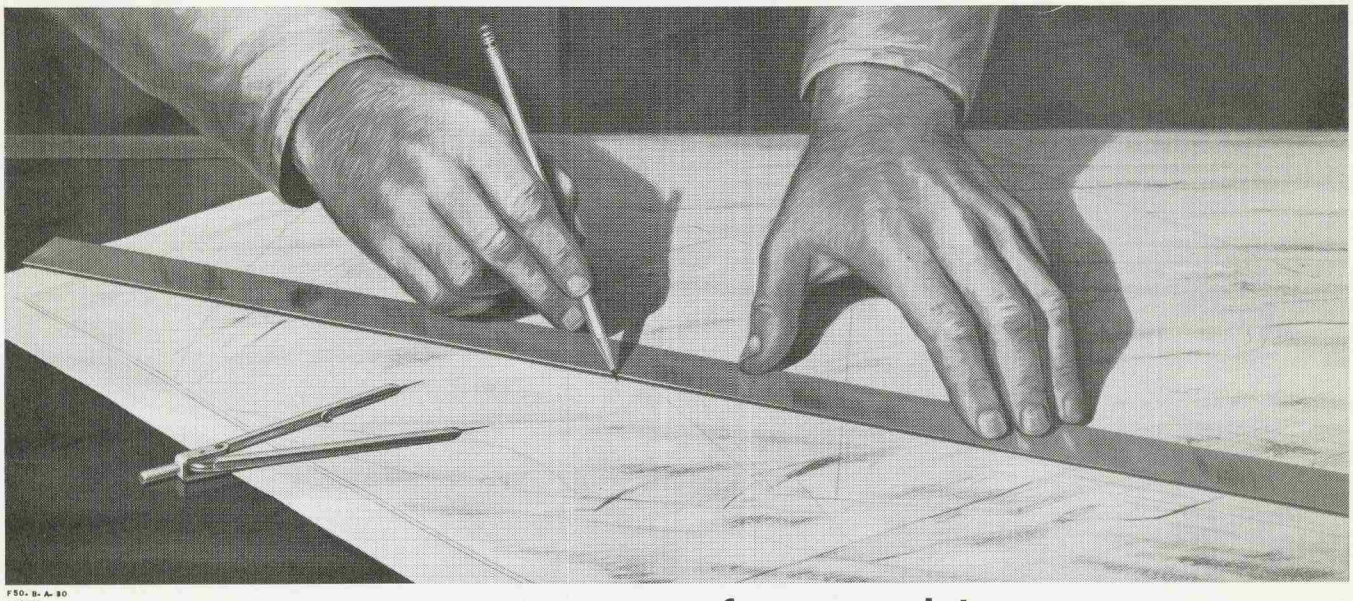
Because of high outside air temperature the true airspeed will be higher than normal and a longer landing roll will result.

2. Avoid heavy braking during landing roll. Small increments of braking with the drag chute deployed will stop the aircraft in a reasonably short distance without excessive tire wear. Heavy braking may cause tire failure.

BEFORE LEAVING THE AIRCRAFT.

1. Leave the canopy open to permit air circulation in the cockpit unless blowing sand or dust is expected.

2. Have crew chief install the protective covers for the pitot boom, engine air intake ducts and tailpipe. Also install canopy and electronic compartment sun shade if airplane is to be parked in sun for an extended period.



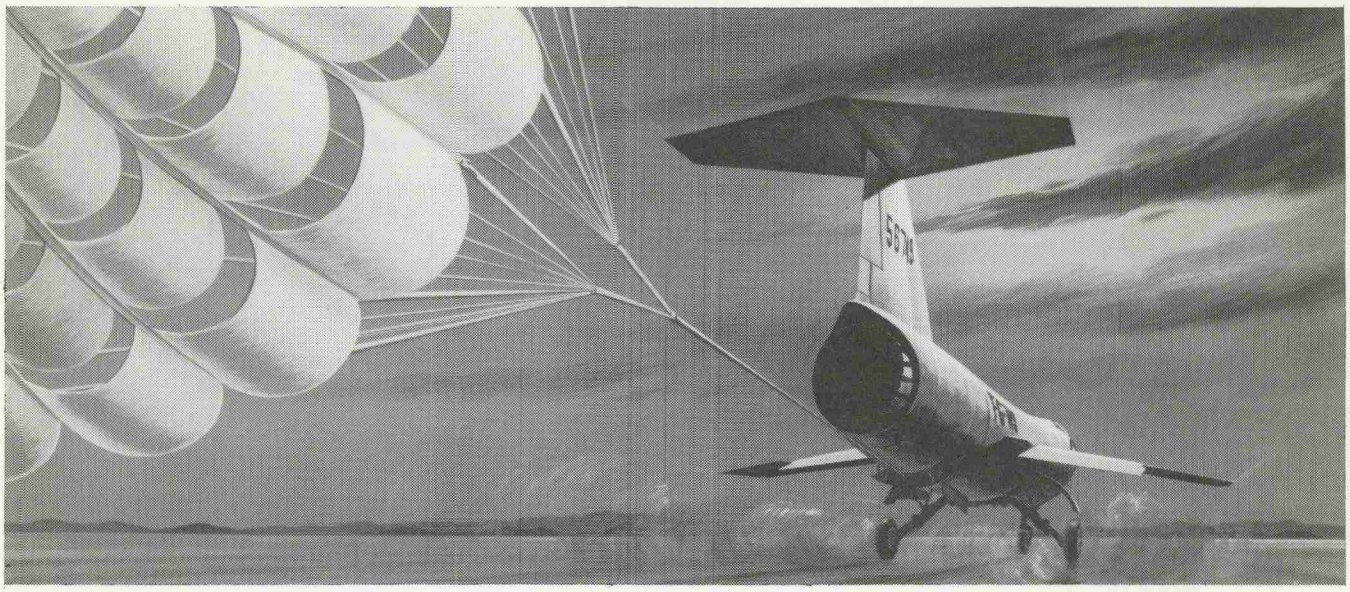
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performance data

APPENDIX I

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SUPPLEMENT
FLIGHT MANUAL



USAF SERIES
F-104A
AIRCRAFT

TITLE UNCLASSIFIED

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

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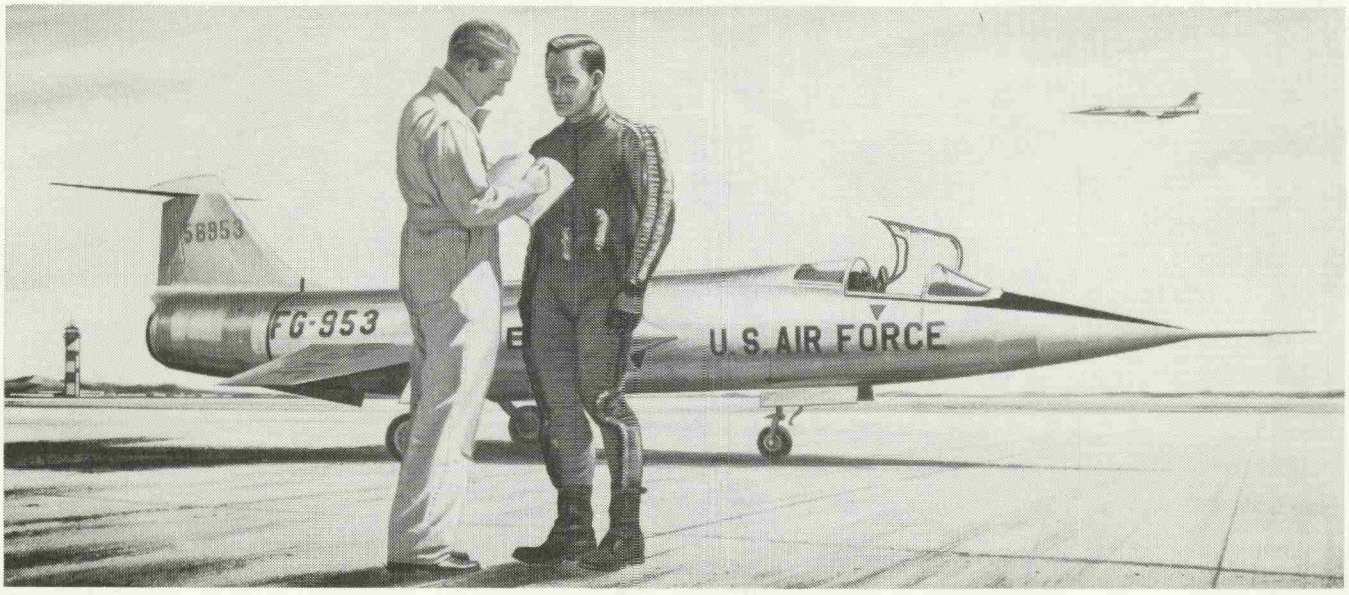
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FLIGHT MANUAL, T.O. 1F-104A-1

NOTE

This supplement contains confidential information pertaining to the F-104A Aircraft. Refer to the unclassified Flight Manual, T.O. 1F-104A-1, for all information not covered in this supplement.

This supplement will be revised, as necessary, the same as the unclassified Flight Manual. As confidential information is declassified, it will be removed from this supplement and simultaneously included in the unclassified manual. When there no longer is any confidential information pertaining to this airplane, this supplement will be discontinued.



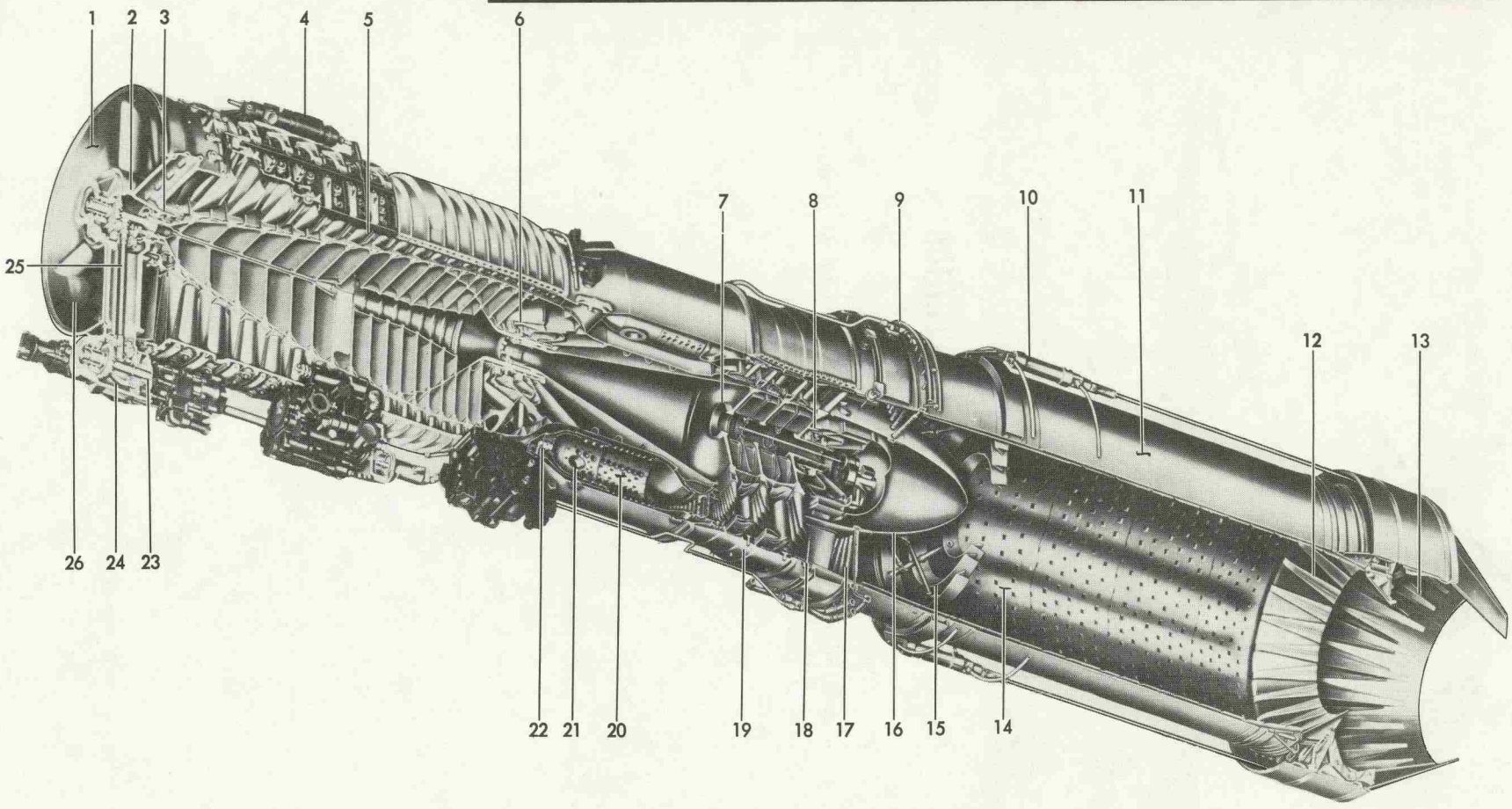
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description

SECTION I

All information on description is contained in the unclassified Flight Manual, T.O. 1F-104A-1 except for the following:

J79 TURBOJET ENGINE WITH AFTERBURNER

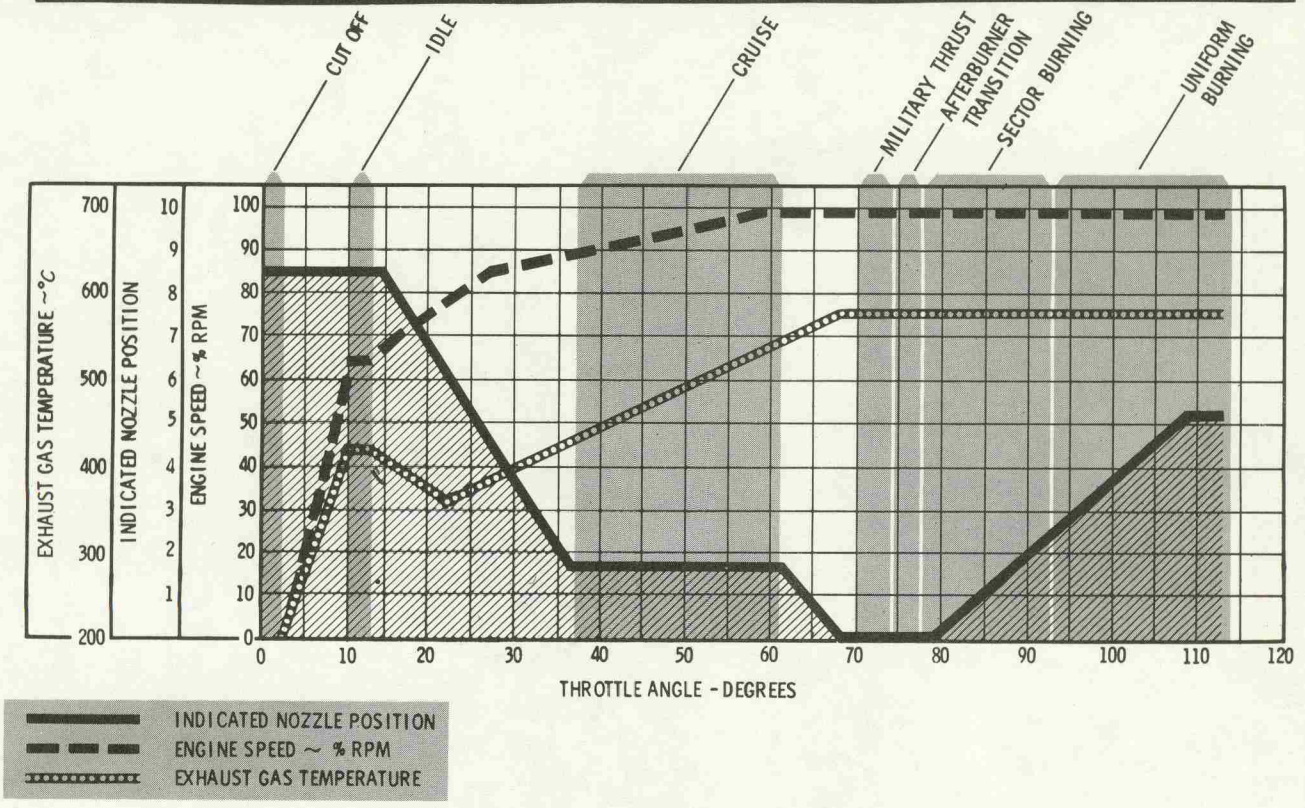


1 AIR INTAKE
2 FRONT GEAR CASE
3 NO. 1 BEARING HOUSING
4 VARIABLE VANE ACTUATOR
5 17 STAGE COMPRESSOR SECTION
6 NO. 2 BEARING HOUSING
7 THREE STAGE TURBINE WHEEL
8 NO. 3 BEARING HOUSING
9 AFTERBURNER FUEL MANIFOLDS
10 EXHAUST NOZZLE FLAPS ACTUATOR
11 TAIL PIPE
12 PRIMARY EXHAUST NOZZLE FLAPS
13 SECONDARY EXHAUST NOZZLE FLAPS
14 TAILPIPE LINER
15 FLAME HOLDER
16 EXHAUST CONE
17 AFTERBURNER SPRAY BARS
18 TAILPIPE TEMPERATURE THERMOCOUPLE
19 3 STAGE TURBINE SECTION
20 COMBUSTION CHAMBER
21 CROSS FIRE DUCT
22 FUEL NOZZLE
23 HORIZONTAL ACCESSORY DRIVE SHAFT
24 TRANSFER GEAR CASE
25 VERTICAL ACCESSORY DRIVE SHAFT
26 VARIABLE INLET GUIDE VANES

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Figure 1-3

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



NOTE

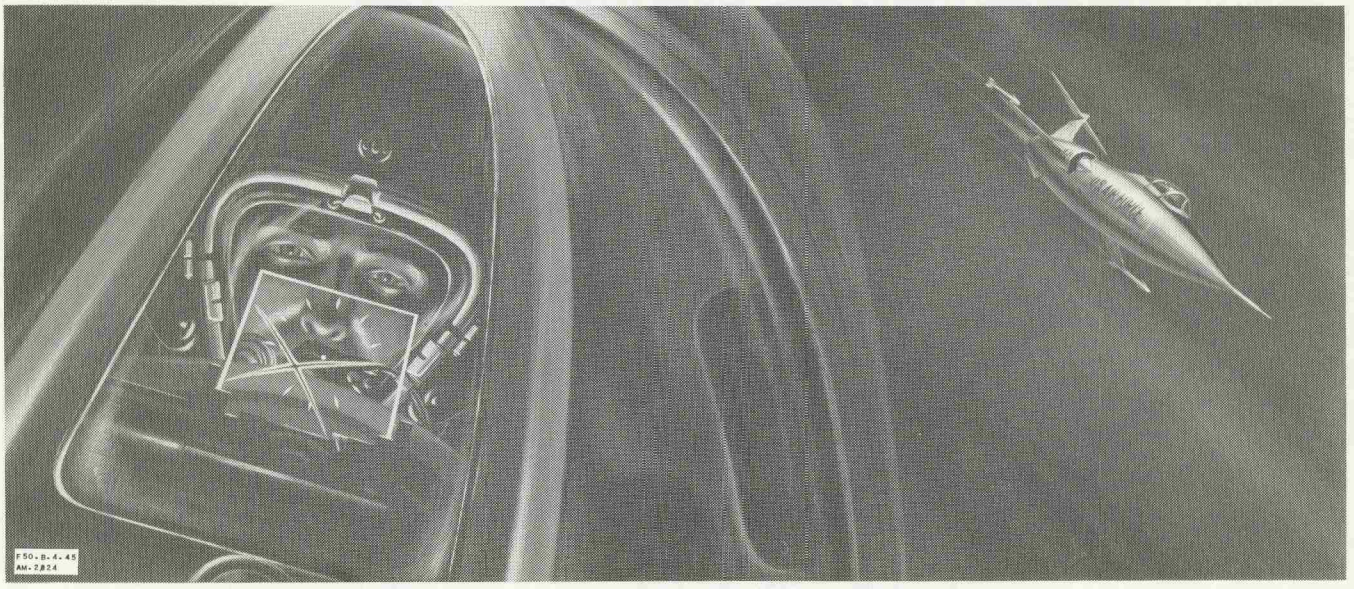
THE NOZZLE NORMALLY WILL FOLLOW THE MECHANICAL SCHEDULE AS DEPICTED BY THE BOUNDARY OF THE DIAGONALLY STRIPED AREA UNTIL EXHAUST GAS TEMPERATURE REACHES 575° C; AT THIS TIME, THE ELECTRONIC TEMPERATURE CONTROL WILL MODULATE NOZZLE AREA TO MAINTAIN THIS EXHAUST GAS TEMPERATURE. THEREFORE, THE INDICATED NOZZLE POSITION WILL NORMALLY BE IN EXCESS OF THE MECHANICAL SCHEDULE IN THE MILITARY AND AFTERBURNER REGION. NOZZLE POSITION INDICATIONS WILL VARY SLIGHTLY BETWEEN AIRPLANES AND MAY BE AS MUCH AS 0 TO 2 AT MILITARY THRUST. EXHAUST GAS TEMPERATURE IS THE MOST POSITIVE INDICATION THAT THE PROPER AREA IS BEING MAINTAINED. SHOULD THE NOZZLE AREA CONTROL SYSTEM FAIL TOWARD THE CLOSED POSITION, THE BOUNDARY OF THE DIAGONALLY STRIPED AREA REPRESENTS THE "FLOOR" OR MINIMUM AREA TO WHICH THE NOZZLE CAN CLOSE.

Figure 1-13

NOTE

For Sections II and III, refer to unclassified Flight Manual, T.O. 1F-104A-1.





auxiliary equipment

SECTION IV

All information on auxiliary equipment is contained in the unclassified Flight Manual, T.O. 1F-104A-1, except for the following:

ARMAMENT EQUIPMENT.

The armament consists of GAR-8 missiles. The missiles are fired by the camera-armament trigger switch on the control stick grip which also actuates an N-9 camera to record firing accuracy. Target searching and tracking information is provided by the AN/ASG-14 (MOD. I) fire control system. This system includes a light weight radar set which is effective to an altitude of 70,000 feet and is used in conjunction with an optical infrared sight system.

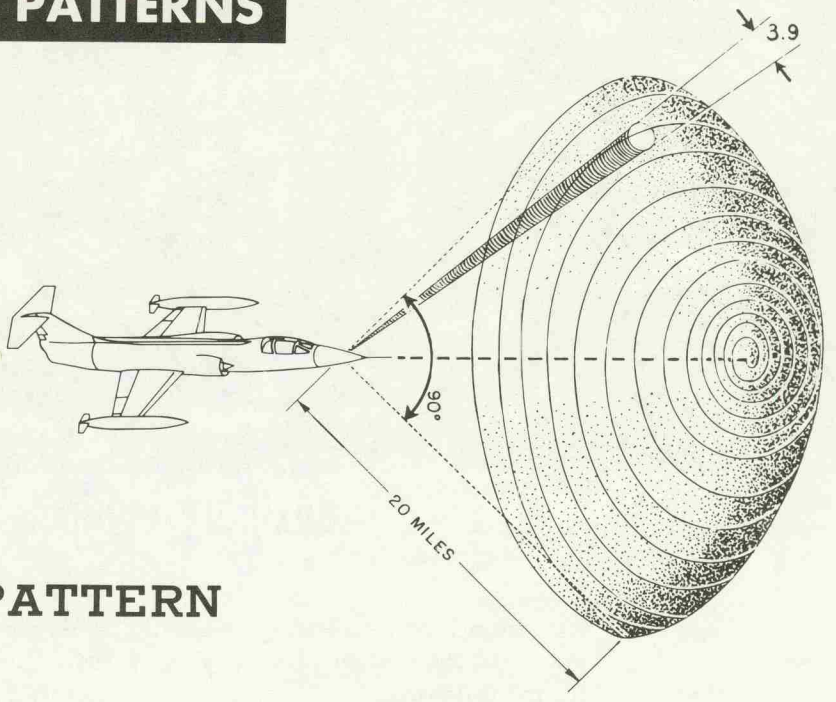
AN/ASG-14 (MOD. I) FIRE CONTROL SYSTEM.

The radar system, optical sight, and infrared sight combine to form a well integrated fire control system with a high kill potential. Although these systems complement each other, neither system is wholly dependent on any other for operation. This means a mission can be successfully completed with one or more of the systems inoperative. For instance, during the day a target could possibly be acquired visually without the use of the radar system and the attack completed. At night either the radar system or the infrared sight can be used alone for detecting and tracking the target to complete an intercept although the latter system will have considerably less search capability. In an extreme case, the GAR-8 missile alone can be used to complete a mission. During the initial phase of the attack, the target will normally be detected by the radar in search mode. The target presentation will appear on the radar scope in the form of an arc indicating target range and position relative to the interceptor. The aircraft should then be flown toward the target until within visual range, at which time the optical sight may be utilized to direct the remainder of the attack. An infrared sight may be used in conjunction with the optical sight to augment the night attack capabilities of the fire control system.

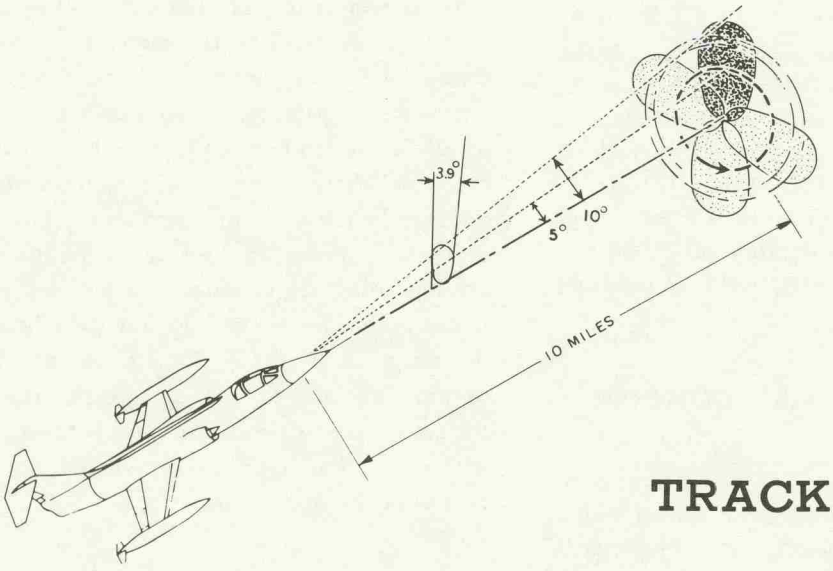
RADAR SYSTEM.

The AN/ASG-14 (MOD. I) radar system is a light weight search and tracking radar system. Its primary function is to provide the pilot with steering range information for tracking an airborne target, after the

ANTENNA PATTERNS



SEARCH PATTERN



TRACK PATTERN

F50-B-4-56

Figure 4-10

interceptor has been directed to the target area and radar contact established. It is inherent with this radar system (antenna scan and indicator display method) that a tail-on pursuit will be established as the interceptor pilot steers to obtain a target boresight presentation on the radar scope. This is the most desirable position for effecting a missile launch. The target tail-on aspect is optimum for infrared seeking devices (IR sight and GAR-8 missile); thus, the fire control and armament components are highly compatible and comprise an efficient and effective system. There are three pilot selected modes of operation: search, track and ECM HOM (Electronic Counter-Measures and Homing). The search mode provides recurrent radar coverage of a 90° conical volume forward, at a 3 second scan rate, and a 20 nautical mile range scale on the indicator. The track mode provides recurrent radar coverage of a 20° conical volume forward, at a 0.1 second scan rate, a 10 mile nautical range scale on the indicator, display of relative opening or closure rate to the target, and target lock-on capability for automatic range tracking. The ECM HOM mode provides capability of homing on jamming signals. The radar system is passive in this mode and therefore offers no range information. A.C. power for operation of the radar system is furnished by the No. 2 a.c. bus, while the d.c. requirements are received from the d.c. monitored bus.

Controls and Indicators.

All controls and indicators for the radar system, except the track action button, are located on the radar and armament control panel (26, figure 1-7) on the lower instrument panel.

Master Selector Switch. The main radar system control is a four-position, rotary, master selector switch (13, figure 4-11). This switch (in conjunction with the radar track action button on the control stick grip) enables the pilot to select the desired mode of operation. With the master selector switch in the OPER (operate) position, the transmitter is activated and the antenna scans a search or track pattern (figure 4-10), depending upon the mode of operation. The radar system begins to warm up as soon as power is supplied to the aircraft. To allow a warm-up period for the rest of the radar system before the transmitter is turned on, a time delay of approximately three minutes is incorporated between the time at which the engine is started and that at which the transmitter can be used. If the master selector switch is at the OPER position when the engine is started, the antenna and transmitter will be turned on automatically after the time delay, thus putting the radar system in operational status. If the master selector switch is in RESET STBY when the engine is started, both the transmitter and the antenna are inoperative, and both will remain so until the end of the warm-up period, even if the switch is turned to the OPER position during warm-up. While the engine is operating, the radar system may be temporarily deactivated by turning the master selector switch to the RESET STBY position, thereby stopping the transmitter and antenna. Since the time delay is effective only from a cold start, the radar system will be immediately operative when the master selector switch is turned to OPER or ECM HOM. The ECM HOM position enables the pilot to home on jamming transmission, since the radar transmitter is turned off (thus removing the normal radar echoes from the radar scope), while the antenna continues to scan as it does in the regular modes. The only indications on the radar scope will be those due to the source of the interference or some other source with roughly the same frequency. The final position on the master selector switch is the PUSH EMER (push for emergency operation) setting. Operation of the master selector switch to this position allows the radar system to be made fully operational immediately after the engine is started without waiting for the expiration of the time delay period.

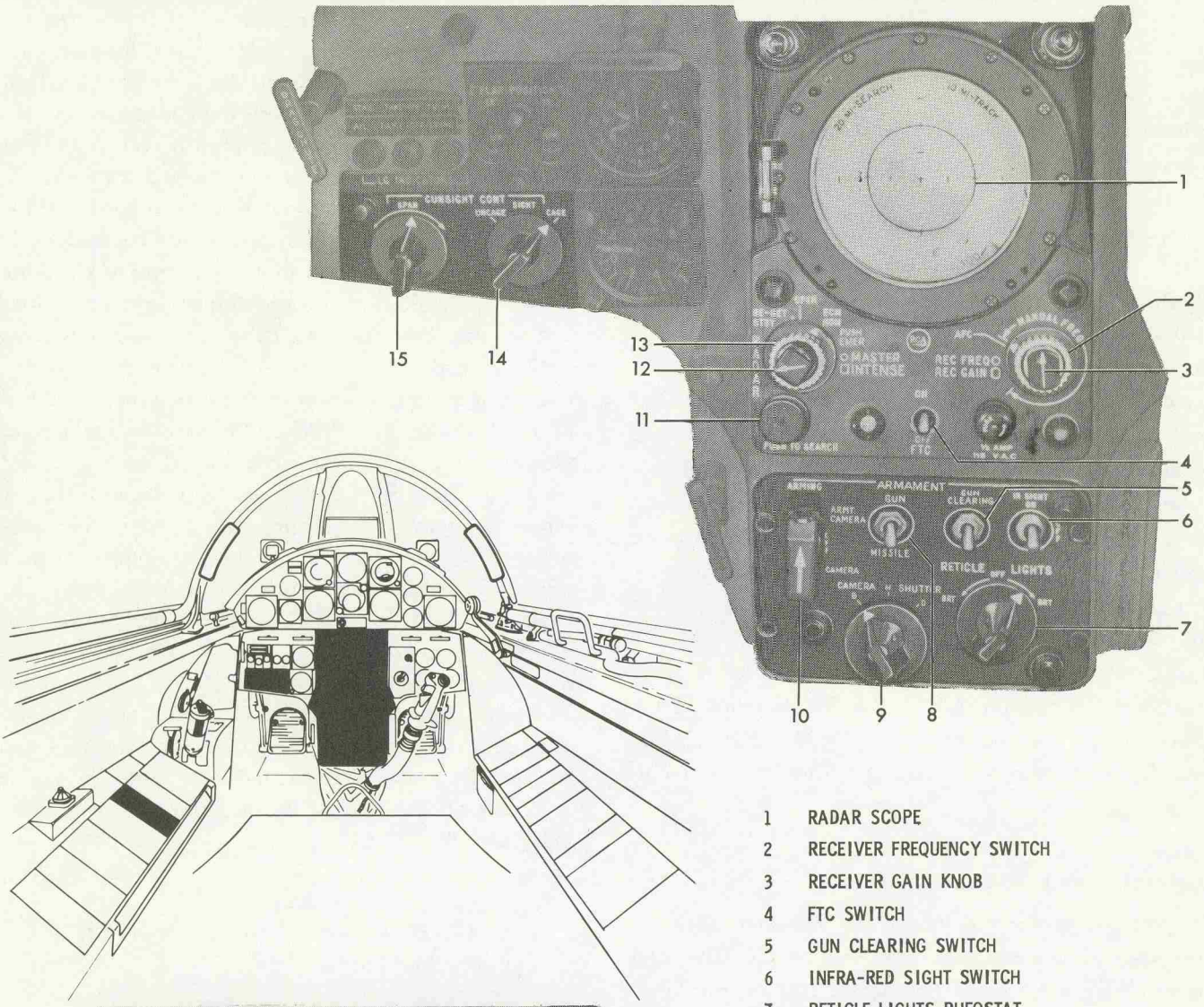
Note

The aircraft is delivered with the PUSH EMER position inoperative.

Radar Scope Intensity Rheostat. The intensity (INTENS) rheostat (12, figure 4-11) is a square knob mounted concentrically with the master selector switch. Radar scope presentation brilliance is adjusted by rotating this rheostat.

Receiver Frequency Switch. Both automatic and manual frequency control are incorporated in the radar system. When the receiver frequency (REC FREQ) switch (2, figure 4-11) is turned to the extreme counterclockwise (AFC) position, an in-out switch is operated and the receiver is automatically tuned for maximum response to the echoes from the transmitter with which it is associated. When the receiver frequency switch is turned clockwise out of detent, manual frequency operation is set up allowing the pilot to tune the receiver manually.

RADAR AND ARMAMENT CONTROL PANEL



1 RADAR SCOPE
2 RECEIVER FREQUENCY SWITCH
3 RECEIVER GAIN KNOB
4 FTC SWITCH
5 GUN CLEARING SWITCH
6 INFRA-RED SIGHT SWITCH
7 RETICLE LIGHTS RHEOSTAT
8 WEAPON SELECTOR SWITCH
9 CAMERA SHUTTER SELECTOR SWITCH
10 ARMING SWITCH
11 PUSH TO SEARCH BUTTON (LOCK-ON INDICATOR)
12 RADAR SCOPE INTENSITY RHEOSTAT
13 MASTER SELECTOR SWITCH
14 MECHANICAL CAGE SWITCH
15 TARGET SPAN SELECTOR SWITCH
16 MISSILE SIGNAL VOLUME RHEOSTAT
17 MISSILE FIRING SELECTOR SWITCH

F50-C-4-55

Figure 4-11

This manual operation is critical in its tuning and should be used only when automatic frequency control fails or during ECM HOM mode operation.

Receiver Gain Knob. The receiver gain (REC GAIN) knob (3, figure 4-11), mounted concentrically with the receiver frequency switch, determines the sensitivity of the radar system to the echoes picked up by the antenna. The knob should be kept in its maximum clockwise position. Gain adjustment is a pre-flight position and should not be changed during flight except to overcome jamming.

Push To Search Button and Lock-on Indicator. The function of the push to search button (11, figure 4-11) is to return the radar system to the search mode after the track mode has been used. A red lock-on indicator (11, figure 4-11) is mechanically integrated with the push-to-search button. This indicator will be illuminated when tracking has been successfully initiated and the radar system is locked on to a target. There is an additional amber lock-on light by the optical sight combining glass.

FTC Switch. Fast time constant (FTC) circuitry has been incorporated as an anti-jamming measure, and to eliminate long formless trains of cloud, ground or sea return echoes. This switch (4, figure 4-11) should be left on unless it is felt the target is being degraded or weakened.

Radar Track Action Button. Changing the radar system from search mode to track mode is accomplished by means of the push-button type track action button (4, figure 1-23) on the control stick grip. This button also slews the range strobe out and in.

Radar Test Panel. A test panel (18, figure 1-8) located on the left console is provided primarily for pre-flight tests of the system by ground crew personnel. An eleven position rotary selector switch (figure 4-12) enables a quick check of various circuits under operating conditions, a good-bad indication being obtained from a monitor meter located just above the switch. Each position of the switch is designated both by nomenclature



Figure 4-12

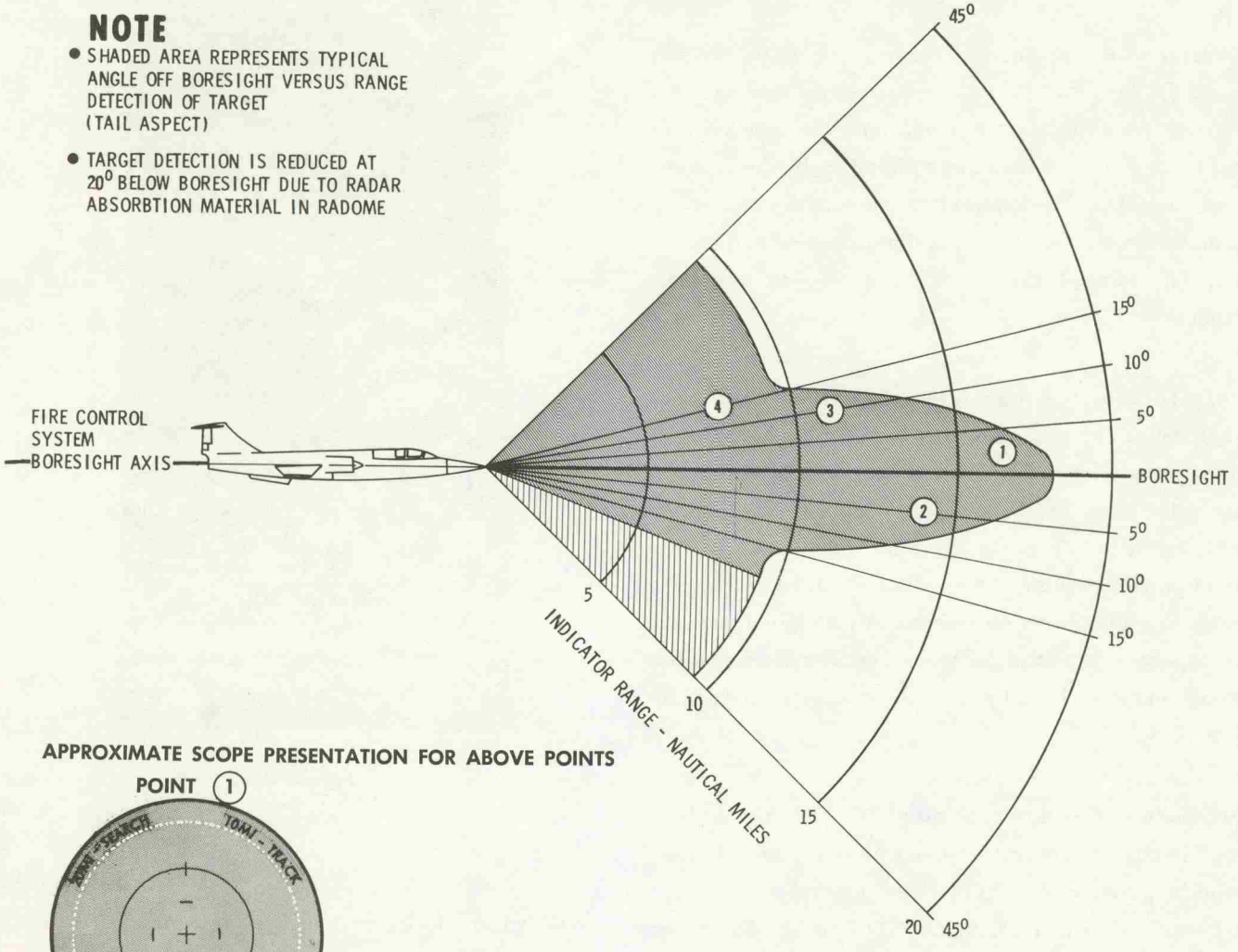
which shows the circuit selected for test, and by color code which shows on which of the three meter scales (correspondingly color-coded) the indication is to be checked. An on-scale reading indicates that the circuit is functioning normally while an off-scale reading indicates the circuit requires adjustment.

Radar Scope. The radar scope (1, figure 4-11) is located on the radar and armament control panel. Two different clamp-on filters are used to cover the scope face. A light green polaroid filter is used for daytime flying. Its primary function is to minimize sun glare and bright ambient lighting interference with the scope presentation. The second filter is a deep red adjustable one used for night flying to reduce glare and brightness emanating from the scope. Ground personnel will install the proper filter for the mission.

RADAR DETECTION ENVELOPE

NOTE

- SHADED AREA REPRESENTS TYPICAL ANGLE OFF BORESIGHT VERSUS RANGE DETECTION OF TARGET (TAIL ASPECT)
- TARGET DETECTION IS REDUCED AT 20° BELOW BORESIGHT DUE TO RADAR ABSORBTION MATERIAL IN RADOME



APPROXIMATE SCOPE PRESENTATION FOR ABOVE POINTS

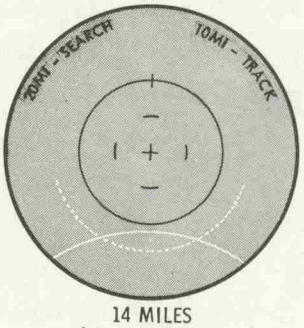
POINT (1)

17 MILES
ALMOST BORESIGHT

NOTE

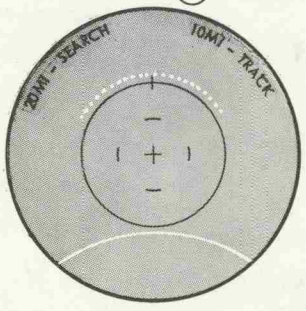
AT RANGES LESS THAN 7 MILES
TARGET APPEARS AS A SOLID ARC

POINT (2)



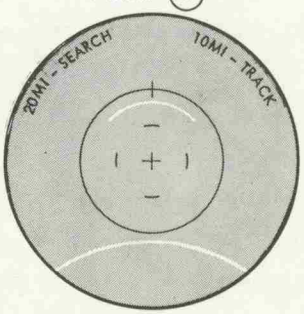
14 MILES
5° OFF BORESIGHT

POINT (3)



11 MILES
10° OFF BORESIGHT

POINT (4)



8 MILES
15° OFF BORESIGHT F50-C-1-111

Figure 4-13

Range Rate Indicator. (Rate of closure.) A range rate indicator is integrated with the radar scope to show, in the track mode, relative range rates of the target and interceptor from 100 knots opening to 400 knots closing.

RADAR SYSTEM OPERATION.

Search Mode.

To operate in the search mode, turn the master selector switch to OPER. The radar is now in search mode operation. Check the following control positions: receiver gain knob full clockwise, receiver frequency switch in AFC detent, FTC switch on. Then adjust the radar scope intensity rheostat until background noise (in clutter free area) is just discernible. Setting the rheostat too high can result in a display which might obscure a target (figure 4-14). Slight readjustment may be desirable at times during the flight. In search mode the maximum displayed range is 20 nautical miles; hence, the engraved markers on the radar scope represent 15, 10, and 5 miles. The antenna beam scans a 90° conical volume (figure 4-10) ahead of the aircraft once every 3 seconds (approximately). Complete coverage of this area is accomplished by spiraling the antenna, about the radar boresight axis in a series of decreasing spiral diameters, from the maximum off-boresight angle (45°) to boresight. Since it takes the antenna approximately 3 seconds to nod from its maximum angle off to the center (one frame time) a target will be painted only once every 3 seconds. This causes a search mode target to pulsate on the radar scope at a 3 second rate. The radar boresight (also optical sight and IR sight) is 2¼° below the longitudinal axis of the aircraft. This condition coupled with the aircraft's angle of attack dictates an altitude separation between target and interceptor to align the target with the aircraft's fire control system boresight axis. This separation is on the order of 300 feet per mile. A boresighted target at 5 miles will be 1500 feet above the aircraft; at 10 miles it will be 3000 feet above, etc. A typical case is illustrated in figure 4-16. The search mode target range versus off-boresight angle envelope (figure 4-13) should be thoroughly studied so that optimum radar usage can be obtained. The shape of the radar detection envelope will remain essentially the same for any target. Ranges at all angles will increase or decrease proportionally depending upon the target size. The example shown in figure 4-13 illustrates a maximum usable boresight range of 18 miles for a given

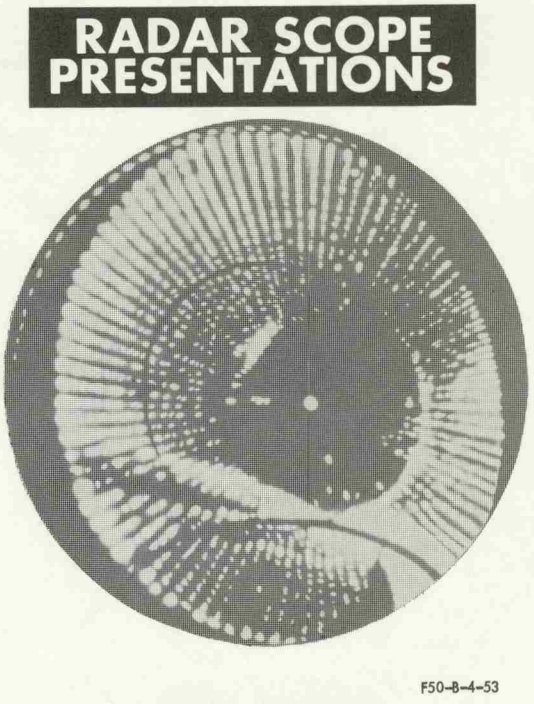


Figure 4-14

target. Therefore, the maximum usable ranges, against this same target, decrease as the target angle off-boresight increases, as shown, to where 9 miles would be maximum at angles off greater than 15°. This particular example is typical of a large jet bomber. Because of this search mode detection envelope, the relationship of airplane angle of attack, fire control system and missile boresight must be fully understood so that the airplane nose up attitude can be used to best radar advantage. This is accomplished by allowing for an altitude separation between target and interceptor. Flying co-altitude presents another disadvantage in that the control target detection would be in the 6 o'clock indicator position where the horizon clutter paints. Target recognition is naturally more difficult in this area. Target information displayed on the radar scope will immediately show the clock position of the target relative to the interceptor radar boresight axis (A, figure 4-15) and range to the target. Commencing with A, figure 4-15, consider the interceptor as the center point position of the radar scope and interpret this presentation to determine the target's position and range relative to you. The target displayed is a small arc at the 9 o'clock position, thus the target is to your left (at an angle not greater than 45°) and at a range of 7 to 8 miles (denoted by the distance from the center of the radar scope to the arc). The interceptor should then be steered in the direction of the target arc

RADAR SCOPE PRESENTATIONS

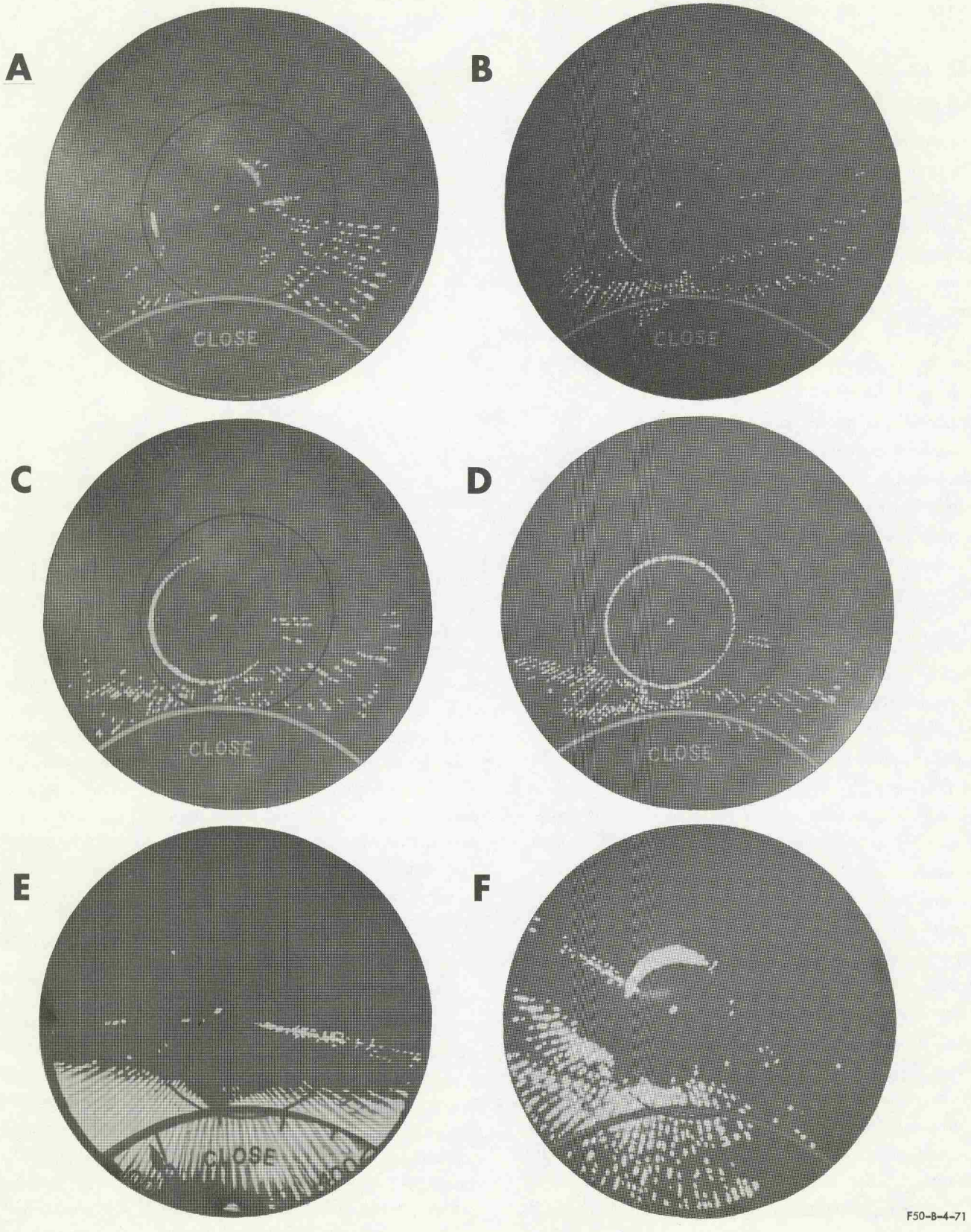
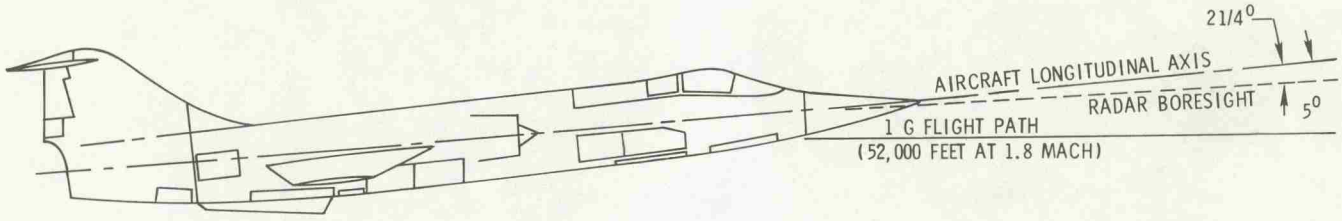


Figure 4-15

FIRE CONTROL SYSTEM BORESIGHT



NOTE ANGLES ARE EXAGGERATED IN THIS DRAWING FOR PURPOSES OF ILLUSTRATION

F50-C-1-112

Figure 4-16

(left) with a resulting increase of target arc length (B, figure 4-15). Continuing to steer in the direction of the arc will result in corresponding presentations (C and D, figure 4-15). A complete circle indicates that the target is boresighted (in line with the radar boresight axis). The length of the target arc is an indication of the target's angle off-boresight. (Figure 4-13 illustrates the relationship.) Return signals from the ground are also displayed on the radar scope in search mode due to the wide angle of antenna scan. In straight and level flight and with the FTC switch off (which disables anti-clutter circuit), the ground return (hereafter referred to as clutter) appears as in E, figure 4-15. Targets could not be discerned within this area but the clutter display is useful as a horizon reference, thus this clutter is referred to as horizon clutter. F, figure 4-15 reveals the effect of the anti-clutter circuit when the FTC switch is on. It is now possible to discern a target in this clutter area, while the broken up horizon clutter still serves as a horizon reference. The position of the horizon clutter in F, figure 4-15 indicates that the interceptor is in a left bank and also shows another ground clutter display which is a normal occurrence when the interceptor banks. It is the crescent-shaped clutter appearing at the 12 o'clock position and is referred to as altitude clutter since its range on the radar scope does correspond to the interceptor's altitude above the terrain. Its clock position must be disregarded since it is not a true relative position indication. The example is shown here for purposes of recognition so as not to confuse the altitude clutter presentation with a target arc. The radar scope is normally void of altitude clutter in straight and level flight unless the terrain below is particularly rugged.

Track Mode.

Track mode is initiated by momentarily depressing the track action button. At this time the antenna reverses its spin direction and counterweights pull back one half of the antenna reflector (disk) slightly. This forms a fan shaped beam of approximately 4° by 10° (figure 4-10). The antenna does not spiral in track mode but spins about a fixed point (offset slightly from the boresight axis) and a 20° conical volume is scanned. Therefore, the track mode envelope provides essentially a constant range versus angle off boresight (±10°). The maximum

RADAR SCOPE PRESENTATIONS

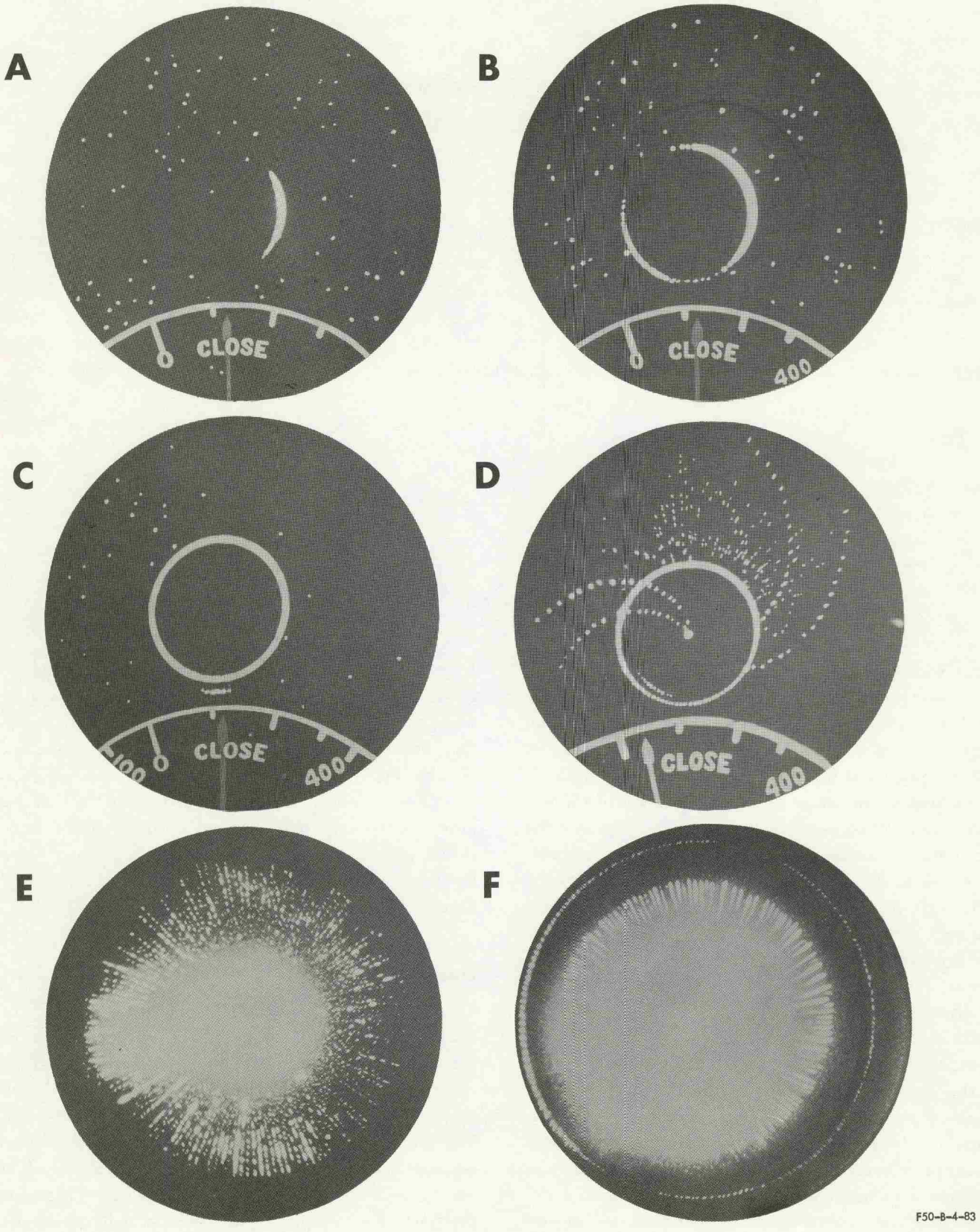


Figure 4-17

displayed range on the radar scope in the track mode is 10 nautical miles and the engraved markers now represent 7.5, 5, and 2.5 miles. Track mode should not be initiated until the search mode display indicates target range less than 10 miles and arc length 90° or greater. This will assure that the target is within the track cone. Readjustment of the radar scope intensity rheostat may be required when switching modes. The track mode target presentation is interpreted in the same manner as in search mode. There is the addition of the range strobe in track operation. The strobe appears as a circle and slews out and in with successive actuation of the track action button and remains stationary when the button is released. An automatic slew feature of the radar will continuously slew the strobe from 200 yards to 3000 yards whenever it is manually slewed into this range. The strobe will then automatically lock on to the closest target it sees within this range. Manual lock-on, of targets beyond 1.5 miles, is accomplished by holding the track action button down and slewing the strobe to the target. Release the button when the strobe coincides with the target to acquire lock-on. If desired, the range strobe can be slewed toward the target and stopped just short of the target range. As target range decreases, lock-on will occur when the target and strobe coincide. The radar lock-on indicator as well as one by the left side of the optical sight combining glass will illuminate to indicate lock-on and the range strobe disappears from the display. When lock-on is accomplished, the range rate meter on the radar scope will show the opening or closure rate in knots (true speed) and range information is provided to the optical sight. Typical track target presentations are shown in A, B and C, figure 4-17. Commencing with A, a target to your right at 3 miles is displayed. Steering right in the direction of the arc results in a near boresight (B) and then boresight (C). A range strobe is not present in these photographs as lock-on exists. The track presentation is normally entirely void of any ground clutter. Altitude clutter may appear during interceptor bank angles of 45° or greater or when flying over water. In the latter case, it takes on the appearance of a boresighted target, but is recognizable as altitude clutter by remaining at a range corresponding to the interceptor altitude. A classic example of interference from other radars is illustrated in D, figure 4-17. This would appear the same in search mode also. Since the antenna scans only a 20° volume ahead of the aircraft in track mode, at a 0.1 second scan or spin rate, the target display appears continuously (does not pulsate as in search mode). This allows the pilot to track the target with greater precision and maintain target boresight which will assure a satisfactory overlap with the missile boresight cone.

ECM HOM Mode.

ECM HOM mode is selected by rotating the master selector switch to ECM HOM. In this mode the radar scope sweep is noticeably shortened for positive indication of ECM HOM mode. The radar system becomes passive, i.e. the transmitter is disabled, and only the receiver is operative allowing the system to be used as a direction finder. During radar system operation, jamming devices or other radar equipment may interfere with target selection and acquisition. The display presented on the radar scope may take a variety of forms, both symmetrical and irregular, depending on the particular type of interference encountered. When ECM HOM mode is selected, the receiver gain knob must be full clockwise, then the receiver frequency switch must be turned out of AFC detent and rotated until the interference is tuned in. The receiver gain knob may then be adjusted along with the radar scope intensity rheostat, to produce a non-blooming, nicely defined presentation on the radar scope. Frequent readjustment of the receiver frequency switch may be necessary. E, figure 4-17 shows an off-boresight source of continuous wave jamming tuned in ECM HOM mode. The pilot would steer toward the center of this splash of interference just as he would turn toward an off-boresighted target arc. When the interference becomes a circular pattern, F, figure 4-17, the source is boresighted. No range information is available from the radar system in ECM HOM mode. Chaff released from an aircraft within the zone of radar detection produces indications on the display that resemble targets when the radar system is in search or track modes. These indications are readily detected. Since chaff has little or no horizontal velocity, it will always appear as the innermost target or targets on the radar scope. ECM HOM mode usually will be selected only when jamming totally obliterates the target and renders search and track mode useless. There are many different forms of radar jamming that are clever and almost impossible to combat. Therefore, the successful tracking of a target through jamming will depend upon the pilot's experience and skill.

NORMAL OPERATION OF THE RADAR SYSTEM.

Before Take-off.

1. Four radar circuit breakers—In. (Check)
2. Master selector switch—RESET STBY.

After Take-off.

1. Master selector switch—OPER.
2. Receiver frequency switch—AFC.
3. FTC switch—ON.
4. Receiver gain knob—Extreme clockwise (maximum gain) position.
5. Radar scope intensity rheostat—Desired brilliance.
6. Proceed with mission.

EMERGENCY OPERATION OF THE RADAR SYSTEM.

No Radar Scope Presentation.

If the display on the radar scope disappears completely, use the following procedure:

1. Check the circuit breakers on the left console to determine if one of them has popped out. To reset a circuit breaker, depress it momentarily. If the power failure was due to a temporary overload the breaker will hold.
2. If the defect is still present, turn the master selector switch to the RESET STBY position and return to the OPER position.
3. If the previous steps are ineffective, remove the fuse on the radar and armament control panel and replace it with the spare which is mounted on clips to the left of the radar scope.

Erratic Radar Scope Presentation.

If the presentations on the radar scope are erratic, the radar test panel may be used to check certain circuits in the radar system as follows:

1. Rotate the eleven position rotary selector switch to each position in turn, checking that the indication lies within the range of the scale corresponding to the color-coded dot revealed by the cut-out in the flange of the selector knob.
2. Since the circuits checked are all within the radar nose package, the pilot can do nothing about an abnormal indication except when the indication is obtained in the AFC XTAL positions. With defective automatic frequency control the radar receiver will be detuned. By operating the receiver frequency switch to the MANUAL FREQ range and adjusting for an optimum presentation, the pilot will be able to complete the mission. (The receiver frequency switch may require periodic readjustment.)

Note

An abnormal reading does not necessarily indicate that the radar system can not be used at least to some extent, although its performance will not be up to standard.

Erratic in Search Mode Only.

If the radar system becomes defective only when operated in the search mode, it is possible to initiate and complete an attack in the track mode. The following procedure may be used to locate and acquire the target:

1. With the master selector switch in the OPER position, depress the radar track action button and hold it depressed until the strobe has run out to the edge of the radar scope.
2. Using the normal operating procedure previously outlined, adjust the display and pick out a target.
3. Again depress the radar track action button and hold it down until the strobe is coincident with the target.

OPTICAL SIGHT SYSTEM.

The optical sight is a gyro computing sight which automatically computes proper lead angle for aerial gunnery attacks. During the missile mode attack the optical sight

is mechanically caged. The mechanically caged sight presents to the pilot a zero lead angle which is one degree blow the physical boresight of the GAR-8 missile. The GAR-8 missile has a search cone of 56° after release, thereby being completely compatible in tracking with the fixed AN/ASG-14 optical sight. Electrical power for the optical sight is derived from the No. 1 a.c. bus and the d.c. monitored bus.

Reticle Lights Rheostat.

The reticle light rheostat (7, figure 4-11) controls the intensity of the reticle on the optical sight combining glass. The reticle brightness can be adjusted by turning the rheostat from OFF toward BRT in either direction. The reticle lamp is a dual filament lamp. The second filament (counterclockwise rotation) is used as a standby. Use the first filament (clockwise rotation) for optimum reticle brightness. Power is derived from the d.c. monitored bus.

Mechanical Cage Switch.

The mechanical cage switch (14, figure 4-11) is provided to cage the sight gyro. The gyro may be caged by turning the switch clockwise to the CAGE position and may be uncaged by moving the switch counterclockwise to the UNCAGE position.

CAUTION

The gyro should be mechanically caged for all take-offs and landings to avoid damaging the sight.

Target Span Selector Switch.

The target span selector switch (SPAN) (15, figure 4-11) enables the pilot to adjust the sight reticle size to some definite dimension on the target by turning the switch. Any desired span setting from 12 to 60 feet can be set into the sight. The selected span numbers will appear on the lower right side of the optical sight combining glass whenever the range numerals indicate 3000 feet or over. When the sight is mechanically caged and the radar and manual range grip in detent position, the optical sight is fed a fixed 1200 foot range signal.

Electrical Cage Switch.

The sight reticle image may be stabilized at approximately zero lead angle, electrically, by depressing the electrical cage button on the throttle (3, figure 1-6). The sight will be uncaged and operable when the electrical cage button is released. The mechanical caging of the optical sight will override this feature and will hold the sight gyro at zero lead angle, regardless of the position of the electrical cage button. Electrically caging the optical sight during level flight with the sight mechanically uncaged will cause the sight gyro to stabilize to approximately 1-2 mils of the zero lead angle and will not alter target range numerals in the sight presentation.

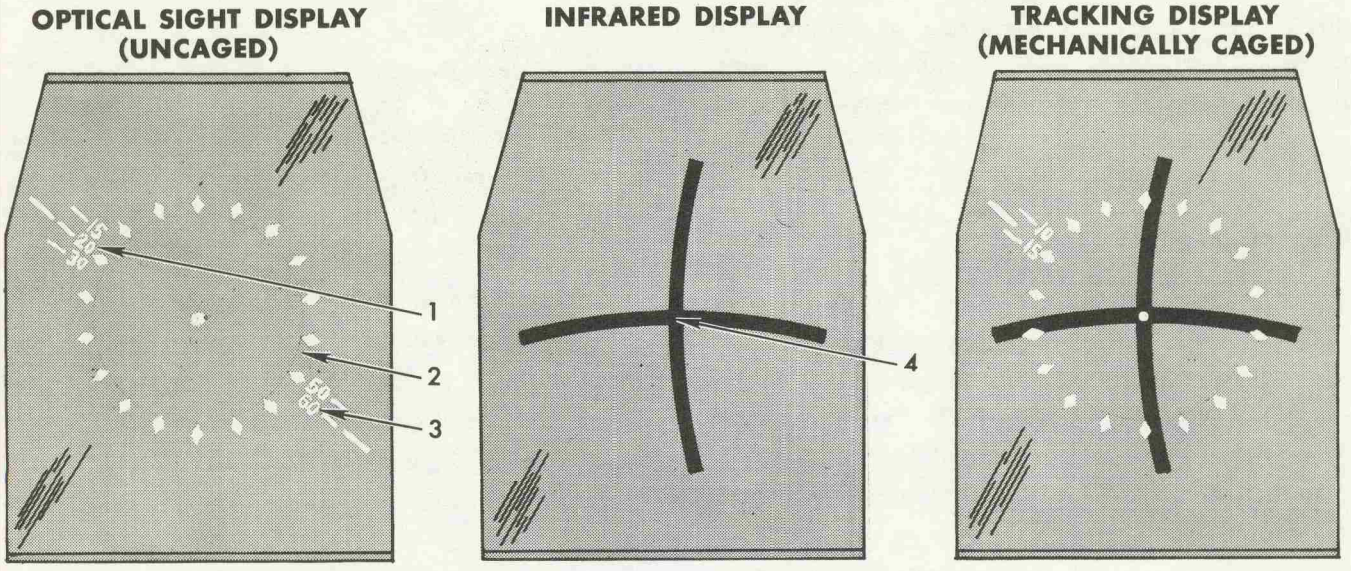
Radar and Manual Range Grip.

The throttle incorporates both radar ranging and manual ranging controls. Turning the radar and manual range grip (2, figure 1-6) to the full counterclockwise position until the detent is engaged allows radar range to be supplied to the sight head from the radar system unless the optical sight is mechanically caged. When the optical sight is in radar mode and mechanically caged, a fixed range signal of 120 feet is fed into the optical sight. The manual range feature of the optical sight is not used in a missile mode attack.

Sight Combining Glass Display.

The sight combining glass display (figure 4-18) is composed of a variable diameter reticle image, target range, and target span numerals. The range numerals at the 10 o'clock position are in hundreds of feet. Figures visible are 10 through 50 (1000 to 5000 feet). Target span numerals which are located at the 4 o'clock position are in actual feet. Span numerals range from 12 to 60 feet.

SIGHT COMBINING GLASS DISPLAY



1. RANGE NUMERALS (HUNDREDS OF FEET)
2. RETICLE IMAGE
3. TARGET SPAN NUMERALS (FEET)
4. TARGET LOCATION INDICATED BY INTERSECTION OF TWO ARCS

F50-B-4-61

Figure 4-18

The optical sight will present a 50 mil diameter reticle when target span setting is 60 feet, sight is mechanically caged, and the radar and manual range grip is in the detent position.

Armament Ground Test Panel.

A sight head and firing override test panel (20, figure 1-8) is located on the left console. This panel is used by ground maintenance personnel only.

INFRARED SIGHT SYSTEM.

With the use of the infrared (IR) sight, attacks can be made in cases where the pilot is unable to see the target visually. The infrared radiation emitted by the target is scanned by a disc having two pairs of slits and a field of view of approximately $\pm 7°$ from zero lead angle. The energy passing through a certain portion of the slits is detected by an infrared sensitive photocell and turned into electrical energy. The electrical energy is amplified and actuates a neon lamp. A disc similar to the scanning disc rotates in front of this lamp. Whenever the lamp is lit, an illuminated arc will be projected onto the combining glass of the optical sight (figure 4-18). Since there are two pairs of slits and the arcs in each pair are effectively perpendicular, the lines will form a cross. With the optical sight mechanically caged (missile mode), the pilot merely superimposes the fixed optical sight pipper over the infrared cross. Maximum range of the infrared unit is dependent on atmospheric conditions, type of aircraft, exhaust gas temperature, and area of relatively hot skin surfaces near the engine. A large four engine jet bomber target at 14-15 miles range will appear as a single cross. As target range decreases, the IR sight will paint two targets (2-engine clusters). If target range is decreased even further, the IR sight may actually define each engine by a thick horizontal arc and as many properly spaced vertical arcs as there are engines. The main intersection of the various arcs defines each engine or each aircraft as in the case when chasing a close flying formation. If the IR sight views no definite target, the general background return appears as an infinite number of small targets. The IR sight then paints many targets or intersections which appear like a basketweave pattern.

Infrared Sight Switch.

The infrared (IR) sight switch, (6, figure 4-11) is the main control for the infrared sight system. When the switch is in the ON position, power from the d.c. monitored bus energizes the IR stroke lamp, and a number of red lines in a basketweave pattern will appear on the sight combining glass. These random lines will break up into the form of a cross (figure 4-18) when the infrared unit senses a target. The intersection of this cross is the target position.

GAR-8 MISSILE SYSTEM.

The GAR-8 is a passive infrared-homing air-to-air guided missile. It homes on infrared energy radiated by heated parts of the target. Since the missile is a passive homing device, it does not need to transmit a signal for guidance and is, therefore, relatively impervious to jamming. Launch conditions are a function of range, altitude, missile seeker tone and G load at launch time. When launched, the Gar-8 missile turns into a pursuit course to the target. If the target maneuvers, the missile turns at about four times the turning rate of the target to maintain the required course. After launching, the aircraft is free to maneuver since all corrections originate within the missile. The missile is 5.0 inches in diameter, 9 feet in length and weighs 155 pounds. It consists of four sections: guidance and control section, warhead, influence fuse, and rocket motor. The guidance and control section contains the optical system for tracking, a hot gas-operated control servo with aerodynamic fins to control the flight of the missile to the target, the electronic components to convert target signals into missile control signals, a hot gas-driven generator to supply electrical power during missile flight, and the contact fuse. While the missile is on the aircraft, stand-by electrical power is furnished by the aircraft's No. 2 a.c. bus and d.c. monitored bus through an umbilical connection to the launcher. The warhead, a 25-pound controlled fragmentation type, is detonated by the action of either the contact or the influence fuse. When exploded, it puts out approximately 1500 metal fragments that travel 6000 feet per second and are capable of penetrating 1 inch of aluminum or $\frac{3}{8}$ inch steel plate at 30 feet. The influence fuse converts near misses into kills. If the missile passes within 30 feet of the target, this fuse detonates the warhead. The motor is a standard 5 inch HVAR rocket motor which accelerates the missile to Mach 1.7 above the speed of the launching aircraft. This section also carries the fixed aft stabilizing fins and the suspension lugs.

Arming Switch.

A guarded arming switch, (10, figure 4-11) powered from the d.c. monitored bus, is used to energize the weapon selector circuit. The armament switch positions are: ARMT CAMERA, OFF, and CAMERA.

Weapon Selector Switch.

The weapon selector switch (8, figure 4-11) is a two position switch. The switch is labeled GUN and MISSILE. When placed in the MISSILE position, the missile firing selector switch is energized.

Missile Firing Selector Switch.

The missile firing selector switch (17, figure 4-11) is used to select either or both missiles for firing. The switch has four positions labeled SAFE, LH, RH, and BOTH. With missiles installed and the aircraft electrical system energized, closing the missile power breakers will operate the missile seeker heads regardless of the position of the missile firing selector switch or other firing circuits. With the missile firing selector switch in the SAFE position, the missiles cannot be launched. To launch either or both missiles, the missile firing selector switch must be placed in LH, RH, or BOTH; the weapon selector switch must be in MISSILE; the guarded arming switch in the AMT CAMERA position and the camera and armament trigger switch depressed into the second detent. Hold down the trigger switch 1-2 seconds to allow missile squib-gas generator build-up before missile motor ignition.

Note

One missile may track the other if both missiles are fired together.

Missile Signal Volume Rheostat.

The missile signal volume rheostat (16, figure 4-11) is used to adjust target signal versus background signal of the missile selected by the missile firing selector switch. The missile signal from the right hand missile is heard on the pilot's earphones when the missile firing selector switch is set in the BOTH position. The missile seeker audio tone may be checked on the ground by passing a flashlight across the seeker head at a distance of approximately 10 feet.

Camera-Armament Trigger Switch.

The missile firing and camera circuits are energized by depressing the camera-armament trigger switch (3, figure 1-23) on the control stick grip. The trigger has two detent positions. With the arming switch in the CAMERA position, the camera will operate when the trigger is depressed to either detent position. With the arming switch in the ARMT CAMERA position, the first detent position of the trigger completes the camera circuit and the second detent position fires the missiles. Electrical power for the trigger is from the d.c. monitored bus.

Note

A delay feature on the N-9 camera system will maintain the camera in a running condition up to three seconds after the trigger is released. To extend the camera overrun time it will be necessary to hold the trigger in the first detent position.

Missile Jettison Controls.

The external stores jettison controls may be used to jettison the missiles and launchers. (Refer to External Stores Release Selector Switch and External Stores Release Button in Section I.)

WARNING

Do not jettison bare launchers as wind tunnel tests show unsafe jettison characteristics.

NORMAL OPERATION OF MISSILE LAUNCHING SYSTEM.

Switch Sequence and Firing Procedure.

After the pilot is in the cockpit and normal prestart operations have been completed, the missiles should be checked. With external power on, the seeker heads should be in operation and the pilot will be able to make an audio check to ascertain satisfactory missile seeker operation. When the missiles are installed on the launchers, protective covers are in place over the missile heads. As the pilot first listens to the audio signal on the selected missile, a very faint signal will be received. When ground crewmen remove the covers prior to taxiing, a louder signal will be heard as the seeker system picks up the infrared radiations from the surrounding area. A light should be directed into the seeker head with a corresponding signal increase in the earphones; this will provide a satisfactory check of seeker operation. On climb out each missile should be checked on another aircraft in the flight. This check will help ascertain that the missile is still in operation, give the pilot an indication of relative signal strength as the target changes position in the seeker view, and most important give the pilot a chance to check correct boresight and maximum signal strength from the missile with the mechanically caged pipper. The missile seeker head has a null area of approximately 10 mils at its center, (very similar to human night vision) and best detects radiation slightly off center. Each aircraft should have a fairly constant boresight vs maximum signal discrepancy which should be placarded in the cockpit.

Note

The missile seeker system derives power from the aircraft electrical system and operates throughout the flight.

Missile Firing.

To get a hit the missile must be given every advantage and should not be considered infallible. Basically, there are two areas or envelopes which the aircraft must be in before launching the missile: (1) The radiation envelope of the target and (2) the performance envelope of the missile. The radiation envelope of the target will be dependent on its configuration: prop or jet, number of engines, and altitude. The performance envelope of the missile is determined by the maximum and minimum guidance range of the missile and the maximum G that the missile can pull at a given altitude. When the attacking aircraft is in the firing envelope; the pilot must place and hold the pipper (mechanically caged) on the target for at least two seconds; maintain no more than the required G's on his plane; depress the trigger switch on the control stick for about 1 second, or until the missile has fired. (There is a short time delay which is explained in this section.) A general rule of thumb for firing the GAR-8 Missile can be approximated by using one mile maximum firing range for each 10,000 feet of altitude. These figures will hold true for target

speeds of .9 to 1.5 Mach number and a closure rate of .3 Mach or more. Below 40,000 feet, pull no more than 2 G's during launch time. Above 40,000 feet, pull no more than 1.6 G's during launch time. The low G factor may sound very limiting but it should be remembered that there is little reason to generate higher turning rates when tracking a target from a distance of 10,000 to 15,000 feet (2 to 3 miles). However, higher G turns may be made prior to the 2-3 seconds required for tracking and firing the missiles. Launching the GAR-8 missile will be a function of range, altitude, missile seeker target signal, and G load at launch point. Once the missile has been fired the pilot may maneuver to avoid contact, press the attack with another missile, or close to gun fire range. Due to the fact that the missile will maneuver itself to the target with no more action from the attacking aircraft, the pilot will be free for further action.

Note

There is no automatic firing sequence from one missile to the other. The pilot must manually select the remaining missile.

Before Take-off.

1. Arming switch—OFF.
2. Weapon selector switch—As Desired.
3. Reticle lights rheostat—ON.
4. Missile firing selector switch—SAFE, then alternate between LH and RH for audio check.

Intercept.

1. Arming switch—ARMT CAMERA.
2. Weapon selector switch—MISSILE.
3. Missile firing selector switch—LH, RH or BOTH.
4. Mechanical cage switch—CAGE.
5. Fly pipper onto target and check G's.
6. Listen for audio signal.
7. Check range.
8. Depress trigger until missiles fire (approximately 1 second).

Note

- Use the same procedure for night attacks, except that the infrared sight should be used in conjunction with the optical sight. (Refer to Infrared Sight System in this Section.)
- Due to a time lag of approximately ¾ second between the ignition of the servo grains in the missile power generator and the firing of the rocket motor, the trigger must be held in the second detent position until the missile has left the launcher.
- Launchers are not considered tip stores; therefore, turn roll damper on with bare launchers installed.

Before Landing.

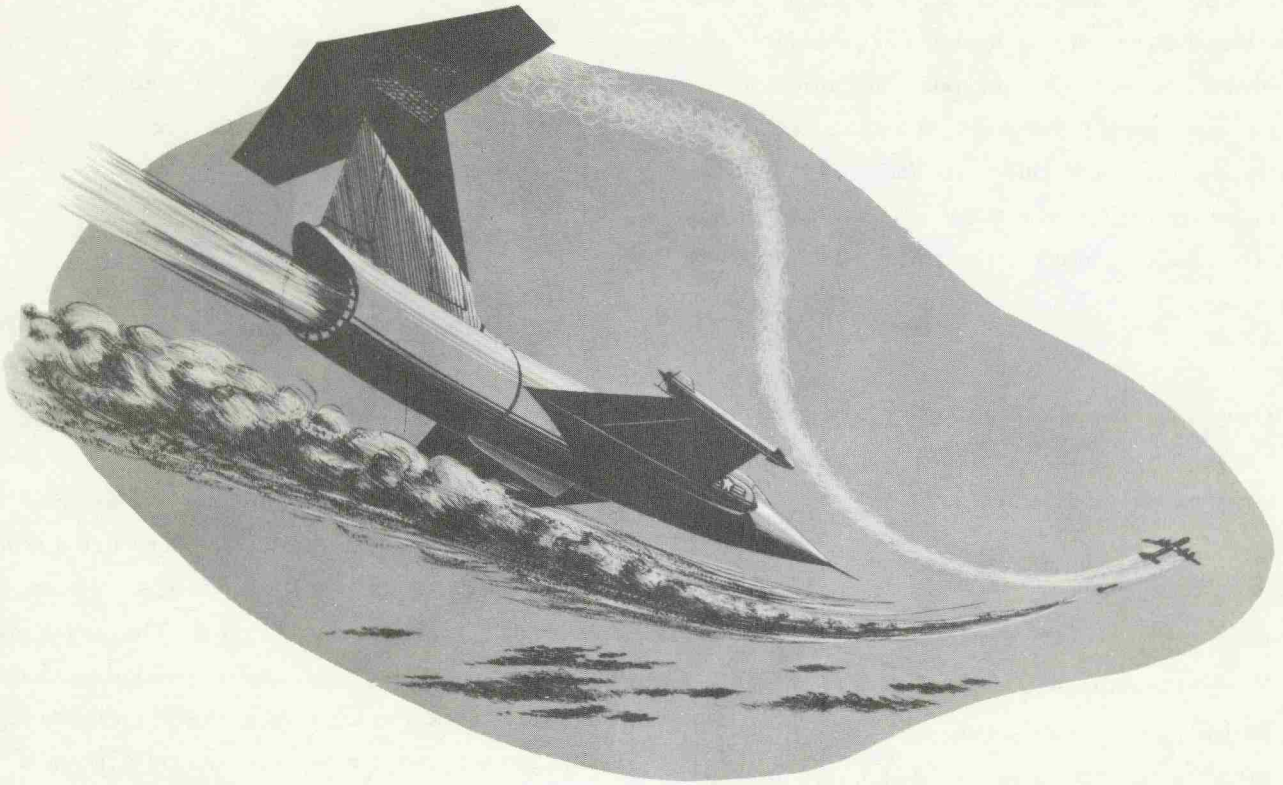
1. Arming switch—OFF.
2. Missile firing selector switch—SAFE.

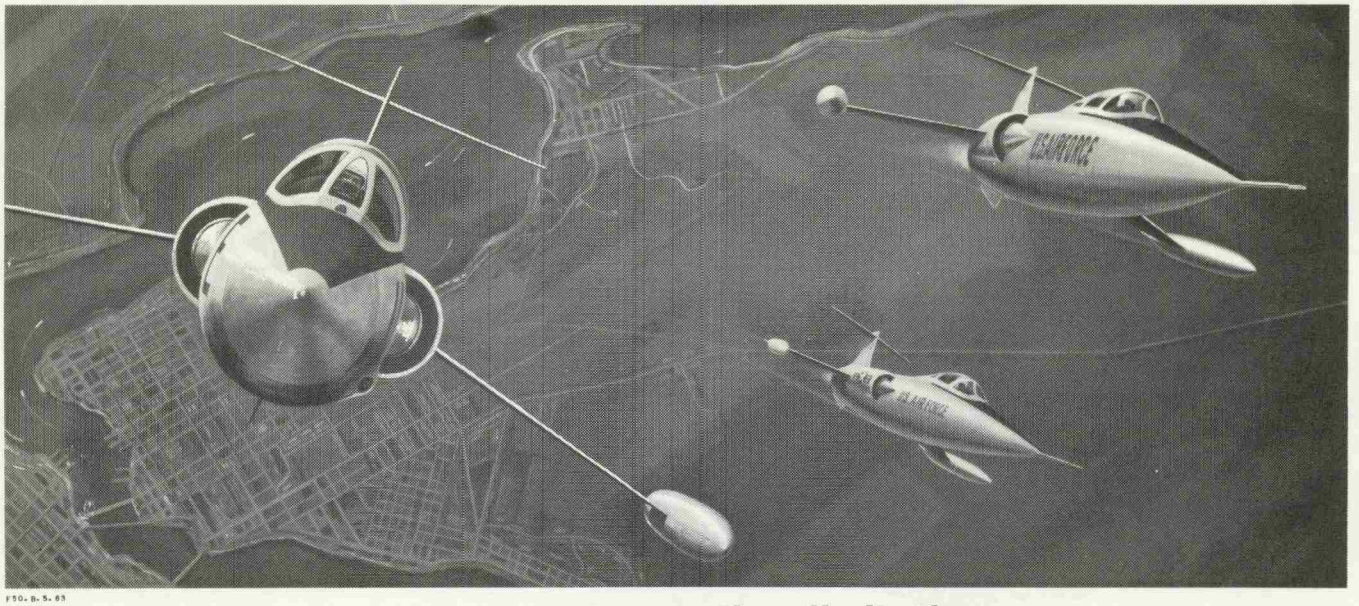
N-9 CAMERA SYSTEM.

A type N-9 16 mm motion picture camera is mounted on the left side of the optical sight head. Through a series of prisms, the camera is able to record simultaneously the optical sight display and the target. The camera may be used to record "hits" or it may be used independently for camera gunnery practice. Adjustments on the camera enable ground crewmen to select camera overrun time (0 to 3 seconds), shutter speed, and lens diaphragm ("f" stop) according to film grade and the type mission planned. The pilot may further adjust the shutter opening while airborne to suit prevailing sky conditions by positioning the camera shutter selector switch.

Camera Shutter Selector Switch.

While in flight, the pilot may readily adjust the camera shutter by positioning the camera shutter selector switch (9, figure 4-11) to the bright (B), hazy (H), or dull (D) positions. Electrical power for the shutter selector and the camera motor is supplied by the d.c. monitored bus.





F50-8-5-63
AM2827

operating limitations

SECTION V

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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. Further explanation of the markings will be covered in the text under appropriate paragraph headings.

ENGINE LIMITATIONS.

See the instrument markings of figure 5-1.

Note

The allowable steady state EGT limit of 585°C is imposed for maximum engine life. However, afterburner operation up to 615°C for a limit of 5 minutes, does not impose a condition requiring overtemperature inspection. Therefore, 615°C EGT can be classified as an acceptable limit during afterburner operation not exceeding 5 minutes. Engine life can be preserved if power settings are reduced to hold EGT at 585°C, whenever it does not adversely affect flying safety.

POWER DEFINITIONS AND TIME LIMITS.

Military Thrust.

Military Thrust is obtained with a full non-afterburning throttle setting and is time-limited as follows:

45 seconds for ground running
30 minutes below 35,000 feet
2 hours above 35,000.

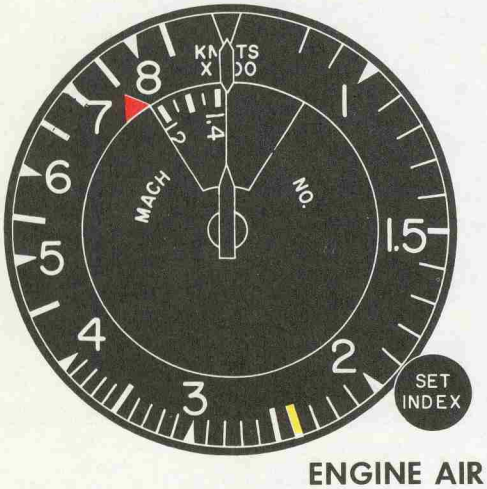
Maximum Thrust.

Maximum Thrust is obtained with a full afterburning throttle setting and has the same time limits as Military Thrust.

INSTRUMENT MARKINGS

BASED ON JP-4 FUEL

AIRSPEED



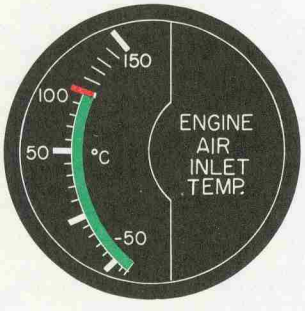
240 KIAS, maximum with flaps in LAND position.

Pointer moves to indicate 750 Knots EAS design maximum. Since the pointer moves according to EAS it will read 750 KIAS at sea level, increasing to 799 Knots at 20,000 feet.

CAUTION

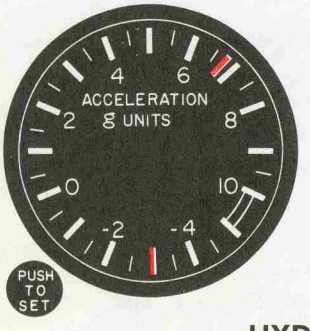
High speed limit applies to the aircraft without external stores. See Airspeed Limitations for limits with external stores

ENGINE AIR INLET TEMPERATURE



100°C Maximum.
-70° to 100°C Operating range.

ACCELEROMETER



SYMMETRICAL MANEUVER LIMITS

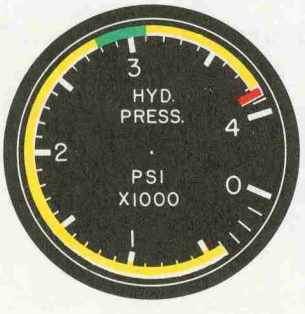
+ 7.33G Maximum with no external stores and 2100 pounds or less internal fuel remaining.

- 3.0G Minimum with 2100 pounds or less internal fuel remaining.

WARNING

Acceleration limits vary with fuel loading and external configuration. Refer to figure 5-4 for complete limits

HYDRAULIC SYSTEMS PRESSURE

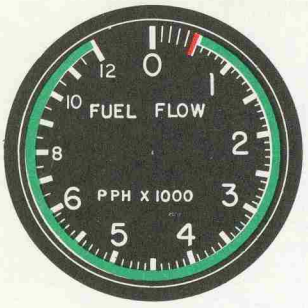


	No. 1 System	No. 2 System
400-2800 PSI	Permissible with high flow demands on system	
		400-2300 PSI Flight Controls have priority over utility system.
	Shows malfunction with no flow demands on system.	
2800-3100 PSI	Normal.	
3100-3850 PSI	Permissible surge during rapid control surface movement.	
	Shows malfunction with control surface static.	
3850 PSI	Maximum.	

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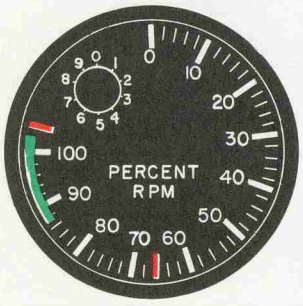
Figure 5-1

FUEL FLOW



500 lb/hr Minimum.
500-12,000 lb/hr Normal operating range.

TACHOMETER



66% Minimum
87-101% Normal operating range
104% Maximum with T_2 reset.
101% Maximum without T_2 reset.
99-101% Normal take-off range with Military Thrust.
107% Maximum overspeed.
105-107% 5 second maximum.
(If exceeded, engine inspection required).

OIL PRESSURE



12 psi Minimum at IDLE thrust.
12-65 psi Continuous operation.
35-65 psi at take-off rpm.
65 psi Maximum (Ground operation)
65-71 psi Permissible range with T_2 reset.
71 psi maximum with full T_2 reset.

EXHAUST GAS TEMPERATURE



100°C Minimum.
100°-585°C Steady state.
100°-545°C Continuous.
545°-585°C 30 minute maximum* below 35,000 feet, 2 hour maximum above 35,000 feet.
585°C Maximum steady state.
585°-1000°C Time limited.
Acceleration or afterburner operation:
615°C, 5 minutes maximum.
635°C, 6 seconds maximum.
675°-735°C, 3 second maximum.
Start:
705°C, 30 second maximum.
725°C, 20 second maximum.
750°C, 10 second maximum.
830°C, 5 second maximum.
1000°C Maximum allowable over-temperature

NOTE
* OPERATION IN EXCESS OF THIS TIME LIMIT WILL REDUCE ENGINE LIFE.

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MAXIMUM AIRSPEED LIMITATIONS

NO EXTERNAL STORES

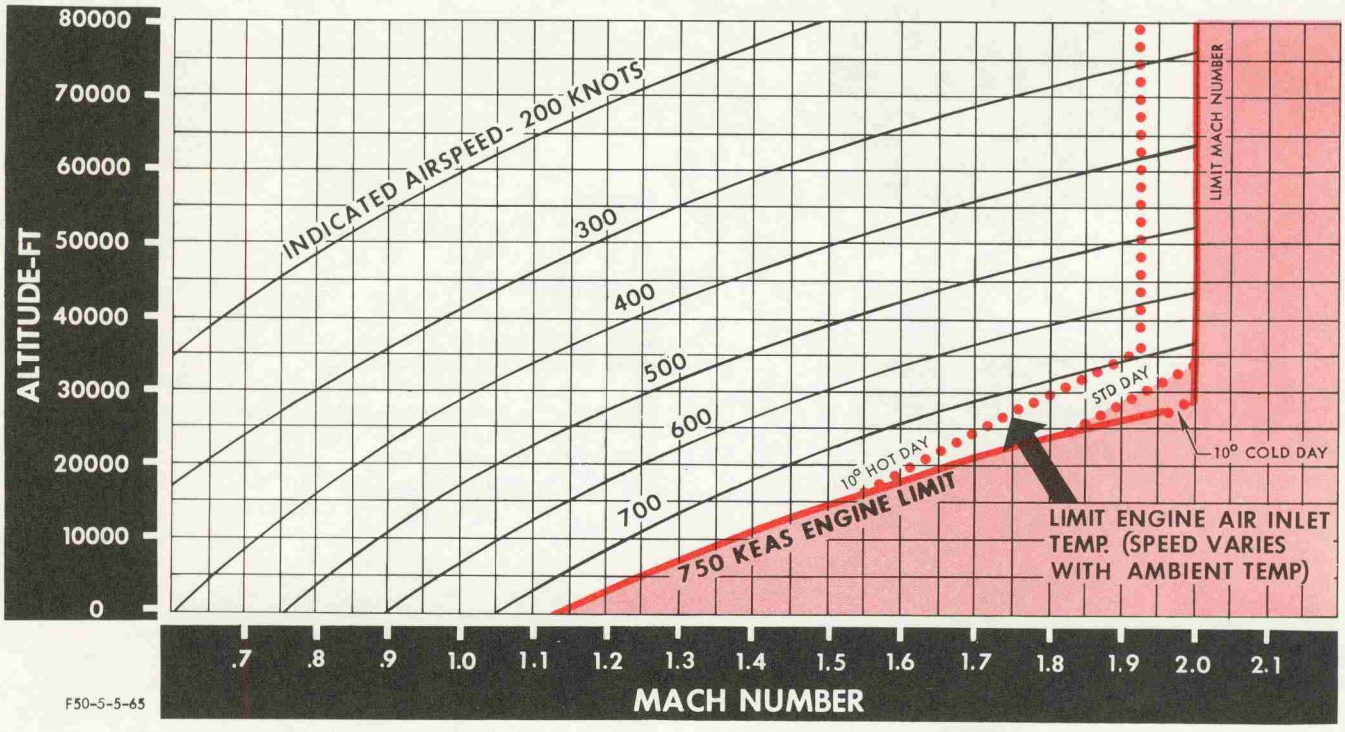


Figure 5-2

Maximum Continuous Thrust.

Maximum continuous thrust (sometimes called normal rated thrust) is obtained at an exhaust gas temperature of 545°C and has no time limit.

ENGINE AIR INLET TEMPERATURE.

The engine air inlet temperature limit is an engine limit which is reached as a function of the outside air temperature and flight Mach number. Therefore, the speed where the limit temperature condition is reached will vary with altitude and from day to day at a given altitude. Note on figure 5-2 that this limit is encountered at approximately Mach 1.92 at high altitude when the ambient temperature is 10°C hotter than standard. Under these conditions, this limit restricts the maximum permissible speed of the airplane rather than limit Mach number.

T_2 RESET LIMITATION.

Reset may occur as low as 70° CIT on the cockpit indicator during full afterburner accelerations. When reset occurs, the rpm shall increase at least 2½% more than the rpm prior to reset. After reset the rpm should be between 102 and 104%.

ROLL STABILITY AUGMENTER LIMITATIONS.

The roll stability augmenter should be turned off before reaching 575 KIAS with wing tip stores installed. With tip stores installed and the roll stability augmenter operating, wing torsional oscillations sufficient to cause structural damage may be experienced at high indicated airspeeds. Missile launchers are not considered as tip stores; therefore, the roll stability augmenter should be left on when carrying bare launchers.

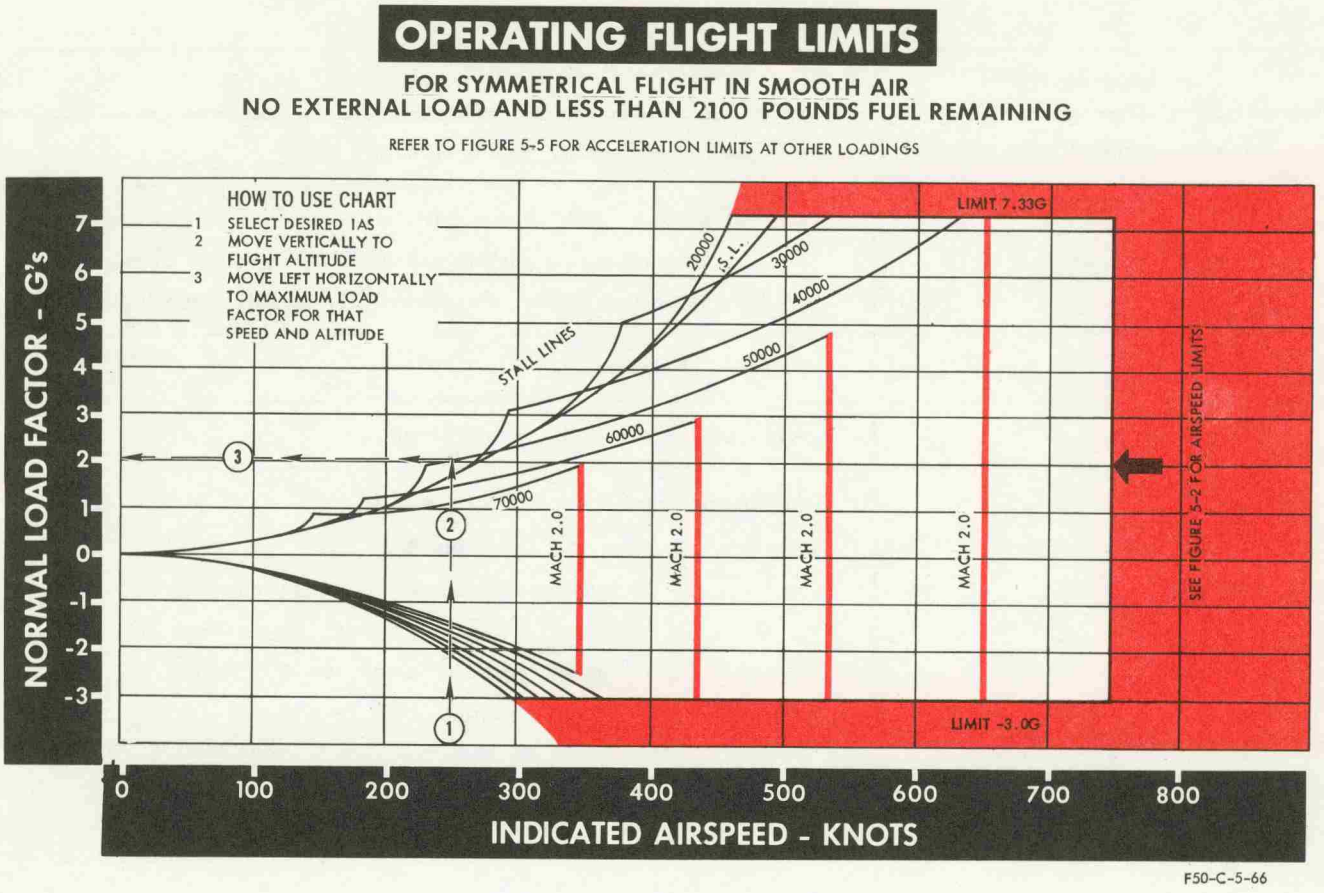


Figure 5-3

AIRSPEED LIMITATIONS.

Canopy, Part-Open position ..... 260 KIAS

Landing gear transient operation (normal or emergency system) ..... 260 KIAS

Windshield rain remover operation ..... 295 KIAS

Landing gear down and locked ..... 295 KIAS

Wing flaps

Landing setting ..... 240 KIAS

Drag chute operation ..... 200 KIAS

Full Speed Brakes ..... Mach 1.6

CAUTION

On unmodified aircraft, only the locked or part-open position of the canopy should be used when the aircraft is in motion. In-flight, the part-open position should be used only in an emergency and at speeds below its limiting airspeed.

MAXIMUM ALLOWABLE AIRSPEED.

The maximum allowable airspeed for any external stores configuration is presented in figure 5-5. In addition, the limits for the no external stores configuration are shown pictorially in figure 5-2.

EXTERNAL STORES JETTISON LIMITS.

Refer to figure 5-4.

PROHIBITED MANEUVERS.

PITCH-UP AND SPINS.

Intentional pitch-up and spins are prohibited because of the high loads that are imposed on the aircraft. These

EXTERNAL STORES JETTISON LIMITS	
EXTERNAL STORES	LIMIT SPEED
GAR-8 Missiles & Launchers	Any speed below Mach 1.4 in 1G flight.
Missile Launchers	Do not jettison. Wind tunnel tests show unsafe jettison characteristics. Landing with a misfired missile is safe because all danger of firing is passed within 10 seconds after an attempted firing.
Pylon Tanks	Any speed below Mach 1.5 in 1G flight.
Pylons	Any speed below Mach 0.9 in 1G flight.
Tip Tanks	Any speed below Mach 0.9 in 1G flight. **Note** In an emergency, empty tip tanks may be jettisoned at supersonic speeds. If possible, jettisoning should be accomplished at or below Mach 1.5 in 1G flight.

Figure 5-4

loads can be of sufficient magnitude to cause structural damage to the aircraft.

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.

INVERTED FLIGHT.

Prolonged inverted flying or any maneuvers resulting in prolonged negative load factor will result in engine flame-out as a result of fuel starvation. Do not fly inverted for longer than 20 seconds.

AILERON ROLL LIMITATIONS.

In order to avoid high structural loads approaching limit values, aileron rolls are restricted as follows:

Wing Flaps Retracted.

Entry Load Factor of 0.5G and Above. Full deflection rolls are limited to 360 degrees.

Entry Load Factor Below 0.5G. Full deflection rolls are limited to 180° and must have either the pitch or the yaw stability augmenter operative. Recovery must be initiated at approximately 90 degrees. With both pitch and yaw stability augmenters inoperative, aileron rolls are prohibited.

Entry Load Factor Below Zero G. All rolls below zero G entry load factor must be executed with extreme caution and should be avoided whenever possible.

Take-off (Maneuvering) Flaps Extended.

Limitations are as follows:

1. Rolls are limited to 360°. High roll rates can develop from moderate aileron application.

MAXIMUM ALLOWABLE AIRSPEED AND ACCELERATION LIMITS

SYMMETRICAL MANEUVERS

AIRPLANE CONFIGURATION		OPERATING CONDITIONS				
		AIRSPEED LIMITS	ACCELERATION LIMITS			
			Using Pylon or Tip Fuel	5830 Lbs. (Using Aux. Tank)	4900 Lbs. Internal Fuel Remaining	2100 Lbs. Internal Fuel Remaining
No External Stores		750 Knots, E.A.S., Mach 2.0, or Engine Air Inlet Temperature Limit, whichever occurs first	—	4.6 −3.0	5.4 −3.0	7.33 −3.0
GAR-8 Missiles			—	4.5 −2.8	5.2 −3.0	6.7 −3.0
Missile Launchers			—	4.6 −3.0	5.4 −3.0	7.1 −3.0
Tip Tanks		Above 25,000 feet—750 KIAS, Mach. 1.9 or Engine Air Inlet Temperature Limit, whichever occurs first Below 25,000 feet—575 KIAS.	4.1 −2.6	4.2 −2.6	4.8 −2.8	5.9 −3.0
Pylon Tanks		• Pylon tanks with .025 inch skin thickness on horizontal vanes—575 KIAS or Mach 1.5, whichever occurs first • Pylon tanks with .032 inch skin thickness on horizontal vanes—750 knots, E.A.S. or Mach 1.5, whichever occurs first	4.4 −2.7	4.6 −3.0	5.3 −3.0	6.9 −3.0
GAR-8 Missiles & Pylon Tanks			4.2 −2.5	4.5 −2.8	5.0 −3.0	6.4 −3.0
Missile Launchers & Pylon Tanks			4.4 −2.8	4.6 −3.0	5.2 −3.0	6.8 −3.0
Tip Tanks & Pylon Tanks		• Above 25,000 feet—Mach 1.5 • Below 25,000 feet—575 knots I.A.S. • Do not exceed 500 knots I.A.S. with empty tip tanks and more than residual fuel in pylon tanks	3.8 −2.6	4.1 −2.6	4.6 −2.8	5.7 −3.0
Take-off Flaps	Modified Airplane During Extension	370 knots I.A.S. or indicated Mach 0.84. There is no Mach limitation if 290 KIAS is not exceeded	• For positive acceleration limits see applicable external configuration above • The minimum negative g limit is −1.0 g.			
	Modified Airplanes Extended or Retracting	500 knots I.A.S. or indicated Mach 0.84. There is no Mach limitation if 350 KIAS is not exceeded				
	Unmodified Airplanes	295 KIAS				

UNSYMMETRICAL MANEUVERS

Rolling Pullouts	No External Stores	Mach 1.9 to 2.0	—	3.3	3.5	4.0
		Below Mach 1.9	⅔ of the symmetrical limits shown above			
	Any External Store Installed	Any permissible speed shown for symmetrical maneuvers				
	Take-off Flaps	See Aileron Roll Limitations				
Rolling Pushovers	No External Stores	Any permissible speed shown for symmetrical maneuvers	See Aileron Roll Limitations			
	Any External Store Installed					
	Take-off Flaps	See Aileron Roll Limitations				

Figure 5-5

2. Either the pitch or yaw stability augmenter must be operative for all rolling maneuvers.

3. Aileron rolls are prohibited for entry load factors less than 1-G.

4. Do not conduct 1-G aileron rolls at speeds exceeding 450 KIAS; however, rolling pullouts may be conducted at speeds above 450 KIAS if entry load factor is 2.0 G's or more.

Note

Application of some back stick pressure during rolls entered at or below 1G is beneficial to provide a more rapid termination of the rolling maneuver and a smoother transition to normal flight.

ACCELERATION LIMITATIONS.

One of the primary factors determining the acceleration limitations on an aircraft is center of gravity position. In this aircraft all of the internal fuel is located in the fuselage resulting in a significant movement of the center of gravity position from forward to aft as fuel is used. As external fuel tanks are located close to the center of gravity of the aircraft, there is minor movement of the center of gravity with expenditure of their fuel. This movement of the c.g. with internal fuel expenditure results in a fairly wide variation of permissible load factor. This information is too complex to permit instrument markings which would cover all operating conditions; therefore, a tabular listing is included in figure 5-5 for this purpose. Note that as lower fuel loadings are reached an increase in maximum permissible load factor is realized.

CAUTION

The instrument markings of figure 5-1 are for the combat condition only as this is the most likely range where maximum load factors will be utilized. Cognizance must be taken of the chart limits, therefore, when operating at higher fuel loadings.

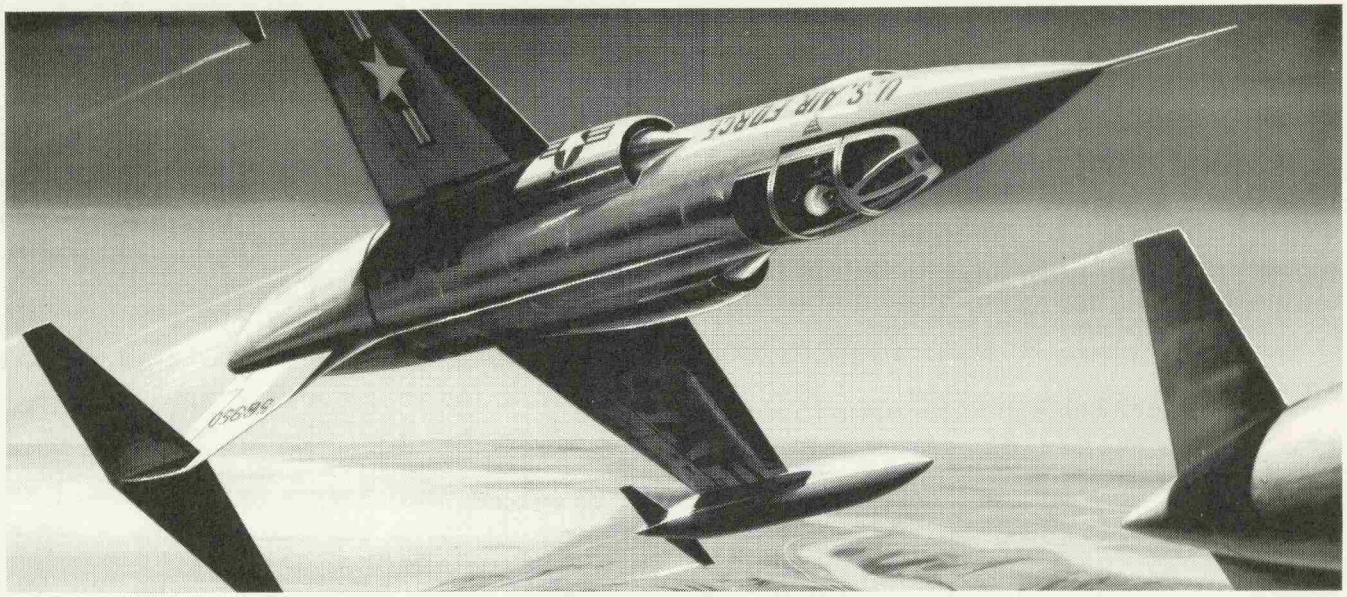
The G-limits shown in the upper portion of the table are for symmetrical maneuvering such as straight pullups or steady turns. In rolling pullouts or rolling pushovers, lower G-limits must be observed because of the higher structural loads that are imposed on the aircraft. These limits are shown in the lower portion of figure 5-5.

MANEUVERING BOUNDARIES.

An operating flight limits diagram is shown in figure 5-3. This diagram shows the maximum maneuvering load factors over the operating range of speeds and altitudes. It includes the load factor and airspeed combination at which the stall occurs (see Stalls, in Section VI) as well as the maximum allowable structural g-limits for the clean configuration. Use of the diagram is illustrated. Since the maximum allowable airspeed depends on different limitation at various altitude ranges, 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, reference should be made to the airspeed limitations of figure 5-2, in conjunction with this diagram for complete limits. Use of figure 5-2 will also permit the determination of the corresponding Mach number for any airspeed and altitude combination if the maximum load factor for a particular Mach is desired.

WEIGHT LIMITATIONS.

There are no weight limitations.



flight characteristics

SECTION VI

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INTRODUCTION.

The operational speed and altitude capabilities of this aircraft are considerably greater than previous fighter aircraft. These capabilities include level flight and climb speeds of Mach 2.0 and altitudes in excess of 90,000 feet. Flight characteristics and handling qualities of the aircraft are excellent over the entire operating range.

MACH NUMBER.

Except for possibly the low speed stall, 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 results 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, therefore, a given flight characteristic will generally be more pronounced. For these reasons, reference to flight speed will generally be made in terms of Mach number rather than airspeed. The airspeed for a desired Mach number at any altitude may be determined by reference to figure 5-2. Enter the bottom of the chart with Mach number, move vertically to the flight altitude using the scale on the left and read IAS from the curved airspeed lines. Interpolation is necessary between the airspeed lines.

AIRSPEED AND ALTITUDE ERRORS.

The Mach number used in this section is indicated Mach number. At subsonic speeds this Mach number differs from true Mach number because of the error in the static pressure sensed by the airspeed nose boom. This error increases as Mach 1.0 is approached and abruptly reduces to zero when the fuselage bow shock wave moves downstream of the static pressure pickup. Once this shock wave moves past the static pressure pickup the airspeed senses true static pressure. Thus, at supersonic speeds, the indicated Mach is true Mach. In an acceleration from subsonic to supersonic speed, the Mach number will appear to hesitate at about Mach .96 due to this rapidly rising error and then increase abruptly to about 1.04. At the same time, the indicated altitude will increase approximately 2,000 feet depending on flight altitude and rate of change of pressure. In normal operation of the aircraft, indicated Mach number will suffice; however, for determination of true airspeed on cross country flight or for maintaining a desired true pressure altitude this error must be considered. Correction curves for this purpose are provided in Part I of the Appendix.

AIRCRAFT CONFIGURATION.

The over-all configuration of the aircraft was chosen with an emphasis upon high Mach number flight while maintaining conventional landing and take-off characteristics. Since the appearance of this aircraft is somewhat unique, 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 item arousing the most interest on the aircraft, is the large 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; therefore, the empennage arrangement will be discussed first and then the negative dihedral.

EMPENNAGE.

The vertical tail of the aircraft, aside from its very high effectiveness, is conventional in most respects. The unconventional empennage item is the location of the horizontal tail on top of the fin. This location of the horizontal tail was determined from extensive wind tunnel tests of the tail located in many positions from below the fuselage to its present 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. Although a low horizontal tail location gave a stable stall, it resulted in undesirable flight characteristics at angles of attack below the stall particularly through the transonic speed range. In addition, a low tail position configuration had substantially higher supersonic drag and thus would have lowered the supersonic performance capability. In addition to superior performance capability, the high tail configuration resulted in an absolute minimum of transonic trim changes and provides high stabilizer effectiveness throughout the speed range.

DIHEDRAL EFFECT.

The position of the stabilizer atop the fin makes it act as an end plate to the vertical fin. The effective aspect ratio of the 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 tail 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 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 stabilizer location, provides the equivalent dihedral effect of 15 to 20 degrees 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 degrees of dihedral. It is to be emphasized that despite the negative wing dihedral, the airplane possesses a normal positive dihedral effect, as you 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 full span leading edge flap is deflected for landing and take-off in order to delay flow separation over the sharp leading edge at the higher angles of attack. Inboard trailing edge flaps incorporating boundary layer control and conventional outboard ailerons are provided. The boundary layer control system blows high energy air

over the trailing edge flap in the LAND position, thereby delaying flow 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.

On modified aircraft maneuvering capabilities of the aircraft may be improved by the use of maneuvering flaps. Maneuvering flaps are attained by lowering the wing flaps to take-off position with the gear retracted. This permits an increase of approximately 1G in the available load factor over that of the aircraft with flaps up, thereby improving the turn radius of the aircraft.

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 shocks decelerate the air flow 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.

STALLS.

The airflow characteristics associated with the high fineness ratio fuselage and the sharp leading edge, low aspect ratio wing at high angles of attack, combined with the high horizontal tail position, result in a pronounced pitch-up characteristic in the fully stalled condition. Beyond a certain point, this pitch-up is uncontrollable and will result in severe gyrations 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 at the large angles of pitch and yaw that are encountered in such a maneuver. In addition to this type of characteristics at the stall, it is possible to develop stalling angles of attack very readily and rapidly in abrupt maneuvering such as quick pull-ups, even though relatively small amounts of stabilizer are used. This is the result of the combination of high stabilizer effectiveness and the high inertia in pitch of the modern supersonic aircraft. The usual stall warnings are inadequate under these conditions to preclude the attainment of an excessive angle of attack. Because of these characteristics, this aircraft incorporates an auto-pitch control system which provides adequate warning by initiating corrective action at the proper time to prevent the attainment of a high enough angle of attack to encounter pitch-up under any operating condition.

AUTO-PITCH (KICKER) CONTROL SYSTEM.

This device senses the approach to the full stall and abruptly moves the stick forward to slightly ahead of neutral. This action provides pilot warning and automatically applies airplane nose down stabilizer to initiate pilot follow-through. It is in effect a built-in artificial stall that occurs just ahead of the aerodynamic stall. In addition to sensing angle of attack, this device senses pitch rate which provides anticipation in high pitch rate maneuvers such as quick pull-ups. Under these conditions, the "stick kick" occurs earlier in terms of load factor or angle of attack than it would in steady turns. This earlier "kick" is necessary, as the inertia of the aircraft will cause the angle of attack to continue to increase after the "kick." The earlier "kick" allows for this angle of attack overshoot and effectively limits the peak angle of attack. As long as you attempt to operate the aircraft beyond the safe maneuvering range, the auto-pitch control system will continue to actuate and continue to force the stick forward. The auto-pitch control only 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 insures adequate nose down corrective action. If the aircraft is trimmed close to a high angle of attack condition, additional pilot follow-through and corrective action is necessary. The kicker is inoperative any time the flaps are in either the TAKE-OFF or LAND position on unmodified aircraft. On modified aircraft, the kicker is inoperative whenever the gear is down or landing flaps are extended. Thus, on modified aircraft, the kick feature is available for high-speed maneuvering with take-off flaps but is inoperative when the gear is down to prevent undesirable kicker operation during take-off and landing. A control stick shaker (stall warning system) is incorporated in the flight control system. Operation of the shaker indicates

MANEUVERING BOUNDARIES OF AUTO-PITCH CONTROL SYSTEM

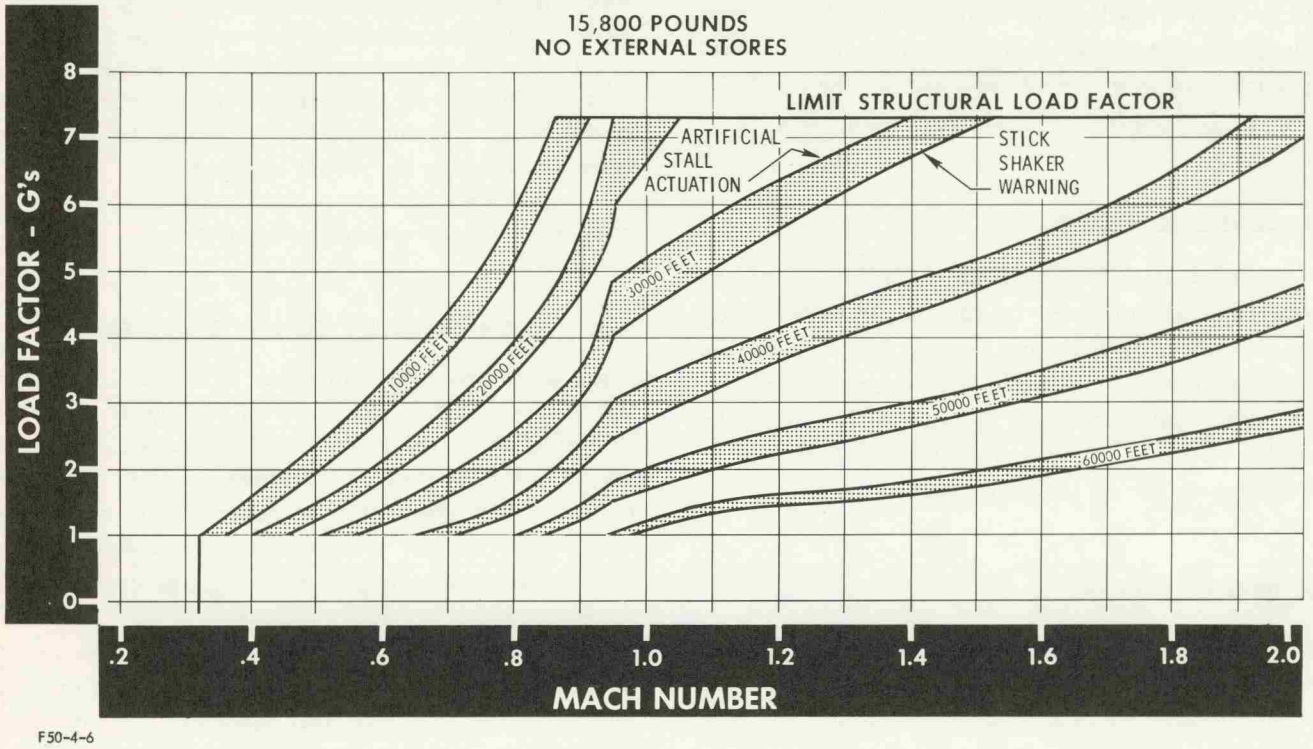


Figure 6-1

an approaching stall. This device is set to operate just prior to the kicker operating point which permits maximum utilization of the aircraft's available maneuvering envelope without incurring the kicker operation. Figure 6-1 shows the maneuvering boundary of the kicker and stick shaker in terms of Mach number and figure 5-3 presents the kicker boundary in terms of indicated airspeed.

UNACCELERATED STALLS.

Since auto-pitch control actuation is, in effect, a built-in stall, the word "stall" is used in the following discussions to define the point of auto-pitch control operation. The low speed "stall" is preceded by a rather wide speed band of heavy airframe buffet. This buffeting builds up from mild to heavy and then remains heavy with further reduction in airspeed. Speeds below the onset of heavy buffet do not represent a useful operating range of the aircraft. With further reduction in speed, the aircraft becomes laterally unstable. This lateral instability may get stronger 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

Stall approaches in aircraft configurations where the kicker is inoperative should be terminated at stick shaker action, lateral instability, or wing drop. In addition, the stall warnings in abrupt high-pitch rate maneuvers are inadequate; therefore, this type of maneuvering must also be avoided. The kicker is inoperative with the wing flaps extended to either the take-off or land position on unmodified aircraft. On modified aircraft, the kicker is inoperative whenever the gear is down or land flaps are extended.

ACCELERATED STALLS.

In the subsonic region, stall characteristics are similar to those described under unaccelerated stalls in that a

MINIMUM OPERATING SPEEDS—KNOTS

AND MINIMUM CONTROL SPEEDS (VALUES IN PARENTHESIS)

ALTITUDE SEA LEVEL TO 10000 FEET		GEAR AND FLAPS UP POWER ON OR OFF			TAKEOFF FLAPS GEAR UP OR DOWN POWER ON OR OFF			GEAR AND LANDING FLAPS DOWN POWER ON		
EXTERNAL CONFIGURATION	BANK ANGLE	0°	40°	60°	0°	40°	60°	0°	40°	60°
	LOAD FACTOR	1.0 G	1.3 G	2.0 G	1.0 G	1.3 G	2.0 G	1.0 G	1.3 G	2.0 G
NO EXTERNAL STORES OR PYLON TANKS	21500 POUNDS	235 (210)	270 (240)	330 (295)	215 (195)	245 (225)	305 (275)	175 (165)	200 (185)	245 (230)
	18500 POUNDS	220 (195)	250 (220)	310 (275)	200 (180)	230 (205)	285 (255)	160 (150)	185 (175)	230 (215)
	14000 POUNDS	190 (170)	215 (195)	270 (240)	175 (160)	200 (180)	245 (225)	140 (130)	160 (150)	200 (185)
MISSILES OR MISSILES PLUS PYLON TANKS	21500 POUNDS	225 (205)	260 (235)	320 (290)	210 (190)	240 (220)	285 (270)	170 (160)	195 (180)	240 (225)
	18500 POUNDS	210 (190)	240 (215)	300 (270)	195 (180)	220 (205)	275 (250)	155 (150)	180 (170)	220 (210)
	14000 POUNDS	185 (165)	210 (185)	260 (235)	170 (155)	190 (175)	240 (220)	135 (130)	155 (150)	195 (185)
TIP TANKS OR TIP TANKS PLUS PYLON TANKS	24500 POUNDS	235 (210)	265 (240)	330 (300)	215 (200)	245 (225)	305 (280)	175 (170)	200 (190)	250 (240)
	21500 POUNDS	220 (200)	250 (225)	310 (280)	200 (185)	230 (215)	285 (265)	165 (160)	190 (180)	235 (225)
	18500 POUNDS	205 (185)	230 (210)	290 (260)	185 (175)	215 (200)	265 (245)	155 (150)	175 (165)	220 (205)
	14000 POUNDS	175 (160)	200 (185)	250 (225)	165 (150)	185 (170)	230 (215)	135 (125)	150 (145)	190 (180)

NOTE:

1. MINIMUM OPERATING SPEEDS ARE THE SPEEDS AT WHICH AUTOMATIC STICK SHAKER ACTION IS EXPERIENCED.
2. MINIMUM CONTROL SPEEDS ARE THE SPEEDS AT WHICH:
   a) KICKER IS EXPERIENCED - FLAPS UP & MANEUVERING FLAPS
   b) LATERAL INSTABILITY IS EXPERIENCED - TAKE-OFF FLAPS AND GEAR DOWN
   c) WING DROP IS EXPERIENCED - LANDING FLAPS
3. FULL STALL WILL BE ENCOUNTERED IF THERE IS FURTHER REDUCTION OF SPEED
4. SPEEDS IN EXCESS OF LIMITS SHOWN FOR INTERPOLATION PURPOSES ONLY

F50-C-6-74

Figure 6-2

band of natural airframe buffet and lateral instability warning precedes the "stall." In the transonic region, the speed or "G" band of natural buffet and lateral instability warning gradually reduces and is indicated to be non-existent above approximately Mach .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 to stick shaker warning or the minimum control speeds of figure 6-2 may be executed at any reasonable altitude; however, it is recommended that 25,000 feet 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 14,000 pounds (normal landing weight with 1,000 pounds of fuel remaining) are described in the following paragraphs. The addition of wing tip stores will lower the typical speeds. See figure 6-2 for magnitude. Increase speeds approximately 5 knots for each additional 1,000 pounds of fuel remaining.

Gear and Flaps Up.

In the clean configuration, the aircraft starts buffeting at about 215 knots, this buffeting becomes heavy at 205 knots. The stick shaker operates below about 190 knots. Lateral instability will be experienced at approximately 180 knots increasing in intensity to the kicker operation at 170 knots.

Take-Off Flaps—Gear Up or Down.

With take-off flaps extended, the stick shaker action is the initial stall warning at 175 knots. Moderate airframe buffet is experienced at 162 knots with lateral instability at 158 knots, the minimum control speed. On modified aircraft, with gear up, the kicker will actuate at 158 knots.

Gear Down and Landing Flaps.

In this configuration and with sufficient engine rpm for boundary layer control operation, there is no airframe buffet or significant lateral instability. At 25,000 feet with 100% rpm, the stick shaker action is felt at 150 knots with some wing drop and a general lowering of over-all stability occurring at the minimum control speed of 140 knots. Under landing conditions, these characteristics occur approximately 10 knots lower due to the increased boundary layer control effectiveness at the lower altitude.

WARNING

Stall approaches in aircraft configurations where the kicker is inoperative should be terminated at stick shaker action, lateral instability, or wing drop. In addition, the stall warnings in abrupt high-pitch rate maneuvers are inadequate; therefore, this type of maneuvering must also be avoided. The kicker is inoperative with the wing flaps extended to either the take-off or land position on unmodified aircraft. On modified aircraft, the kicker is inoperative whenever the gear is down or land flaps are extended.

Ground Effect.

Due to the ground effect during take-off 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 be experienced. In addition, the high pitch angles required for flight at these low speeds will be excessive and can result in tail dragging.

PITCH-UP.

SUBSONIC PITCH-UP FROM 1-G FLIGHT.

Before it is possible to encounter pitch-up, the pilot must ignore the heavy buffet and stick shaker action which warns of an impending stall. He must also overpower the automatic pitch control system and ignore the lateral instability which increases in magnitude as pitch-up is approached. If a 1-G pitch-up is encountered at subsonic speed, recovery can be accomplished if the control stick is pushed forward as soon as pitch-up is detected. Pitch-up cannot be checked if it is allowed to go beyond the initial stages. The oscillations in roll and yaw will increase in intensity as the aircraft pitches to an extreme nose-up attitude of about 60 degrees. If the internal fuel loading is greater than 2200 pounds, the aircraft has only slight tendencies to spin, and pitch-up recovery is best effected by neutralizing the controls and waiting for the oscillations to stop. As much as 15,000 feet may be lost before recovery is complete. If pitch-up has been allowed to develop fully with less than this fuel loading, (which results in a more aft C.G. location) the aircraft will probably enter a spin before pitch-up recovery can be effected, regardless of control positions.

SUBSONIC PITCH-UP FROM ACCELERATED FLIGHT.

The same stall warnings are present when entered from accelerated flight; however, the abruptness of pitch-up is so rapid that it may be impossible to prevent. In this case, controls should be neutralized and held until the ensuing oscillations subside or a spin develops. Recovery is initiated when the aircraft is in or near a vertical nose-down attitude. If a spin develops, the spin recovery procedure should be used.

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. Stall warnings at supersonic speeds are provided by stick shaker action and the artificial stall response of the automatic pitch control. Natural stall warnings such as buffeting or lateral instability may be small or nonexistent.

SPINS.

NORMAL SPINS.

With more than 2200 pounds of fuel, the aircraft exhibits only slight tendencies to spin. With less than this loading, the aircraft will be likely to spin following pitch-up, regardless of stick and rudder positions. If the aircraft spins, it will, in all probability, be to the right since large gyroscopic forces caused by the engine during pitch-up make spins to the left improbable. The pilot may become disoriented during spin entry because of the extreme oscillations in pitch, roll, and yaw. If there is any doubt as to the direction of rotation, a recovery should be made for a spin to the right. Altitude loss during a spin recovery will be approximately 2,000 feet per revolution, with each revolution requiring about 5 seconds.

RECOVERY.

Note

Be sure aircraft is "clean" before starting spin recovery.

Spin recovery is effected by applying aileron into the direction of the spin, rudder in the opposite direction, and the stick full forward. Controls should be held in this position until rotation stops and the aircraft is in or near a vertical nose-down attitude. This may require two or more complete revolutions with an increase of pitch and roll oscillations just prior to spin recovery. Altitude required for complete recovery once rotation is stopped can be as much as 10,000 to 12,000 feet. Pull-out should not be rushed and speed should be allowed to increase high enough to prevent a secondary spin entry during pull-out. If rotation has not stopped upon reaching 15,000 feet, it is recommended that the aircraft be abandoned.

INVERTED SPINS.

Inverted spins has not been investigated, however, standard recovery procedure for inverted spins is to neutralize all controls.

VERTICAL STALL RECOVERY.

In the event extremely steep or vertical climb paths are maintained beyond 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 arc 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" type manuever will probably be experienced at the apex of the flight path as the aircraft reverses direction of flight and heads back toward the ground. Altitude required for complete recovery to level flight can be as much as 15,000 to 20,000 feet. In all probability, a pitch-up will not be encountered; however, the possibility should not be precluded. In such cases, follow the proper recovery procedure.

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, caution should be used 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. The design of the stabilizer control trim system results in a reduction of available aircraft nose down stabilizer with large amounts of aircraft nose up trim; therefore, to insure adequate nose down pitch control avoid unnecessary trimming in practice stall approaches.

AILERONS.

The ailerons are also fully powered and have stick forces supplied by feel springs. This results in essentially constant stick forces for a given amount of deflection regardless of airspeed. The ailerons which are capable of developing extremely high roll rates have decreased travel when the wing flap lever is in the UP position to reduce their effectiveness and thus avoid inertial coupling tendencies in rolls over the wide operating limits of the aircraft.

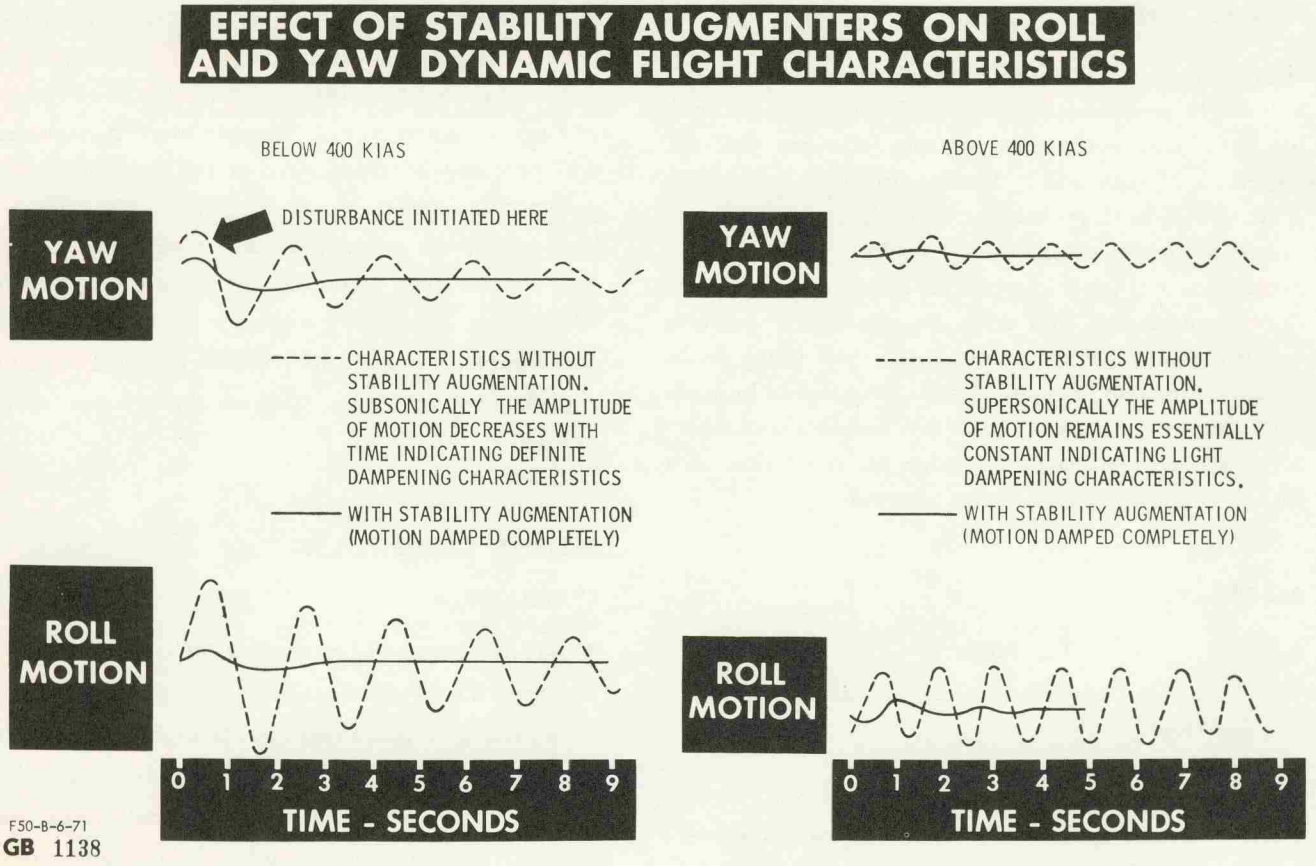


Figure 6-3

RUDDER.

Operation and effectiveness of the rudder are conventional at low speeds except for the relatively high breakout force of the rudder lock at neutral. The rudder is effective above approximately 70 knots. Above flaps up speed there is no need for rudder control as the aircraft is coordinated in normal maneuvering without its use. Use of the rudder at high Mach numbers is not recommended. Because the rudder system is unboosted, very little control is available if operated at high indicated airspeeds and its use merely untrips the rudder lock and results in aerodynamic interaction from the yaw damper.

STABILITY AUGMENTERS.

The aircraft's dynamic response and handling characteristics are greatly improved through the use of stability augmenters about all three aircraft axes. 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 gun 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 with tip stores installed (see Section V). Figures 6-3 and 6-4 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 authority. With trim lost the aircraft will be yawed slightly and yaw damping will be noticeably reduced. At speeds below 400 knots, a low level of positive damping will be present and oscillation characteristics will be the same as illustrated in figure 6-3 for the yaw damper OFF case for speeds below 400 knots. Above 400 knots, damping of the oscillation will occur for disturbances in excess of ± 1.0 degree sideslip; however, a residual oscillation of ± 1.0 degree sideslip or less may be encountered. Although the oscillations will be uncomfortable, they present no hazard to safe flight. These high-speed oscillatory characteristics will be the same as illustrated for the yaw damper OFF case for speeds above 400 knots in figure 6-3. The damping of the oscillation is improved

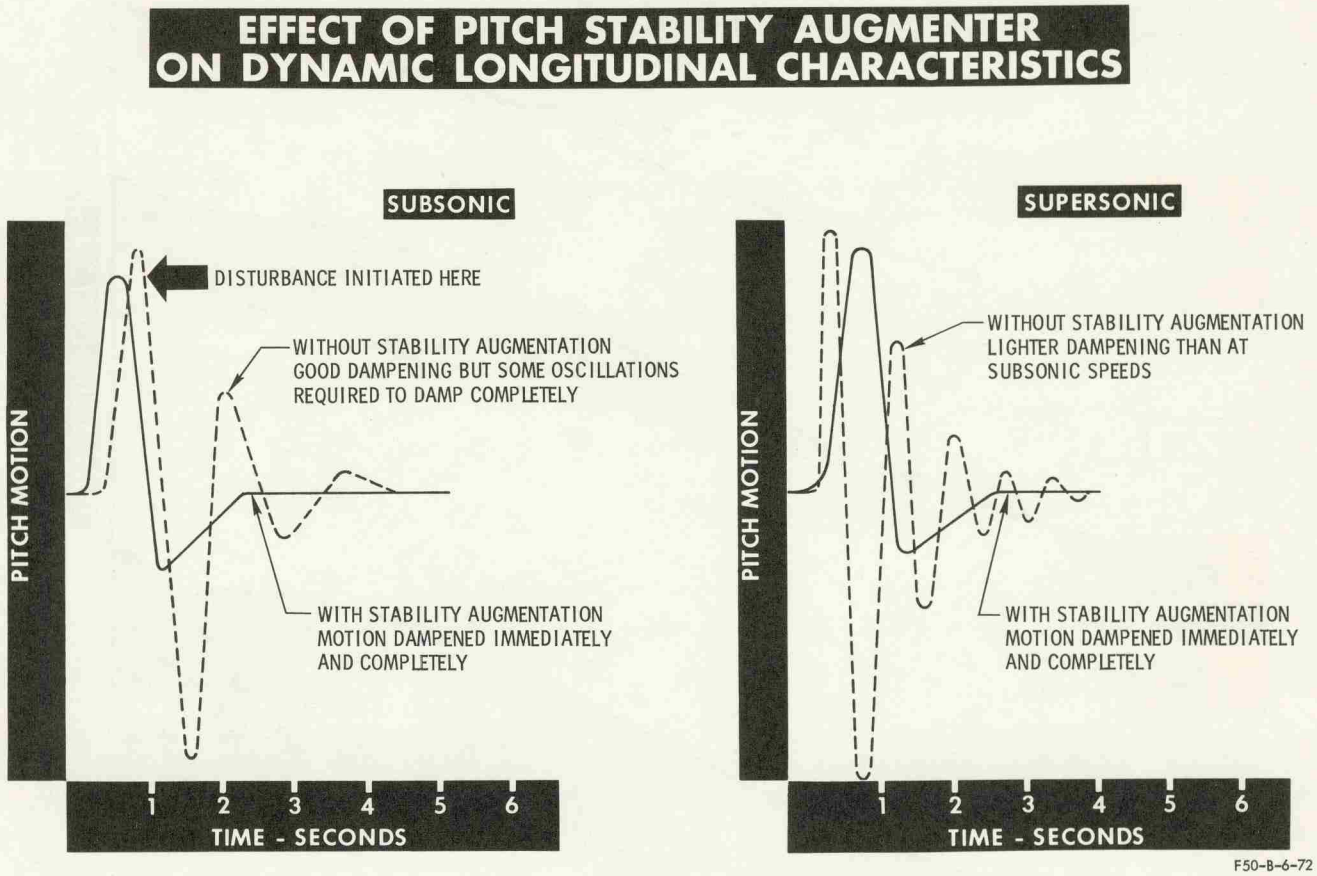


Figure 6-4

in accelerated flight resulting in a reduction in oscillation amplitude, even though the initial pilot impression is the opposite because of the increase in oscillation frequency that also occurs. No attempt should be made to dampen oscillations with the flight controls as the frequency of the oscillation is too high for adequate pilot response and oscillation will become worse. Successful GAR-8 missile firings may be accomplished even though the yaw augmenter has failed because the oscillation amplitudes are within the cone of the IR target seeker. Failure of the roll augmenter system will not reduce the aircraft's effectiveness, but the aircraft will be more sensitive to roll disturbances in turbulent air. Flight may be continued and, in smooth air, little or no effect will 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 will be impaired.

SPEED BRAKES.

Speed brake effectiveness is proportional to airspeed. At high speeds (within the limitations specified in Section V) they produce tremendous drag without objectionable buffeting and only a mild nose-up trim change. Combined with engine power reduction they provide exceptionally rapid deceleration, particularly at the lower altitudes.

PERFORMANCE CAPABILITIES.

THRUST AND DRAG.

The relatively low drag and the high thrust to weight ratio of this aircraft results 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 capabil-

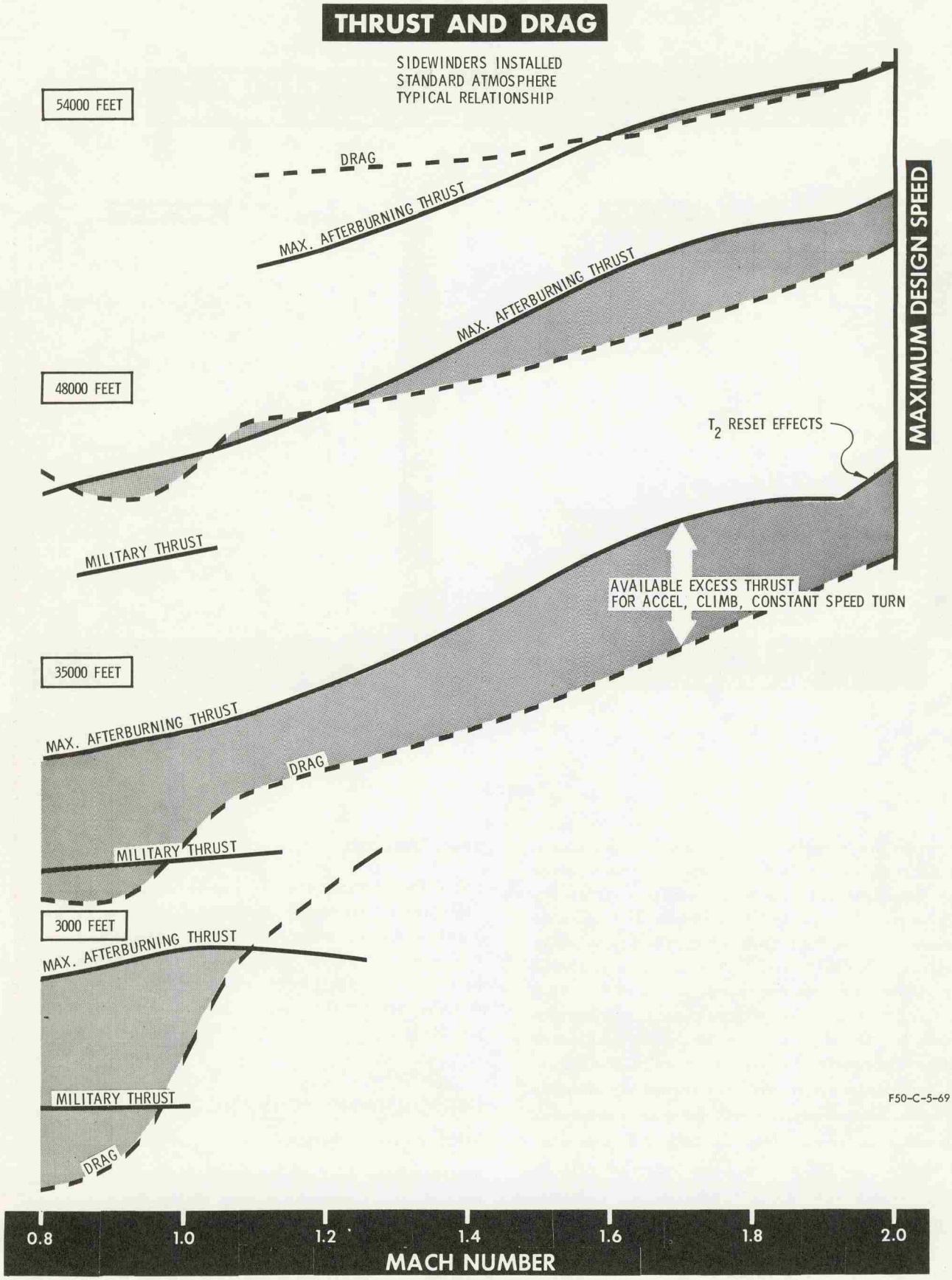
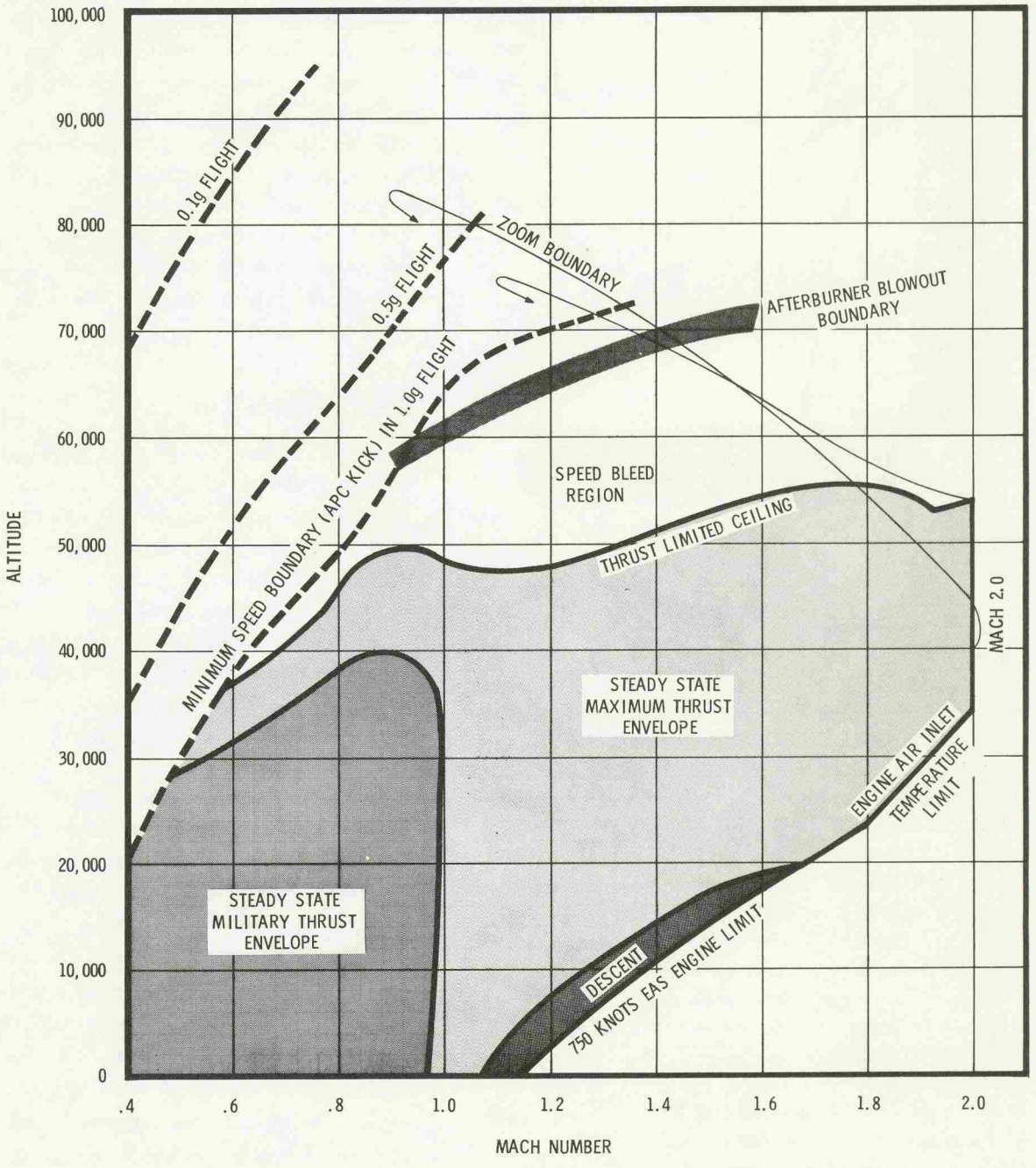

THRUST AND DRAG
SIDEWINDERS INSTALLED
STANDARD ATMOSPHERE
TYPICAL RELATIONSHIP
54000 FEET
DRAG
MAX. AFTERBURNING THRUST
MAX. AFTERBURNING THRUST
48000 FEET
T_2 RESET EFFECTS
MAXIMUM DESIGN SPEED
MILITARY THRUST
AVAILABLE EXCESS THRUST
FOR ACCEL, CLIMB, CONSTANT SPEED TURN
35000 FEET
MAX. AFTERBURNING THRUST
DRAG
MILITARY THRUST
3000 FEET
MAX. AFTERBURNING THRUST
MILITARY THRUST
DRAG
F50-C-5-69
0.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

Figure 6-5

FLIGHT ENVELOPE

SIDEWINDERS INSTALLED
GROSS WEIGHT 16000 LB
STANDARD ATMOSPHERE



F50-C-5-92

Figure 6-6

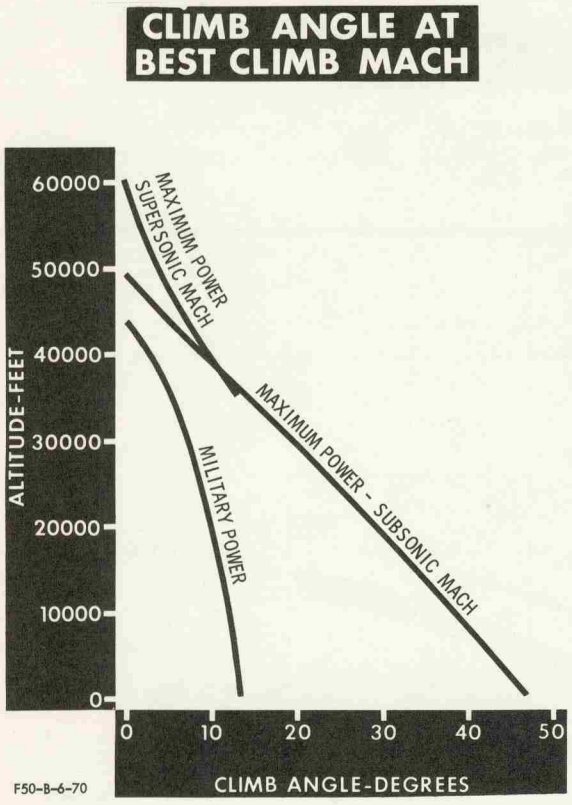


Figure 6-7

ities. Figure 6-5 shows typical thrust and drag variations over the flight range of speed and altitude. At a given speed, the difference between the thrust and drag represents the excess thrust that is available to produce rate of climb or to 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 the thrust and drag are equal and therefore represent the stabilized level flight conditions. The thrust lines shown are for the Maximum power operation. Obviously, at lower power settings the entire level of thrust is lowered, resulting in less excess thrust and lower stabilized level flight speeds. At near sea level altitudes, 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, maneuvering flight without loss of speed or acceleration. As altitude is increased, due to the relative shape of the two curves, the drag exceeds the thrust at subsonic speed at a lower altitude than at supersonic speed. This results in a considerable variance 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 be supersonic 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 maximum excess power (excess thrust $\times$ velocity) occurs at approximately Mach 1.7 number indicating that best rate of climb and maneuvering exist at this speed. The maximum level flight altitude will be obtained also at this speed as it is the last point that the thrust and drag curves touch.

PERFORMANCE ENVELOPE.

The performance envelope or the speed and altitude capability of this aircraft, contains three separate and distinct regions. These are illustrated in figure 6-6. 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 altitudes above 10,000 feet to altitudes well over 50,000 feet. Note the increase in ceiling under standard temperature conditions of approximately 6,000 feet obtainable at high supersonic speeds. This ceiling is also dependent on aircraft gross weight 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 30,000 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 results in a steep climb angle of approximately 45 degrees at sea level, decreasing as altitude is increased. Care should be exercised following take-off to anticipate the high forward acceleration of the aircraft as climb speed is approached and to assume the proper climb attitude to insure maximum performance. Figure 6-7 shows the relative angles for the various climb configurations.

ZOOM CLIMBS.

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 altitudes 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 attained 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 40,000 to 45,000 feet and Mach 2.0 approximately 4,000 feet may be gained, with proper pilot technique, for each Mach .1 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 2,000 feet gained for each Mach .1 lost from the thrust limited ceiling. Maximum altitude is obtained in a zoom initiated from maximum permissible speed in the 40,000 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 pre-planned 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 as only limited maneuverability is available afterwards; hence ground control must know the approximate time and distance required to execute the required zoom. In tactical situations, zoom climbs will probably not be initiated until target detection has been obtained on the radar scope. A pull-up 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 pre-selected altitude, it is best to make a pull-up of approximately 1.5-G, attaining a maximum climb angle of approximately 20 degrees. The push-over to the preselected altitude should be started after approximately one-half the desired altitude gain. The conscious effort of pushing over is minimized by the natural tendency of the aircraft to nose down as airspeed is reduced. In zooms to above 65,000 feet, afterburner blowout will be experienced as the minimum operating pressure level of the afterburner is crossed. The speed and altitude combination for this boundary is approximated on figure 6-6. The boundary is shown as a fairly wide band because the blowout point is affected by individual afterburner performance and is also sensitive to pull-up or pushover technique. Experienced pilot technique will permit maintaining afterburner operation as long as possible. As blowout is sensed, the throttle should be retarded slowly to maintain operation. When blowout occurs, the throttle should be retarded out of afterburning to close the nozzle to minimum area to maintain engine operation and to prevent engine overspeed. Caution should be exercised not to allow the engine to exceed the allowable overspeed. EGT should be monitored to maintain operation within limits. It is possible for the engine to go into stall as the afterburner blows out. Throttle reduction to IDLE or possibly engine shutdown will be necessary in order to control EGT. If the latter is necessary, restart can be accomplished during the descent following the zoom. Zooms accomplished to attain maximum altitude are initiated as stated previously, from 40,000 feet or wherever in this altitude range, maximum Mach number is permissible. An initial pull-up of 2.0 to 2.5 G's 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. A maximum climb attitude of approximately 45 degrees will be reached at about 70,000 feet. From this point on climb angle will decrease to zero as the aircraft is allowed to arch over the peak of the zoom in a ballistic flight path. Indicated airspeeds as low as 50 knots can be expected; however, the aircraft load factor will have decreased to close to zero G and airplane angle of attack will never exceed approximately 11 degrees (the stick shaker setting). Lower than level flight indicated airspeeds are possible without stalling due to the accompanying low load factor. Zoom climbs initiated from 60,000 feet can be expected to reach 80,000 feet, whereas the optimum zoom from 40,000 feet

CHANGE IN GO-AROUND CAPABILITY WITH AIRSPEED, ALTITUDE, TEMPERATURE AND WEIGHT

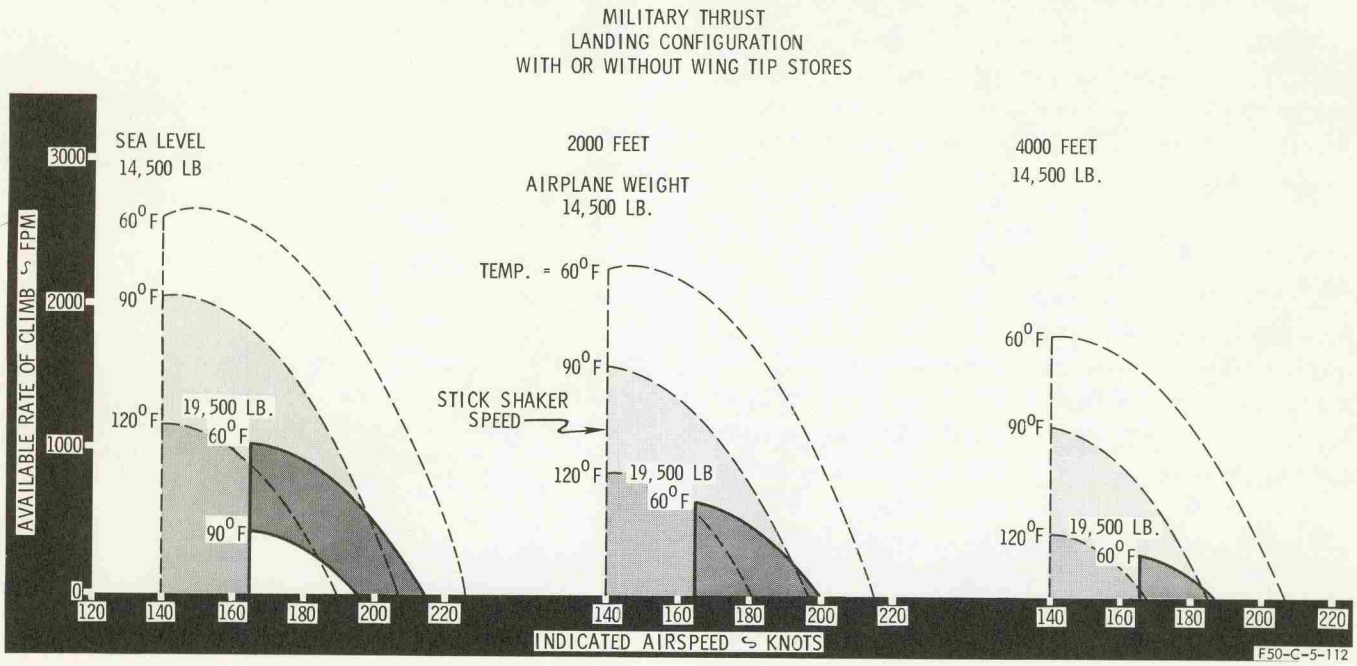


Figure 6-8

will exceed 90,000 feet. 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 85,000 to 87,000 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 cabin pressurization can be expected to decrease, causing suit inflation, if engine stall is encountered above the afterburner blowout point. If engine flameout occurs or it is necessary to shut down the engine, electrical power will not be lost because of the high windmilling engine speed at high altitude. If the flight path is leveled out prior to afterburner blowout, the afterburner will blow out as speed bleeds to the blowout boundary. The natural descent angle from the peak altitude of a zoom can be expected to be as steep as, or in proportion to, the ascending climb angle.

GO-AROUND.

The available excess thrust to accomplish a go-around varies with aircraft configuration, airspeed, gross weight, altitude and ambient temperature. In the landing configuration, use of Military Thrust for go-around or even approach may be inadequate as extremes of these variables are approached and afterburning thrust becomes necessary. As a measure of go-around potential, the available rate of climb with Military Thrust is shown in figure 6-8 as affected by these variables. It will be noted that with full internal fuel weight and at high ambient temperature the available rate of climb at sea level altitude is less than 100 feet per minute at recommended final approach speed, indicating that afterburning would probably be necessary to accomplish a comfortable go-around. Study of this figure will permit familiarization with the relationship of approach speeds to maximum level flight speeds and available rate of climb as a measure of acceleration capability, etc. Additional performance data readable to greater accuracy is presented in Appendix I. Awareness of the available performance under expected operating conditions is essential to optimum utilization of the aircraft.

LEVEL FLIGHT CHARACTERISTICS.

LOW SPEEDS.

Low speed handling characteristics are good. Longitudinal and lateral control are positive and effective down to the minimum useable speeds. Sideslips 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 down 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 .88 to Mach .90. This is so slight that it will be barely perceptible. In the 450 to 550 knot speed region, at medium and low altitudes small amplitude, residual yaw oscillations may be experienced on some aircraft. This characteristic is considered normal although it is evidence of less than optimum operation of the yaw stability augmenter in the individual airplane.

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 variance in trim over the supersonic speed range, the magnitude of the change is very small as indicated in figure 6-9. 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 6-12). A directional trim change which requires increasing trim with increasing Mach number may be experienced or detected on some airplanes. Generally, the sideslip is to the right 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.

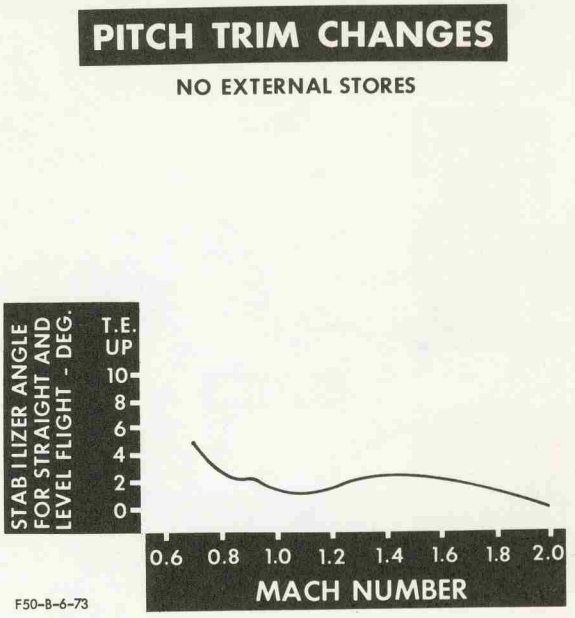


Figure 6-9

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 6-10 shows the difference in time, fuel, and distance requirements to accelerate at various altitudes. The variance with altitude is emphasized in figure 6-11 which shows the optimum altitude for the minimum fuel is 35,000 feet. This will vary somewhat with the existing atmospheric temperature but usually occurs in the 35,000 to 40,000 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 will be available in this range.

MANEUVERING FLIGHT.

Maneuvering stick forces are moderate and generally unaffected by Mach number at a given altitude. See figure 6-12. Forces will, however, vary with altitude, increasing as altitude is increased. This is because forces vary with the amount of stabilizer deflection and bobweight effect, and are not dependent on airloads at the

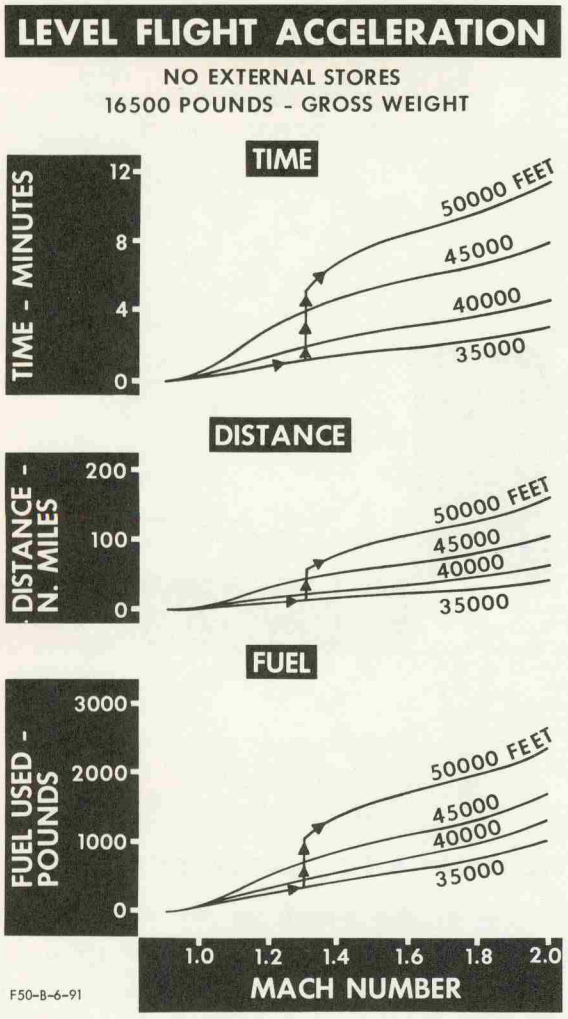


Figure 6-10

control surface as is the case in unboosted or semi-boosted control systems. As a result, the aircraft will feel more stable and be more comfortable in maneuvers at altitude where greater stabilizer deflection is required. In turning flight, adverse sideslip is not experienced, eliminating the necessity to use the rudder above flaps up speed.

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 6-13 shows this normal increase in turn radius due to Mach number. Figures 6-14 and 6-15 show the maneuvering capabilities of this aircraft without a speed loss for various altitudes in terms of both load factor and turn radius. Notice that at any altitude the maximum constant speed G's are available at the speed 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 6-14 also illustrates the constant Mach and constant load factor capability in a climbing turn. For example, it can be seen that ample constant speed, constant G capability is available to permit a 1.5 G turn with sufficient excess thrust remaining to climb quite readily to approximately 50,000 feet at which altitude there is only enough excess thrust to maintain the 1.5 G turn. Higher load factors then shown on these figures can be utilized to accomplish the desired maneuvering in decelerating flight.

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 airplane is easily controlled. Due to the high rates of descent that

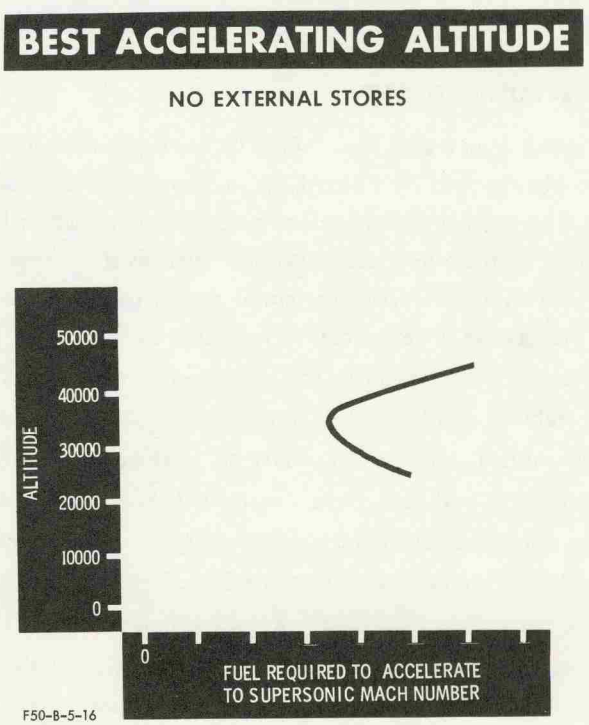


Figure 6-11

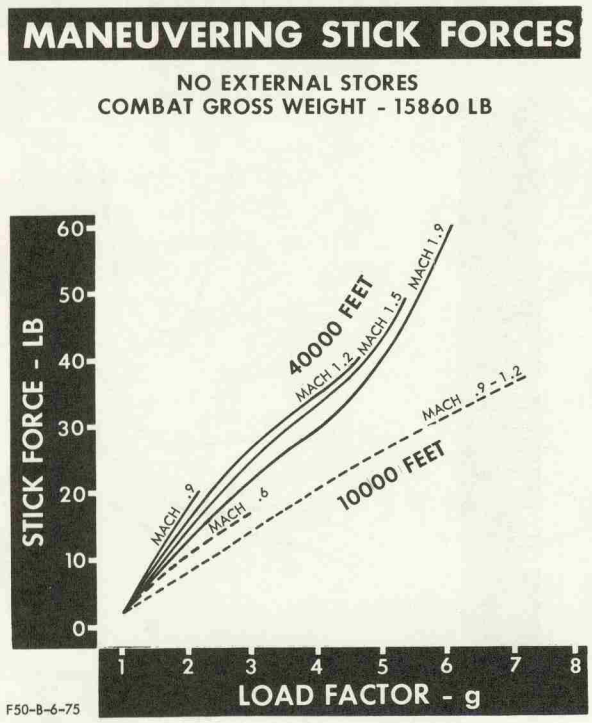


Figure 6-12

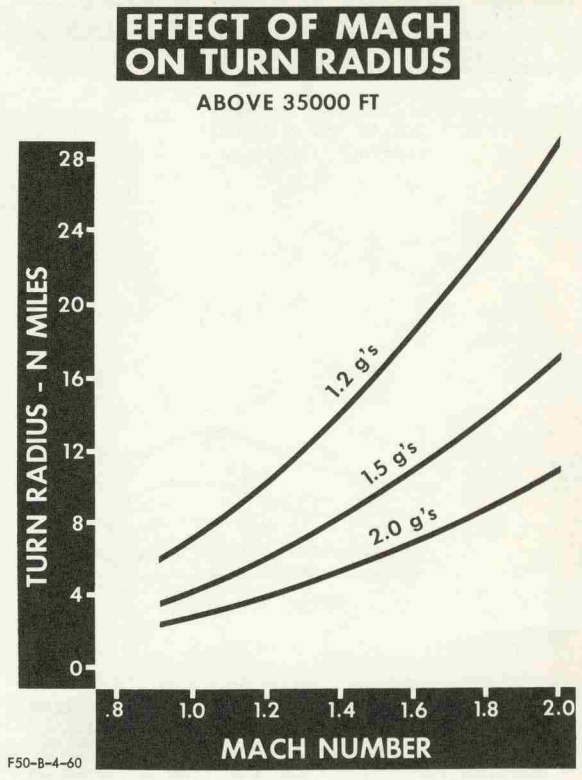


Figure 6-13

are possible, steep dives should be executed with caution. Altitude loss in dive recovery is shown in figure 6-16 for any combination of speed, dive angle and pull-out load factor. The altitude loss in recovery is the same for this aircraft for the same conditions 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.

FLIGHT WITH EXTERNAL STORES.

Flight characteristics with tip stores or pylon tanks installed are essentially the same as without external stores. Of course performance is decreased 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.

FLIGHT WITH ASYMMETRICAL LOAD.

GAR-8 MISSILES.

Installation of wing tip stores increases the efficiency of the wing and produces additional lift. Installation of a single missile results, therefore, in more lift on one wing than on the other, creating a rolling moment. The rolling moment is easily controllable under all flight conditions. It is most pronounced under high lift conditions such as low airspeed or in accelerated flight. On take-off, the wing which has the missile installed will roll up immediately after becoming airborne. Formation take-offs involving any aircraft in the one-missile configuration is prohibited. During negative load factor maneuvers the wing with the missile will roll down. This characteristic is also noticeable during longitudinal acceleration and deceleration. The former produces a wing-up rolling tendency, whereas during deceleration the missile wing rolls down. When firing missiles, a slight wing drop will be noticeable to the side of the fired missile. This is due to the reduced lift on that side when the missile leaves the wing tip.

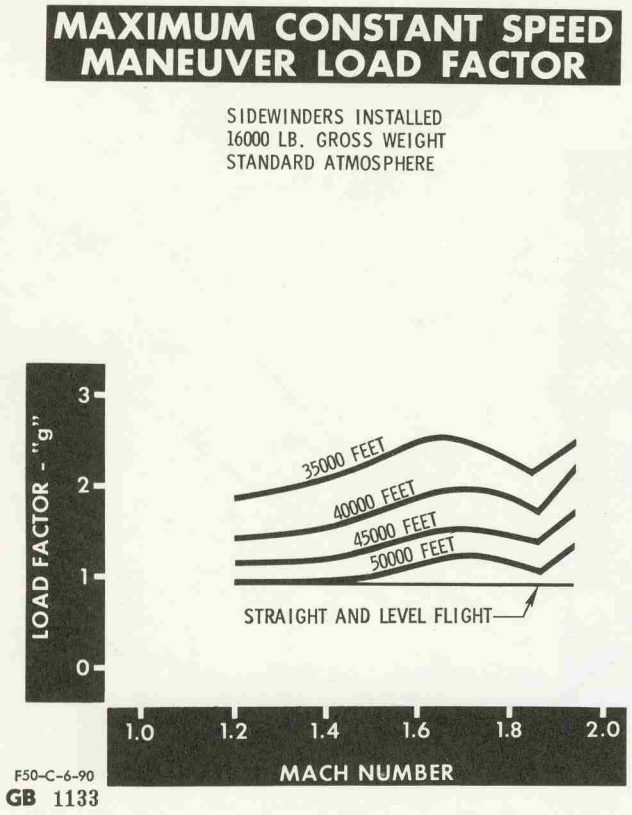


Figure 6-14

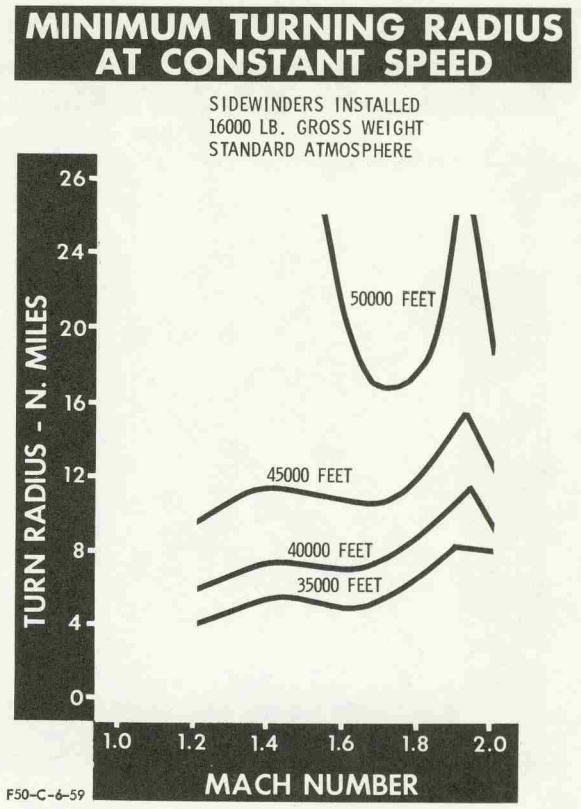


Figure 6-15

ASYMMETRIC TIP TANK FUEL LOAD.

Adequate control is available for landing with one tip tank full and one tank empty under smooth air conditions; however, consideration should be given to the added aileron requirements under strong or gusty 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. 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.

FORMATION FLYING.

Because of the rapid airplane acceleration characteristics and the segmented burning characteristics of the afterburner, a pilot of low proficiency may find himself quite busy while flying afterburner formation; however, after a learning period of two or three flights, a good close formation take-off and flight can be made. There is no problem associated with Military Thrust formation take-offs and climbs.

AFTERBURNER FORMATION TAKE-OFFS.

Due to the rapid acceleration of the aircraft, it is necessary for the leader to use a power setting somewhat less than full uniform to allow the wingman to maintain his position. Large variations in thrust with a relatively small throttle movement are obtained when operating with the afterburner, particularly where the afterburner shifts from sector to uniform burning. If the wingman modulates through this point during take-off, he will fall behind and then overtake the leader rapidly as maximum power is attained. It is preferable to remain in uniform burning throughout the take-off roll. Because of the short span of time involved in accomplishing a take-off, the wingman may find some difficulty in stabilizing his position, in which case the incremental use of speed brakes in the low speed range becomes a logical aid to maintaining position. The wingman should become airborne with the leader; gear and take-off flaps should be retracted on a signal from the leader. After gear and flap retraction, the leader may advance power to almost full uniform. If afterburner light is accomplished simultaneously between two aircraft and power

ALTITUDE LOSS IN DIVE RECOVERY

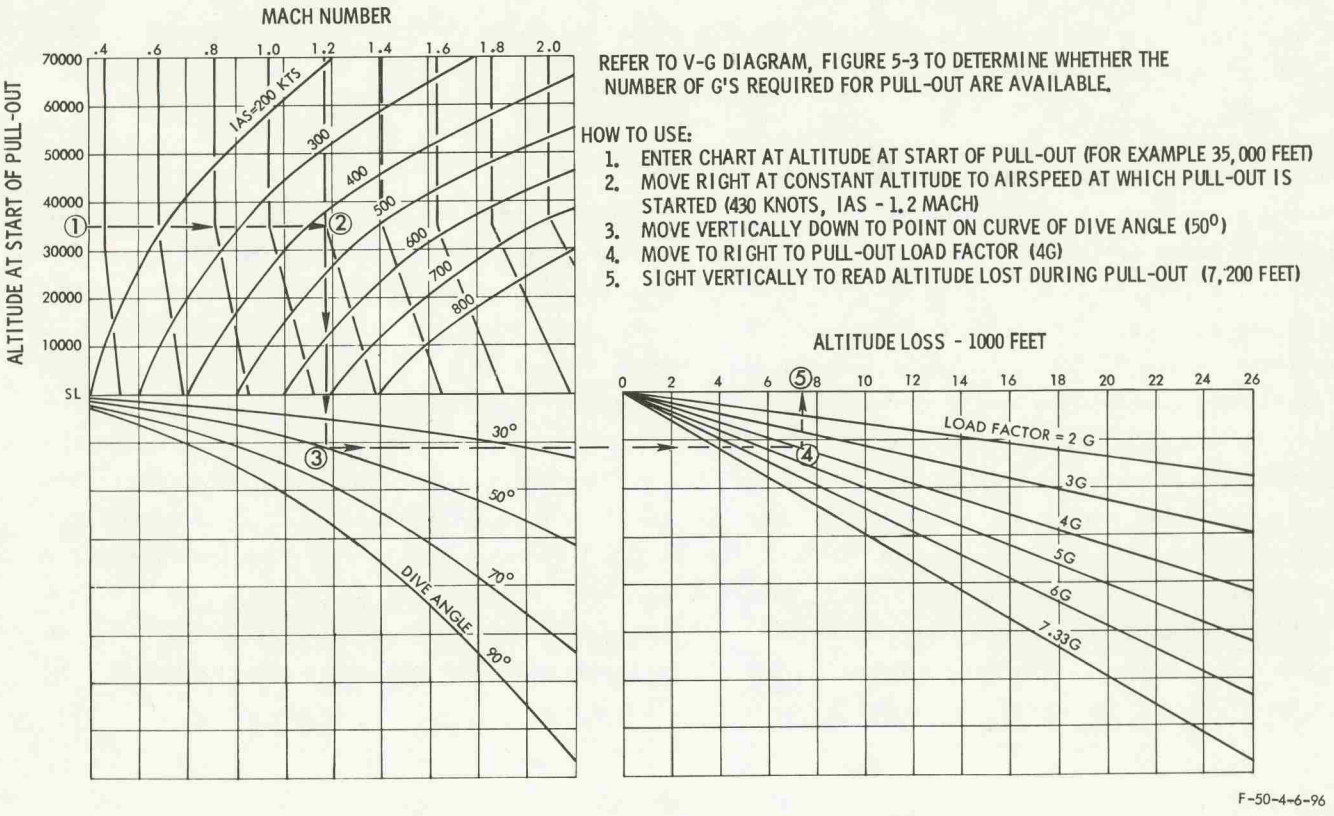


Figure 6-16

adjustments made immediately, the wingman will be able to maintain a satisfactory position throughout take-off with small power changes.

AFTERBURNER FORMATION CLIMB.

If the wingman is in formation at take-off, he can easily maintain position throughout an afterburner climb to subsonic ceiling. The leader must select a power setting less than full throttle to allow the wingman to make power adjustments. Formation join-up from single ship take-offs can be made if the leader turns after take-off and retards his throttle to sector burning. A leader who maintains an excessively high power setting probably will reach cruise altitude before his wingman can reach formation climb position.

SUPERSONIC FORMATION.

In close and combat formation at speeds up to Mach .9 no critical or safety of flight regions of operation are involved. However, as the aircraft reaches supersonic flight, shocks begin to form. These shocks attach themselves to such places as the nose, wing, ducts and the tail. The bow wave and the tail shock are parallel and effectively form one wide, disturbed area. This shock 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 airplane as the speed is increased. When this occurs, the reaction is much the same as if the aircraft had flown through a jet wake. Yawing and rolling disturbances will develop as the aircraft passes in and out of the shock wave. If the wing airplane passes completely through the wave on a course parallel and level with the lead aircraft, the passing airplane will experience one yawing cycle. As the nose of the wing airplane enters the wave, the aircraft will yaw toward the leader. As the airplane becomes enveloped by the wave, it will yaw away; and as it leaves the wave, it will again yaw in, completing one oscillation.

WARNING

If the overtake rate is in the vicinity of Mach .01 or .02 or if the passage through the wave lasts approximately two seconds, there is a possibility of structural damage due to high sideslip angles.

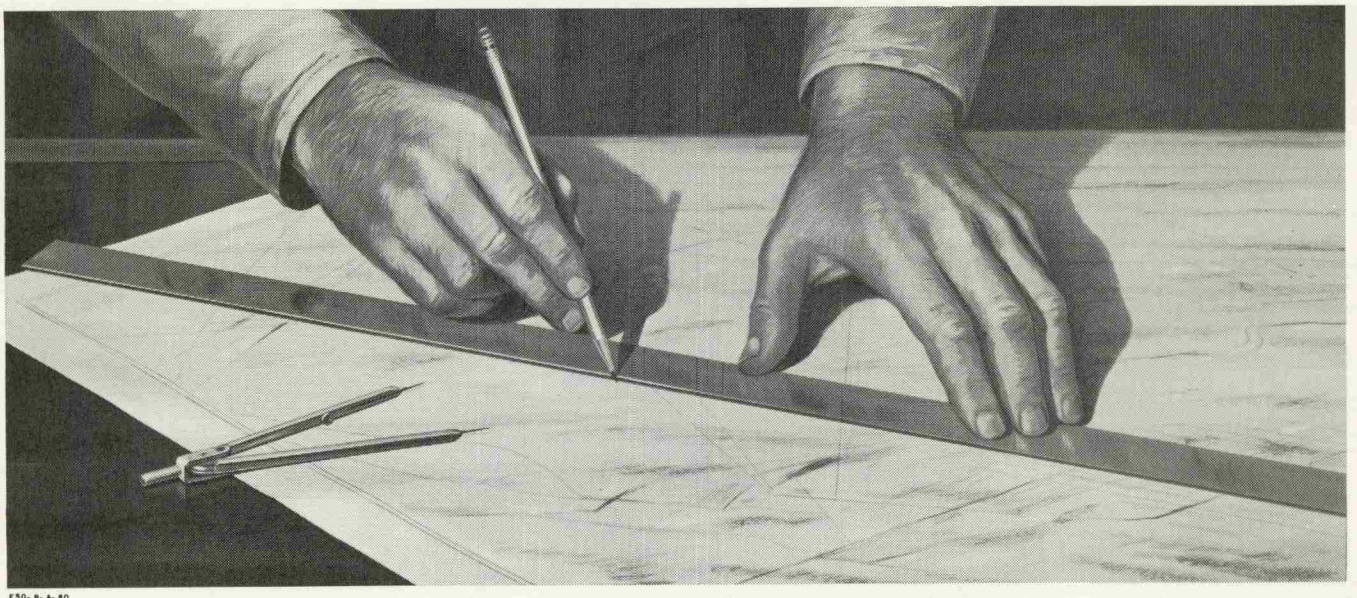
If the wing airplane passes directly above or below the lead airplane, pitching moments instead of yawing moments will be experienced. The first movement will be toward the lead airplane, then away, then back again as passage through the flow field is completed. If the wingman remains behind the shock cone, no difficulty will be experienced. This may be accomplished by dropping back as Mach number is increased. When flying tactical formation at supersonic speeds, the wing airplane may receive random actuations of the stick shaker and kicker while crossing through the wave. If kicker operation is experienced, the wing airplane will fall behind, but that is all. The most serious problem encountered in supersonic formation flying is afterburner cycling induced by throttle movement; it varies in intensity between airplanes. Occasionally an afterburner will blow out if an attempt is made to stabilize between sector burning and uniform burning. It is, therefore, quite important that the leader establish a power setting sufficiently above or below the afterburner switchover point so that the wingman can maintain position.

BOUNDARY LAYER CONTROL.

The installation of boundary layer control to effect low 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 82%. 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.

NOTE

For Sections VII through IX, refer to the unclassified Flight Manual, T.O. 1F-104A-1.



F50- B- A- 80
AM2822

performance data

APPENDIX I

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Part 1—Introduction

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INTRODUCTION

SCOPE AND ARRANGEMENT.

This appendix contains performance data necessary for accurate pre-flight planning, and is applicable to aircraft operating in the following configurations with the J79-GE-3A engine: without external stores; with two wing tip mounted GAR-8 (Sidewinder) missiles; with 165-gallon tip tanks; with 200-gallon pylon tanks and GAR-8 missiles; or with 200-gallon pylon tanks and 165-gallon tip tanks. The material is grouped into "Parts" for planning general phases of each flight. Descriptive text in each Part discusses and explains the use of the charts provided in the Part. Profile charts are included for subsonic cruise, endurance, and refueling missions. A mission planning section at the end of the appendix shows how the individual performance charts for each part of a flight can be combined for overall planning purposes. High speed maneuvering performance data are included in Part 9.

PERFORMANCE DATA BASIS.

Subsonic flight planning information shown in this appendix is based on F-104A flight test data. All charts are based on performance under standard atmospheric conditions; however, corrections for non-standard atmospheric conditions have been included on the charts, when possible.

FUEL AND FUEL DENSITY.

All performance and operating weight ranges included in this appendix are based on operation with JP-4 fuel at a nominal fuel density of 6.5 pounds per U.S. gallon.

SYMBOLS	DEFINITIONS
IAS	Indicated airspeed, airspeed indication uncorrected for instrument error. Where this symbol is used on the performance charts, mechanical error in the instrument is assumed to be zero.
ΔVi	Airspeed position error correction.
CAS	Calibrated airspeed. Indicated airspeed corrected for position error: $CAS = IAS + \Delta Vi$
ΔVc	Airspeed compressibiilty correction
EAS	Equivalent airspeed. Calibrated airspeed corrected for compressibility: $EAS = CAS + \Delta Vc$
TAS	True Airspeed: $TAS = EAS \times 1/\sqrt{\sigma}$ ($1/\sqrt{\sigma}$ from Figure A1-6)
Δ H	Altimeter position error correction
ICAO	International Civil Aviation Organization
M	Mach number
N. Mi.	Nautical miles

Figure A1-1

If fuel density is known to be different from the nominal value, enter the performance charts with an aircraft weight which has been corrected to reflect the difference in weight of fuel on board.

SYMBOLS AND DEFINITIONS.

Symbols used throughout this appendix are listed and defined in the table, figure A1-1.

MACH NUMBER CORRECTION.

Indicated Mach number can be corrected for position error as follows: Enter figure A1-2 with indicated Mach number and read true Mach number directly from the calibration curve.

AIRSPEED POSITION ERROR CORRECTION.

Position error correction can be obtained and applied to I.A.S. to determine C.A.S. as follows: Enter figure A1-3 with indicated airspeed (corrected for individual mechanical instrument error); and algebraically add the position error correction value read from the curve to the corrected instrument reading to obtain calibrated airspeed. Algebraically subtract the position error correction from calibrated airspeed value to obtain indicated airspeed.

AIRSPEED–MACH NUMBER CONVERSION.

True Mach number, True Airspeed or Calibrated Airspeed can be obtained from the Airspeed-Mach Number curve as follows: Enter figure A1-4 or A1-5 with Calibrated Airspeed and proceed vertically to the altitude desired, read the True Mach Number. If standard temperature exists at that altitude also read the True Airspeed from the speed lines provided. If non-standard free air temperature exists, proceed horizontally to the base line (from intersection of CAS and Alt.) then move down to the desired temperature and read True Airspeed from the right hand scale. Using the same type procedure, the curve may be entered with True Airspeed or True Mach Number to determine the other two unknown values. Two Airspeed—Mach Number

curves are presented for convenience. Figure A1-4 covers the cruise range of the aircraft. Figure A1-5 provides data for the high speed operating area.

COMPRESSIBILITY CORRECTION TO C.A.S.

Compressibility correction can be otbained and applied to C.A.S. for determination of E.A.S. as follows: Enter figure A1-6 with calibrated airspeed and algebraically add the compressibility correction value read from the curve to the calibrated airspeed to obtain equivalent airspeed.

ALTIMETER POSITION ERROR CORRECTION.

Position error correction can be obtained and applied to indicated pressure altitude to obtain actual flight pressure altitude as follows: Enter figure A1-7 with indicated Mach number (corrected for individual mechanical instrument error); and algebraically add the position error correction value read from the curve to the corrected altimeter instrument reading to obtain correct pressure altitude. (Algebraically subtract the position error correction from the pressure altitude value to obtain indicated altitude).

Note

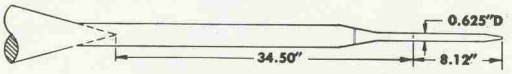
Failure to use position error and airspeed compressibility correction data properly can result in an apparent inability to realize predicted airplane performance.

STANDARD ALTITUDE TABLE.

The standard altitude table (figure A1-8) provides reference temperature, pressure, air density, and sonic speed information which may be of assistance in overall flight planning.

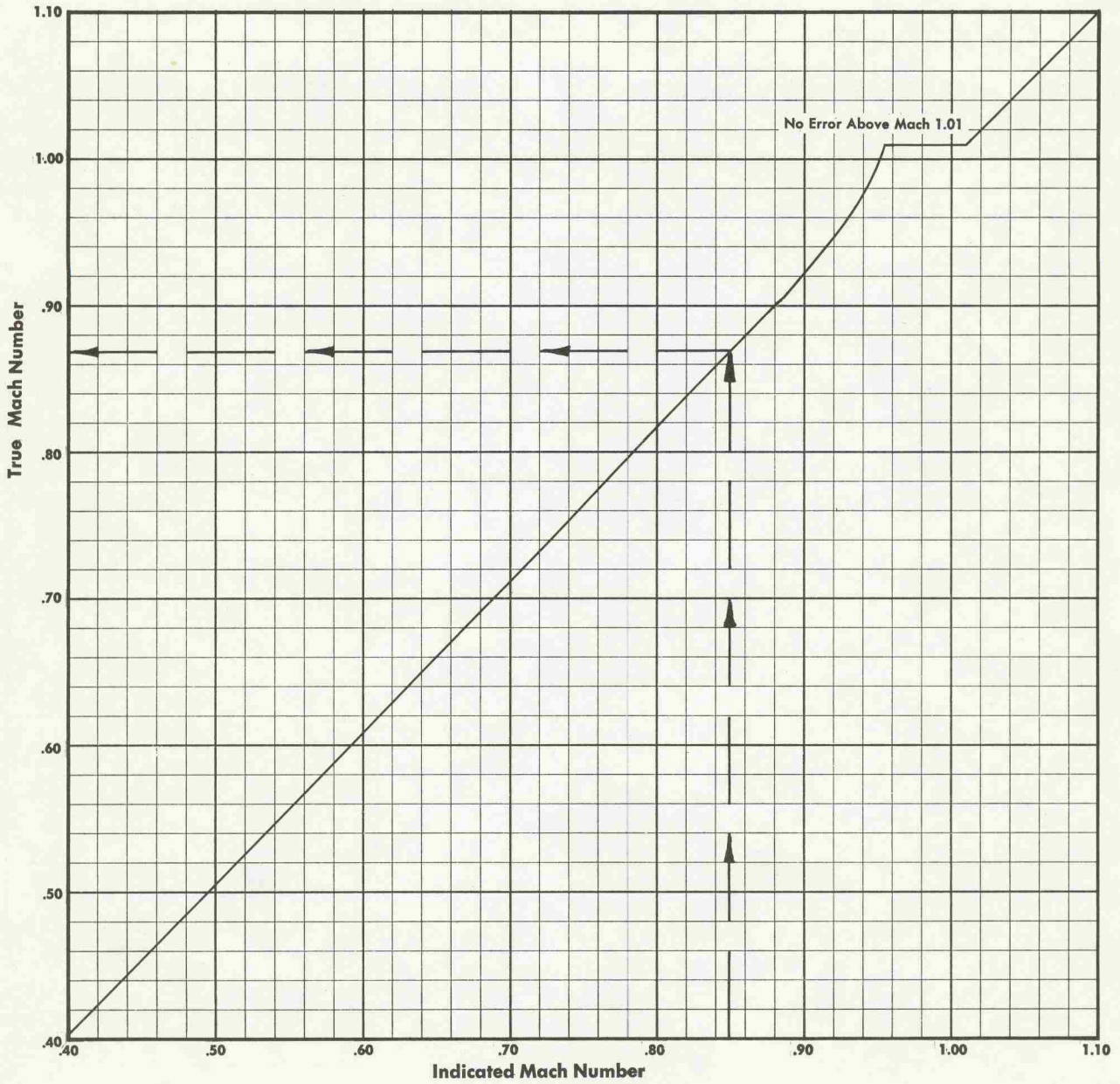
MACH NUMBER CORRECTION

MODEL: F-104A
DATA AS OF: 31 DECEMBER 1958
DATA BASED ON: FLIGHT TEST



MA-1 Airspeed Head

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

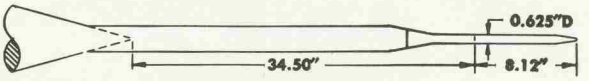


NOTES: Data Based On MA-1 Head Mounted On A 28.5 Inch Boom.
Static Orifices Located 34.5 Inches Ahead Of Nose Of Airplane.

Figure A1-2

AIRSPEED POSITION ERROR CORRECTION

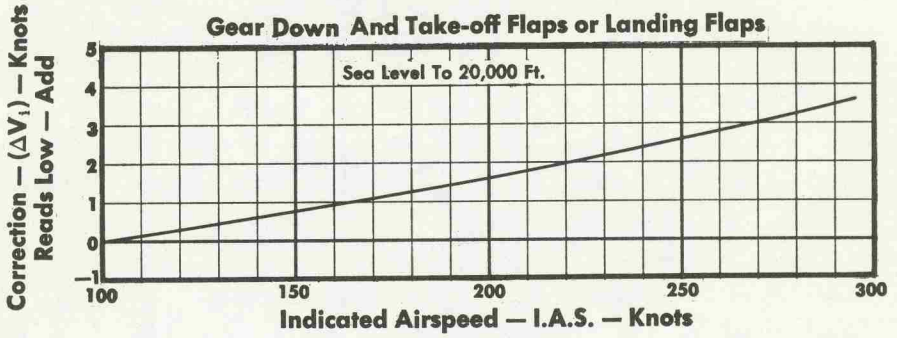
MODEL: F-104A
DATA AS OF: 31 DECEMBER 1958
DATA BASED ON: FLIGHT TEST

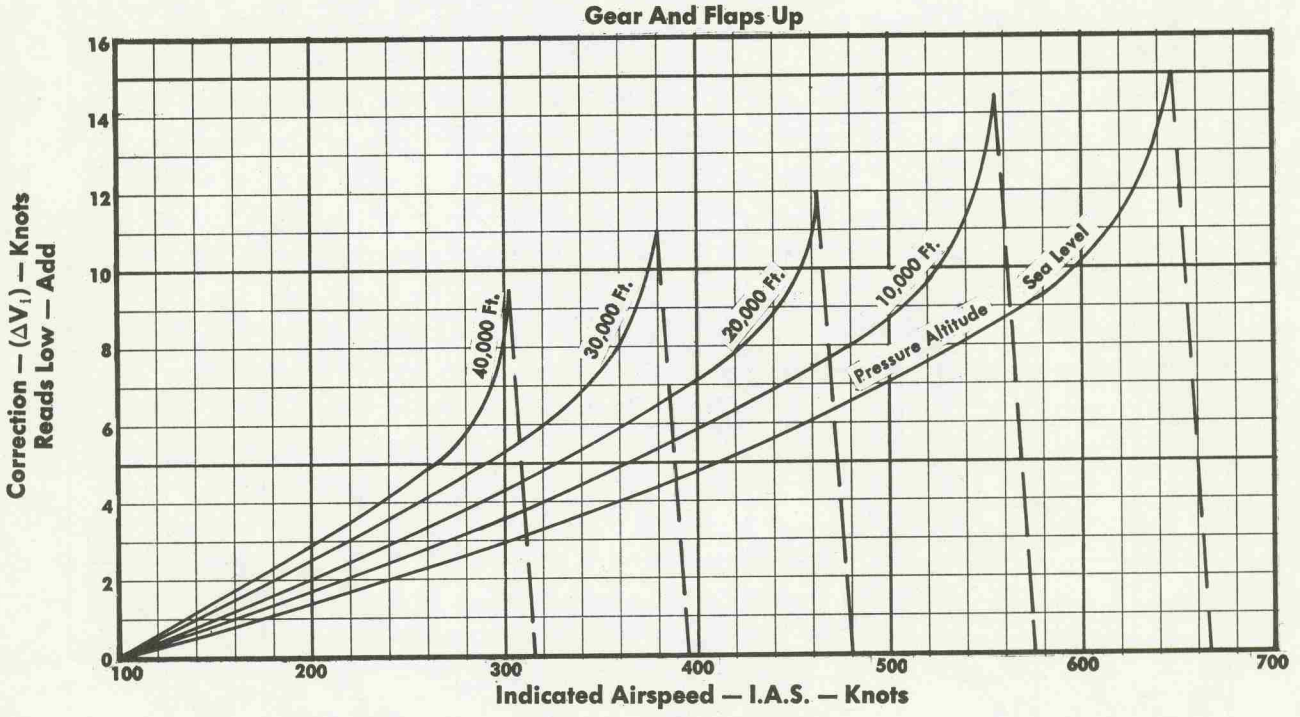


MA-1 Airspeed Head

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

Assume Zero Correction During Ground Run





REMARKS: Add Position Error Correction To Indicated Airspeed To Obtain Calibrated Airspeed (C.A.S.)

C.A.S. = I.A.S. + ΔV_i I.A.S. = C.A.S. − ΔV_i

NOTES: Data Based On MA-1 Head Mounted On A 28.5 Inch Boom, Static Orifices Located 34.5 Inches Ahead Of Nose Of Airplane.

Figure A1-3

AIRSPEED — MACH NUMBER CURVES

0.3 TO 1.2 MACH NUMBER

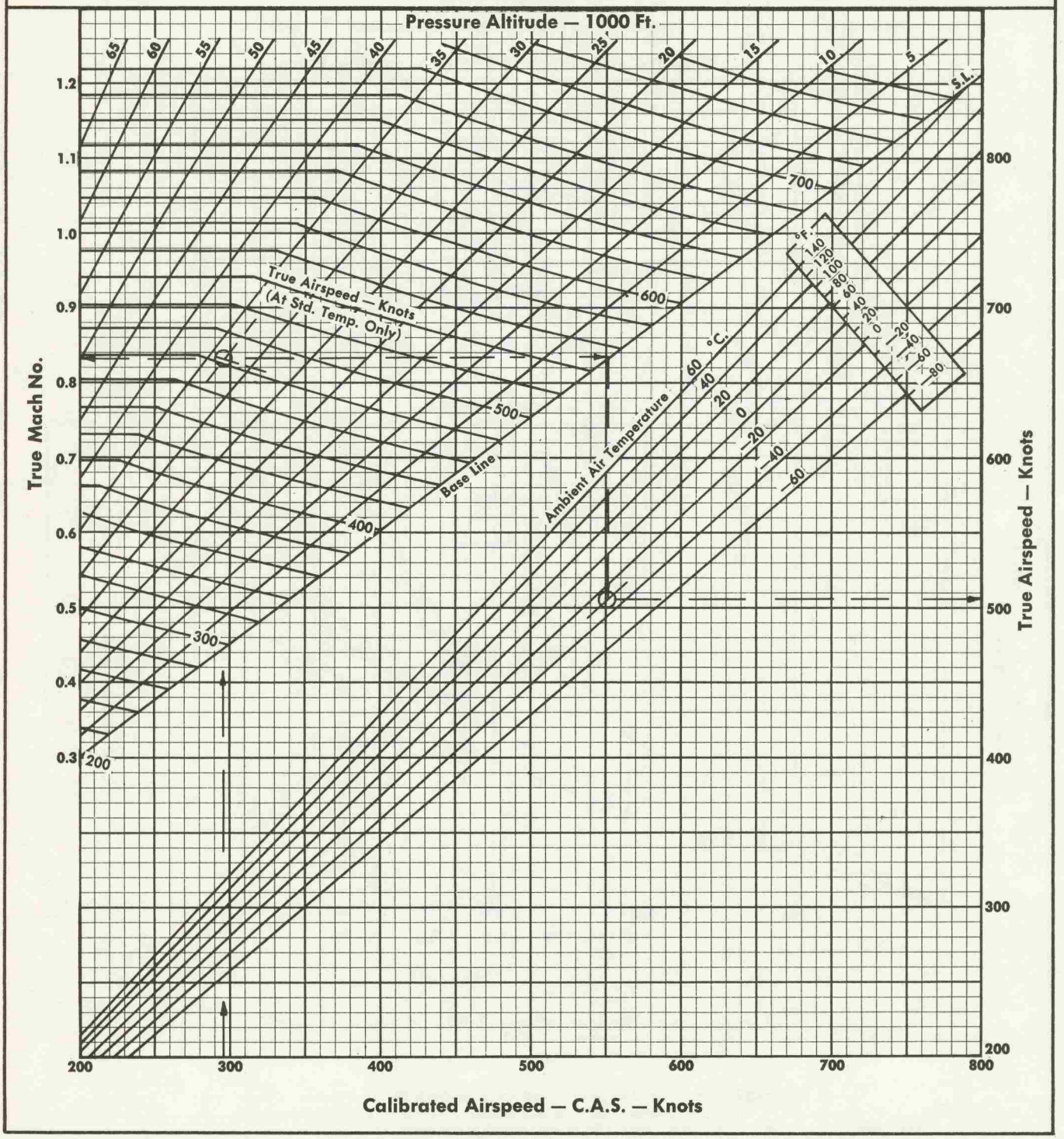


Figure A1-4

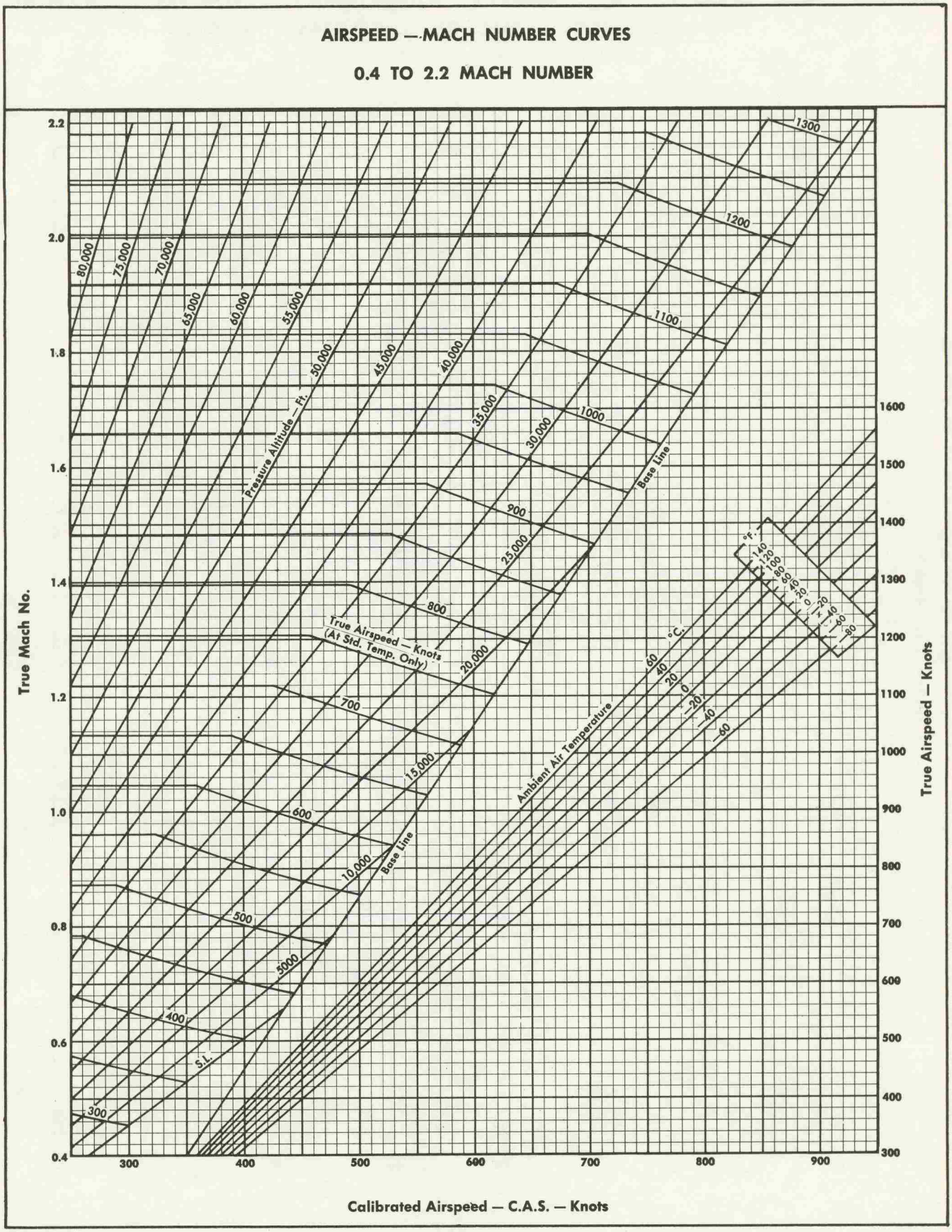

AIRSPEED — MACH NUMBER CURVES
0.4 TO 2.2 MACH NUMBER
True Mach No.
Calibrated Airspeed — C.A.S. — Knots
True Airspeed — Knots
Pressure Altitude — Ft.
80,000
75,000
70,000
65,000
60,000
55,000
50,000
45,000
40,000
35,000
30,000
25,000
20,000
15,000
10,000
5000
S.L.
Base Line
True Airspeed — Knots (At Std. Temp. Only)
1300
1200
1100
1000
900
800
700
600
500
400
300
Ambient Air Temperature
°C.
°F.

Figure A1-5

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED

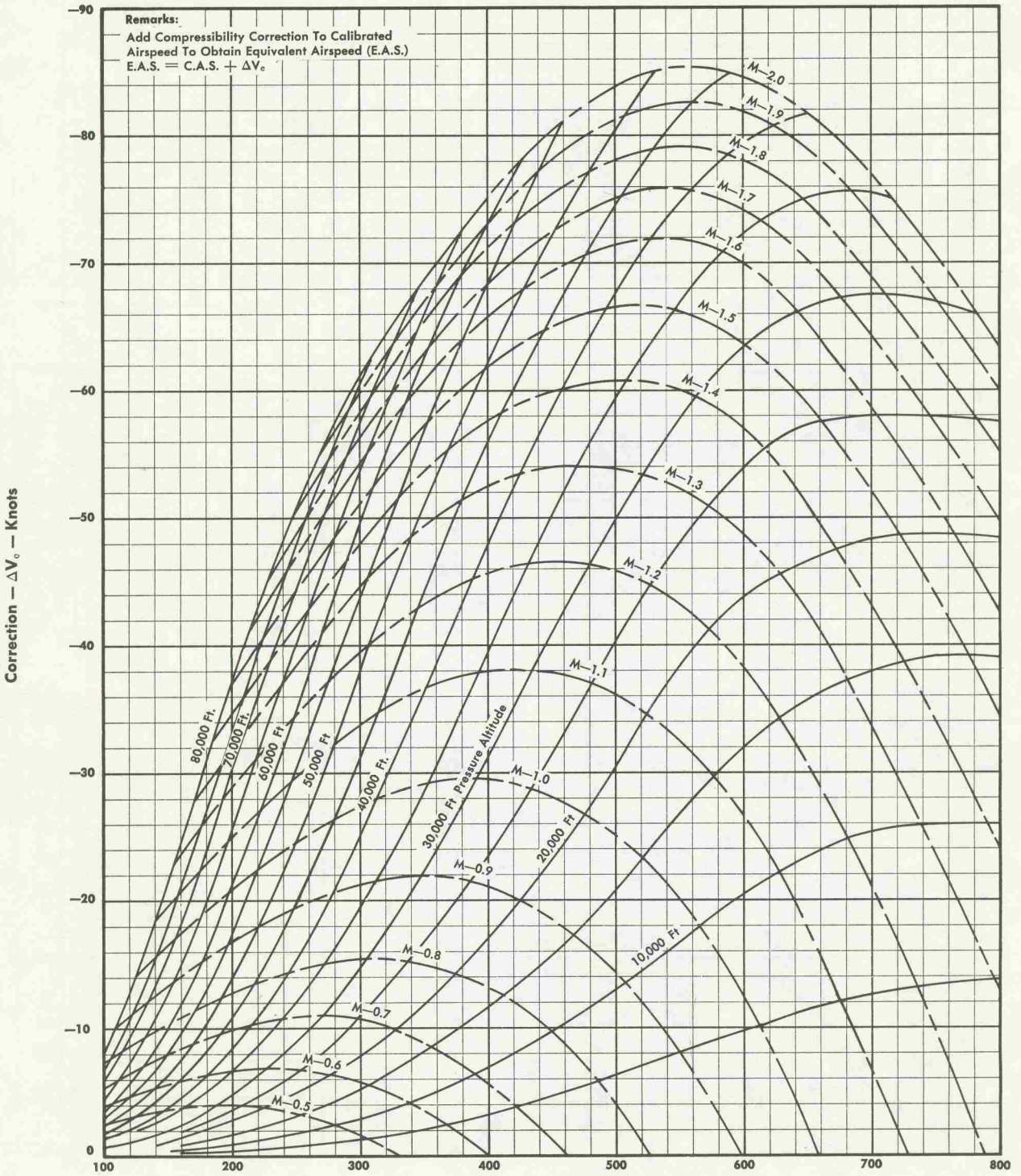
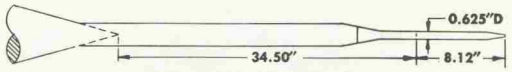


Figure A1-6

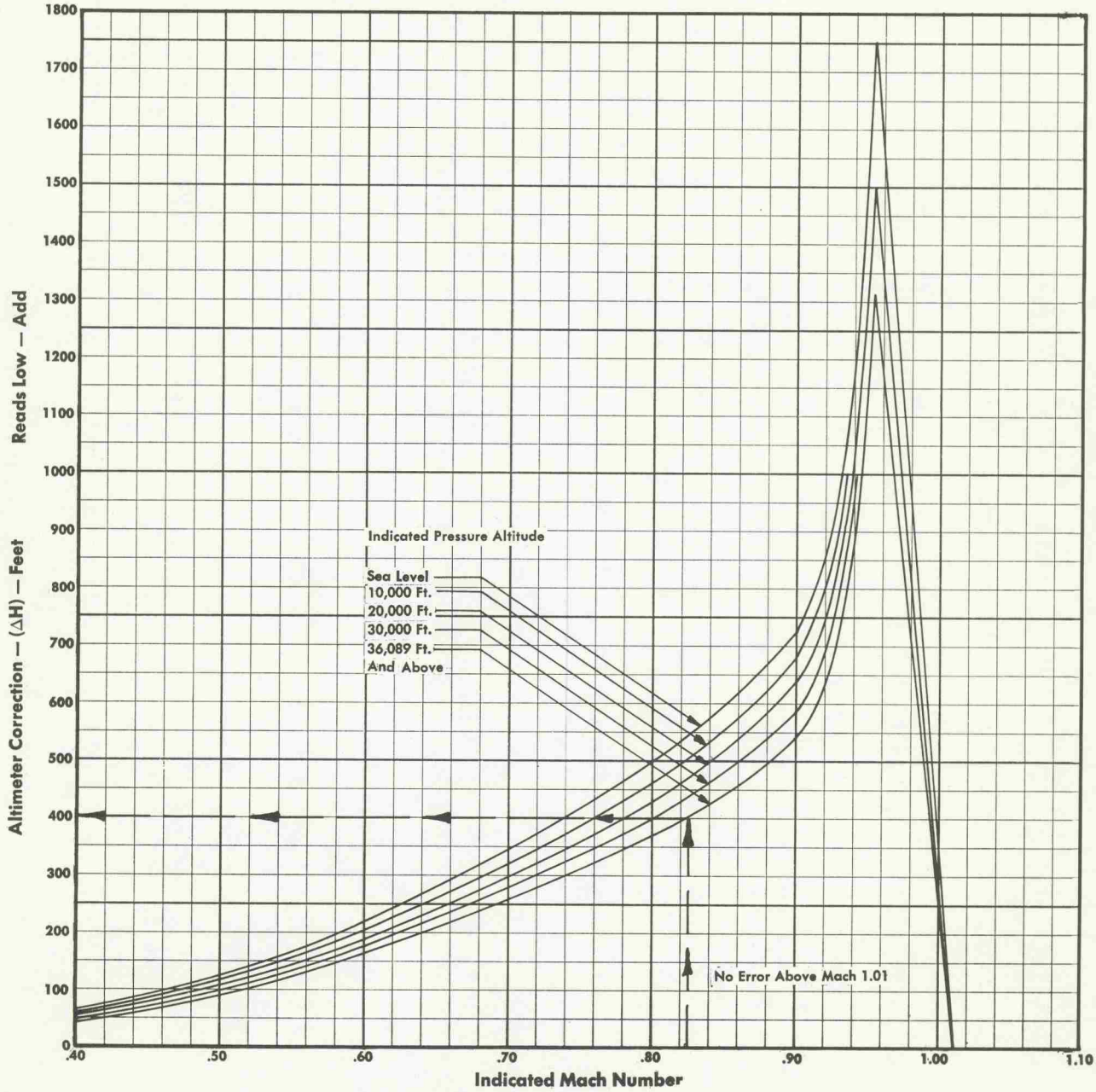
ALTIMETER POSITION ERROR CORRECTION

MODEL: F-104A
DATA AS OF: 31 DECEMBER 1958
DATA BASED ON: FLIGHT TEST



MA-1 Airspeed Head

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



REMARKS: Add Correction To Indicated Pressure Altitude To Obtain Correct Pressure Altitude.

NOTES: Data Based On MA-1 Head Mounted On A 28.5 Inch Boom. Static Orifices Located 34.5 Inches Ahead Of Nose Of Airplane.

Figure A1-7

STANDARD ALTITUDE TABLE

STANDARD SEA LEVEL AIR:
T = 59°F
P = 29.921 IN. OF HG.

W = .076475 LB/CU. FT. ρ_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		SPEED OF SOUND RATIO a/a_o	PRESSURE	
			DEG. F	DEG. C		IN. OF HG.	RATIO ρ/ρ_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 A1-8

TEMPERATURE CONVERSION CHART

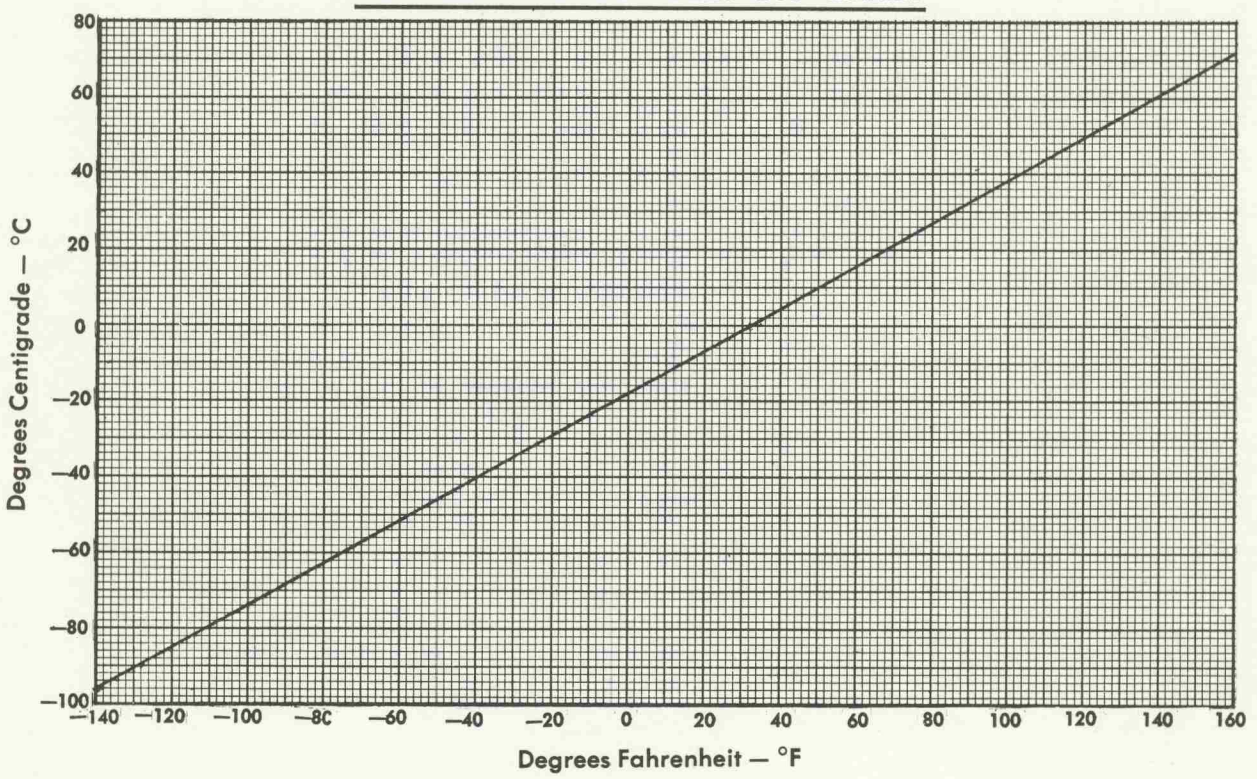


Figure A1-9

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Part 2–Take-Off

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TAKE-OFF

TAKE-OFF PLANNING AND CHARTS.

This Part presents Maximum and Military Trust take-off performance charts. The distances are based on normal acceleration using the normal take-off technique described in Section II. They are applicable to all configurations and weights. Special labeled lines are included for convenience which represent performance for typical configurations with full fuel load at the ramp. (A 150 pound allowance has been made to account for engine start, ground maneuver, and taxi). The gross weights used for these typical performance lines are as follows:

No External Stores	18,850 lb.
Two GAR-8 Missiles	19,270 lb.
Tip Tanks	21,382 lb.
Pylon Tanks & GAR-8's	22,364 lb.
Pylon Tanks & Tip Tanks	24,476 lb.

A planning procedure and charts are also included to provide data for monitoring progress at marked runway distances. A "Go No-Go" concept is introduced as part of this procedure, whereby take-off acceleration is checked as late in the ground run as safely possible. Acceleration is related to total runway length available so that, in the event of subnormal acceleration, a decision to continue or abort the take-off may be made before the refusal distance is reached. Definitions of the terms used in the planning procedure, plus the purpose of each chart in the planning concept, are given in the following paragraphs.

GROUND RUN DISTANCE AND TAKE-OFF SPEED.

Take-off ground run distance is the actual length of runway used from the start of the ground run until the aircraft becomes airborne. Distances are shown for Maximum and Military Thrust operation on Figures A2-2 and A2-7 for normal and maximum performance take-offs. Data for maximum performance speed schedules are included on the Maximum Thrust charts for those conditions where it is desirable to avoid high speed take-offs.

Note

Take-off ground run may be considered to be unclassified provided that it is not correlated with gross weight and combat figuration. This will preclude the necessity of classifying DD Form 175 because of the inclusion of take-off ground run data.

REFUSAL SPEED.

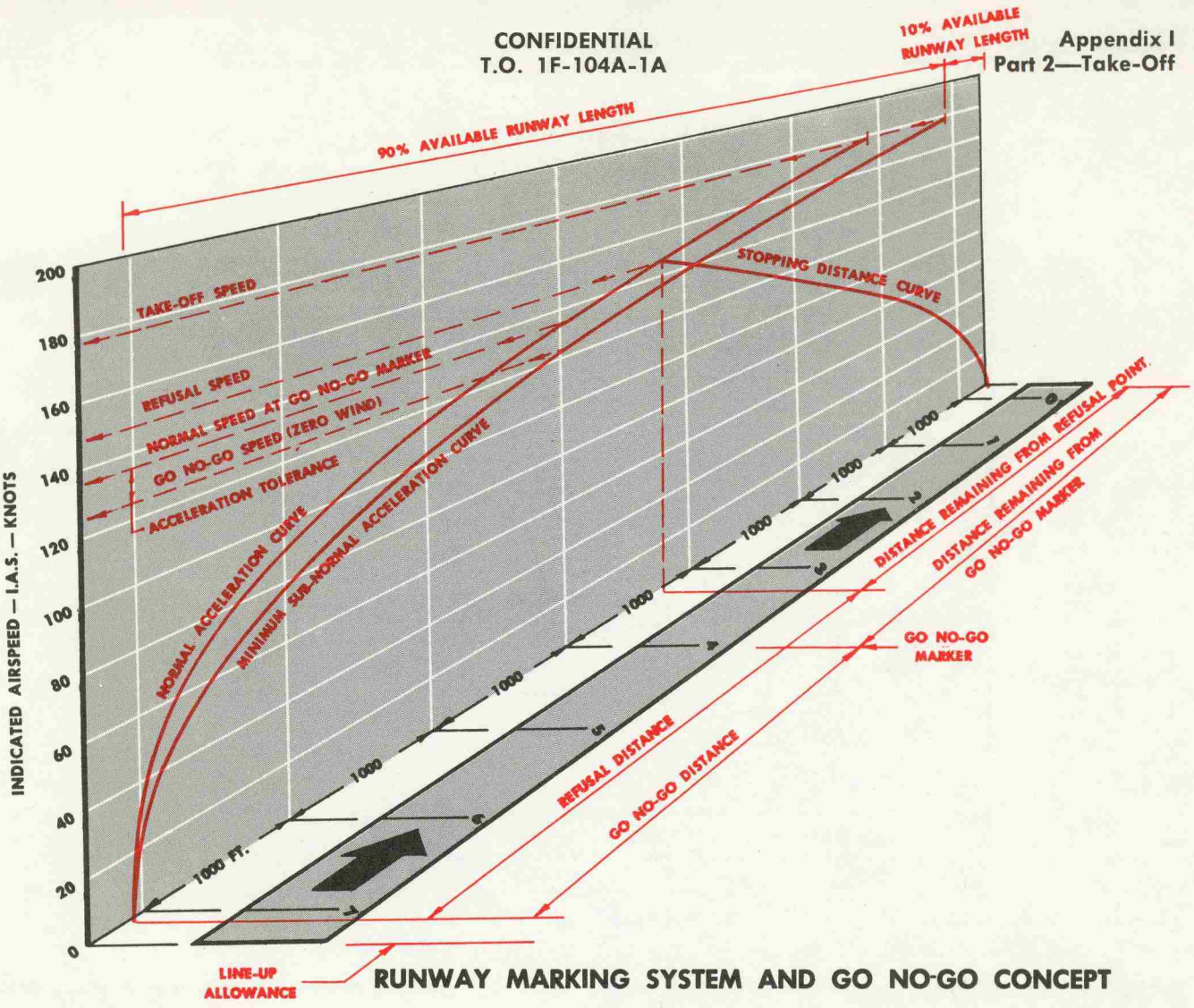
Refusal speed is the highest speed to which the aircraft can be accelerated, assuming normal acceleration, and still be stopped on the remaining runway. The refusal speed charts in this part take into account gross weight, pressure altitude, temperature and actual runway length (Figures A2-3 and A2-8). Refusal speeds are presented for level runways with surface conditions varying from a coefficient of friction (μ) of 0.3 to 0.

REFUSAL DISTANCE.

Refusal distance is the distance at which the aircraft will reach the refusal speed with normal acceleration and is obtained from the velocity during take-off ground run charts (Figures A2-4 and A2-9). These charts are plots of ground run distance versus indicated airspeed and show normal acceleration guide lines that indicate the speed-distance relationships for all required take-off conditions of gross weight, field pressure altitude, and air temperature.

GO NO-GO DISTANCE.

Since the refusal point usually will not occur exactly on a runway marker the last marker before the refusal point is reached must be used as the decision point. This marker, where the final decision to continue or abort the take-off is made, is called the go no-go marker. The distance that the aircraft has traveled in reaching the go no-go marker is the go no-go distance. The take-off should be aborted if the speed of the aircraft is below a specific speed at the go no-go marker. This minimum allowable speed at the go no-go marker is called the go no-go speed.



RUNWAY MARKING SYSTEM AND GO NO-GO CONCEPT

GO NO-GO SPEED.

The go no-go speed is the normal speed at the go no-go marker corrected for acceleration tolerance and runway wind component. The procedure for determining the go no-go data is as follows.

1. Determine available runway length by subtracting the line-up allowance from the runway length.

2. Determine the refusal speed and refusal distance.

3. Determine the distance remaining from the refusal point by subtracting the refusal distance from the available runway length.

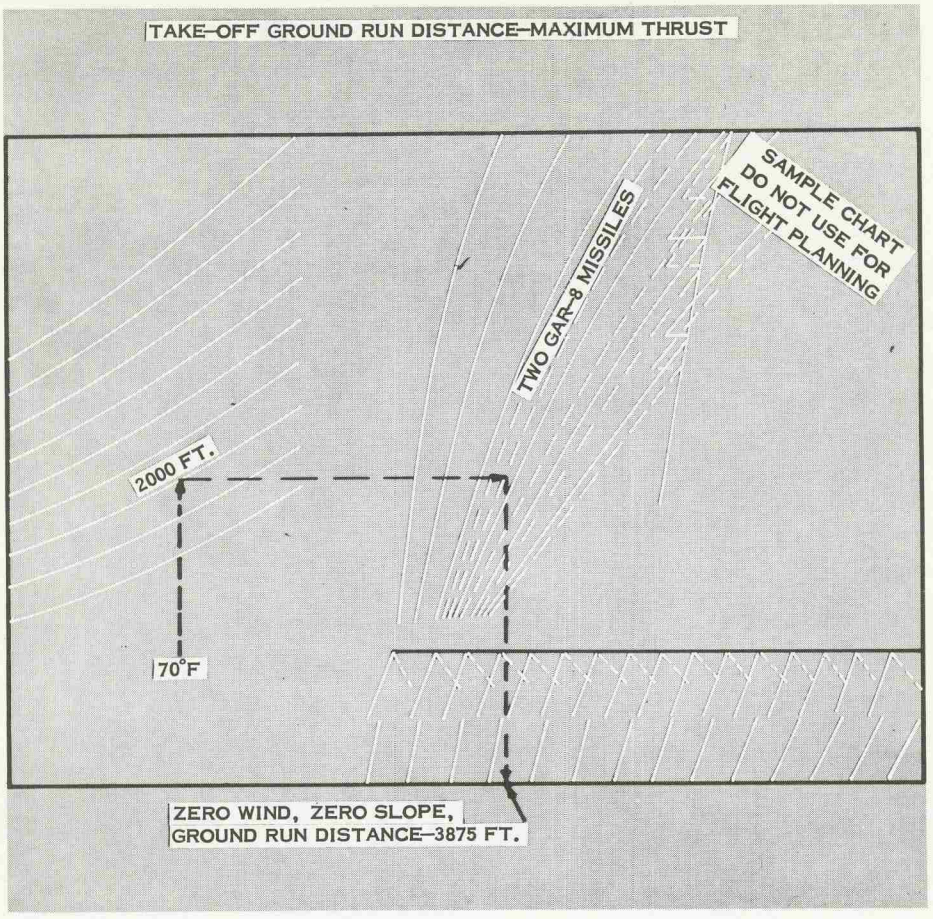
4. Determine the go no-go marker as the closest marker where at least the distance in step 3 remains to the end of the runway.

5. Determine the distance remaining from the go no-go marker to the end of the runway. This is one-half of the odd figure over exact thousands of feet of runway length added to the go no-go marker distance.

6. Determine the go no-go distance (distance from the line-up point to the go no-go point) by subtracting the distance remaining (found by Step 5) from the available runway length. This distance is used to enter the velocity during take-off ground run chart to determine the go no-go speed.

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 take-off. The acceleration tolerance is based on the minimum allowable take-off acceleration which will result in a zero wind take-off ground run no greater than 90 percent of the runway available. Acceleration tolerance data are presented on Figure A2-6 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 might otherwise be



SAMPLE PROBLEM.

Determine take-off performance for the following conditions:

- Maximum Thrust
- Configuration—Two GAR-8 Missiles*
- Gross weight—19,270 Lb.*
- Field ambient air temperature—70°F
- Field pressure altitude—2000 Ft.
- Wind—15 knots from 045 degrees.
- Runway—7300 feet long, 030 degrees heading, dry, hard surface, 1% downhill slope.
- Line-up allowance—250 feet from end of runway.

*Enter on take-off data card.

1. From Figure A2-2 (Take-off Ground Run Distance) obtain the zero wind, zero slope ground run distance .....3,875 ft.*

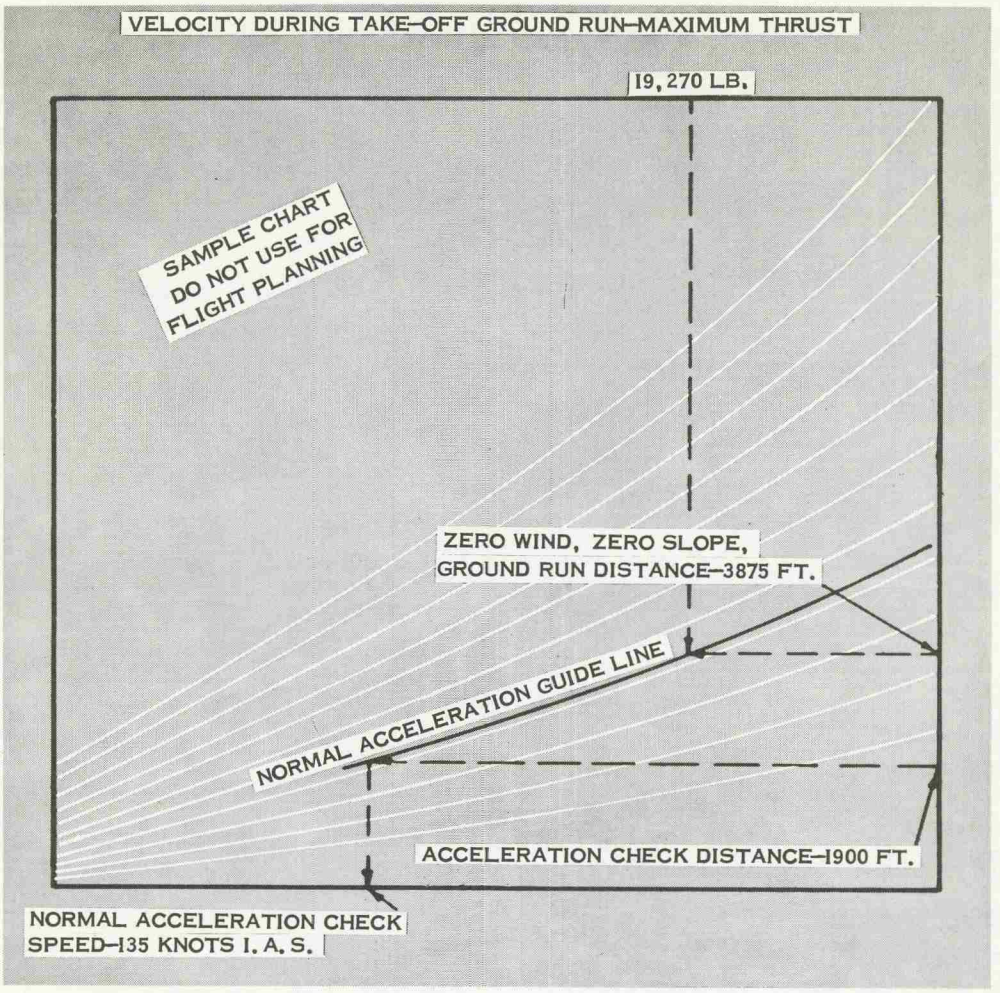
2. Available runway length (7,300 ft. runway length—250 ft. line-up allowance).....7,050 ft.

3. The acceleration check marker, for light weight operation is the *second marker from brake release* .....5,000 ft. remaining marker.

permitted. There must be a serious flaw in the functioning of the airplane and a take-off should not be attempted if the acceleration is deficient to the extent of exceeding the maximum allowable tolerance.

ACCELERATION CHECK DISTANCE.

The final decision to continue or to abort the take-off should be made at the go no-go marker. However, a preliminary check should be made on the progress of



Sample Problem Continued

4. One-half of the odd number of feet over exact thousand feet of runway length, added to distance in item 5, is the distance to the end of the runway from acceleration check marker (150 ft. + 5,000 ft.) . . . . . .5,150 ft.

5. Acceleration check distance is (available runway length 7,050 ft.—5,150 ft.) . . . . . .1,900 ft. from brake release.

6. Enter Figure A2-4 (Velocity During Take-off Ground Run) at the gross weight (19,270 lb.) and zero wind take-off ground run distance (3,875 ft.) to establish the normal acceleration guide line.

7. Using Figure A2-4 (Velocity During Take-off Ground Run) enter ground run distance scale at acceleration check distance (1,900 ft.) and where it intersects the normal acceleration guide line, read normal speed at acceleration distance . . . 135 knots.

8. From Figure A2-1 at a reported wind velocity of 15 knots and an angle between wind and runway of 45°–30° or 15 degrees, the runway component of the wind is . . . . . .14.5 knots.

9. Runway wind component (14.5 knot headwind) added to normal speed at acceleration distance gives final normal acceleration check speed at acceleration check marker . . . . . .149.5 knots.

the take-off prior to reaching the go no-go marker. This preliminary check should normally be made at the marker 2000 feet short of the go no-go marker. The distance traveled in reaching this marker is called the acceleration check distance.

For conditions where the normal speed (corrected for wind component) at this marker is less than the minimum range of the airspeed indicator, make the accelera-

tion check at the marker that is 1000 feet short of the go no-go marker. For light weight operation it is the distance to the second runway marker from brake release.

ACCELERATION CHECK SPEED.

The normal speed at the acceleration check marker may be found from the velocity during take-off ground run curve at the point where the normal acceleration curve crosses the acceleration check distance line. Upon reaching the acceleration check marker the speed of the aircraft should be compared with the predicted normal speed at the check marker. If the speed of the aircraft is lower than the normal speed at the acceleration check marker, it is possible that the take-off may have to be aborted and it will be absolutely necessary that a careful speed check be made upon reaching the go no-go marker.

TOTAL DISTANCE TO CLEAR A 50-FOOT OBSTACLE.

The total distance to clear a 50-foot obstacle is the sum of the ground run distance and the air distance traveled while attaining a height of 50-foot above the runway. Total distance charts are provided for Maximum and Military Thrust operation on Figures A2-5 and A2-10. Distances are shown for a range of altitude, temperature, wind, and slope conditions. The obstacle speeds are also shown on the charts. The total distance to clear a 100-foot obstacle may be approximately obtained by multiplying the *AIR* distance to the 50-foot obstacle by 1.5 and adding to this the ground run distance.

RUNWAY WIND COMPONENT.

Normal acceleration speeds at the go no-go point and the acceleration check point must be corrected for headwind. The correction is made by adding runway headwind component directly to the no-wind speeds. Figure A2-1 presents a chart for computing runway wind component. This chart may also be used to obtain crosswind component speeds. Normal take-off planning procedure should ignore the effect of headwind on take-off 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 wind conditions are not excessive.

RUNWAY MARKING SYSTEM.

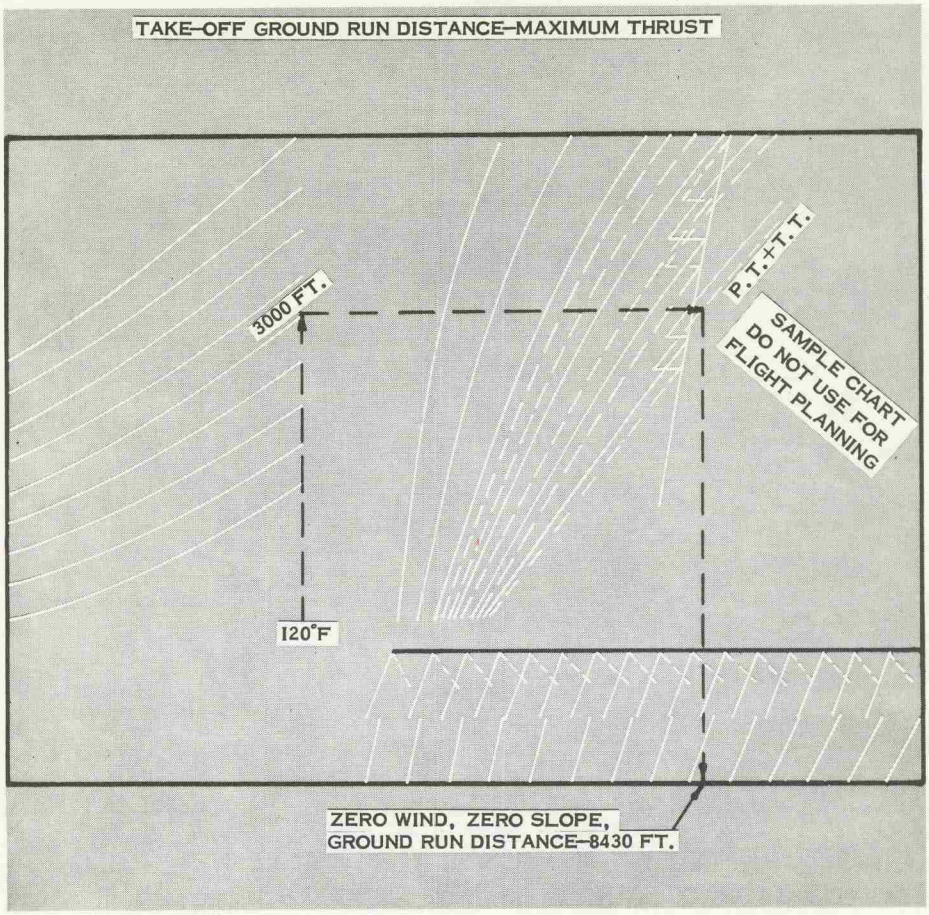
The take-off planning procedure presented in this handbook is based on the "distance remaining" marker concept. This criteria for numbering and placement of runway distance markers to reflect distance remaining, as established by current Air Force documents, is illustrated in the diagram presented on page A2-3. The markers are placed 1000 feet apart along the runway and are numbered to reflect the number of thousand foot intervals remaining. If the runway length is an odd figure (i.e., 7500 ft.) then the last or zero distance marker will be one-half of the odd figure over exact thousands of feet [½x(7500-7000)] from the end of the runway. Therefore, at any given marker, say No. 5, the aircraft will be 5000 feet plus one-half the odd figure over exact thousands of feet from the end of the runway.

LINE-UP ALLOWANCE.

The distance that the aircraft has traveled at any particular marker depends on the length of the runway and the position of the aircraft on the runway when the brakes were released. The distance from the starting end of the runway at brake release to the aircraft is called the line-up allowance. If the aircraft is 250 feet on a 7500 foot runway, brakes would be released at the No. 7 marker and the airplane would have traveled 2000 feet at the No. 5 marker.

TAKE-OFF PLANNING PROBLEM.

The procedures for take-off planning are presented by two sample problems. The methods used for solution of the example apply to the actual performance charts discussed. Reference sample charts (mentioned in parenthesis) are provided for illustration purposes only and are not to be used for actual performance calculations.



COMPLETE TAKE-OFF ANALYSIS SAMPLE PROBLEM.

Determine take-off performance for the following conditions:

- Maximum Thrust*
- Configuration—Pylon Tanks + Tip Tanks
- Gross weight—24,475 pounds*
- Field Pressure Altitude—3,000 feet.*
- Field ambient air temperature—120°F.
- Wind—20 knots from 060 degrees.
- Runway—10,250 feet long, 035 degree heading, dry, hard surface, 1% downhill slope.
- Line-up allowance—250 feet from end of runway.

*Enter on take-off data card.

1. From Figure A2-2 (Take-off Ground Run Distance) obtain the zero wind, zero slope ground run distance, for use in determining the go no-go requirements . . . . . 8,430 ft.*

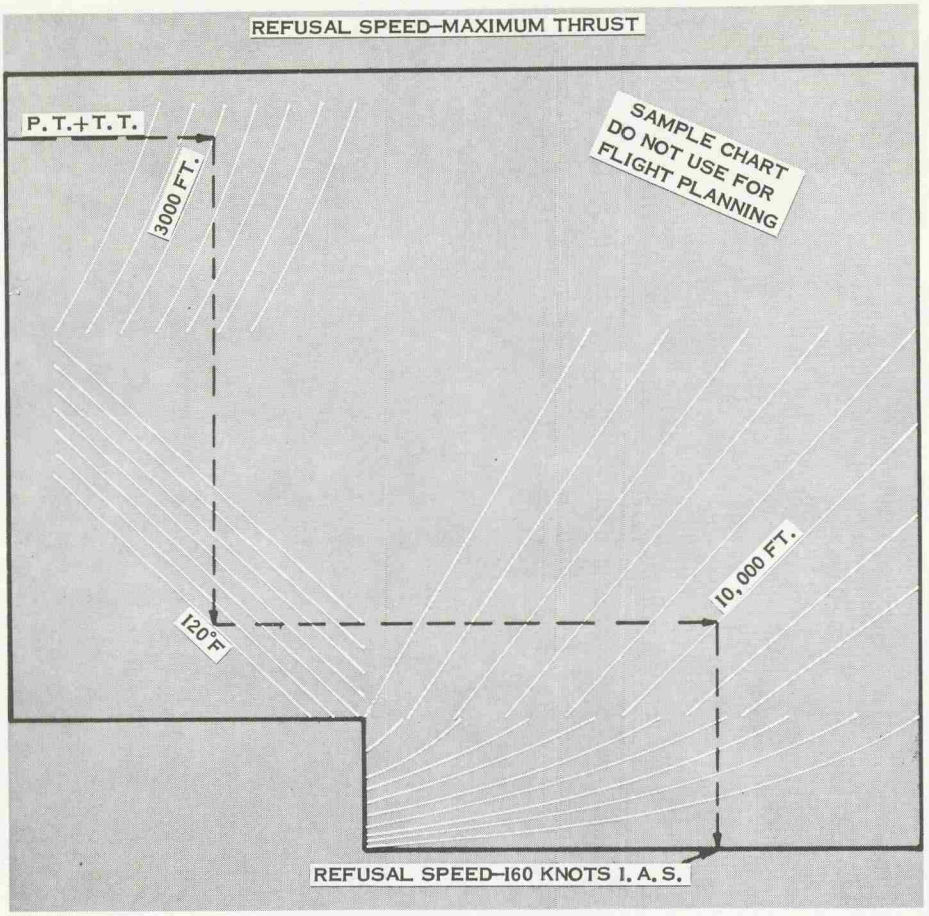
NOTE

Use Maximum Performance Take-Off Speed schedule.

2. Available runway length is the runway length less the line-up allowance 10,250 ft.* −250 ft. . . . . 10,000 ft.

3. The refusal speed is determined from Figure A2-3 (Refusal Speed) . . . . . 160 knots

4. Enter Figure A2-4 (Velocity During Take-off Ground Run) at the take-off gross weight (24,476 lb.) and take-off ground run distance (8,430 ft.) to establish the normal acceleration guide line.



Sample Problem Continued

5. Using the same chart as in item 4, enter (from the bottom) at the refusal speed (160 knots). At the intersection of the refusal speed and the normal acceleration guide line read refusal distance . . . . . 4,500 ft.

6. The runway distance remaining from the refusal point is the available runway length less the refusal distance (10,000 ft.—4,500 ft.) . . . 5,500 ft.

7. The go no-go marker is the closest marker where at least the distance found in item 6 remains to the end of the runway . . . . . 6,000 ft. remaining marker.

8. One-half of the odd number of feet over the exact thousand feet of runway length (125 ft.), added to the go no-go marker (6,000 ft.) determines actual distance remaining from go no-go marker to the end of runway . . . . . 6,125 ft.

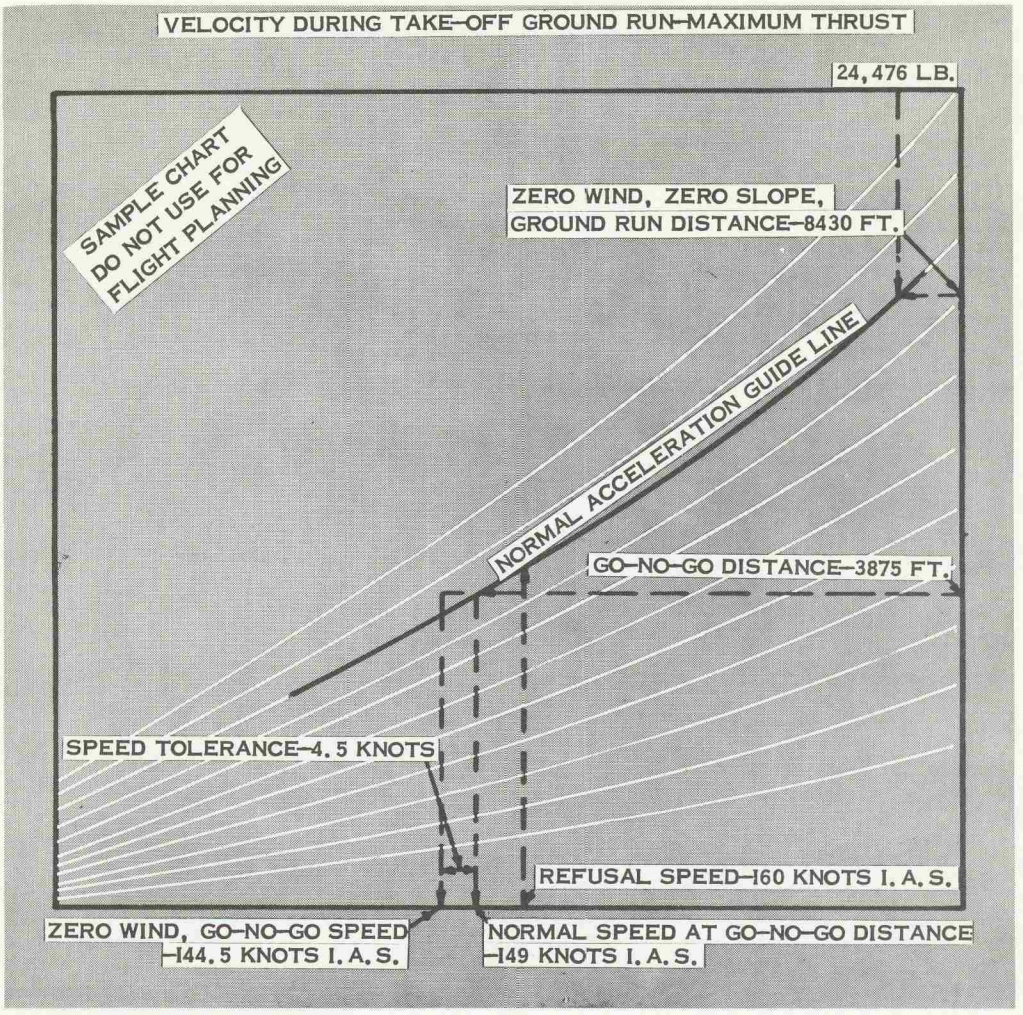
9. The go no-go distance is the distance remaining at the go no-go marker subtracted from the available runway length (10,000 ft.—6,125 ft.) . . . . . 3,875 ft.

10. Using Figure A2-4 (Velocity During Take-Off Ground Run) enter at the go no-go distance (3,875 ft.) and where this distance intersects the normal acceleration guide line read down to find the normal speed at the go no-go distance . . . . 149 knots.

11. From Figure A2-6, the allowable speed tolerance is 45 knots. The zero wind go no-go speed is (149 knots—4.5 knots speed tolerance. . . . . 144.5 knots.

12. From Figure A2-1, at a reported wind velocity 20 knots and an angle between wind and runway of (60° —35°) 25 degrees, the runway component of the wind is . . . . . 18 knot headwind.

13. The final go no-go speed, corrected for headwind is (144.5 knots + 18 knots) . . . . 162.5 knots.* If the speed of the airplane is below 162.5 knots at the go no-go marker (6,000 ft. marker), the take-off



Sample Problem Continued

should be aborted, using maximum wheel braking and drag chute.

14. The acceleration check marker is (1,000 ft. short of the go no-go marker) at the 7,000 ft. remaining marker.

15. One-half of the odd number of feet over exact thousand feet of runway length, added to distance in item 14 is the distance to the end of the runway from acceleration check marker (125 ft. +7,000 ft.) . . . . . .7,125 ft.

16. Acceleration check distance is (available runway length 10,000 ft.—7,125 ft. distance in item 15) . . . . . .2,875 ft. distance from brake release.

17. Enter Figure A2-4 (Velocity During Take-off Ground Run) at ground run distance 2,875 ft. and where it intersects the normal acceleration guide line read normal speed at acceleration distance (zero wind) . . . . . .128 knots.

18. Runway wind component (18 knot headwind) added to normal speed at acceleration distance gives final normal acceleration check speed at acceleration check marker . . . . . .146 knots. When the acceleration check speed is marginal, the speed at the go no-go point should be given close attention.

19. Referring to Figure A2-5, the total distance to take-off and clear a 50-foot obstacle, based on normal acceleration, zero wind, and zero slope is . . . . . .17,300 ft.

20. If the acceleration of the aircraft is normal, assuming an 18 knot headwind (runway component) and a 1% downhill slope, the take-off ground run distance will be (Figure A2-2) . . . . . .7,300 ft.

21. Normal distance to take-off and clear a 50-foot obstacle for the 18 knot headwind (runway component) and 1% downhill slope (Figure A2-5). . . . . .15,000 ft.

WIND COMPONENT CHART

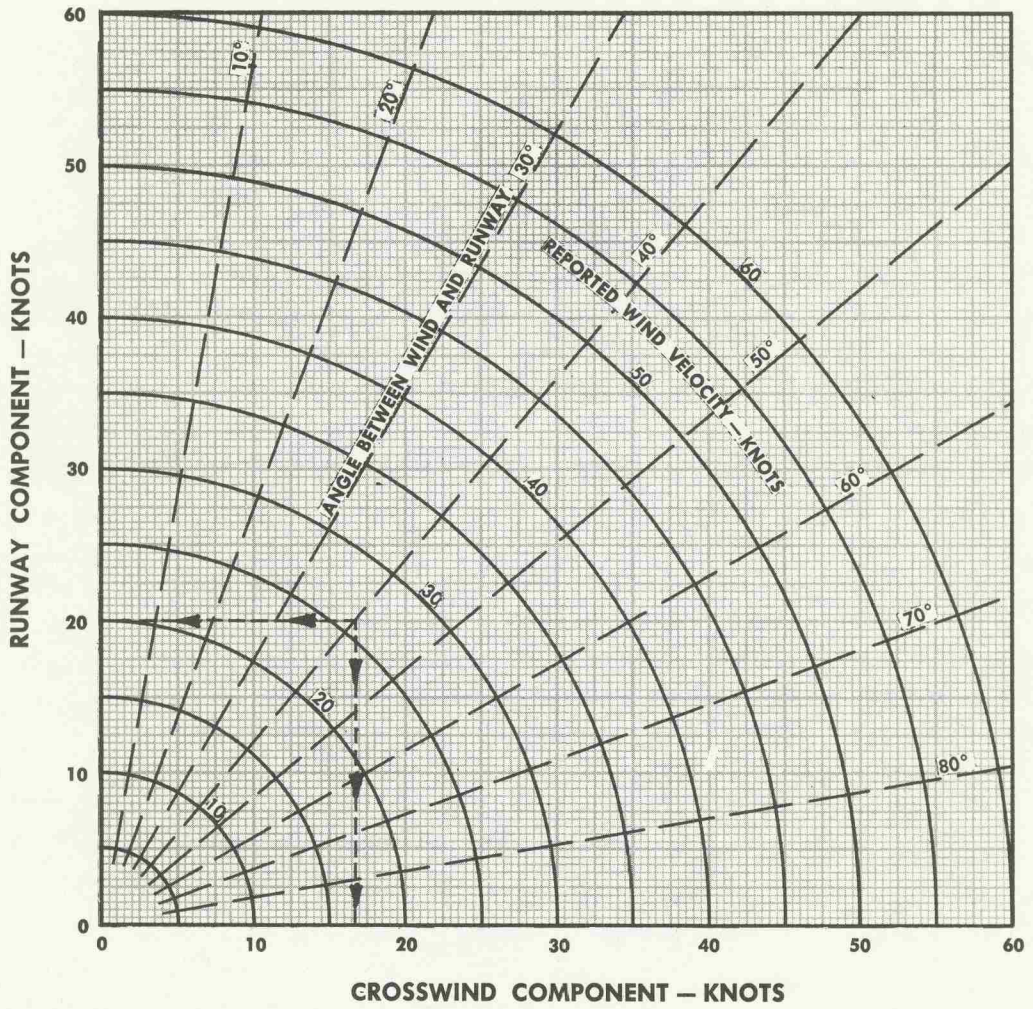


Figure A2-1

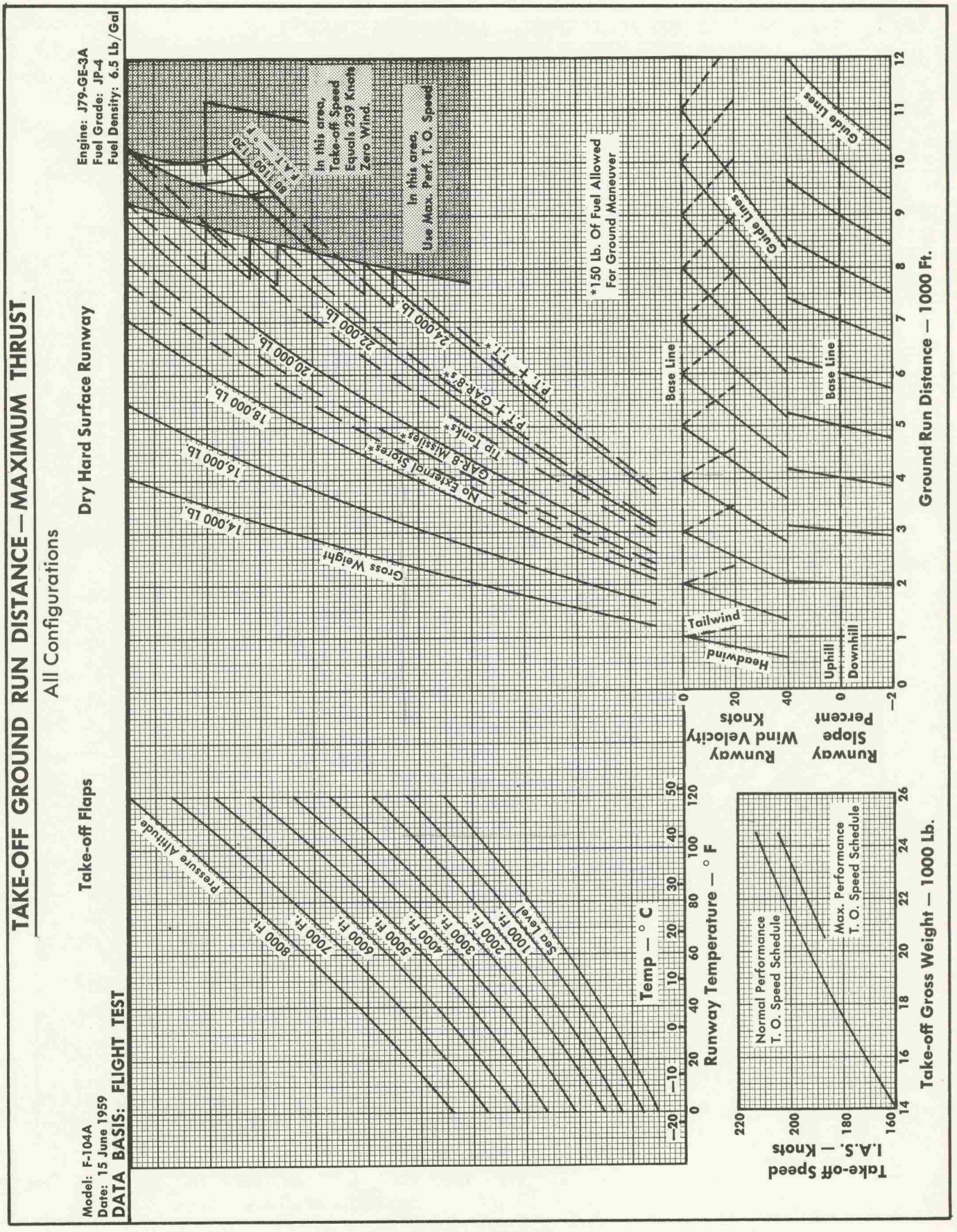

TAKE-OFF GROUND RUN DISTANCE—MAXIMUM THRUST
All Configurations
Take-off Flaps
Dry Hard Surface Runway
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Gross Weight
14,000 Lb.
16,000 Lb.
18,000 Lb.
20,000 Lb.
22,000 Lb.
24,000 Lb.
Pressure Altitude
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
5000 Ft.
6000 Ft.
7000 Ft.
8000 Ft.
Temp — °C
Runway Temperature — °F
No External Stores*
GAR-8 Missiles*
Tip Tanks*
P.T. + GAR-8's*
P.T. + T.T.*
*150 Lb. Of Fuel Allowed For Ground Maneuver
In this area, Take-off Speed Equals 239 Knots Zero Wind.
In this area, Use Max. Perf. T.O. Speed
Base Line
Guide Lines
Runway Slope Percent
Uphill
Downhill
Runway Wind Velocity Knots
Headwind
Tailwind
Ground Run Distance — 1000 Ft.
Take-off Speed I.A.S. — Knots
Normal Performance T. O. Speed Schedule
Max. Performance T. O. Speed Schedule
Take-off Gross Weight — 1000 Lb.

Figure A2-2

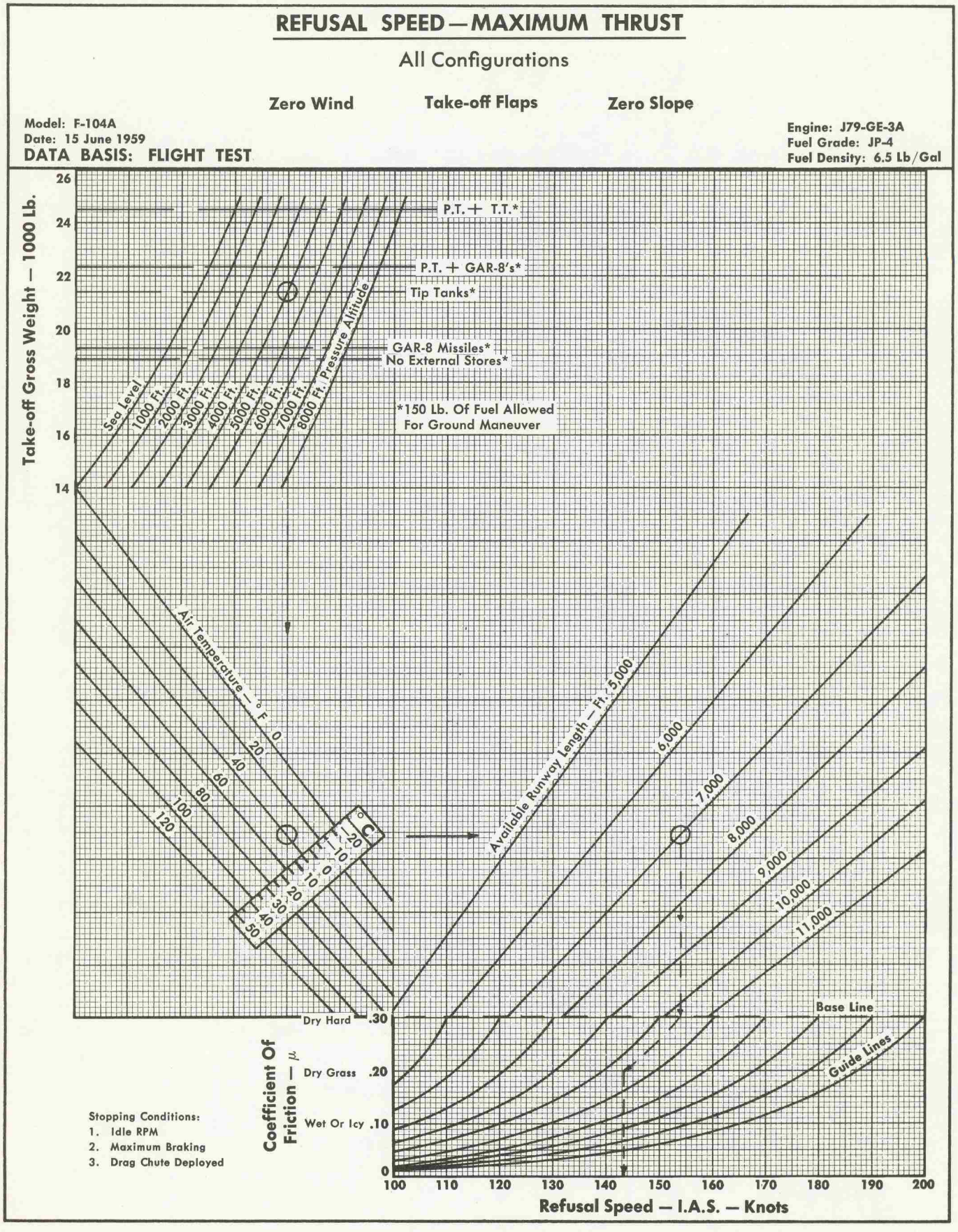

REFUSAL SPEED—MAXIMUM THRUST
All Configurations
Zero Wind
Take-off Flaps
Zero Slope
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Take-off Gross Weight — 1000 Lb.
P.T. + T.T.*
P.T. + GAR-8's*
Tip Tanks*
GAR-8 Missiles*
No External Stores*
*150 Lb. Of Fuel Allowed For Ground Maneuver
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
5000 Ft.
6000 Ft.
7000 Ft.
8000 Ft. Pressure Altitude
Air Temperature — °F
Available Runway Length — Ft. 5,000
6,000
7,000
8,000
9,000
10,000
11,000
Base Line
Guide Lines
Coefficient Of Friction — μ
Dry Hard .30
Dry Grass .20
Wet Or Icy .10
Stopping Conditions:
1. Idle RPM
2. Maximum Braking
3. Drag Chute Deployed
Refusal Speed — I.A.S. — Knots

Figure A2-3

VELOCITY DURING TAKE-OFF GROUND RUN—MAXIMUM THRUST

All Configurations

Zero Wind **Take-off Flaps** **Zero Slope**

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

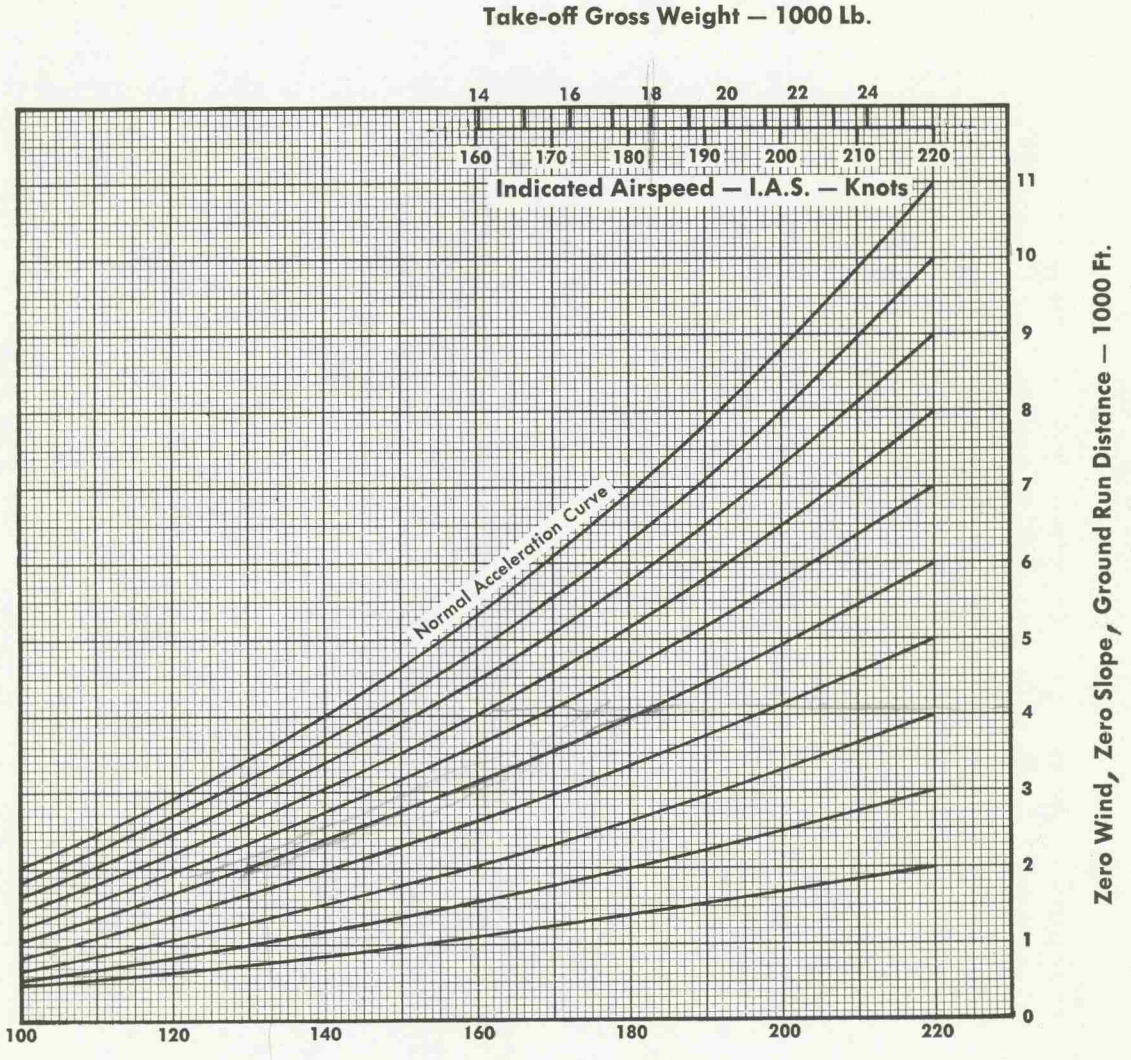


Figure A2-4

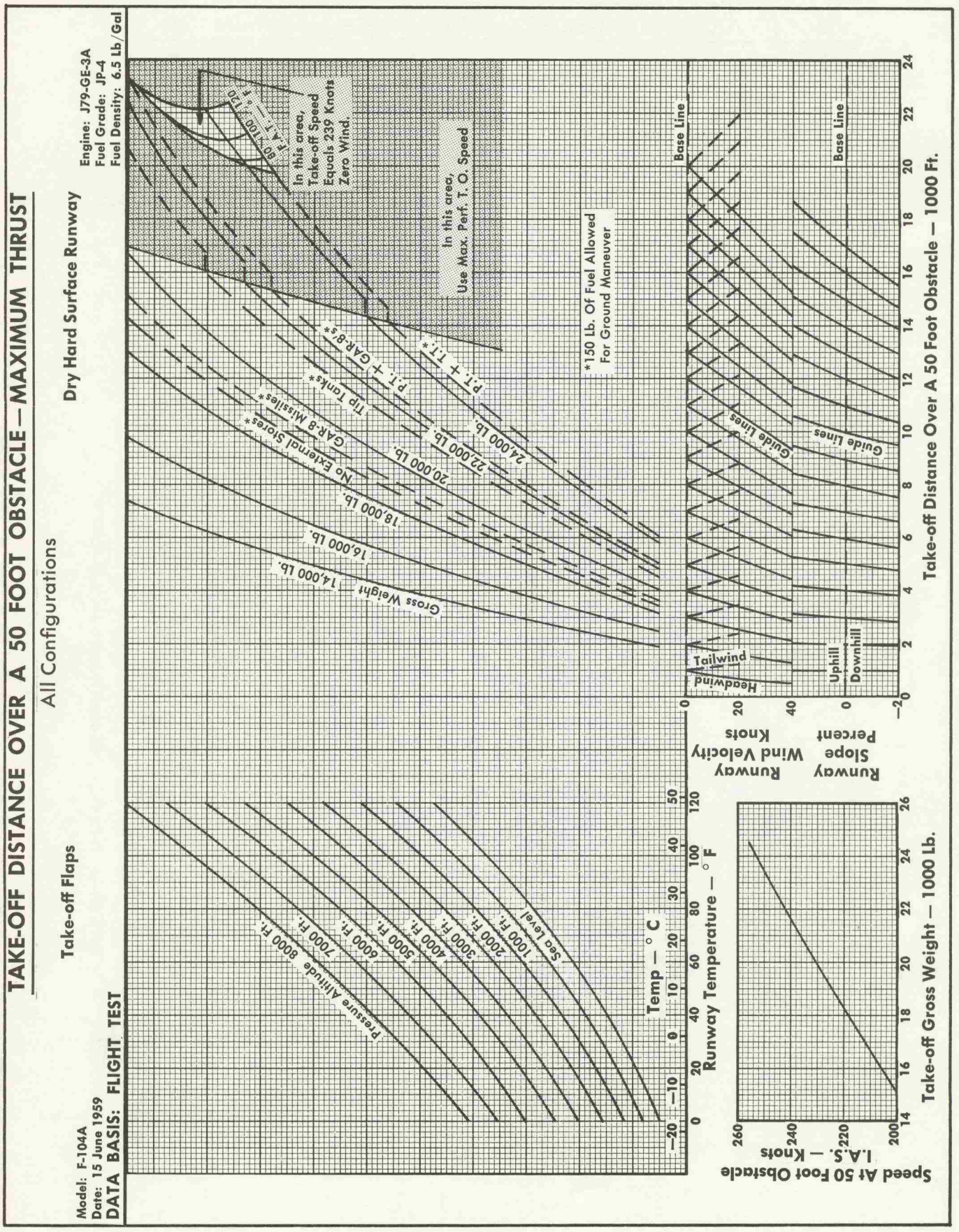

TAKE-OFF DISTANCE OVER A 50 FOOT OBSTACLE—MAXIMUM THRUST
All Configurations
Take-off Flaps
Dry Hard Surface Runway
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
DATA BASIS: FLIGHT TEST
Model: F-104A
Date: 15 June 1959
In this area, Take-off Speed Equals 239 Knots Zero Wind.
In this area, Use Max. Perf. T.O. Speed
*150 Lb. Of Fuel Allowed For Ground Maneuver
Gross Weight — 14,000 lb.
16,000 lb.
18,000 lb.
No External Stores*
GAR-8 Missiles*
Tip Tanks*
P.T. + GAR-8's*
20,000 lb.
22,000 lb.
24,000 lb.
P.T. + T.T.*
Pressure Altitude
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
5000 Ft.
6000 Ft.
7000 Ft.
8000 Ft.
Temp — °C
Runway Temperature — °F
Take-off Gross Weight — 1000 Lb.
Speed At 50 Foot Obstacle I.A.S. — Knots
Runway Slope Percent
Runway Wind Velocity Knots
Headwind
Tailwind
Uphill
Downhill
Base Line
Guide Lines
Take-off Distance Over A 50 Foot Obstacle — 1000 Ft.

Figure A2-5

AIRSPEED ACCELERATION TOLERANCE

All Configurations

Hard Surface Level Runway

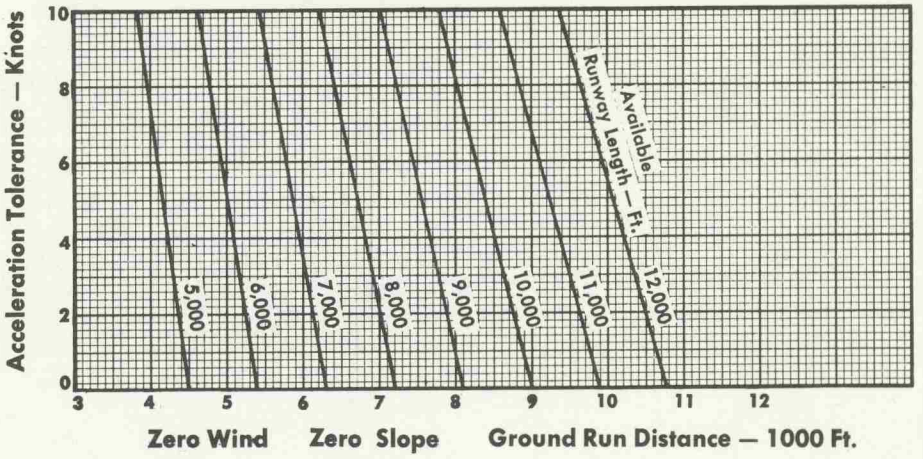
Zero Wind

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Take-off Flaps

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

MAXIMUM THRUST



MILITARY THRUST

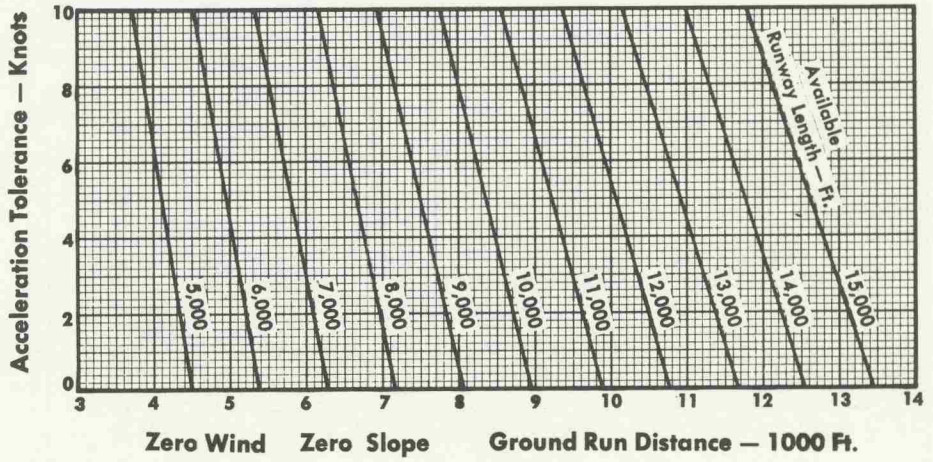


Figure A2-6

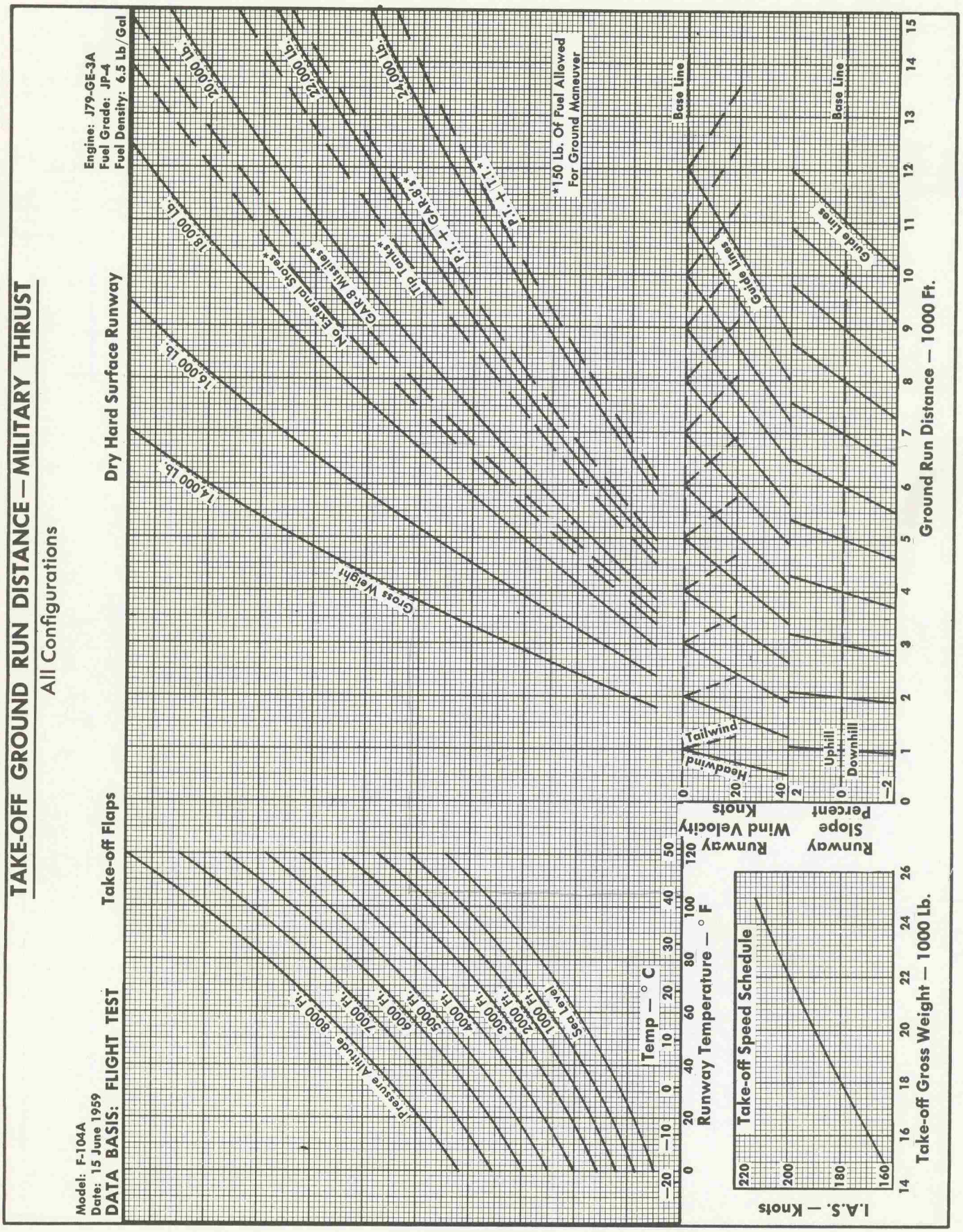

TAKE-OFF GROUND RUN DISTANCE—MILITARY THRUST
All Configurations
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Take-off Flaps
Dry Hard Surface Runway
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
*150 Lb. Of Fuel Allowed For Ground Maneuver
Gross Weight
14,000 Lb.
16,000 Lb.
18,000 Lb.
20,000 Lb.
22,000 Lb.
24,000 Lb.
No External Stores*
GAR-8 Missiles*
ITP Tanks*
P.T. + GAR-8's*
P.T. + T.T.*
Pressure Altitude
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
5000 Ft.
6000 Ft.
7000 Ft.
8000 Ft.
Temp — °C
Runway Temperature — °F
Runway Wind Velocity
Knots
Headwind
Tailwind
Runway Slope
Percent
Uphill
Downhill
Base Line
Guide Lines
Ground Run Distance — 1000 Ft.
Take-off Speed Schedule
I.A.S. — Knots
Take-off Gross Weight — 1000 Lb.

Figure A2-7

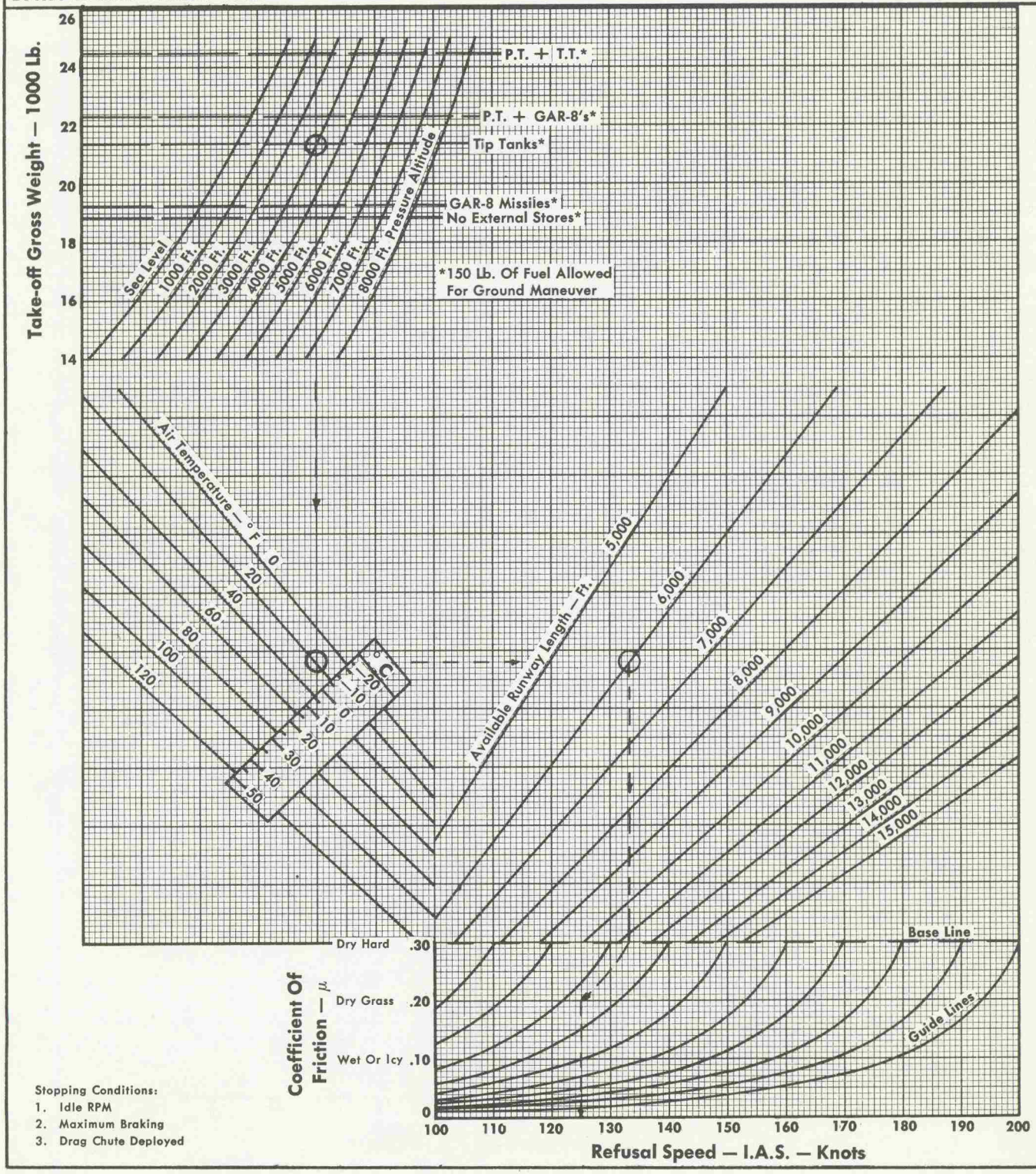
REFUSAL SPEED—MILITARY THRUST

All Configurations

Zero Wind **Take-off Flaps** **Zero Slope**

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



Stopping Conditions:
1. Idle RPM
2. Maximum Braking
3. Drag Chute Deployed

Figure A2-8

VELOCITY DURING TAKE-OFF GROUND RUN—MILITARY THRUST

All Configurations

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Zero Wind Take-off Flaps Zero Slope

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

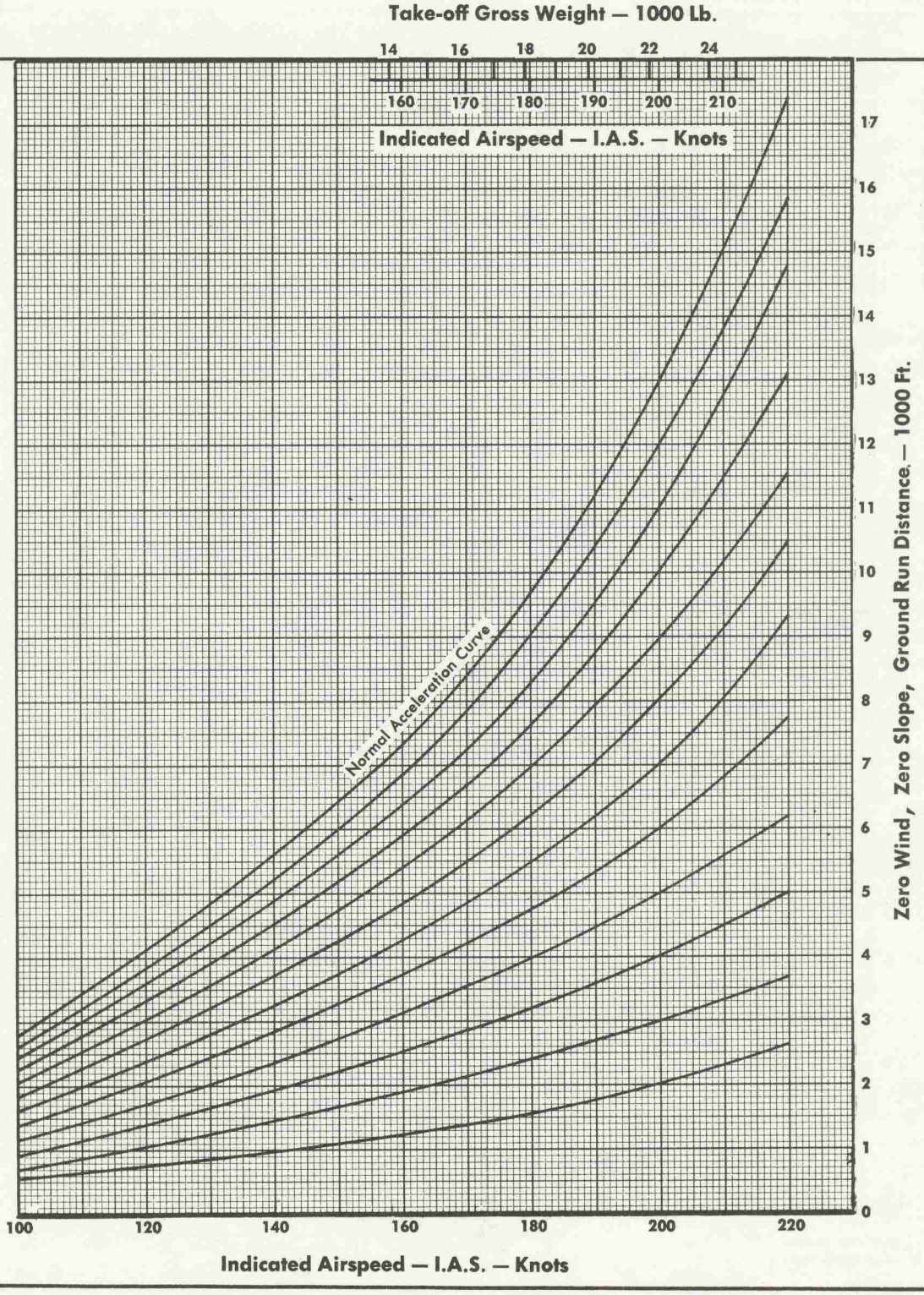


Figure A2-9

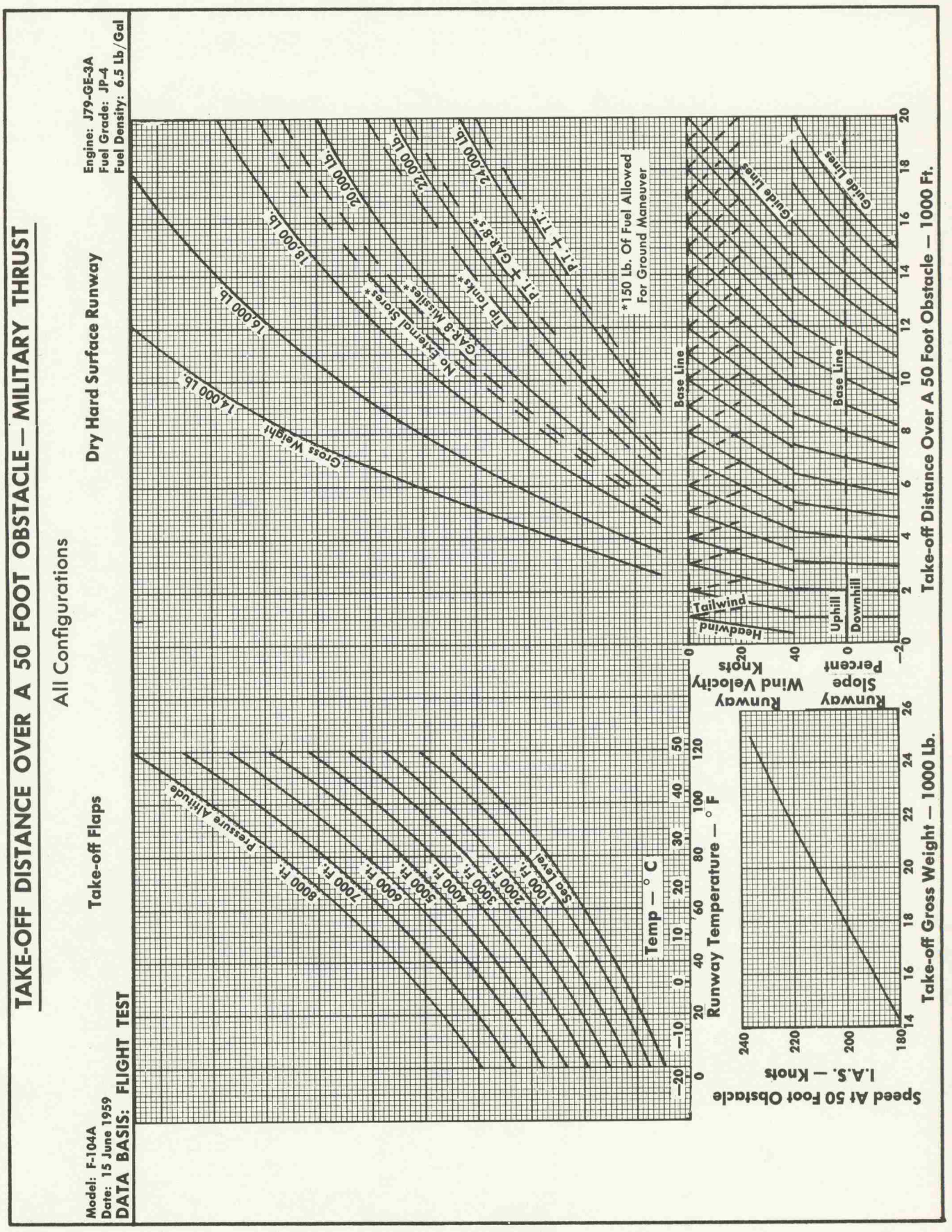

TAKE-OFF DISTANCE OVER A 50 FOOT OBSTACLE— MILITARY THRUST
All Configurations
Take-off Flaps
Dry Hard Surface Runway
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Gross Weight
14,000 Lb.
16,000 Lb.
18,000 Lb.
20,000 Lb.
22,000 Lb.
24,000 Lb.
Pressure Altitude
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
5000 Ft.
6000 Ft.
7000 Ft.
8000 Ft.
Temp — °C
Runway Temperature — °F
P.I. + T.T.*
P.I. + GAR-8's*
Tip Tanks*
No External Stores*
GAR-8 Missiles*
*150 Lb. Of Fuel Allowed For Ground Maneuver
Speed At 50 Foot Obstacle I.A.S. — Knots
Take-off Gross Weight — 1000 Lb.
Runway Slope Percent
Runway Wind Velocity Knots
Headwind
Tailwind
Uphill
Downhill
Base Line
Guide Lines
Take-off Distance Over A 50 Foot Obstacle — 1000 Ft.

Figure A2-10

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Part 3 – Climb

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Page numbers in bold face denote illustrations.

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Supersonic Climb	A3-2
Fuel Allowances	A3-3
Climb Speed Schedule	A3-3
Effect of Free Air Temperature on Climb Performance	A3-3
Determination of Equivalent Weight	A3-3
Use of Charts	A3-3
Time and Fuel Used From Brake Release to Climb Speed Maximum and Military Thrust	**A3-5**
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Military Thrust Climb Performance at Mach 0.88	**A3-6**
Military Thrust Rate of Climb at Mach 0.88	**A3-7**
Maximum Thrust Climb Performance at Mach 0.925	**A3-8**
Maximum Thrust Rate of Climb at Mach 0.925	**A3-9**
Maximum Thrust Climb Performance at Mach 1.6	**A3-10**
Maximum Thrust Climb Performance at Mach 1.7	**A3-11**
Maximum Thrust Rate of Climb at Mach 1.7	**A3-12**
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Maximum Thrust Climb Performance at Mach 1.9	**A3-14**
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Part 3 – Climb

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Maximum Thrust Climb Performance at Mach 1.7	**A3-21**
Maximum Thrust Rate of Climb at Mach 1.7	**A3-22**
Maximum Thrust Climb Performance at Mach 1.8	**A3-23**
Maximum Thrust Climb Performance at Mach 1.9	**A3-24**
Combat and Service Ceiling Chart	**A3-25**
Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles, or Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks	
Military Thrust Climb Performance at Mach 0.825	**A3-26**
Military Thrust Rate of Climb at Mach 0.825	**A3-27**
Maximum Thrust Climb Performance at Mach 0.90	**A3-28**
Maximum Thrust Rate of Climb at Mach 0.90	**A3-29**
Combat and Service Ceiling Chart	**A3-30**

SUBSONIC CLIMB.

Subsonic standard day climb performance is shown in this Section for Military and Maximum Thrust operation without external stores, with two GAR-8 missiles or 165 gallon tip tanks, and with pylon tanks and two GAR-8 missiles, or pylon tanks and tip tanks installed. Optimum climb speeds for these configurations are listed below and on the appropriate charts. The climb performance charts show distance and time to climb versus gross weight for altitudes ranging from sea level to the combat ceiling (altitude for 500 feet per minute rate of climb). Guide lines indicate aircraft weight reduction in climb due to fuel consumption. Rate of climb and climb fuel flow data can be obtained for any point during climb to altitude by use of the constant weight rate of climb charts which are also presented in this section. Since the amount of fuel used and time to take-off and accelerate to climb speed are critical for accurate flight planning, this information is presented for a complete range of take-off gross weights. Summaries provide combat and service ceilings versus gross weight at optimum climb speeds for the two climb power settings.

SUPERSONIC CLIMB.

Presentations similar to those described above are given to provide supersonic climb performance for Maximum Thrust operation in the no-external-stores and GAR-8 missile configurations. Climb performance data are pro-

vided from a base altitude of 35,000 feet to the combat ceiling for climb speeds from Mach 1.6 to 1.9. Constant weight rate of climb information is presented for the Mach 1.7 speed schedule.

FUEL ALLOWANCES.

Information regarding fuel used and time to accelerate to climb speed from brake release is presented for all configurations on Figure A3-1. A note is included which indicates the allowance that should be added to the fuel consumed to allow for ground maneuvers. The value shown is based on flight test data. It may be necessary to adjust this value for abnormal periods of ground operation, or if service experience indicates that some other value may be more applicable. Time and fuel used to 350 KIAS are also indicated on the chart. This provides data for time and fuel used from brake release to climb speed for any combination of Maximum or Military Thrust when the thrust setting is changed at 350 KIAS.

CLIMB SPEED SCHEDULE.

The Climb Speed Schedule noted on each chart will provide the climb performance shown. The speeds are given in terms of true and indicated Mach Number. Other climb techniques, such as flying at constant IAS until the recommended climb speed schedule is reached, will not appreciably affect the time, distance and fuel used.

	Military Thrust	Maximum Thrust
No External Stores at Mach	0.88	0.925
GAR-8 Missiles or tip tank at Mach	0.88	0.925
Pylon Tanks and GAR-8 Missiles or Pylon Tanks and tip tanks at Mach	0.825	0.90

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 if it weighs more than it actually does. When the free air temperature is lower than standard (a cold day) the aircraft will perform as if it weighs less than it actually does. Use of an "equivalent weight" permits utilization of the standard day charts under non-standard temperature conditions. Substitute an equivalent weight number for the actual airplane gross weight at start of climb when reading the climb charts. Equivalent weight is obtained by applying a weight increment ($\triangle$ Weight) to actual airplane gross weight. A $\triangle$ Weight versus $\triangle$ Temperature curve is provided on each climb performance chart, and gives the weight for any $\triangle$ Temperature up to 40°F deviation from standard.

DETERMINATION OF EQUIVALENT WEIGHT.

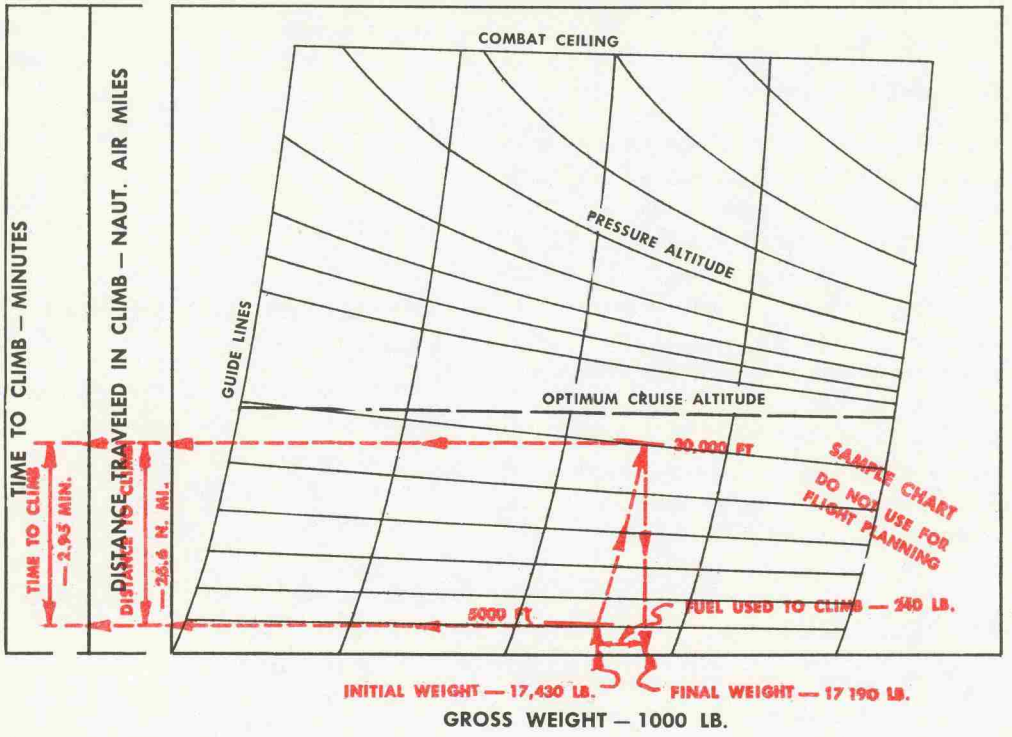
The procedure for determining equivalent weight is as follows:

1. Determine the average O.A.T. (ambient air temperature) during climb to altitude from meteorological forecasts.
2. Enter the Standard Altitude Table (Figure A1-8) at the average pressure altitude during climb and obtain the standard day temperature.
3. Determine the difference between the standard day temperature (Step 2) and the O.A.T. (Step 1). This is the $\triangle$ Temperature.
4. Using the $\triangle$ Weight versus $\triangle$ Temperature curve with the applicable climb performance chart, enter the chart at the $\triangle$ Temperature from Step 3 and read the $\triangle$ Weight.
5. Determine the equivalent gross weight by using the suitable non-standard temperature equation:
   a. Hot day: Equivalent weight = actual airplane gross weight + $\triangle$ Weight.
   b. Cold Day: Equivalent weight = actual airplane gross weight − $\triangle$ Weight.
6. Enter the applicable rate of climb or climb performance chart using the equivalent weight calculated in Step 5.

USE OF CHARTS.

A sample problem illustrates how to use the climb performance charts: A "time and distance prediction" climb chart is shown in sample chart form to further illustrate their use.

MILITARY THRUST CLIMB PERFORMANCE AT MACH 0.88



SAMPLE PROBLEM.

Determine the standard day climb performance for a military thrust climb at Mach 0.88 at an initial airplane gross weight of 17,430 pounds from 5,000 feet to 30,000 fet. The airplane has no external stores. To determine distance, time, and fuel required for the climb:

1. Enter Figure A3-2 (Military Thrust Climb Performance at Mach 0.88) at the initial gross weight (17,430 lb.) and initial pressure altitude (5,000 ft.). Read:

Initial distance . . . . . 3.8 n.mi.
Initial time . . . . . 0.40 min.

2. Trace a curve parallel to the nearest guide line until it intersects the final altitude (30,000 ft.). Read:

Final distance . . . . . 29.4 n.mi.
Final time . . . . . 3.35 min.
Final gross weight . . . . . 17,190 lb.

3. Distance to climb is then
(29.4 n.mi. −3.8 n.mi.) or . . . . . 25.6 n.mi.

4. Time to climb equals
(3.35 min. −.40 min.) or . . . . . 2.95 min.

5. Fuel used to climb equals
(17,430 lb. −17,190 lb.) . . . . . 240 lb.

A correction for the effects of non-standard air temperatures is included. For example, if the average air temperature were predicted to be 20°F above standard during the climb, note that the equivalent weight correction is plus 2,120 lb. Enter the climb performance curve at an equivalent weight 2,120 pounds greater than the actual weight of 17,430 pounds or 19,550 pounds. Determine the hot day performance in the same manner as described above.

Though the sample problem did not specify a climb to optimum cruise altitude or combat ceiling, a continuation of the guide line traced on Figure A3-2 (Military Thrust Climb Performance at Mach 0.88) will provide the optimum cruise altitude and combat ceiling plus the resultant gross weight at each altitude. Read:

Optimum cruise altitude . . . . . 32,200 ft.
Gross weight . . . . . 17,130 lb.
Combat ceiling . . . . . 41,200 ft.
Gross weight at combat ceiling . . . . . 16,880 lb.

TIME AND FUEL REQUIRED FROM BRAKE RELEASE TO CLIMB SPEED

Standard Day **Zero Wind**

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

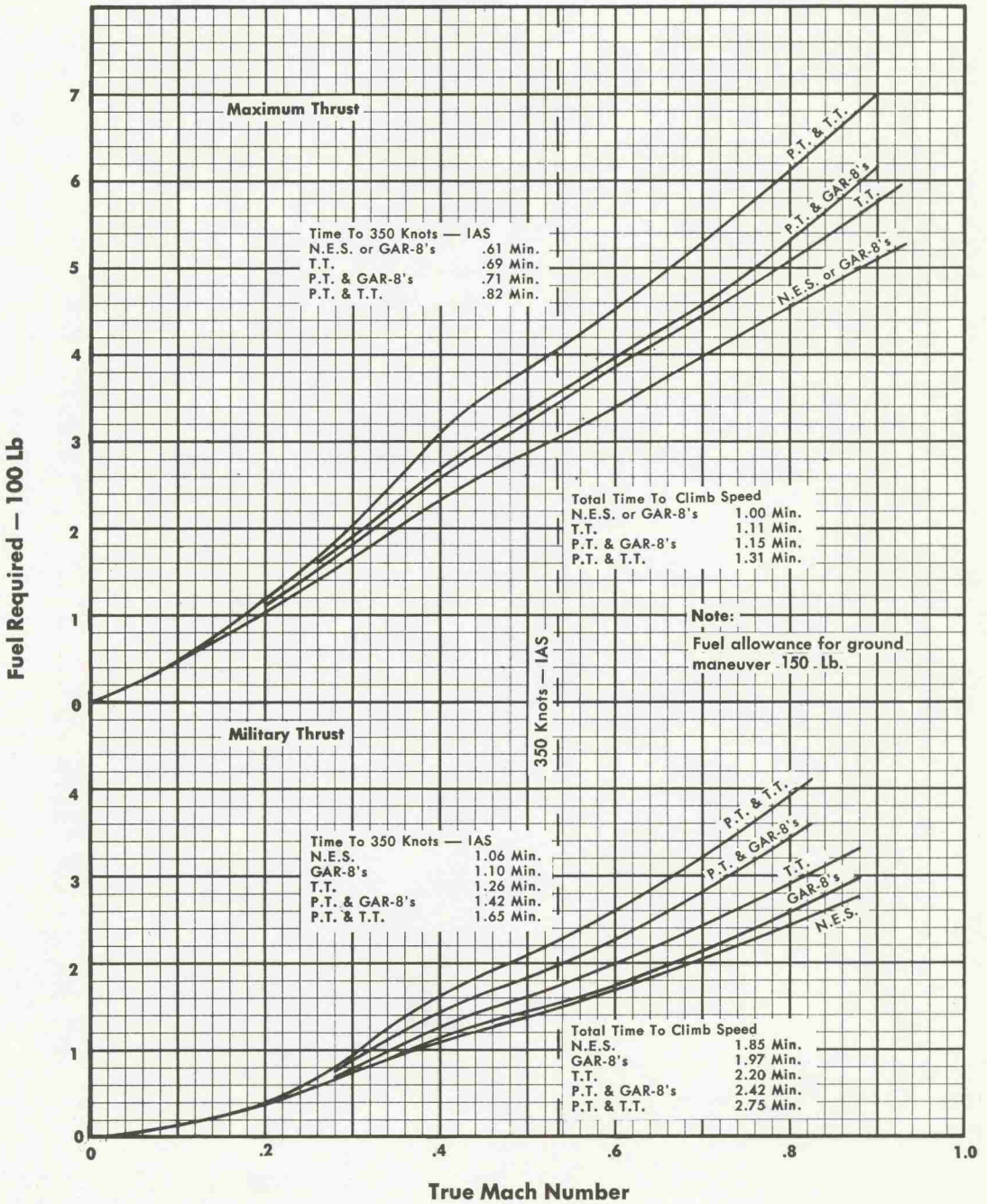
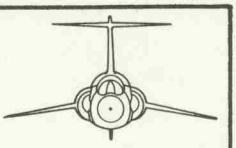


Figure A3-1

MILITARY THRUST CLIMB PERFORMANCE AT MACH 0.88



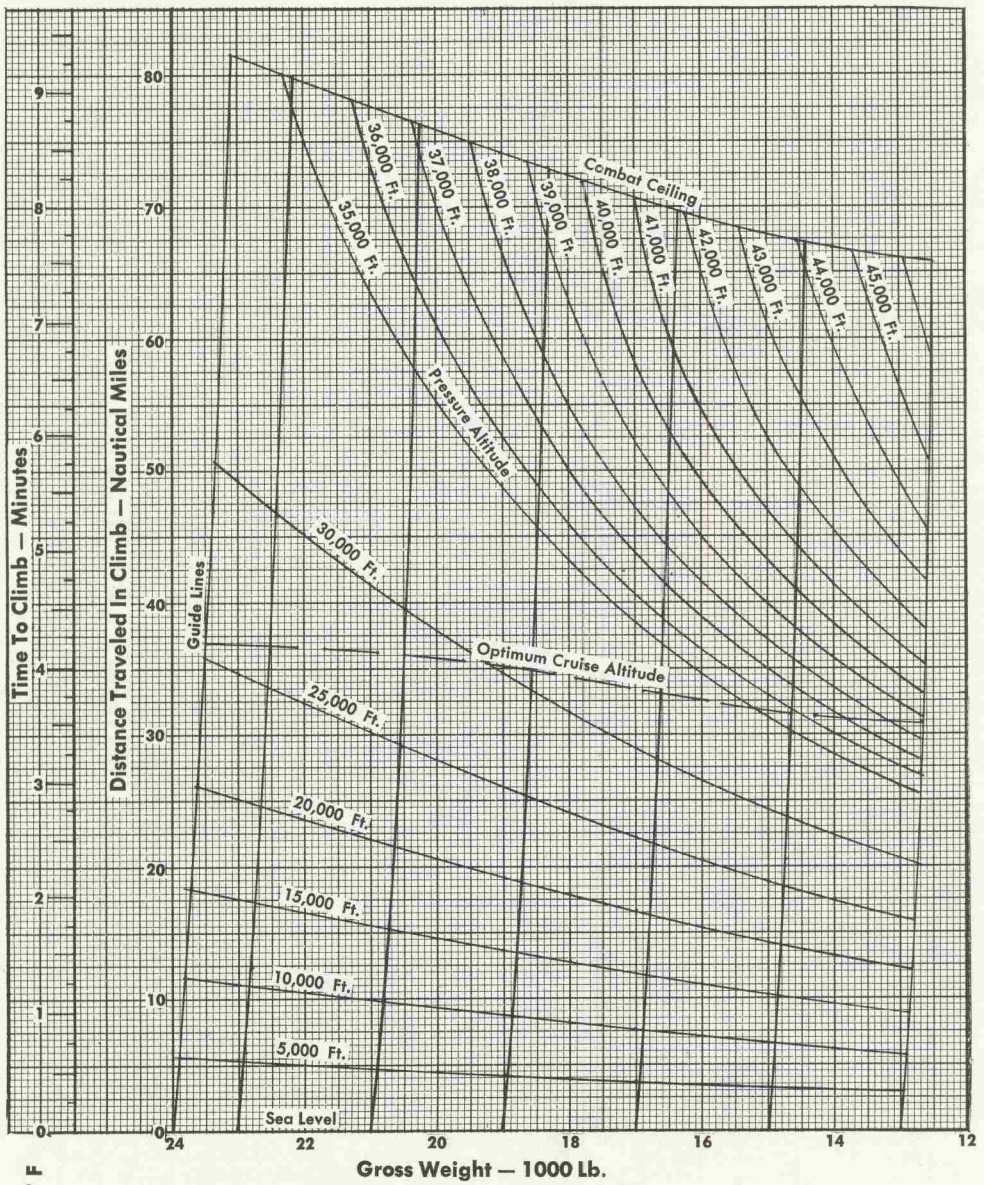
No External Stores

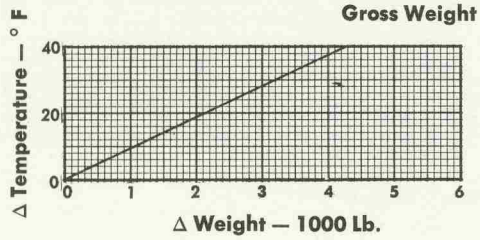
Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed — True Mach — 0.88
Indicated Mach — 0.86

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal





Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-2

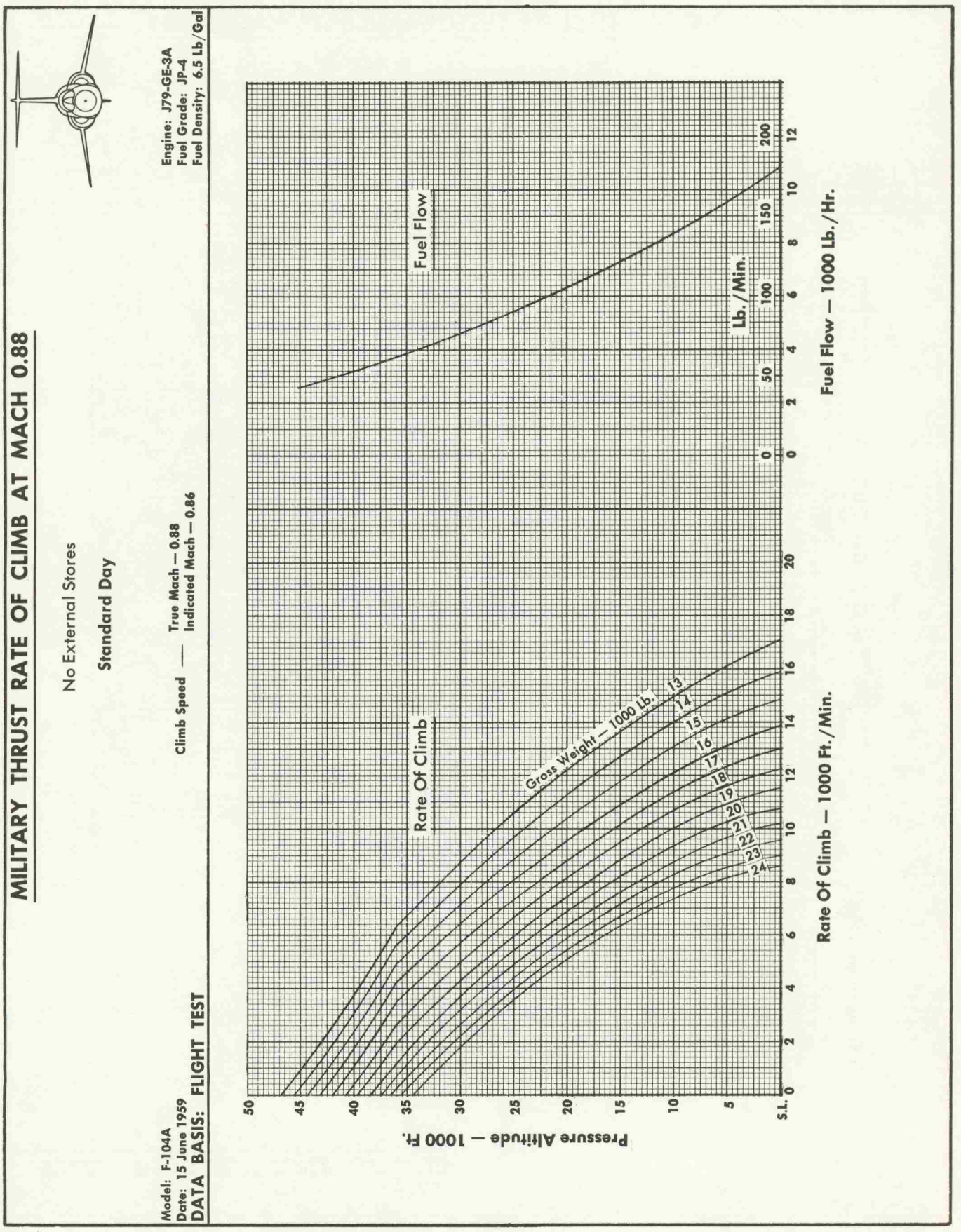
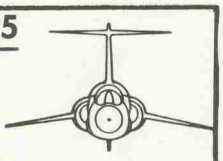

MILITARY THRUST RATE OF CLIMB AT MACH 0.88
No External Stores
Standard Day
Climb Speed — True Mach — 0.88
Indicated Mach — 0.86
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Rate Of Climb
Fuel Flow
Gross Weight — 1000 Lb.
13
14
15
16
17
18
19
20
21
22
23
24
Pressure Altitude — 1000 Ft.
S.L.
5
10
15
20
25
30
35
40
45
50
Rate Of Climb — 1000 Ft./Min.
Lb./Min.
0
50
100
150
200
Fuel Flow — 1000 Lb./Hr.
0
2
4
6
8
10
12
14
16
18
20

Figure A3-3

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 0.925



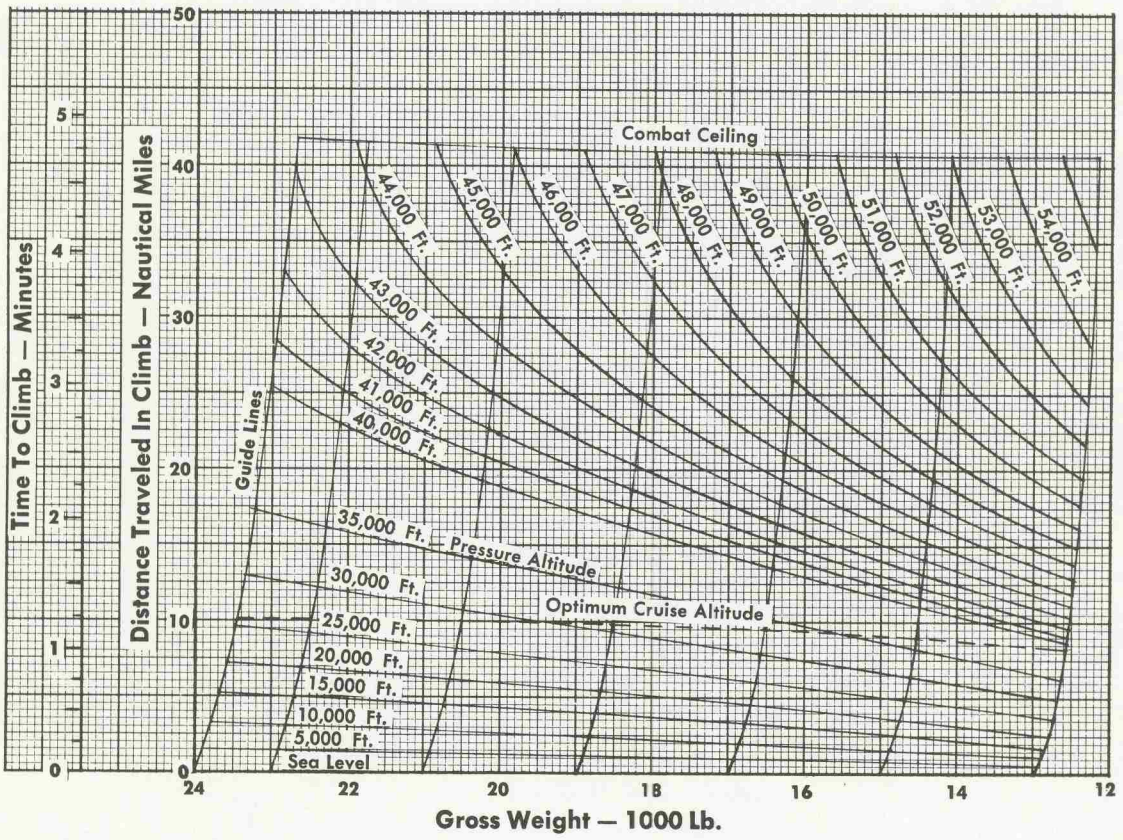
No External Stores

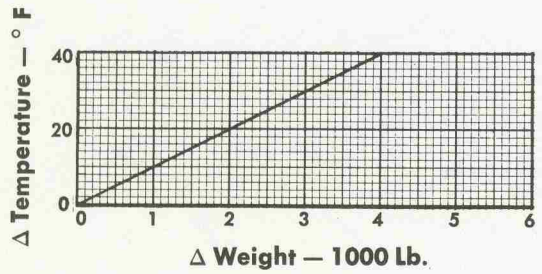
Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed— True Mach — 0.925
Indicated Mach — 0.900

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal





Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-4

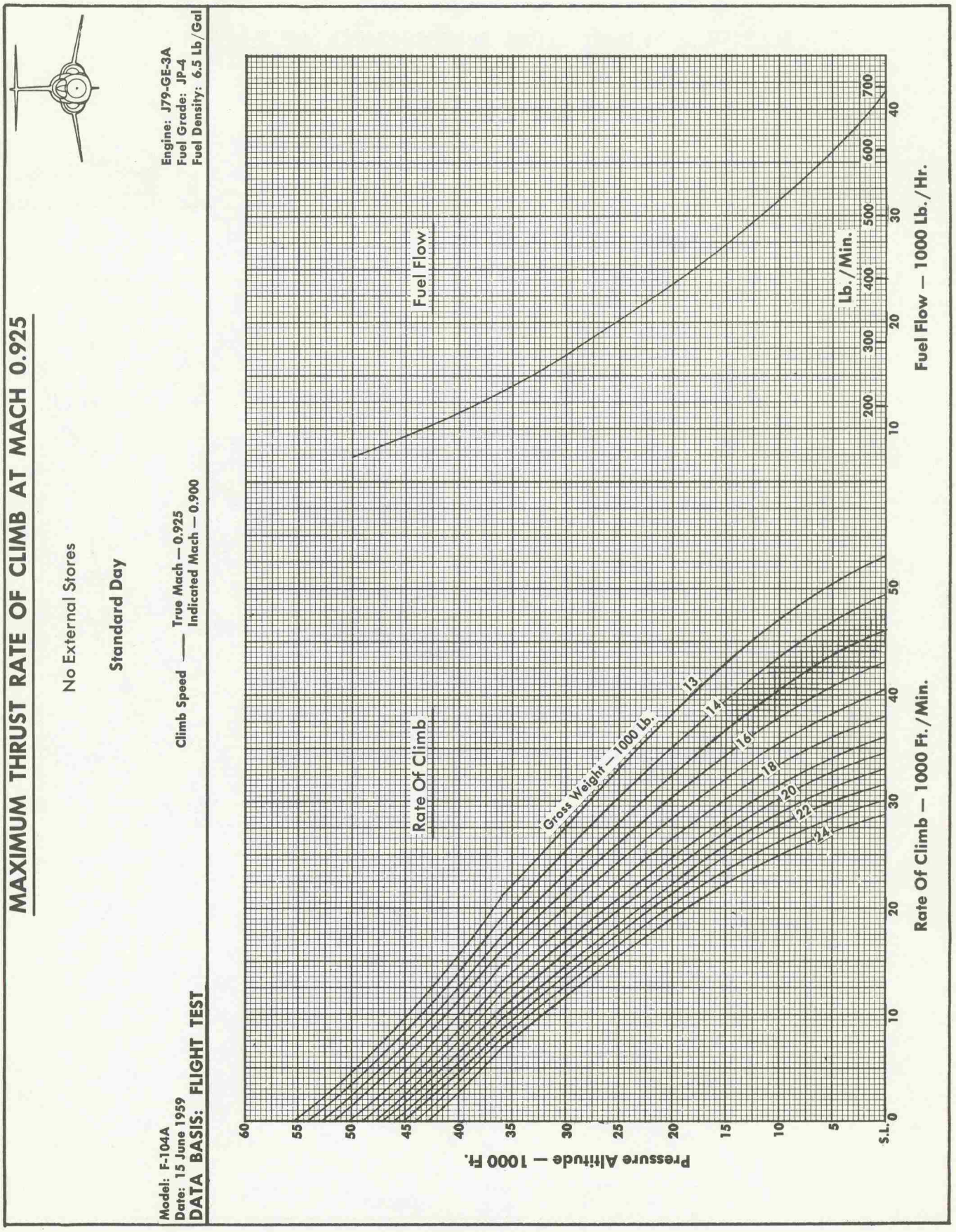

MAXIMUM THRUST RATE OF CLIMB AT MACH 0.925
No External Stores
Standard Day
Climb Speed — True Mach — 0.925
Indicated Mach — 0.900
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Rate Of Climb
Fuel Flow
Gross Weight — 1000 Lb.
13
14
16
18
20
22
24
Pressure Altitude — 1000 Ft.
60
55
50
45
40
35
30
25
20
15
10
5
S.L.
0
10
20
30
40
50
Rate Of Climb — 1000 Ft./Min.
Lb./Min.
200
300
400
500
600
700
10
20
30
40
Fuel Flow — 1000 Lb./Hr.

Figure A3-5

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.6

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

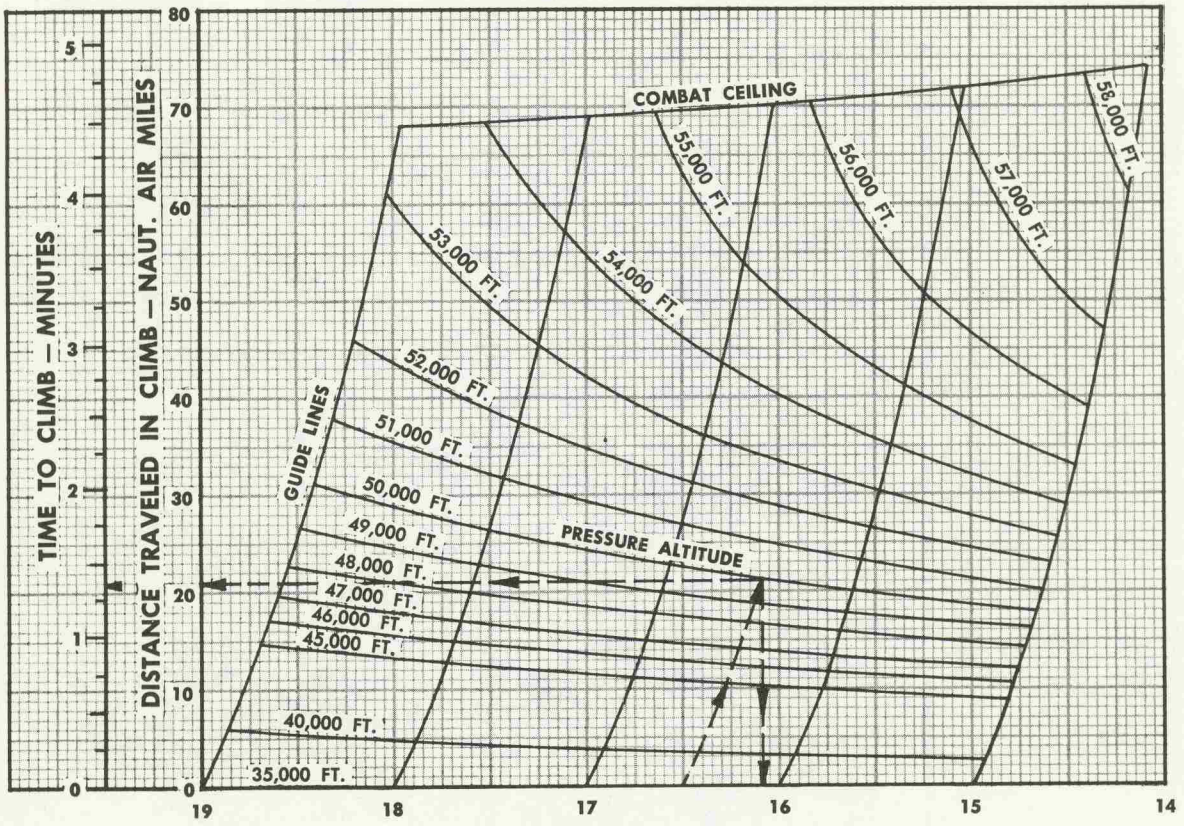


Figure A3-6

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.7

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

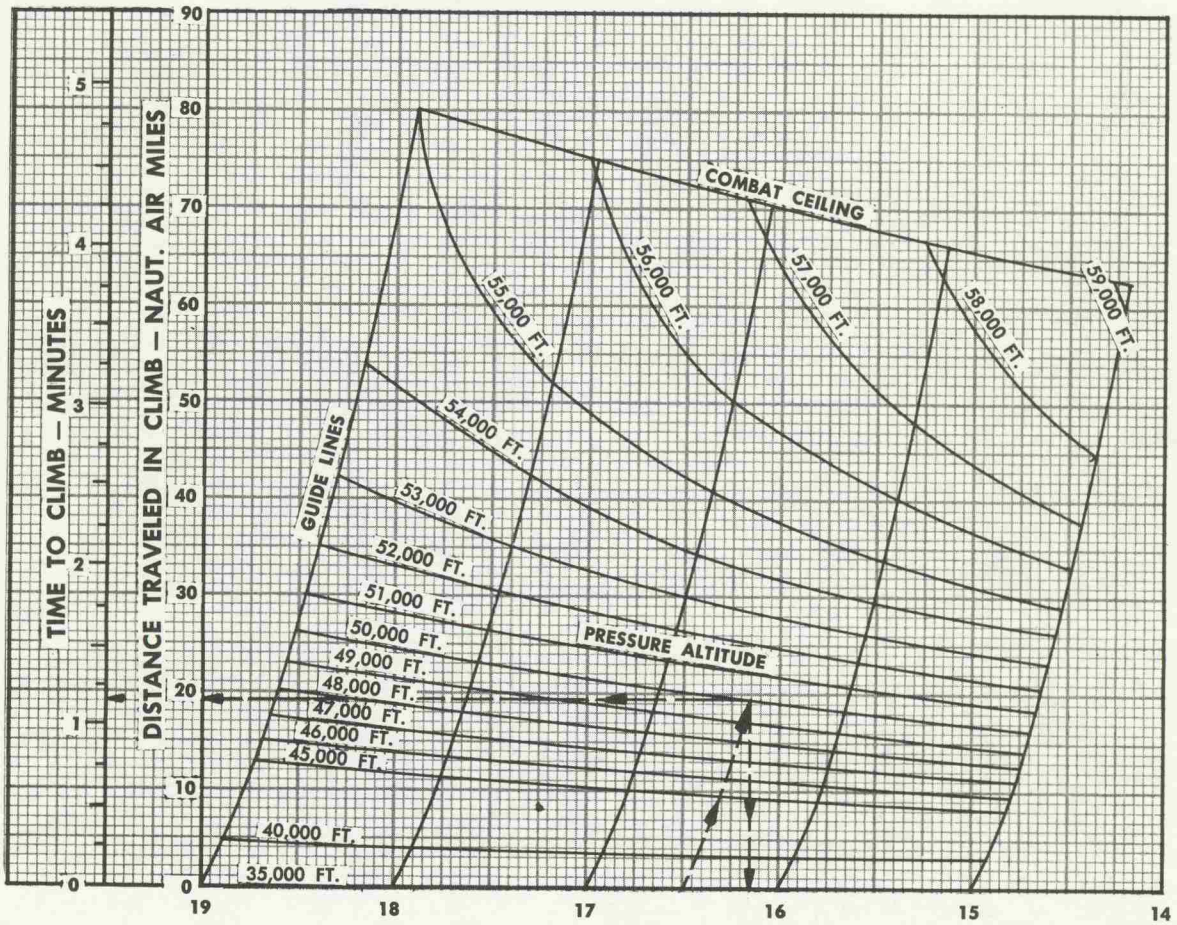


Figure A3-7

MAXIMUM THRUST RATE OF CLIMB AT MACH 1.7

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

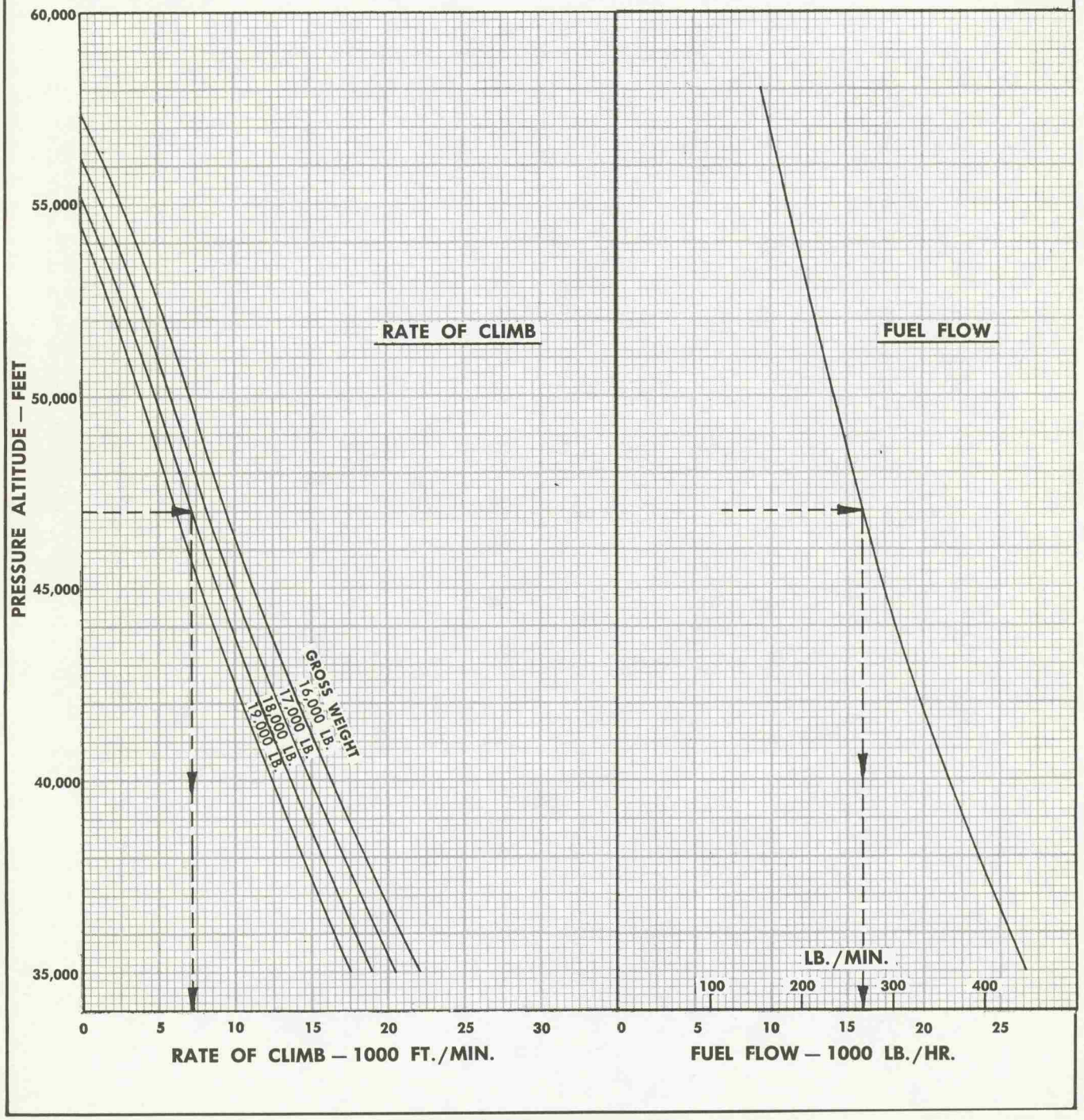


Figure A3-8

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.8

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

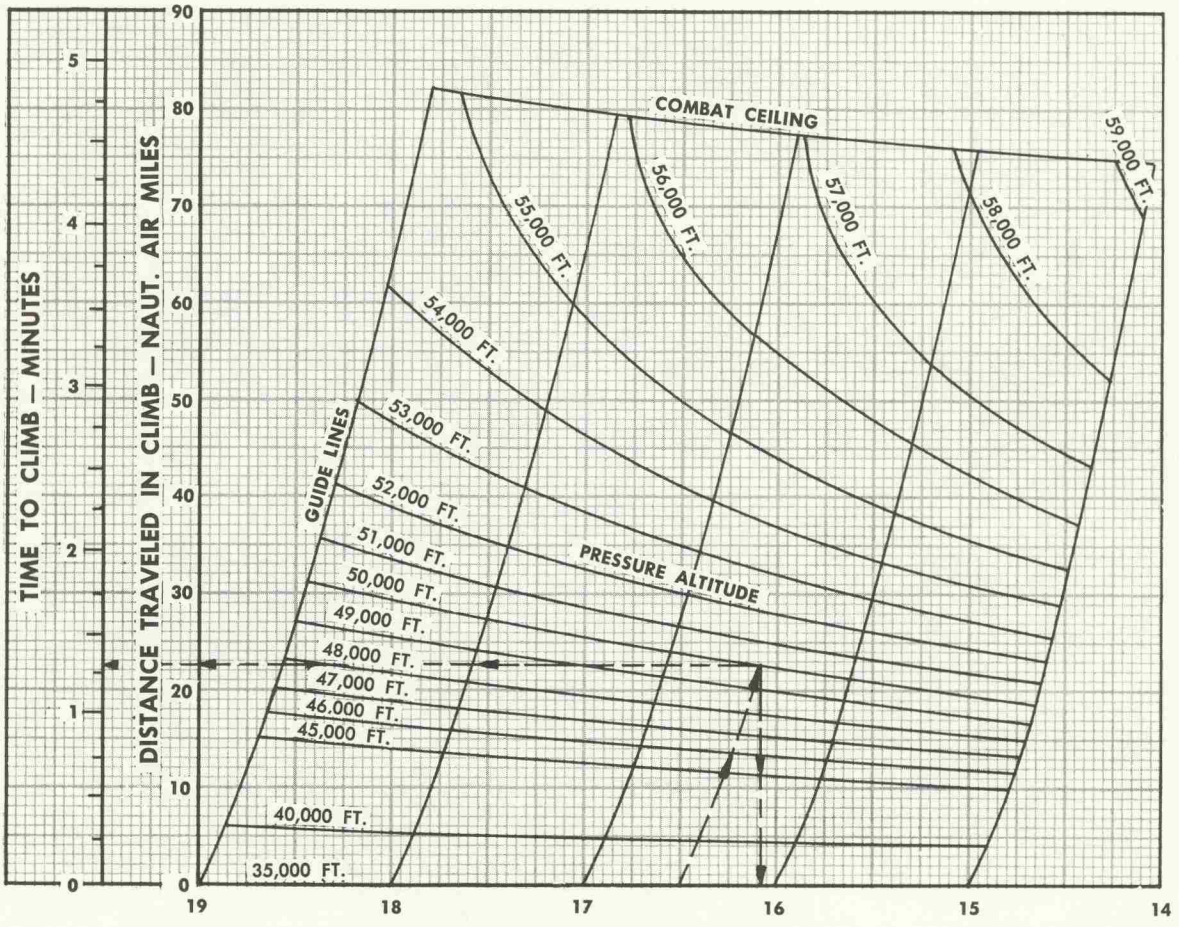


Figure A3-9

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.9

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

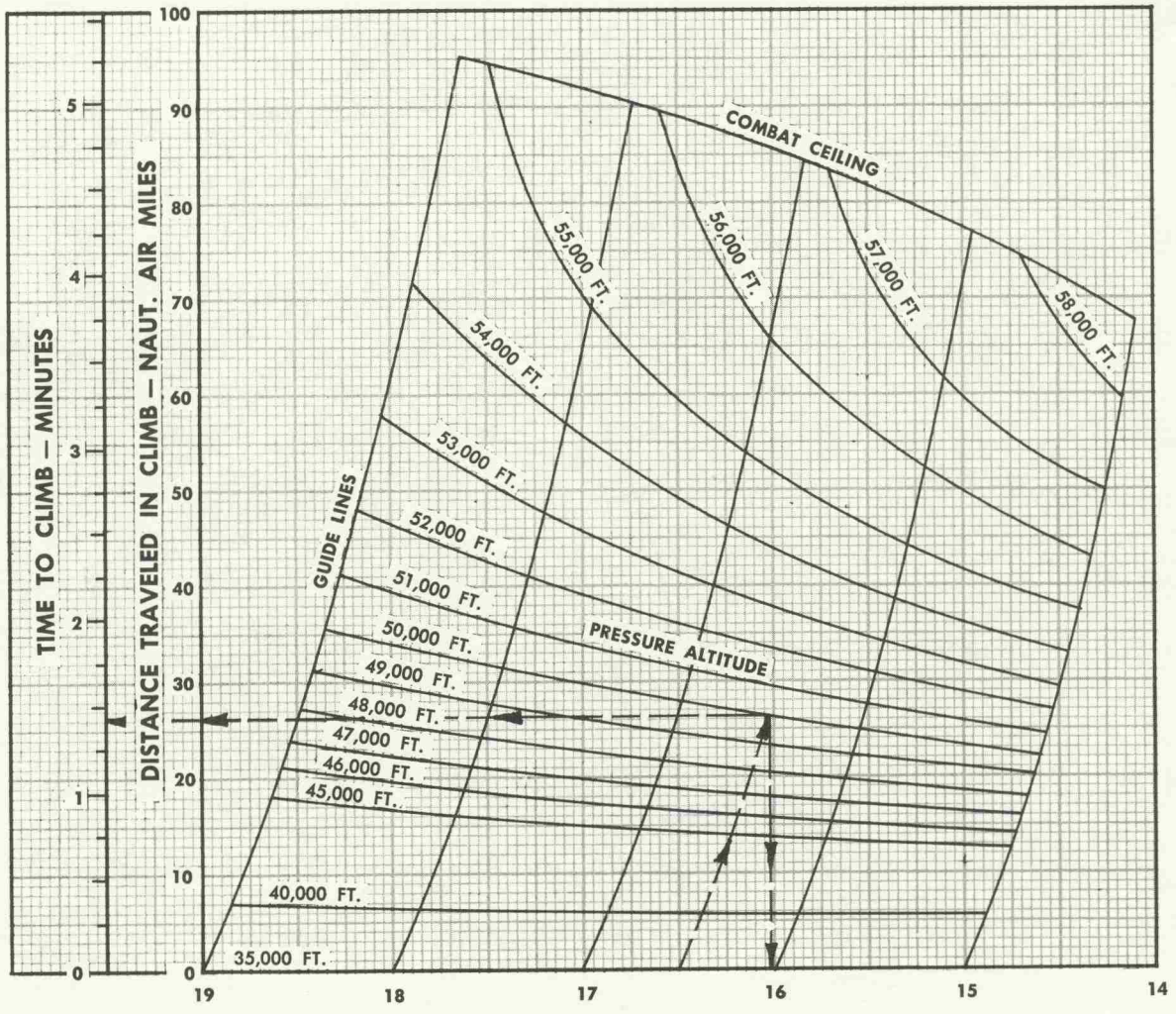
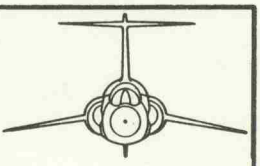


Figure A3-10

COMBAT AND SERVICE CEILING CHART



MAXIMUM THRUST

MILITARY THRUST

No External Stores

Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

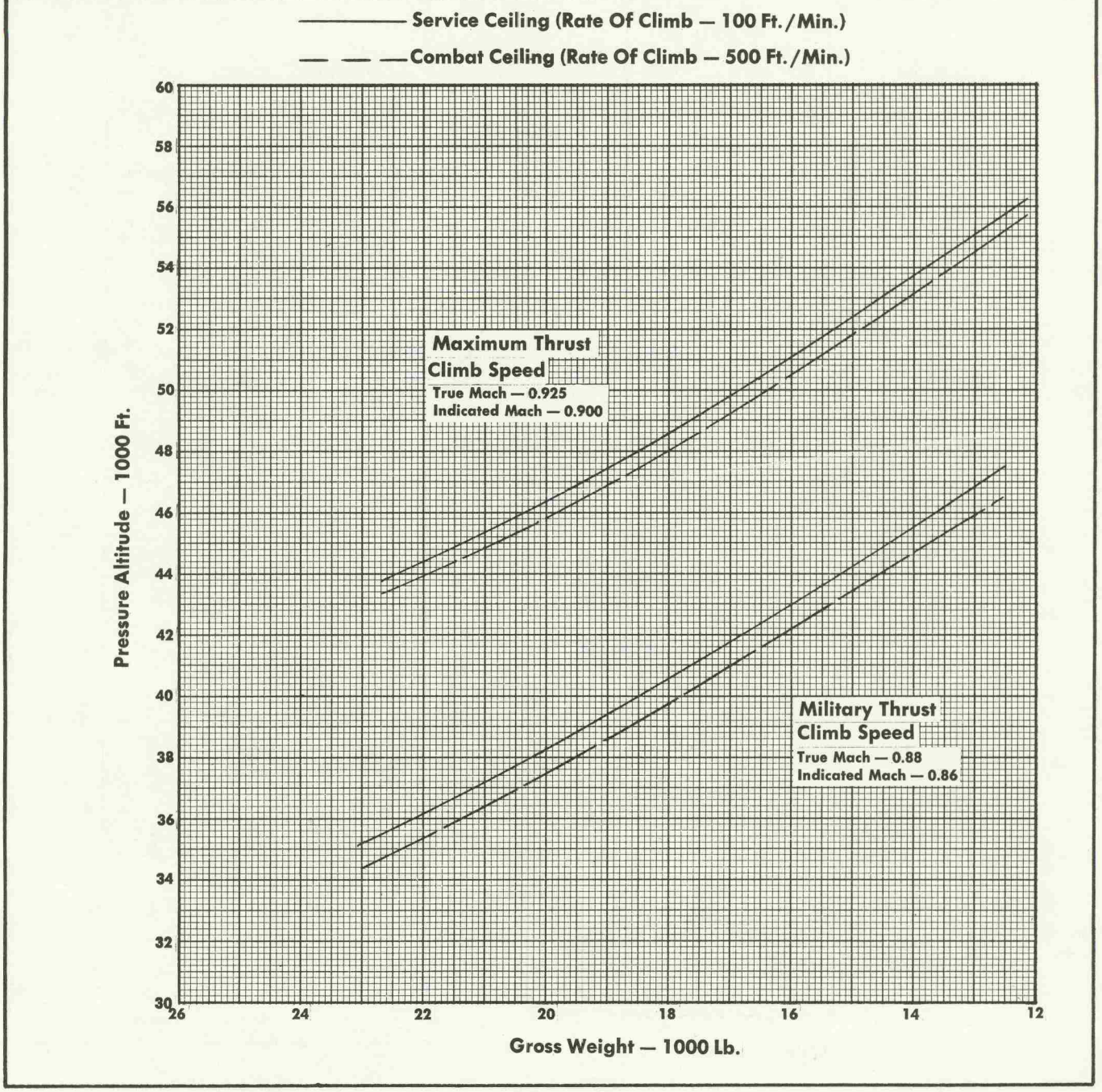


Figure A3-11

MILITARY THRUST CLIMB PERFORMANCE AT MACH 0.88

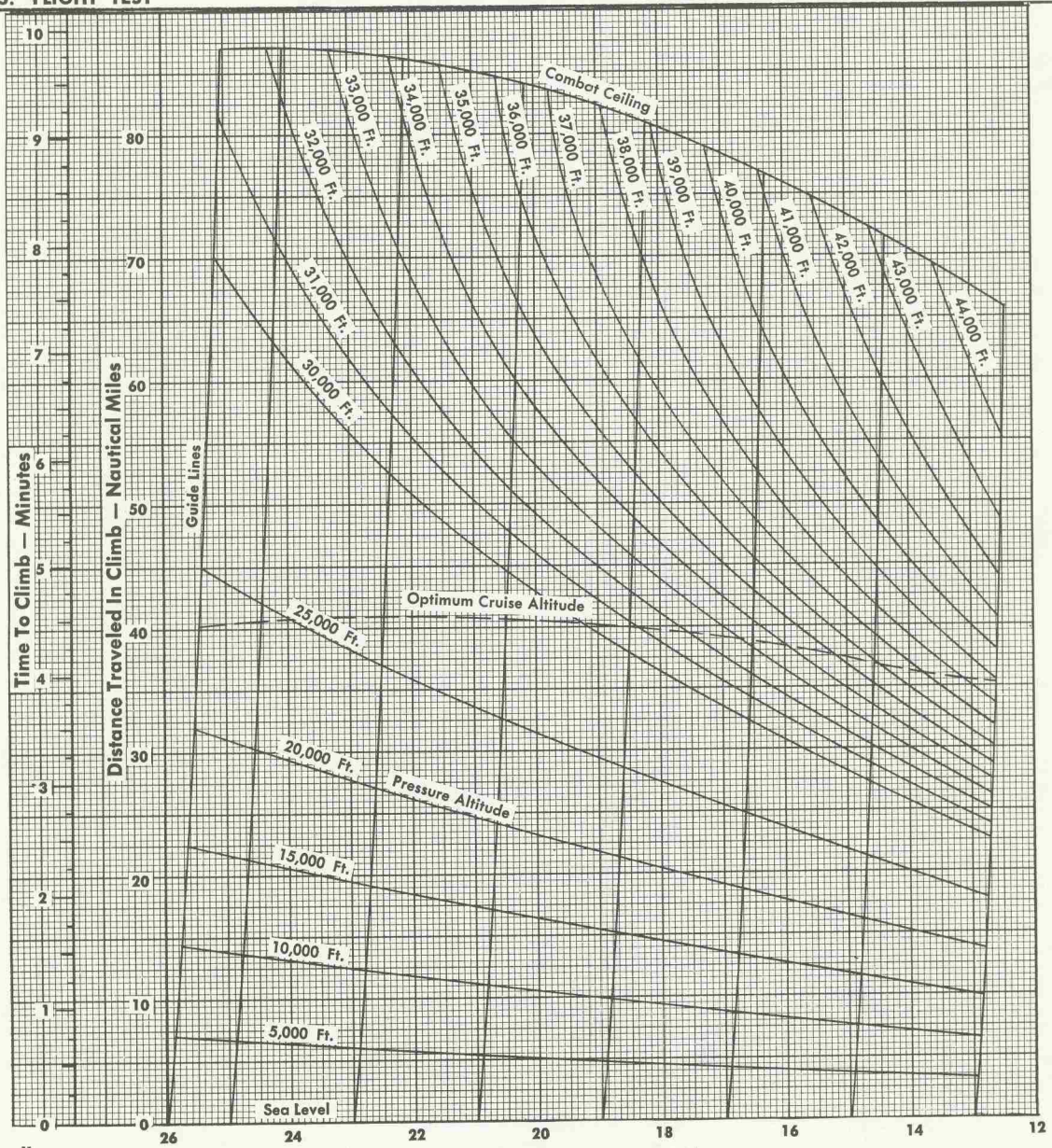
Two GAR-8 Missiles or Two 165 Gallon Tip Tanks

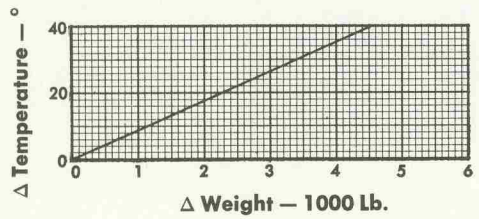
Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed — True Mach — 0.88 / Indicated Mach — 0.86

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal





Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-12

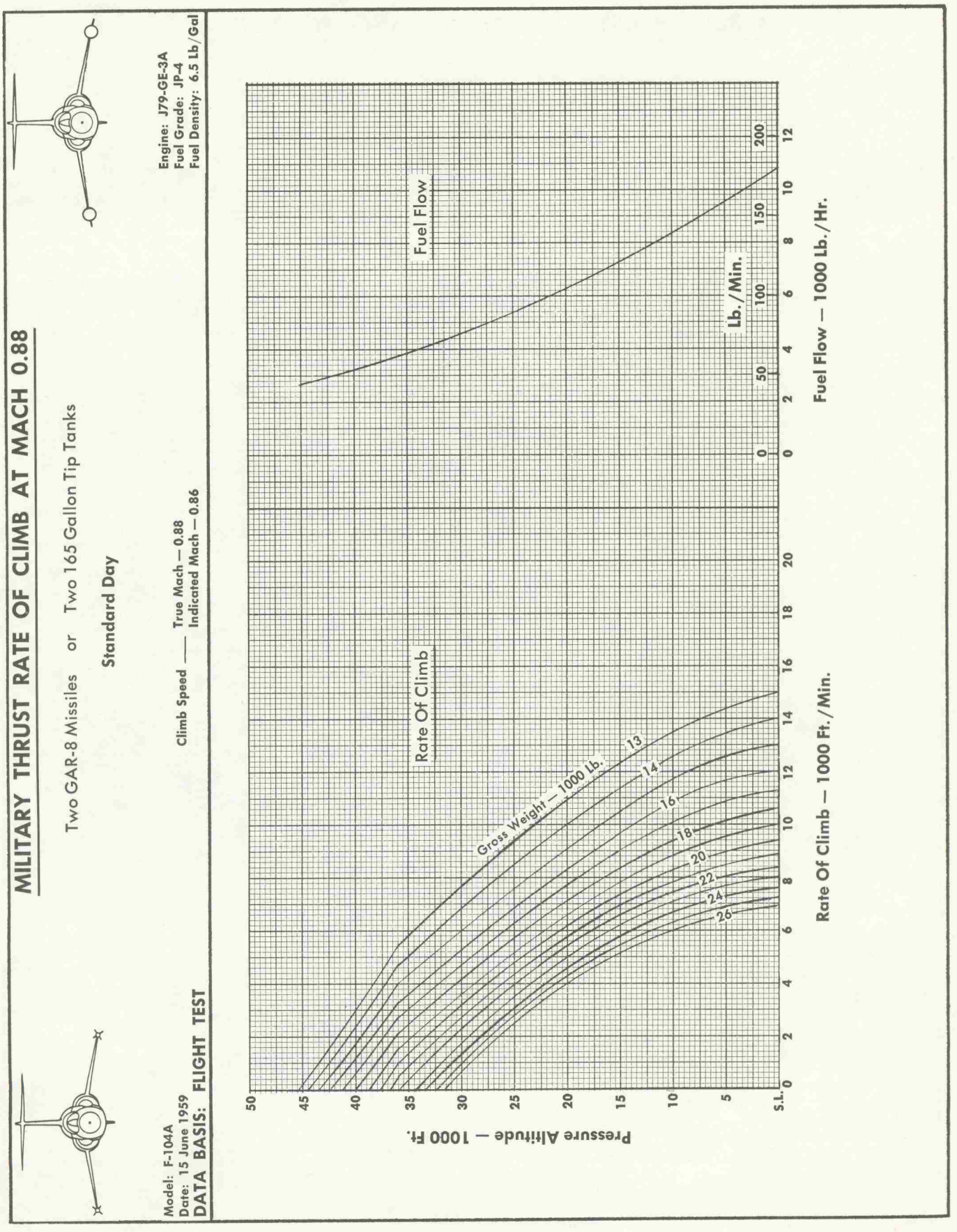
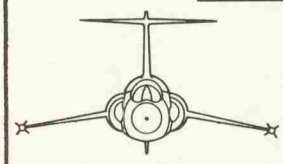

MILITARY THRUST RATE OF CLIMB AT MACH 0.88
Two GAR-8 Missiles or Two 165 Gallon Tip Tanks
Standard Day
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Climb Speed —— True Mach — 0.88
Indicated Mach — 0.86
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Rate Of Climb
Fuel Flow
Gross Weight — 1000 Lb.
13
14
16
18
20
22
24
26
Lb./Min.
Pressure Altitude — 1000 Ft.
Rate Of Climb — 1000 Ft./Min.
Fuel Flow — 1000 Lb./Hr.

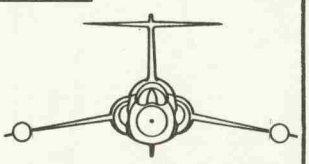
Figure A3-13

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 0.925



Two GAR-8 Missiles or Two 165 Gallon Tip Tanks

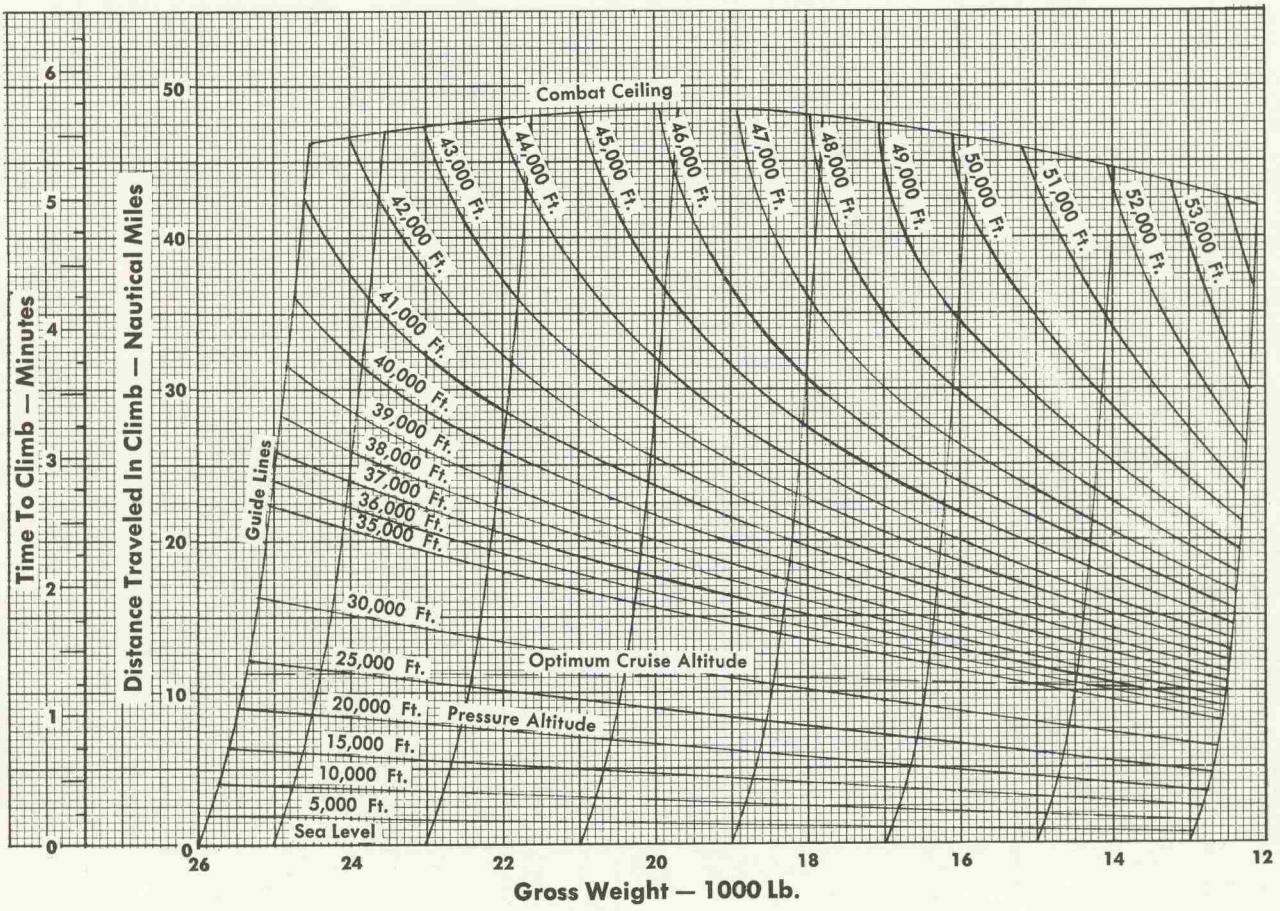
Standard Day

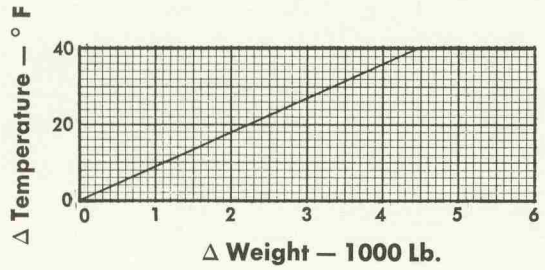


Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed— True Mach — 0.925 / Indicated Mach — 0.900

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal





Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-14

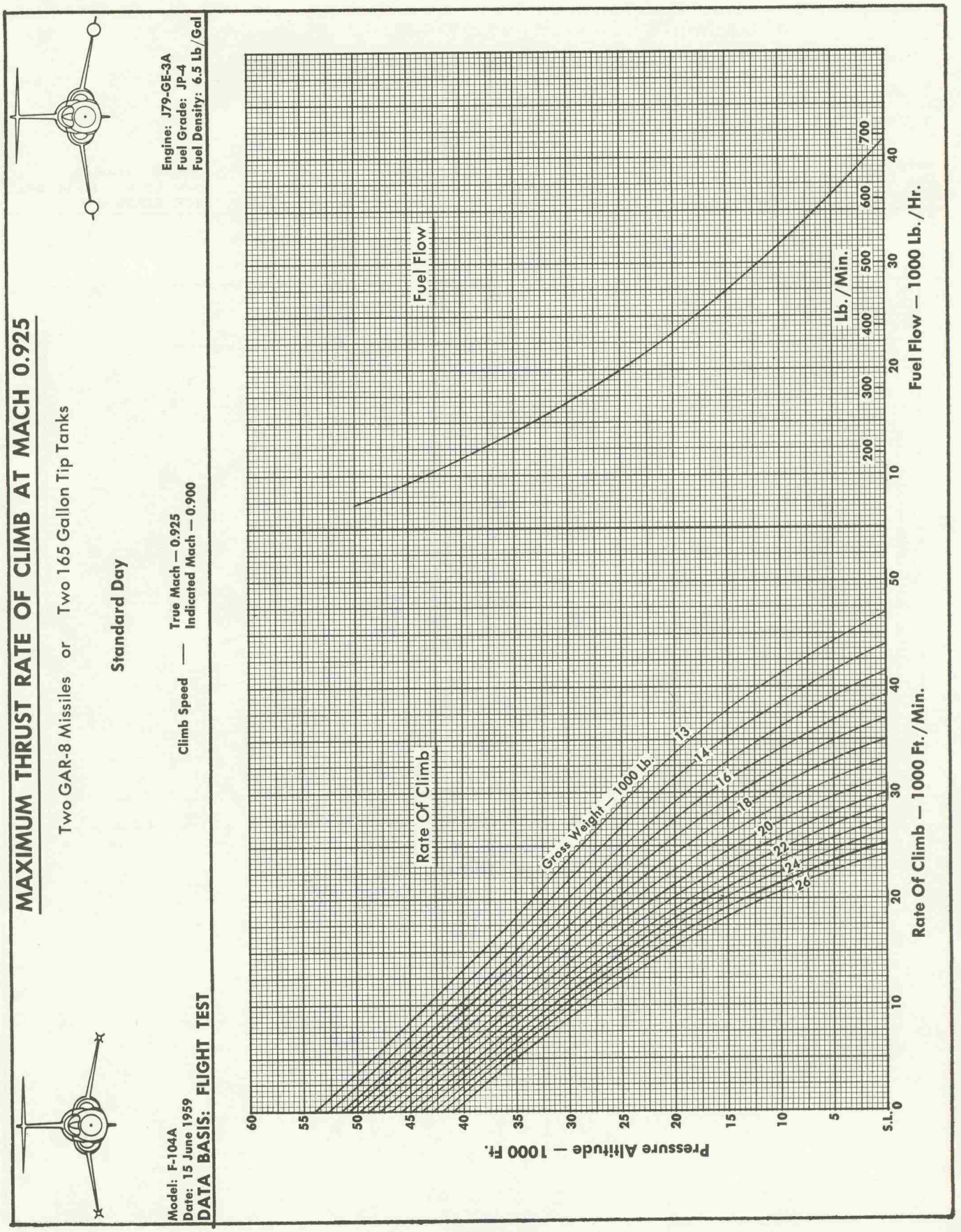

MAXIMUM THRUST RATE OF CLIMB AT MACH 0.925
Two GAR-8 Missiles or Two 165 Gallon Tip Tanks
Standard Day
Climb Speed — True Mach — 0.925
Indicated Mach — 0.900
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Fuel Flow
Rate Of Climb
Gross Weight — 1000 Lb.
13
14
16
18
20
22
24
26
Pressure Altitude — 1000 Ft.
Rate Of Climb — 1000 Ft./Min.
Fuel Flow — 1000 Lb./Hr.
Lb./Min.
S.L.
0
5
10
15
20
25
30
35
40
45
50
55
60
200
300
400
500
600
700

Figure A3-15

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.6

Two GAR-8 Missiles

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

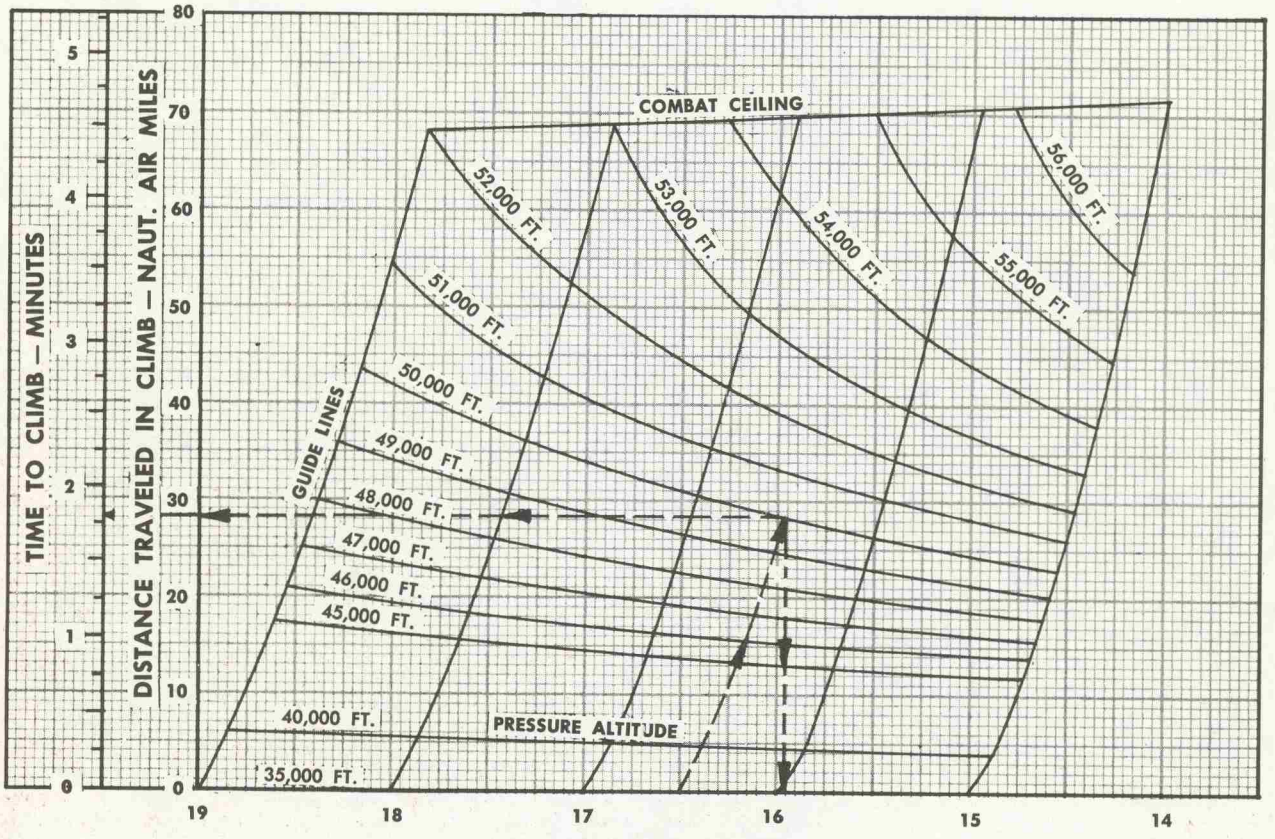


Figure A3-16

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.7

Two GAR-8 Missiles

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

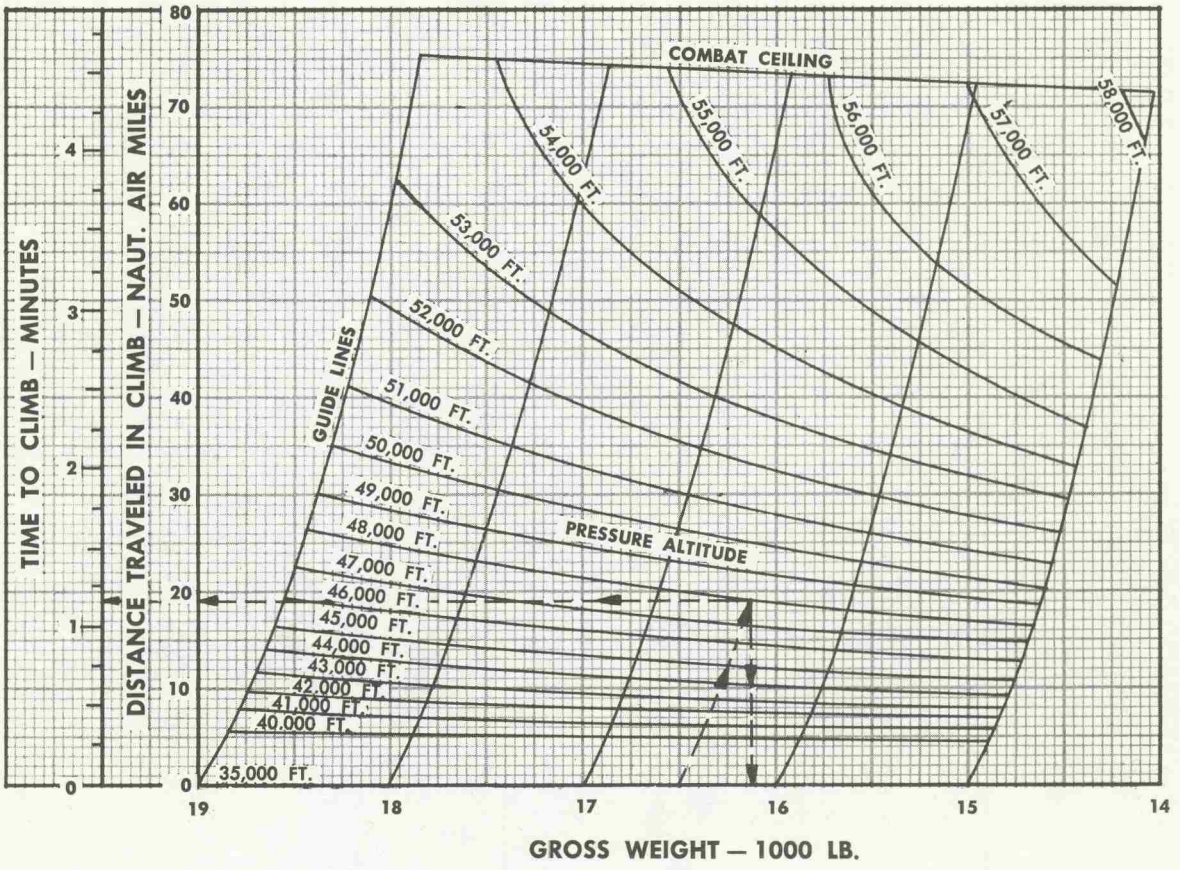


Figure A3-17

MAXIMUM THRUST RATE OF CLIMB AT MACH 1.7

Two GAR-8 Missiles

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

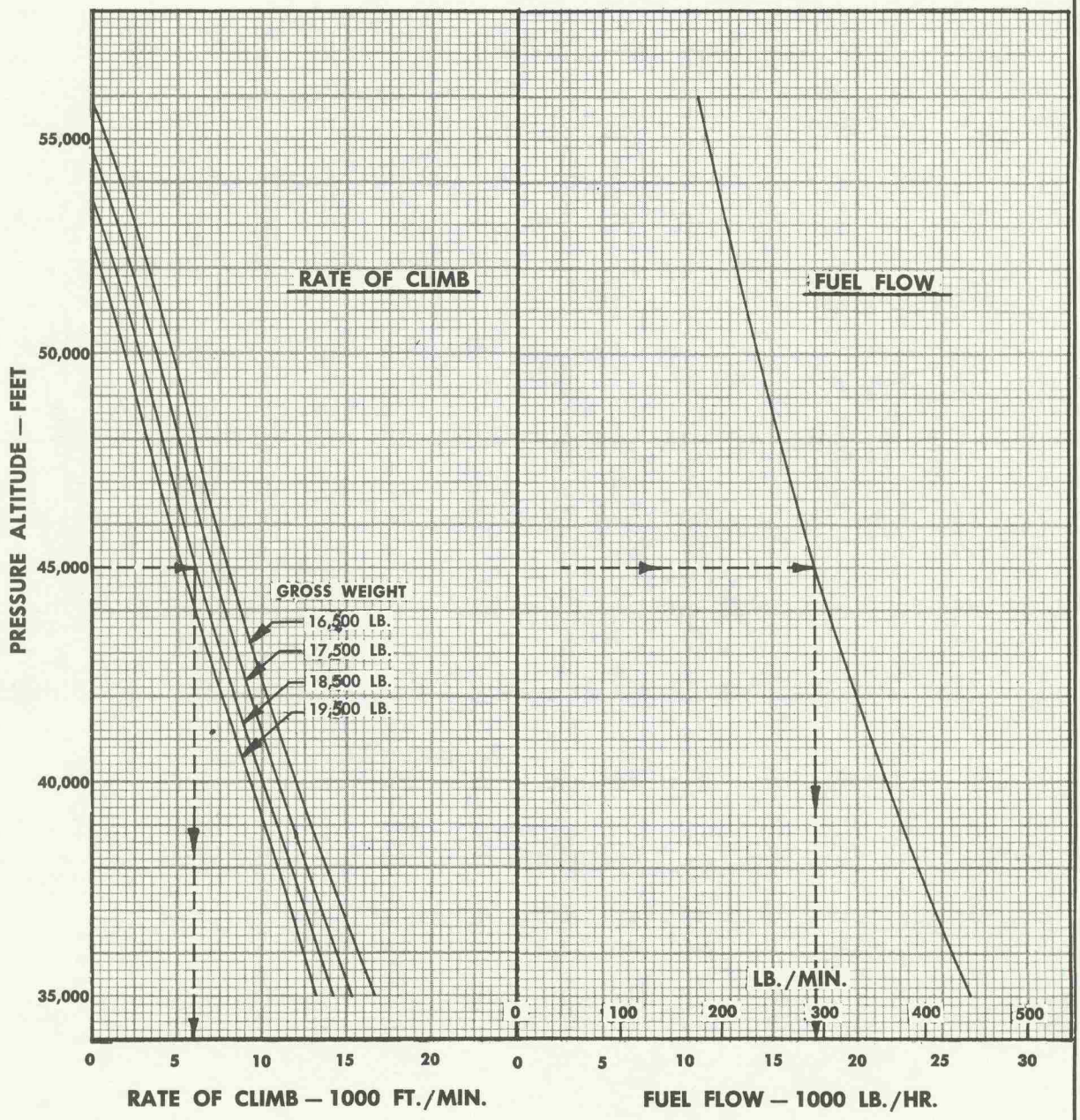


Figure A3-18

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.8

Two GAR-8 Missiles

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

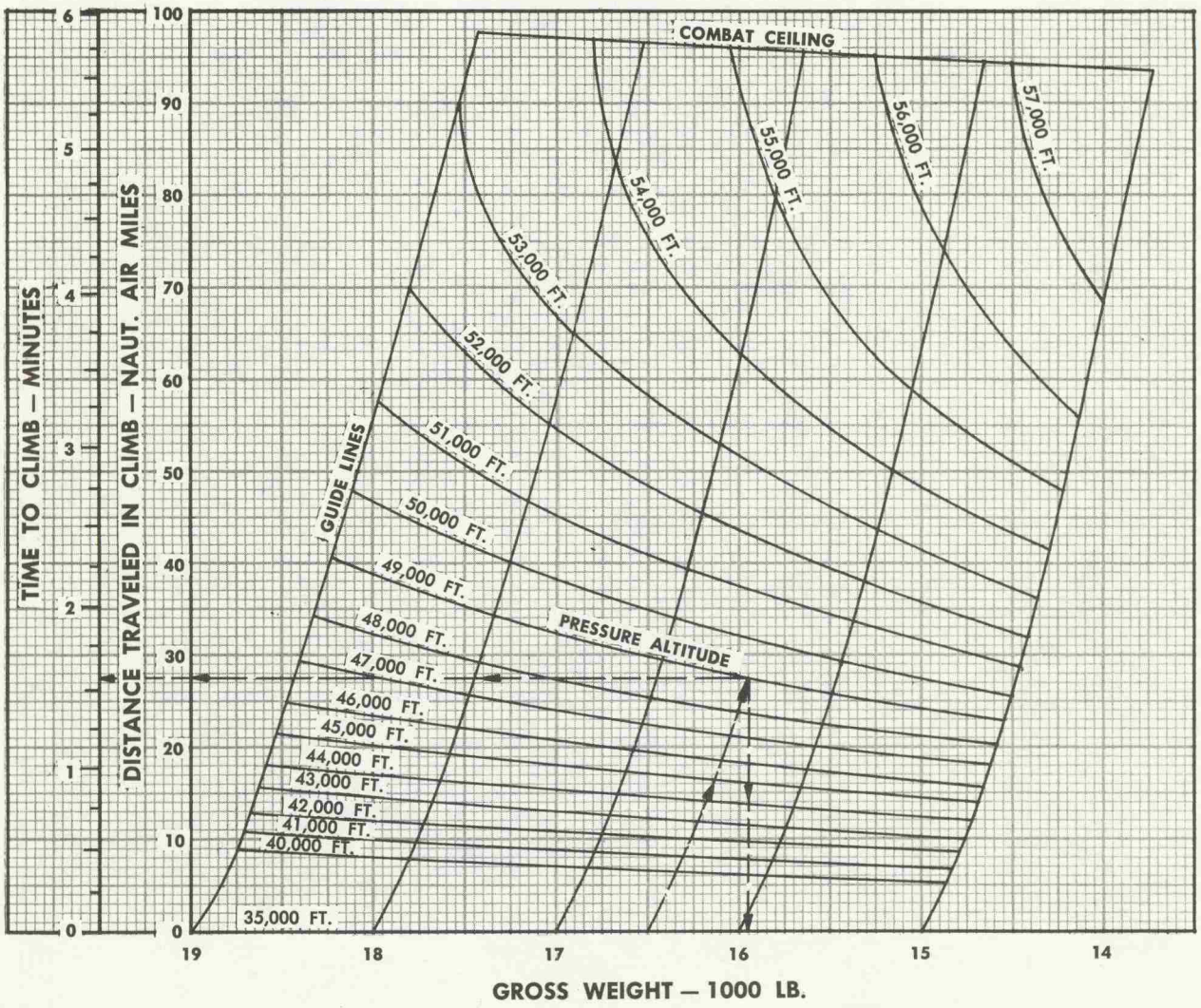


Figure A3-19

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 1.9

Two GAR-8 Missiles

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

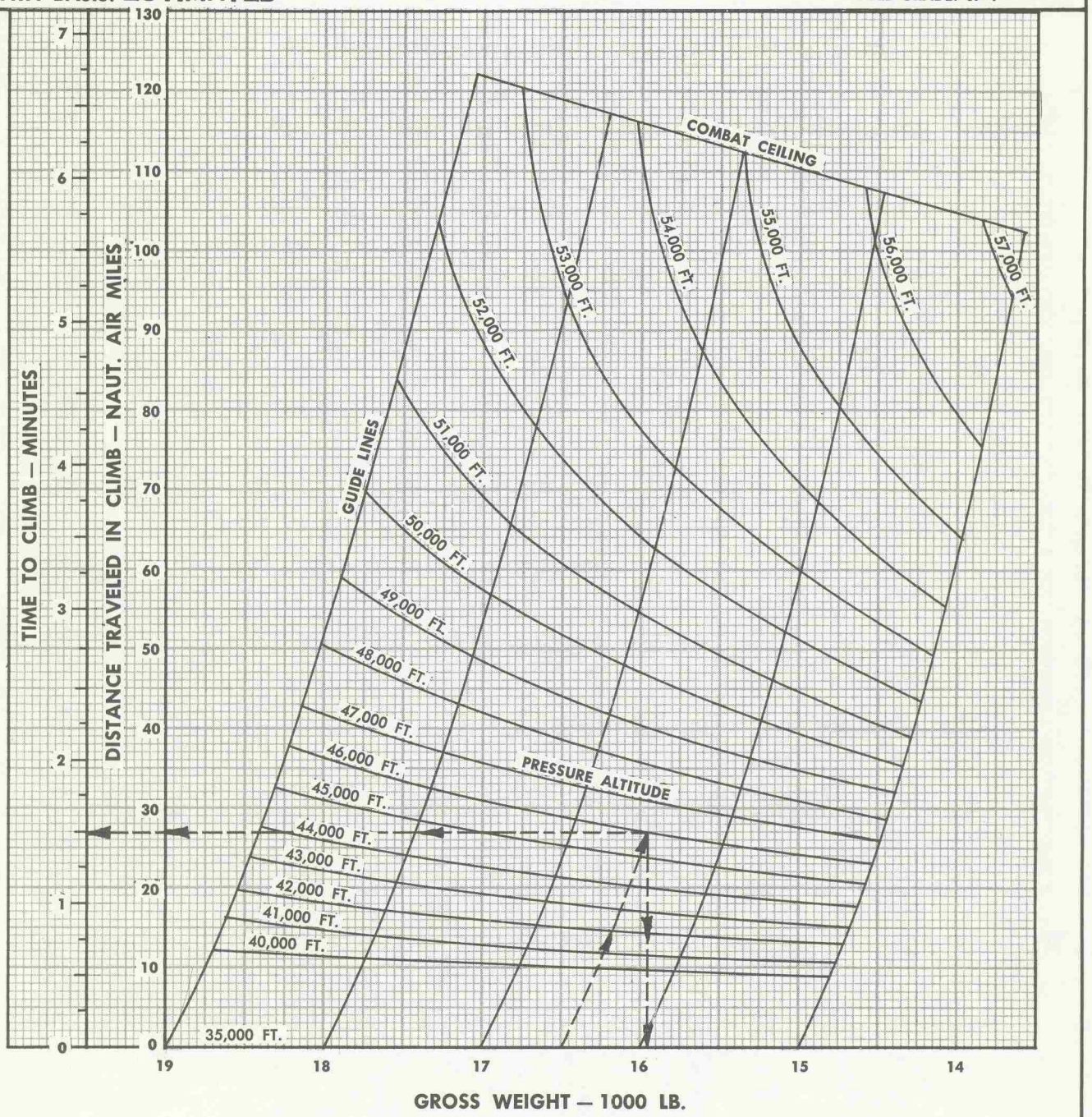
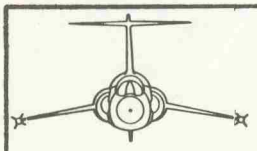
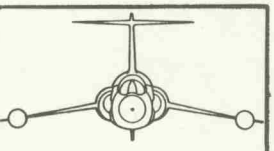


Figure A3-20



COMBAT AND SERVICE CEILING CHART



MAXIMUM THRUST

MILITARY THRUST

Two GAR-8 Missiles or Two 165 Gallon Tip Tanks

Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

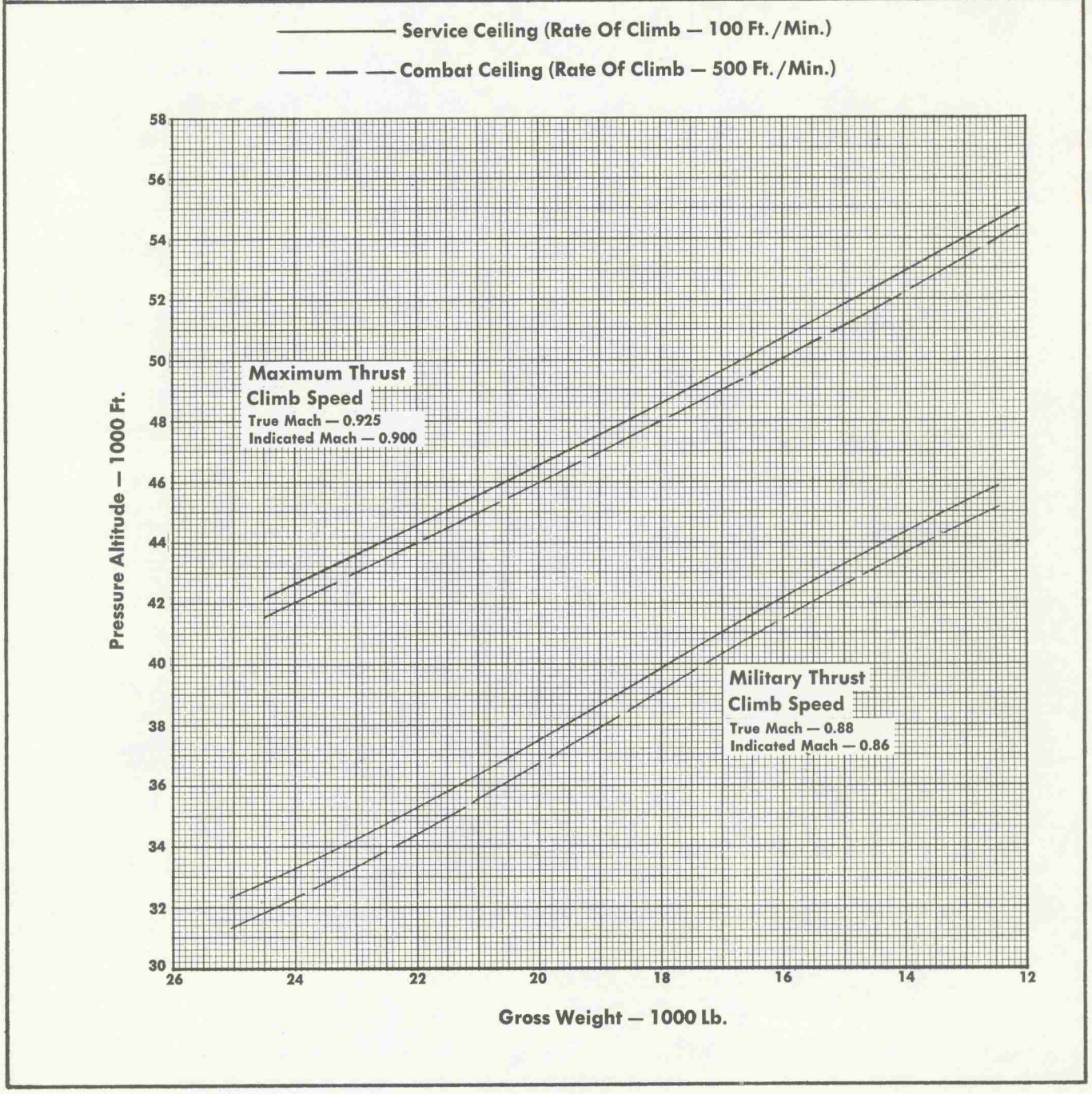
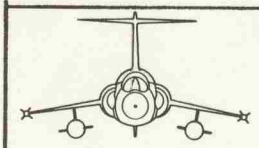
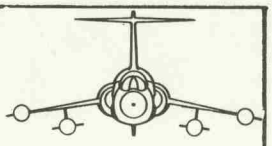


Figure A3-21



MILITARY THRUST CLIMB PERFORMANCE AT MACH 0.825

Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles
or
Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks

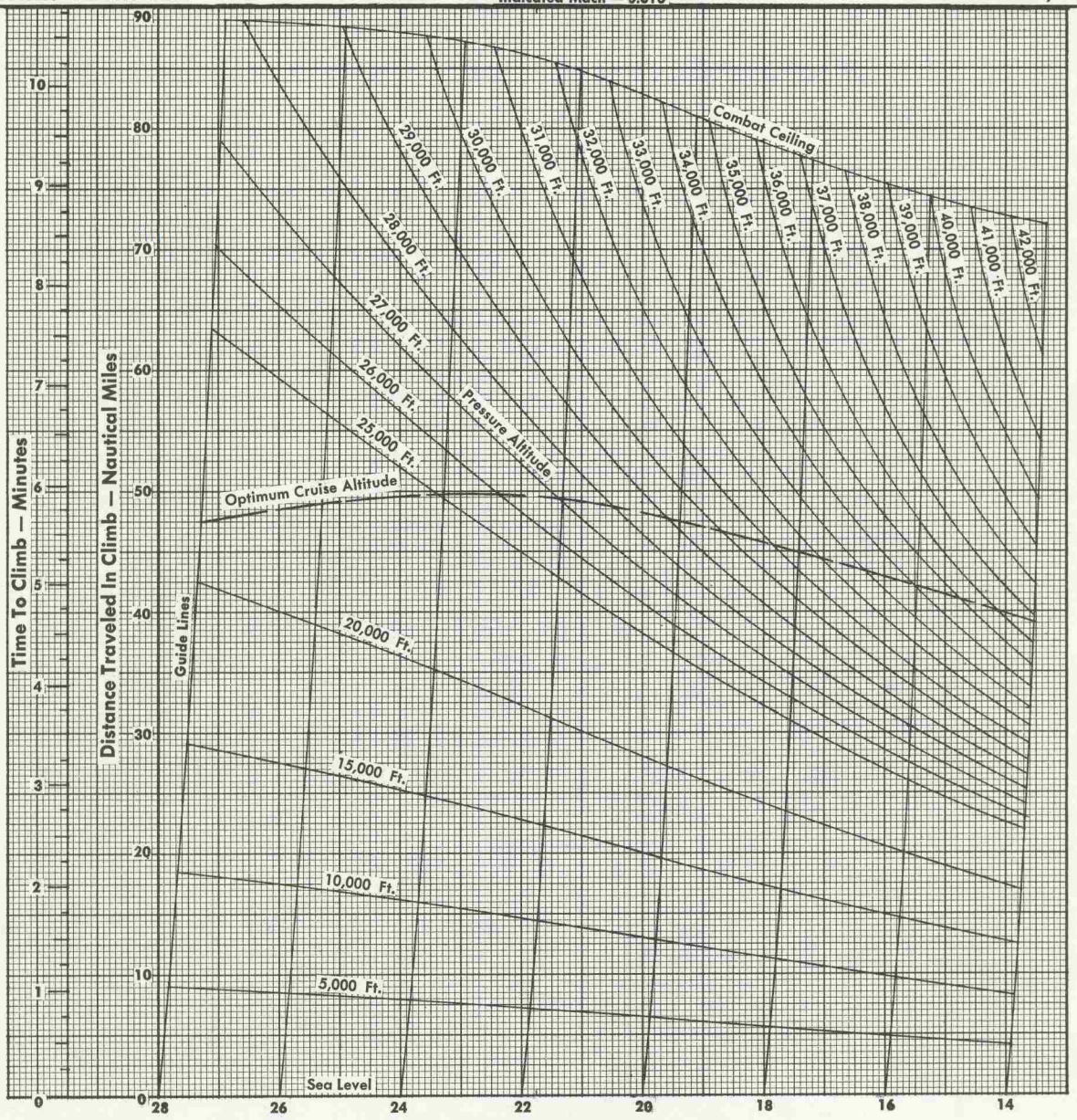


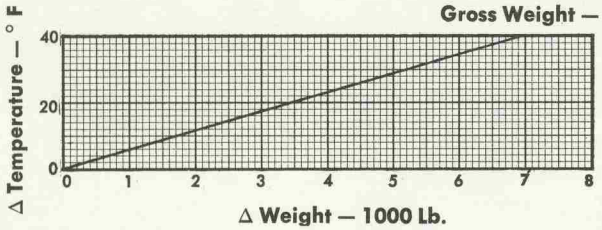
Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed — True Mach — 0.825
Indicated Mach — 0.810

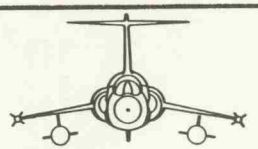
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



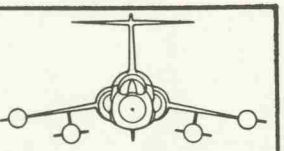


Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-22



MILITARY THRUST RATE OF CLIMB AT MACH 0.825



Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles
or
Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Standard Day

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Climb Speed — True Mach — 0.825
Indicated Mach — 0.810

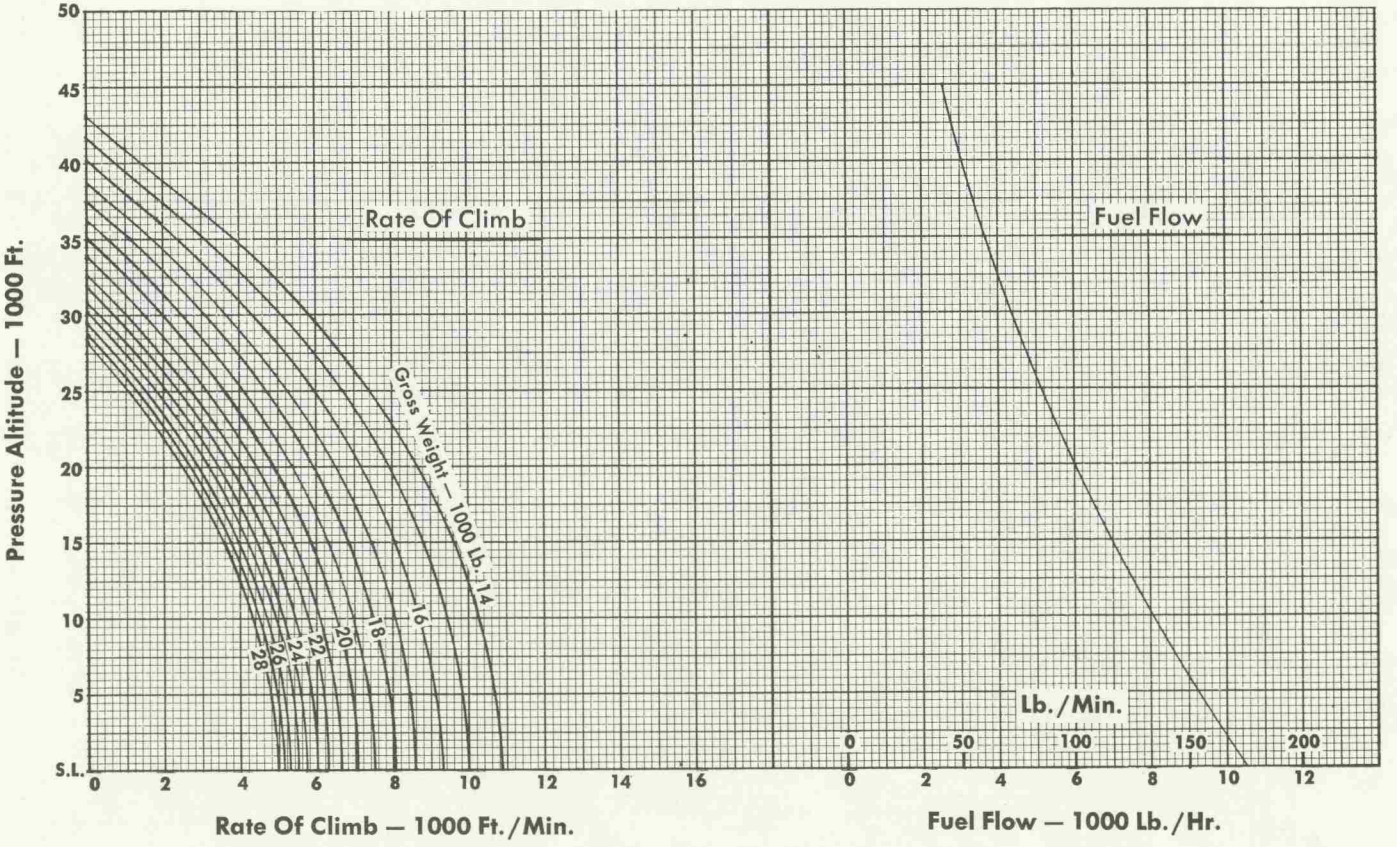


Figure A3-23

MAXIMUM THRUST CLIMB PERFORMANCE AT MACH 0.90

Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles

or

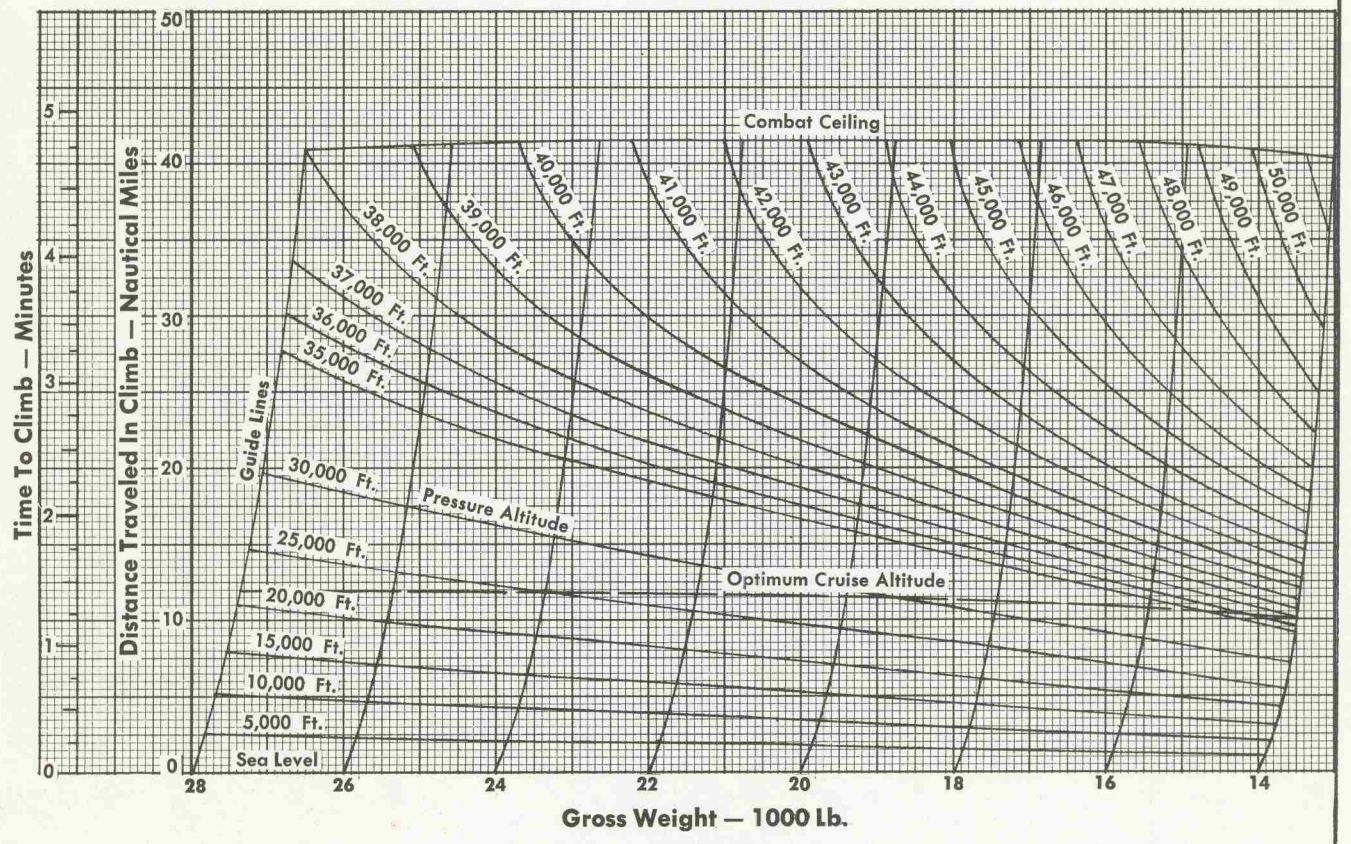
Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks

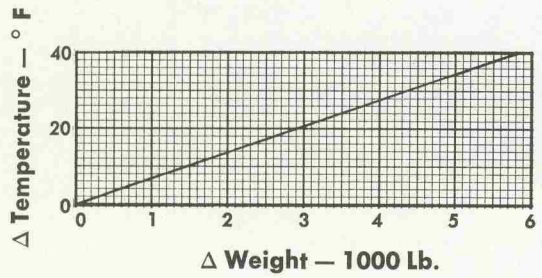
Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Climb Speed — True Mach — 0.90
Indicated Mach — 0.88

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal





Remarks:
Determine equivalent weight for non-standard temperature conditions. Equivalent weight equals actual aircraft gross weight + Δ Weight (For hot days) or — Δ Weight (For cold days) From Δ Temp Vs Δ Weight curve.

Figure A3-24

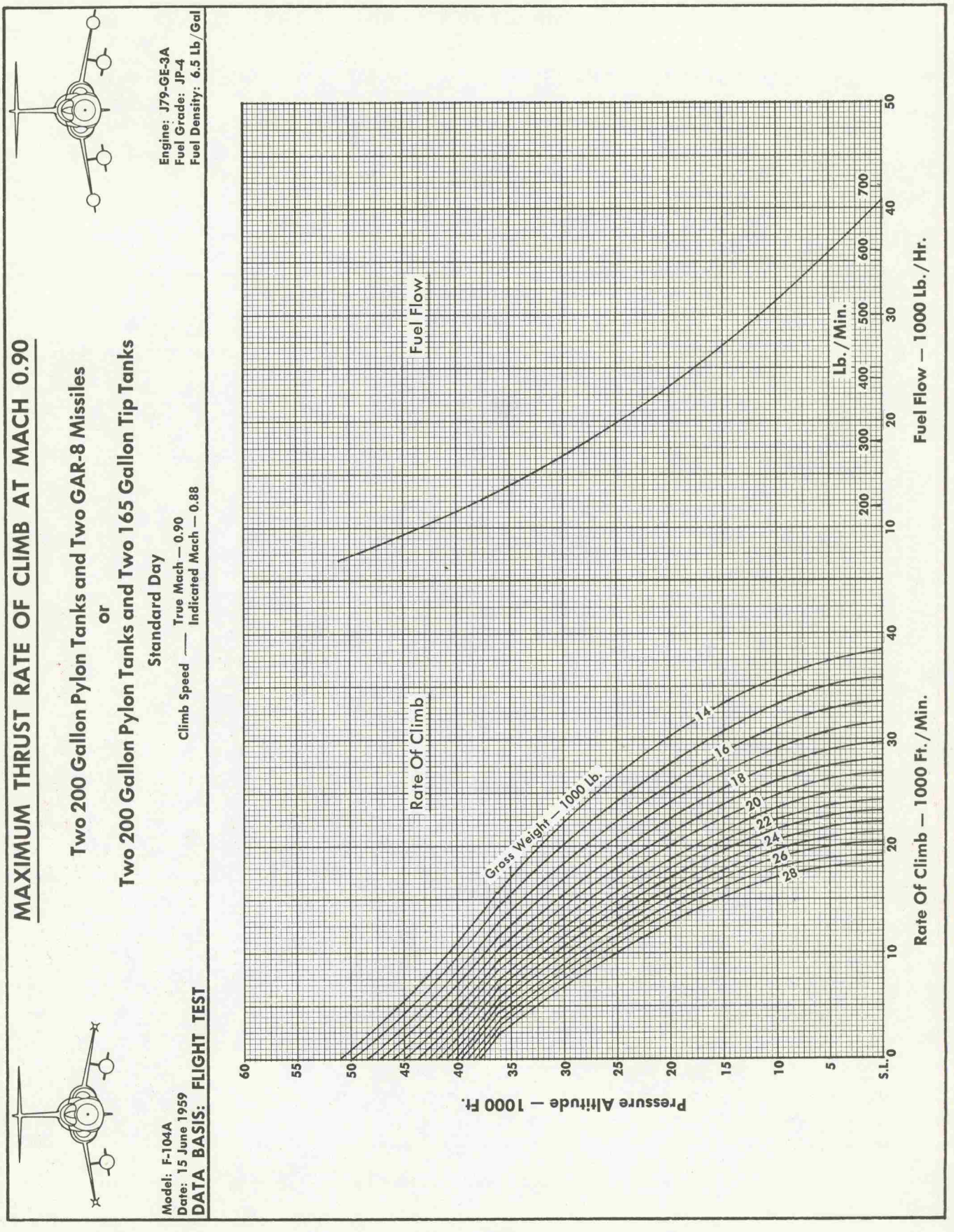
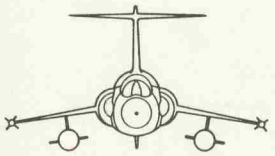
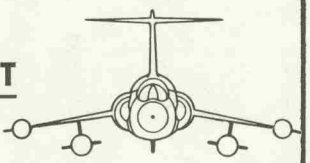

MAXIMUM THRUST RATE OF CLIMB AT MACH 0.90
Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles
or
Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks
Standard Day
Climb Speed — True Mach — 0.90
Indicated Mach — 0.88
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Rate Of Climb
Fuel Flow
Gross Weight — 1000 Lb.
14
16
18
20
22
24
26
28
Lb./Min.
Pressure Altitude — 1000 Ft.
Rate Of Climb — 1000 Ft./Min.
Fuel Flow — 1000 Lb./Hr.

Figure A3-25

COMBAT AND SERVICE CEILING CHART



MAXIMUM THRUST — MILITARY THRUST



Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles

or

Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks

Standard Day

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

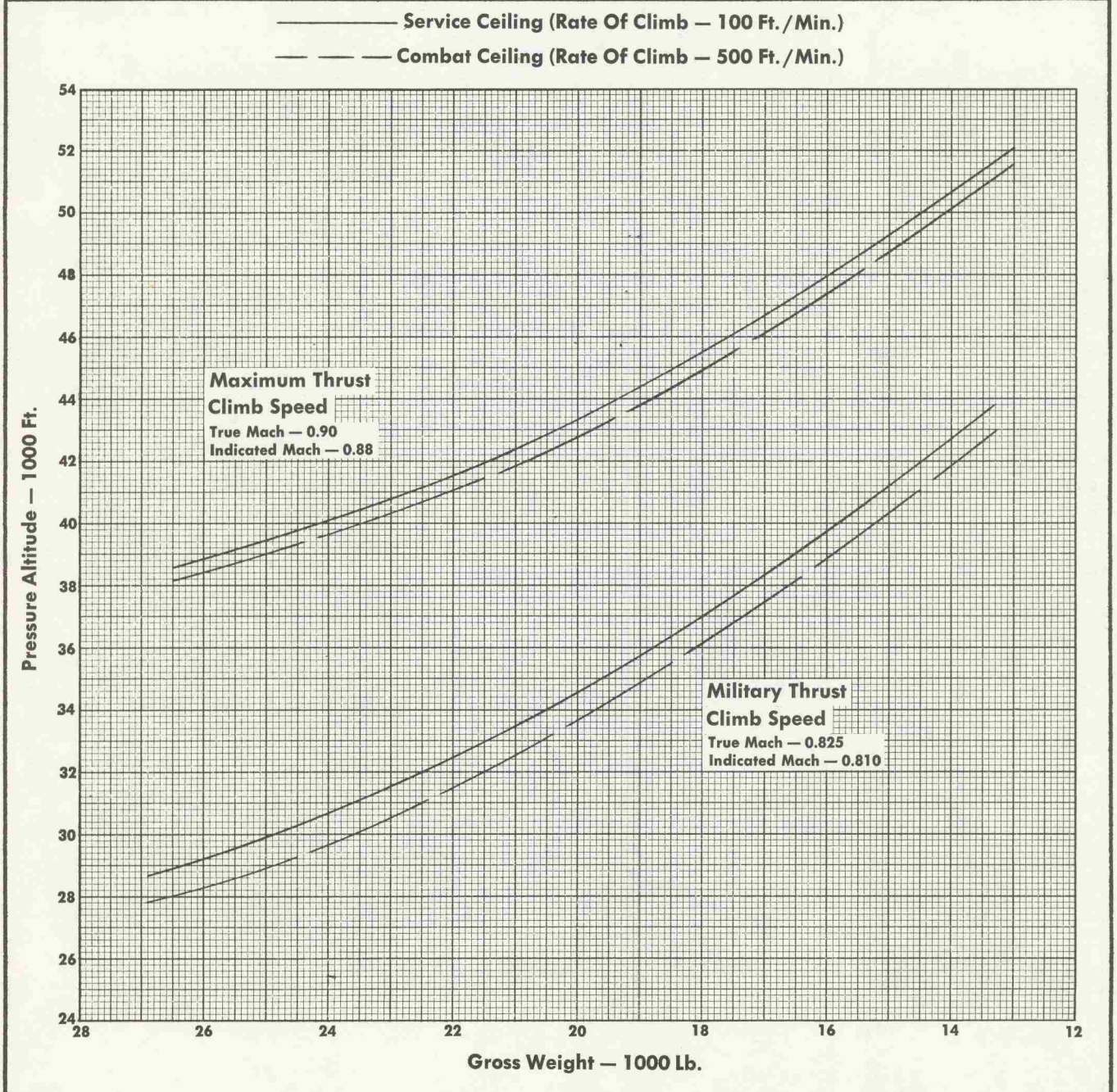


Figure A3-26

Part 4 — Range

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Sea Level **A4-28**
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Nautical Miles Per 1000 lb. of Fuel — Two 165 Gallon Tip Tanks

Sea Level **A4-34**
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Nautical Miles Per 1000 lb. of Fuel — Two GAR-8 Missiles

Sea Level **A4-40**
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Nautical Miles Per 1000 lb. of Fuel — Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks

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Nautical Miles Per 1000 lb. of Fuel — Two 200 Gallon Pylon Tanks and Two Tip GAR-8 Missiles

Sea Level **A4-53**
15,000 Feet **A4-54**
20,000 Feet **A4-55**

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Page numbers in bold face denote illustration

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Nautical Miles Per 1000 lb. of Fuel — Two 200 Gallon Pylon Tanks and Two Tip GAR-8 Missiles (Continued)

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Range Factor Chart **A4-60**

RANGE

RANGE PROFILE CHARTS.

The range profile charts present the range characteristics of the aircraft in a convenient pictorial form and provide a simplified method of flight planning for the majority of missions where cruise control must be considered. This is done by pictorially presenting the relationship between time, fuel, and distance obtainable for recommended cruise procedures. The range data presented consist of three types of charts: Mission Profiles, Intercept Profiles and Optimum Return Profiles.

MISSION PROFILE CHARTS.

CHART SCOPE.

The Mission Profile charts present the level flight cruise and the optimum cruise (cruise-climb) range capabilities of the aircraft. The charts give the time and fuel required to fly a given distance under no-wind conditions at any cruising altitude from sea level to the optimum cruise altitude of the aircraft.

CHART BASIS.

A mission sequence of Maximum Thrust take-off and Military Thrust acceleration to climb speed, Military Thrust climb, and maximum range cruise is assumed. Fuel allowances have been included for ground maneuver, take-off, and acceleration to climb speed. The charts do not include any reserve for loiter, descent, or landing at the end of the cruise. The charts are based on a fully loaded gross weight of 19,000 pounds with a fuel load of 5,902 pounds for the no-external stores configuration; 21,532 pounds (8,034 pounds of fuel) for the 165-gallon tip tank configuration, 19,420 pounds (5,902 pounds of fuel) for the GAR-8 missile configuration; 24,626 pounds (10,530 pounds of fuel) for the 165-gallon tip tank plus pylon tank configuration; and 22,514 pounds (8,398 pounds of fuel) for the GAR-8 missile plus pylon tank configuration. Minor variations in aircraft basic weight do not affect the usability of the charts.

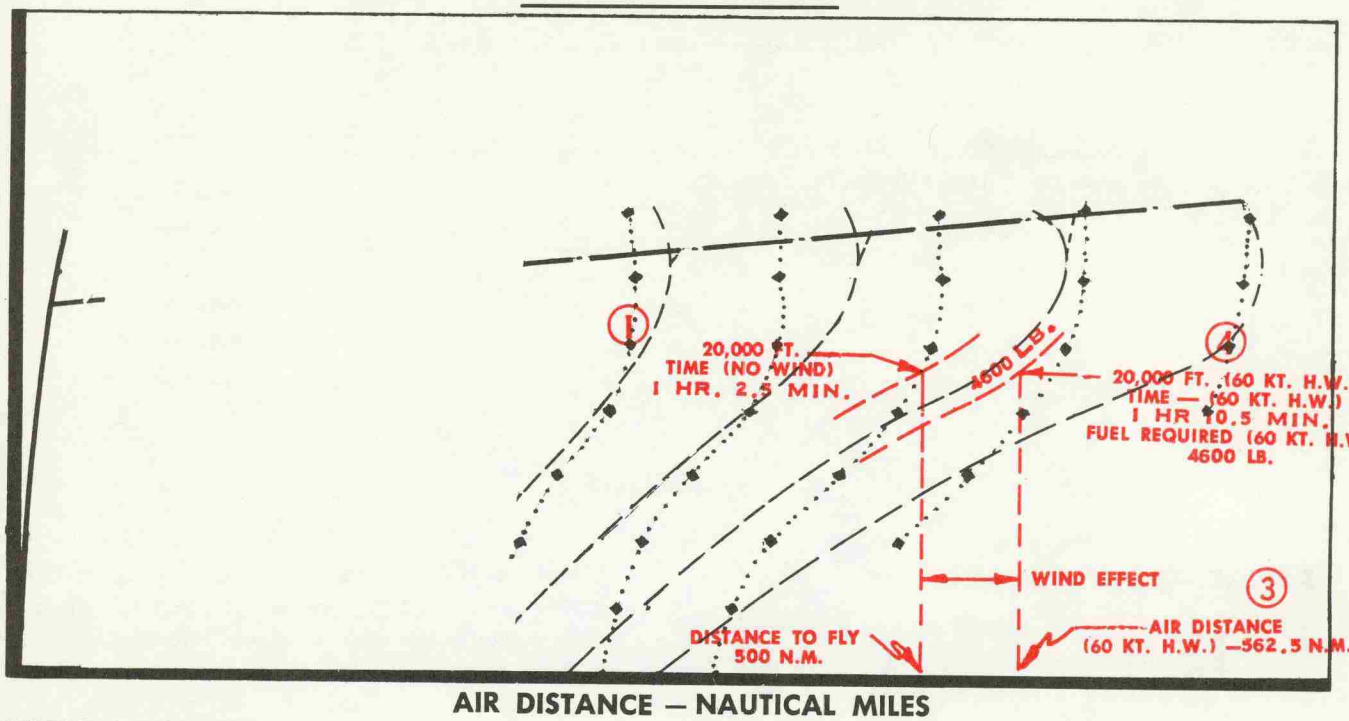
CHART ARRANGEMENT.

In general, the Mission Profile charts are read from left to right in a flight sequence consisting of an initial climb followed by cruising flight. The initial climb path is indicated by a solid line which shows the distance traveled in the climb from the point at which the initial climb speed was reached. The total range attainable (not including distance to accelerate to climb speed or to descend) is indicated for any desired value of fuel consumed and altitude by the dashed "pounds consumed" lines. For example: At a desired level flight cruise altitude of 20,000 feet and a range of 700 nautical air miles in the GAR-8 missile configuration (see Figure A4-3), a total of 5,500 pounds of fuel will be used from the time the engine is started until the aircraft is 700 nautical air miles away at 20,000 feet. Cruise at optimum cruise conditions (cruise-climb) would require 4460 pounds. The time required to climb to any altitude and cruise out is indicated by the dotted lines. This time is measured from the brake release point but does not include descent time.

TABULATED DATA.

Each Mission Profile chart includes tabulated speeds which apply to the climb and cruise portions of a flight. These indicated Mach number schedules must be used in order to obtain the aircraft performance shown. Indicated altitudes for cruise assume that the altimeter is set at 29.92 in. Hg and has no appreciable instrument error.

MISSION PROFILE



SAMPLE PROBLEM.

Determine the time, fuel required, and recommended cruise speed to cruise 500 nautical miles at a level flight cruise altitude of 20,000 feet on a standard day with a 60-knot headwind in the GAR-8 missile configuration (use Figure A4-3).

1. Enter profile at 500 nautical miles and 20,000 feet to obtain the no wind time . . . . . 1 hr. 2.5 min. (1.04 hr.)

2. Air distance due to wind (60 × 1.04) . . . . . 62.5 n.mi.

3. Total air distance with headwind (500 + 62.5) . . . . . 562.5 n.mi.

4. Re-enter profile with total air distance (with wind) and cruise altitude (20,000 feet) to obtain fuel required with wind . . . . . 4,600 lb. and time required with wind . . . . . 1 hr. 10.5 min.

5. Determine recommended indicated cruise speed from table . . . . . IND Mach 0.78

A suitable adjustment of indicated flight altitude is desirable if the instrument calibration is known. Engine cruise rpm and fuel flow values represent initial approximate settings for use in establishing the cruise phase of a flight. Non-standard air conditions and the operating characteristics of individual engines can have a decided effect on the settings that are necessary to maintain cruise airspeeds.

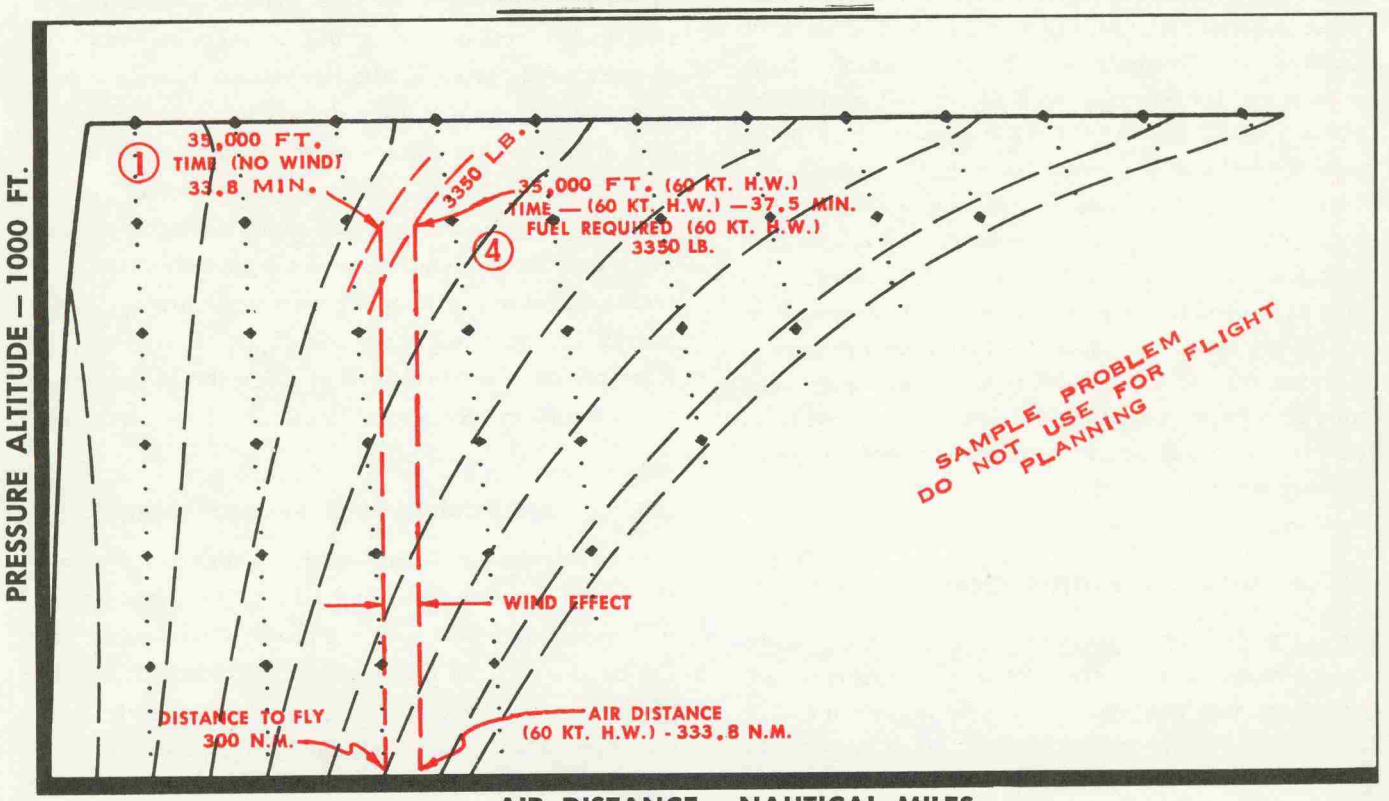
USE OF MISSION PROFILE FOR RETURN PERFORMANCE.

Return performance can be determined by use of the charts on an incremental basis for any fuel loading or fuel quantity used. Enter the Mission Profile chart at the cruise altitude and fuel consumed value to obtain initial distance and time readings. Add the desired cruise distance to the initial distance reading to obtain the final distance reading. Enter the chart at the final distance value and cruise altitude to obtain the final fuel and time consumed readings. The difference between the initial and final fuel and time values is the fuel and time required to cruise the desired distance.

WIND EFFECT ON CRUISE PERFORMANCE.

The effect of wind should be applied to the flight plan to obtain true fuel and time required to fly a given ground distance. Wind effect can be determined by entering the profile chart with total air distance with wind. Total air distance with wind may be approximated by adding the air distance due to wind (wind component parallel to course x time, no wind) to the desired cruise distance. A more accurate method of accounting for the effect of wind on range is as follows: Compute the average true

INTERCEPT PROFILE



SAMPLE PROBLEM.

Determine the time and fuel required to make an intercept at 35,000 ft. at a distance of 300 N. Miles. Assume that it is a standard day with a 60-knot headwind and that the aircraft is in the GAR-8 missile configuration. The aircraft has 5,902 lb. of fuel on board at engine start.

1. Enter the profile at 300 n.mi. and 35,000 ft. to obtain the no wind time . . . 33.8 min. (0.563 hr.)

2. Air distance due to wind
(60 × 0.563) . . . . . 33.8 n.mi.

3. Total air distance with headwind
(300 + 33.8) . . . . . 333.8 n. mi.

4. Re-enter the profile with total air distance (with wind) and intercept altitude (35,000 ft.) to obtain fuel required with wind . . . . . 3350 lb.
and time required with wind . . . . . 37.5 min.

5. Fuel remaining for maneuver and return
(5,902 − 3,350) . . . . . 2,552 lb.

6. Determine speed to intercept from table. . . .
. . . . . IND. Mach 0.93

7. RPM and EGT. . . . . 100% and 545°C

airspeed (TAS = distance ÷ no wind time) and correct for wind component to obtain average ground speed (GS = average TAS − headwind, or + tailwind). Compute the air distance to be flown by multiplying ground distance by (TAS/GS). Re-enter the profile chart at the air distance with wind and cruising altitude to obtain the fuel required with wind.

USE OF MISSION PROFILE CHARTS.

A sample problem is provided to insure proper understanding of the use of the Mission Profile range charts. The use of these charts is further illustrated by a sample chart.

INTERCEPT PROFILE CHART.

This type of profile chart shows the time and fuel necessary to fly a given distance in a minimum amount

of time. It is based on the use of Maximum Thrust for take-off and climb and Normal Thrust for cruise. Normal Thrust is obtained by setting 100% RPM and an EGT (Exhaust Gas Temperature) of 545°C. Profile charts are presented for operation with no external stores, with GAR-8 missiles and with GAR-8 missiles plus pylon tanks. The charts are based on a fully loaded gross weight of 19,000 pounds for the no external stores configuration, 19,420 pounds for the GAR-8 missile configuration and 22,514 pounds for the GAR-8 missile plus pylon tank configuration. The fuel used is presented in 500 lb. increments and includes fuel used for ground maneuver, take-off and acceleration to climb speed. The time values presented on these charts are measured from brake release. No allowance or reserve has been included for loiter, descent or landing.

USE OF INTERCEPT PROFILE CHART.

The Intercept Profile charts are similar to the Mission Profiles in use. The Intercept Profiles, however, should be restricted to flights where time is the important factor. The Mission Profiles are used for maximum range flight speed schedules. Flight planning with the Intercept Profile is accomplished in the same manner as with the Mission Profile. The tabulated data in the lower right hand corner is based on an initial cruise weight and standard temperature conditions. A sample chart shows how to use the Intercept Profile for flight planning.

OPTIMUM RETURN PROFILE CHART.

The Optimum Return Profile chart shows the maximum range which can be obtained with any amount of fuel remaining at a given altitude by making a Military Thrust climb to cruise-climb altitude and then cruise-climbing until all fuel is expended. The chart does not show level flight cruise range except for the case when the fuel quantity remaining is so small that there is no advantage in climbing to cruise-climb altitude. This type of profile is useful for planning the return from a mission and for maximum radius flight. No fuel allowance is made in the charts for loiter, descent or landing. Wind effects as discussed on the Mission Profile chart also apply to the Optimum Return Profile.

OPTIMUM RETURN LEVEL FLIGHT RANGE.

There is no advantage in climbing to cruise-climb altitude with small amounts of fuel on board because an equal or greater range can be obtained at the existing or intermediate flight altitudes. This is shown by the two shaded areas on the profiles. In one area, you climb to the altitude given by the line of best range for constant altitude flight (heavy dashed line) and cruise in level flight at this altitude at the recommended Mach number shown. In the other area, you maintain present altitude and cruise out in level flight at the recommended Mach number. In situations where large amounts of fuel are on board and a level flight cruise return is desired instead of the cruise-climb technique, the return performance may be determined by using the Mission Profile chart on an incremental basis.

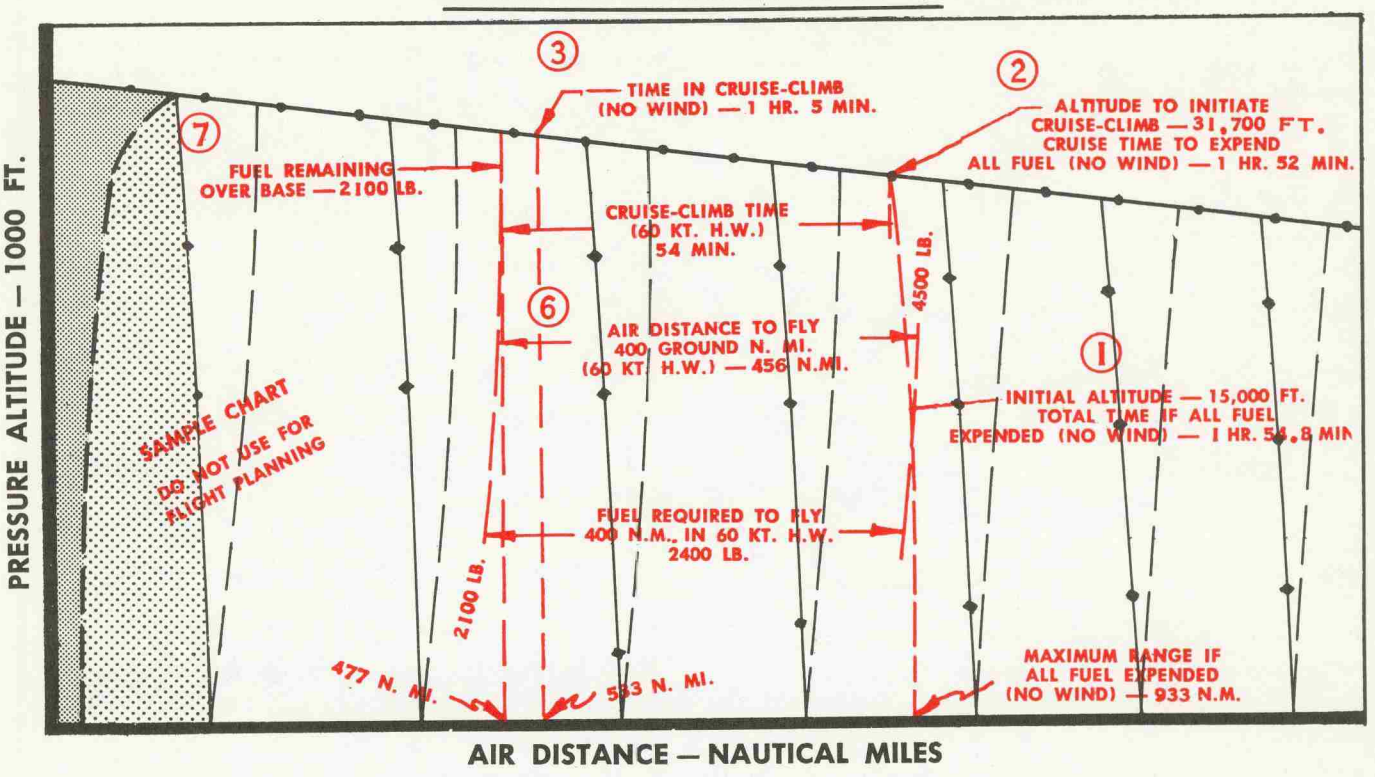
USE OF OPTIMUM RETURN PROFILE CHARTS.

You can determine the maximum possible range attainable from a given point and with a given amount of fuel remaining by entering the proper Optimum Return Profile chart at the fuel remaining and actual altitude of the aircraft. Read the range available directly from the chart. Time of flight is the sum of time to climb (if climb is necessary) plus the time at cruise altitude. The area on the chart in which the initial condition falls determines the cruising procedure to use as stated in the note for each area. Use the chart in a similar manner to obtain the fuel required to fly a given distance which is less than the maximum range available. Reserve range available becomes that distance in excess of the desired distance to be flown, or maximum range available minus desired flight distance. Note that the effect of wind must be applied to obtain the true fuel and time required to fly a given ground distance. Multiply ground distance to be flown by (TAS/GS) to find the corresponding chart value applicable for the existing wind.

NAUTICAL MILES PER POUND OF FUEL CHARTS.

The Nautical Miles per Pound of Fuel charts present cruise data throughout the speed range from maximum endurance to Military Thrust. Gross weights encompassing the maximum and minimum probable flight weights are given at altitudes from sea level up to the combat ceiling of the aircraft for all normal aircraft configurations. Each chart includes specific range (nautical miles per pound) which will result from the various speed settings, fuel flow, and standard day engine rpm settings.

OPTIMUM RETURN PROFILE



SAMPLE PROBLEM.

An example problem is provided to show the use of the optimum return charts. A sample chart is presented with the problem as a further aid in understanding the function of the charts.

Determine the fuel and time required to fly to a base 400 nautical miles away in a 60-knot headwind. The aircraft is in the GAR-8 missile configuration and is at 15,000 feet with 4500 pounds of fuel remaining. Use Figure A4-9.

1. Enter the profile at 15,000 feet and 4,500 pounds of fuel on board to establish initial point. In this area a Military Thrust climb followed by cruise-climb is required. Read directly the maximum range attainable by expending all fuel (zero wind) . . . . . .933 n.mi.

2. Follow the climb path guide lines to the line for cruise-climb flight and read altitude to initiate cruise-climb procedure . . . . . .31,700 ft.

Also read the time in cruise-climb to expend all fuel (zero wind) . . . . . .1 hr. 52 min.

Time to climb from 15,000 ft. to 31,700 ft. . 2.8 min.

Total time to climb and cruise-climb until all fuel is expended (zero wind)
(1 hr. 52 min. + 2.8 min.) . . . . . .1 hr. 54.8 min.

3. Enter profile at (933 n.mi. — 400 n.mi.) 533 nautical miles and intersect cruise-climb flight path. Read time in cruise-climb (zero wind). . . 1 hr. 5 min.

4. Time required to fly 400 nautical miles in zero wind . . . . . .(1 hr. 54.8 min. — 1 hr. 5 min.)
0 hr. 49.8 min. (0.83 hr.)

5. Average true airspeed for 400 miles
(400 ÷ 0.83) = 481 kts.
Airspeed ÷ ground speed (481 ÷ 421) 1.14

6. Total air distance with wind
(400 × 1.14) . . . . . .456 n.mi.

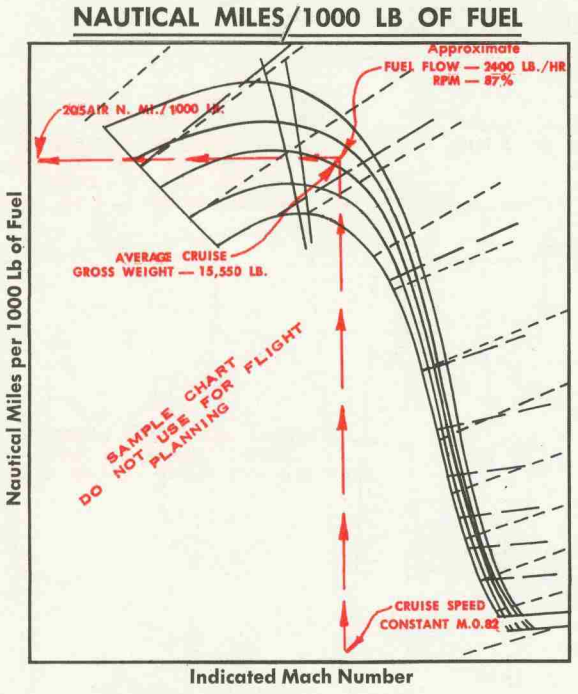
7. Re-enter profile at (933 — 456). . . .477 n.mi. and intersect cruise-climb flight path. Read fuel remaining over base (with wind) . . . . . .2,100 lb.

Fuel used for return leg (4500 — 2,100). . 2,400 lb.

Subtract final time reading (58 min.) from time reading to expend all fuel (1 hr. 54.8 min.) to obtain time for leg . . . . . .56.8 min.

Cruise-climb time (with wind)
(56.8 min. — 2.8 min.) . . . . . .54 min.

Note that the power setting to use is that given for cruise-climb. (The value given for 30,000 ft. applies only to level flight with less than 250 lb. of fuel remaining.



SAMPLE PROBLEM.

Determine the range available, time required, fuel flow, and RPM to cruise in level flight at a constant true Mach 0.84 at 30,000 ft. The initial cruise gross weight is 17,000 lb. and a 1000 lb. reserve is desired. Assume that it is a standard day with a 60 knot headwind and the airplane is in the No External Stores configuration. The empty gross weight is 13,098 lb.

1. True Mach Number 0.84

2. From Figure A1-2 determine Indicated Mach. 0.82

3. True Airspeed (from Fig. A4-12) at Indicated Mach of 0.82 . . . . . 493 knots

4. Ground Speed, V_g, (493 kts. — 60 kts.) . . . . . 433 kts.

5. Empty gross weight . . . . . 13,098 lb.

6. Gross weight with 1000 lb. reserve (13,098 lb. + 1000 lb.) . . . . . 14,098 lb.

7. Average cruise gross wt. ([17,000 lb. + 14,098 lb.]/2) . . . . . 15,550 lb.

8. Enter the chart at Mach 0.82 Ind. and 15,550 lb. Read average air n.mi./1000 lb. of fuel . . . . . 205 n.mi./1000 lb.

9. Determine average ground n.mi/1000 lb. by using the formula: ground n.mi./1000 = air n.mi. × V_g/V_t (205 × 433/493) . . . . . 180 N.M./1000 lb.

10. Fuel to be used in cruise (17,000 lb. — 14,098 lb.) . . . . . 2,902 lb.

11. Range available (180 × 2902/1000) . . . . . 521 n.mi.

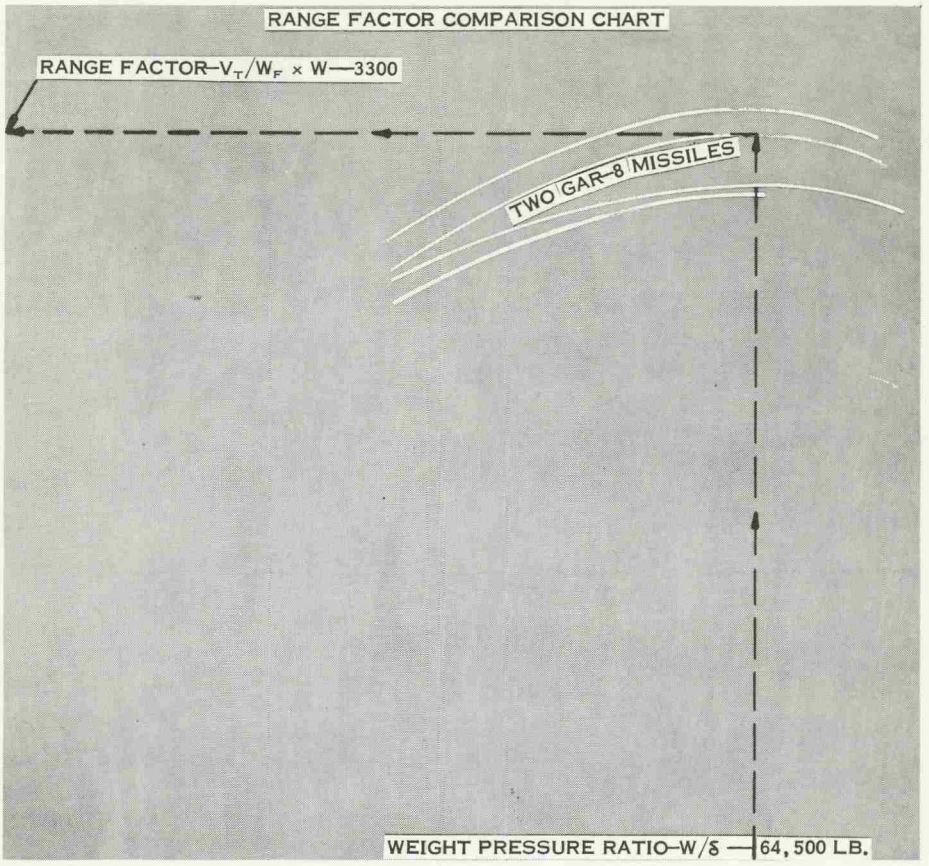
12. Time required (521/433) . . . . . 1.20 hr. (1 hr. 12 min.) . . . . . 0.94 hr. (56.4 min.)

13. From chart read fuel flow and RPM . . . . . approximately 87% RPM . . . . . approximately 2400 lb/hr.

Also included are curves of recommended cruise Mach number and speeds for maximum endurance. Specific range is plotted vs. True Airspeed (TAS) with subscales of Indicated Mach Number and Indicated Airspeed (IAS).

USE OF NAUTICAL MILES PER POUND OF FUEL CHARTS.

A sample problem is provided to insure proper understanding of the use of the Nautical Miles per Pound of Fuel Chart. The use of these charts is further illustrated by a sample chart.



SAMPLE PROBLEM.

Determine the range available and the cruising altitude for an optimum cruise flight. The aircraft is in the GAR-8 Missile configuration. The initial cruising weight is 17,000 pounds and 2,000 lb. of fuel are to be used. It is a standard day with zero wind.

1. Weight at start of cruise . . . . . 17,000 lb.
2. Weight at end of cruise (17,000 lbs. — 2000 lb.) . . . . . 15,000 lb.
3. Avg. gross weight during cruise . . . 16,000 lb.
4. From Figure A4-36 the optimum cruise W/δ for Two GAR-8 Missiles is . . . 64,500 lb. The range factor is . . . . . 3,300 lb.mi/lb.
5. The range available is

$$(3000 \times \frac{2000}{1600} \ldots\ldots\ldots\ldots\ldots 412 \text{ n.mi.}$$

6. At the initial cruise gross weight (17,000 lb.) the pressure ratio δ (P/P_0) is (17,000/64,500) 0.264
7. From the standard altitude table (Figure A1-8 the pressure altitude or a δ (P/P_0) of 0.264 is approximately . . . . . 32,500 Ft. This is the altitude for the beginning of the cruise-climb at a gross weight of 17,000 pounds.
8. By the same method, the final altitude for the cruise-climb at a gross weight of 15,000 lb. is approximately . . . . . 35,200 Ft.
9. Cruise speed for the flight is 0.833 True Mach.

RANGE FACTOR CHART.

The range factor is defined as $V_T/W_f \times W$ (Pound miles per pound of fuel used). This number when divided by the average gross weight during the cruise and multiplied by the fuel to be used during cruise gives the range available in a cruise-climb at the W/δ which corresponds to the range factor (δ is the pressure ratio P/P_0). In equation form:

$$\text{Range Available} = \frac{\text{Range factor} \times \text{Fuel to be used in cruise}}{\text{Avg. gr. wt. during cruise}}$$

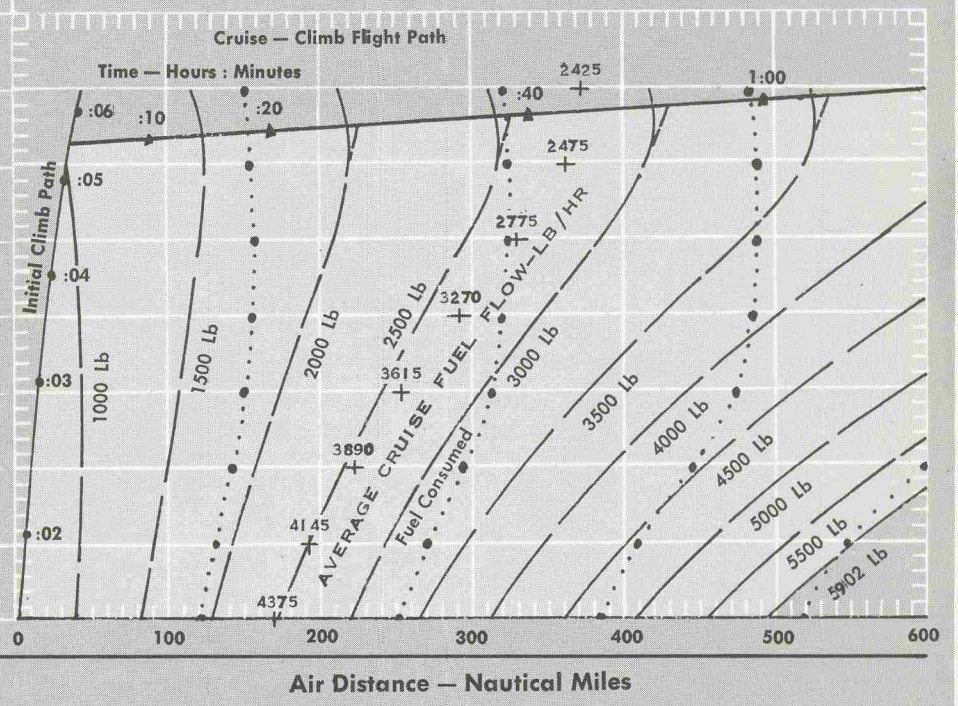
Range factors are presented for values of W/δ which cover the subsonic operational range of the aircraft. The purpose of this chart is to show the optimum cruise W/δ for each configuration and the corresponding Mach schedule and range factor. The chart also allows a rapid comparison of range factors and W/δ's for the following configurations: No External Stores, Two 165 Gallon Tip Tanks, Two GAR-8 Missiles, Two 200 Gallon Pylon Tanks and Two GAR-8 Missiles, and Two 200 Gallon Pylon Tanks and Two 165 Gallon Tip Tanks.

MISSION PROFILE

Climb Data			
Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	573	S.L.

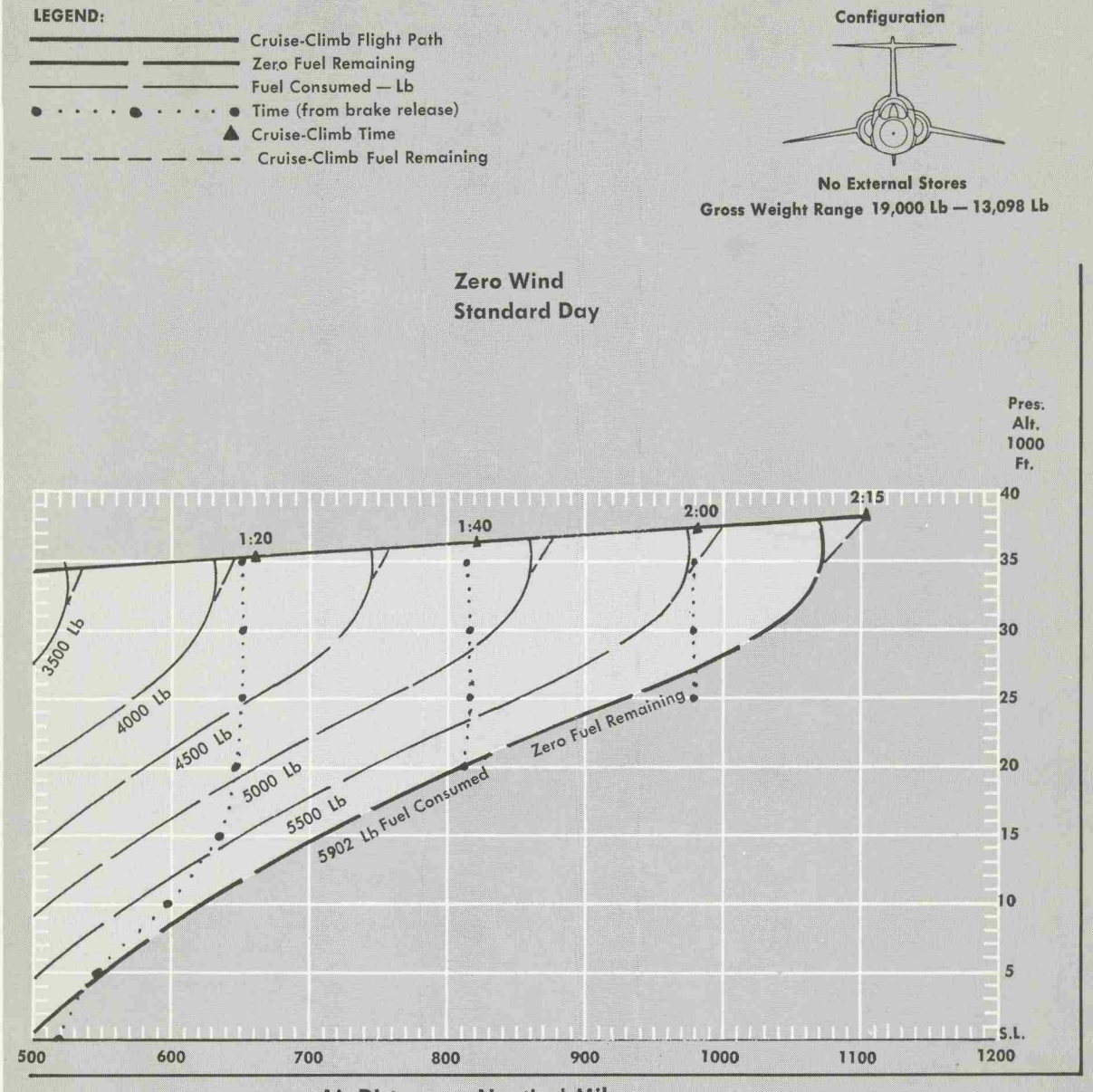
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



NOTE:
1. Fuel allowance for ground maneuver, Maximum Thrust take-off to 350 Knots, and Military Thrust acceleration from 350 Knots IAS to climb speed —570 Lb.
2. No allowance made for loiter, descent or landing.
3. Use Military Thrust for climb. (See Military Thrust Climb Chart for detailed information.)
4. Cruise at recommended Mach number.

Figure A4-1 (Sheet 1 of 2)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
Optimum Cruise (Cruise-Climb)		.86	.84	310–260	505–490	90.0–88.0	2710–1930
35,000	34,510	.86	.84	290	495	93.5	2735
30,000	29,550	.84	.82	315	495	89.0	2670
25,000	24,570	.82	.80	340	495	88.0	2900
20,000	19,600	.81	.79	385	495	88.0	3240
15,000	14,620	.78	.77	390	485	88.5	3700
10,000	9,640	.71	.70	390	455	88.5	4050
5,000	4,650	.64	.63	385	415	88.5	4310
S. L.	—210	.60	.59	395	400	88.5	4520

* Initial Approximate Values

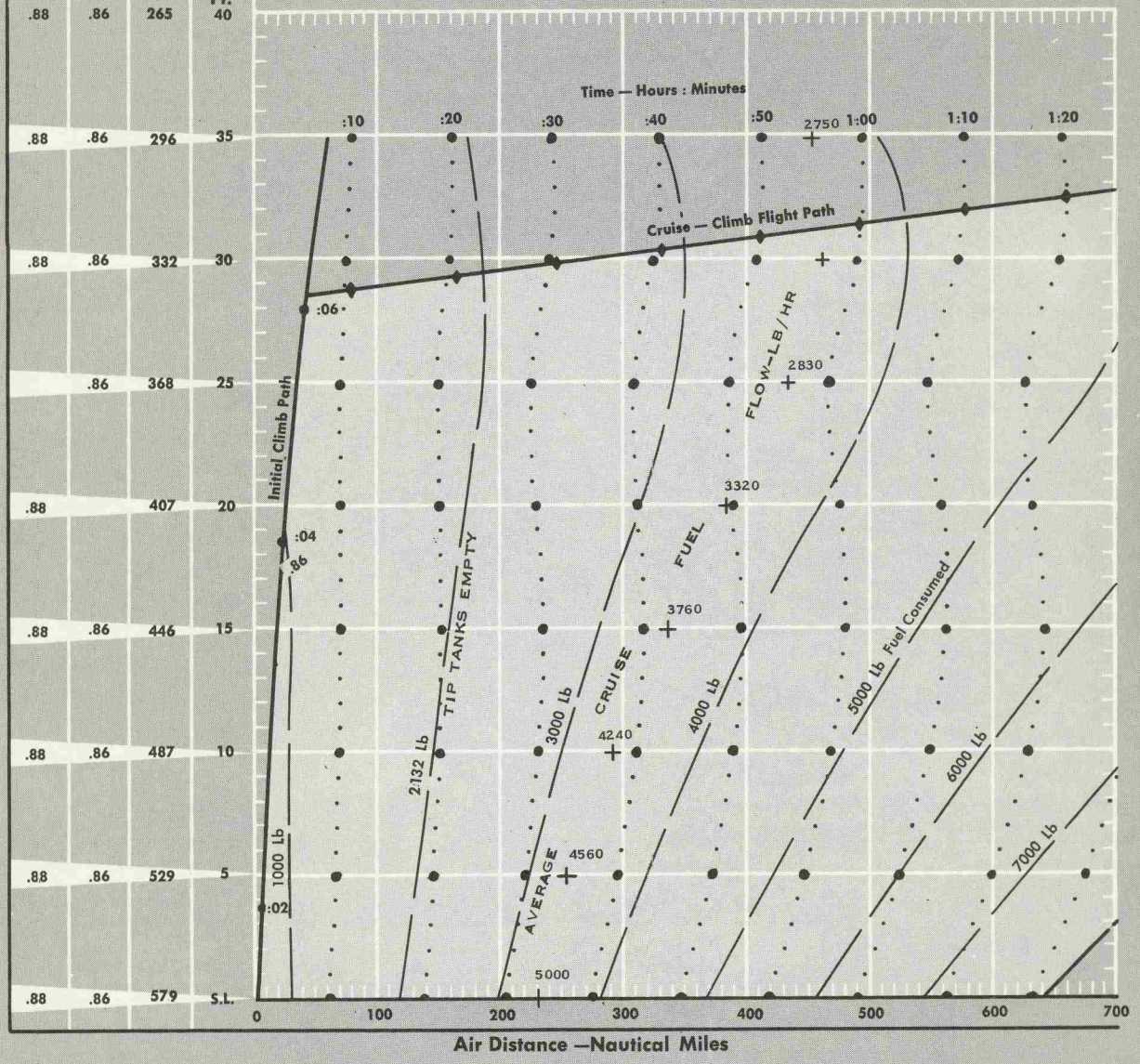
Figure A4-1 (Sheet 2 of 2)

MISSION PROFILE

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

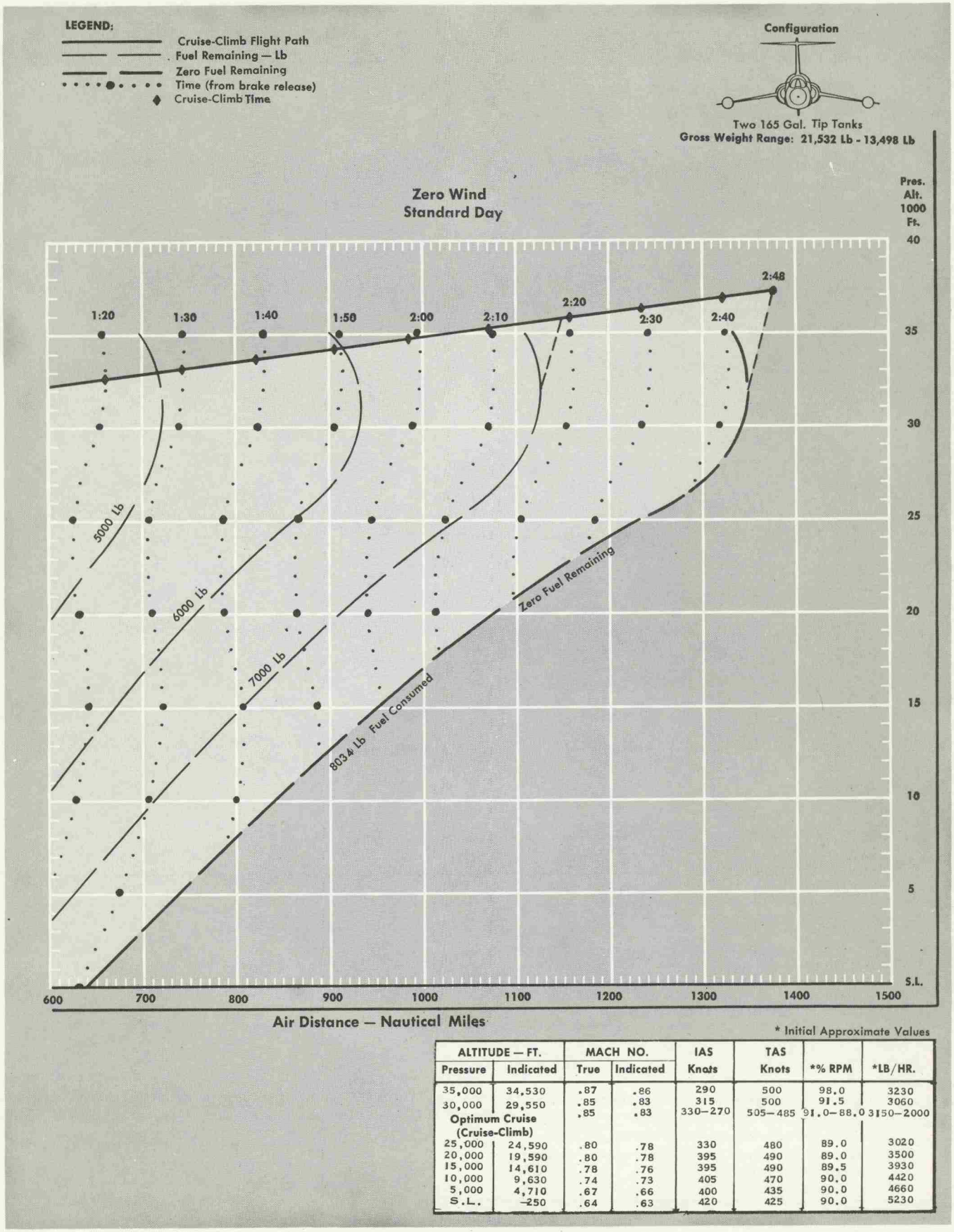
Climb Data — Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
	.86	368	25
.88		407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	579	S.L.



NOTE:

1. Fuel allowance for ground maneuver, Maximum Thrust take-off to 350 Knots, and Military Thrust acceleration from 350 Knots IAS to climb speed — 650 Lb.
2. No allowance made for loiter, descent or landing.
3. Use Military Thrust for climb. (See Military Thrust Climb Chart for detailed information.)
4. Cruise at recommended Mach number.

Figure 4-2 (Sheet 1 of 2)



* Initial Approximate Values

ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
35,000	34,530	.87	.86	290	500	98.0	3230
30,000	29,550	.85	.83	315	500	91.5	3060
Optimum Cruise (Cruise-Climb)		.85	.83	330–270	505–485	91.0–88.0	3150–2000
25,000	24,590	.80	.78	330	480	89.0	3020
20,000	19,590	.80	.78	395	490	89.0	3500
15,000	14,610	.78	.76	395	490	89.5	3930
10,000	9,630	.74	.73	405	470	90.0	4420
5,000	4,710	.67	.66	400	435	90.0	4660
S.L.	–250	.64	.63	420	425	90.0	5230

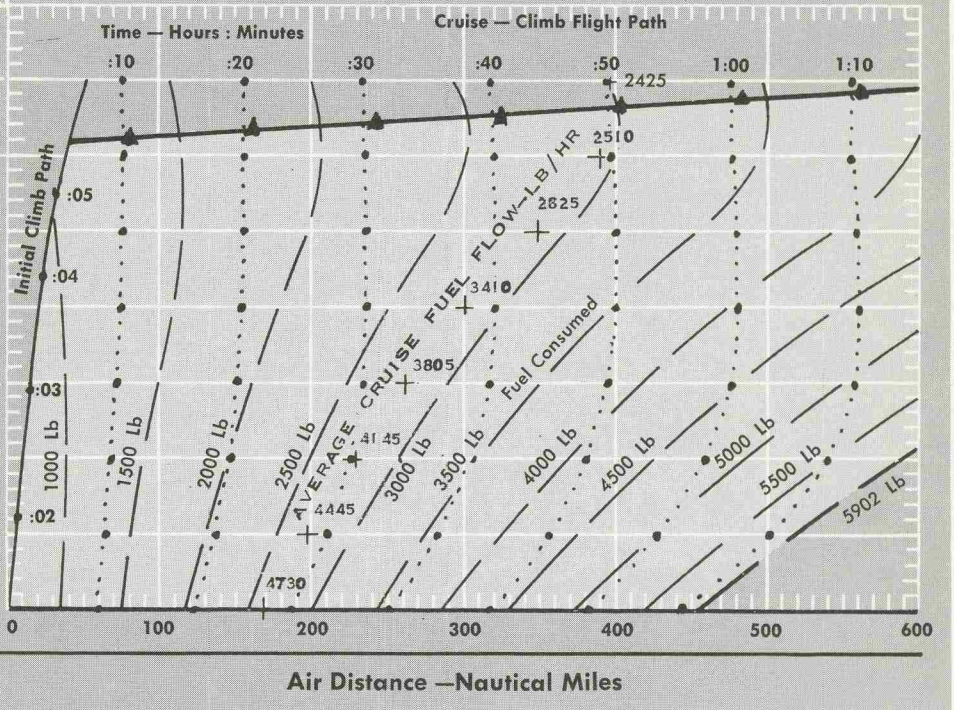
Figure 4-2 (Sheet 2 of 2)

MISSION PROFILE

Climb Data			
Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	579	S.L.

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



NOTE:

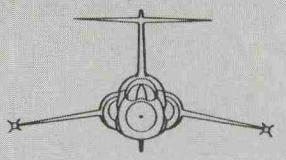
1. Fuel allowance for ground maneuver, Maximum Thrust take-off to 350 Knots, and Military Thrust acceleration from 350 Knots IAS to climb speed — 600 Lb
2. No allowance made for loiter, descent or landing.
3. Use Military Thrust for climb. (See Military Thrust Climb Chart for detailed information.)
4. Cruise at recommended Mach number.

Figure A4-3 (Sheet 1 of 2)

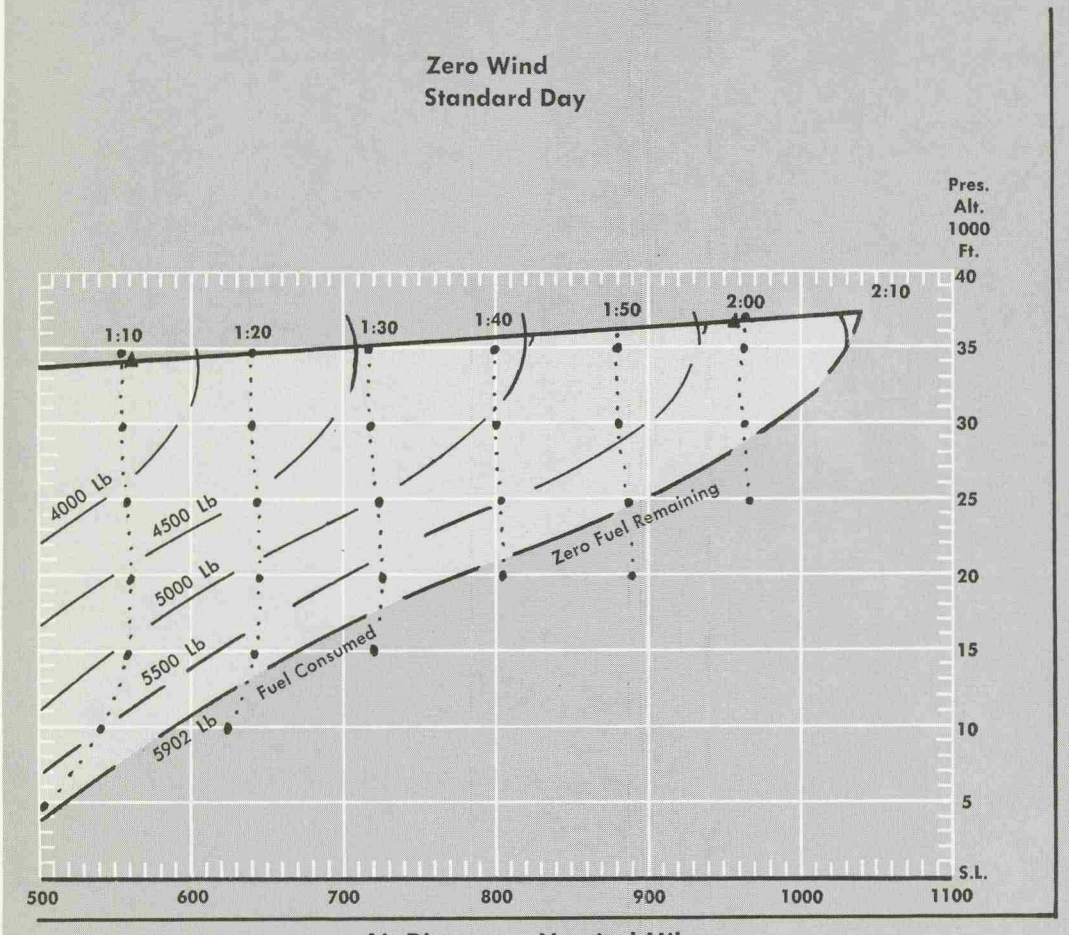
LEGEND:

- Cruise-Climb Flight Path
- Zero Fuel Remaining
- Fuel Consumed — Lb
- Time (from brake release)
- Cruise-Climb Time
- Cruise-Climb Fuel Remaining

Configuration



Two GAR-8 Missiles
Gross Weight Range 19,420 Lb – 13,518 Lb



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
35,000	34,600	.84	.82	285	485	94.0	2780
Optimum Cruise (Cruise-Climb)		.83	.81	305–265	490–475	91.0–88.0	2710–1960
30,000	29,600	.82	.80	310	485	88.5	2750
25,000	24,600	.81	.79	335	490	89.0	2980
20,000	19,600	.80	.78	370	490	89.0	3420
15,000	14,620	.77	.75	390	485	89.5	3820
10,000	9,650	.73	.72	405	465	89.5	4410
5,000	4,740	.65	.64	395	425	90.0	4600
S. L.	—210	.60	.59	390	395	90.0	4930

* Initial Approximate Values

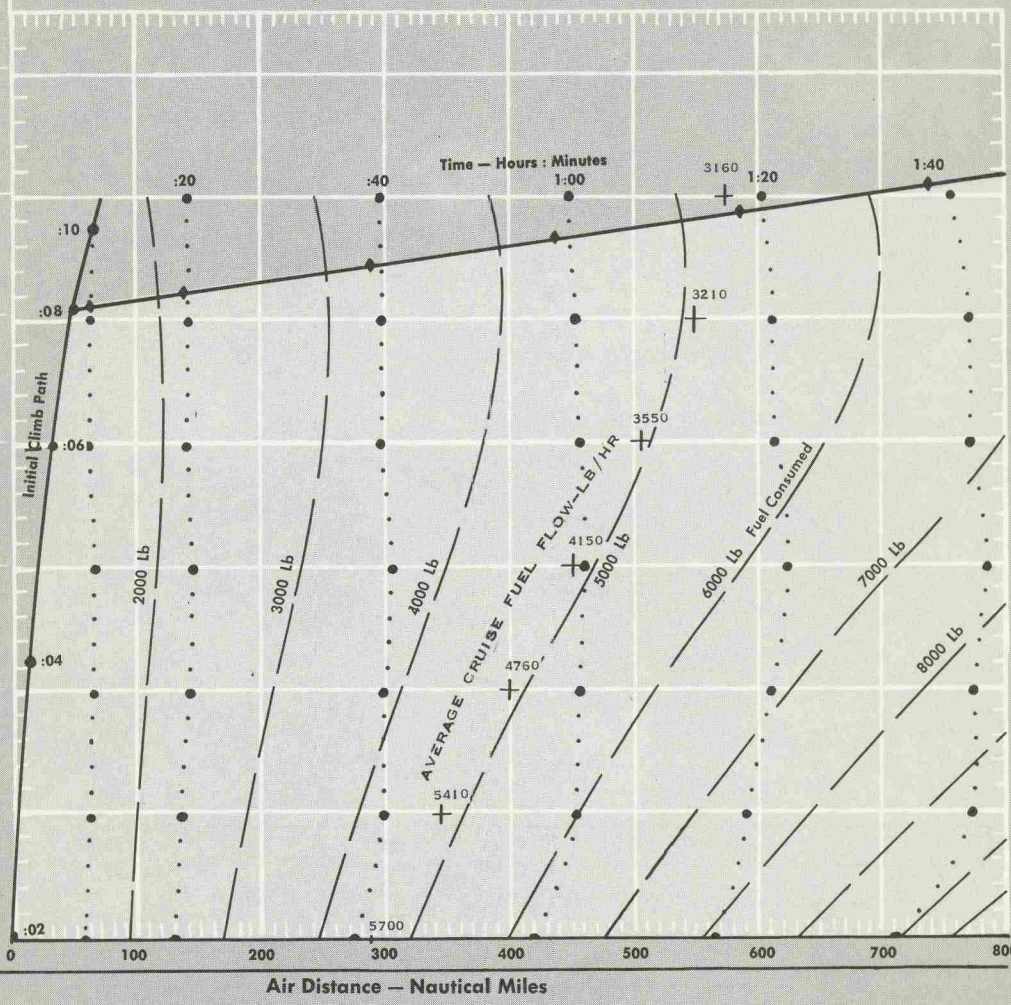
Figure A4-3 (Sheet 2 of 2)

MISSION PROFILE

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

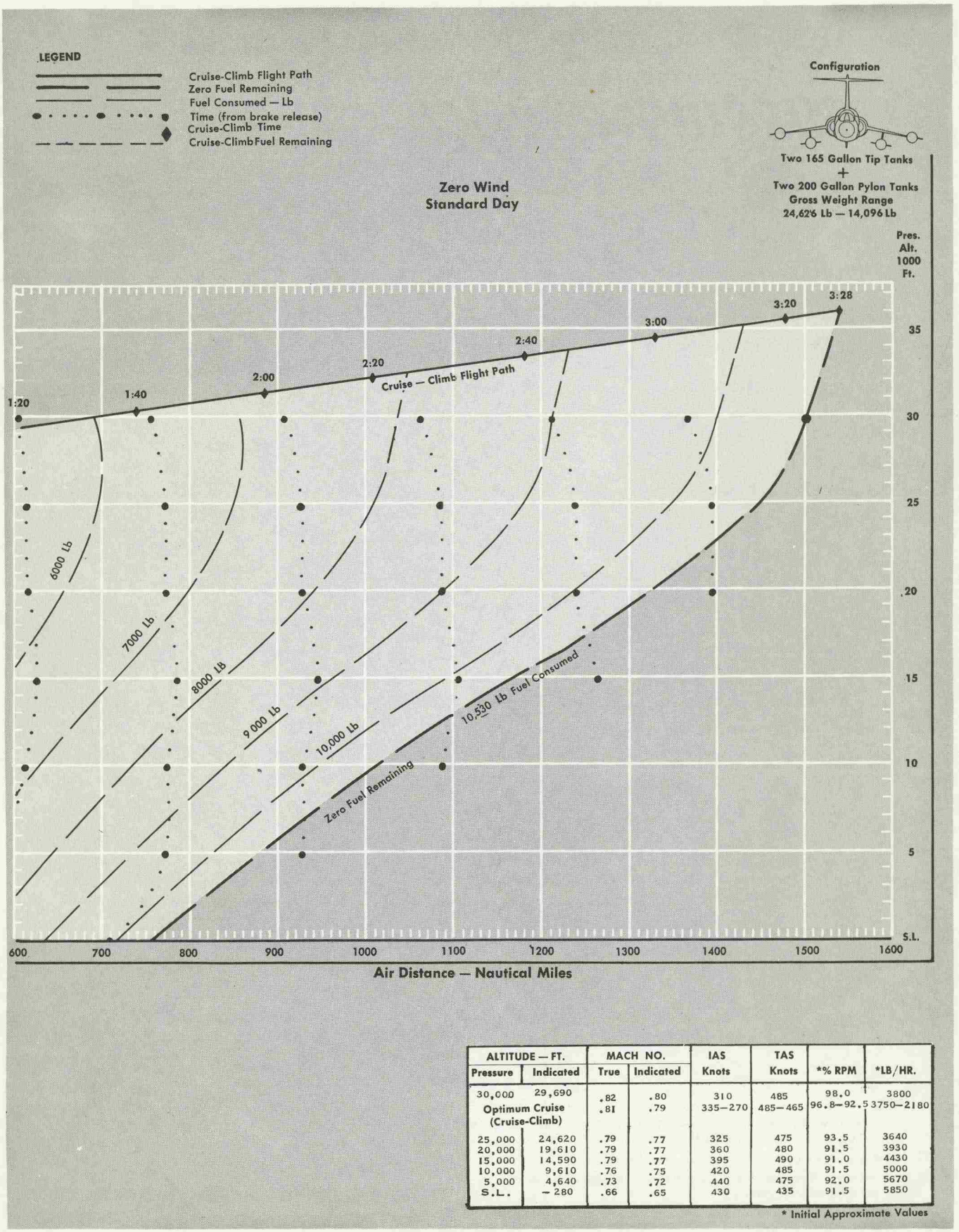
Climb Data — Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.825	.81	277	35
.825	.81	309	30
.825	.81	343	25
.825	.81	379	20
.825	.81	417	15
.825	.81	456	10
.825	.81	496	5
.825	.81	538	S.L.



NOTE:

1. Fuel allowance for ground maneuver, Maximum Thrust take-off to 350 Knots, and Military Thrust acceleration from 350 Knots IAS to climb speed — 740 Lb.
2. No allowance made for loiter, descent or landing.
3. Use Military Thrust for climb. (See Military Thrust Climb Chart for detailed information.)
4. Cruise at recommended Mach number.

Figure A4-4 (Sheet 1 of 2)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
30,000	29,690	.82	.80	310	485	98.0	3800
Optimum Cruise (Cruise-Climb)		.81	.79	335—270	485—465	96.8—92.5	3750—2180
25,000	24,620	.79	.77	325	475	93.5	3640
20,000	19,610	.79	.77	360	480	91.5	3930
15,000	14,590	.79	.77	395	490	91.0	4430
10,000	9,610	.76	.75	420	485	91.5	5000
5,000	4,640	.73	.72	440	475	92.0	5670
S.L.	— 280	.66	.65	430	435	91.5	5850

* Initial Approximate Values

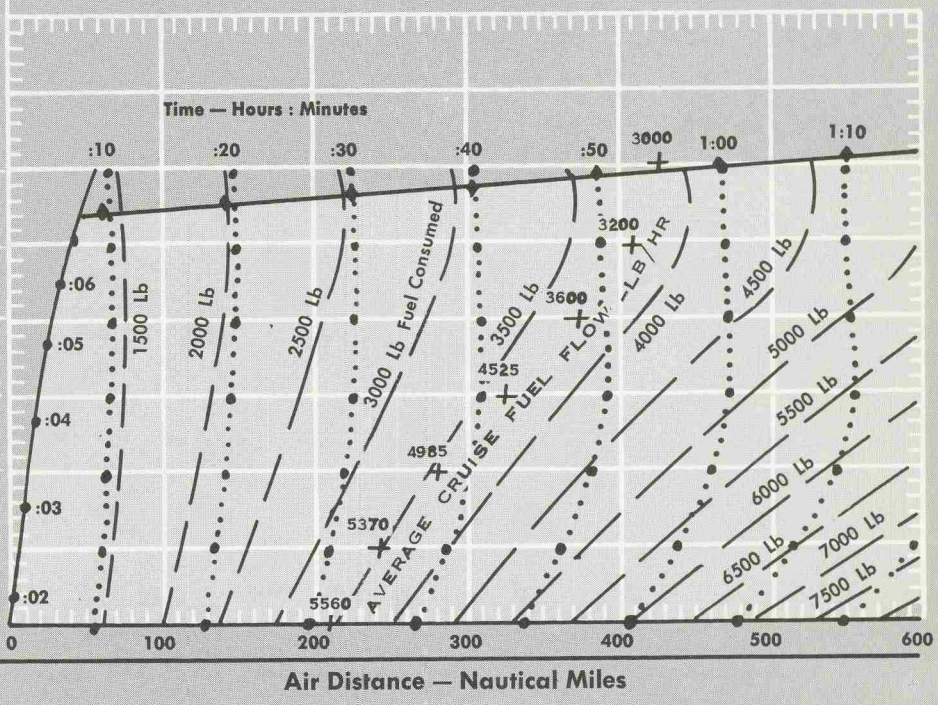
Figure A4-4 (Sheet 2 of 2)

MISSION PROFILE

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

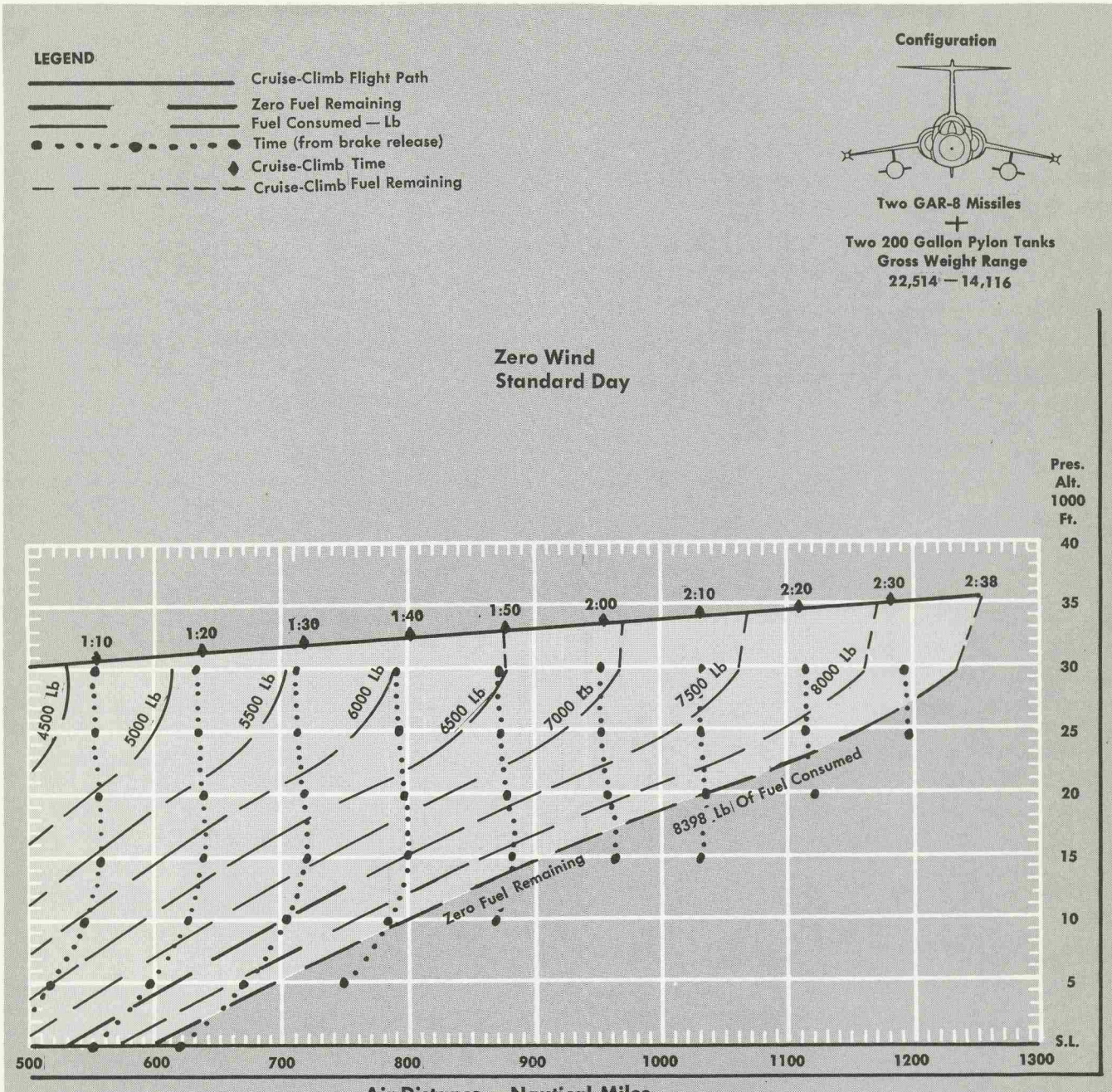
Climb Data — Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.825	.810	247	40
.825	.810	277	35
.825	.810	309	30
.825	.810	343	25
.825	.810	379	20
.825	.810	417	15
.825	.810	456	10
.825	.810	496	5
.825	.810	538	S.L.



NOTE:

1. Fuel allowance for ground maneuver, Maximum Thrust take-off to 350 Knots, and Military Thrust acceleration from 350 Knots IAS to climb speed — 666 Lb
2. No allowance made for loiter, descent or landing.
3. Use Military Thrust for climb. (See Military Thrust Climb Chart for detailed information.)
4. Cruise at recommended Mach number.

Figure A4-5 (Sheet 1 of 2)



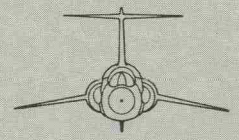
ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
30,000	29,590	.82	.80	305	480	95.0	3410
Optimum Cruise (Cruise-Climb)		.82	.80	325–270	490–470	94.0–90.0	3460–2230
25,000	24,590	.81	.79	335	485	92.0	3470
20,000	19,600	.79	.78	365	485	91.0	3800
15,000	14,600	.78	.77	395	490	91.0	3750
10,000	9,680	.75	.74	415	480	91.0	3750
5,000	4,670	.71	.70	425	460	91.0	5270
S. L.	—270	.64	.63	420	425	91.5	5790

* Initial Approximate Values

Figure A4-5 (Sheet 2 of 2)

INTERCEPT PROFILE

Configuration



No External Stores
Gross Weight Range
19,000 Lb — 13,098 Lb

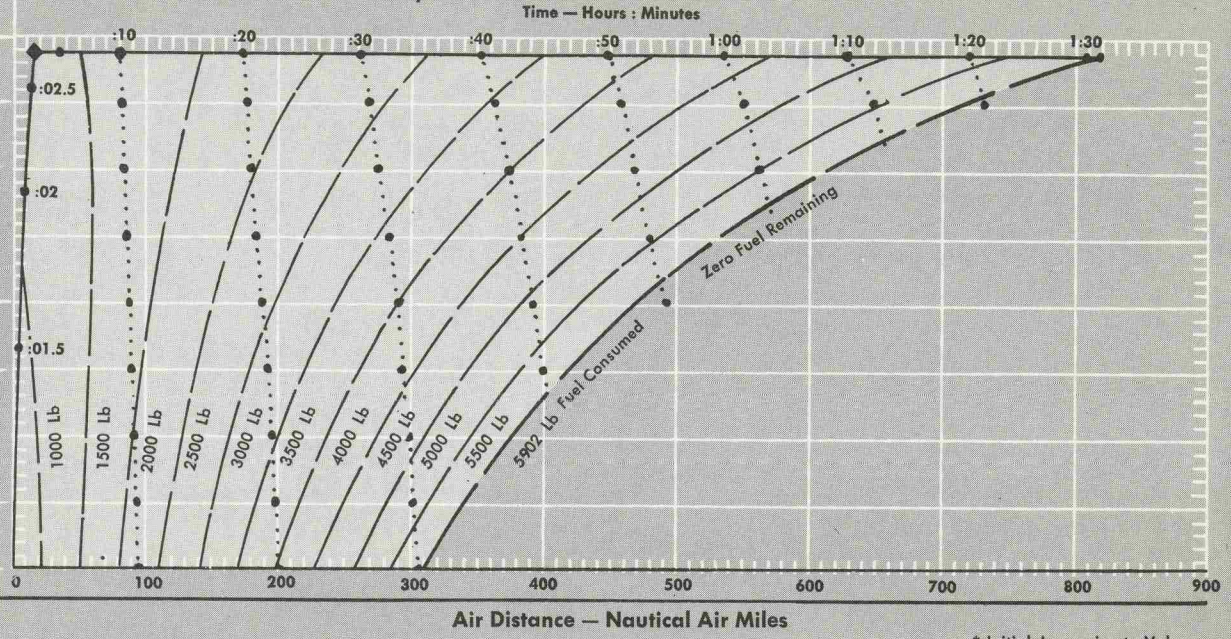
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

LEGEND:
- Zero Fuel Remaining
- Fuel Consumed
- Time (from brake release)
- Normal Thrust Cruise Ceiling

Zero Wind Standard Day

Climb Data — Maximum Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.925	.90	278	40
.925	.90	313	35
.925	.90	350	30
.925	.90	388	25
.925	.90	430	20
.925	.90	470	15
.925	.90	513	10
.925	.90	556	5
.925	.90	602	S.L.



* Initial Approximate Values

ALTITUDE — FT.		MACH NO.		IAS	TAS			EGT
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR	° C
38,700	37,650	.95	.92	295	545	100	3340	545
35,000	34,250	.98	.94	335	565	100	3750	545
30,000	29,210	.98	.94	375	580	100	4550	545
25,000	24,170	.98	.94	415	590	100	5440	545
20,000	19,150	.98	.94	455	600	100	6540	545
15,000	14,120	.98	.94	495	610	100	7740	545
10,000	8,800	.97	.94	535	620	100	8500	545
5,000	3,480	.96	.93	575	625	100	10140	545
S. L.	—1,350	.95	.92	620	630	100	10760	545

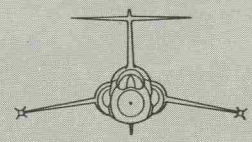
NOTE:

1. Fuel allowance for ground maneuver and Maximum Thrust take-off and acceleration to climb speed — 676 Lb.
2. No Allowance or reserve made for loiter, descent or landing.
3. Use Maximum Thrust for climb. (See Maximum Thrust Climb Chart for detailed information.)
4. Cruise at Normal Thrust (100% RPM) and EGT of 545° C.
5. No distance credit is allowed for time necessary to accelerate to climb speed.

Figure A4-6

INTERCEPT PROFILE

Configuration



Two GAR-8 Missiles
Gross Weight Range
19,420 Lb – 13,518 Lb

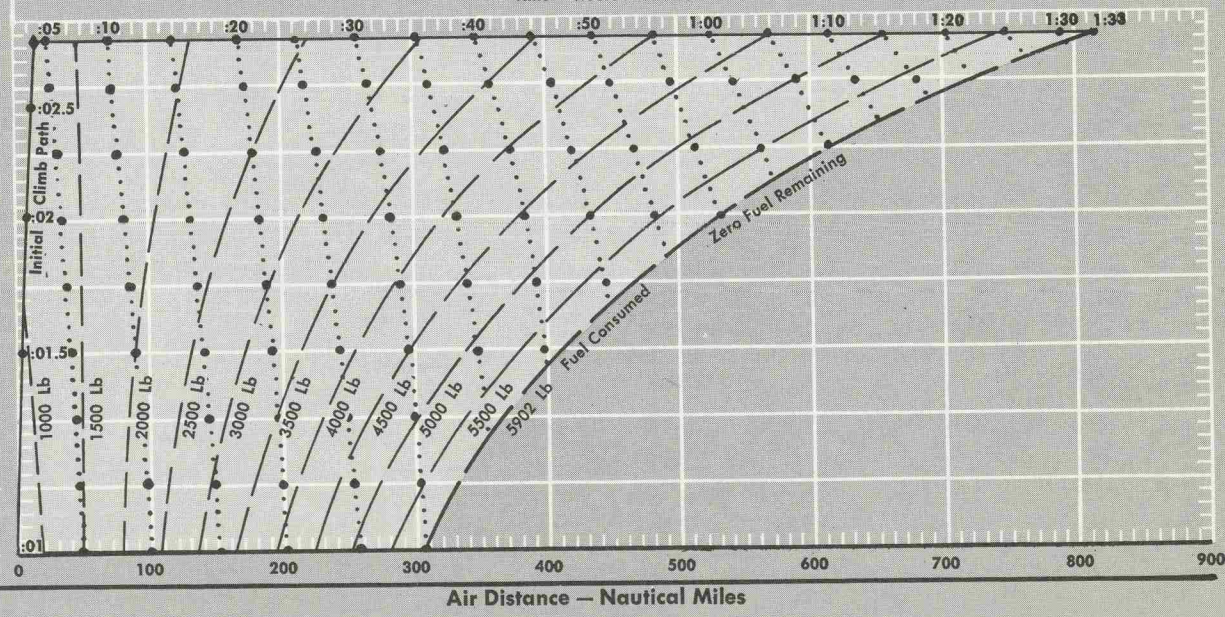
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

LEGEND:
- Zero Fuel Remaining
- Fuel Consumed
- Time (from brake release)
- Normal Thrust Cruise Ceiling

Climb Data — Maximum Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.925	.90	278	40
.925	.90	313	35
.925	.90	350	30
.925	.90	388	25
.925	.90	430	20
.925	.90	470	15
.925	.90	513	10
.925	.90	556	5
.925	.90	602	S.L.

Zero Wind Standard Day



* Initial Approximate Values

ALTITUDE — FT.		MACH NO.		IAS	TAS			EGT
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR	° C
38,400	37,860	.93	.90	290	535	100	3290	545
35,000	34,280	.96	.93	325	555	100	3690	545
30,000	29,210	.98	.94	370	575	100	4530	545
25,000	24,030	.97	.94	410	585	100	5350	545
20,000	19,000	.97	.94	450	600	100	6300	545
15,000	14,100	.96	.93	490	600	100	7300	545
10,000	9,080	.96	.93	535	610	100	8300	545
5,000	4,240	.94	.91	565	610	100	9400	545
S. L.	—720	.93	.90	605	615	100	10600	545

NOTE:

1. Fuel allowance for ground maneuver and Maximum Thrust take-off and acceleration to climb speed — 676 Lb.
2. No Allowance or reserve made for loiter, descent or landing.
3. Use Maximum Thrust for climb. (See Maximum Thrust Climb Chart for detailed information.)
4. Cruise at Normal Thrust (100% RPM) and EGT of 545° C.
5. No distance credit is allowed for time necessary to accelerate to climb speed.

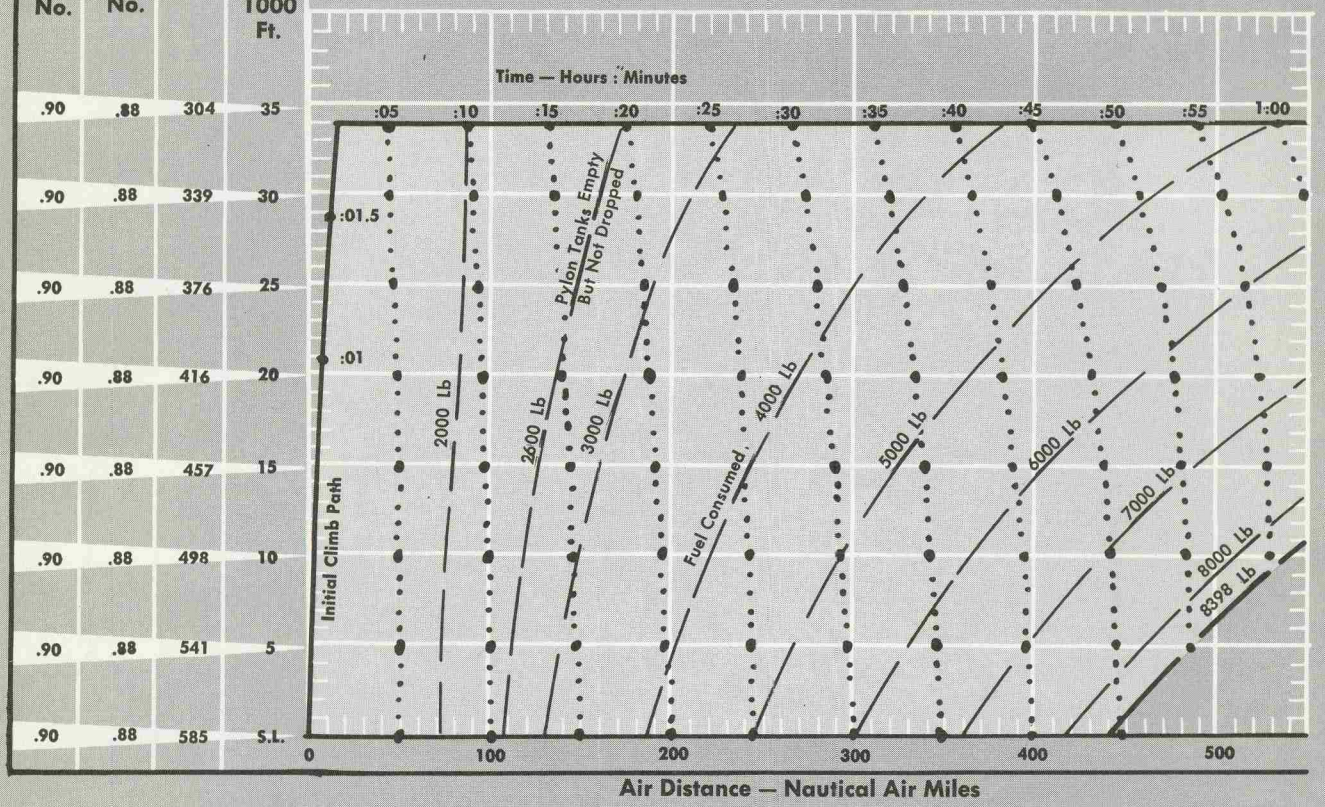
Figure A4-7

INTERCEPT PROFILE

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Climb Data — Maximum Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.90	.88	304	35
.90	.88	339	30
.90	.88	376	25
.90	.88	416	20
.90	.88	457	15
.90	.88	498	10
.90	.88	541	5
.90	.88	585	S.L.



NOTE:

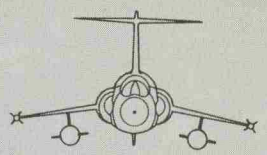
1. Fuel allowance for ground maneuver and Maximum Thrust take-off and acceleration to climb speed —765 Lb
2. No Allowance or reserve made for loiter, descent or landing.
3. Use Maximum Thrust for climb. (See Maximum Thrust Climb Chart for detailed information.)
4. Cruise at Normal Thrust (100% RPM) and EGT of 545° C.
5. No distance credit is allowed for time necessary to accelerate to climb speed.

Figure A4-8 (Sheet 1 of 2)

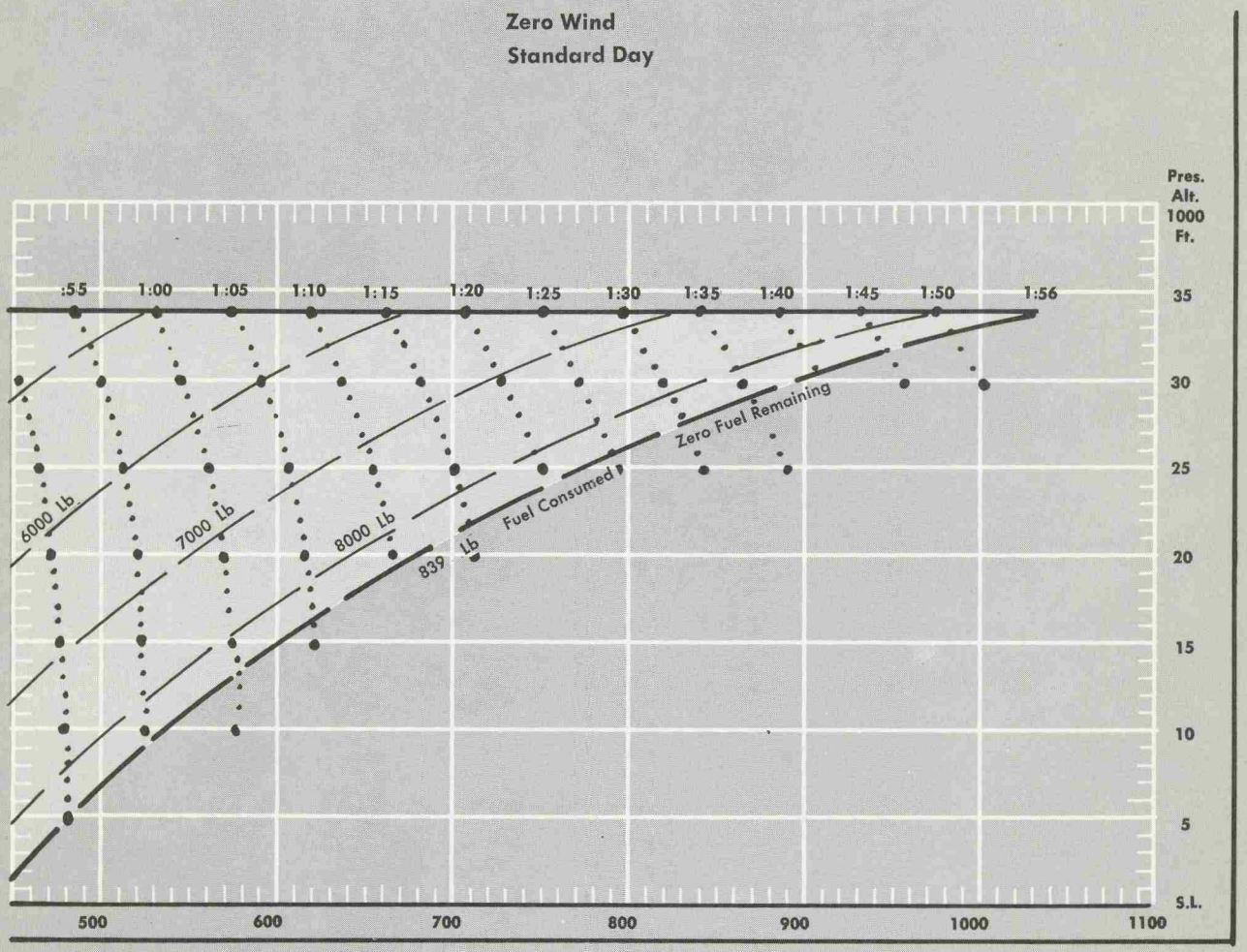
LEGEND:

Zero Fuel Remaining
Fuel Consumed — Lb
Time (from brake release)
Normal Thrust Cruise Ceiling

Configuration



Two GAR-8 Missiles + Two 200 Gal. Pylon Tanks
Gross Weight Range : 22,514 — 14,116 Lb



ALTITUDE — FT.		† MACH NO.		† IAS	† TAS			EGT
Pressure	Indicated	True	Indicated	Knots	Knots	% RPM	*LB/HR	° C
34,000	33,220	.93	.90	325	540	100	3640	545
30,000	29,000	.95	.92	360	555	100	4390	545
25,000	23,870	.95	.92	400	570	100	5250	545
20,000	18,830	.95	.92	440	585	100	6200	545
15,000	13,880	.95	.92	480	590	100	7200	545
10,000	8,950	.94	.91	520	600	100	8280	545
5,000	4,030	.94	.91	560	605	100	9420	545
S.L.	−930	.93	.90	600	610	100	10600	545

* Initial Approximate Values
† Average

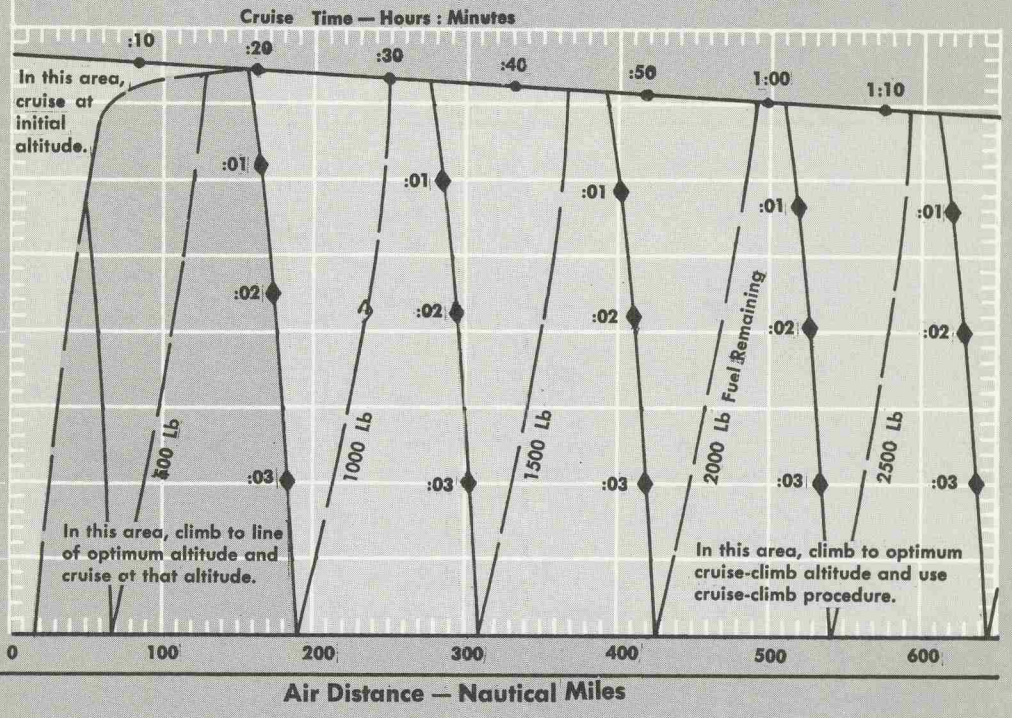
Figure A4-8 (Sheet 2 of 2)

OPTIMUM RETURN PROFILE

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

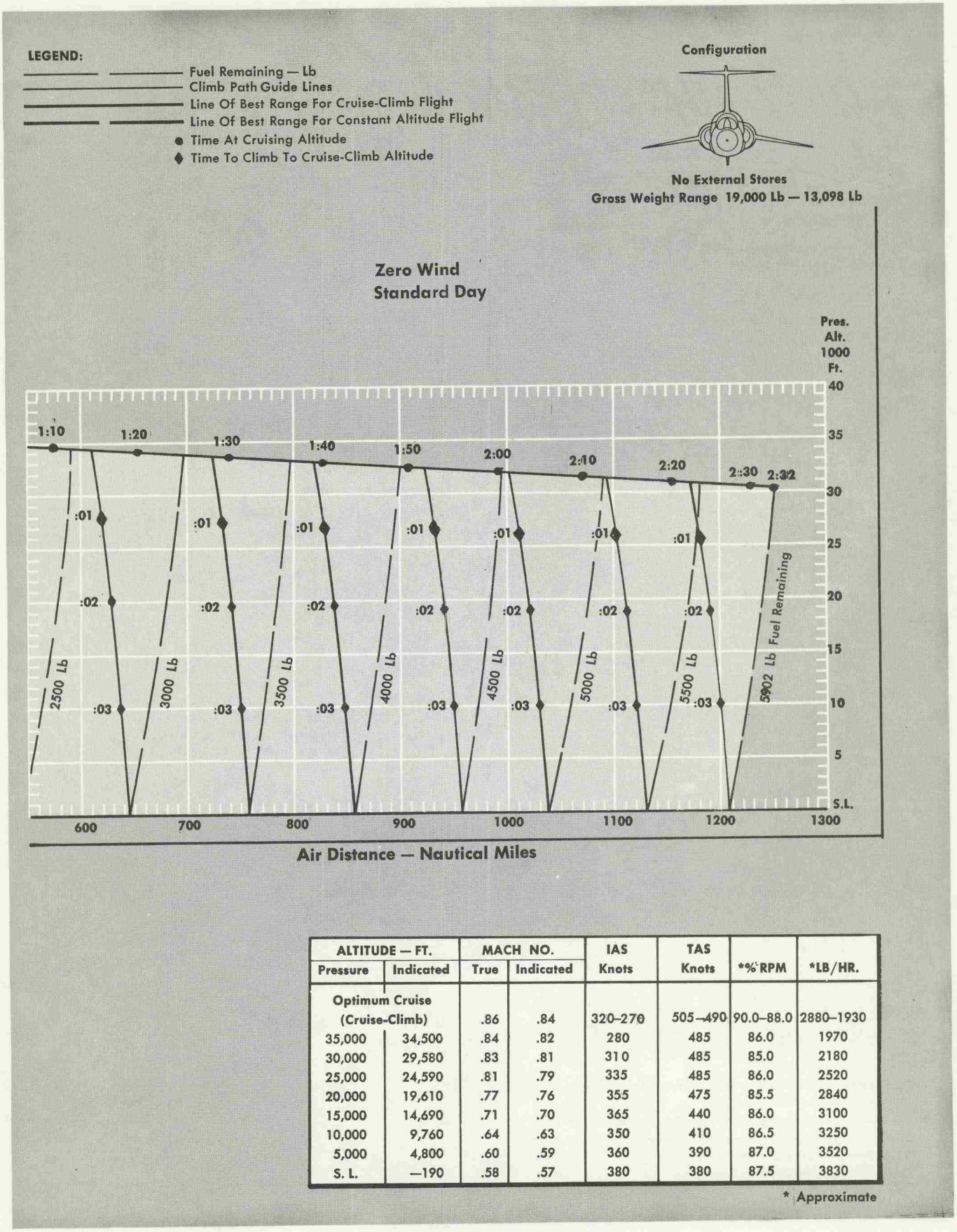
Climb Data			
Military Thrust Climb			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	573	S.L.



NOTE:

1. Fuel at any point includes Military Thrust climb, if required, to Cruise-Climb altitude.
2. No allowance or reserve made for loiter, descent or landing.
3. Best cruise condition is determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach Number.

Figure A4-9 (Sheet 1 of 2)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
Optimum Cruise (Cruise-Climb)		.86	.84	320–270	505–490	90.0–88.0	2880–1930
35,000	34,500	.84	.82	280	485	86.0	1970
30,000	29,580	.83	.81	310	485	85.0	2180
25,000	24,590	.81	.79	335	485	86.0	2520
20,000	19,610	.77	.76	355	475	85.5	2840
15,000	14,690	.71	.70	365	440	86.0	3100
10,000	9,760	.64	.63	350	410	86.5	3250
5,000	4,800	.60	.59	360	390	87.0	3520
S. L.	—190	.58	.57	380	380	87.5	3830

* Approximate

Figure A4-9 (Sheet 2 of 2)

OPTIMUM RETURN PROFILE

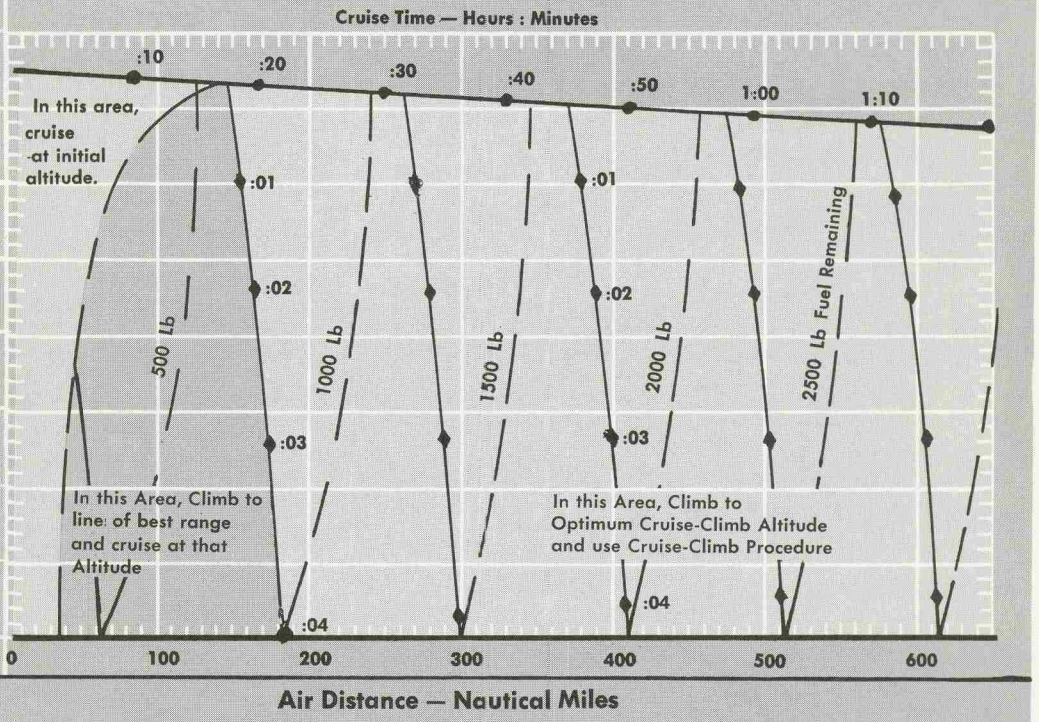
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Climb Data

Military Thrust Climb

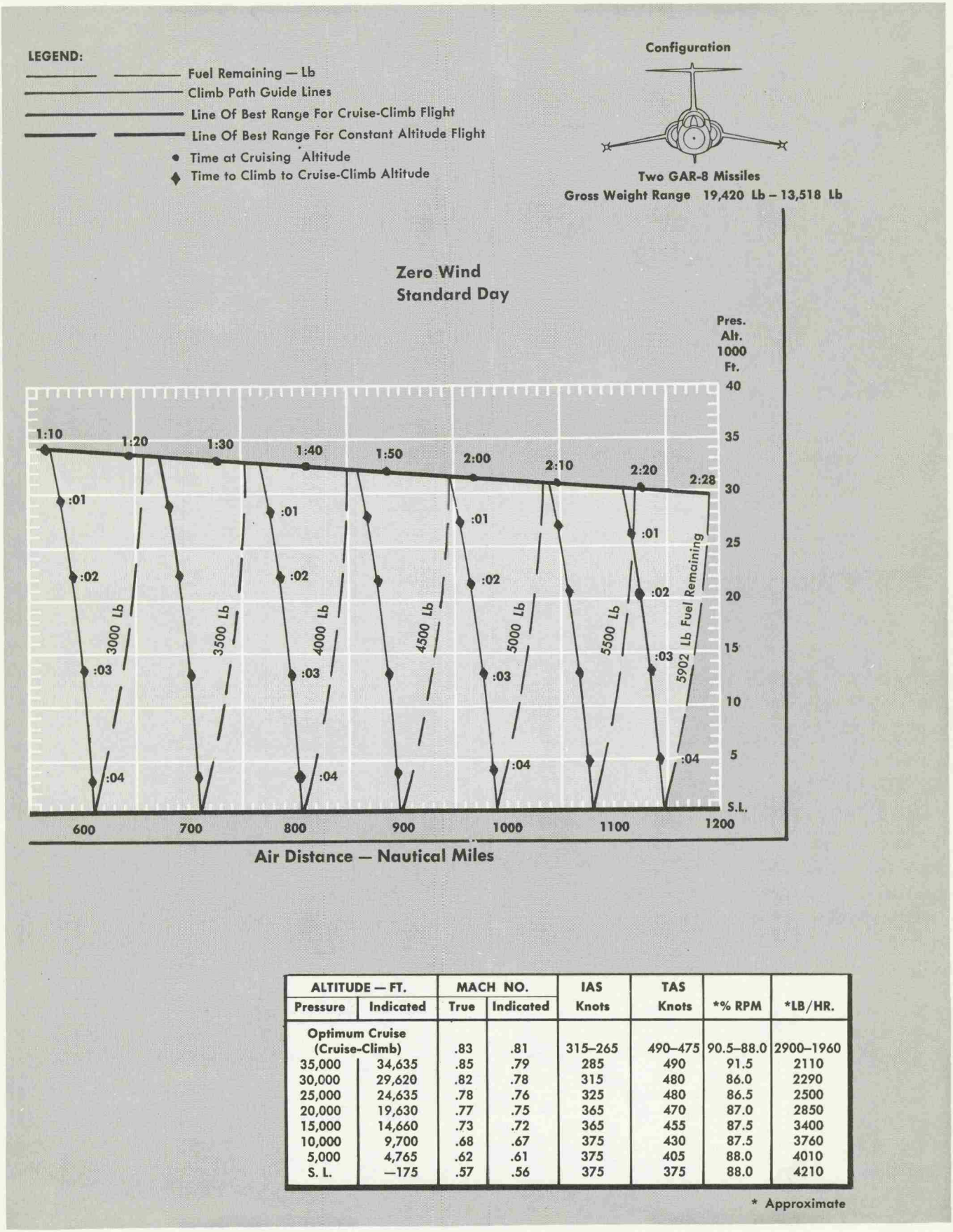
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	579	S.L.



NOTE:

1. Fuel at any point includes Military Thrust climb, if required, to Cruise-Climb altitude.
2. No allowance or reserve made for loiter, descent or landing.
3. Best cruise condition is determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach Number.

Figure A4-10 (Sheet 1 of 2)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
Optimum Cruise (Cruise-Climb)		.83	.81	315–265	490–475	90.5–88.0	2900–1960
35,000	34,635	.85	.79	285	490	91.5	2110
30,000	29,620	.82	.78	315	480	86.0	2290
25,000	24,635	.78	.76	325	480	86.5	2500
20,000	19,630	.77	.75	365	470	87.0	2850
15,000	14,660	.73	.72	365	455	87.5	3400
10,000	9,700	.68	.67	375	430	87.5	3760
5,000	4,765	.62	.61	375	405	88.0	4010
S. L.	—175	.57	.56	375	375	88.0	4210

* Approximate

Figure A4-10 (Sheet 2 of 2)

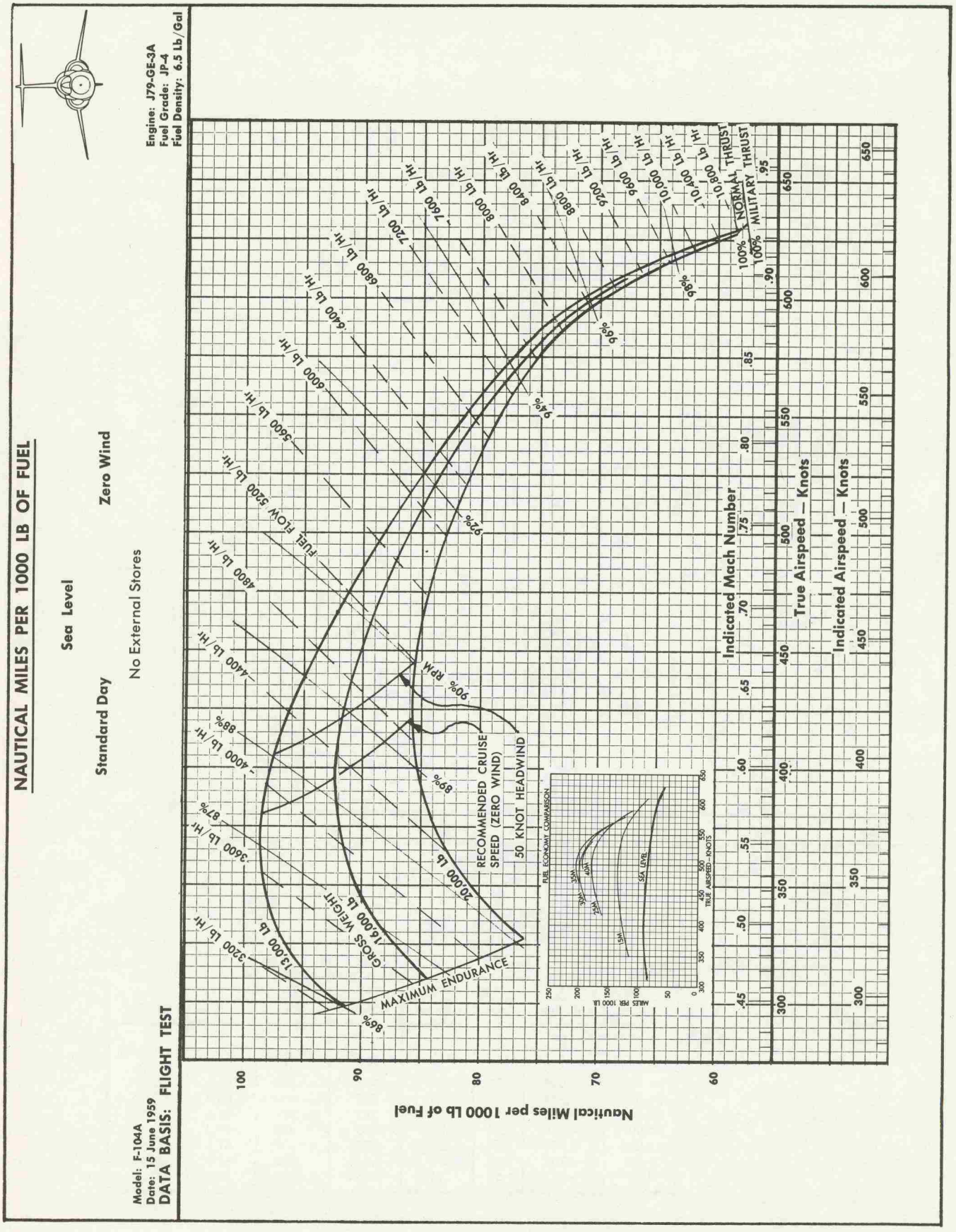

NAUTICAL MILES PER 1000 LB OF FUEL
Sea Level
Standard Day
Zero Wind
No External Stores
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 Lb of Fuel
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots
100% NORMAL THRUST
100% MILITARY THRUST
FUEL FLOW 5200 LB/HR
4800 LB/HR
4400 LB/HR
4000 LB/HR
3600 LB/HR
3200 LB/HR
5600 LB/HR
6000 LB/HR
6400 LB/HR
6800 LB/HR
7200 LB/HR
7600 LB/HR
8000 LB/HR
8400 LB/HR
8800 LB/HR
9200 LB/HR
9600 LB/HR
10,000 LB/HR
10,400 LB/HR
10,800 LB/HR
90% RPM
92%
94%
96%
98%
86%
87%
88%
89%
GROSS WEIGHT
20,000 LB
16,000 LB
13,000 LB
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
MAXIMUM ENDURANCE
FUEL ECONOMY COMPARISON
SEA LEVEL
35M
40M
30M
25M
15M
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS

Figure A4-11

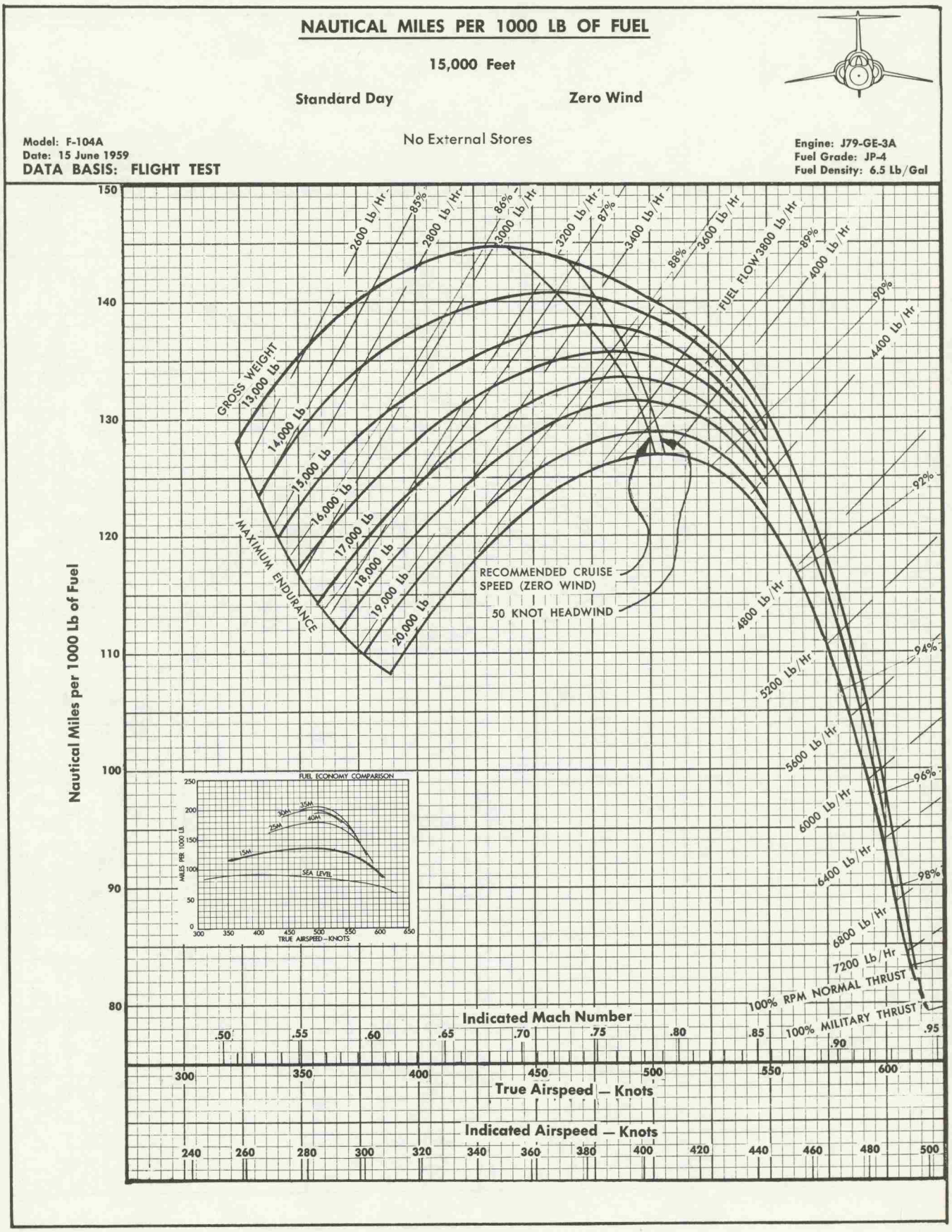

NAUTICAL MILES PER 1000 LB OF FUEL
15,000 Feet
Standard Day
Zero Wind
No External Stores
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
FUEL FLOW
2600 Lb/Hr
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
6800 Lb/Hr
7200 Lb/Hr
85%
86%
87%
88%
89%
90%
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
SEA LEVEL
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-12

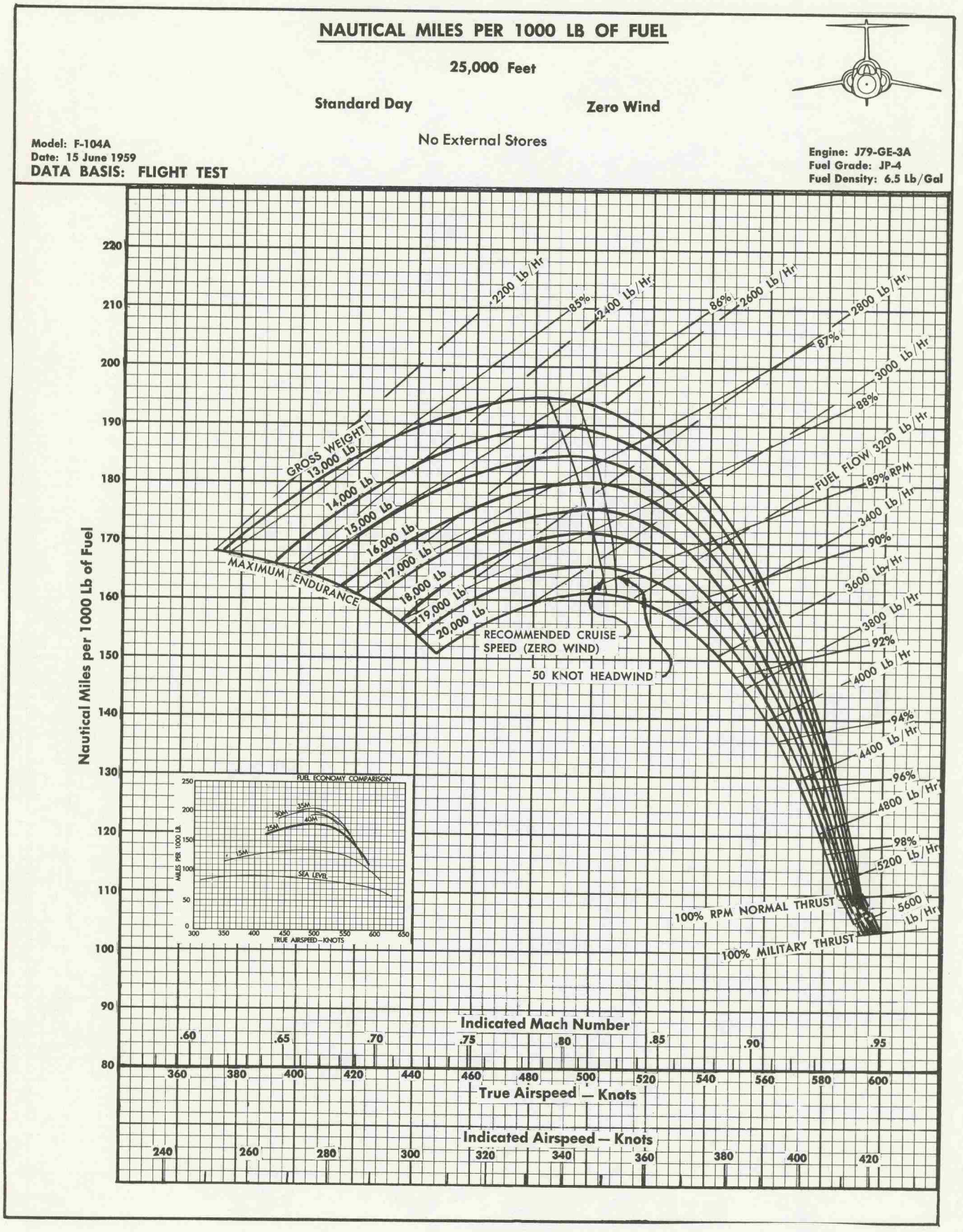

NAUTICAL MILES PER 1000 LB OF FUEL
25,000 Feet
Standard Day
Zero Wind
No External Stores
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
2200 Lb/Hr
2400 Lb/Hr
2600 Lb/Hr
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
85%
86%
87%
88%
89% RPM
90%
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL FLOW
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
SEA LEVEL
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-13

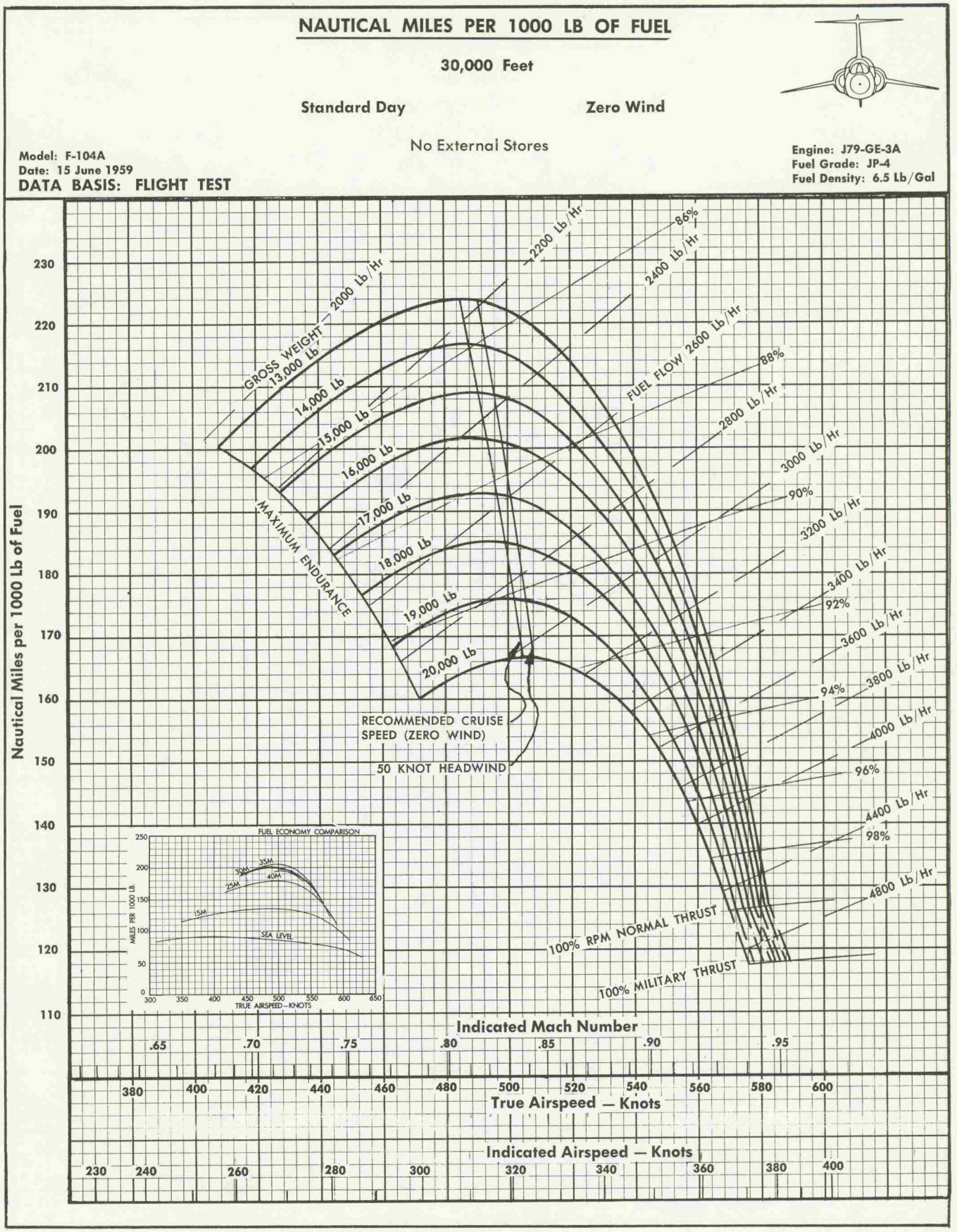

NAUTICAL MILES PER 1000 LB OF FUEL
30,000 Feet
Standard Day
Zero Wind
No External Stores
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
2000 Lb/Hr
2200 Lb/Hr
2400 Lb/Hr
2600 Lb/Hr
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
FUEL FLOW
86%
88%
90%
92%
94%
96%
98%
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB.
TRUE AIRSPEED—KNOTS
SEA LEVEL
15M
25M
30M
35M
40M
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-14

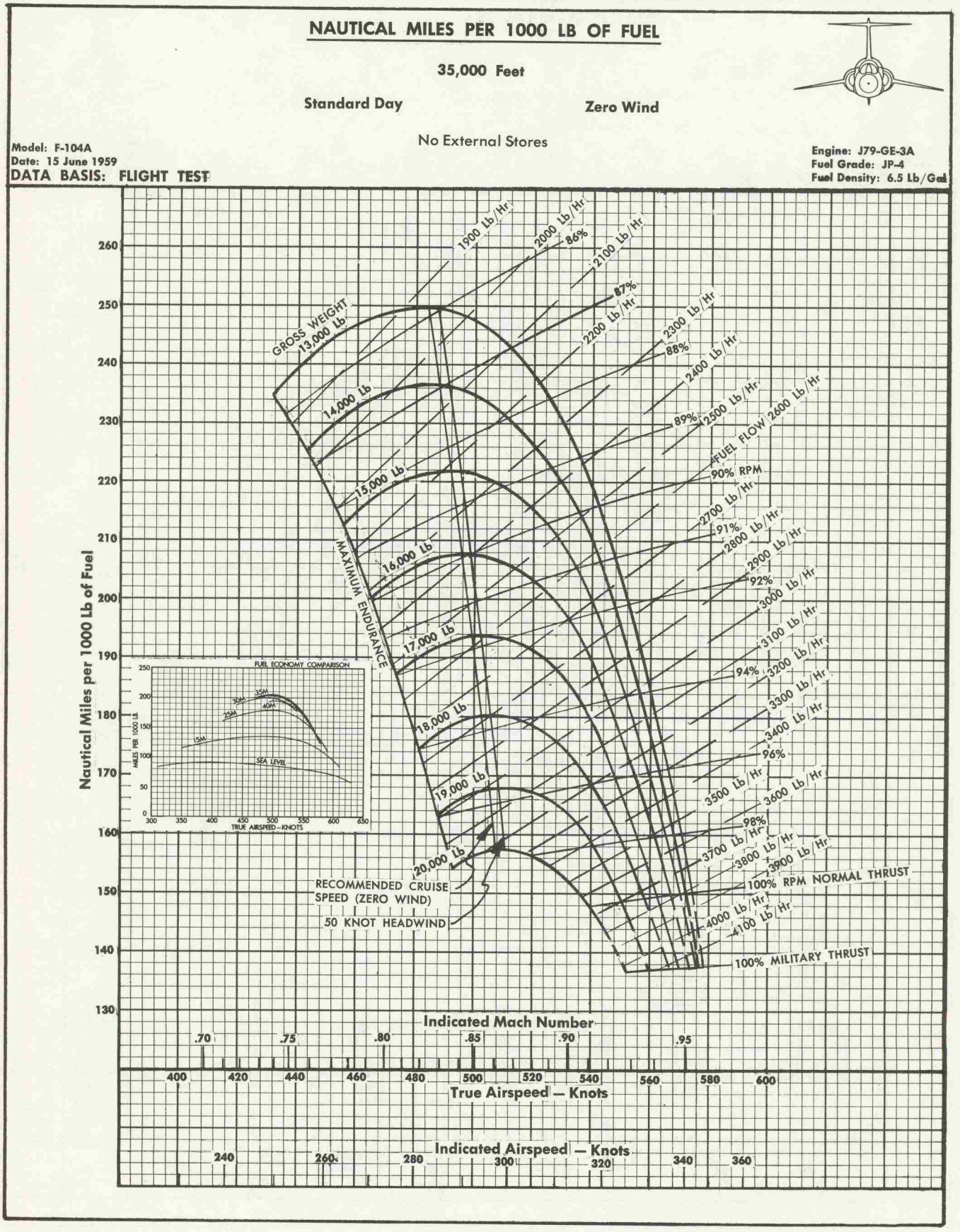

NAUTICAL MILES PER 1000 LB OF FUEL
35,000 Feet
Standard Day
Zero Wind
No External Stores
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT 13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
1900 Lb/Hr
2000 Lb/Hr
2100 Lb/Hr
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
FUEL FLOW 2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
3500 Lb/Hr
3600 Lb/Hr
3700 Lb/Hr
3800 Lb/Hr
3900 Lb/Hr
4000 Lb/Hr
4100 Lb/Hr
86%
87%
88%
89%
90% RPM
91%
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
SEA LEVEL
15M
25M
30M
35M
40M
Indicated Mach Number
.70
.75
.80
.85
.90
.95
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-15

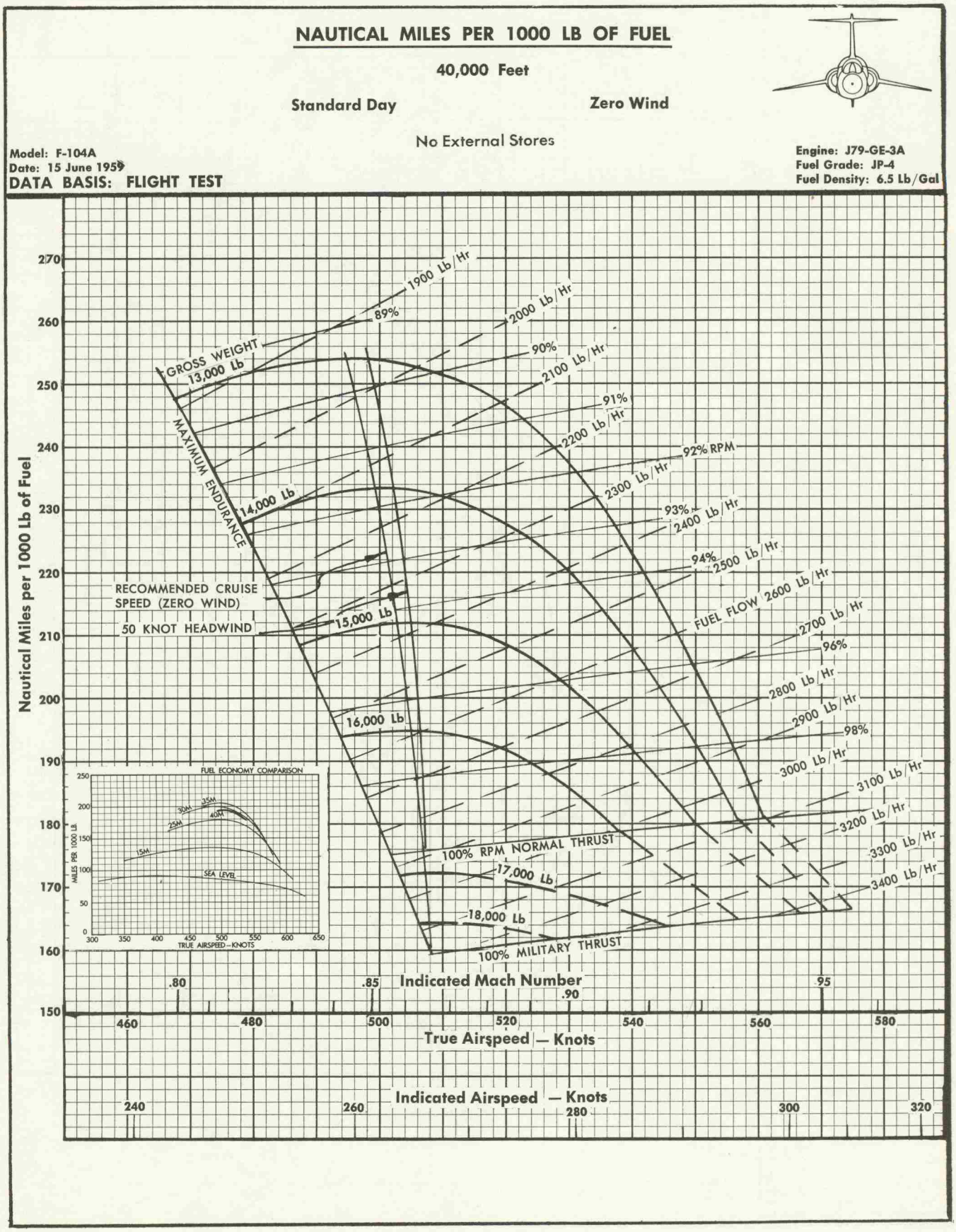

NAUTICAL MILES PER 1000 LB OF FUEL
40,000 Feet
Standard Day
Zero Wind
No External Stores
Model: F-104A
Date: 15 June 195
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
1900 Lb/Hr
2000 Lb/Hr
2100 Lb/Hr
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
FUEL FLOW 2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
89%
90%
91%
92% RPM
93%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
SEA LEVEL
15M
25M
30M
35M
40M
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-16

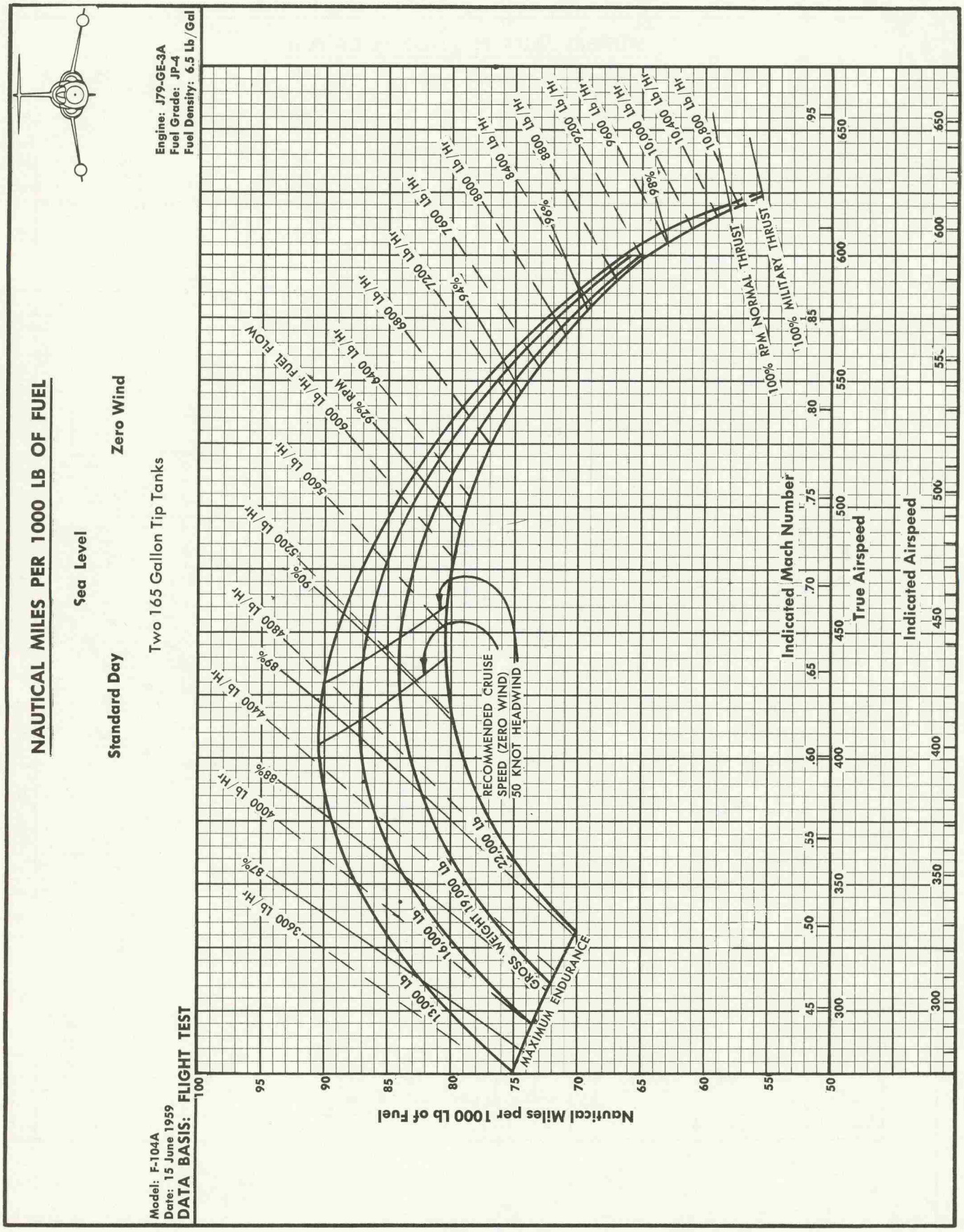

NAUTICAL MILES PER 1000 LB OF FUEL
Sea Level
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 Lb of Fuel
Indicated Mach Number
True Airspeed
Indicated Airspeed
100% RPM NORMAL THRUST
100% MILITARY THRUST
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
MAXIMUM ENDURANCE
GROSS WEIGHT 19,000 Lb
22,000 Lb
16,000 Lb
13,000 Lb
FUEL FLOW
10,800 Lb/Hr
10,400 Lb/Hr
10,000 Lb/Hr
9600 Lb/Hr
9200 Lb/Hr
8800 Lb/Hr
8400 Lb/Hr
8000 Lb/Hr
7600 Lb/Hr
7200 Lb/Hr
6800 Lb/Hr
6400 Lb/Hr
6000 Lb/Hr
5600 Lb/Hr
5200 Lb/Hr
4800 Lb/Hr
4400 Lb/Hr
4000 Lb/Hr
3600 Lb/Hr
98%
96%
94%
92% RPM
90%
89%
88%
87%

Figure A4-17

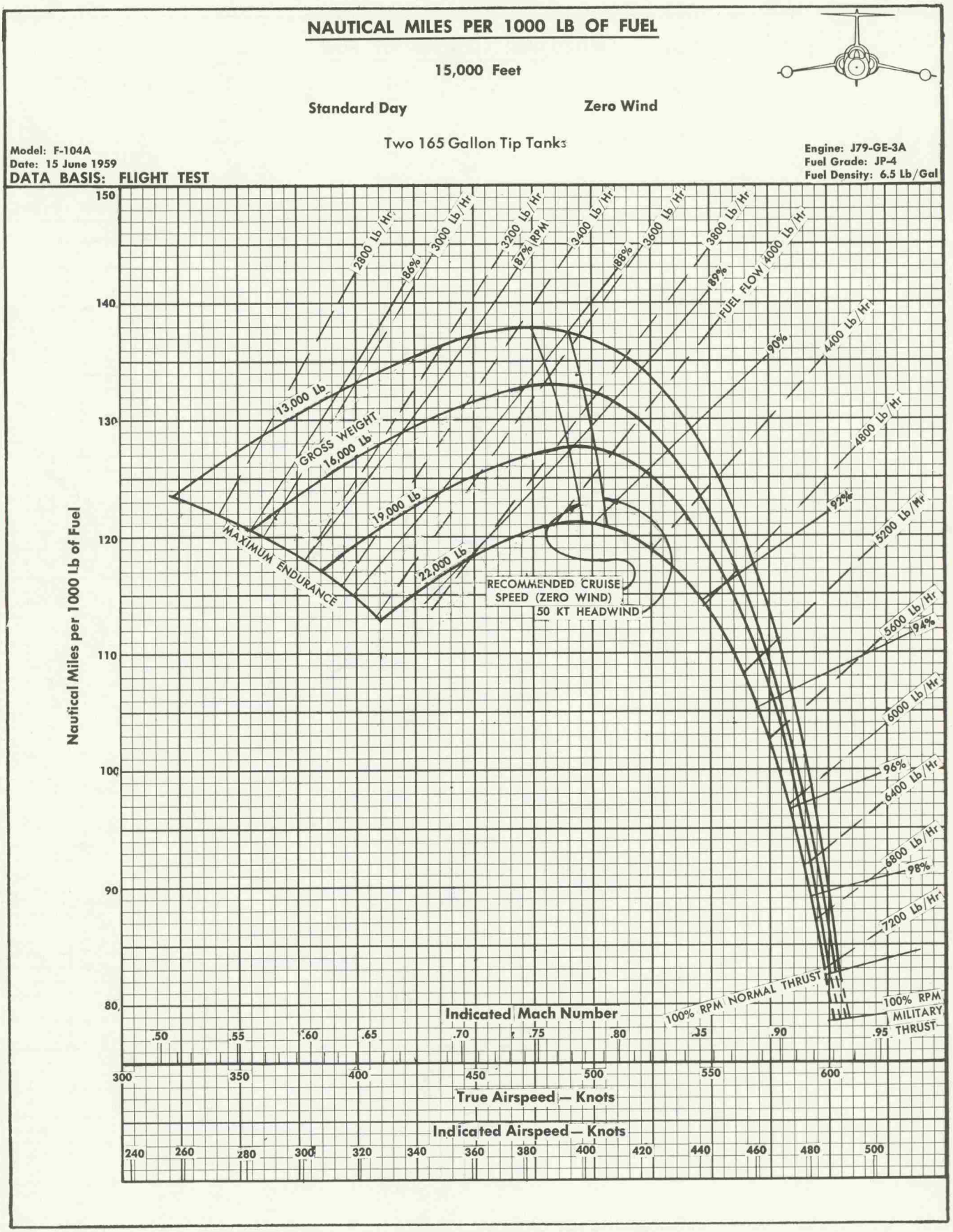
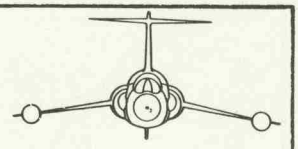

NAUTICAL MILES PER 1000 LB OF FUEL
15,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
150
140
130
120
110
100
90
80
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
6800 Lb/Hr
7200 Lb/Hr
86%
87% RPM
88%
89%
90%
92%
94%
96%
98%
FUEL FLOW
13,000 Lb
GROSS WEIGHT
16,000 Lb
19,000 Lb
22,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KT HEADWIND
100% RPM NORMAL THRUST
100% RPM MILITARY THRUST
Indicated Mach Number
.50 .55 .60 .65 .70 .75 .80 .85 .90 .95
300 350 400 450 500 550 600
True Airspeed — Knots
Indicated Airspeed — Knots
240 260 280 300 320 340 360 380 400 420 440 460 480 500

Figure A4-18

NAUTICAL MILES PER 1000 LB OF FUEL



25,000 Feet

Standard Day

Zero Wind

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Two 165 Gallon Tip Tanks

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

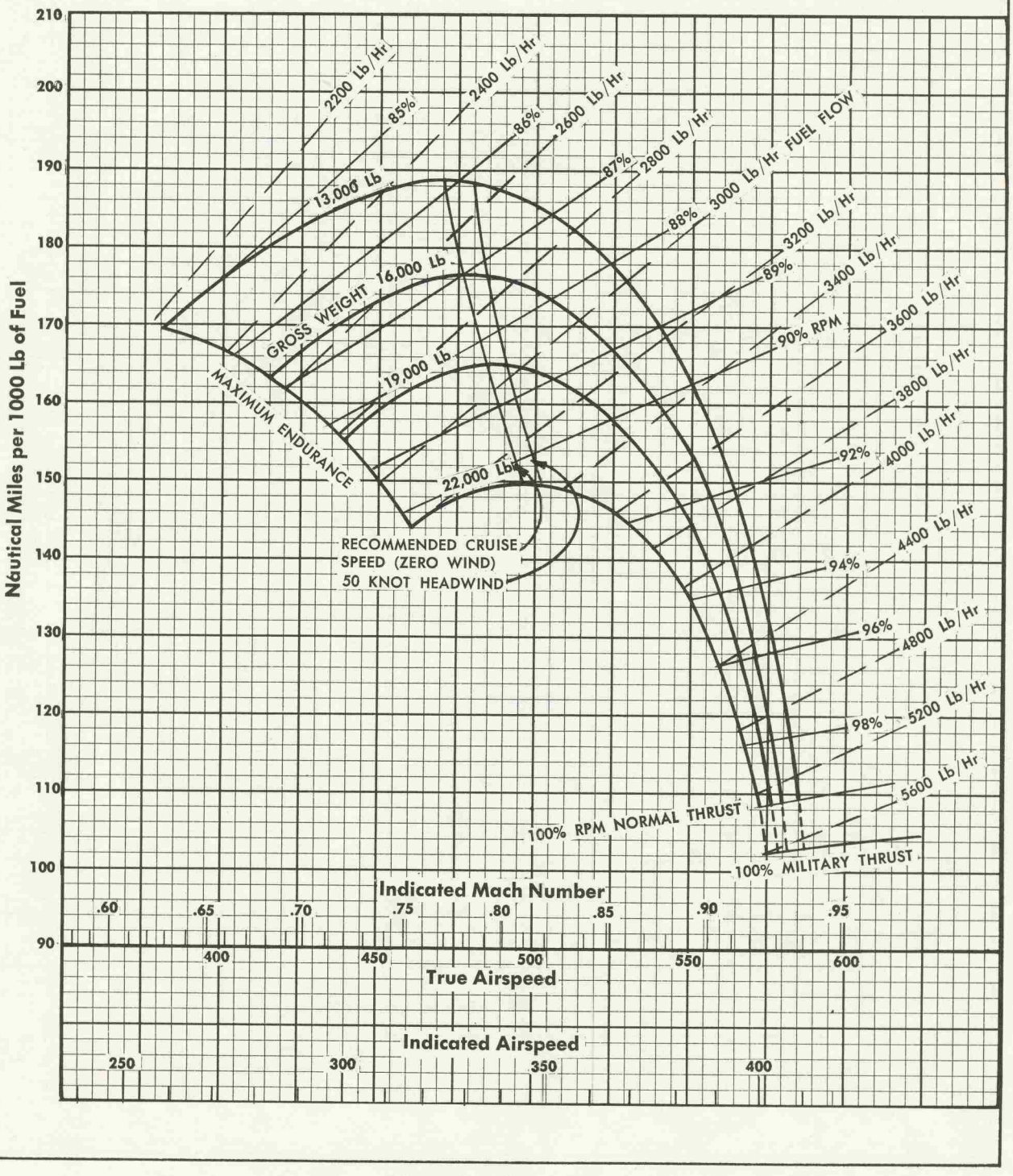


Figure A4-19

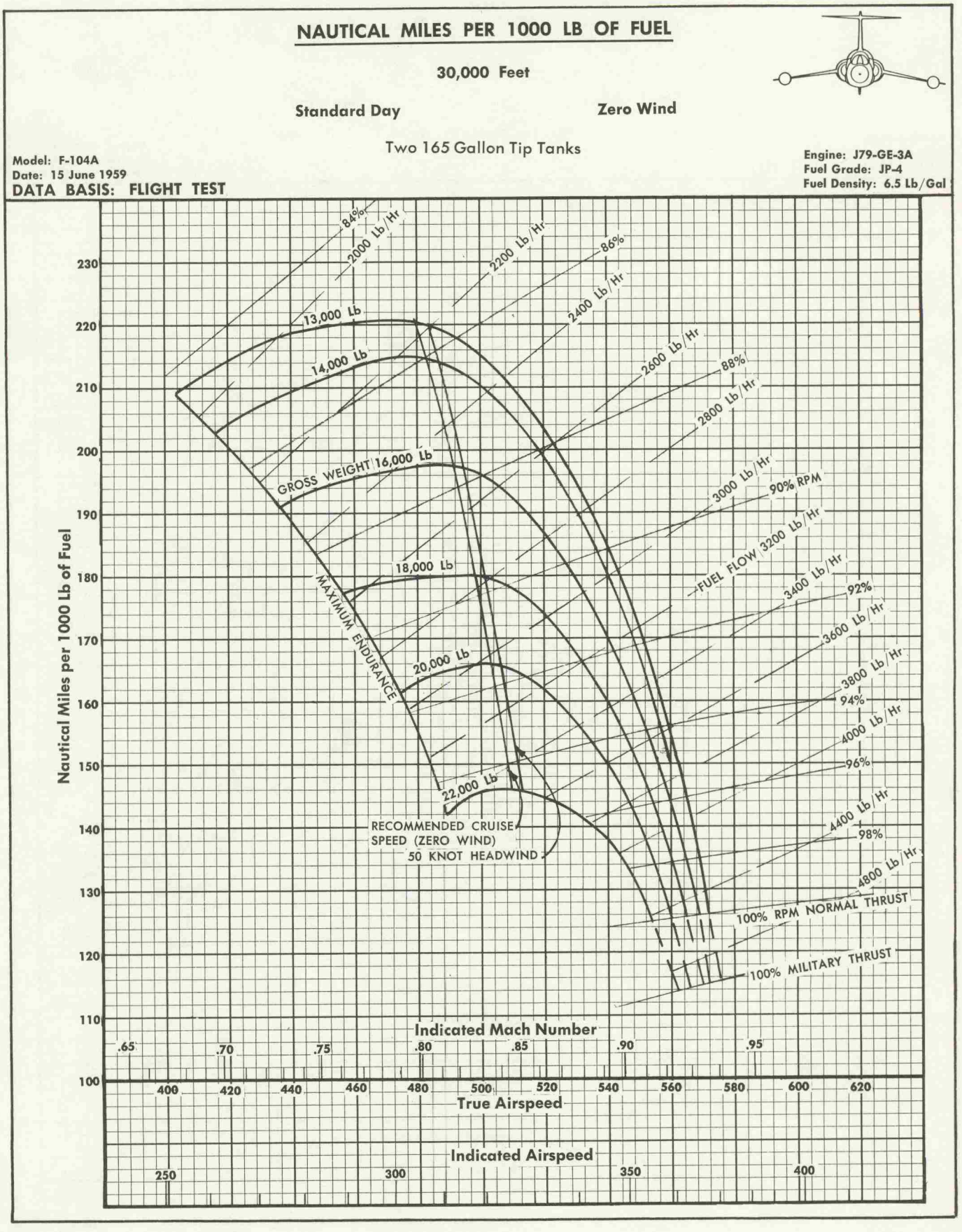

NAUTICAL MILES PER 1000 LB OF FUEL
30,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
230
220
210
200
190
180
170
160
150
140
130
120
110
100
84%
2000 Lb/Hr
2200 Lb/Hr
86%
2400 Lb/Hr
13,000 Lb
14,000 Lb
2600 Lb/Hr
88%
2800 Lb/Hr
GROSS WEIGHT 16,000 Lb
3000 Lb/Hr
90% RPM
FUEL FLOW 3200 Lb/Hr
18,000 Lb
MAXIMUM ENDURANCE
3400 Lb/Hr
92%
3600 Lb/Hr
20,000 Lb
3800 Lb/Hr
94%
4000 Lb/Hr
96%
22,000 Lb
4400 Lb/Hr
RECOMMENDED CRUISE
SPEED (ZERO WIND)
50 KNOT HEADWIND
98%
4800 Lb/Hr
100% RPM NORMAL THRUST
100% MILITARY THRUST
Indicated Mach Number
.65
.70
.75
.80
.85
.90
.95
400
420
440
460
480
500
520
540
560
580
600
620
True Airspeed
Indicated Airspeed
250
300
350
400

Figure A4-20

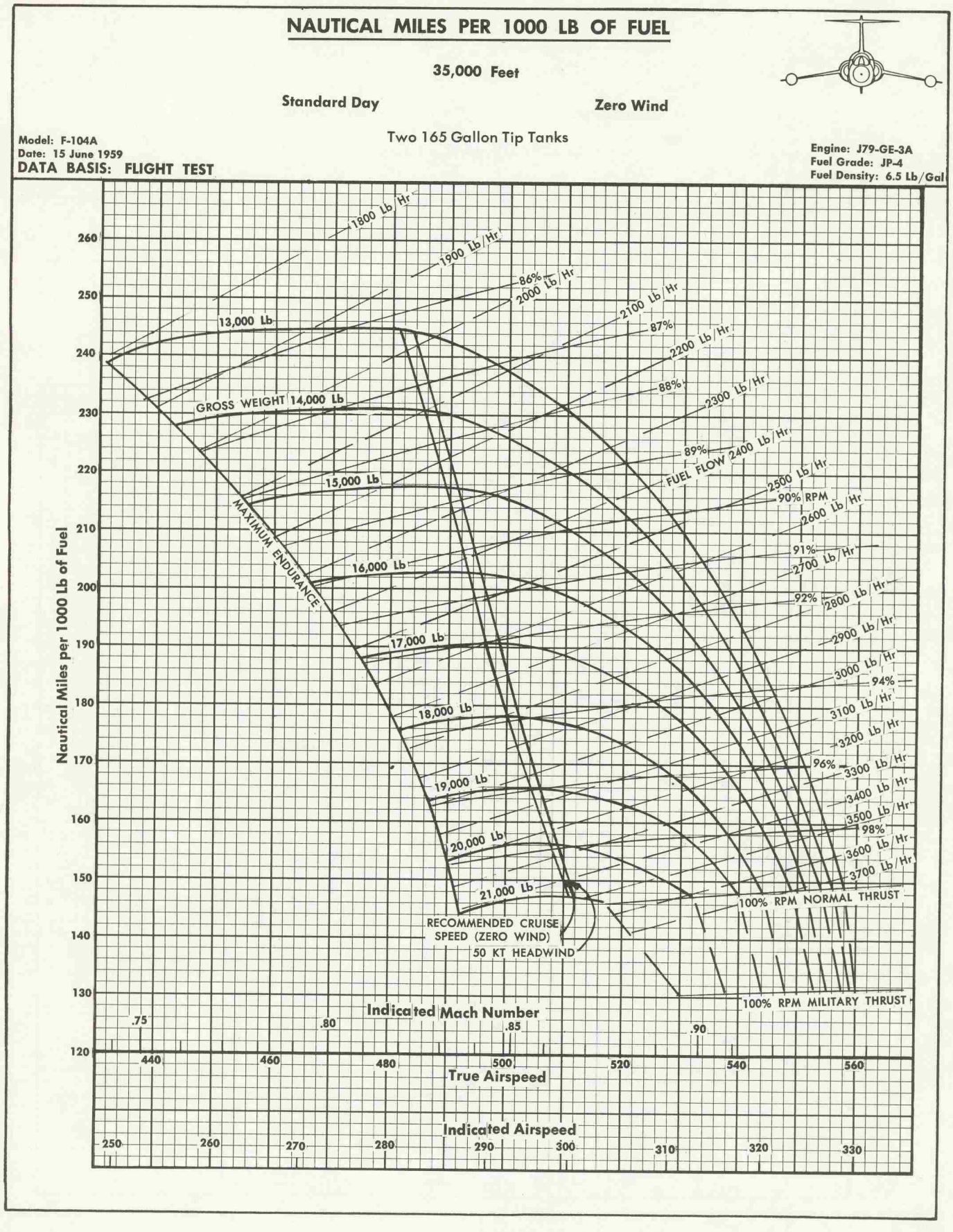
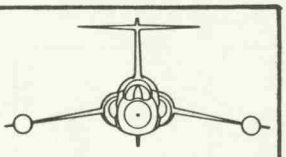

NAUTICAL MILES PER 1000 LB OF FUEL
35,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
Indicated Mach Number
True Airspeed
Indicated Airspeed
GROSS WEIGHT 14,000 Lb
13,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
21,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KT HEADWIND
FUEL FLOW
1800 Lb/Hr
1900 Lb/Hr
2000 Lb/Hr
2100 Lb/Hr
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
3500 Lb/Hr
3600 Lb/Hr
3700 Lb/Hr
86%
87%
88%
89%
90% RPM
91%
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% RPM MILITARY THRUST

Figure A4-21

NAUTICAL MILES PER 1000 LB OF FUEL



40,000 Feet

Standard Day — Zero Wind

Two 165 Gallon Tip Tanks

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

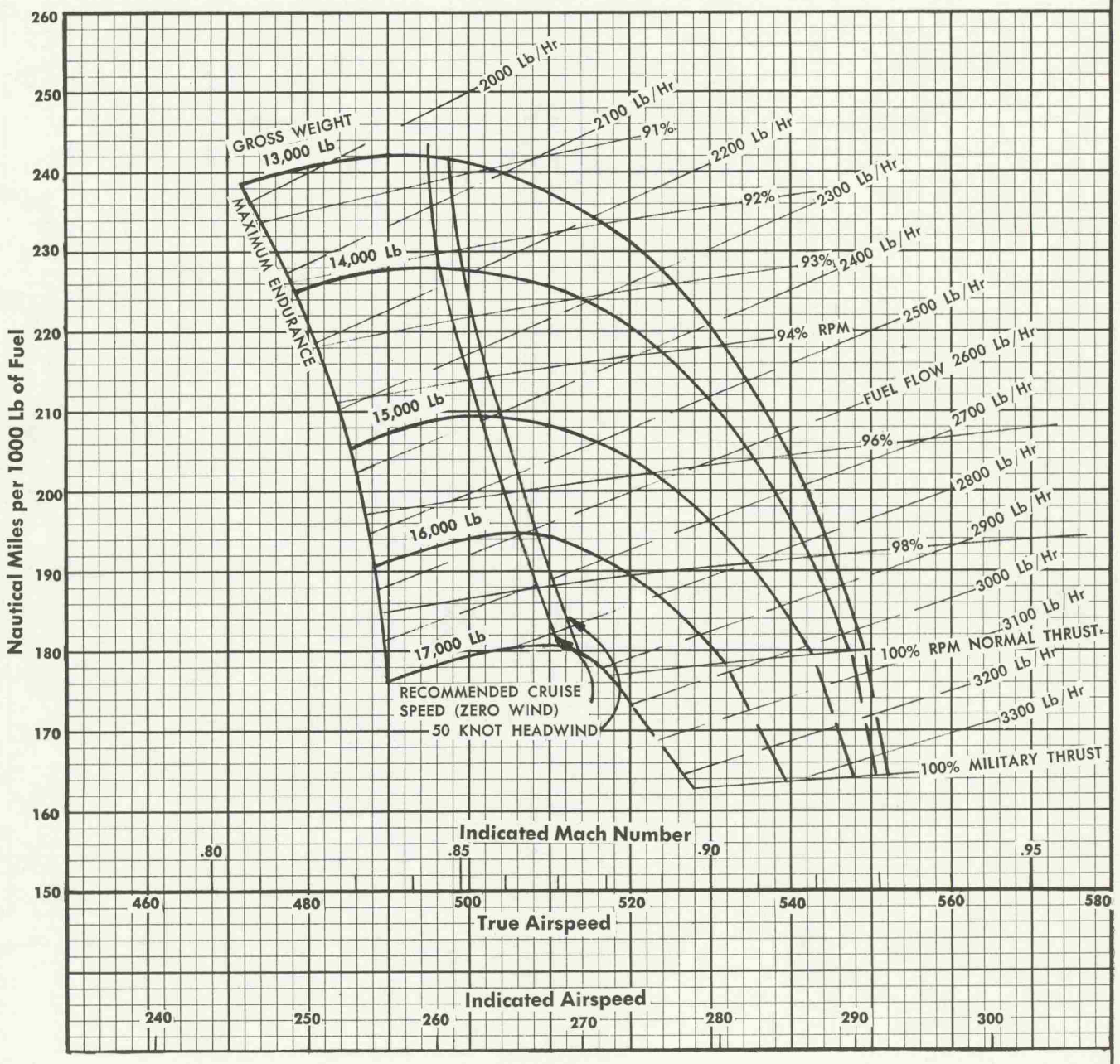


Figure A4-22

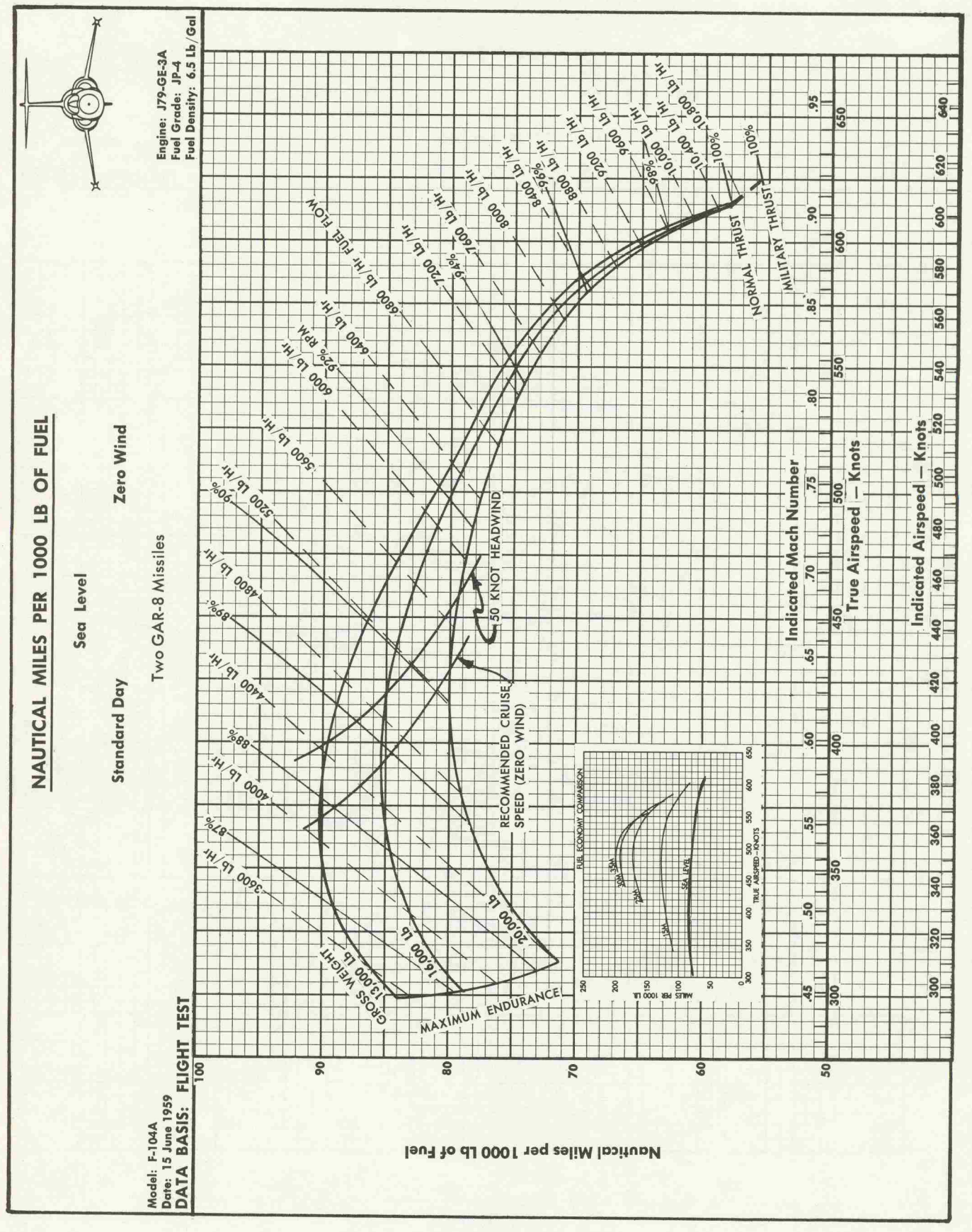

NAUTICAL MILES PER 1000 LB OF FUEL
Sea Level
Standard Day
Zero Wind
Two GAR-8 Missiles
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 lb of Fuel
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots
NORMAL THRUST
MILITARY THRUST
100%
98%
96%
94%
92%
90%
89%
88%
87%
10,800 LB/HR
10,400 LB/HR
10,000 LB/HR
9600 LB/HR
9200 LB/HR
8800 LB/HR
8400 LB/HR
8000 LB/HR
7600 LB/HR
7200 LB/HR
6800 LB/HR
6400 LB/HR
6000 LB/HR
5600 LB/HR
5200 LB/HR
4800 LB/HR
4400 LB/HR
4000 LB/HR
3600 LB/HR
RPM
FUEL FLOW
50 KNOT HEADWIND
RECOMMENDED CRUISE SPEED (ZERO WIND)
MAXIMUM ENDURANCE
GROSS WEIGHT
13,000 LB
16,000 LB
20,000 LB
FUEL ECONOMY COMPARISON
SEA LEVEL
35M
30M
25M
20M
15M
MILES PER 1000 LB
TRUE AIRSPEED - KNOTS

Figure A4-23

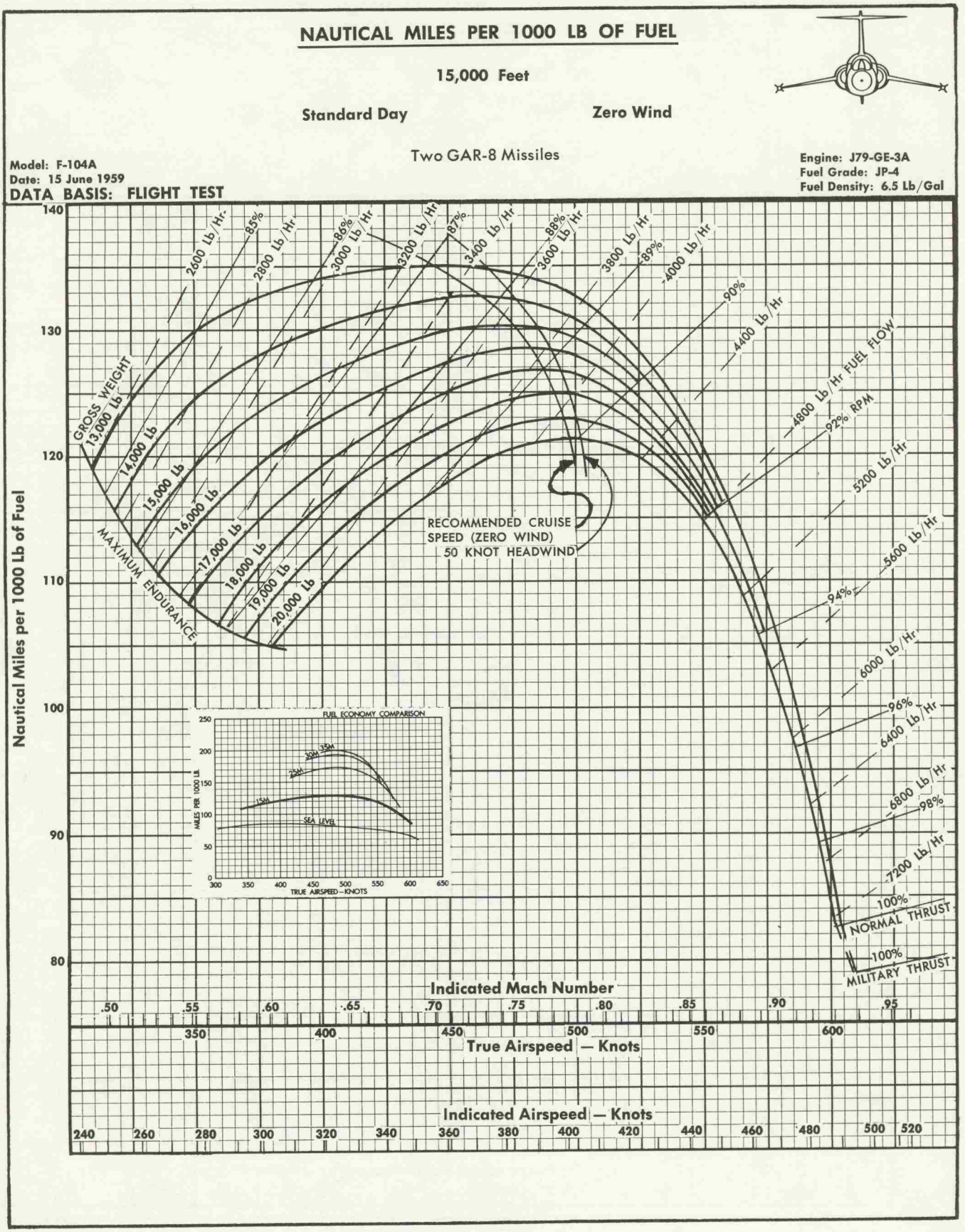

NAUTICAL MILES PER 1000 LB OF FUEL
15,000 Feet
Standard Day
Zero Wind
Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND) 50 KNOT HEADWIND
2600 Lb/Hr
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr FUEL FLOW
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
6800 Lb/Hr
7200 Lb/Hr
85%
86%
87%
88%
89%
90%
92% RPM
94%
96%
98%
100%
NORMAL THRUST
100%
MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB.
TRUE AIRSPEED—KNOTS
35M
30M
25M
15M
SEA LEVEL
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-24

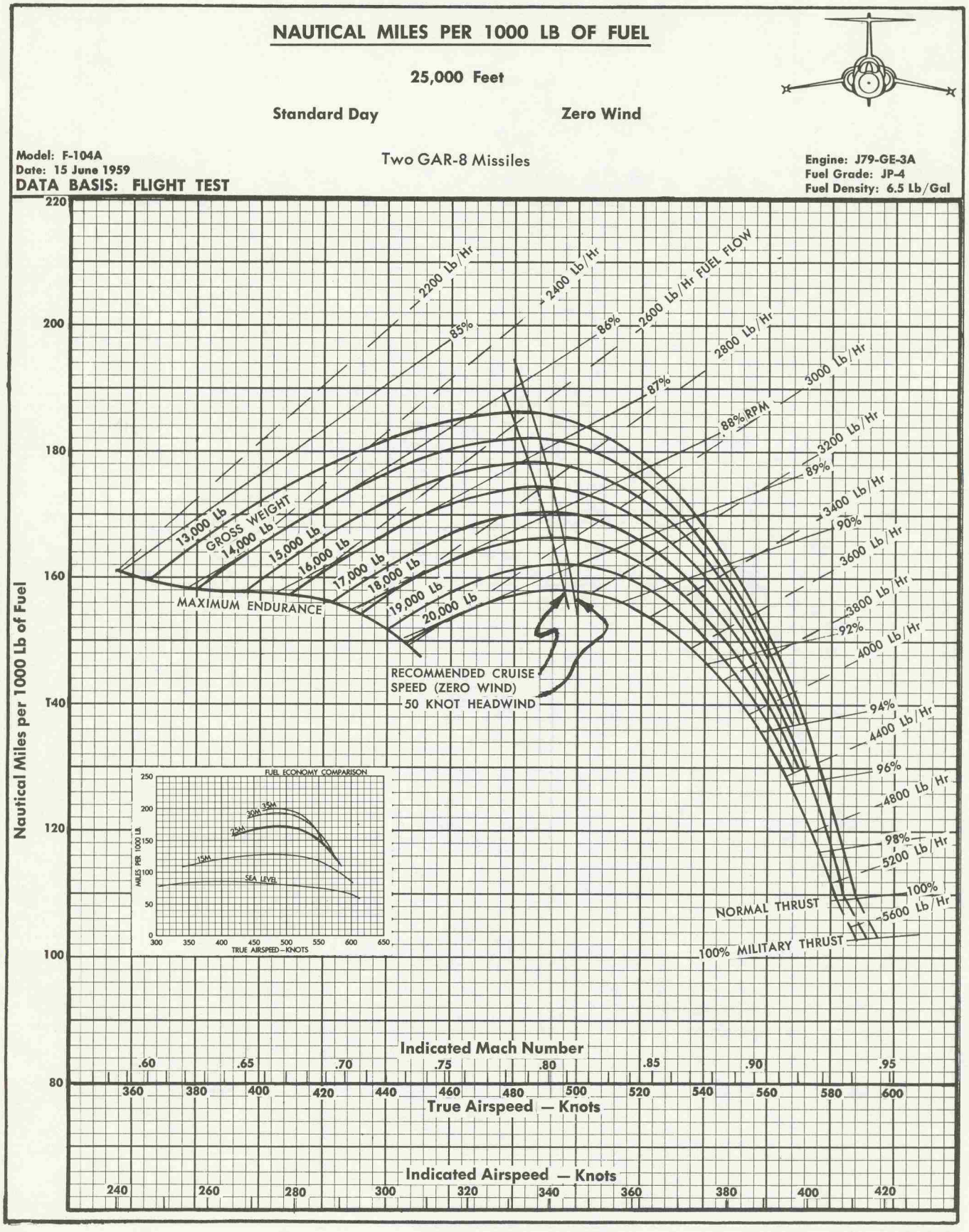

NAUTICAL MILES PER 1000 LB OF FUEL
25,000 Feet
Standard Day
Zero Wind
Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
220
200
180
160
140
120
100
80
2200 Lb/Hr
2400 Lb/Hr
2600 Lb/Hr FUEL FLOW
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
85%
86%
87%
88% RPM
89%
90%
92%
94%
96%
98%
100%
13,000 Lb
GROSS WEIGHT
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
NORMAL THRUST
100% MILITARY THRUST
FUEL ECONOMY COMPARISON
MILES PER 1000 LB.
TRUE AIRSPEED—KNOTS
30M
35M
25M
15M
SEA LEVEL
Indicated Mach Number
.60
.65
.70
.75
.80
.85
.90
.95
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-25

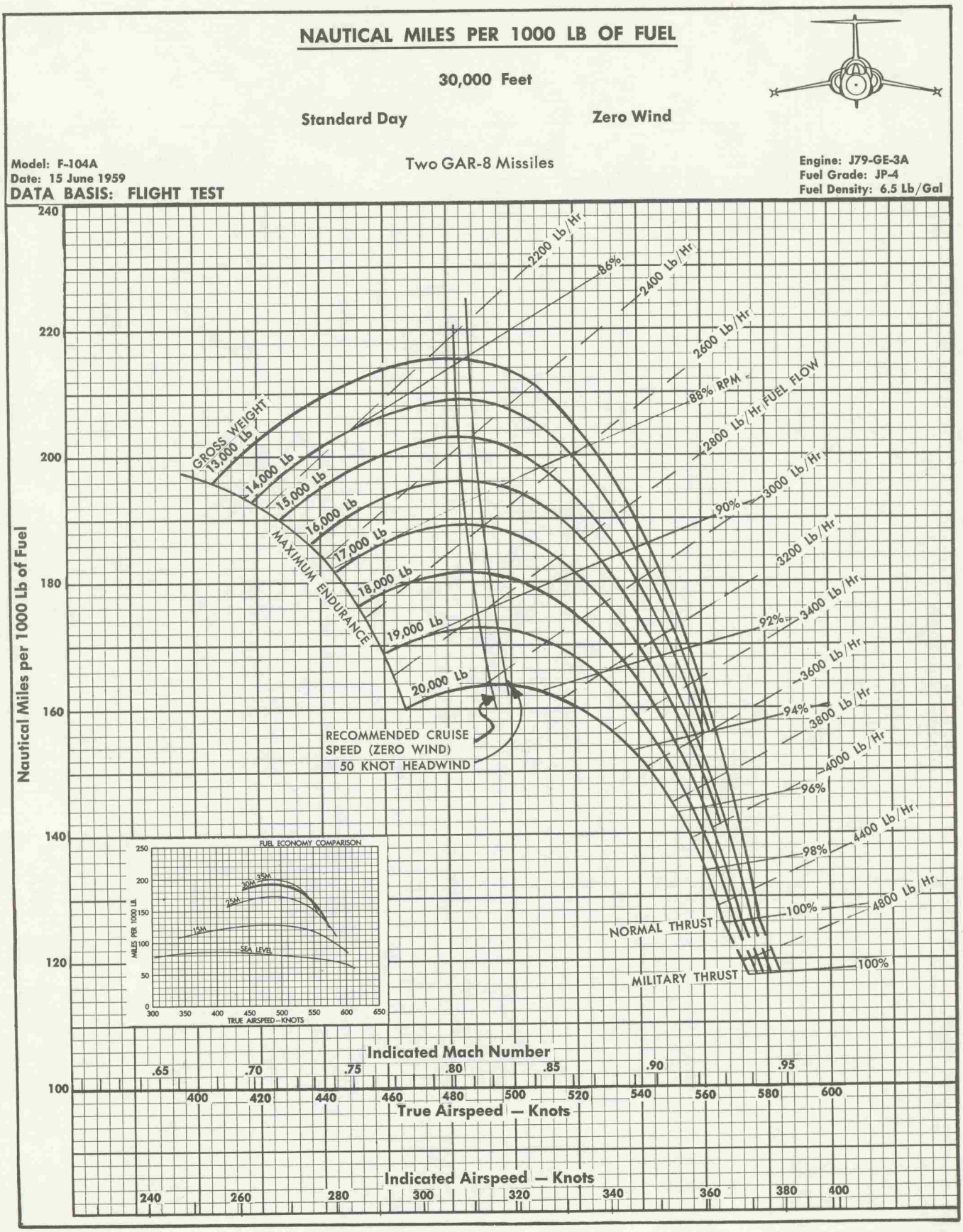

NAUTICAL MILES PER 1000 LB OF FUEL
30,000 Feet
Standard Day
Zero Wind
Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
20,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
2200 Lb/Hr
86%
2400 Lb/Hr
2600 Lb/Hr
88% RPM
2800 Lb/Hr FUEL FLOW
90%
3000 Lb/Hr
3200 Lb/Hr
92%
3400 Lb/Hr
3600 Lb/Hr
94%
3800 Lb/Hr
4000 Lb/Hr
96%
4400 Lb/Hr
98%
4800 Lb/Hr
100%
NORMAL THRUST
MILITARY THRUST
100%
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
35M
30M
25M
15M
SEA LEVEL
TRUE AIRSPEED—KNOTS
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-26

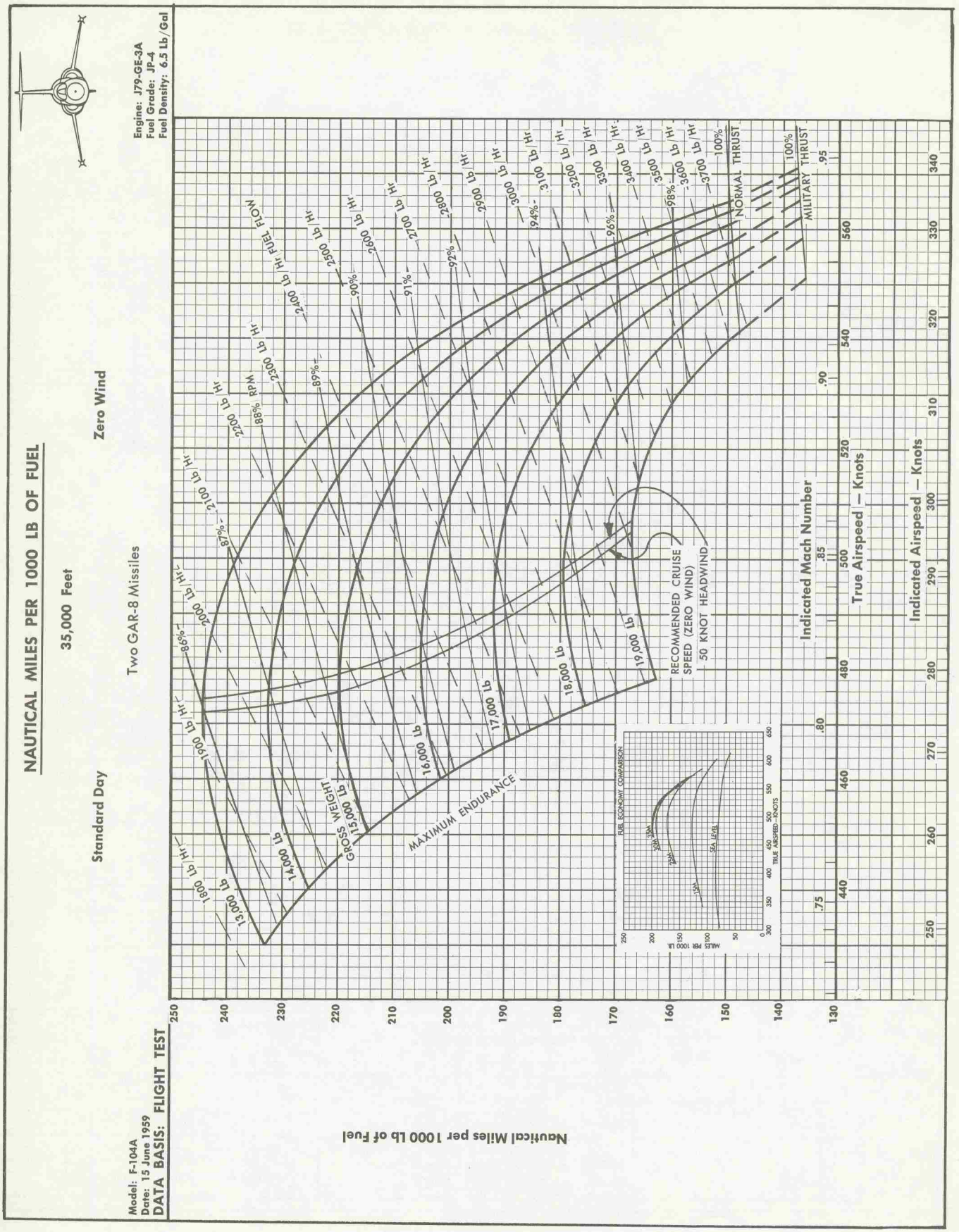

NAUTICAL MILES PER 1000 LB OF FUEL
35,000 Feet
Standard Day
Zero Wind
Two GAR-8 Missiles
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 lb of Fuel
Indicated Airspeed — Knots
True Airspeed — Knots
Indicated Mach Number
MAXIMUM ENDURANCE
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
NORMAL THRUST
MILITARY THRUST
100%
98%
96%
94%
92%
91%
90%
89%
88% RPM
87%
86%
FUEL FLOW
1800 Lb/Hr
1900 Lb/Hr
2000 Lb/Hr
2100 Lb/Hr
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
3500 Lb/Hr
3600 Lb/Hr
3700 Lb/Hr
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
SEA LEVEL

Figure A4-27

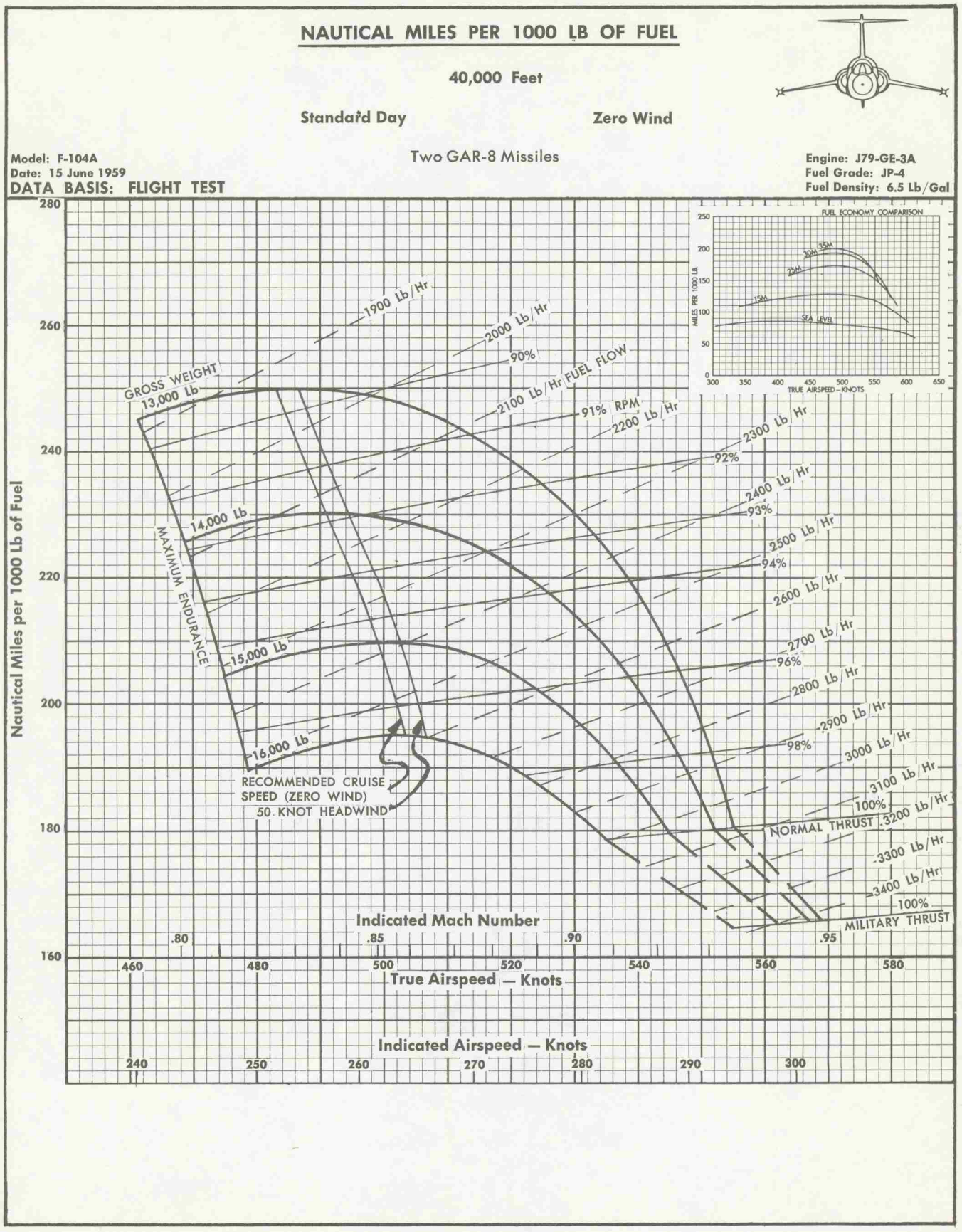

NAUTICAL MILES PER 1000 LB OF FUEL
40,000 Feet
Standard Day
Zero Wind
Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
FUEL ECONOMY COMPARISON
MILES PER 1000 LB
TRUE AIRSPEED—KNOTS
30M
35M
25M
20M
15M
SEA LEVEL
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
13,000 Lb
14,000 Lb
15,000 Lb
16,000 Lb
MAXIMUM ENDURANCE
1900 Lb/Hr
2000 Lb/Hr
2100 Lb/Hr FUEL FLOW
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
90%
91% RPM
92%
93%
94%
96%
98%
100%
NORMAL THRUST
100%
MILITARY THRUST
RECOMMENDED CRUISE
SPEED (ZERO WIND)
50 KNOT HEADWIND
Indicated Mach Number
.80
.85
.90
.95
True Airspeed — Knots
Indicated Airspeed — Knots
280
260
240
220
200
180
160
460
480
500
520
540
560
580
240
250
260
270
280
290
300

Figure A4-28

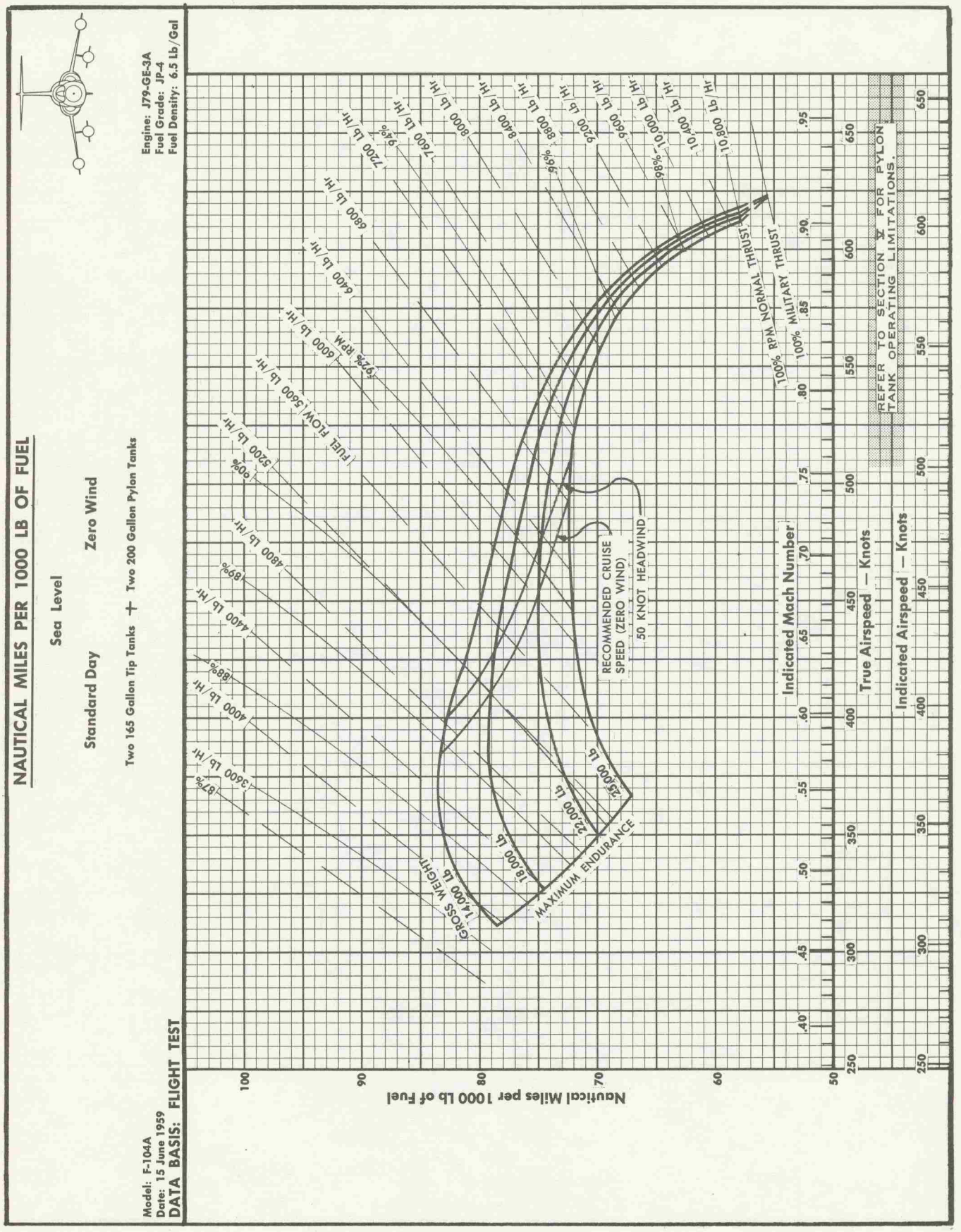

NAUTICAL MILES PER 1000 LB OF FUEL
Sea Level
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 Lb of Fuel
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots
100% RPM NORMAL THRUST
100% MILITARY THRUST
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
MAXIMUM ENDURANCE
GROSS WEIGHT
14,000 LB
18,000 LB
22,000 LB
25,000 LB
FUEL FLOW 5600 LB/HR
6000 LB/HR
5200 LB/HR
4800 LB/HR
4400 LB/HR
4000 LB/HR
3600 LB/HR
6400 LB/HR
6800 LB/HR
7200 LB/HR
7600 LB/HR
8000 LB/HR
8400 LB/HR
8800 LB/HR
9200 LB/HR
9600 LB/HR
10,000 LB/HR
10,400 LB/HR
10,800 LB/HR
92% RPM
96%
98%
87%
88%
89%
84%
REFER TO SECTION V FOR PYLON TANK OPERATING LIMITATIONS.

Figure A4-29

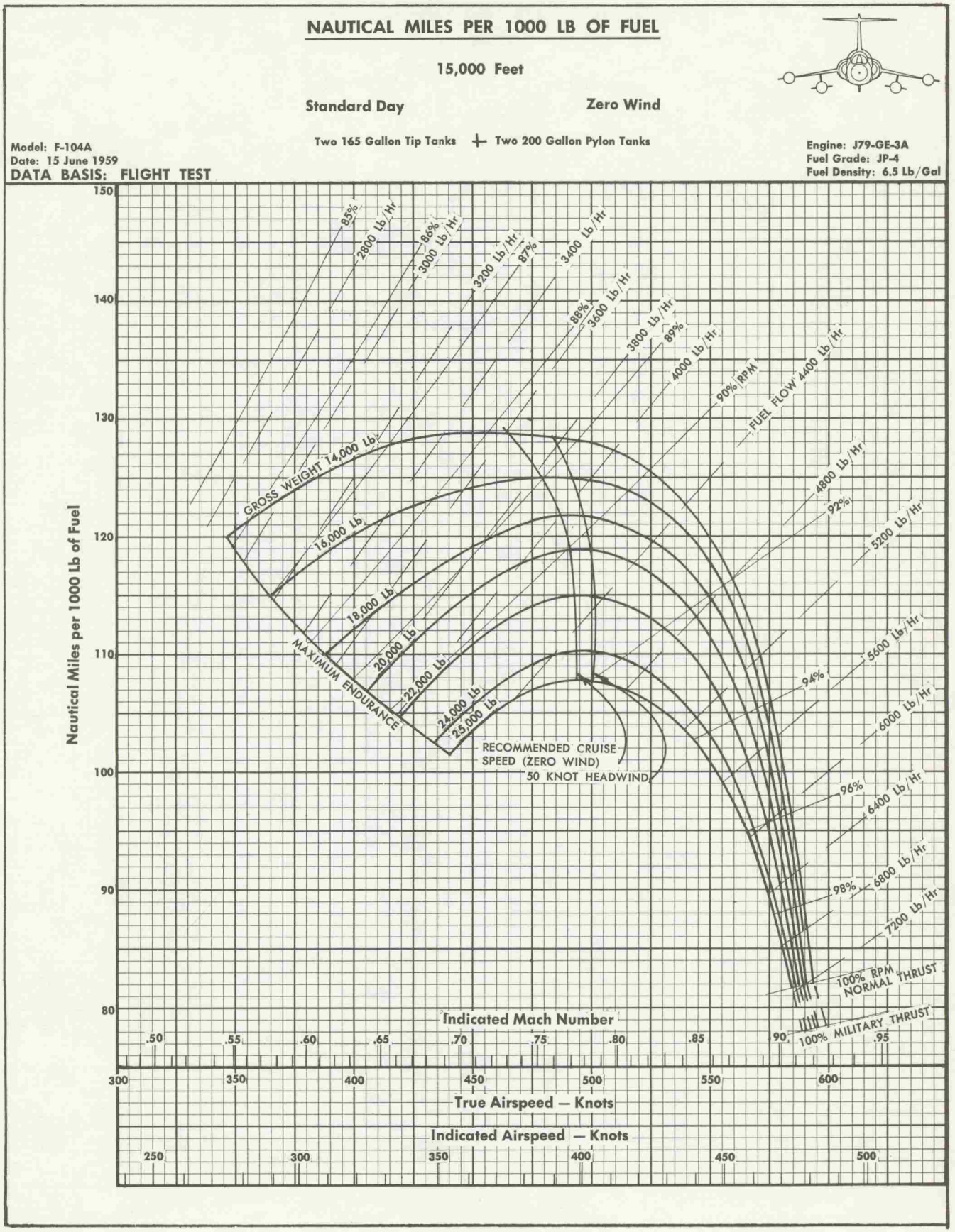

NAUTICAL MILES PER 1000 LB OF FUEL
15,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
150
140
130
120
110
100
90
80
85%
2800 Lb/Hr
86%
3000 Lb/Hr
3200 Lb/Hr
87%
3400 Lb/Hr
88%
3600 Lb/Hr
3800 Lb/Hr
89%
4000 Lb/Hr
90% RPM
FUEL FLOW 4400 Lb/Hr
4800 Lb/Hr
92%
5200 Lb/Hr
5600 Lb/Hr
94%
6000 Lb/Hr
96%
6400 Lb/Hr
6800 Lb/Hr
98%
7200 Lb/Hr
100% RPM NORMAL THRUST
100% MILITARY THRUST
GROSS WEIGHT 14,000 Lb
16,000 Lb
18,000 Lb
20,000 Lb
22,000 Lb
24,000 Lb
25,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
Indicated Mach Number
.50
.55
.60
.65
.70
.75
.80
.85
.90
.95
300
350
400
450
500
550
600
True Airspeed — Knots
Indicated Airspeed — Knots
250
300
350
400
450
500

Figure A4-30

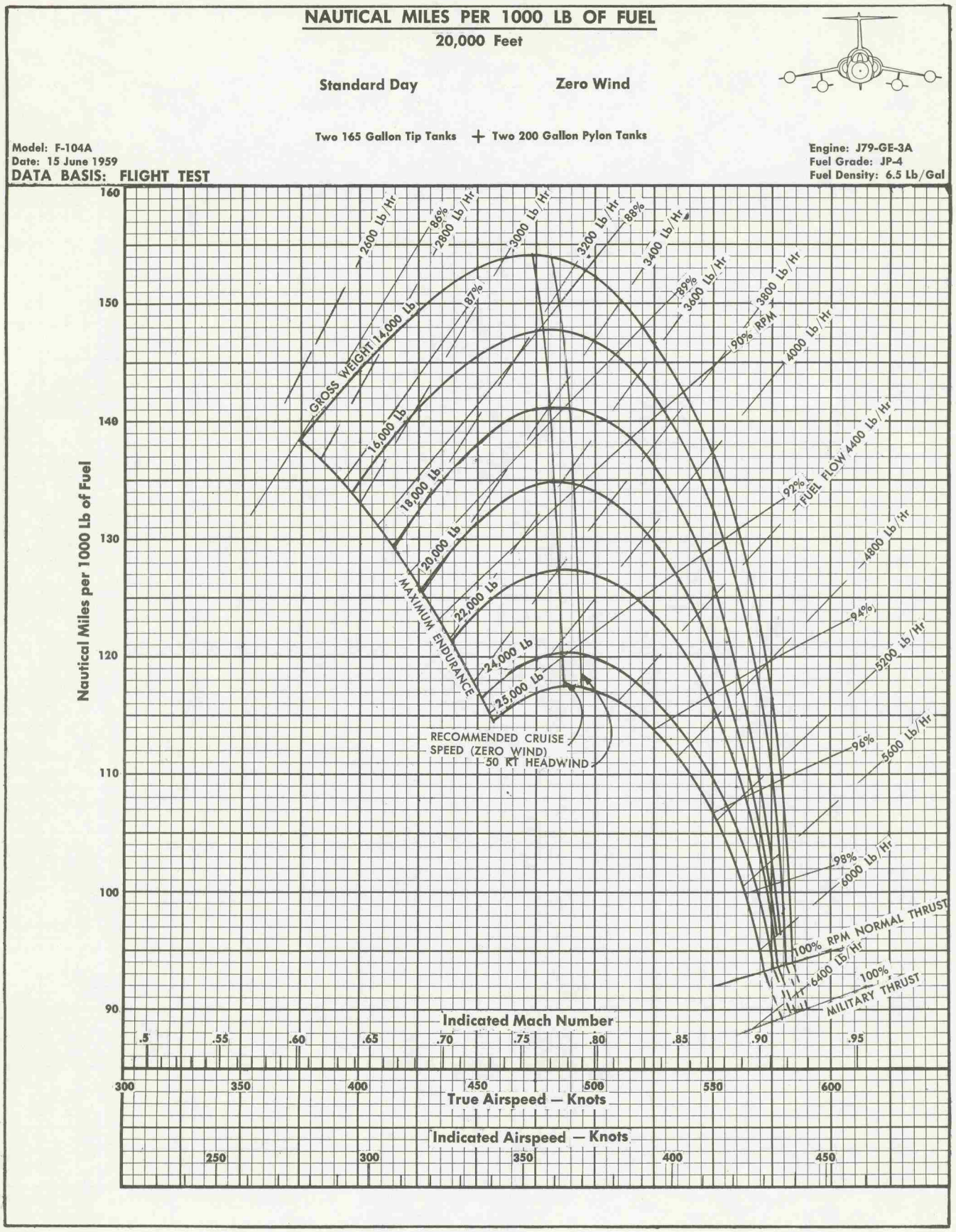
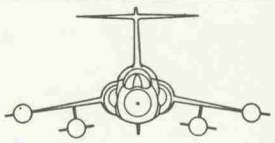

NAUTICAL MILES PER 1000 LB OF FUEL
20,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
160
150
140
130
120
110
100
90
GROSS WEIGHT
14,000 Lb
16,000 Lb
18,000 Lb
20,000 Lb
22,000 Lb
24,000 Lb
25,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KT HEADWIND
2600 Lb/Hr
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
86%
87%
88%
89%
90% RPM
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL FLOW
Indicated Mach Number
.5
.55
.60
.65
.70
.75
.80
.85
.90
.95
300
350
400
450
500
550
600
True Airspeed — Knots
Indicated Airspeed — Knots
250
300
350
400
450

Figure A4-31

NAUTICAL MILES PER 1000 LB OF FUEL



25,000 Feet

Standard Day — Zero Wind

Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

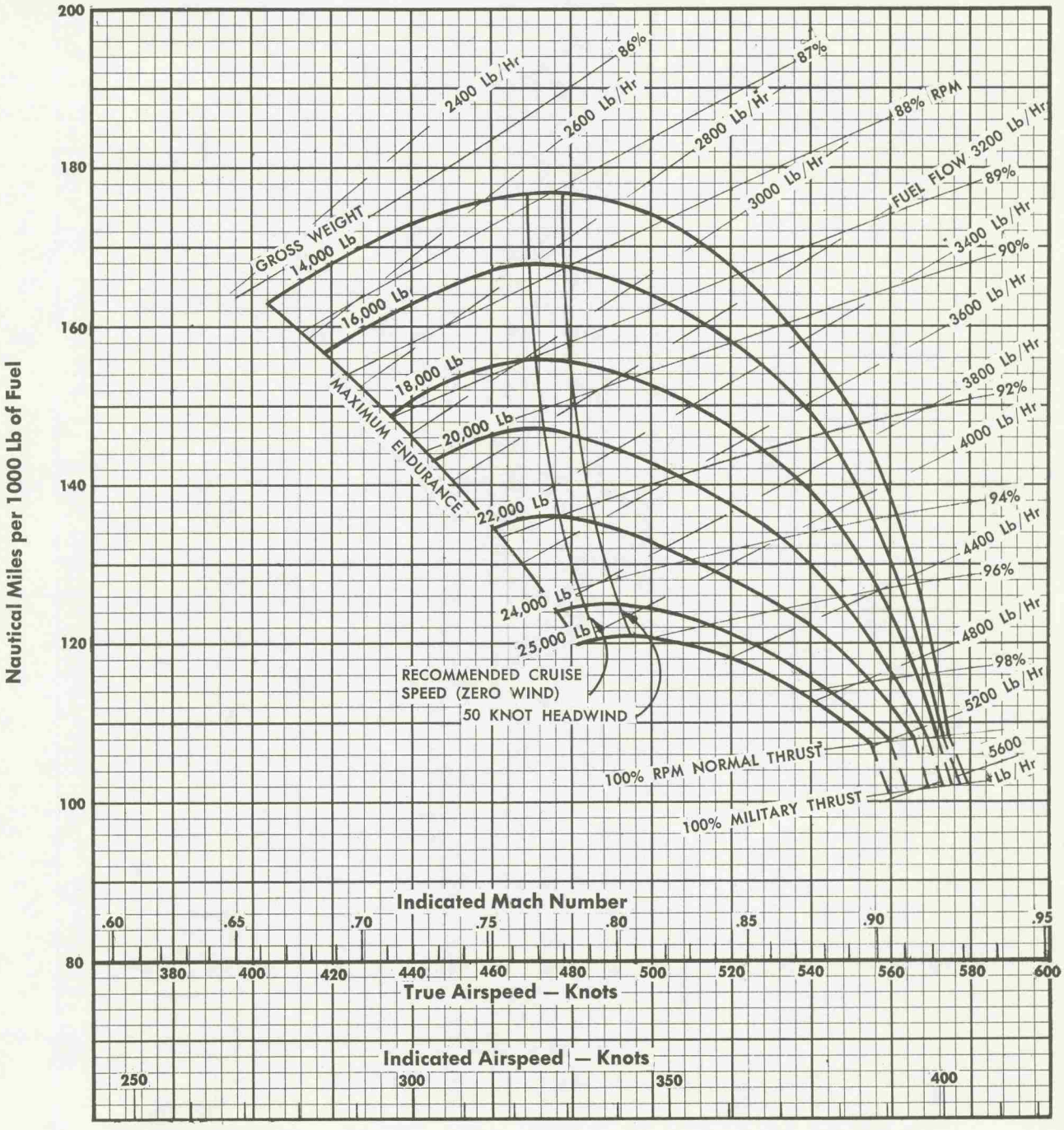


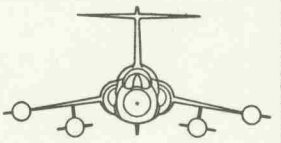
Figure A4-32

NAUTICAL MILES PER 1000 LB OF FUEL

30,000 Feet

Standard Day

Zero Wind



Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

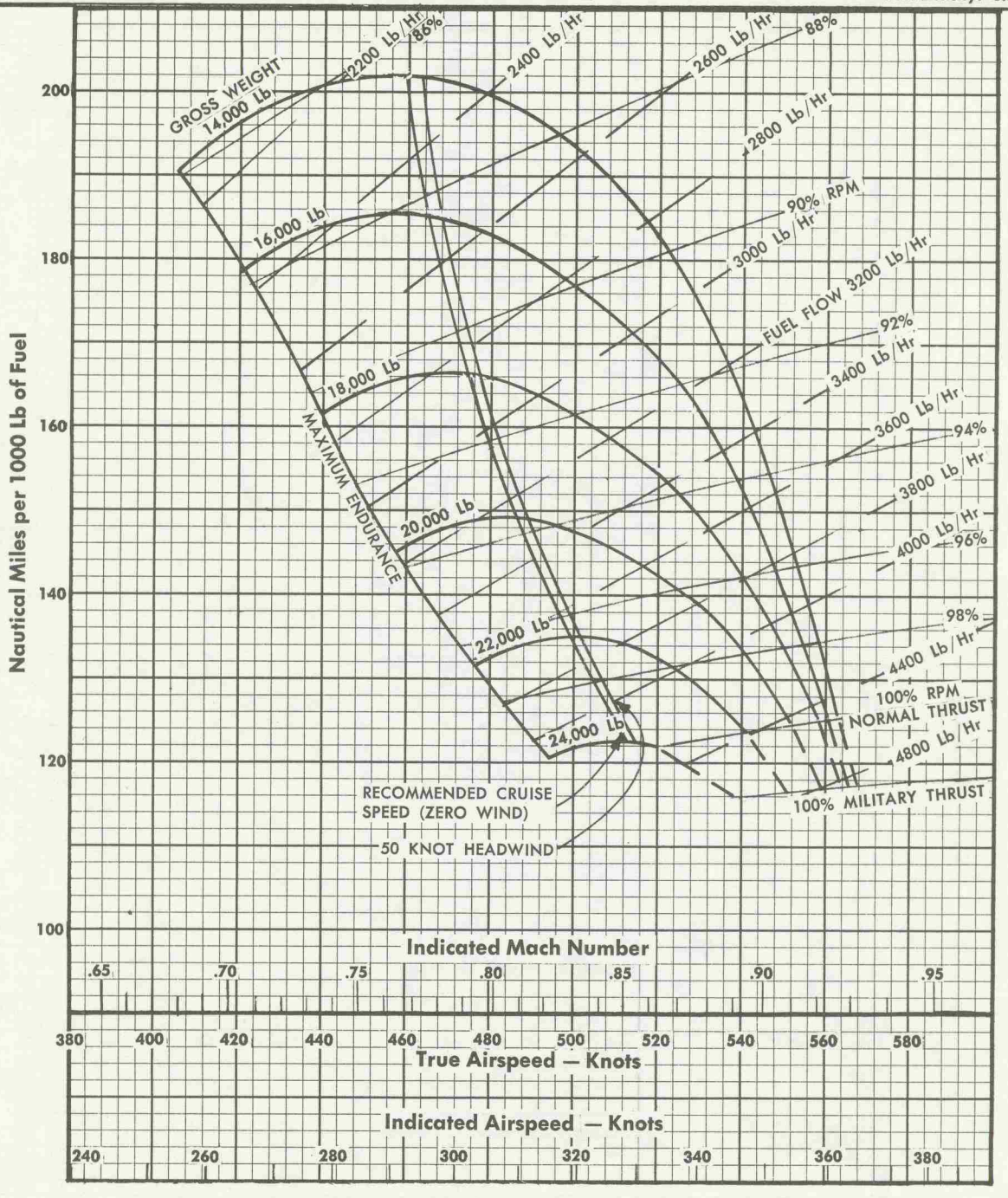
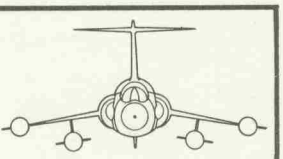


Figure A4-33

NAUTICAL MILES PER 1000 LB OF FUEL

35,000 Feet

Standard Day **Zero Wind**



Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

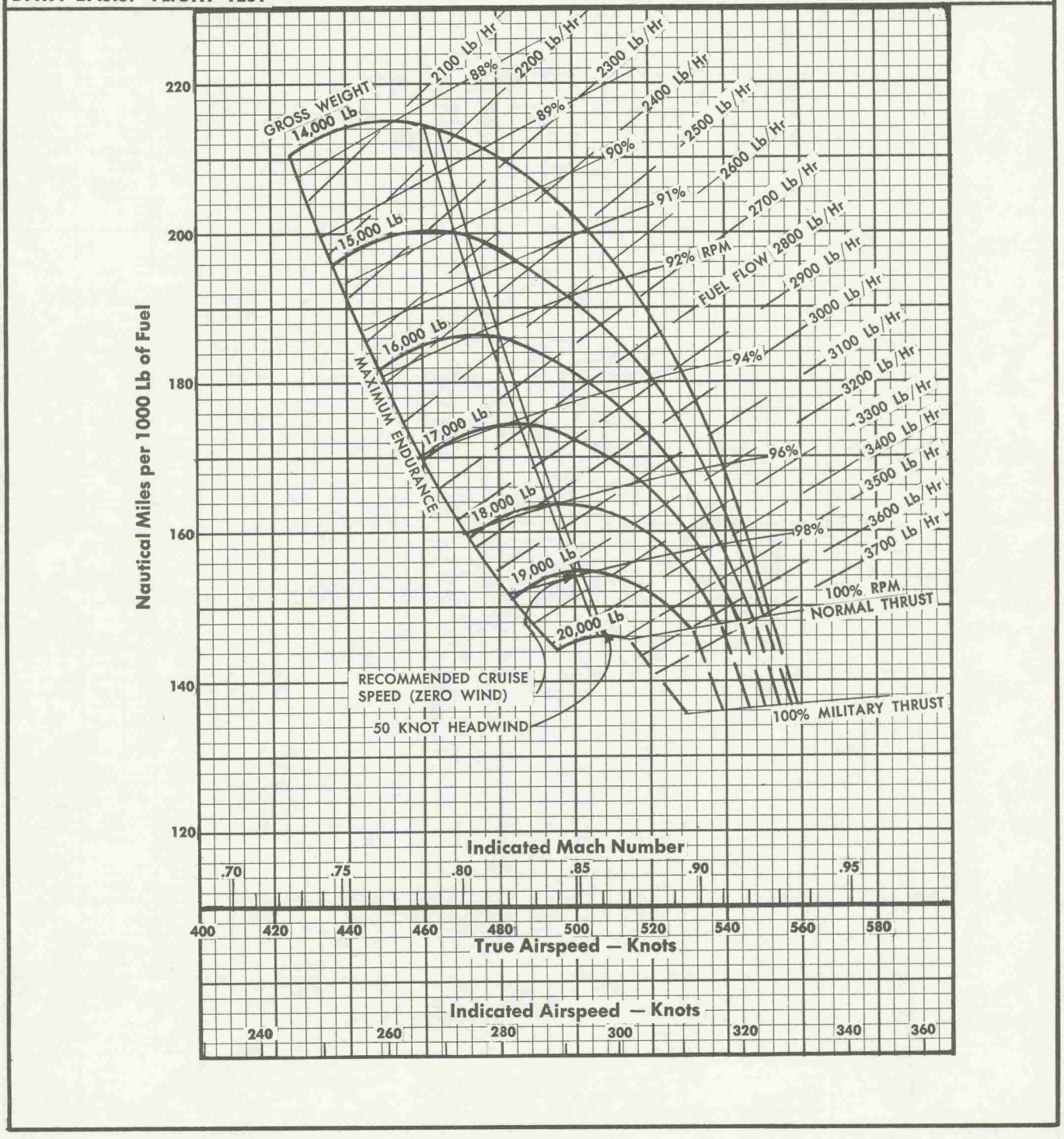


Figure A4-34

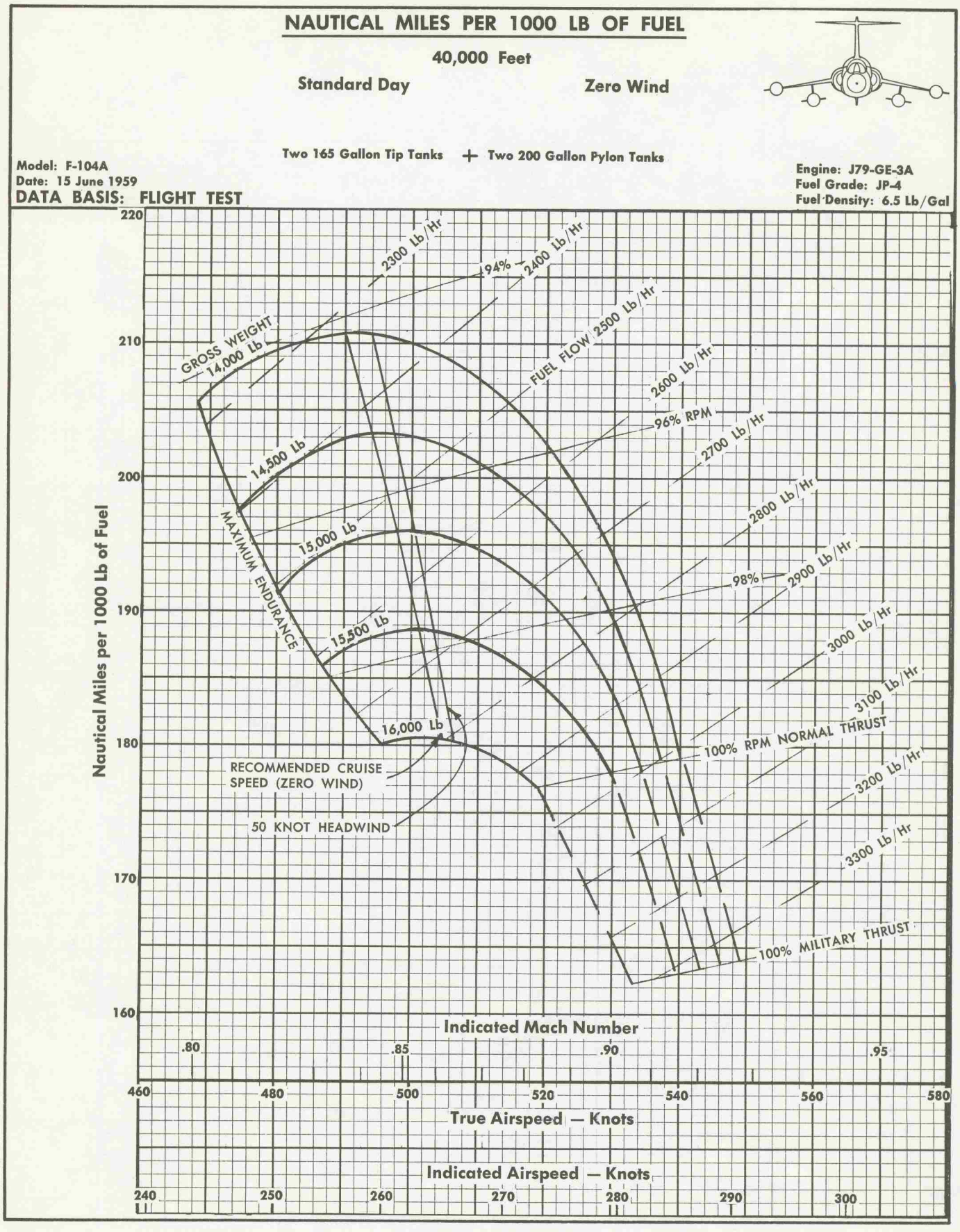

NAUTICAL MILES PER 1000 LB OF FUEL
40,000 Feet
Standard Day
Zero Wind
Two 165 Gallon Tip Tanks + Two 200 Gallon Pylon Tanks
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
220
210
200
190
180
170
160
GROSS WEIGHT
14,000 Lb
14,500 Lb
15,000 Lb
15,500 Lb
16,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE
SPEED (ZERO WIND)
50 KNOT HEADWIND
2300 Lb/Hr
2400 Lb/Hr
94%
FUEL FLOW 2500 Lb/Hr
2600 Lb/Hr
96% RPM
2700 Lb/Hr
2800 Lb/Hr
98%
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
100% RPM NORMAL THRUST
3200 Lb/Hr
3300 Lb/Hr
100% MILITARY THRUST
Indicated Mach Number
.80
.85
.90
.95
460
480
500
520
540
560
580
True Airspeed — Knots
Indicated Airspeed — Knots
240
250
260
270
280
290
300

Figure A4-35

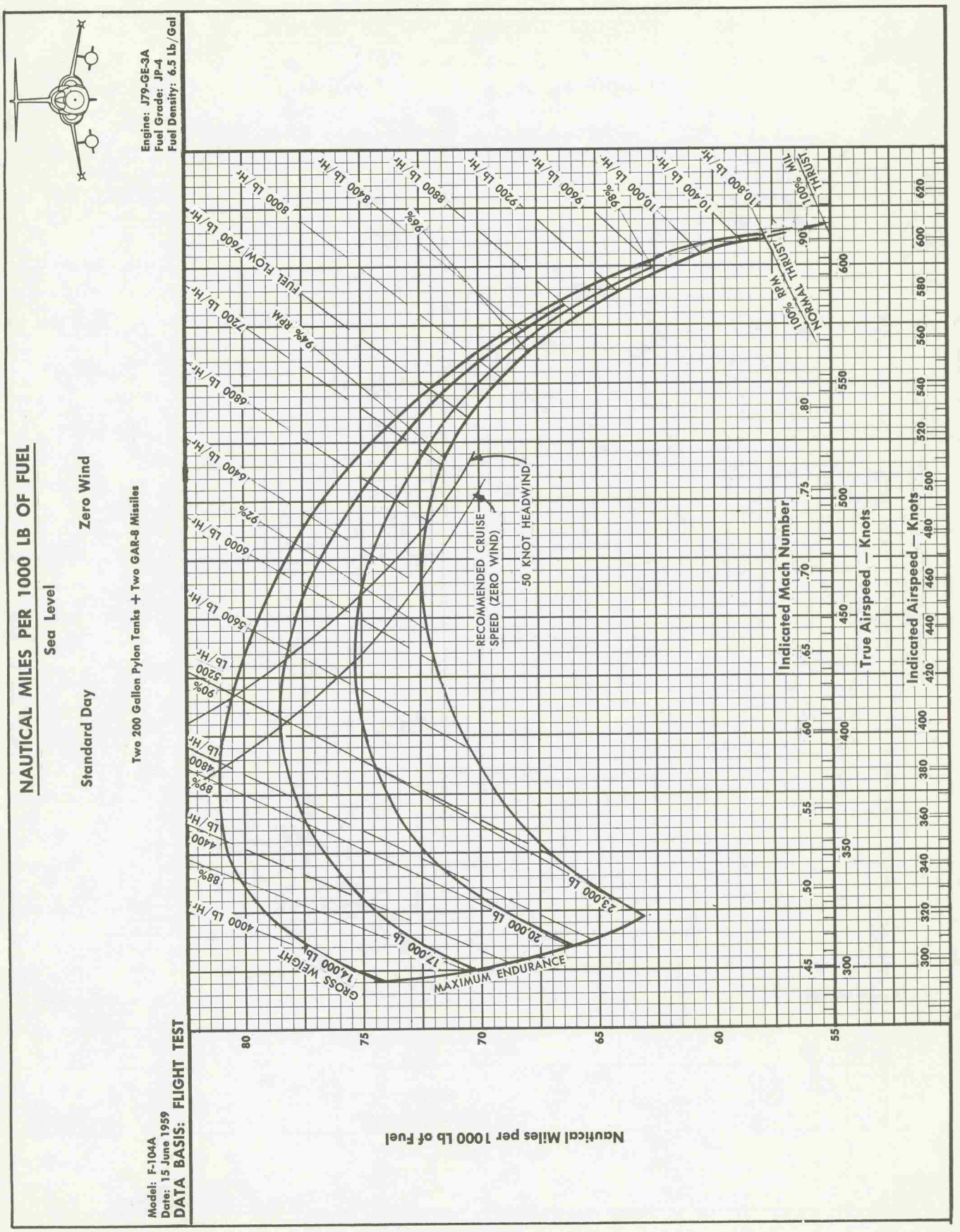

NAUTICAL MILES PER 1000 LB OF FUEL
Sea Level
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Nautical Miles per 1000 Lb of Fuel
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
MAXIMUM ENDURANCE
GROSS WEIGHT
14,000 LB
17,000 LB
20,000 LB
23,000 LB
FUEL FLOW
4000 LB/HR
4400 LB/HR
4800 LB/HR
5200 LB/HR
5600 LB/HR
6000 LB/HR
6400 LB/HR
6800 LB/HR
7200 LB/HR
7600 LB/HR
8000 LB/HR
8400 LB/HR
8800 LB/HR
9200 LB/HR
9600 LB/HR
10,000 LB/HR
10,400 LB/HR
10,800 LB/HR
88%
89%
90%
92%
94% RPM
96%
98%
100% RPM
NORMAL THRUST
100% MIL THRUST

Figure A4-36

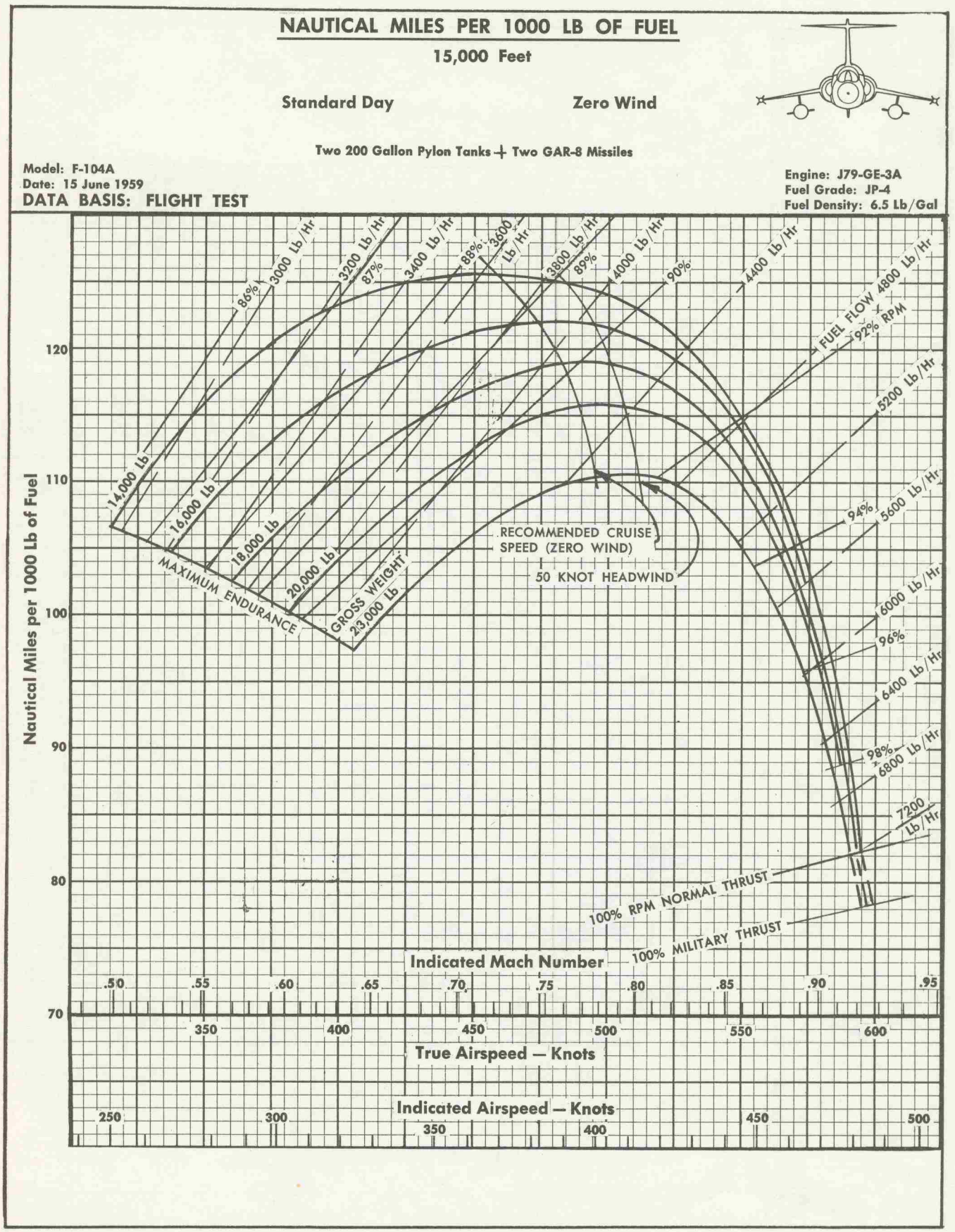

NAUTICAL MILES PER 1000 LB OF FUEL
15,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
120
110
100
90
80
70
14,000 Lb
16,000 Lb
18,000 Lb
20,000 Lb
GROSS WEIGHT 23,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
86%
3000 Lb/Hr
87%
3200 Lb/Hr
88%
3400 Lb/Hr
89%
3600 Lb/Hr
90%
3800 Lb/Hr
92%
4000 Lb/Hr
94%
4400 Lb/Hr
96%
4800 Lb/Hr
98%
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
6800 Lb/Hr
7200 Lb/Hr
FUEL FLOW
RPM
100% RPM NORMAL THRUST
100% MILITARY THRUST
Indicated Mach Number
.50 .55 .60 .65 .70 .75 .80 .85 .90 .95
350 400 450 500 550 600
True Airspeed — Knots
Indicated Airspeed — Knots
250 300 350 400 450 500

Figure A4-37

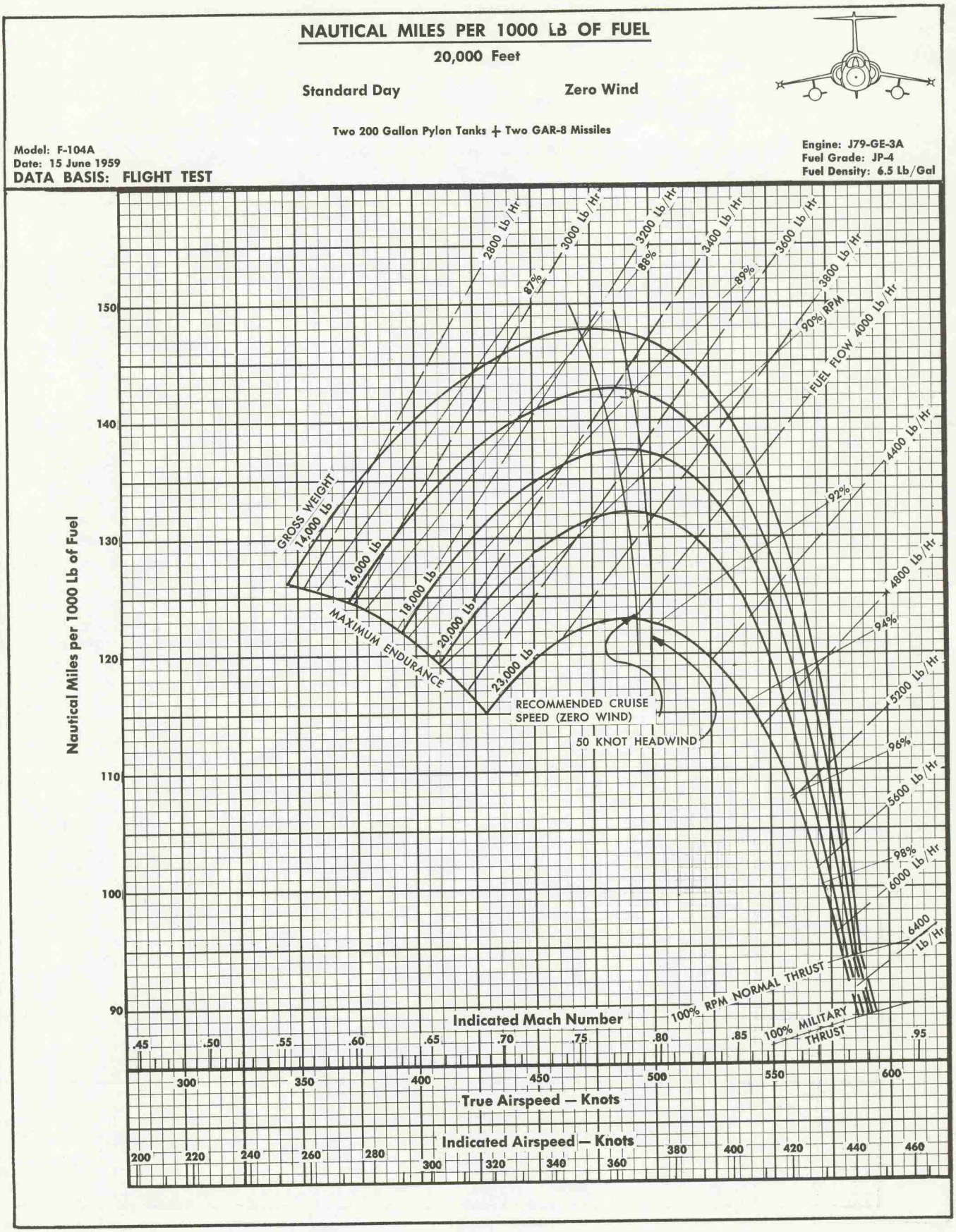

NAUTICAL MILES PER 1000 LB OF FUEL
20,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
14,000 Lb
16,000 Lb
18,000 Lb
20,000 Lb
23,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
2800 Lb/Hr
3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
6000 Lb/Hr
6400 Lb/Hr
87%
88%
89%
90% RPM
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
FUEL FLOW
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-38

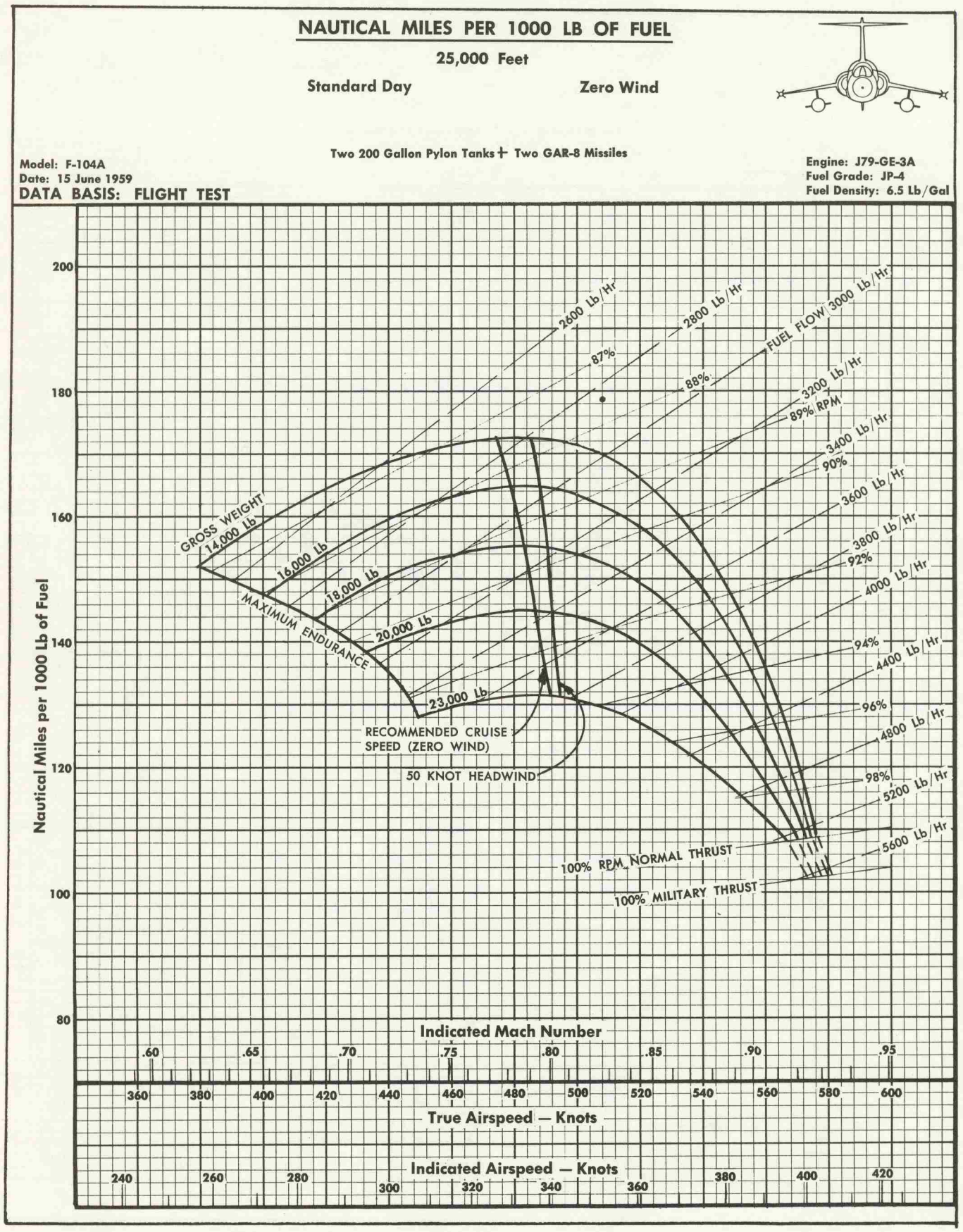

NAUTICAL MILES PER 1000 LB OF FUEL
25,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
200
180
160
140
120
100
80
GROSS WEIGHT
14,000 Lb
16,000 Lb
18,000 Lb
20,000 Lb
23,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
2600 Lb/Hr
2800 Lb/Hr
FUEL FLOW 3000 Lb/Hr
3200 Lb/Hr
3400 Lb/Hr
3600 Lb/Hr
3800 Lb/Hr
4000 Lb/Hr
4400 Lb/Hr
4800 Lb/Hr
5200 Lb/Hr
5600 Lb/Hr
87%
88%
89% RPM
90%
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
Indicated Mach Number
.60
.65
.70
.75
.80
.85
.90
.95
360
380
400
420
440
460
480
500
520
540
560
580
600
True Airspeed — Knots
240
260
280
300
320
340
360
380
400
420
Indicated Airspeed — Knots

Figure A4-39

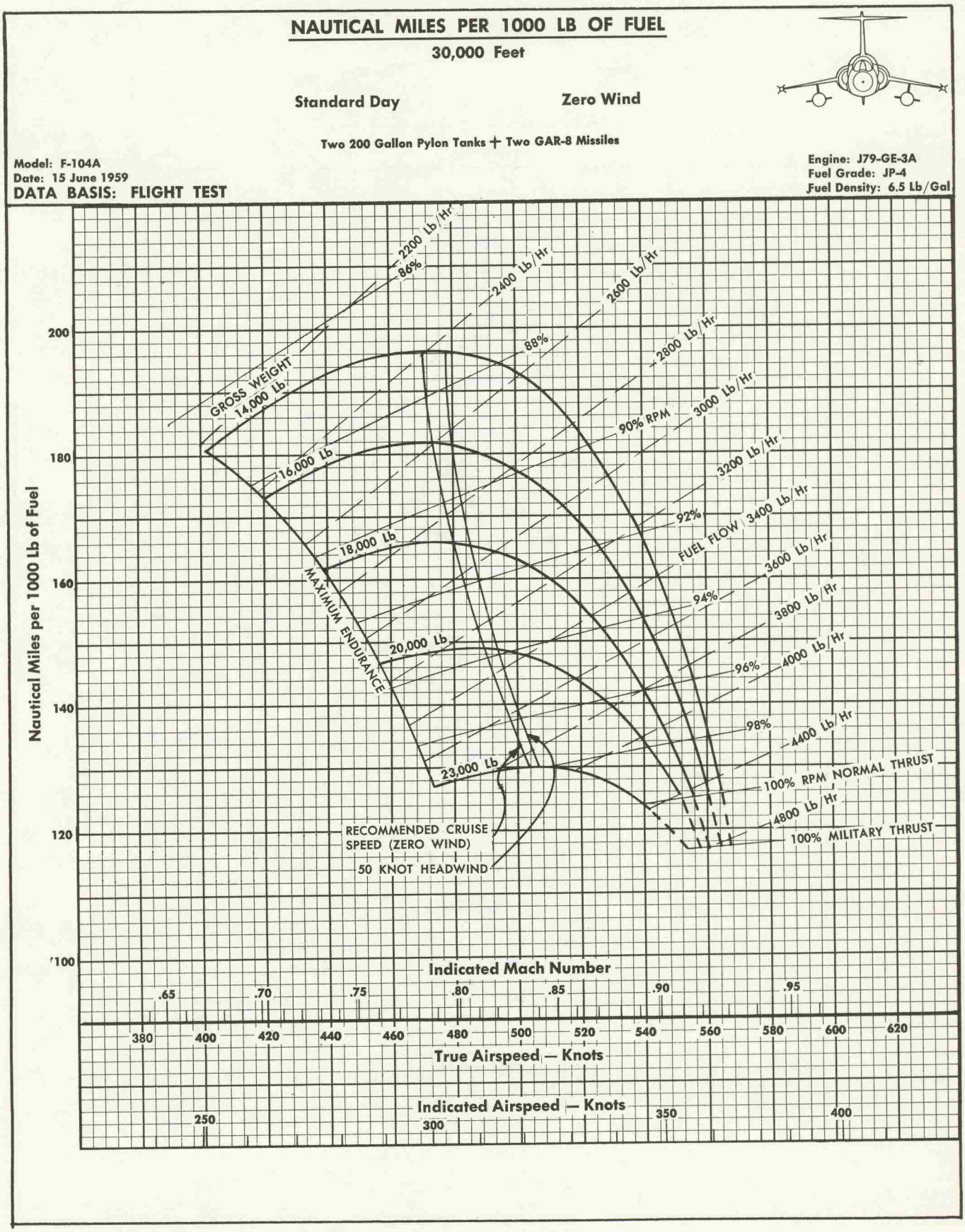

NAUTICAL MILES PER 1000 LB OF FUEL
30,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
200
180
160
140
120
100
2200 Lb/Hr
86%
2400 Lb/Hr
2600 Lb/Hr
88%
2800 Lb/Hr
GROSS WEIGHT
14,000 Lb
90% RPM
3000 Lb/Hr
16,000 Lb
3200 Lb/Hr
92%
FUEL FLOW 3400 Lb/Hr
18,000 Lb
3600 Lb/Hr
MAXIMUM ENDURANCE
94%
3800 Lb/Hr
20,000 Lb
4000 Lb/Hr
96%
98%
4400 Lb/Hr
23,000 Lb
100% RPM NORMAL THRUST
4800 Lb/Hr
100% MILITARY THRUST
RECOMMENDED CRUISE
SPEED (ZERO WIND)
50 KNOT HEADWIND
Indicated Mach Number
.65
.70
.75
.80
.85
.90
.95
380
400
420
440
460
480
500
520
540
560
580
600
620
True Airspeed — Knots
Indicated Airspeed — Knots
250
300
350
400

Figure A4-40

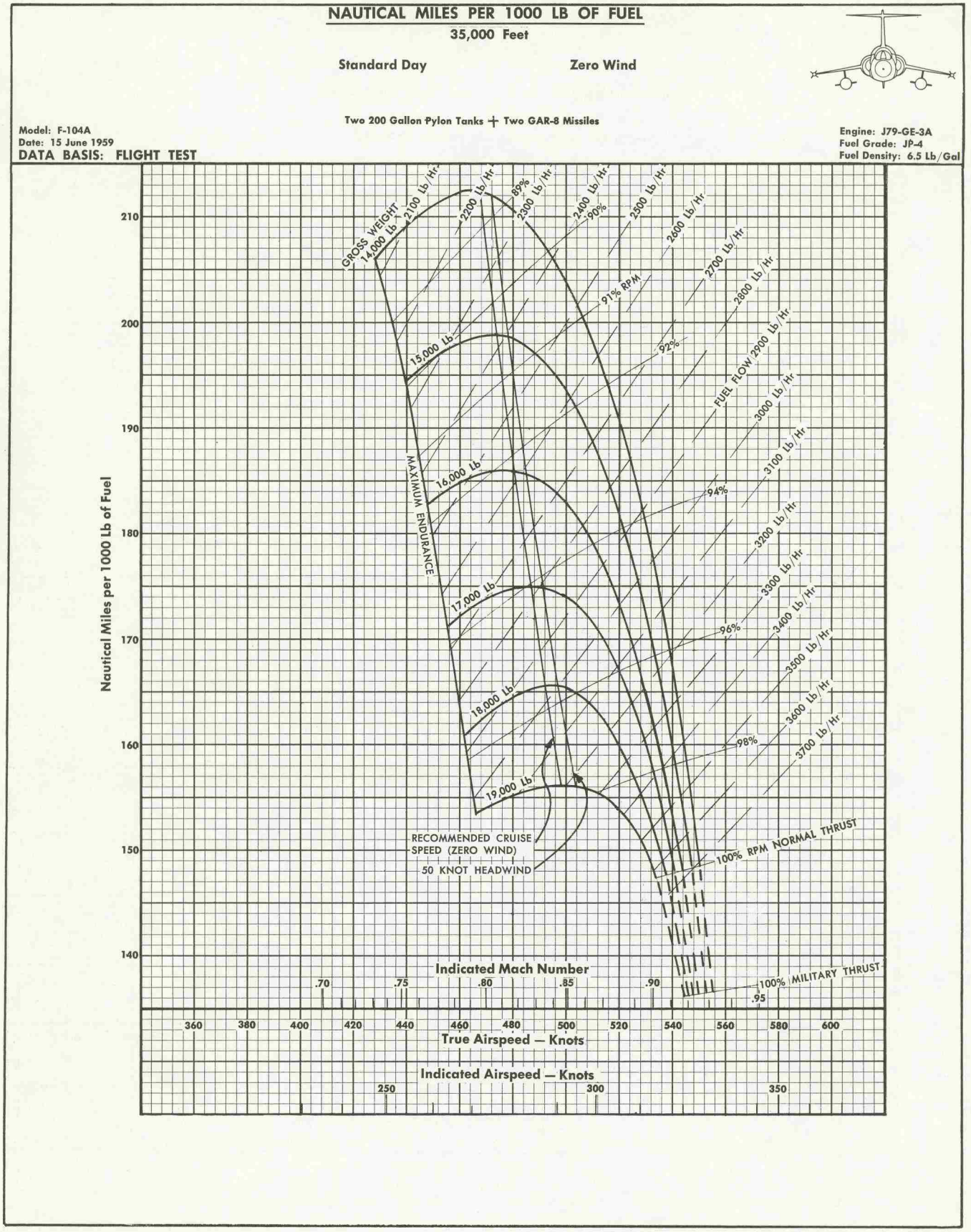

NAUTICAL MILES PER 1000 LB OF FUEL
35,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
GROSS WEIGHT
14,000 Lb
15,000 Lb
16,000 Lb
17,000 Lb
18,000 Lb
19,000 Lb
MAXIMUM ENDURANCE
2100 Lb/Hr
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
2500 Lb/Hr
2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
3500 Lb/Hr
3600 Lb/Hr
3700 Lb/Hr
FUEL FLOW
89%
90%
91% RPM
92%
94%
96%
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
Indicated Mach Number
True Airspeed — Knots
Indicated Airspeed — Knots

Figure A4-41

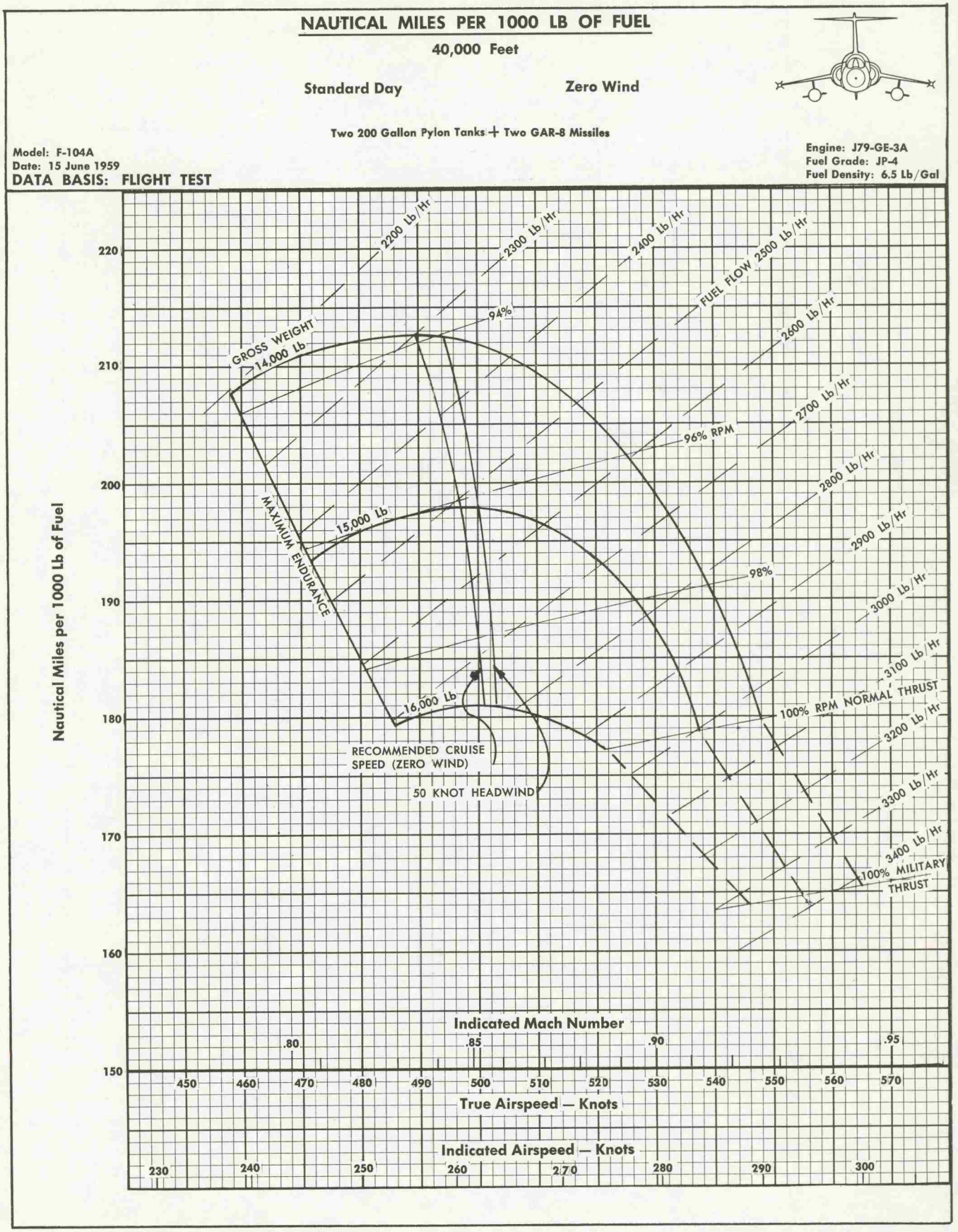

NAUTICAL MILES PER 1000 LB OF FUEL
40,000 Feet
Standard Day
Zero Wind
Two 200 Gallon Pylon Tanks + Two GAR-8 Missiles
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Nautical Miles per 1000 Lb of Fuel
220
210
200
190
180
170
160
150
2200 Lb/Hr
2300 Lb/Hr
2400 Lb/Hr
FUEL FLOW 2500 Lb/Hr
2600 Lb/Hr
2700 Lb/Hr
2800 Lb/Hr
2900 Lb/Hr
3000 Lb/Hr
3100 Lb/Hr
3200 Lb/Hr
3300 Lb/Hr
3400 Lb/Hr
94%
96% RPM
98%
100% RPM NORMAL THRUST
100% MILITARY THRUST
GROSS WEIGHT
14,000 Lb
15,000 Lb
16,000 Lb
MAXIMUM ENDURANCE
RECOMMENDED CRUISE SPEED (ZERO WIND)
50 KNOT HEADWIND
Indicated Mach Number
.80
.85
.90
.95
450
460
470
480
490
500
510
520
530
540
550
560
570
True Airspeed — Knots
Indicated Airspeed — Knots
230
240
250
260
270
280
290
300

Figure A4-42

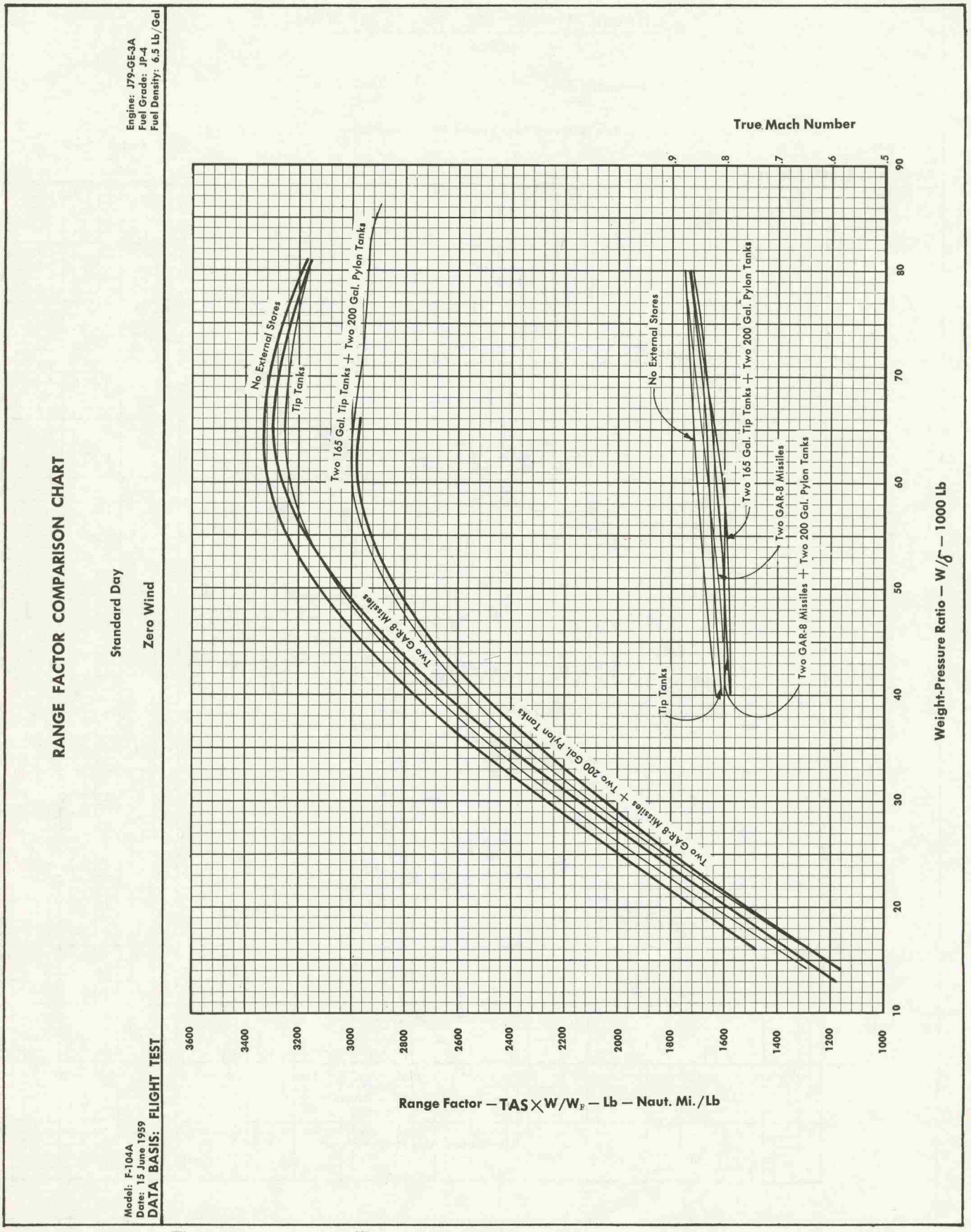

RANGE FACTOR COMPARISON CHART
Standard Day
Zero Wind
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
True Mach Number
.5
.6
.7
.8
.9
10
20
30
40
50
60
70
80
90
Weight-Pressure Ratio — W/δ — 1000 Lb
No External Stores
Tip Tanks
Two 165 Gal. Tip Tanks + Two 200 Gal. Pylon Tanks
Two GAR-8 Missiles
Two GAR-8 Missiles + Two 200 Gal. Pylon Tanks
No External Stores
Two 165 Gal. Tip Tanks + Two 200 Gal. Pylon Tanks
Two GAR-8 Missiles
Two GAR-8 Missiles + Two 200 Gal. Pylon Tanks
Tip Tanks
3600
3400
3200
3000
2800
2600
2400
2200
2000
1800
1600
1400
1200
1000
Range Factor — TAS × W/W_F — Lb — Naut. Mi./Lb

Figure A4-43

Part 5—Endurance

TABLE OF CONTENTS

Page numbers in bold face denote illustration

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Maximum Endurance Profile Charts	A5-1
Optimum Maximum Endurance Profile Charts	A5-1
Climb to Optimum Altitude	A5-1
Use of Optimum Maximum Endurance Profile Charts	A5-2
Maximum Endurance Profile—No External Stores	**A5-4**
Maximum Endurance Profile—GAR-8 Missiles	**A5-5**
Optimum Maximum Endurance Profile—No External Stores	**A5-6**
Optimum Maximum Endurance Profile—Two GAR-8 Missiles	**A5-8**

ENDURANCE

MAXIMUM ENDURANCE PROFILE CHARTS.

These charts show the maximum flight time obtainable with any amount of fuel remaining when flying at a constant altitude. The recommended airspeeds shown in the table must be used to obtain predicted performance. A sample chart illustrates the method of using this type of chart.

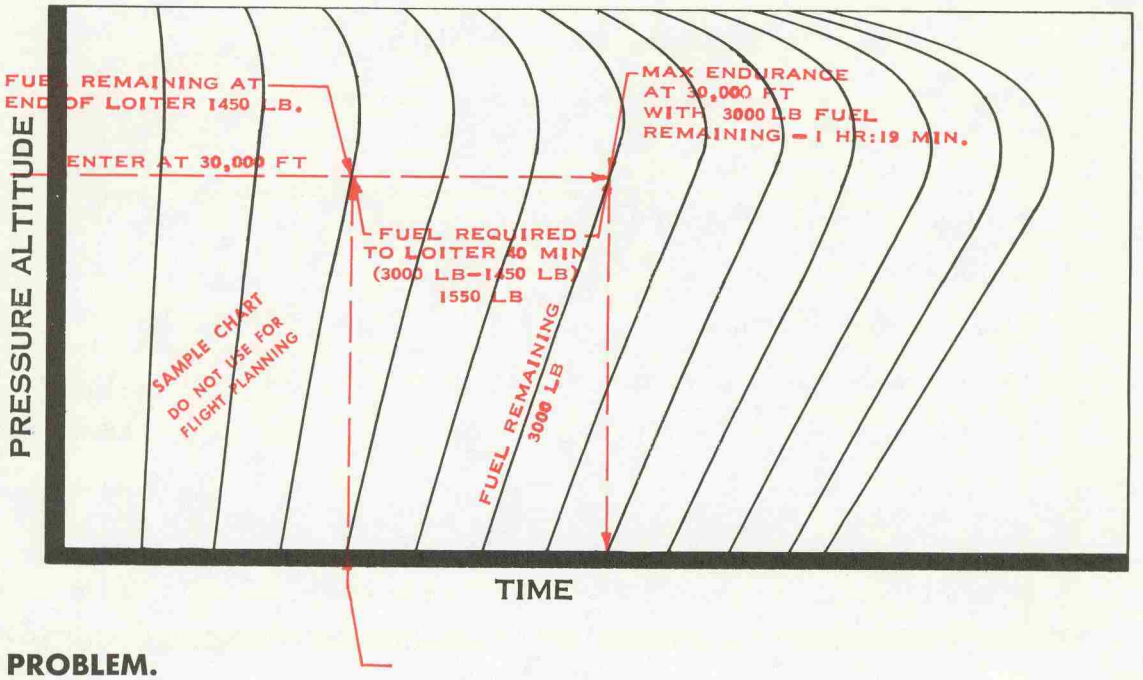
OPTIMUM MAXIMUM ENDURANCE PROFILE CHARTS.

The optimum maximum endurance profiles show the maximum loiter time for any fuel remaining value, based on loitering at optimum altitude. This endurance includes the effect of weight reduction due to the consumption of fuel during loiter. The area above the optimum loitering altitude line also indicates the maximum endurance that can be realized for a specific quantity of fuel remaining and altitude. It can be seen, however, that maximum endurance can be attained only by first descending or climbing to the optimum loitering altitude. The area under the optimum loitering altitude line indicates the altitudes and fuel loads at which a Military Thrust climb should be made to the optimum loitering altitude to obtain maximum endurance. (Climb speed schedules applicable to the various configurations are tabulated on the appropriate profile charts). Constant lines of fuel-on-board are indicated by dashed lines. The fuel lines are based on a Military Thrust climb to optimum altitude (if climb is necessary), loiter at altitude and idle thrust descent at 300 knots IAS to sea level. No allowance is made for a landing reserve. Climb path guide lines are shown in the area below the optimum loitering altitude line by solid lines. Recommended loiter speeds are tabulated for each altitude, as are approximate fuel consumption and engine speed values to be expected.

CLIMB TO OPTIMUM ALTITUDE.

It must be emphasized that for altitudes and fuel loadings where a climb is designated, it is necessary to climb to optimum altitude in order to obtain the endurance

MAXIMUM ENDURANCE PROFILE



SAMPLE PROBLEM.

From the sample chart shown, determine the fuel required to loiter 40 minutes at 30,000 feet with tip GAR-8 missiles installed. The fuel on board at the start of loiter is 3,000 pounds.

1. Read chart at 30,000 feet and 3,000 lbs. fuel remaining . . . . . 1 hr. 19 min.

2. Final time (1:19 — 0:40) . . . . . 0 hr. 39 min.

3. Enter chart at 30,000 feet and time of 39 min. Read fuel on board at end of loiter. . . . . 1,450 pounds

4. Fuel required to loiter [initial fuel load—final load (3000 pounds — 1450 pounds)] . . . . . 1,550 pounds

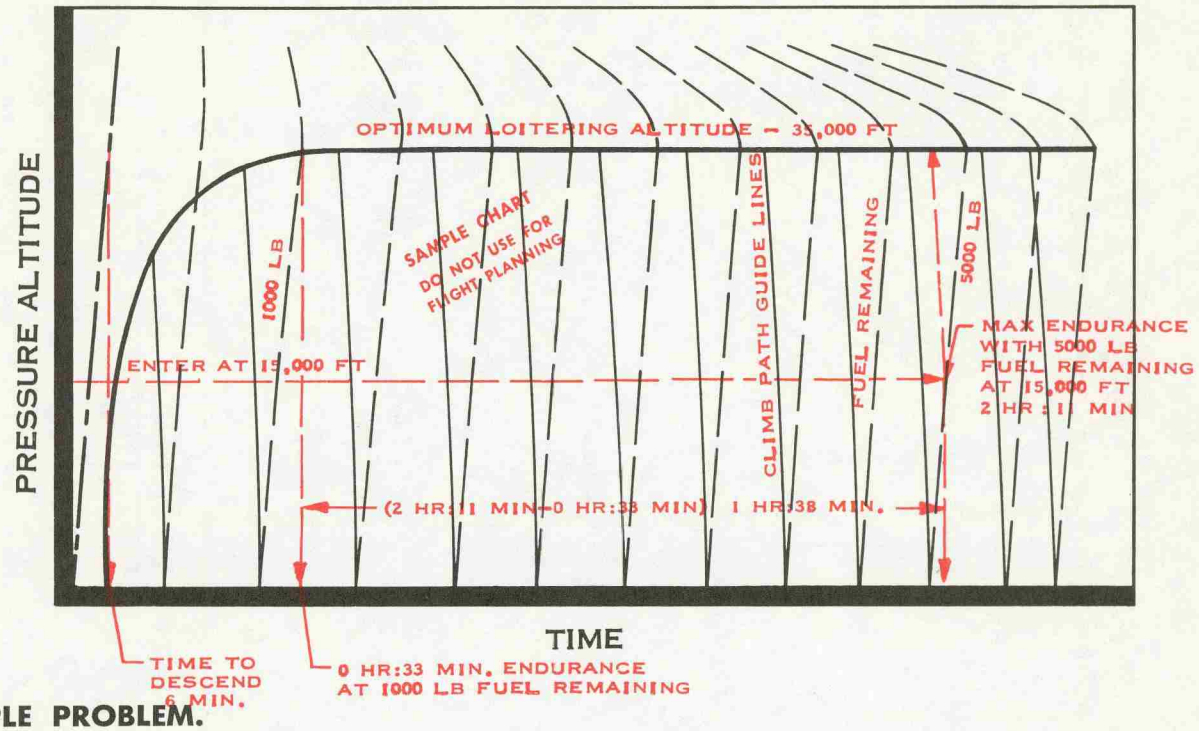
5. Recommended IAS, from tabulated data. . . . . 275 knots

shown on the charts. Climb should not be made above optimum altitudes because to do so will result in less endurance. The climb speed schedule appears on the left side of the chart.

USE OF OPTIMUM MAXIMUM ENDURANCE PROFILE CHARTS.

The chart is utilized by entering at the initial altitude with either the maximum airborne time desired (to determine the fuel requirement) or the fuel remaining (to determine the maximum airborne time available). If the initial point falls in the area above the optimum loitering altitude line, descend to the optimum loitering altitude and read the maximum total endurance, which includes time for a descent to sea level at 300 knots IAS. If the initial point falls in the area below the optimum loitering altitude line, read directly the maximum total endurance. To obtain this endurance it is necessary to climb immediately at Military Thrust to optimum loitering altitude. This value of maximum total endurance also includes descent at 300 knots IAS.

OPTIMUM MAXIMUM ENDURANCE PROFILE



SAMPLE PROBLEM.

From the sample chart shown above, determine the maximum endurance time available and the necessary flight path for the following condition: 5,000 pounds of fuel remaining, initial altitude of 15,000 feet, two GAR-8 missiles installed. It is desired to have 1,000 pounds of fuel remaining at end of loiter as a landing reserve.

1. Enter profile at 15,000 feet and 5,000 pounds of fuel remaining to establish initial point. Total time available if all fuel is expended . . . . . 2 hr. 11 min.

(Note that a climb is required in this area.)

2. Follow climb path guide lines to the optimum loitering altitude line to determine best endurance altitude . . . . . 35,000 ft.

3. Enter profile at 35,000 feet and 1,000 pounds of fuel remaining to find time at end of loiter . . . . . 33 min.

4. Time from start of climb to start of descent (2:11 — :33) . . . . . 1 hr. 38 min.

5. Time to descend from 35,000 feet to sea level . . . . . 6 min.

6. Maximum time available to remain airborne (1:38 + :06) . . . . . 1 hr. 44 min.

MAXIMUM ENDURANCE PROFILE

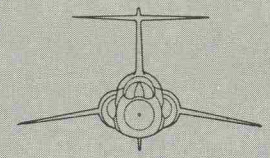
Standard Day

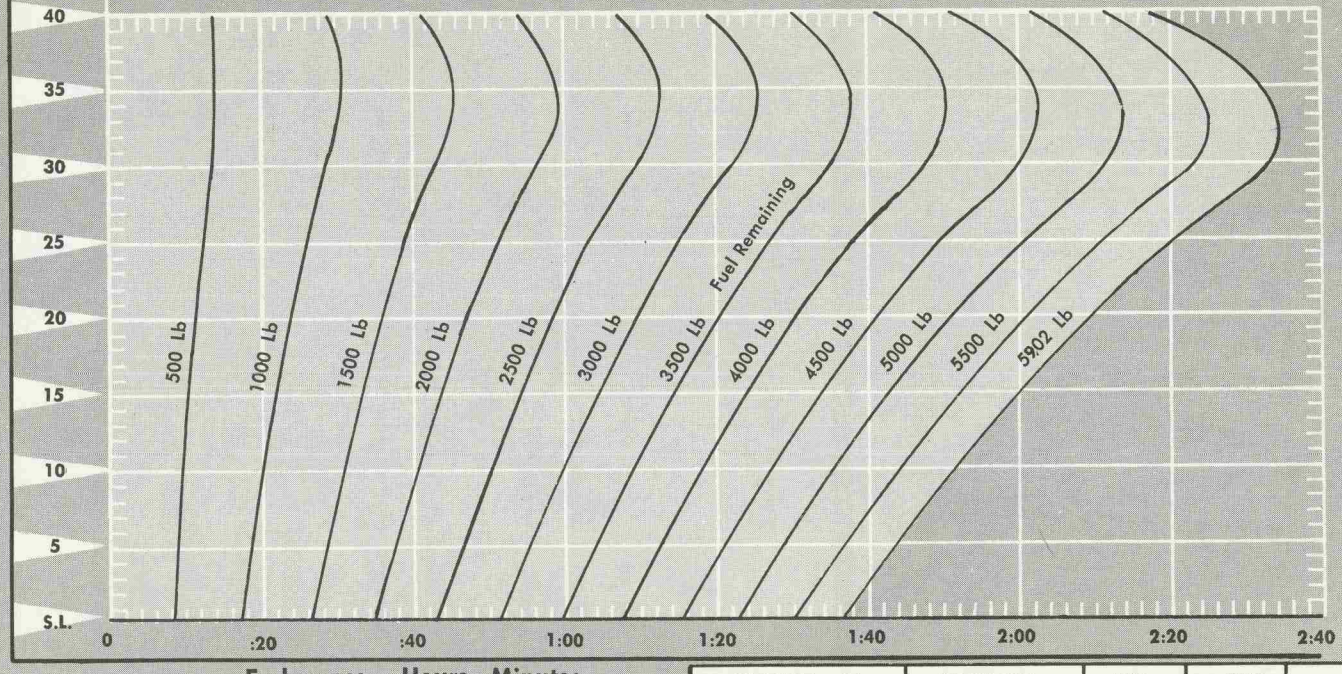
Zero Wind

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Configuration
No External Stores
Gross Weight Range
19,000 Lb — 13,098 Lb





ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
40,000	39,570	.86	.84	250	495	97.0	2570
35,000	34,650	.80	.78	275	465	89.5	2300
30,000	29,690	.74	.73	265	435	87.0	2320
25,000	24,730	.69	.68	280	415	87.0	2560
20,000	19,900	.62	.61	280	380	86.5	2810
15,000	14,850	.56	.55	280	350	86.5	2940
10,000	9,880	.52	.51	285	330	86.0	3150
5,000	4,920	.49	.48	295	320	86.5	3410
S. L.	—70	.47	.46	310	310	86.8	3670

NOTE:
1. Loiter at recommended Mach Number.
2. Maintain Constant Altitude.

* Approximate

Figure A5-1

MAXIMUM ENDURANCE PROFILE

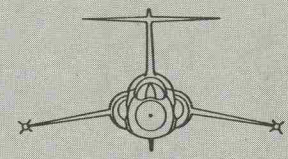
Standard Day

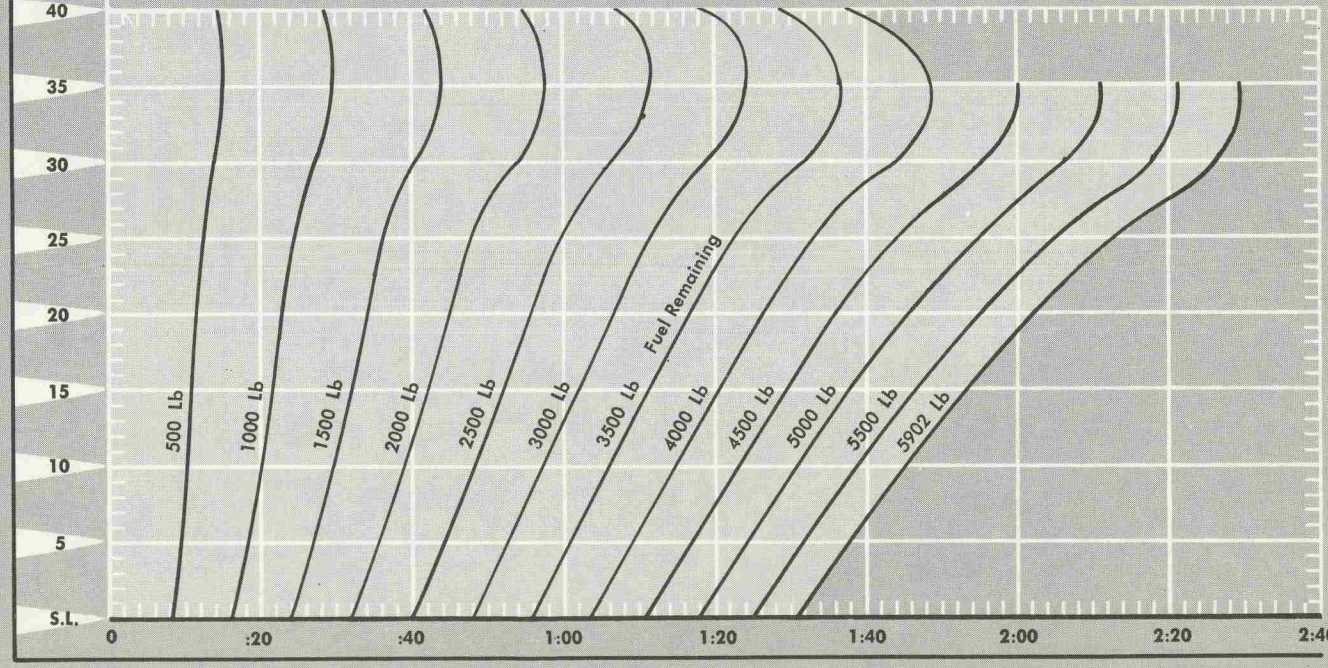
Zero Wind

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Configuration
Two GAR-8 Missiles
Gross Weight Range
19,420 Lb – 13,518 Lb





ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
40,000	39,610	.84	.82	250	480	98.5	2660
35,000	34,640	.81	.79	270	465	90.5	2380
30,000	29,690	.75	.73	275	440	87.5	2390
25,000	24,740	.69	.68	280	415	87.5	2640
20,000	19,820	.60	.59	270	370	87.0	2860
15,000	14,860	.55	.54	275	340	87.0	3100
10,000	9,890	.51	.50	280	325	87.0	3340
5,000	4,900	.48	.47	290	310	87.0	3600
S. L.	—90	.46	.45	300	300	87.2	3900

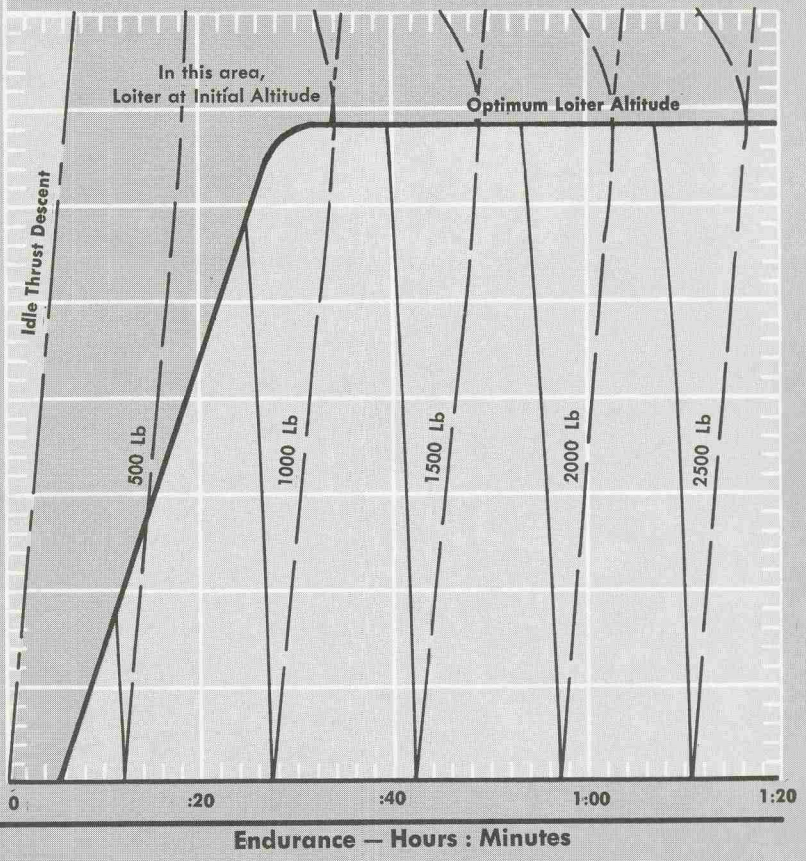
NOTE:
1. Loiter at recommended Mach Number.
2. Maintain Constant Altitude.

* Approximate

Figure A5-2

OPTIMUM MAXIMUM ENDURANCE PROFILE

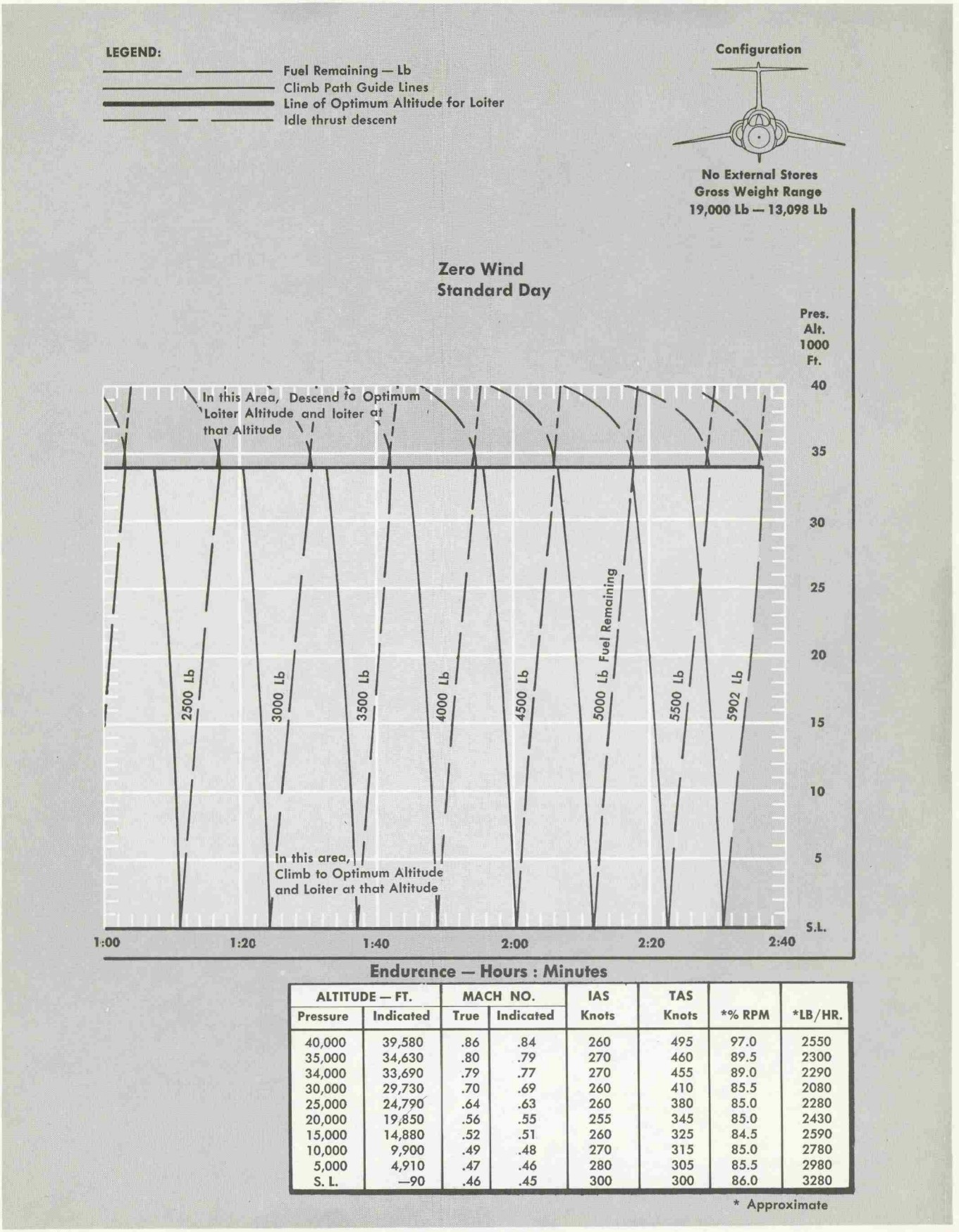
Climb Data			
Military Thrust			
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	573	S.L.



NOTE:

1. Fuel required at any point includes Military Thrust Climb, if required, to altitude, loiter and Idle Thrust descent to Sea Level. (See Military Thrust Climb Chart for detailed information.)
2. Loiter at recommended Mach Number.
3. Recommended descent at Idle Thrust and 300 Knots IAS. (See Descent Chart for detailed information.)
4. No allowance or reserve made for landing.

Figure A5-3 (Sheet 1)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
40,000	39,580	.86	.84	260	495	97.0	2550
35,000	34,630	.80	.79	270	460	89.5	2300
34,000	33,690	.79	.77	270	455	89.0	2290
30,000	29,730	.70	.69	260	410	85.5	2080
25,000	24,790	.64	.63	260	380	85.0	2280
20,000	19,850	.56	.55	255	345	85.0	2430
15,000	14,880	.52	.51	260	325	84.5	2590
10,000	9,900	.49	.48	270	315	85.0	2780
5,000	4,910	.47	.46	280	305	85.5	2980
S. L.	—90	.46	.45	300	300	86.0	3280

* Approximate

Figure A5-3 (Sheet 2)

OPTIMUM MAXIMUM ENDURANCE PROFILE

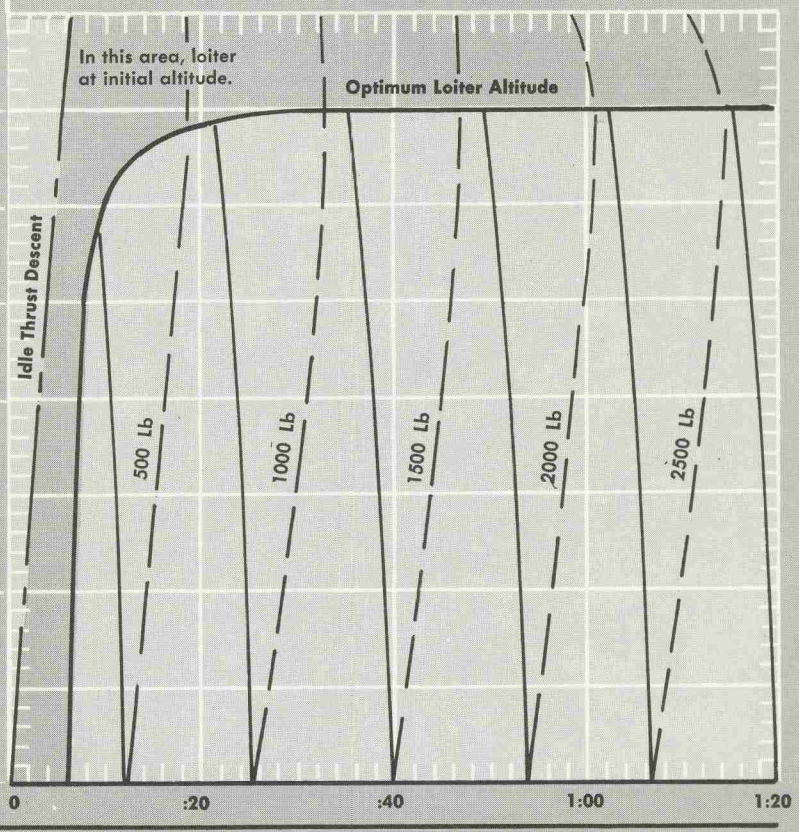
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal

Climb Data

Military Thrust

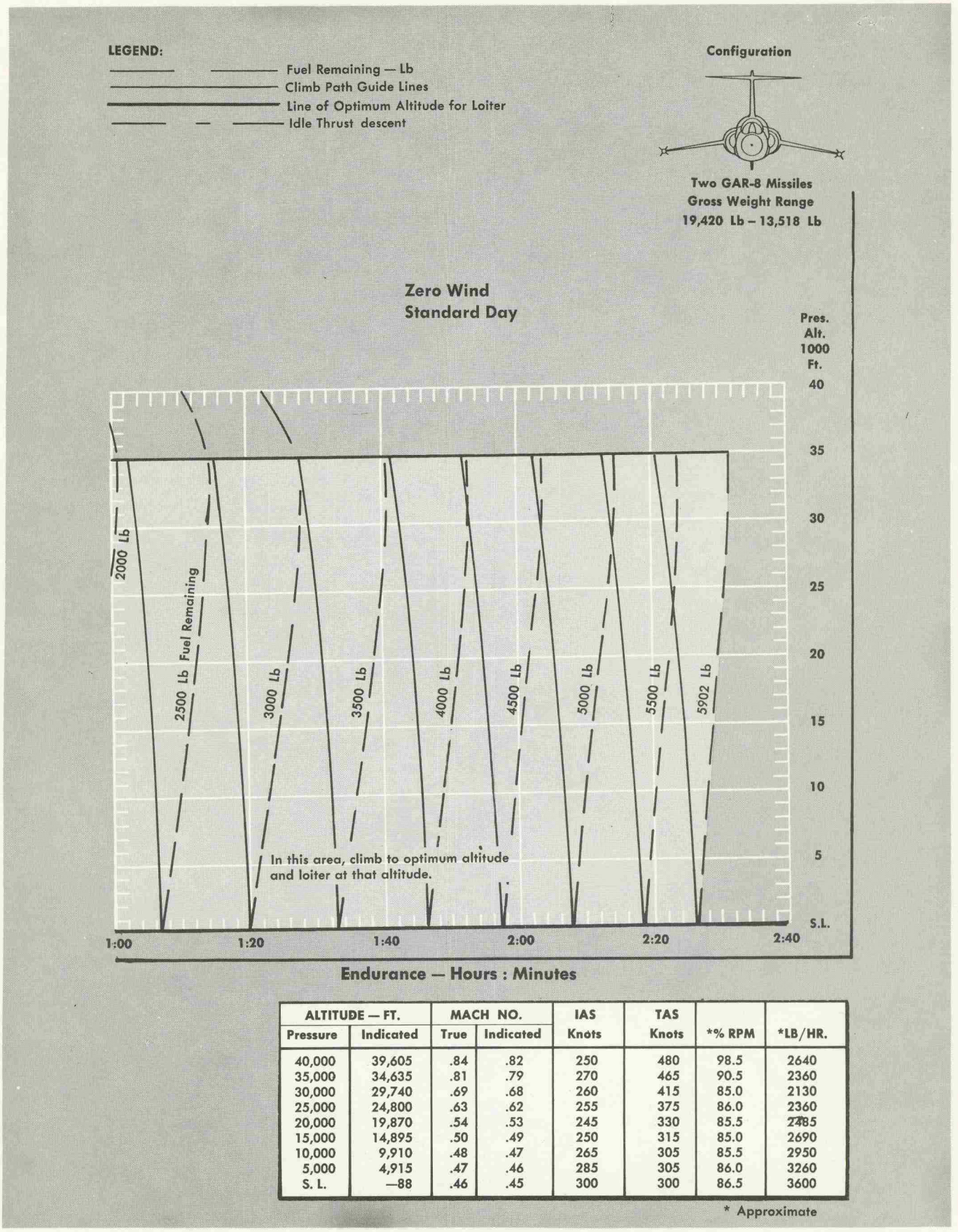
True Mach No.	Ind. Mach No.	IAS Knots	Pres. Alt. 1000 Ft.
.88	.86	265	40
.88	.86	296	35
.88	.86	332	30
.88	.86	368	25
.88	.86	407	20
.88	.86	446	15
.88	.86	487	10
.88	.86	529	5
.88	.86	579	S.L.



NOTE:

1. Fuel required at any point includes Military Thrust Climb, if required, to altitude, loiter and Idle Thrust descent to Sea Level. (See Military Thrust Climb Chart for detailed information.)
2. Loiter at recommended Mach Number.
3. Recommended descent at Idle Thrust and 300 Knots IAS. (See Descent Chart for detailed information.)
4. No allowance or reserve made for landing.

Figure A5-4 (Sheet 1)



ALTITUDE — FT.		MACH NO.		IAS	TAS		
Pressure	Indicated	True	Indicated	Knots	Knots	*% RPM	*LB/HR.
40,000	39,605	.84	.82	250	480	98.5	2640
35,000	34,635	.81	.79	270	465	90.5	2360
30,000	29,740	.69	.68	260	415	85.0	2130
25,000	24,800	.63	.62	255	375	86.0	2360
20,000	19,870	.54	.53	245	330	85.5	2485
15,000	14,895	.50	.49	250	315	85.0	2690
10,000	9,910	.48	.47	265	305	85.5	2950
5,000	4,915	.47	.46	285	305	86.0	3260
S. L.	—88	.46	.45	300	300	86.5	3600

* Approximate

Figure A5-4 (Sheet 2)

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Part 6—Air Refueling

Not Applicable

Part 7—Descent

TABLE OF CONTENTS

Page numbers in bold face denote illustration

Title	*Page*
Descent Performance Presented	A7-1
Use of Descent Chart	A7-1
Descent Performance—300 Knots I.A.S.	**A7-2**

DESCENT

DESCENT PERFORMANCE PRESENTED.

Time, distance, and fuel required to descend from altitude with idle RPM are presented for operation without external stores. The performance presented is based on a constant 300 K.I.A.S. descent with speed brakes, wing flaps, and gear retracted. Inasmuch as the chart is for the no external stores configuration, it may be used as a conservative estimate for the aircraft equipped with GAR-8 Missiles, 165 gallon tip tank, 200 gallon pylon tanks or any combination of these stores. For any of these external store configurations and at a descent speed of 300 K.I.A.S., the fuel used, time and distance traveled will be less than the data presented in Figure A7-1. Refer to Section IX for jet penetration descent procedures and performance.

USE OF THE DESCENT CHART.

The charts may be read directly if sea level is the terminal altitude; otherwise, read them in an incremental basis between the initial and final altitudes. A sample problem is provided to show the use of the descent charts.

SAMPLE PROBLEM.

For an aircraft with GAR-8 Misiles installed, find time, fuel used, and distance travelled in a descent from 35,000 feet to Sea Level.

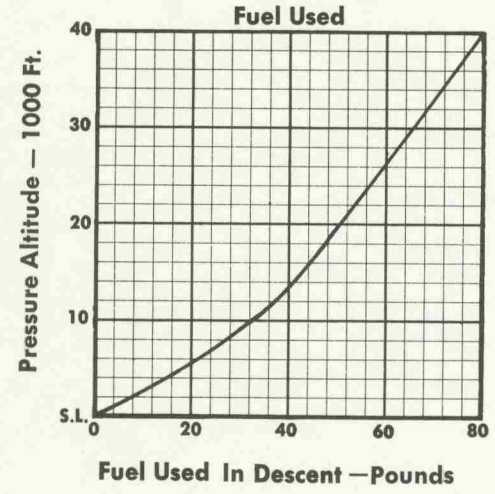
Enter Figure A7-1 at 35,000 ft. and read 73 lbs. of fuel used, 6.3 minutes, and 41.8 Nautical Miles traveled in descent. As stated previously, these values would be less for operation with external stores or with speed brakes extended.

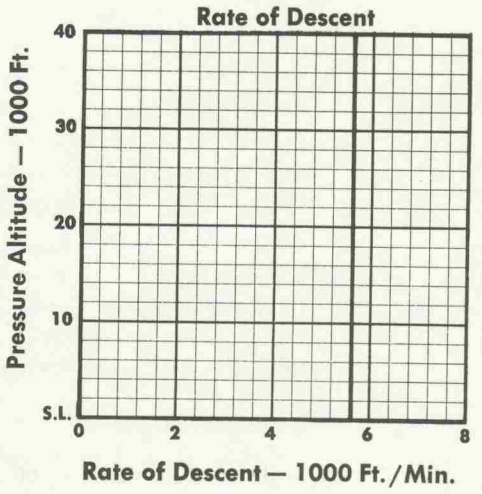
IDLE DESCENT PERFORMANCE — 300 KNOTS IAS

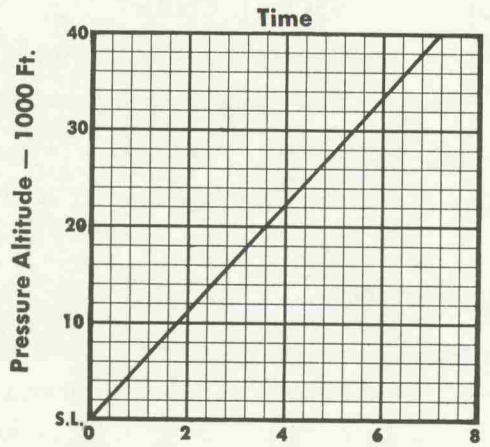
No External Stores

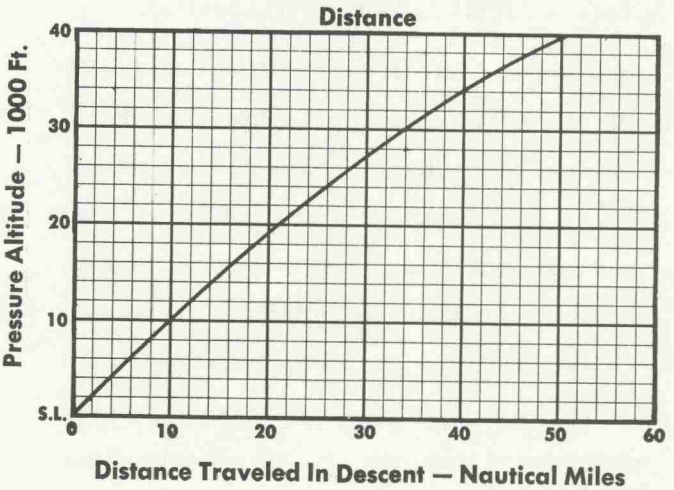
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal









Remarks:

Set Idle Thrust with wing flaps and speed brakes retracted.

Hold 300 knots IAS during descents from altitudes above 40,000 ft.

Refer to Section IX for 85% RPM jet penetration descent performance.

Data based on 14,400 pound gross weight.

Figure A7-1

Part 8 — Landing

TABLE OF CONTENTS

Page numbers in bold face denote illustration

Title	*Page*
Performance Information Presented	A8-1
Data Basis	A8-1
Normal Performance	A8-1
Use of Charts	A8-1
Effect of Wind	A8-1
Landing With Drag Chute	A8-2
Rate of Climb Available for Go-Around	A8-2
Landing With Take-Off Flaps	A8-2
Effect of Wet Surface	A8-2
Effect of Slope	A8-4
Effect of Airspeed	A8-4
Wind Component Chart	**A8-5**
Landing Distance With Drag Chute	**A8-6**
Landing Distance Without Drag Chute	**A8-7**
Available Rate of Climb for Go-Around, Military Thrust	**A8-8**

PERFORMANCE INFORMATION PRESENTED.

Landing ground roll distances and distances required to clear a fifty foot obstacle, land, and stop, are presented for operation with and without drag chute on figures A8-2 and A8-3. Values shown are directly applicable with hard, dry runway surface conditions and for wind component velocities from twenty knot tailwinds to forty knot headwinds. Correction factors are supplied which allow prediction of the effects of runway surface condition, variations from normal speed schedules, runway slopes, and operation with take-off flaps. Additional charts supply a means for determining runway headwind and crosswind velocity components and for predicting rate of climb available at the final approach speed with Military Thrust and landing flaps and gear extended.

DATA BASIS.

Landing distances included in this Part reflect results of flight tests using normal speed schedules for approach and touchdown, and hard wheel braking for the subsequent stop.

NORMAL PERFORMANCE.

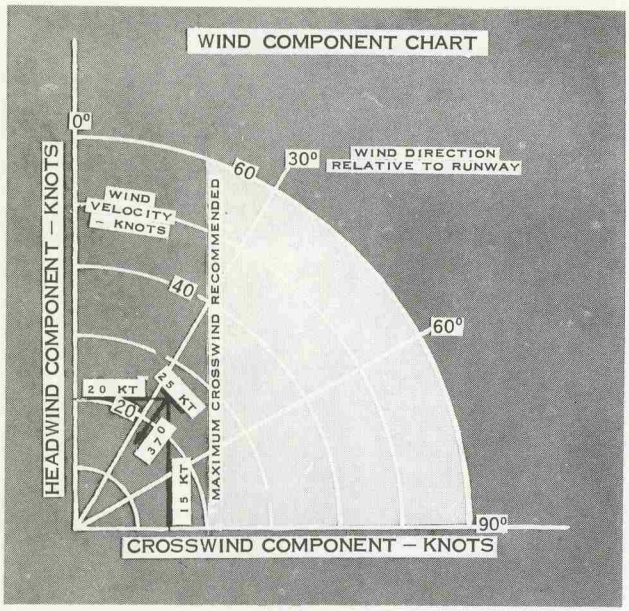
Normal performance used as a basis for landing distance data assumes that the approach flight path slope is approximately $-2\frac{1}{2}°$ at the fifty foot obstacle point, and that airspeed is decreased smoothly according to the recommended schedule during the flare to touchdown. Stopping distances are based on hard wheel braking without skidding.

USE OF CHARTS.

Sample problems are included with illustrative charts to show the use of the landing data charts. The illustrative charts are samples only, and should not be used in flight planning.

EFFECT OF WIND.

Correction grids included on the landing distance charts show the effect of wind component along the runway. Actual surface wind velocities should be used. Wind speeds indicated by an anemometer located considerably above the surface may be corrected for wind gradient by



WIND COMPONENT CHART

SAMPLE PROBLEM.

Determine the headwind and crosswind components for a 25 knot wind velocity blowing 37° relative to the service runway (wind velocity obtained from an anemometer located adjacent to the runway).

1. From Figure A8-1 (Wind Component Chart) determine the cross wind component to be . . 15 knots.

(This is less than the maximum recommended component of 20 knots.)

2. If landings are to be made into the wind, the headwind component is . . . . . 20 knots.

multiplying reported winds by a factor of 7/10. Landings with crosswind components at the runway in excess of 20 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 drag chute deployment.

LANDING WITH DRAG CHUTE.

Use of the drag chute allows a high rate of deceleration to be developed during the early portion of the ground roll. However, any unnecessary delay in its deployment will add considerably to the stopping distance. The performance chart for landings with drag chute is based on making a smooth rotation from touchdown attitude to the nosewheel contact point in about 2¼ seconds, prompt activation of the drag chute with moderate wheel braking during the transition period of about 2½ seconds, and continuous hard wheel braking with drag chute deployed during the remainder of the stop. The effect of a slick runway is not noticeable in the reduction of wheel braking force available during the last half of the stop.

RATE OF CLIMB AVAILABLE FOR GO-AROUND.

Rates of climb available at final approach speeds are given on figure A8-4 for operation with Military Thrust. The data are directly applicable to the wing-tip stores and no external stores configurations. Rates of climb with pylon stores will be slightly less at a given weight because of their additional drag. Flight characteristics and the effect of speed and configuration changes are described in Section VI.

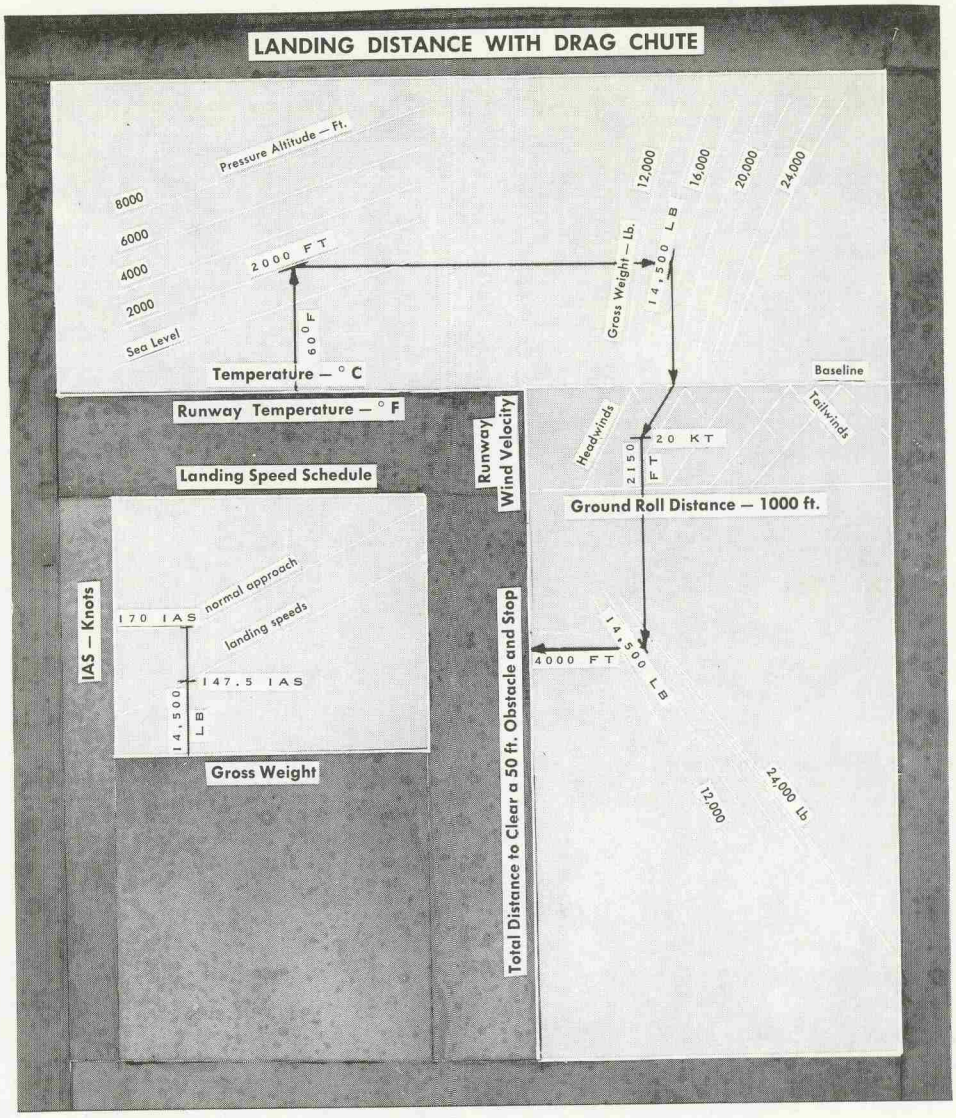
LANDING WITH TAKE-OFF FLAPS.

Performance during landings with take-off flaps can be determined by applying the following factors to data provided for operation with landing flaps.

Distance and Speed Corrections: Take-Off Flaps	Total Distance From 50′ Obstacle		Ground Roll Distance	
	With Chute	No Chute	With Chute	No Chute
Multiply normal landing distance by	1.19	1.13	1.31	1.21
With normal landing speed increased by	20 KIAS at 50′	20 KIAS at 50′	15 KIAS at contact	15 KIAS at contact

EFFECT OF WET SURFACE.

Apply the following factors to normal landing ground distances to determine zero wind ground roll distances on wet runways. With drag chute, using normal landing speed schedules, apply a factor of 1.41 at 14,000 pounds landing weight. Increase the factor by 0.02 for each 1000 pound weight increase. Apply a factor of 1.90 to the ground roll distances shown on figure A8-3 for operation without drag chute.



LANDING DISTANCE

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—14,500 pounds
Runway Air Temperature—60°F.
Field Pressure Altitude—2000 feet
Runway Wind Component—20 knot headwind
Runway Slope—1% uphill
Drag Chute—deployed
Runway Surface—dry and hard

1. From Figure A8-2 (Landing Distance With Drag Chute) the ground roll distance from touchdown to stop for a level runway with a 20 knot headwind is . . . . . 2150 ft.

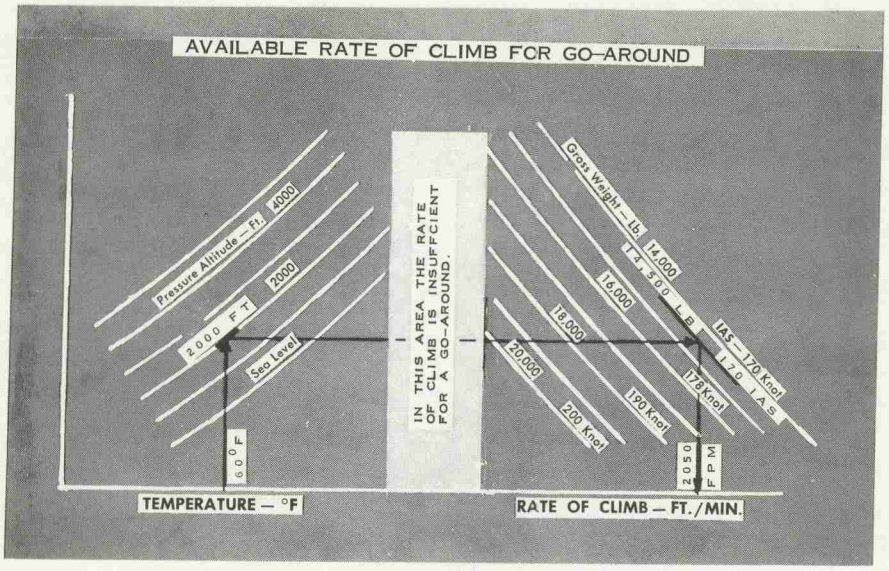
2. The ground roll distance with a 1% uphill slope is determined by decreasing the zero slope value by 3%. (2150 ft. − .03 × 2150 ft.) . . 2085 ft.

3. Touchdown speed is . . . . . 147.5 kts IAS

4. Total distance to clear a 50 foot obstacle and stop, for a level runway, is . . . . . 4000 ft.

5. Total distance with a 1% uphill slope is determined by decreasing the zero slope total distance by 6%. (4000 ft. − .06 × 4000 ft.) . . . . . 3760 ft.

6. Airspeed at the 50 ft. height point is 170 knots IAS.



GO-AROUND RATE OF CLIMB AVAILABLE

SAMPLE PROBLEM.

Determine the rate of climb available for go-around with Military Thrust, landing configuration, for the following conditions:

Gross Weight—14,500 pounds
Air Temperature—60°F.
Pressure Altitude—2,000 ft.
I.A.S.—170 knots

1. From Figure A8-4 (Rate of Climb Available for Go-Around, Military Thrust) determine the available rate of climb to be . . . . . 2,050 fpm.

EFFECT OF RUNWAY SLOPE.

With or without drag chute, ground roll distances should be increased 3% per 1% downhill runway slope, decreased 3% per 1% uphill slope. Total landing distance should be increased 9% per 1% downhill slope, decreased 6% per 1% uphill slope.

EFFECT OF AIRSPEED.

Increase ground roll distances 7½% per 10 knot increase over normal touchdown airspeed.

WIND COMPONENT CHART

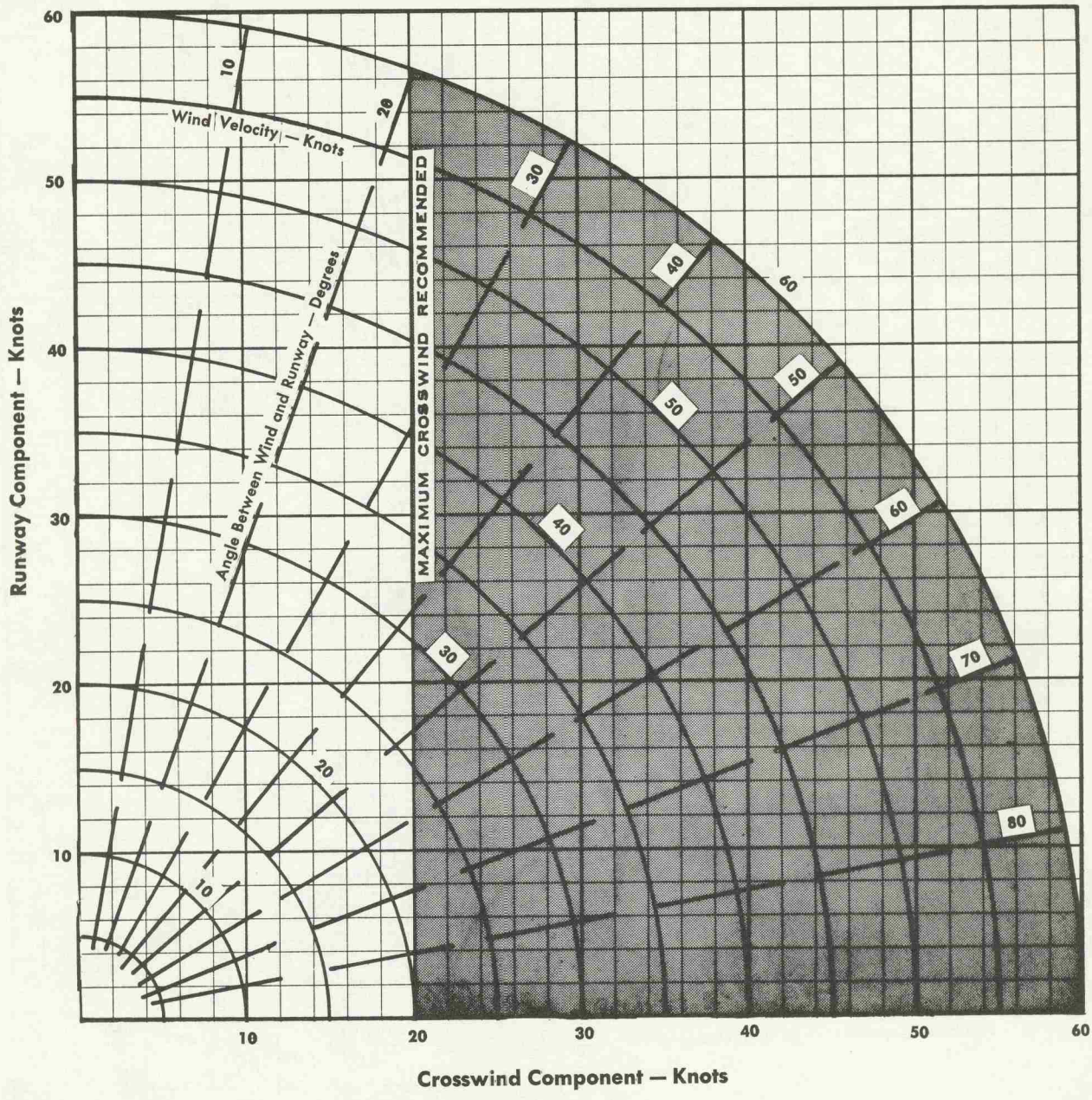


Figure A8-1

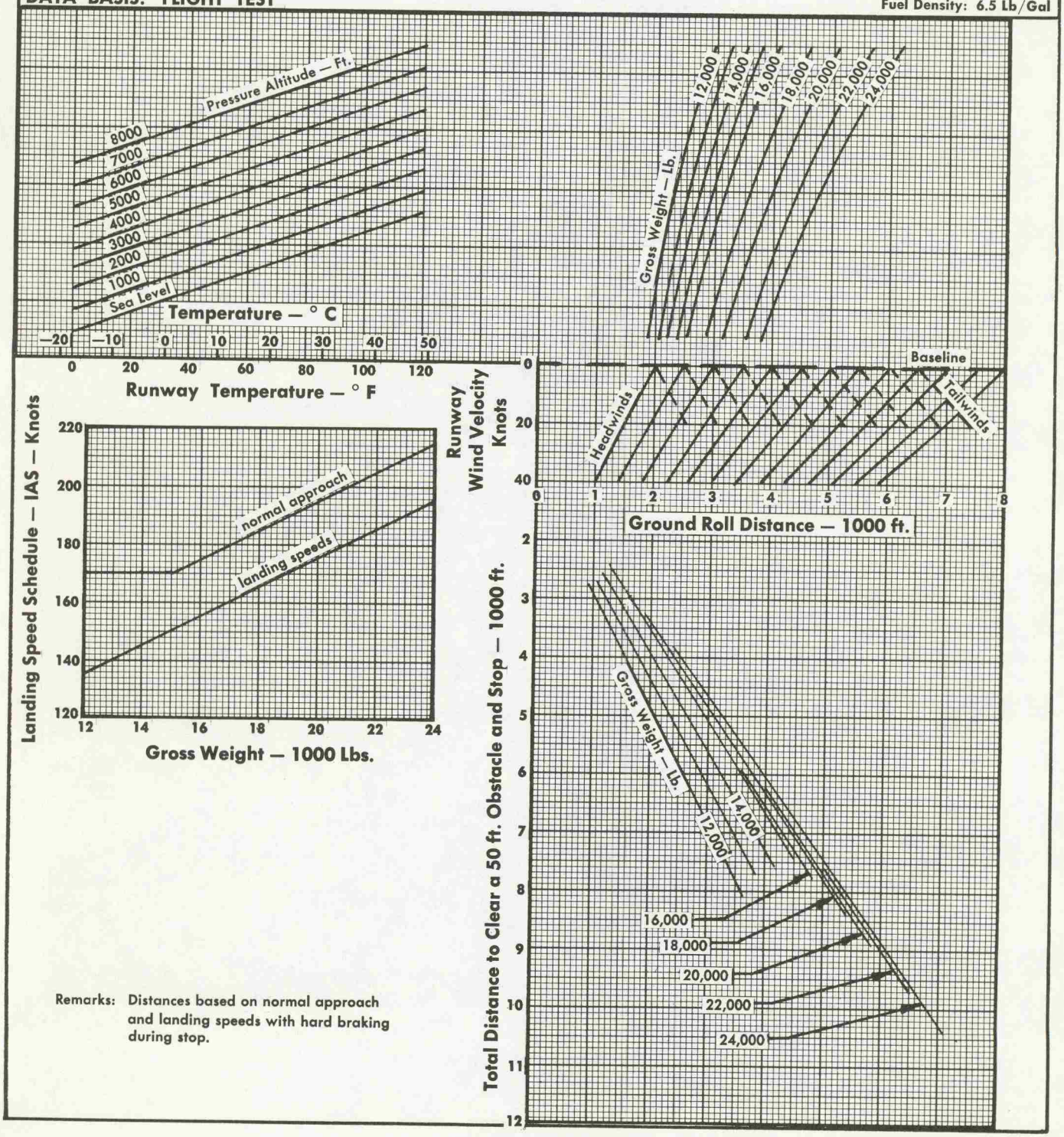
LANDING DISTANCE WITH DRAG CHUTE
BOUNDARY LAYER CONTROL ON

All Wing Stores Configurations
Dry, Hard Surface Runway

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

LANDING FLAPS

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



Remarks: Distances based on normal approach and landing speeds with hard braking during stop.

Figure A8-2

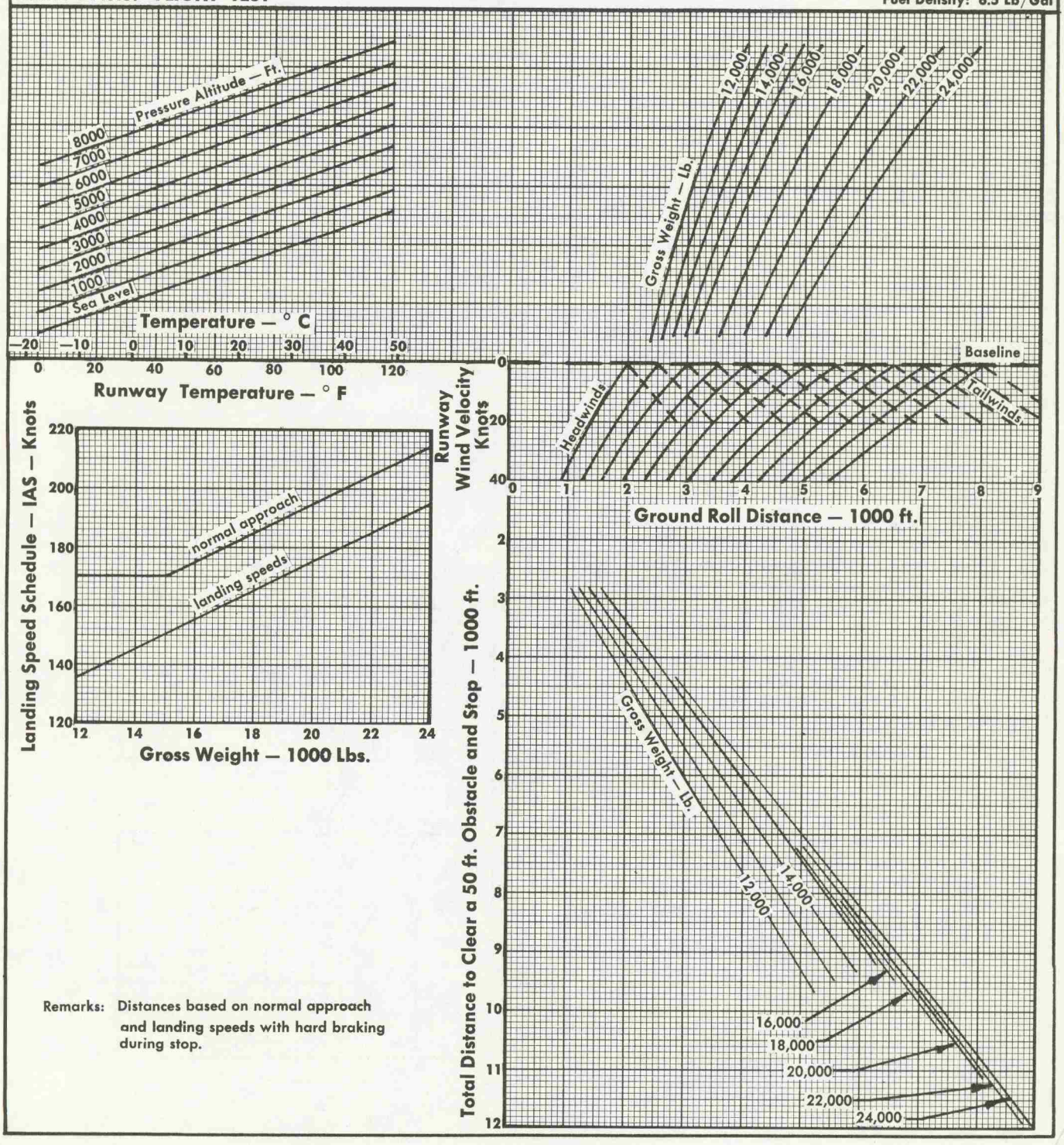
LANDING DISTANCE WITHOUT DRAG CHUTE
BOUNDARY LAYER CONTROL ON

All Wing Stores Configurations
Dry, Hard Surface Runway

Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST

LANDING FLAPS

Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal



Remarks: Distances based on normal approach and landing speeds with hard braking during stop.

Figure A8-3

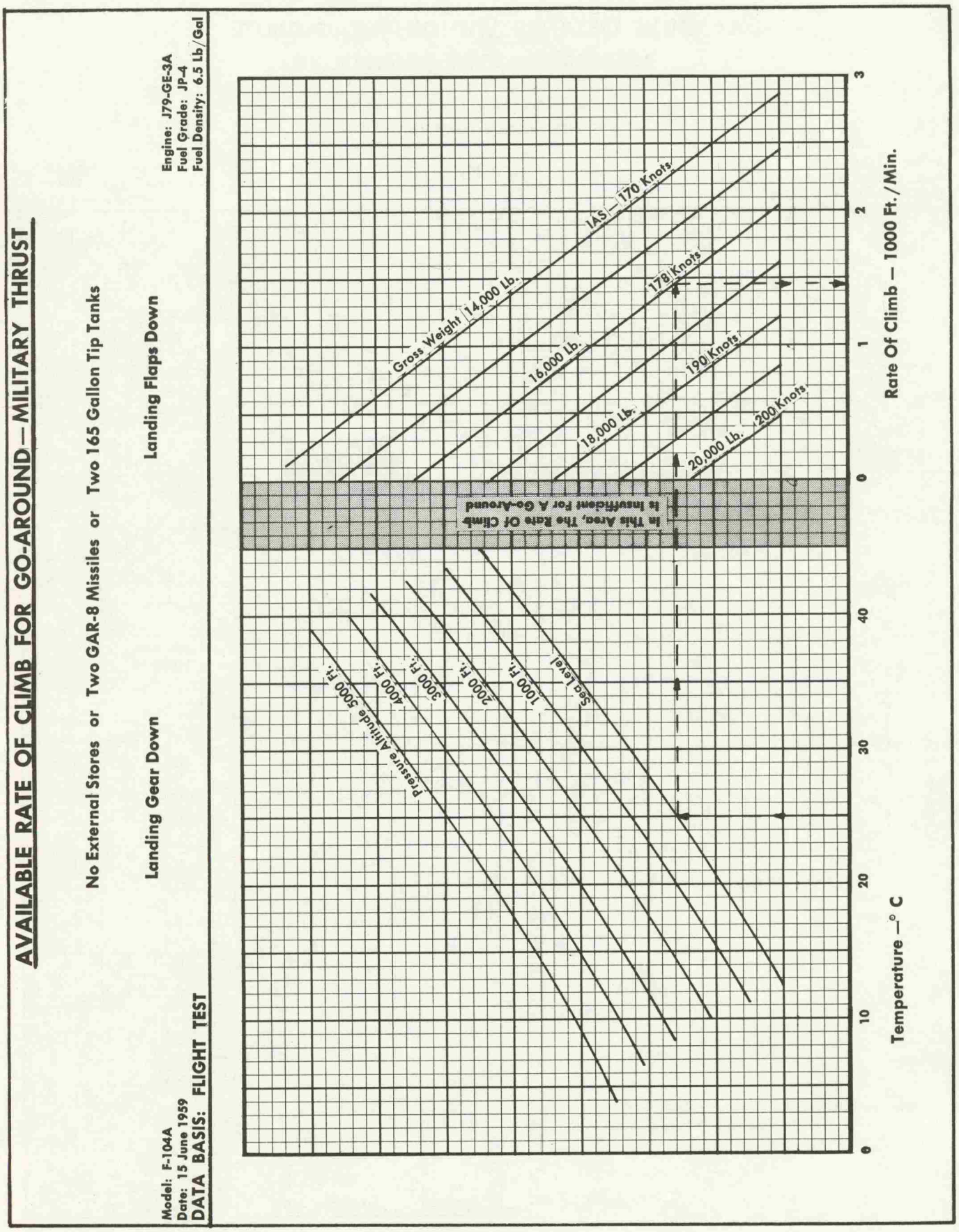

AVAILABLE RATE OF CLIMB FOR GO-AROUND—MILITARY THRUST
No External Stores or Two GAR-8 Missiles or Two 165 Gallon Tip Tanks
Landing Gear Down
Landing Flaps Down
Engine: J79-GE-3A
Fuel Grade: JP-4
Fuel Density: 6.5 Lb/Gal
Model: F-104A
Date: 15 June 1959
DATA BASIS: FLIGHT TEST
Gross Weight 14,000 Lb.
16,000 Lb.
18,000 Lb.
20,000 Lb.
IAS — 170 Knots
178 Knots
190 Knots
200 Knots
In This Area, The Rate Of Climb Is Insufficient For A Go-Around
Sea Level
1000 Ft.
2000 Ft.
3000 Ft.
4000 Ft.
Pressure Altitude 5000 Ft.
Temperature —° C
Rate Of Climb — 1000 Ft./Min.
0
10
20
30
40
0
1
2
3

Figure A8-4

Part 9—Combat Performance

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SAMPLE PROBLEM.

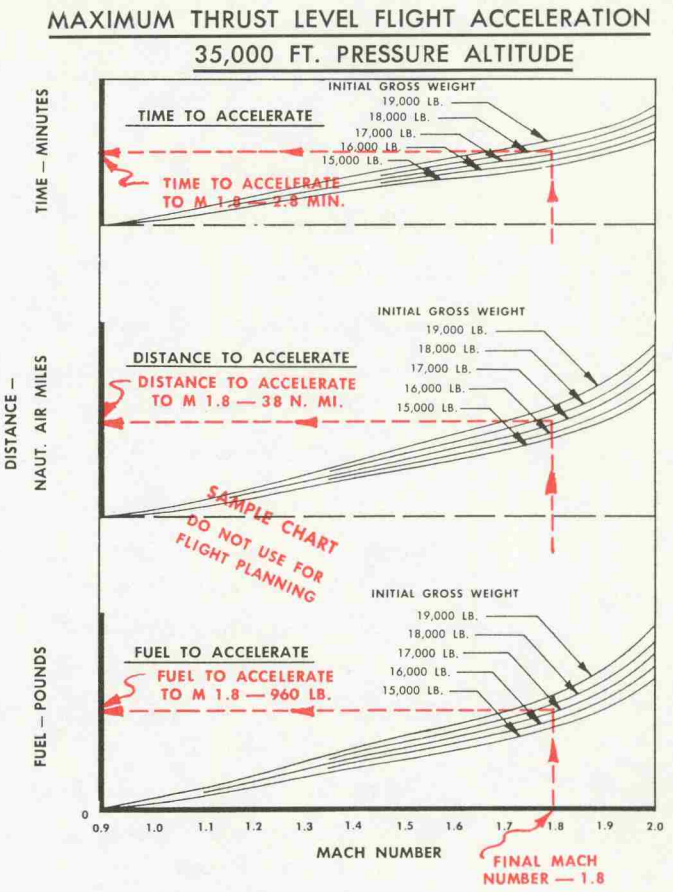
Determine the time, distance, and fuel required to accelerate from Mach 0.9 to Mach 1.8 at 35,000 feet for operation with two GAR-8 missiles. Assume an initial gross weight of 17,000 pounds.

1. Enter Figure A9-4 (Maximum Thrust Level Flight Acceleration—35,000 Feet Pressure Altitude) at a final Mach number of 1.8 and at each 17,000 pound gross weight line, read:

Time to accelerate . . . . . 2.8 min.

Distance to accelerate . . . . . 38 n.mi.

Fuel to accelerate . . . . . 960 lb.



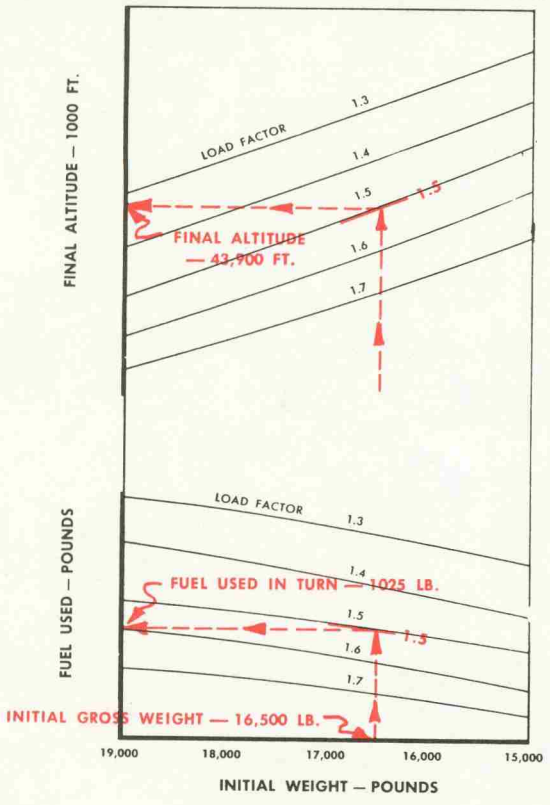
LEVEL FLIGHT ACCELERATION.

Maximum thrust level flight acceleration performance for the aircraft is provided on Figures A9-1 to A9-8. The time, distance, and fuel required to accelerate from an initial cruise speed of Mach 0.9 true (Mach 0.88 indicated) are plotted as functions of the final desired Mach number and initial operating weight. Data are shown for operation without external stores or with two GAR-8 missiles installed at pressure altitudes of 35,000 feet, 40,000 feet, and 45,000 feet. Acceleration data for operation with two 165 gallon tip tanks, or two 200 gallon pylon tanks and two GAR-8 missiles are provided at 35,000 feet pressure altitude only. Maximum acceleration performance occurs at approximately 35,000 feet pressure altitude for all configurations and results in minimum time and fuel required to reach any desired final Mach number. The sample problem and sample chart illustrate the use of the acceleration charts.

CLIMBING TURN PERFORMANCE.

The basic attack procedure for this aircraft, acknowledging the requirements of the GAR-8 missile and the aircraft fire control system, requires a tail attack on the target. Therefore, the aircraft may be required to make a turn of up to 180 degrees to complete the intercept. Appreciable amounts of time and fuel may be required for the maneuver and these items, plus radius of turn and altitude gained, should be considered in overall mission planning. This turn is best accomplished at supersonic speeds and while climbing. Speed and load factor should be maintained constant throughout the maneuver. A speed of Mach 1.9 is found to be optimum in attacks against targets near 65,000 feet. Climbing turn performance based on Mach 1.9 and also Mach 1.8 is presented on Figures A9-9 through A9-15. The data presented are for climbing turns starting at the optimum level flight acceleration altitude of 35,000 feet. The final altitude and fuel used are given as a function of initial gross weight and load factor. Turn radius and time required for the maneuver are also given. Data for 180° turns are provided for the no external stores, two GAR-8 missiles, and two 165 gallon tip tanks configurations. 90° turn performance is also given for the two tip GAR-8 missiles configuration at Mach 1.9. A sample problem and reference chart illustrate the use of the climbing turn charts.

MAXIMUM THRUST 180° CLIMBING TURN



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	20.9	3.6
1.4	17.7	3.1
1.5	15.5	2.7
1.6	13.9	2.4
1.7	12.6	2.2

SAMPLE PROBLEM.

Determine the turning performance for a Mach 1.9 maximum thrust 180° climbing turn. The aircraft is in the two GAR-8 missiles configuration and the initial gross weight is 16,500 pounds. The load factor is to be held constant at 1.5 G's.

1. Enter Figure A9-10 (Maximum Thrust 180° Climbing Turn Performance From 35,000 Feet) at the initial gross weight of 16,500 pounds and 1.5 load factor. Read:

Fuel used . . . . . 1025 lb.

Final altitude . . . . . 43,900 ft.

Turn radius . . . . . 15.5 n.mi.

Time required . . . . . 2.7 min.

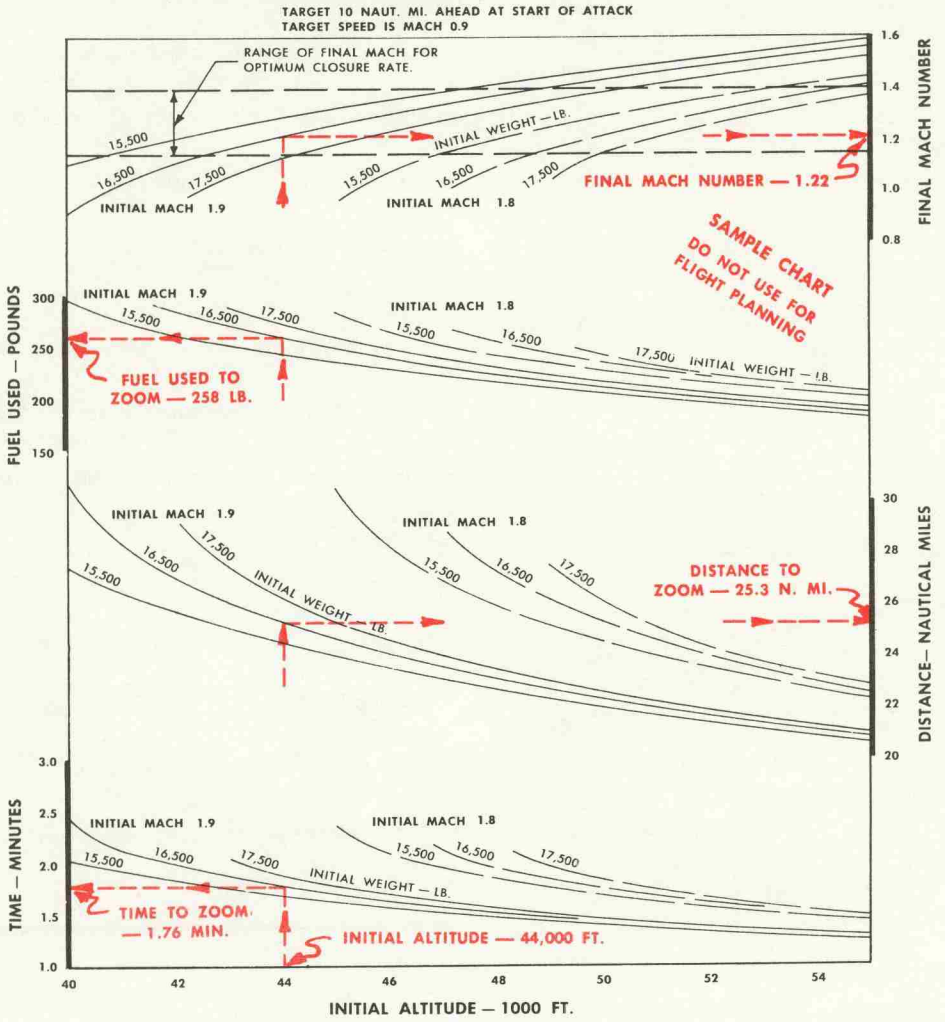
ZOOM PERFORMANCE.

The zoom maneuver is used to attain an altitude above the normal steady state ceiling of the airplane. This maneuver is accomplished by exchanging speed for altitude and requires precise flying technique for maximum performance. The zoom procedure and zoom information described in this Section are not intended for all types of zoom combat. The optimum procedure and resultant performance change for different target speeds and altitudes. However, the planning data provided here are applicable to the specific mission assumed and are illustrated in a form which may be used to present data for other types of combat problems.

ZOOM PROCEDURE.

The zoom information presented on the accompanying charts is based on an attack on a 65,000 foot target cruising at Mach 0.9. The maneuver is started by making a constant load factor pull-up (approximately 1.3 to 1.5 G's) until a boresight is obtained so that the airplane is pointed straight at the target. This boresight is maintained through the remainder of the maneuver, requiring a gentle pushover, averaging approximately 0.8 G, while the aircraft closes on the target. It is assumed that the zoom is initiated from a position 10 nautical miles behind the target, that a wing-tip-mounted GAR-8 missile is fired at a range of five nautical miles and, assuming a miss, that the aircraft continues to close to zero range. During the initial phases of the zoom when the load factor is still greater than 1.0, the climb may be stopped and the aircraft flown in level flight. It will not be possible to maintain a steady speed, however, and the airplane will decelerate to the stall speed unless a descent is initiated.

TIME, DISTANCE, AND FUEL FOR ZOOM ATTACK



SAMPLE PROBLEM.

A zoom attack is to be made on a 65,000 foot, Mach 0.9 target from an initial altitude of 44,000 feet, and initial speed of Mach 1.9. The aircraft has two GAR-8 missiles installed and weighs 16,500 pounds. The target is 10 nautical miles ahead at start of the attack.

1. Enter Figure A9-13 (Time, Distance and Fuel for Maximum Thrust Zoom Attacks on a 65,000 Ft. Target) at the initial altitude of 44,000 feet and read:

Time	1.76 min.
Distance	25.3 n.mi.
Fuel used	258 lb.
Final Mach number	1.22

ZOOM INFORMATION PRESENTED.

The final Mach number, fuel used, distance, and time to close to zero range for zoom attacks on a 65,000 ft. target are presented as a function of initial altitude and initial weight on Figures A9-16, A9-17, and A9-18. Data are given for initial speeds of Mach 1.9 and 1.8. A sample problem and an illustrated sample chart are provided to show further the use of the zoom attack charts.

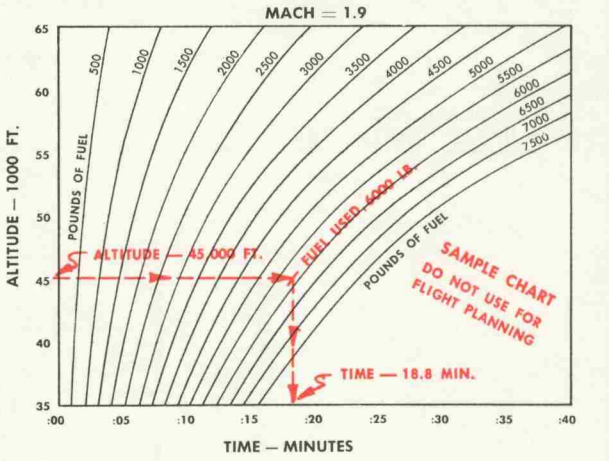
SAMPLE PROBLEM.

Determine the time available for combat at Mach. 1.9 and 45,000 feet if 6000 pounds of fuel are to be used.

1. Enter the Mach 1.9 chart on Figure A9-17 (Maximum Thrust Combat Allowance Chart—Mach 1.9 and 2.0) at 45,000 feet and 6000 pounds of fuel. Read:

Combat time available . . . . . 18.8 min.

MAXIMUM THRUST COMBAT ALLOWANCE CHART



COMBAT ALLOWANCE.

The combat allowance charts show the relationship between combat time and fuel consumption for Maximum Thrust operation at constant speed and altitude. Data are presented for speeds from Mach 1.5 to Mach 2.0 on Figures A9-19, A9-20, and A9-21. The example and sample chart illustrate the use of the combat allowance charts.

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

35,000 FT. PRESSURE ALTITUDE

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

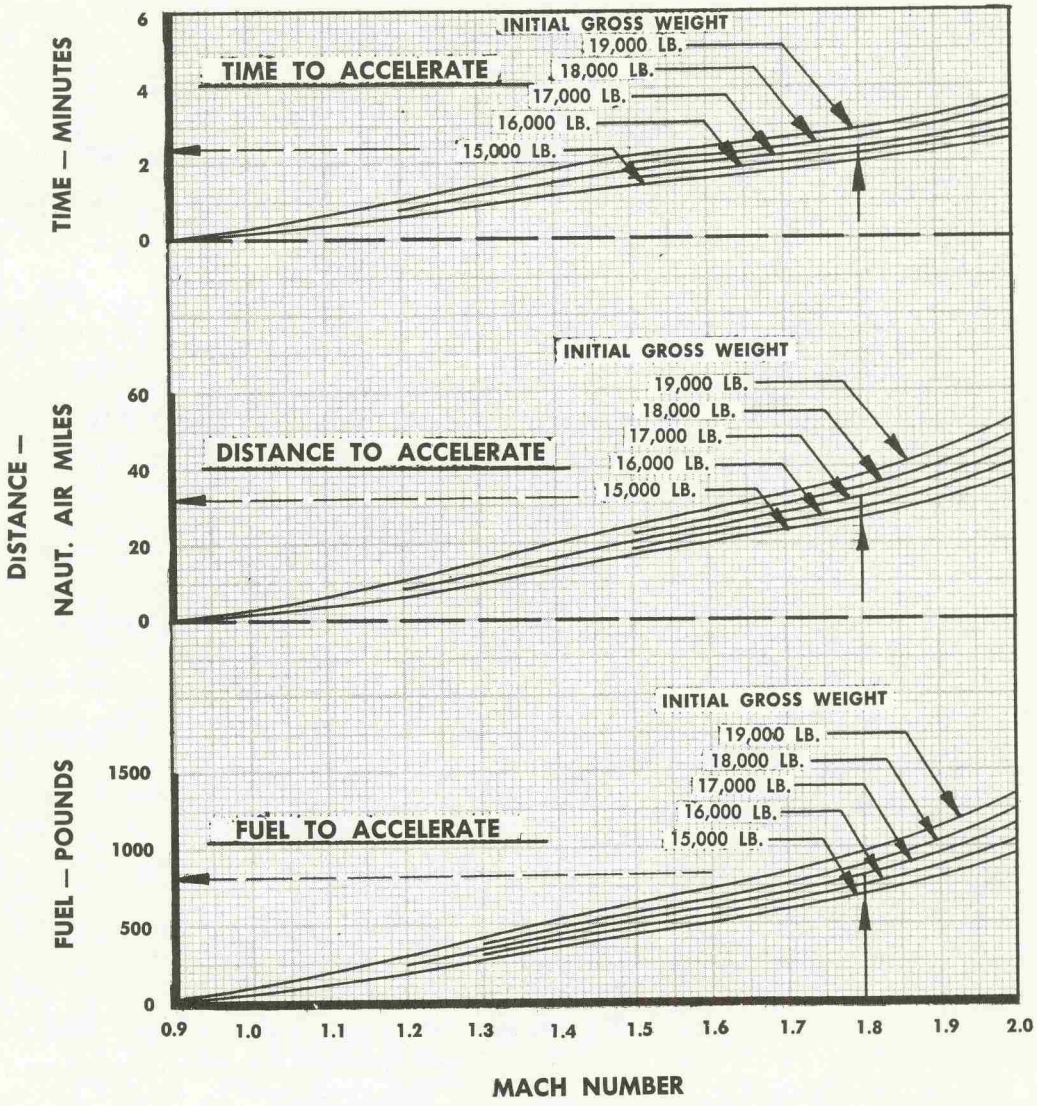


Figure A9-1

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION
40,000 FT. PRESSURE ALTITUDE

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

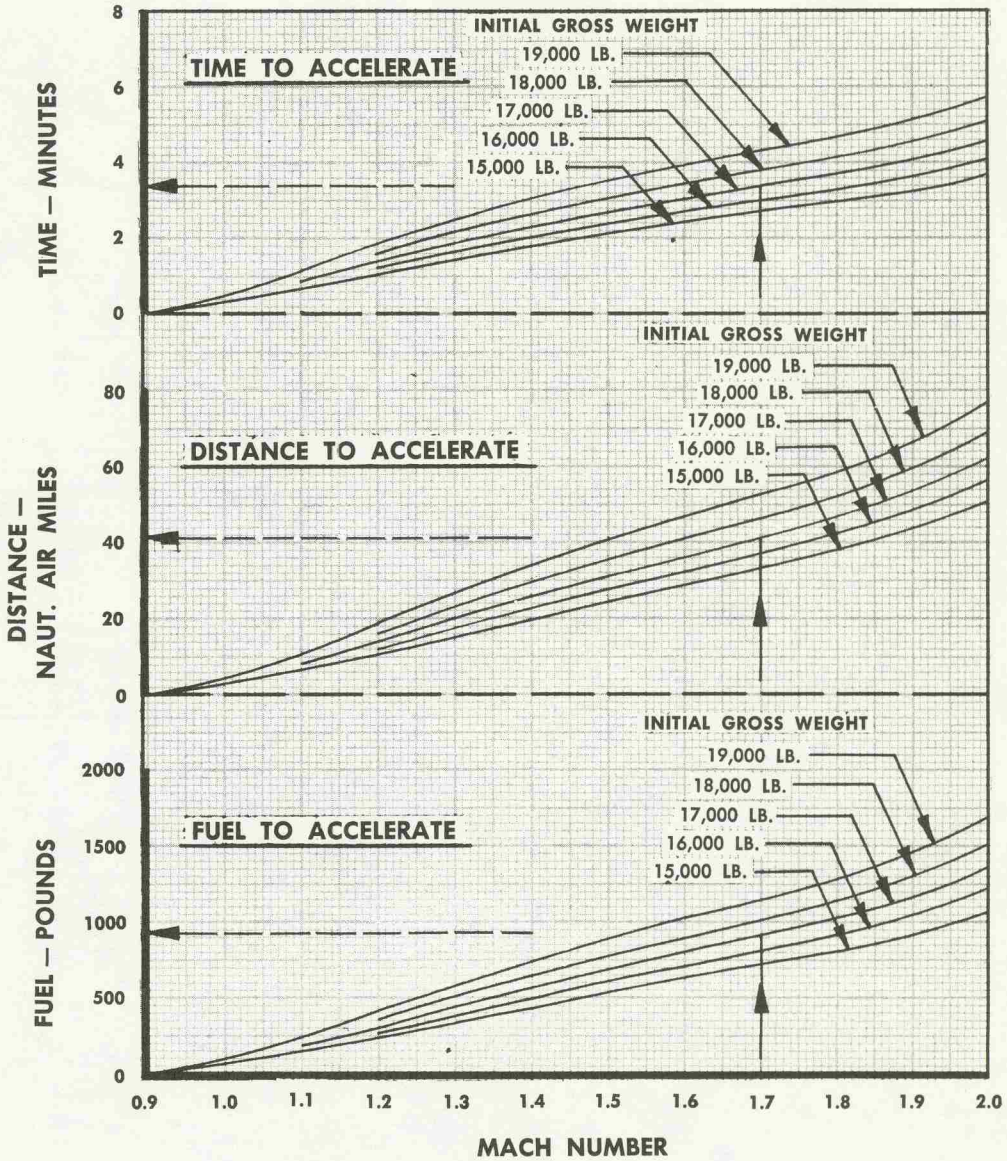


Figure A9-2

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION
45,000 FT. PRESSURE ALTITUDE

NO EXTERNAL STORES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

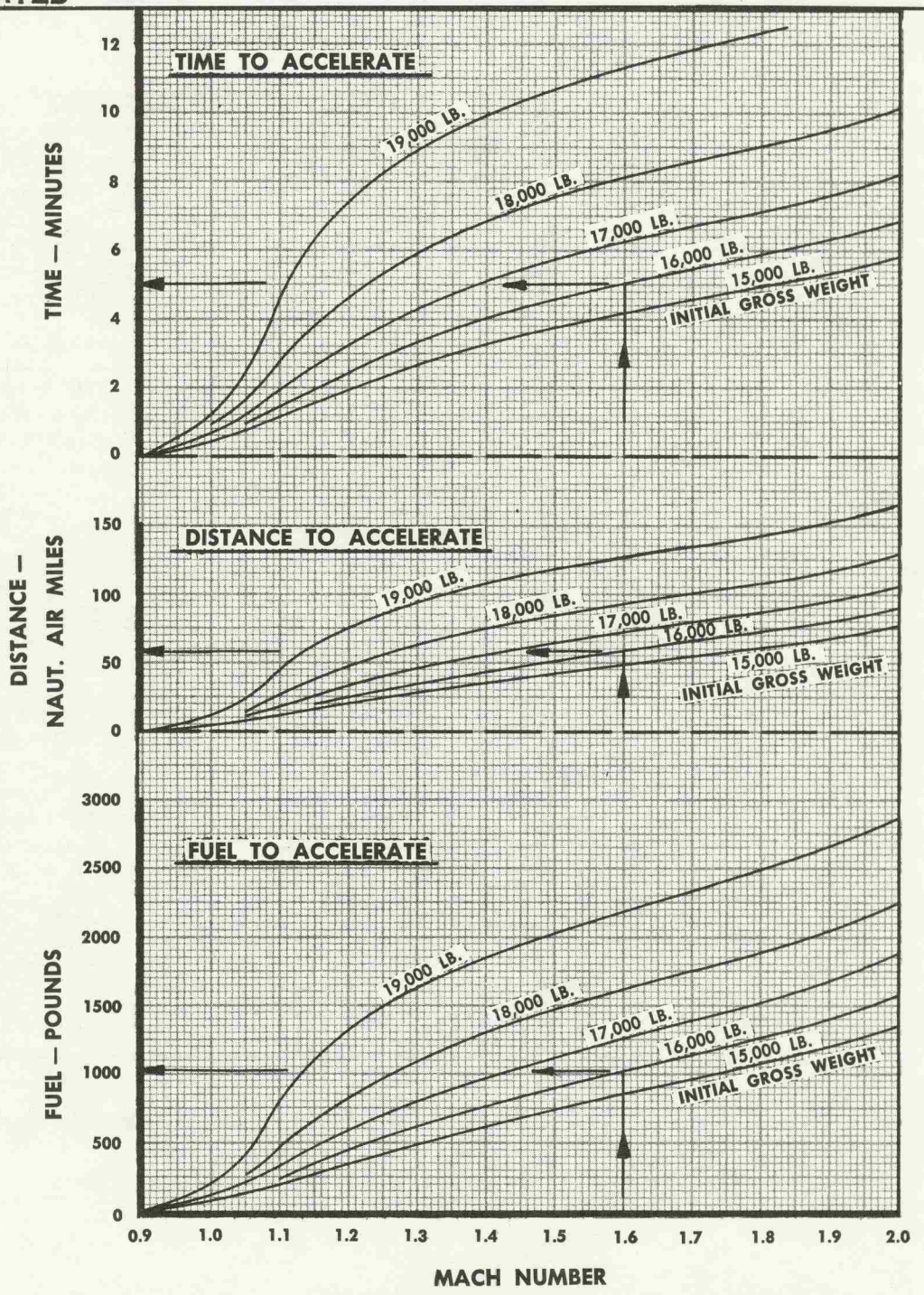


Figure A9-3

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

35,000 FT. PRESSURE ALTITUDE

TWO TIP SIDEWINDER MISSILES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

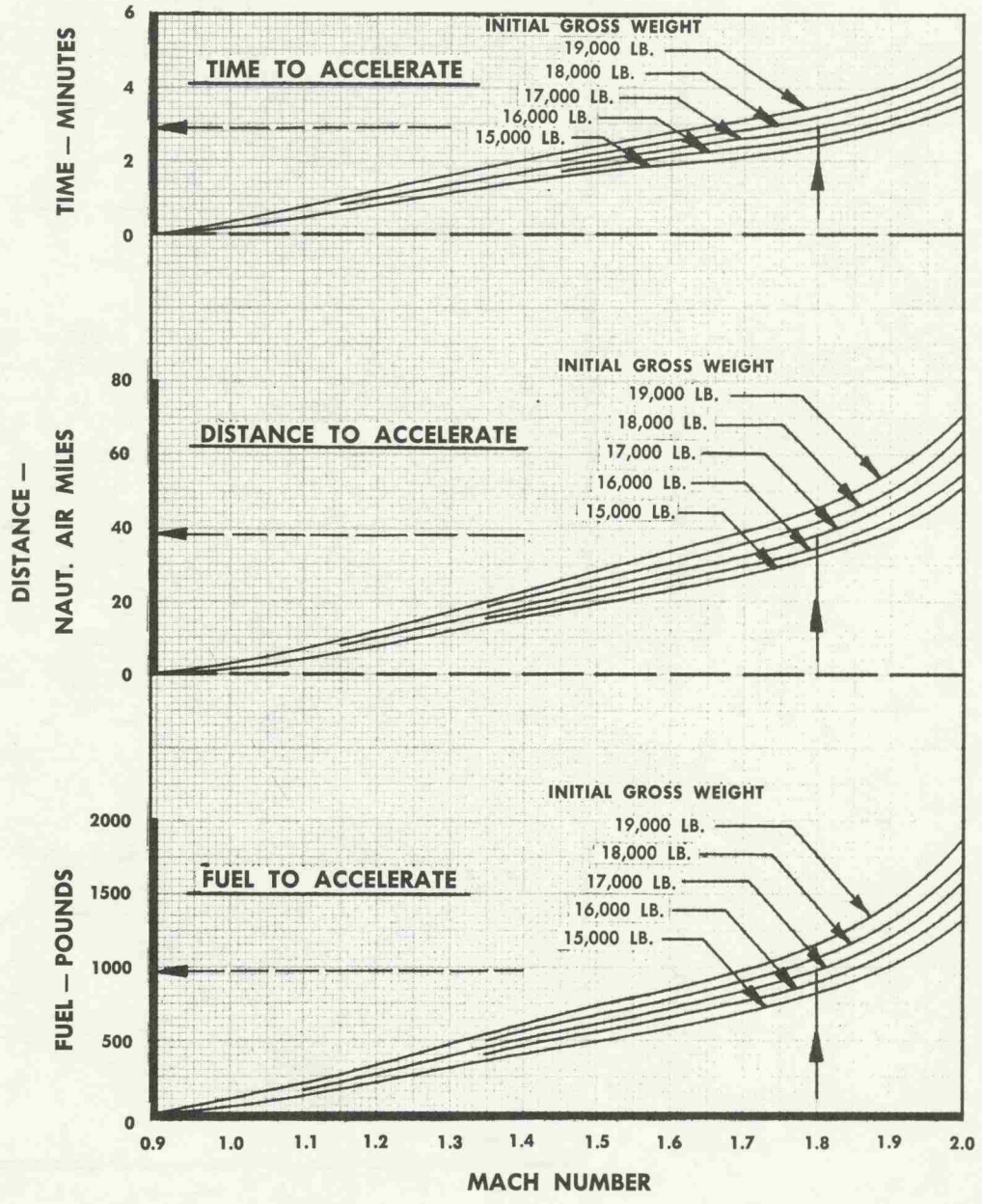


Figure A9-4

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

40,000 FT. PRESSURE ALTITUDE

TWO TIP SIDEWINDER MISSILES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

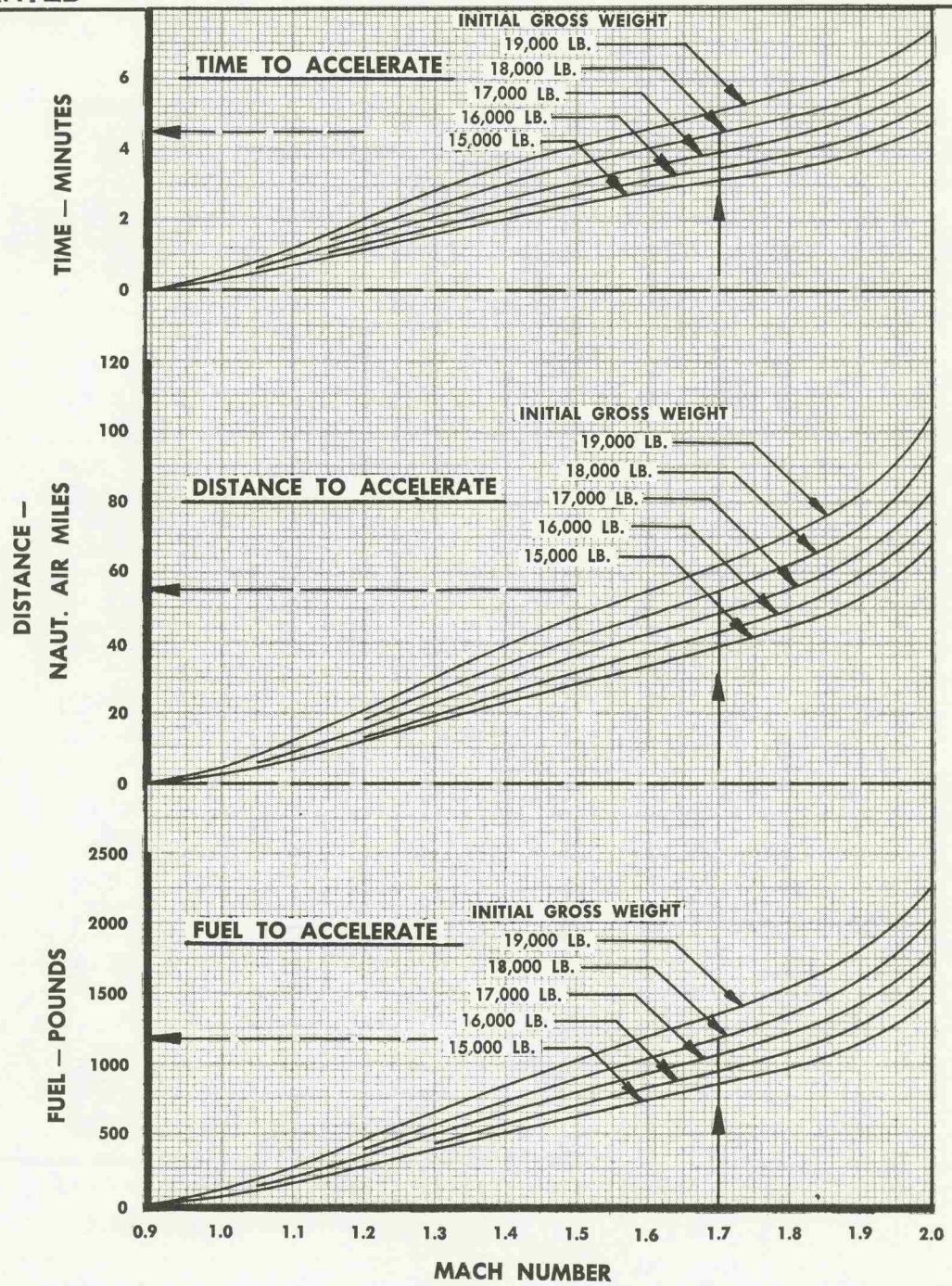


Figure A9-5

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

45,000 FT. PRESSURE ALTITUDE

TWO TIP SIDEWINDER MISSILES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

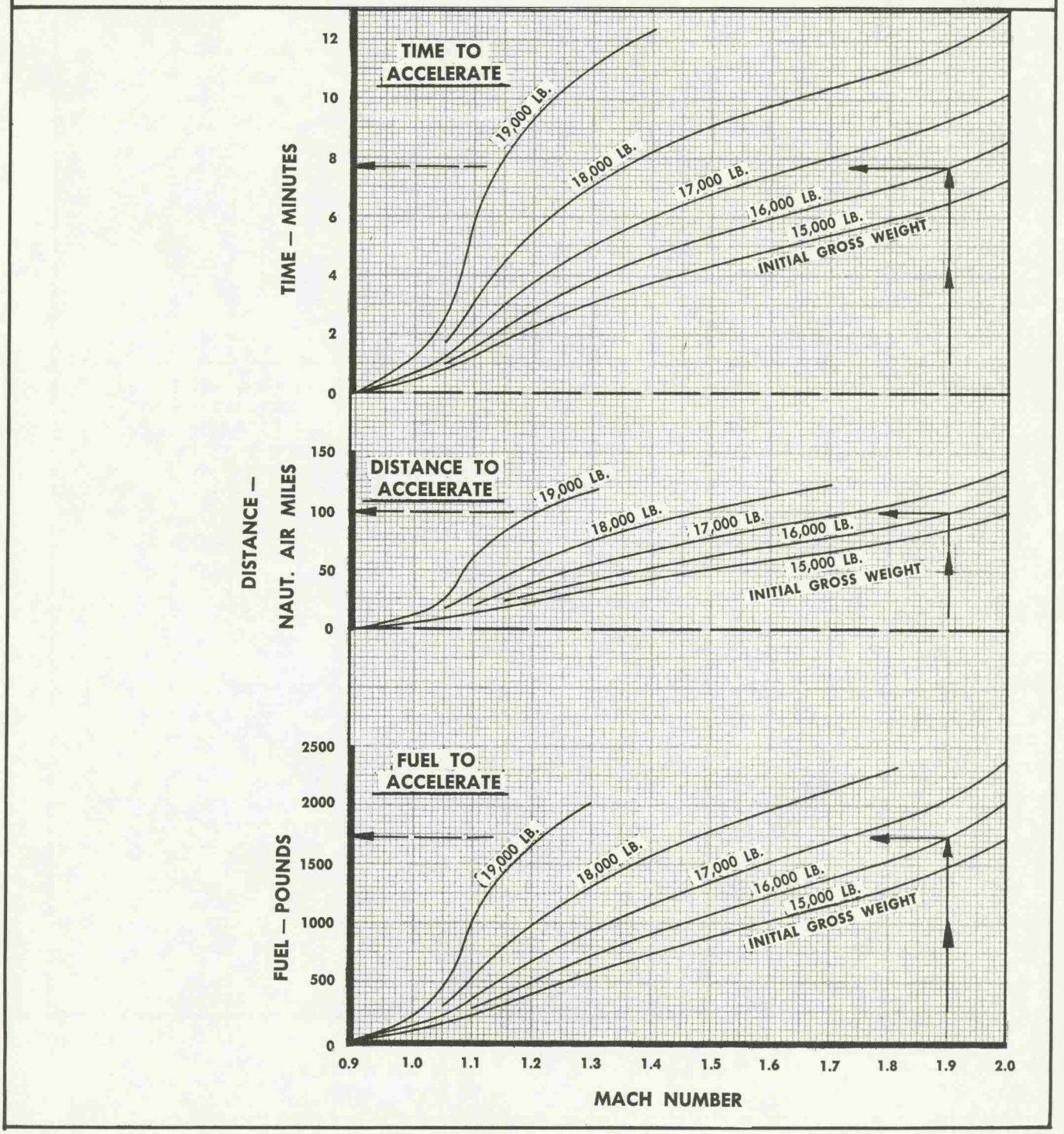


Figure A9-6

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

35,000 FT. PRESSURE ALTITUDE

TWO 165 GALLON TIP TANKS

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

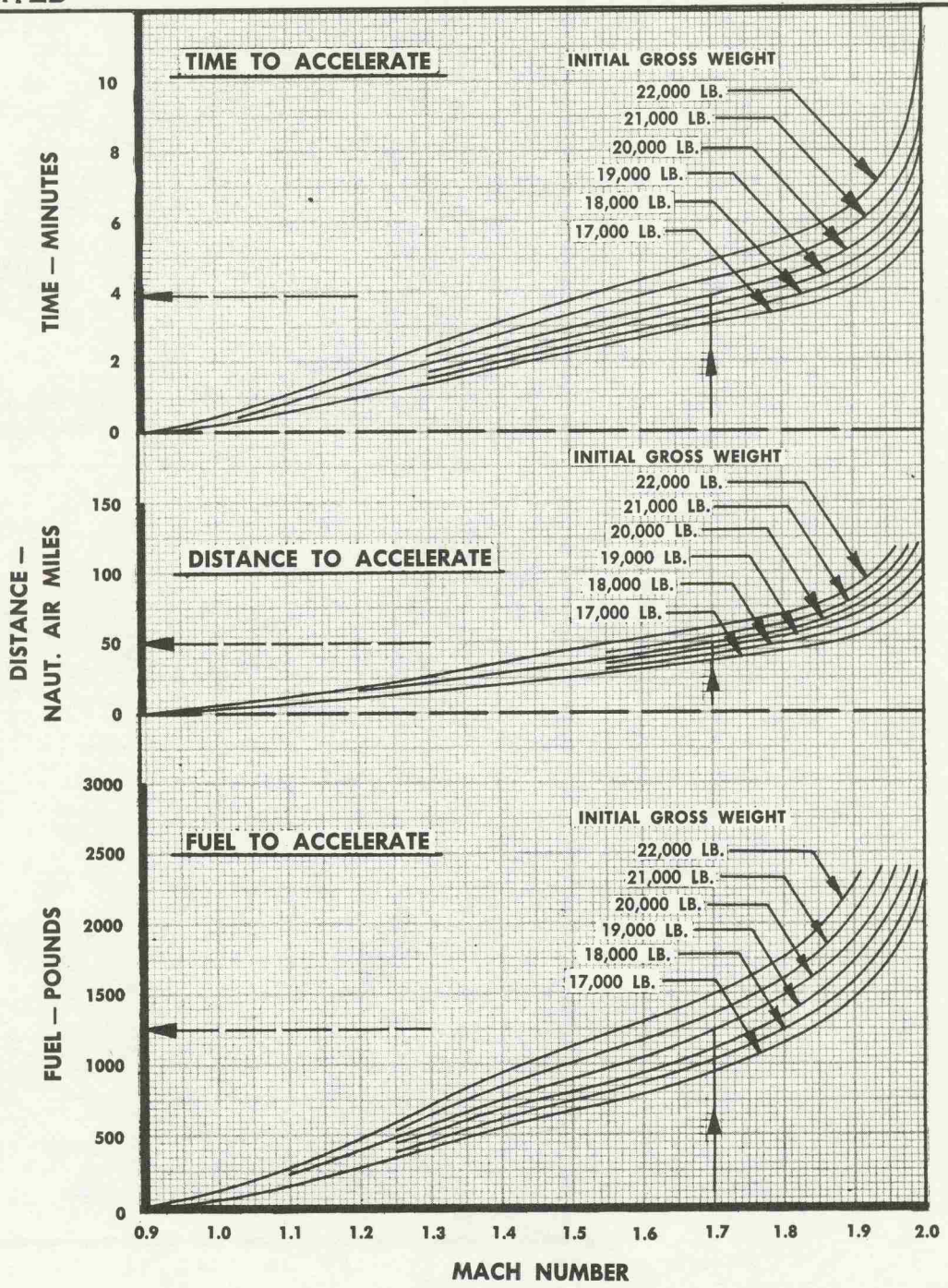


Figure A9-7

MAXIMUM THRUST LEVEL FLIGHT ACCELERATION

35,000 FT. PRESSURE ALTITUDE

TWO 200 GALLON PYLON TANKS AND TWO TIP SIDEWINDER MISSILES

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

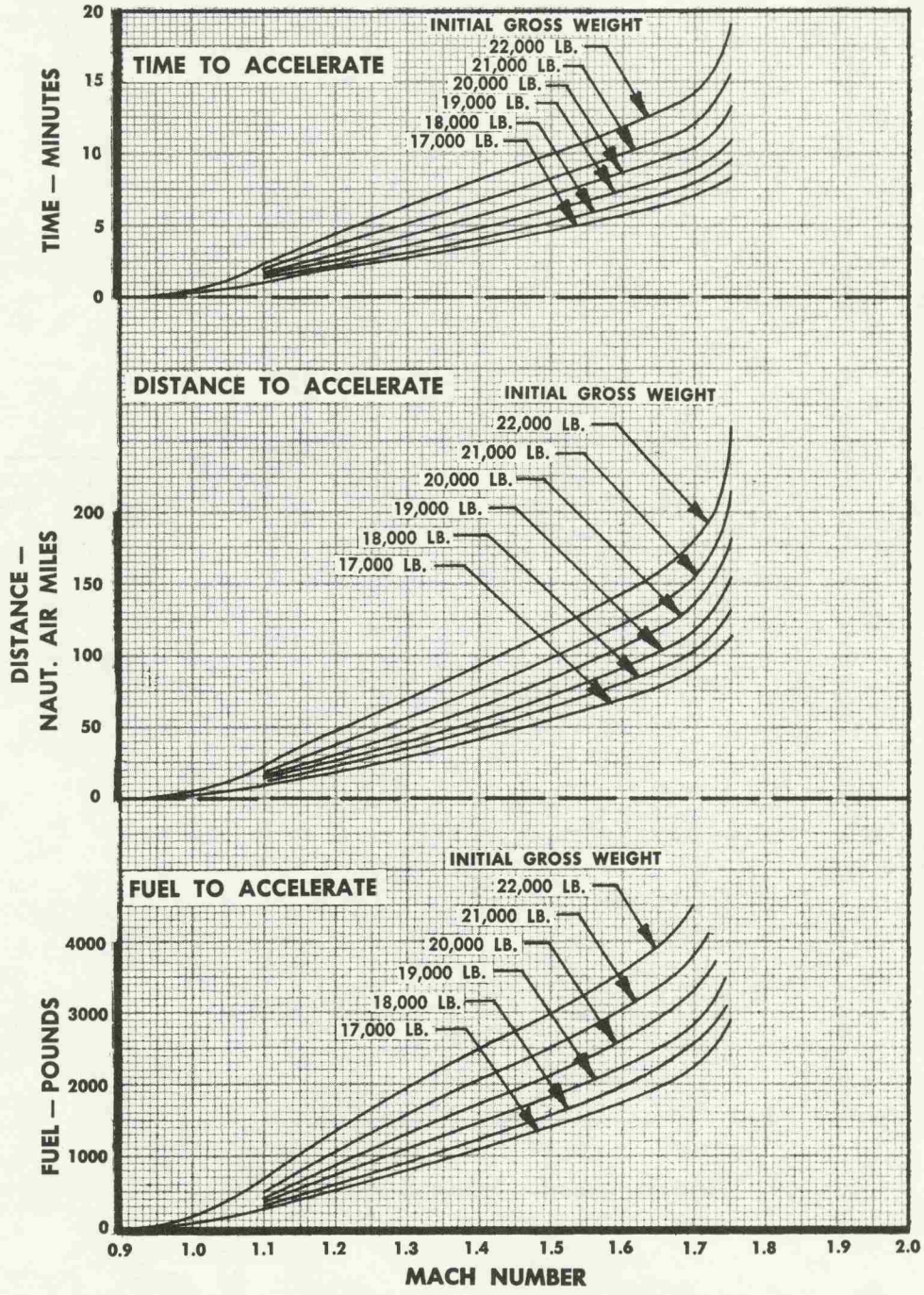


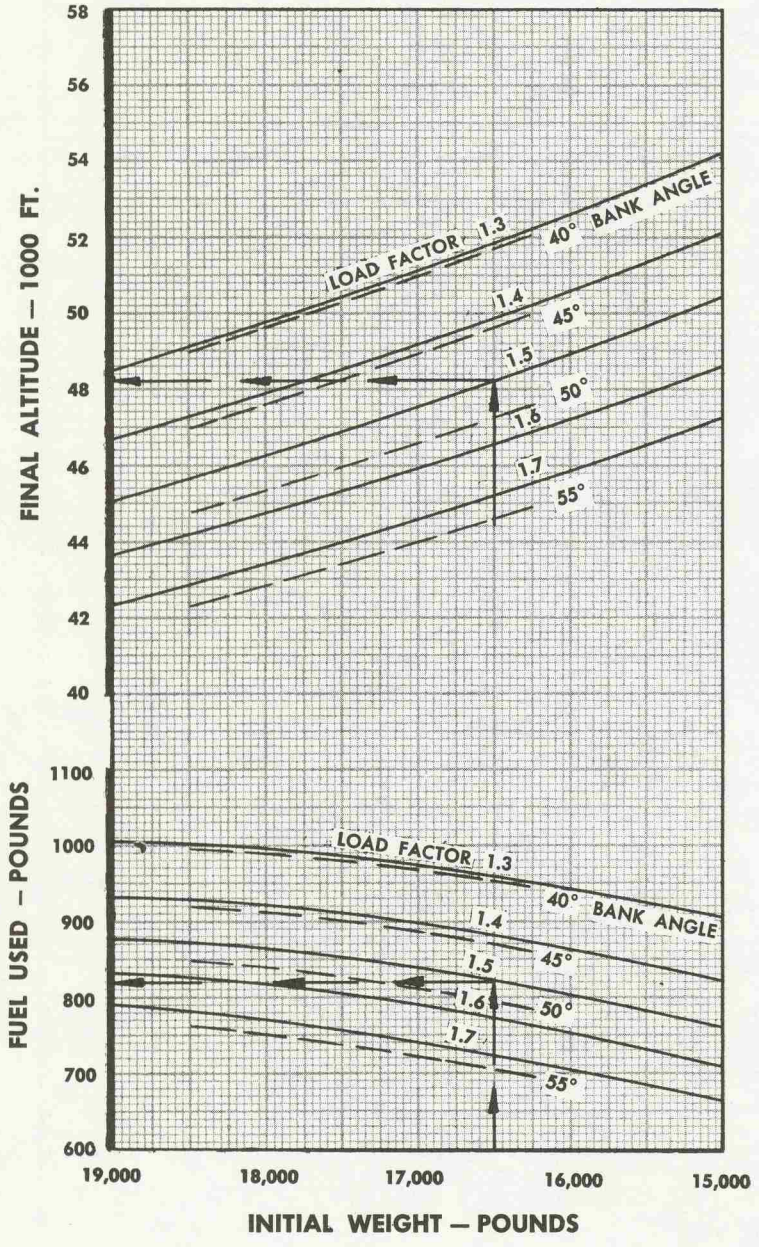
Figure A9-8

MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE
FROM 35,000 FEET
NO EXTERNAL STORES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH 1.8

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	18.7	3.4
1.4	15.7	2.9
1.5	13.9	2.5
1.6	12.4	2.3
1.7	11.3	2.1

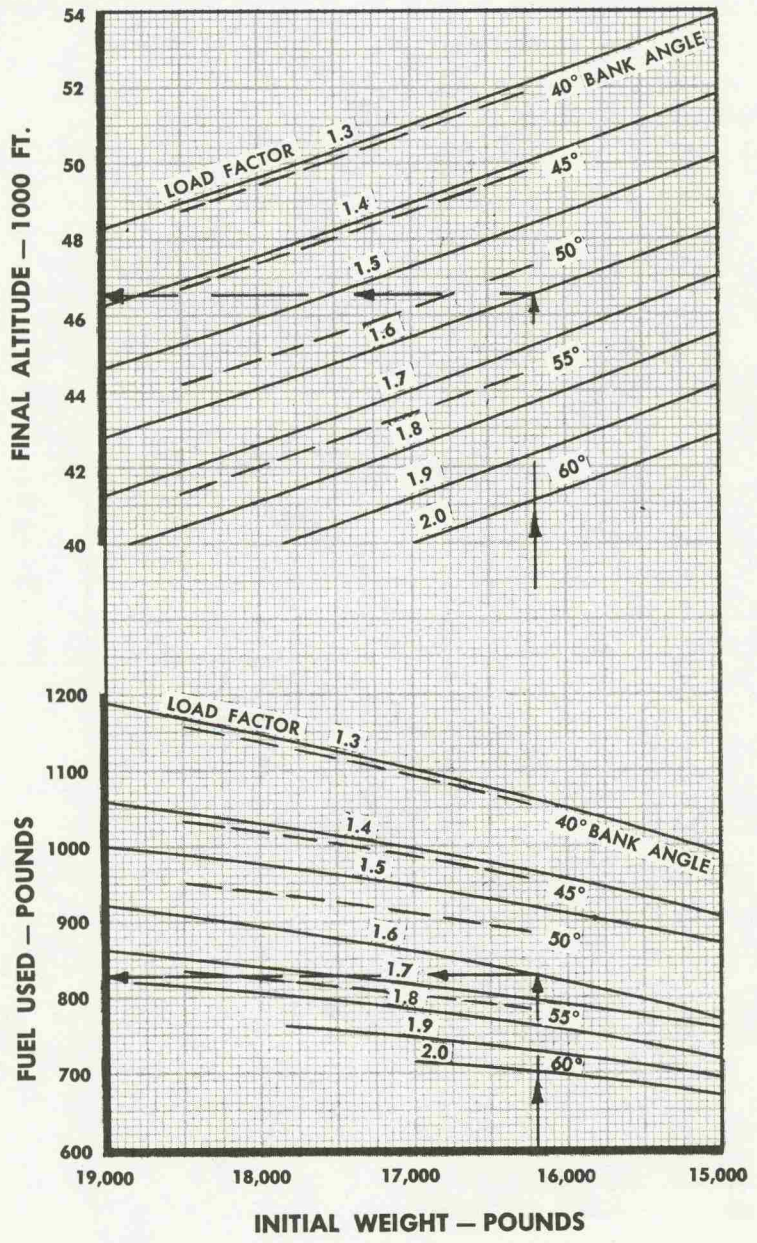
Figure A9-9

MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE
FROM 35,000 FEET
NO EXTERNAL STORES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH = 1.9

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	20.9	3.6
1.4	17.7	3.1
1.5	15.5	2.7
1.6	13.9	2.4
1.7	12.6	2.2
1.8	11.6	2.0
1.9	10.7	1.9
2.0	10.0	1.7

Figure A9-10

MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE

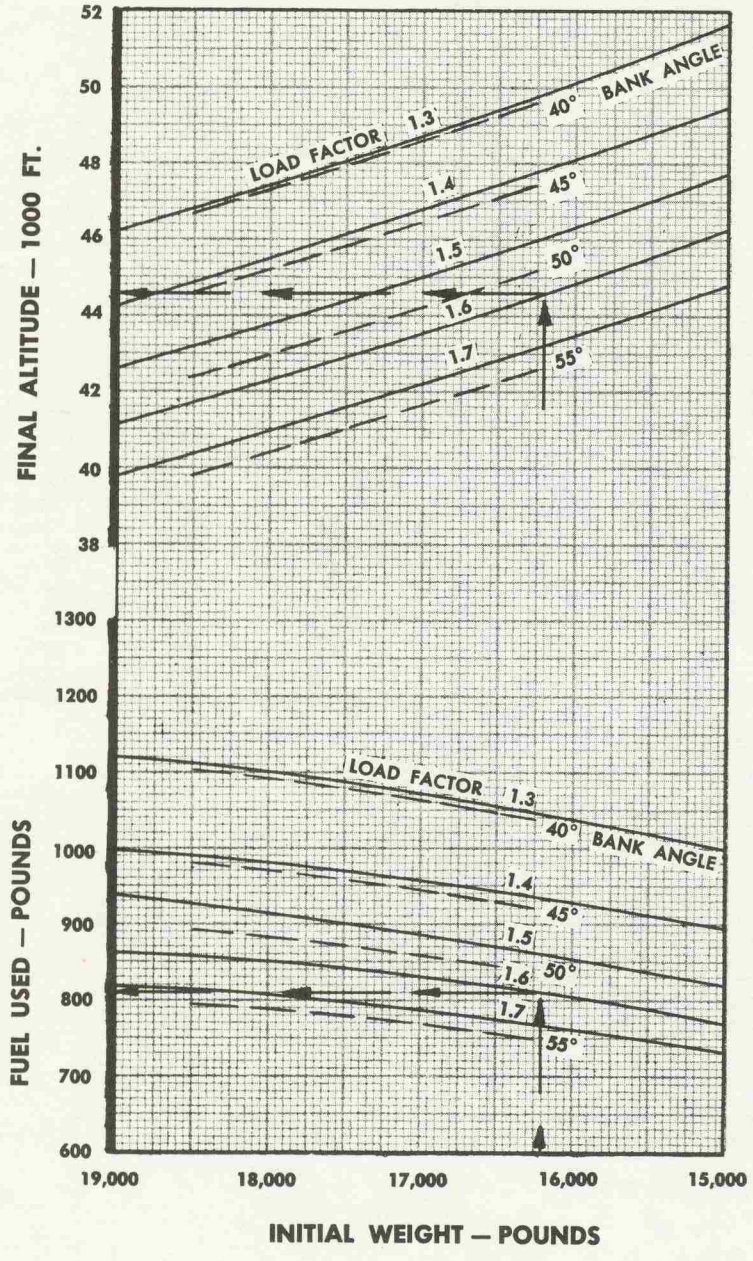
FROM 35,000 FEET

TWO TIP SIDEWINDER MISSILES

MACH 1.8

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	18.7	3.4
1.4	15.7	2.9
1.5	13.9	2.5
1.6	12.4	2.3
1.7	11.3	2.1

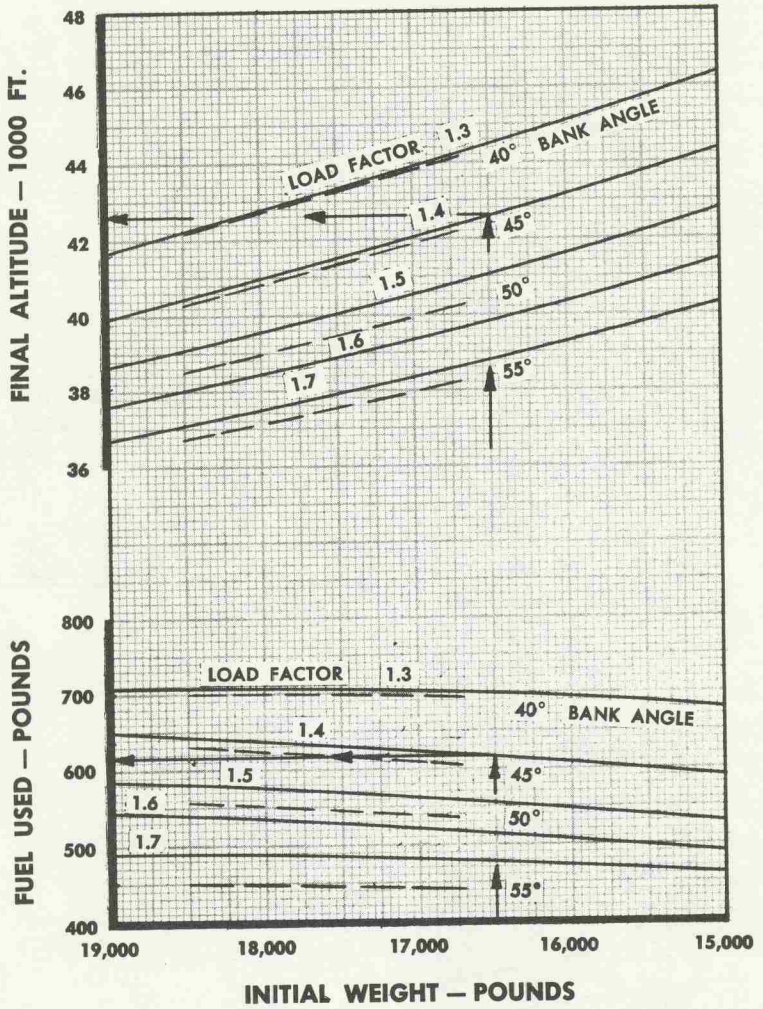
Figure A9-11

MAXIMUM THRUST 90° CLIMBING TURN PERFORMANCE
FROM 35,000 FEET
TWO TIP SIDEWINDER MISSILES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH = 1.9

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	20.9	1.8
1.4	17.7	1.5
1.5	15.5	1.3
1.6	13.9	1.2
1.7	12.6	1.1

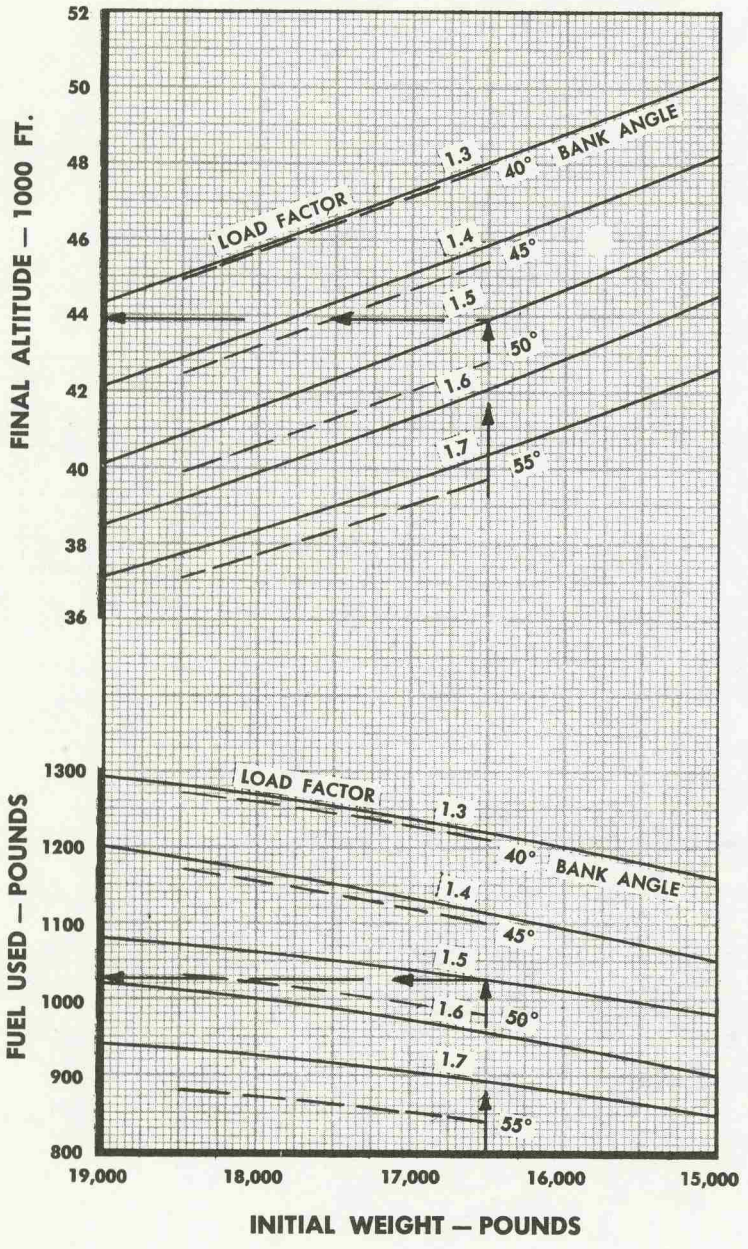
Figure A9-12

MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE
FROM 35,000 FEET
TWO TIP SIDEWINDER MISSILES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH = 1.9

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	20.9	3.6
1.4	17.7	3.1
1.5	15.5	2.7
1.6	13.9	2.4
1.7	12.6	2.2

Figure A9-13

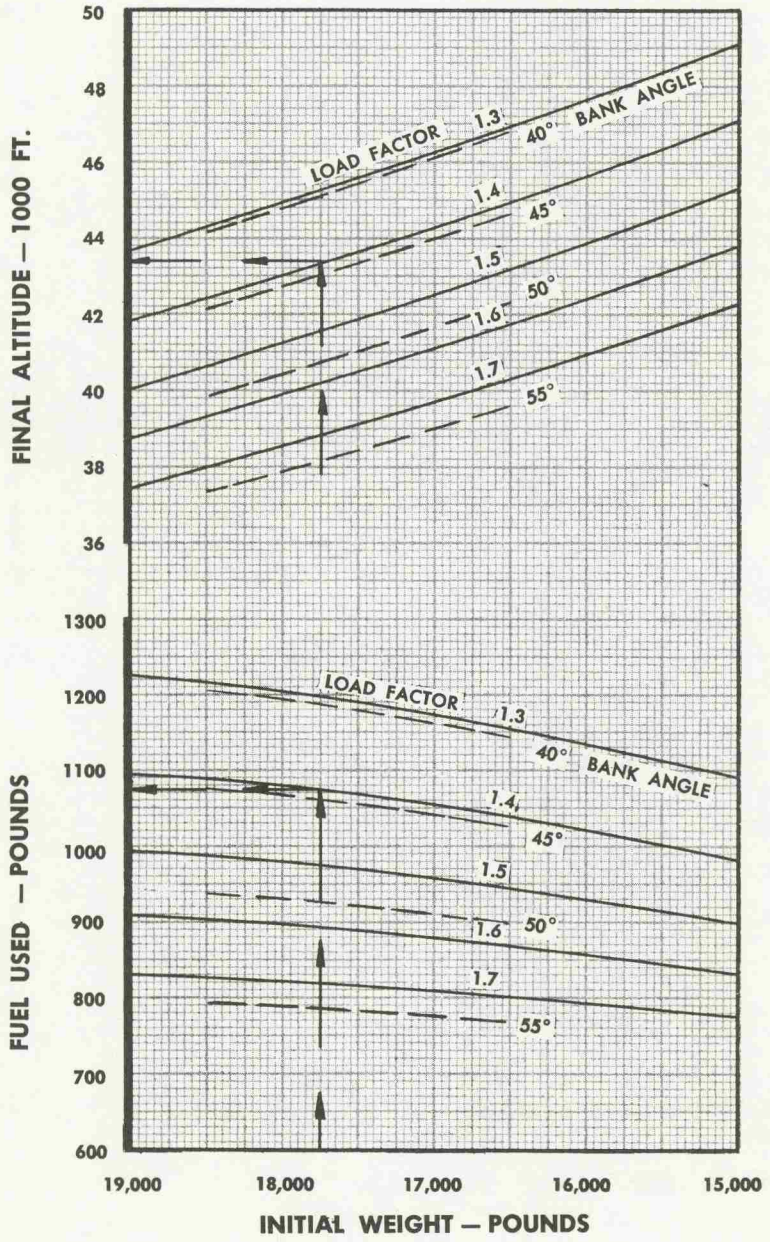
MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE FROM 35,000 FEET

TWO 165 GALLON TIP TANKS

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH 1.8

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	18.7	3.4
1.4	15.7	2.9
1.5	13.9	2.5
1.6	12.4	2.3
1.7	11.6	2.1

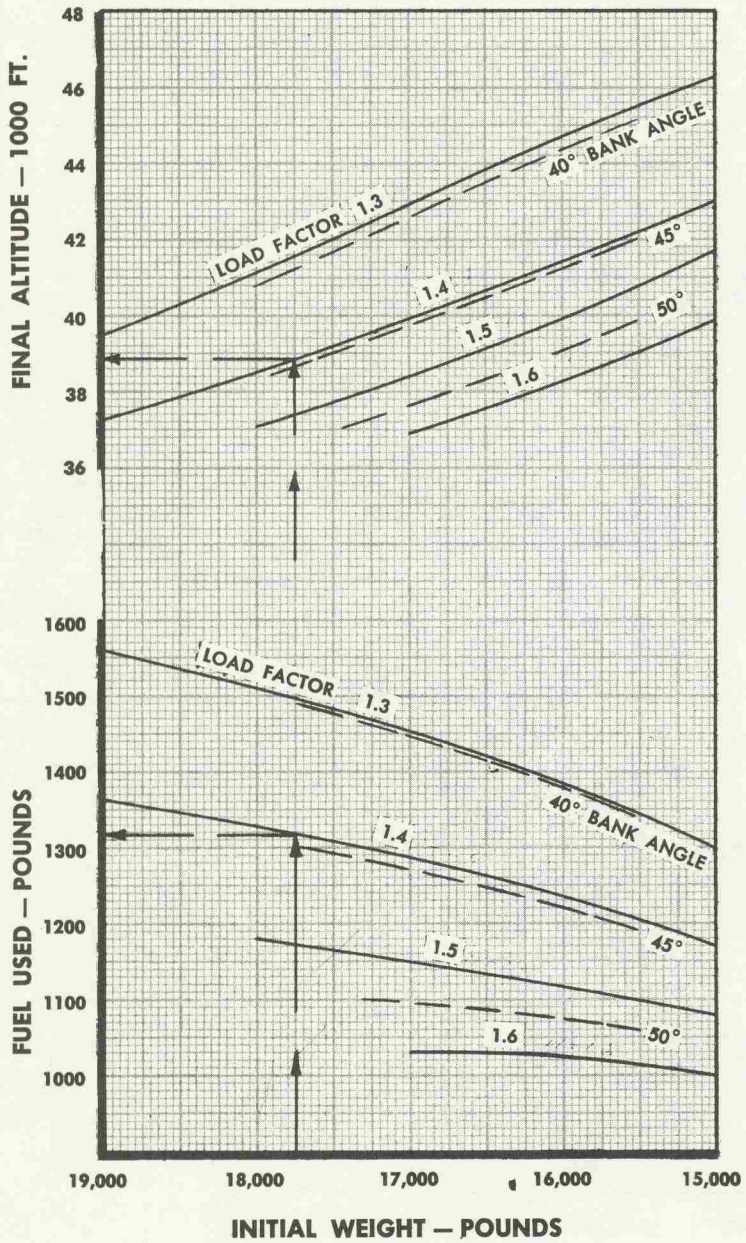
Figure A9-14

MAXIMUM THRUST 180° CLIMBING TURN PERFORMANCE
FROM 35,000 FEET
TWO 165 GALLON TIP TANKS

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

MACH = 1.9

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



LOAD FACTOR	TURN RADIUS N. MI.	TIME REQ'D MIN.
1.3	20.9	3.6
1.4	17.7	3.1
1.5	15.5	2.7
1.6	13.9	2.4

Figure A9-15

TIME, DISTANCE AND FUEL FOR MAXIMUM THRUST ZOOM ATTACKS ON A 65,000 FT. TARGET

NO EXTERNAL STORES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

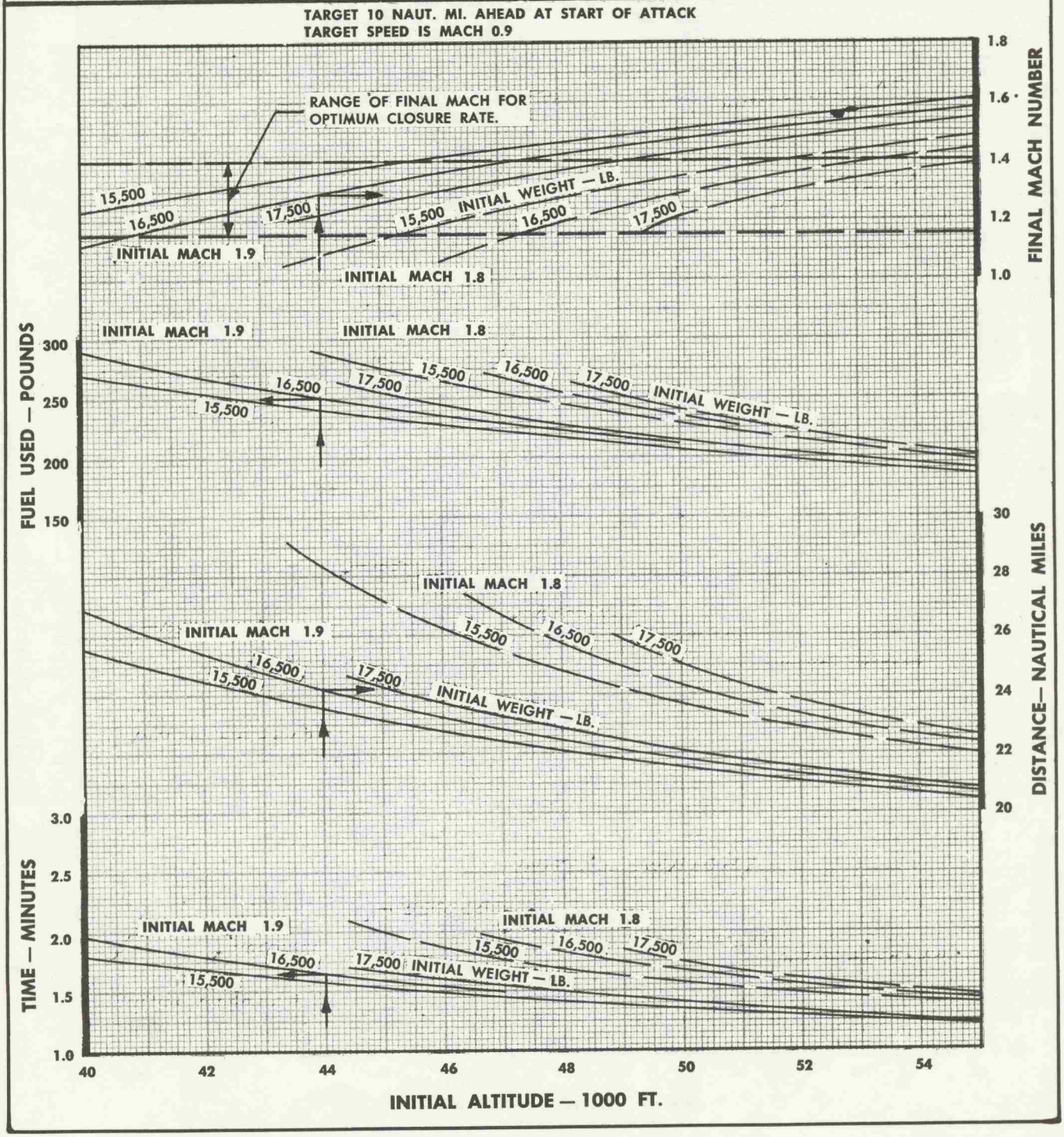


Figure A9-16

TIME, DISTANCE AND FUEL FOR MAXIMUM THRUST ZOOM ATTACKS ON A 65,000 FT. TARGET

TWO TIP SIDEWINDER MISSILES

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

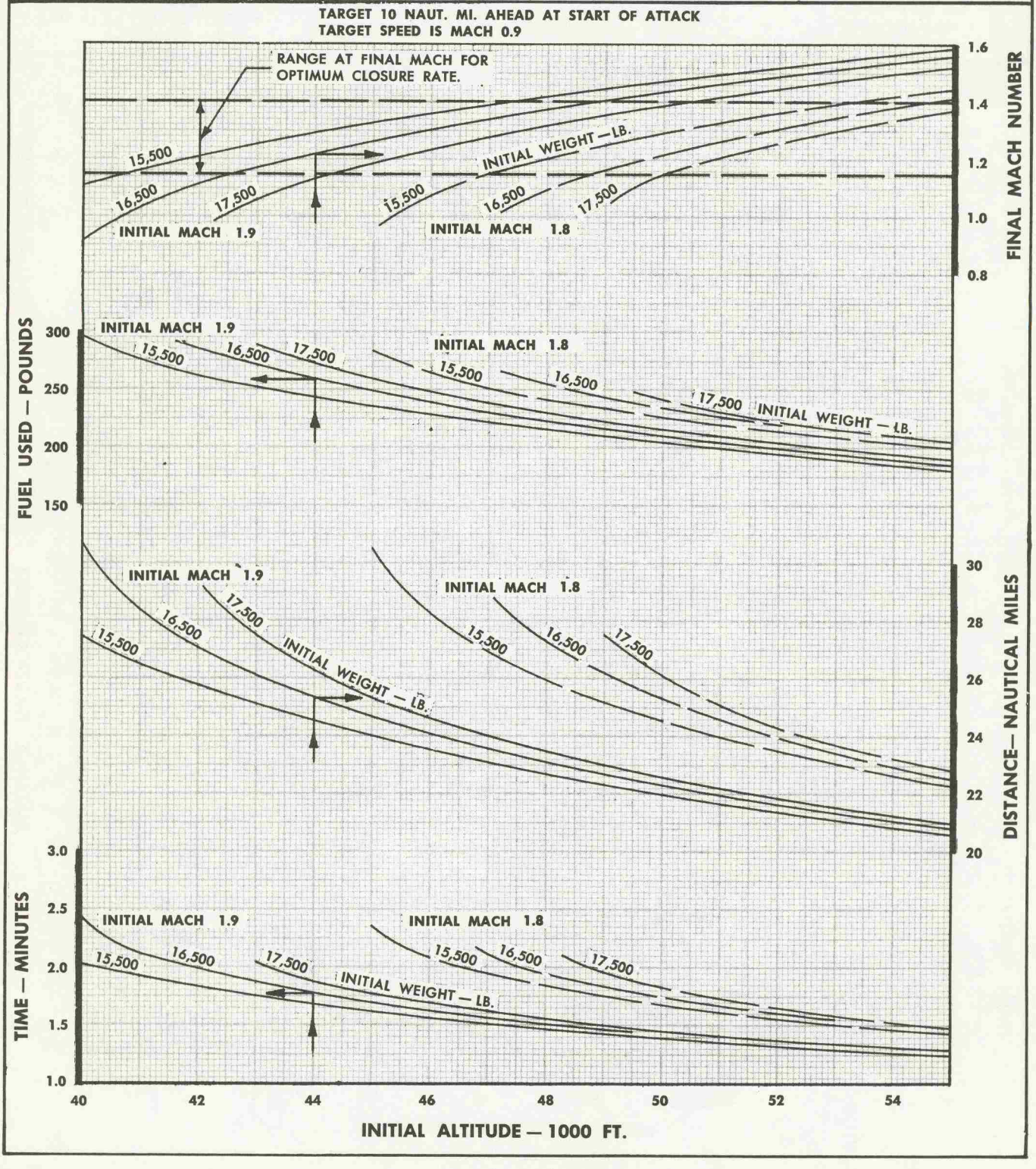


Figure A9-17

TIME, DISTANCE AND FUEL FOR MAXIMUM THRUST ZOOM ATTACKS ON A 65,000 FT. TARGET

TWO 165 GALLON TIP TANKS

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4

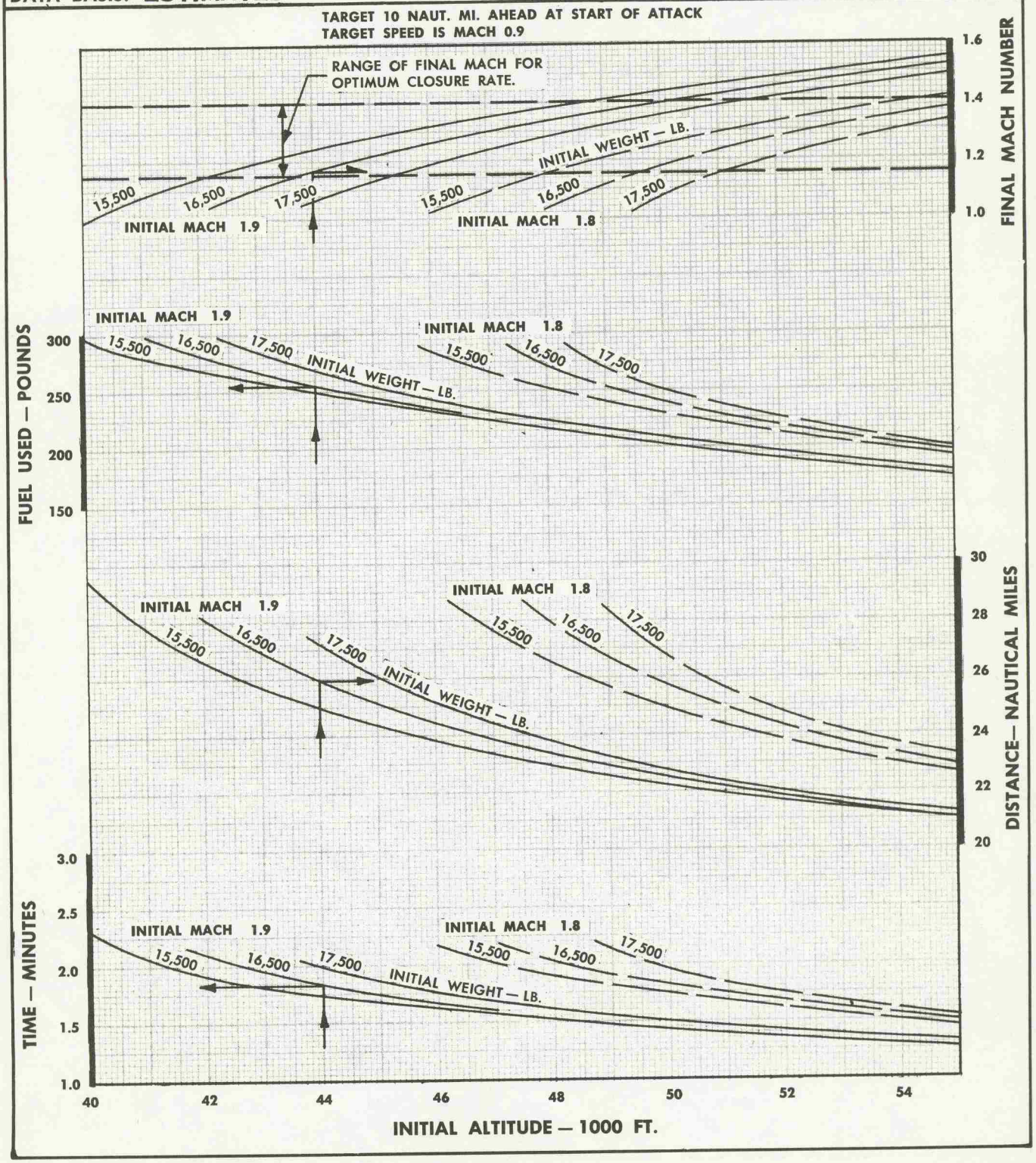


Figure A9-18

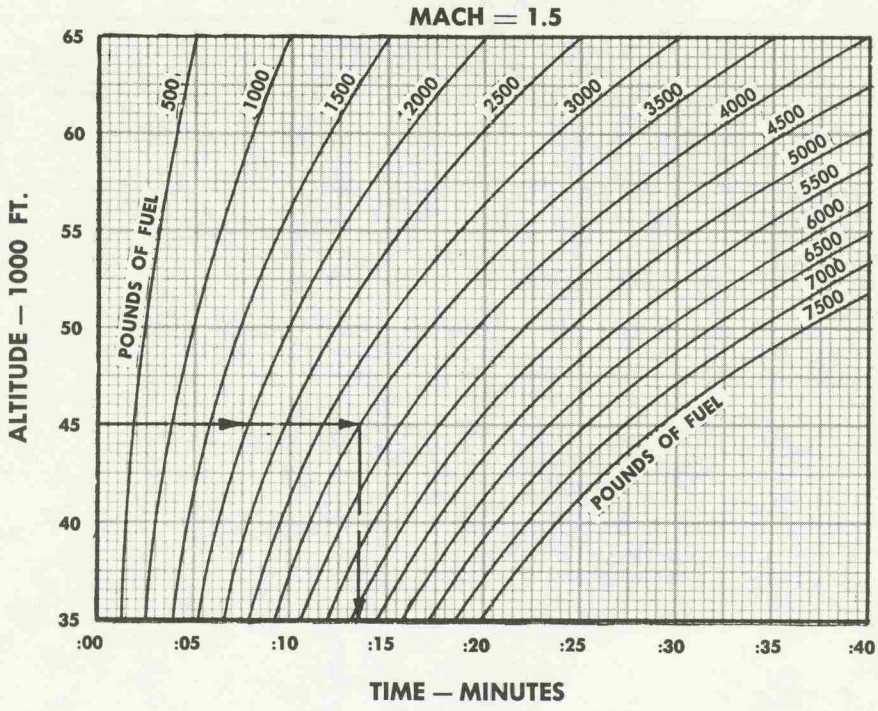
MAXIMUM THRUST COMBAT ALLOWANCE CHART — MACH 1.5 AND MACH 1.6

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

TIME LIMIT = 2 HOURS

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



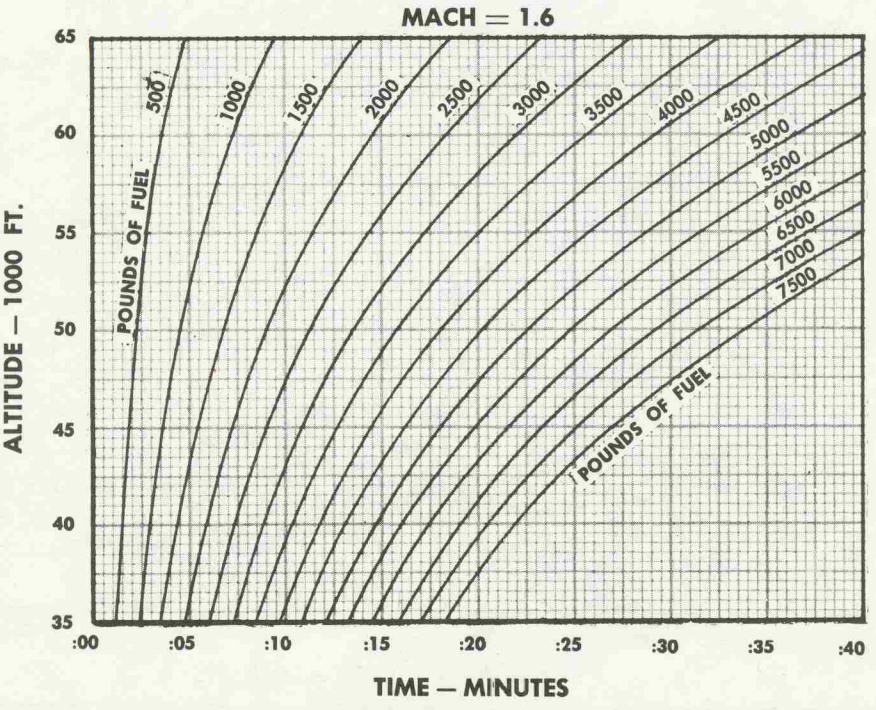


Figure A9-19

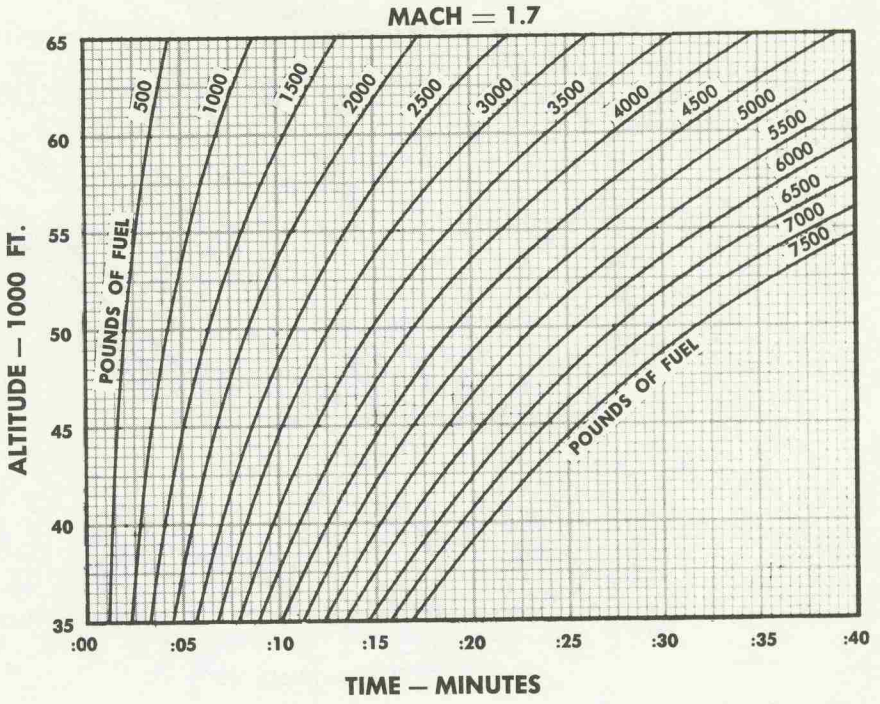
MAXIMUM THRUST COMBAT ALLOWANCE CHART—MACH 1.7 AND MACH 1.8

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

TIME LIMIT = 2 HOURS

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



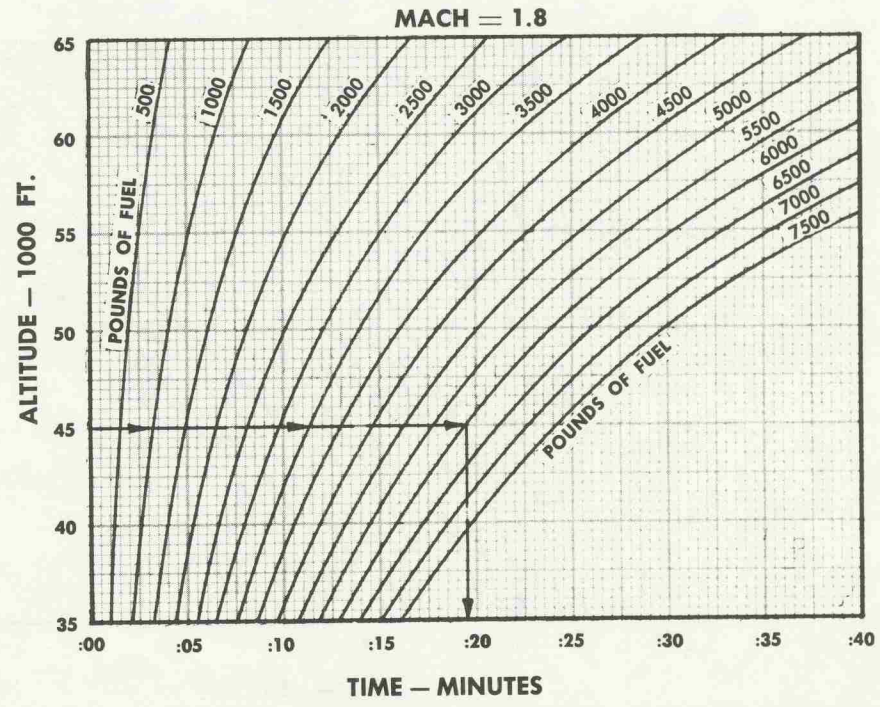


Figure A9-20

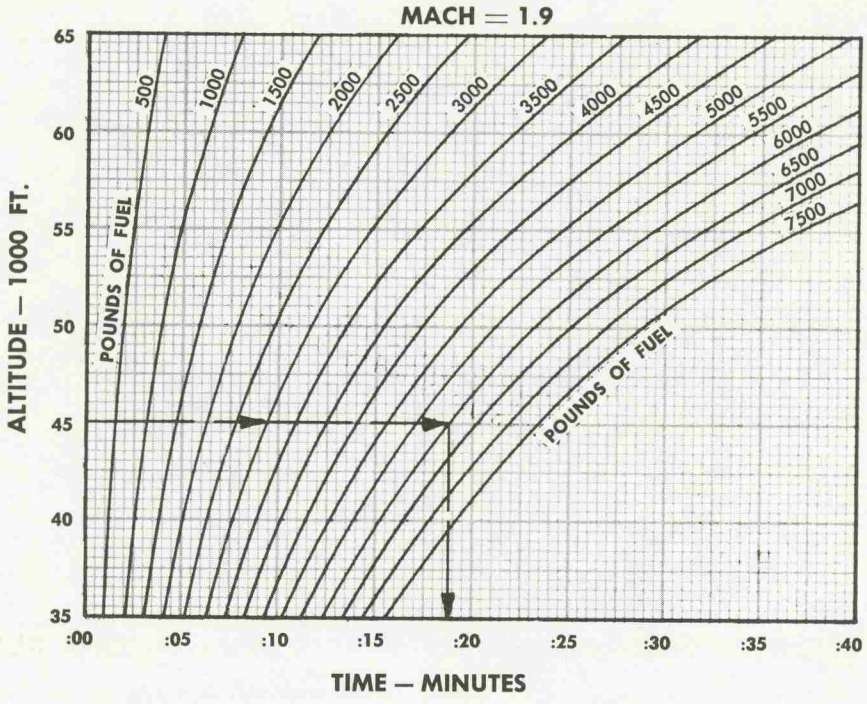
MAXIMUM THRUST COMBAT ALLOWANCE CHART — MACH 1.9 AND MACH 2.0

STANDARD DAY

Model: F-104A
Date: 1 FEBRUARY 1958
DATA BASIS: ESTIMATED

TIME LIMIT = 2 HOURS

ENGINE: J79-GE-3A
FUEL DENSITY: 6.5 LB./GAL.
FUEL GRADE: JP-4



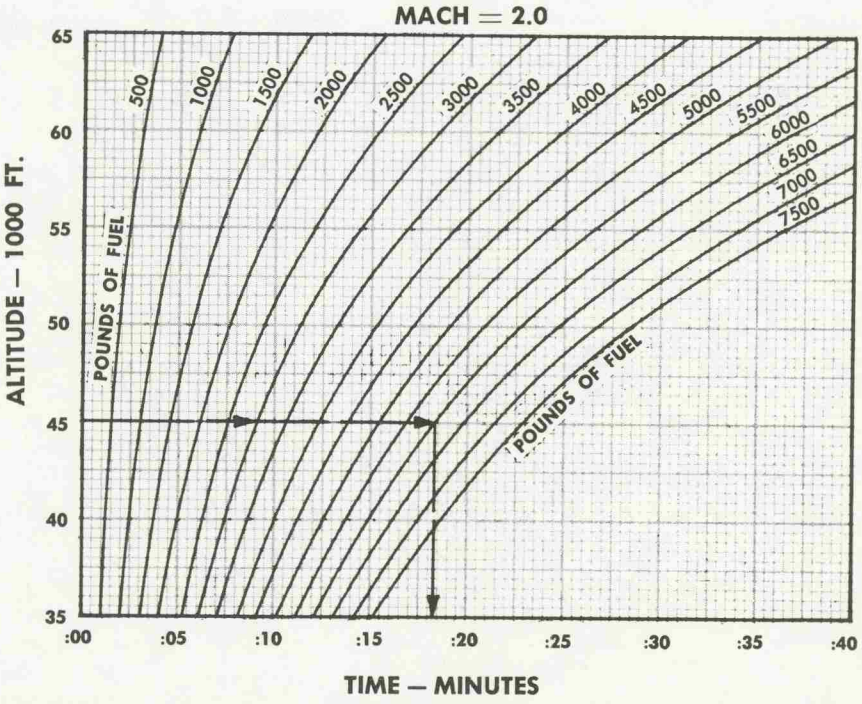


Figure A9-21

Part 10—Mission Planning

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MISSION PLANNING

NEED FOR MISSION PLANNING.

Performance available and fuel consumption resulting from operation at high supersonic speeds have introduced many new problems to interceptor operations. The number of performance variables involved in acceleration to combat speed, high speed turns to the intercept course, and probable climbs or zooms to the point of intercept may require that the major portion of the flight be made under ground controlled intercept procedure to the point where actual fire control contact with the target is made. Time required for these supersonic maneuvers may not be more than five or six minutes. However, the high rate of fuel consumption does not allow much margin of error. One thousand pounds of fuel represents from two to ten minutes of operation at combat speeds with Maximum Thrust, or the equivalent of approximately 225 nautical miles of long range flight. Thus, any delay in making the initial intercept seriously reduces actual combat time available. If afterburner or thrust settings are used, any delay in breaking off the contact can seriously deplete the planned fuel reserve. Accurate mission planning and constant checking of progress against the flight plan becomes necessary to obtain optimum use of the aircraft.

MISSION FLIGHT PLANNING EXAMPLE.

The charts presented in Parts 1 through 9 provide the performance information necessary to plan many different types of missions. The following example illustrates how the performance charts may be intregrated to form a complete intercept mission flight plan.

SAMPLE PROBLEM.

Configuration: Two GAR-8 launchers and missiles. Full internal fuel 908 gal. (5902 lbs). Fully loaded weight prior to starting engine is assumed to be 19,420 lbs. Assumed Mission and Flight sequence:

1. Maximum Thrust take-off and acceleration to 350 KIAS; Military Thrust acceleration from 350 KIAS to climb speed.

2. Military Thrust climb at Mach 0.88 to optimum cruise altitude.

3. Cruise-climb until 140 miles from base.

4. Military Thrust climb to 35,000 feet.

5. Maximum Thrust acceleration at 35,000 feet to Mach 1.9.

6. 180° Maximum Thrust climbing turn at Mach 1.9 and 1.6 G.

7. Zoom to 65,000 feet.

8. Fire GAR-8 missiles.

9. Descend to optimum cruise altitude.

10. Cruise-climb return to base with 1000 lbs. fuel reserve at cruise altitude.

11. Descend and land.

Supplemental Data:

1. Weather: Field ambient air temperature 59 degrees F., field pressure altitude sea level, zero runway slope, zero wind. Assume standard day temperatures and zero wind for other portion of flight.

2. Runway: No. 36, 7,000 feet long, dry, hard surface.

Take-Off, Climb to Optimum Cruise Altitude, and Outbound Cruise.

Obtain take-off data from Part A2. Climb and cruise planning data may be obtained from Part A4 using the applicable mission profile.

The fuel and time required to takeoff, climb to cruise altitude, and cruise to a point 140 nautical miles from base may be determined from Figure A4-2. 140 miles is the "key point" for this problem, the point where the combat phase of the flight i begun. The maximum intercept radius is obtained when he "key point" is the farthest distance from base consistent with return and reserve fuel allowances. A cut-and-try procedure can be used to find maximum "key point" distance, or a simple plot can be drawn to obtain a graphical solution. The steps used to develop such plots are shown with the problem.

Enter Figure A4-2 at 140 nautical miles and read at the intersection of the cruise-climb flight path:

Fuel required .....1,640 lbs.

Gross weight at the end of cruise out
(19,420 lbs. − 1,640 lbs.) .....17,780 lbs.

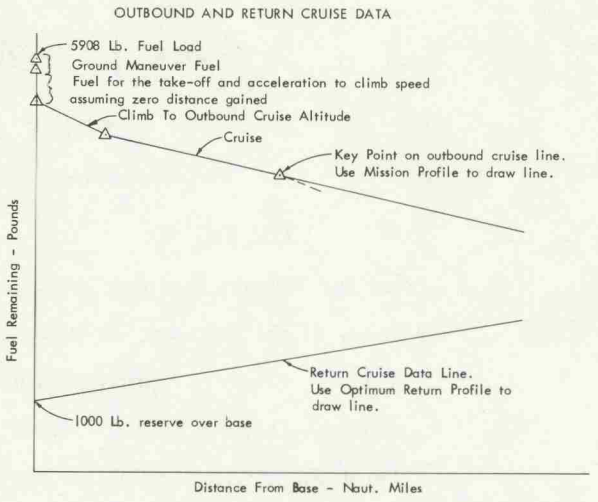


Figure A10-1

Time required in take-off, climb, and cruise out .....18.0 min.

Total time at end of outbound cruise .....18.0 min.

Fuel remaining (5,902 lbs. − 1,640 lbs.) .....4,262 lbs.

These data include:

a. Maximum gross weight at engine start....19,420 lbs.

b. Fuel used in ground maneuver, maximum thrust take-off, acceleration from 350 K.I.A.S. to climb speed.

c. Military Thrust climb at Mach 0.88 true.

d. Time from brake release.

See Figure A10-1 for sample of outbound cruise data plot. Note that return cruise data also can be included at this time, using the no-stores Optimum Return Profile, Figure A4-7.

Cruise performance data and the initial engine setting may be determined from the tabulated data box in Figure A4-2. Read at the Optimum Cruise—Cruise—climb.

Set initial RPM 91%

Approximate initial fuel flow .....2,710 lb/hr.

Trim RPM to obtain indicated cruise Mach No. 0.81

Initial correct pressure altitude .....31,000 ft.

Initial indicated pressure altitude (from Figure A1-7 for an indicated Mach No. of 0.81) (31,000 ft − 410 ft.) ..... 30,590 ft.

Final correct pressure altitude at a distance of 140 N.Mi. from base ..... 31,500 ft.

Final indicated pressure altitude (31,500 ft. − 410 ft.) ..... 31,090 ft.

Time at start of cruise ..... 5.5 min.

Time at a distance of 140 N.Mi. from base ..... 18.0 min.

Time elapsed during cruise-out (18.0 min. − 5.5 min) ..... 12.5 min.

Approximate rate-of-climb during cruise-out (31,500 ft. − 31,000 ft.)/12.5 min. ..... 40 ft/min.

Military Thrust Climb to 35,000 ft.

The time, distance, and fuel required to climb from cruise altitude to 35,000 ft. may be determined from Figure A3-12: For a gross weight at the end of cruise-climb of 17,780 lbs. and a final cruise altitude of 31,500 ft. read:

Gross weight at 35,000 ft. ..... 17,730 lbs.

Fuel used in climb (17,780 lbs. − 17,730 lbs.) ..... 50 lbs.

Fuel remaining at 35,000 ft. (4,262 lbs. − 50 lbs.) ..... 4,212 lbs.

Time at final cruise altitude ..... 4.4 min.

Time at 35,000 ft. ..... 5.4 min.

Time to climb (5.4 min. − 4.4 min.) ..... 1.0 min.

Total elapsed time at 35,000 ft. 18.0 min. + 1.0 min.) ..... 19.2 min.

Distance at final cruise altitude ..... 39.7 N.Mi.

Distance at 35,000 ft. ..... 47.7 N.Mi.

Distance traveled in climb (47.7 N.Mi. − 39.7 N.Mi.) ..... 8.0 N.Mi.

Distance from base 35,000 ft. (140.0 N.Mi. + 8.0 N.Mi.) ..... 148 N.Mi.

Level Flight Acceleration to Mach 1.9.

The time, distance and fuel requirements for an acceleration to Mach 1.9 at 35,000 feet pressure altitude may be determined from Figure A9-4. The effect of weight reduction during acceleration is included in these charts. Performance is determined as follows:

From final Mach number desired (1.9) and initial weight (17,730 pounds) read:

Time to accelerate ..... 3.6 min.

Distance to accelerate ..... 49.0 n.mi.

Fuel to accelerate ..... 1,270 lb.

Final weight (17,730 lb. − 1,870 lb.) ..... 16,460 lb.

Fuel remaining (4,212 lb. − 1,270 lb.) ..... 2,942 lb.

Distance from base (148 n.mi. + 49 no.mi.) ..... 197 n.mi.

Total time at end of acceleration to Mach 1.9 (19.0 min. + 3.6 min.) ..... 22.6 min.

See Figure A10-2 for addition of climb and level acceleration performance data.

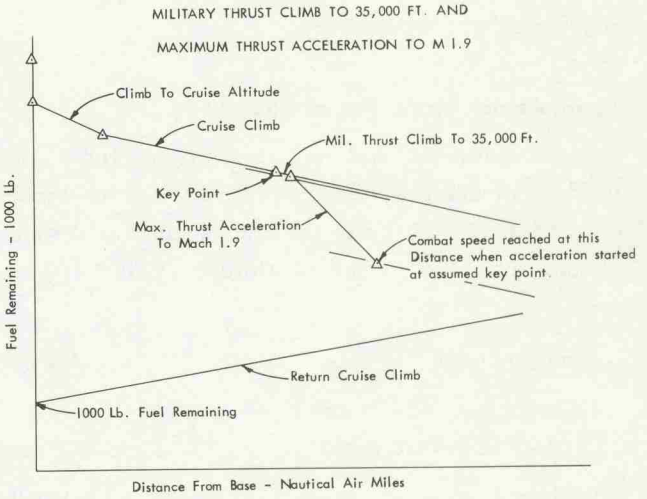


Figure A10-2

180° Maximum Thrust Climbing Turn.

The maximum Thrust Climbing Turn performance for a Mach 1.9, 1.6 G turn may be obtained from figure A9-13.

Load factor ..... 1.6 G

Fuel used in Climbing Turn ..... 958 lb.

Fuel remaining (2,942 lb. − 958 lb.) ..... 1,984 lb.

Final weight (16,460 lb. − 958 lb.) ..... 15,502 lb.

Time required for Climbing Turn ..... 2.40 min.

Turning radius ..... 13.9 n.mi.

Total time at end of 180° climbing turn (22.6 min. + 2.40 min.) ..... 25.0 min.

Altitude at end of Climbing Turn ..... 42,000 ft.

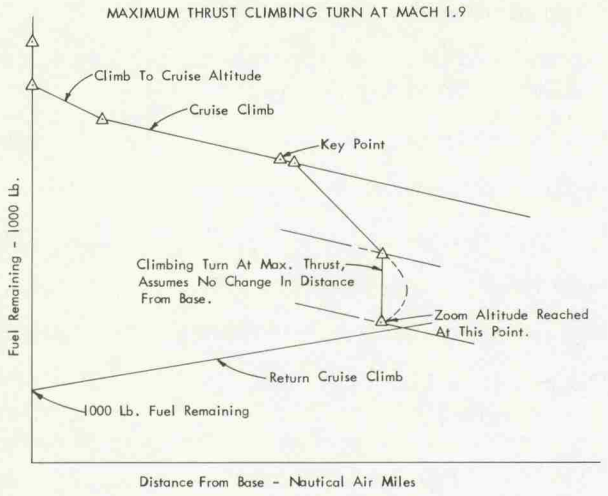


Figure A10-3

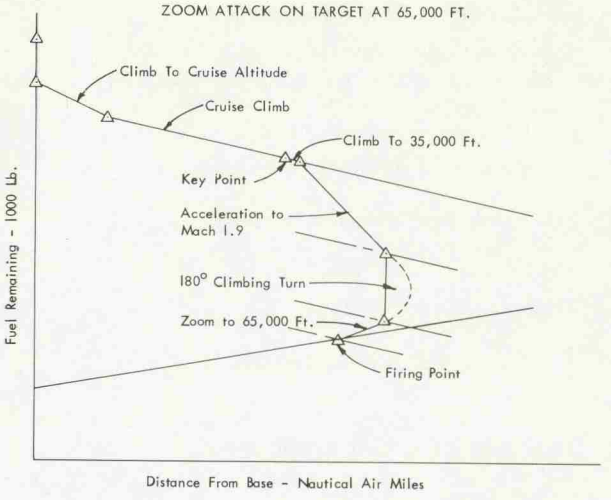


Figure A10-4

See Figure A10-3 for addition of Climbing Turn Fuel allowance.

Zoom Attack on Target at 65,000 ft.

The time, distance, and fuel requirements for a zoom attack from an initial speed of Mach 1.9 may be obtained from Figure A9-17. Enter the chart at the initial weight (15,502 lbs.) and the initial altitude (42,000 ft.) and read:

Time to zoom .....1.8 min.

Distance traveled in zoom (Assume toward base) .....26.0 n.mi.

Fuel used in zoom .....262 lbs.

Final Mach Number 1.2

Gross Weight at 65,000 feet (15,502 lbs. − 262 lbs.) .....15,240 lbs.

Assume that the GAR-8 missiles are fired as 65,000 feet altitude is reached. The final gross weight after intercept is found by subtracting the weight of the missiles (310 lbs.) from the weight at the zoom peak.

Final gross weight after intercept (15,240 lbs. − 310 lbs.) .....14,930 lbs.

Time from brake release to completion of intercept (25.0 min. + 1.8 min.) .....26.8 min.

Distance from base at completion of intercept (197.0 n.mi. − 26.0 n.mi.) .....171.0 n.mi.

Fuel remaining at completion of intercept (1,984 lbs. − 262 lbs.) .....1,722 lbs.

See Figure A10-4 for addition of zoom performance data and firing of missiles.

Descent.

It has been assumed for convenience that the return will start from the distance out at the completion of the intercept. No time, distance, or fuel requirements for the descent to cruise climb altitude have been considered. This is conservative in that if the descent to cruise-climb altitude is made toward the base, additional range and fuel reserves are available. If combat is terminated at the higher altitudes, turn toward base and descend at Idle Thrust and 300 knots I.A.S. to cruise-climb altitude.

Return Cruise-Climb Performance.

The return cruise-climb performance may be determined from Figure A4-7. Optimum return profile (No External Stores) for the 171 nautical air mile return.

Fuel remaining at start of return cruise .....1,722 lb.

Enter figure A4-7 at the intersection of 1,722 lb. fuel remaining with the optimum cruise line.

Distance reading at initial fuel remaining of 1,722 lb. .....420 n.mi.

Distance reading with 1,000 lb. reserve fuel remaining .....249 n.mi.

Range available (420 mi. − 249 mi.) .....171 n.mi.

Initial cruise time reading .....51.0 min.

Final cruise time reading .....30.0 min.

Return cruise time (51 min. − 30 min.) .....21.0 min.

Total flight time on return to base is (26.8 + 21.0 min.) .....47.8 min.

Figure A10-5 shows how the key point radius may be determined for returns with 1,000 pounds fuel reserve or with zero reserve.

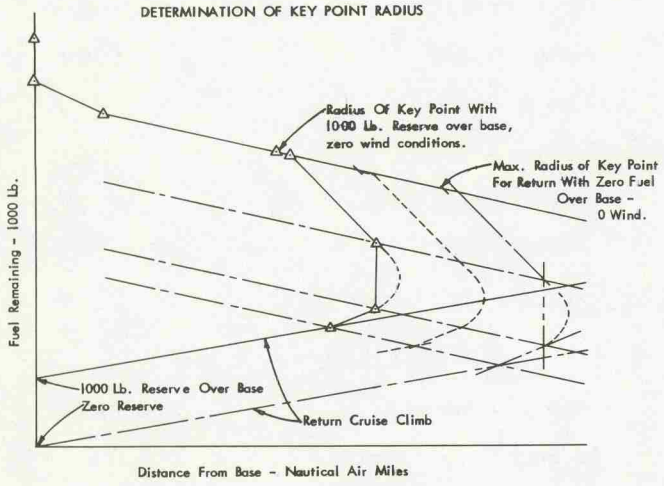


Figure A10-5

Return cruise-climb performance data and engine settings may be determined from the tabulated data box in Figure A4-8 as was done for the initial cruise.

Cruise-climb at indicated Mach No. ..... .84

Determine the initial and final cruising altitudes as follows:

Initial correct pressure altitude at 1,722 lb. of fuel remaining .....35,700 ft.

Initial indicated pressure altitude for indicated cruise Mach number of 0.84 (35,700 − 430 ft.) .....35,270 ft.

Final correct pressure altitude at 1,000 lb. of fuel remaining .....36,700 ft.

Final indicated pressure altitude (36,700 ft. − 425 ft.) .....36,275 ft.

Approximate rate of climb (36,700 ft. − 35,700 ft.)/21.0 = 47.6 ft/min.

Landing Reserve.

Weight decrease during the flight to the end of the return cruise is 19,420 pounds minus 14,208 pounds or 5,212 pounds. The weight or stores expended was 310 pounds so fuel used amounts to 4,902 pounds. Initial fuel load was 908 gallons, which is 5,902 pounds at 6.5 pounds per gallon fuel density. Fuel reserve for descent and landing is, therefore, 5,902 pounds minus 4,902 pounds or 1,000 pounds. This should be adequate. However, reference to the combat allowance charts will show that this entire amount could be consumed in 2.3 minutes at Maximum Thrust at 35,000 feet at Mach 1.6. On arrival at the take-off point and at altitude, the pilot could:

1. Utilizing the cruise-climb procedure, fly to an alternate air field at a distance of 122 nautical miles and have a 500 pound fuel reserve.

2. Cruise-climb to an alternate air field at a distance of 185 nautical miles and have a fuel reserve of 250 pounds.

Figure A10-6 summarizes the mission planning data developed using the key point for return with 1,000 pounds reserve.

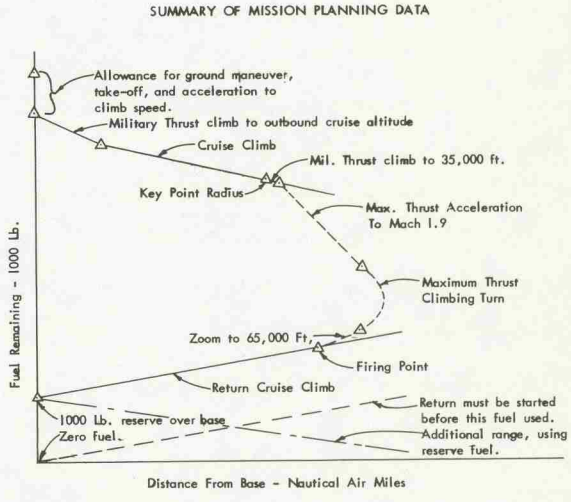


Figure A10-6

Take-Off Data Card.

Take-off planning data may be obtained from Part A2. The airplane take-off gross weight is determined by subtracting the fuel allowance for ground maneuvering (150 pounds, or about 5 minutes at idle thrust) from the gross weight prior to starting the engine. Data which would be needed if a heavy weight landing were to be made immediately after take-off are obtained from Part A8. The data are recorded on the Take-Off Data Card as follows:

TAKE-OFF DATA CARD

CONDITIONS

Gross Weight . . . . . 19,270 lb.
Runway Length . . . . . 7,000 ft.
Runway Temperature . . . . . 60° F.
Pressure Altitude . . . . . SL ft.
Runway Gradient 0%
Runway Wind Component . . . . . 10 kts.

TAKE-OFF

AFTERBURNER, Yes X, No

Take-Off Ground Roll Distance . . . . . 3,230 ft.
(zero wind, zero slope)
Take-Off Speed . . . . . 189 kts. IAS
Acceleration Check Distance . . . . 6000 ft. Marker
Acceleration Check Speed . . . . . 112 kts. IAS

LANDING IMMEDIATELY AFTER TAKE-OFF

Approach Speed . . . . . 197 kts. IAS
Landing Ground Roll . . . . . 3350 ft.
Available Rate of Climb
For Go Around . . . . . 650 ft./min.

See condensed check list in Section II for Take-Off Data Card form.

Landing Data.

The landing performance is determined from Figure A8-1 and recorded on the Landing Data Card. See condensed check list in Section II for Landing Data Card form.

LANDING DATA CARD

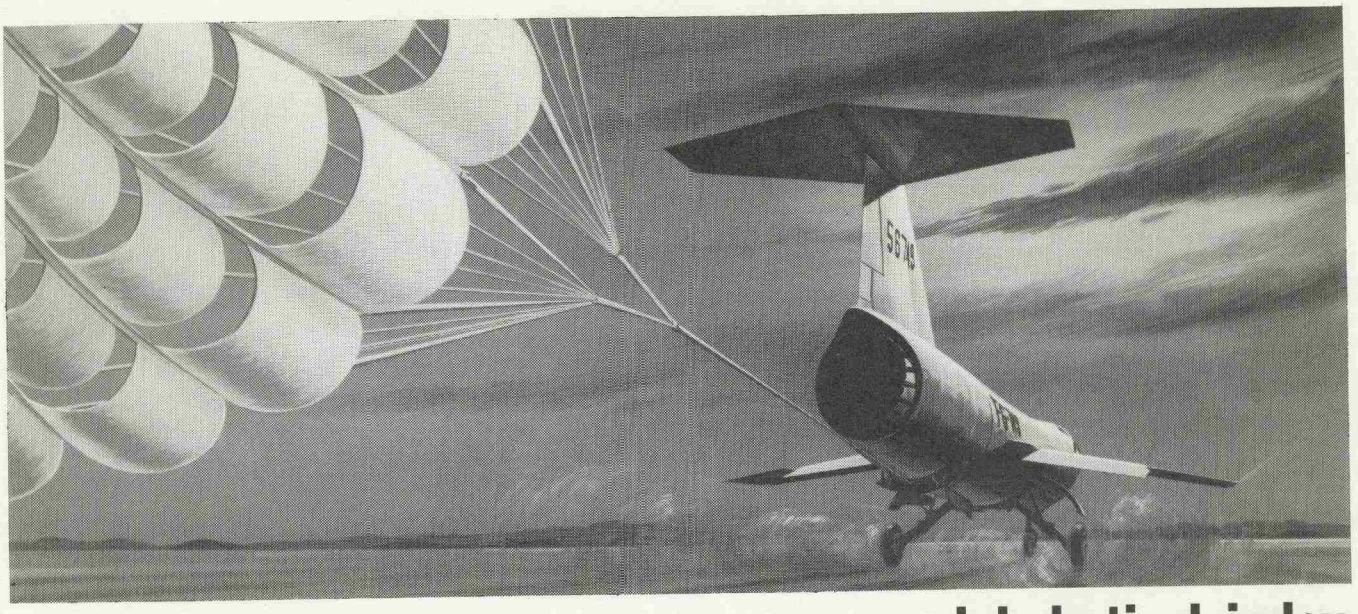
CONDITIONS

Gross Weight . . . . . 14,208 lb.
Runway Length . . . . . 7,000 ft.
Runway Temperature . . . . . 60° F.
Pressure Altitude . . . . . SL ft.
Runway Gradient 0%
Runway Wind Component . . . . . 0 kts.

LANDING

DRAG CHUTE, Yes X, No

Approach Speed . . . . . 170 kts. IAS
Landing Ground Roll . . . . . 2,100 ft.
Available Rate of Climb
For Go-Around . . . . . 2,675 ft./min.



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