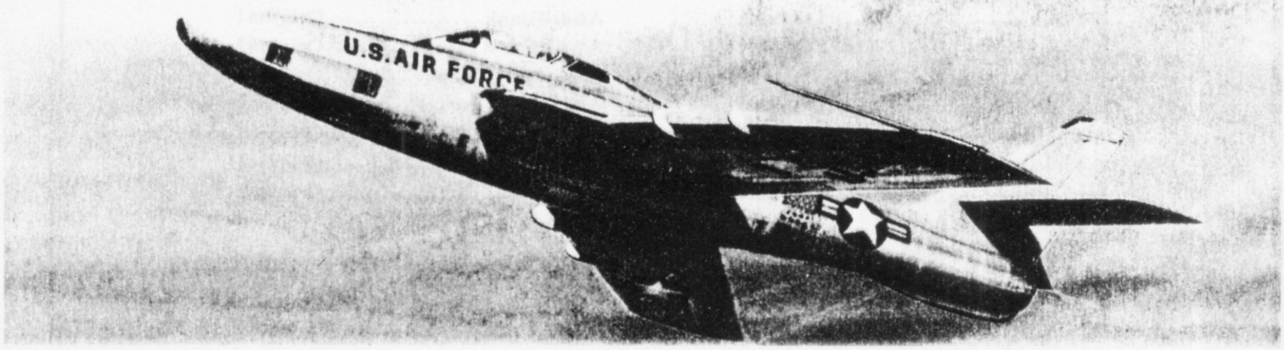
T.O. 1F-84(R)F-1

FLIGHT MANUAL

USAF SERIES

RF-84F-5 AND LATER, AIRCRAFT



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

This publication replaces T.O. 1F–84(R)F–1 dated 15 February 1963. See T.O. Index, T.O. 0–1–1–4A for current status of Safety Supplements, Flight Manuals and Flight Crew Checklists.

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

1 JANUARY 1966

INSERT LATEST CHANGED PAGES. DESTROY SUPERSEDED PAGES.

LIST OF EFFECTIVE PAGES

NOTE: The portion of the text affected by the changes is indicated by a vertical line in the outer margins of the page.

TOTAL NUMBER OF PAGES IN THIS PUBLICATION IS 342**, CONSISTING OF THE FOLLOWING:**

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CURRENT FLIGHT CREW CHECKLIST

T.O.1F-84(R)F-ICL-1

15 February 1963

Changed 1 January 1966

ADDITIONAL COPIES OF THIS PUBLICATION MAY BE OBTAINED AS FOLLOWS:

USAF

USAF ACTIVITIES. — In accordance with T.O. 00-5-2.

NAVY ACTIVITIES. — Submit request to nearest supply point listed below, using form NavAer 140; NASD, Philadelphia, Pa.; NAS, Alameda, Calif.; NAS, Jacksonville, Fla.; NAS, Norfolk, Va.; NAS, San Diego, Calif.; NAS, Seattle, Wash.; ASD, NSC, Guam.

For listing of available material and details of distribution see Naval Aeronautics Publications Index NavAer 00-500.

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SCOPE.

This manual contains the necessary information for safe and efficient operation of the RF-84F. These instructions provide you with a general knowledge of the aircraft, its characteristics, and specific normal, and emergency, operating procedures. Your flying experience is recognized, and therefore, basic flight principles are avoided.

SOUND JUDGMENT.

This manual 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.

PERMISSIBLE OPERATIONS.

The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations (such as asymmetrical loading) are prohibited unless specifically covered herein. Clearance must be obtained from MOAMA (MONEA) before any questionable operation is attempted which is not specifically permitted in this manual.

HOW TO BE ASSURED OF HAVING LATEST DATA.

Refer to T.O. 0-1-1-4A which is issued weekly and devoted solely to the listing of all current Flight Manuals, Safety Supplements, and Checklists. Its frequency of issue and brevity assures an accurate up to date listing of these publications.

STANDARDIZATION AND ARRANGEMENT.

Standardization assures that the scope and arrangement of all Flight Manuals are identical. The manual is divided into ten fairly independent sections to simplify reading it straight through or using it as a reference manual.

SAFETY SUPPLEMENTS.

Information involving safety will be promptly forwarded to you by Safety Supplements. Supplements covering loss of life will get to you in 48 hours by TWX, and those concerning serious damage to equipment within 10 days, by mail. The title page of the Flight Manual and the title block of each Safety Supplement should be checked to determine the effect they may have on existing supplements. You must remain constantly aware of the status of all supplements – current supplements must be complied with but there is no point in restricting your operation by complying with a replaced or rescinded supplement.

CHECKLISTS.

The Flight Manual contains only amplified checklists. Abbreviated checklists have been issued as separate technical orders — see the back of the title page for T.O. number and date of your latest checklist. Line items in the Flight Manual and checklists are identical with respect to arrangement and item number. Whenever a Safety of Flight Supplement affects the abbreviated checklist, write-in the applicable change on the affected checklist page. As soon as possible, a new checklist page, incorporating the supplement will be issued. This will keep hand-written entries of Safety of Flight Supplement information in your checklist to a minimum.

HOW TO GET PERSONAL COPIES.

Each flight crew member is entitled to personal copies of the Flight Manual, Safety of Flight Supplements, and Checklists. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your supply personnel — it is their job to fulfill your Technical Order requests. Basically, you must order the required quantities on the Publication Requirements Table (T.O. 0-3-1). Technical Orders 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver these publications to the flight crews immediately upon receipt.

FLIGHT MANUAL AND CHECKLIST BINDERS.

Loose leaf binders and sectionalized tabs are available for use with your manual. These are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part 1). Binders are also available for carrying your abbreviated checklist. These binders contain plastic envelopes into which individual checklist pages are inserted. They are available in three capacities and are obtained through normal Air Force supply under the following stock list numbers: 7510-766-4268, -4269, and -4270 for 15, 25, and 40 envelope binders respectively. Check with your supply personnel for assistance in securing these items.

WARNINGS, CAUTIONS, AND NOTES.

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

WARNING

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

CAUTION

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

Note

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

YOUR RESPONSIBILITY — TO LET US KNOW.

Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. However, we cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded through your Command Standardization Evaluation Group, to SMAMA, McClellan AFB, California ATTN: SMNEO.

T.O. 1F-84(R)F-1

TCTO IDENTIFICATION

The following TCTO's affecting aircraft operation are covered in this flight manual. Throughout the Manual the TCTO is identified by the last three numbers of the T.O. This is not a complete T.O. listing. Refer to the Basic Index (T.O. 0-1-1) for the complete listing of T.O.'s for these aircraft.

1F-84-263	Relocate Speed Brake Switch.
-267	Modify Fuel Liquidometer System.
-501	Installation of Emergency - Override switch for Air-Refueling.
-502	Install rheostat in non-flight instrument light circuit.
-503	Install AN ARW-3A Camera Mount.
-505	Install sequence jettison provisions of inboard stores.
-513	Installation of Landing Gear operated pneumatic compressor switch.
-521	Reactivate Fuel Deicing System.
-541	Relocate AN ARN-6 Radio Control Panel.
-545	Modify Instrument Panel.
-547	Modify left hand instrument panel.
-557	Installation of probe-drogue Air Refueling System.
-562	Installation of Hytrol Anti-Skid System.
1F-84-564	Modification of Inboard Jettison circuits.
-664	Revise Speed Brake pneumatic compressor sequence circuit.
-674	Provide dimming feature for Landing Gear handle warning light.
-697	Install emergency battery for AN ARC-34 Command Radio.
-702	Modify canopy squib circuit.
-726	Install new landing gear warning signal (beeper).
-728	Modify Drag Chute Control handle.
-755	Installation for type MC-1 seat cushion.
-763	Modify canopy to insure separation.
-778	Installation of type MA-1-AAV8 A altimeter.
-783	Installation of AN ARN-21, AN APX-25, Radio Compass Indicator (RCI) and Radio Altimeter (RA) provisions.
-786	Installation of remote channel frequency indicator and UHF manual tuner.
-801	Incorporate zero-altitude capability to pilot escape system and install seat man separator.

SAFETY SUPPLEMENT SUMMARY

Safety Supplements are numbered with a prefix SS before -1 of the Flight Manual technical order number followed by a number starting with 1 and continuing numerically for each succeeding supplement. The supplements you receive should follow in sequence and if you find you are missing one, check the Safety Supplement Index T.O. 0-1-1-4A to see if it was issued, and, if so, is still in effect. That supplement may have been replaced or rescinded before you received your copy. If it is still active, see your Publications Distribution Officer and get your copy. It should be noted that a supplement number will never be used more than once. An example of the numbered sequence is as follows: -SS-1-1, -SS-1-2, -SS-1-3, etc.

SAFETY SUPPLEMENTS IN THIS REVISION

Number	Date	Short Title	Disposition
1ET	2 May 1960	Zero Delay Lanyard	Page 3-14
-1	30 Aug 1960	Prohibited Maneuvers	Page 5-8
-2	20 Feb 1961	Takeoff and Landing Cross Wind Chart	Page A2-20, B2-14
-3	3 May 1961	Electrical Fires and Smoke Elimination	Page 3-11
-4	12 June 1961	AC Generator Failure	Page 3-22
-5	14 June 1961	Autopilot Restrictions	Page 2-12, 4-27, 5-7
-6	24 July 1961	Air Starts	Page 3-4
-7	8 Jan 1962	Airspeed and Altitude for Ejection	Page 3-5, 3-14
-8	29 Jan 1962	Anti G Suit Instructions	Page 2-5, 3-26
-9	12 Mar 1962	Exterior Inspection Starter Flash	Deleted in Revision
-10	4 May 1962	Canopy Jettison	Page 3-26
-11	8 May 1962	Emergency Procedures	Section III
-13	9 July 1962	Landing Gear Warning System	Page 1-44, 2-12
-14	9 July 1962	Fuel System Management	Page 2-12, 7-6
-15	15 Aug 1962	Performance Data	Page A-1
-17	14 Nov 1962	Slippery Runway Distance Factors	Replaced by -22
-19		Restrict Intentional Spins	Replaced by -20
-20	27 Feb 1963	Restrict Intentional Spins	Page 5-8
-21		Modified Escape System	Replaced by -23
-22	6 Mar 1963	Slippery Runway Distance Factors	Replaced by -25
-23	8 Mar 1963	Modified Escape System	Page 1-53, 1-59, 3-5
-24	14 Mar 1963	Throttle Linkage Failure	Page 3-13
-25	18 Mar 1963	Slippery Runway Distance Factors	Page A7-5, B7-5
-26	21 June 1963	Tumbling After Ejection	Replaced by -37

Number	Date	Short Title	Disposition
-27	26 June 1963	Purging Fuel Transfer Lines	Replaced by -32
-28	26 June 1963	Trajectories of Jettisoned External Tanks	Page 5-8
-29	2 July 1963	TACAN False Indications	Page 4-16
-30	2 July 1963	Routing Anti-G Suit Hose	Replaced by -31
-31	12 July 1963	Routing Anti-G Suit Hose	Page 2-5
-32	22 July 1963	Purging Fuel Transfer Lines	Replaced by -33
-33	23 Jul 1963	Purging Fuel Transfer Lines	Page 2-9, 2-12
-34	23 July 1963	Taxi Restriction With Emergency Brakes	Page 3-26
-35	26 July 1963	Inflight Refueling Restrictions	Replaced by -36
-36	7 Aug 1963	Inflight Refueling Restrictions	Page 4-25
-37	6 Sep 1963	Tumbling After Ejection	Page 1-57
-38	11 Sep 1963	Revise Alternate Start Procedure	Page 2-6
-39	13 Sep 1963	Redesignate Minimum Speed As Stall Speed	Page 6-4
-42	1 Nov 1963	Main Tank Boost Pump Failure	Page 3-18
-46	13 Nov 1964	Zero Lanyard Engagement	Page 3-14

SAFETY SUPPLEMENT STATUS

This portion to be filled in by you when you receive your Flight Manual and to be added to as you receive additional supplements. Refer to Safety Supplement Index T.O. 0-1-1-4A for latest information if any questions arise.

Number	Date	Short Title	Disposition
-41	11 Oct 1963	Emergency Ejection Altitude	
-45	16 Sep 1964	Manual Canopy Jettison	
-47	13 Dec 1965	Escape System Limitations	

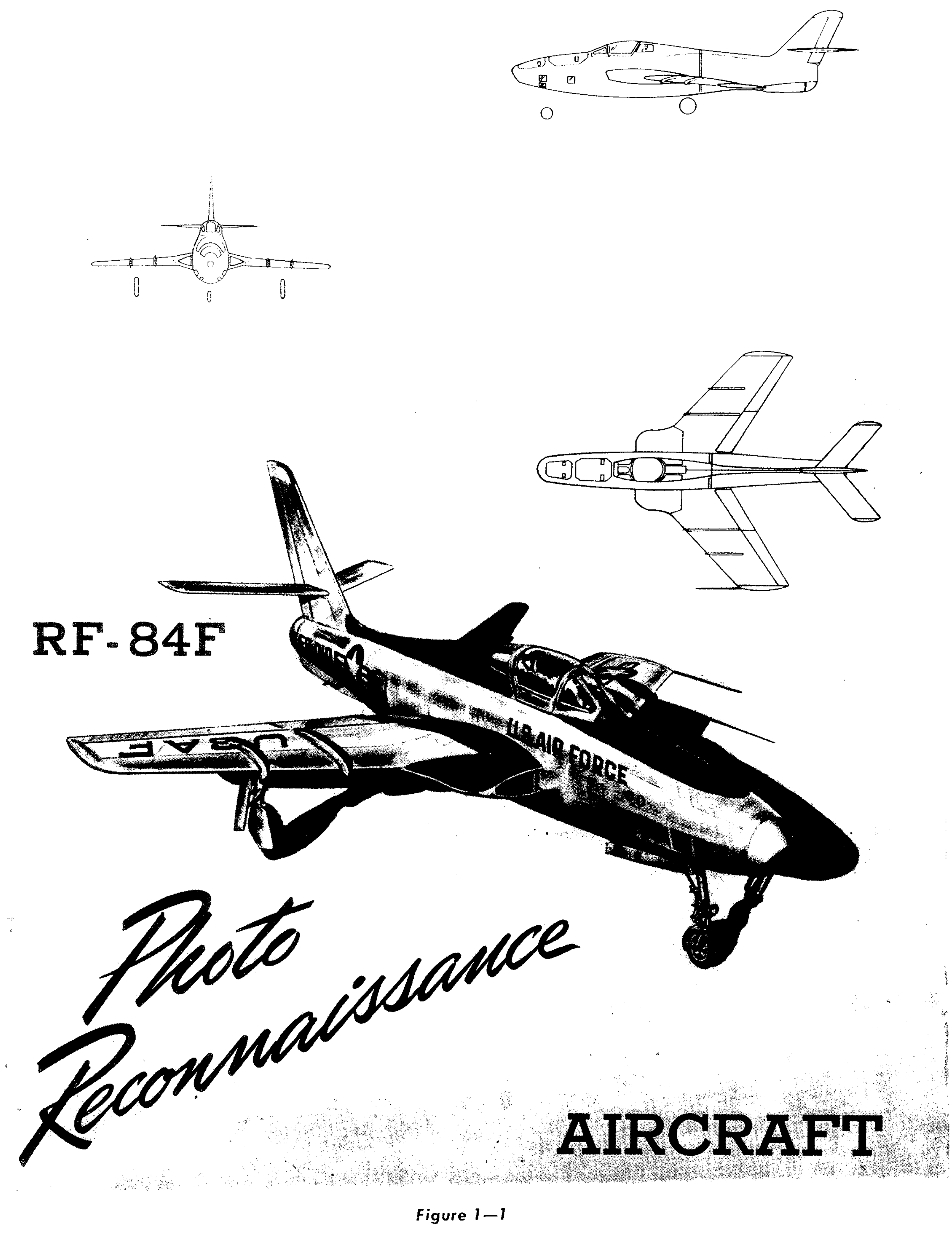
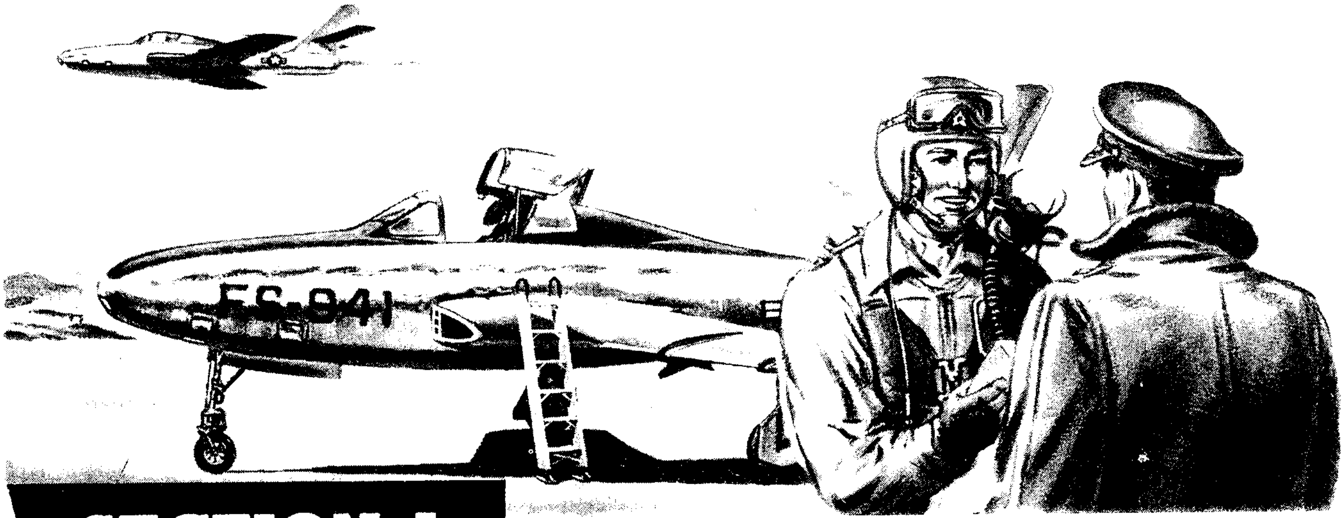


Figure 1—1



SECTION I

Description

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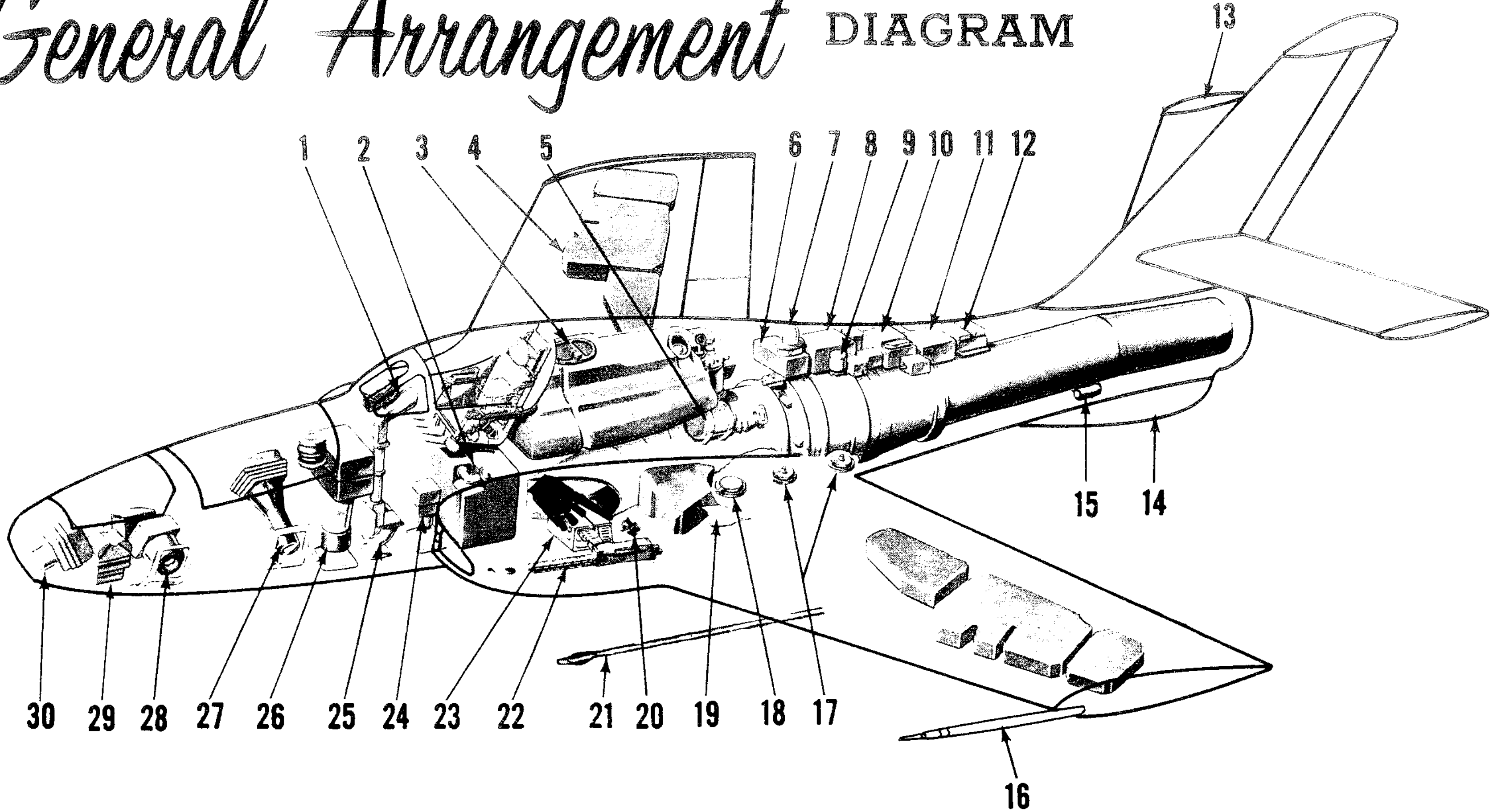
General Arrangement DIAGRAM



1. Gun Sight
2. Forward Fuel Tank
3. Main Fuel Tank
4. Wing Fuel Tank
5. Oil Tank
6. Radio Compass Equipment
7. Radio Compass Loop Antenna
8. Command Radio Equipment
9. Directional Indicator Equipment
10. IFF SIF Radar Equipment
11. Auto Pilot Equipment
12. APW 11 Radar Equipment
13. Command Radio Antenna
14. Drag Chute Compartment
15. Emergency Hydraulic Compartment
16. Pitot Tube
17. APW 11 Radar Antenna
18. IFF SIF Radar Antenna
19. Air Refueling Receiver UNMODIFIED
20. Pneumatic Compressor
21. Air Refueling Probe MODIFIED
22. Guns
23. Ammunition Tanks
24. Camera Photo Cell
25. Viewfinder
26. Prime Vertical Camera
27. Aft Left Oblique Camera
28. Mid L and R Oblique Camera
29. Forward Vertical Camera
30. Forward Oblique Camera

Figure 1—2

THE AIRCRAFT.

The RF-84F (Thunderflash) is a single place, tactical photo-reconnaissance fighter designed by Republic Aviation Corporation. This version of the Thunderstreak is a high performance aircraft with swept-back wings and empennage designed for flight in the subsonic and sonic speed ranges. The aircraft is equipped with a high-thrust turbojet engine and lends itself readily to high and low altitude photo-reconnaissance missions. It can carry exceptionally large external loads and yet retain its high speed performance and stability.

Outstanding design features are air refueling, single-point ground refueling, automatic pilot, pneumatic-type emergency landing gear system and hydraulically actuated controls.

Pilot comfort is assured by providing greater quantities of air at comfortable temperatures and cockpit pressurization at lower effective altitude. Visibility is improved by use of a double-wall canopy, a dry air circulating system for anti-fogging purposes, and a flat front windshield.

AIRCRAFT DIMENSIONS.

The over-all dimensions of the aircraft under normal conditions of gross weight, tire, and strut inflation are approximately as follows.

LENGTH (Including Stabilizers)	47.5 FT
WING SPAN	33.5 FT
HEIGHT (to top of fin)	15.0 FT
TREAD	20.4 FT
GROUND CLEARANCE	
STABILIZER TIP	8.4 FT
WING TIP	3.7 FT
PYLON TANK	1.0 FT

AIRCRAFT GROSS WEIGHT.

The average aircraft gross weight, including pilot, a full load of ammunition and fully serviced with fuel and oil is as follows:

Clean aircraft	18,850 LB.
Clean aircraft plus two 230 gallon tanks	22,400 LB.
Clean aircraft plus two 450 gallon tanks	25,400 LB.
Clean aircraft plus four 230 gallon tanks	25,900 LB.

Note

The above weights are average and are not to be used for Flight Planning. For more exact weight information refer to T.O. 1-1B-40 Weight and Balance Data.

AIRCRAFT BLOCK NUMBERS.

Production changes that affect the aircraft and, or its equipment are identified by the addition of a block number to the aircraft model designation. Suffix letters are added to the block number to identify the aircraft manufactures. In cases where it is of definite value to the pilot in the operation of the aircraft, a code number is used to identify specific paragraphs, illustrations, and procedural steps **not applicable to all aircraft**. The code number appears at the top right-hand corner of paragraphs, in illustrations, or after the title of illustrations, in the vicinity of an applicable part of an illustration and in procedural steps.

USAF Aircraft Serial No.	*Aircraft Model*	*Code*
51-1839 thru 51-1873	RF-84F-5	5
51-1874 thru 51-1938	RF-84F-10	10
51-1939 thru 51-1958, 51-11250 thru 51-11294	RF-84-15 and -16	15
51-11295 thru 51-11297, 51-16996 thru 51-17002, 52-7229 thru 52-7253	RF-84R-20 and -21	20
51-1832, 51-17003 thru 51-17058, 52-7279 thru 52-7377	RF-84F-25 and -26	25
52-7378 thru 52-7472	RF-84F-30 and -31	30
52-7473 thru 52-7475, 52-8717 thru 52-8766, 53-7521 thru 53-7532	RF-84F-35 and -36	35
53-7533 thru 53-7559, 53-7584 thru 53-7616	RF-84F-40 and -41	40
53-7560 thru 53-7583, 53-7617 thru 52-7697	RF-84F-45 and -46	45

MAIN DIFFERENCE TABLE

		RF-84F 5 thru 45		F-84F 25 thru 70	
MISSION		TACTICAL PHOTO-RECONNAISSANCE FIGHTER		FIGHTER BOMBER	
ENGINE AIR INLET		WING ROOTS		FUSELAGE NOSE	
ENGINE SCREENS		5 thru 35 Manual Control	40 thru 45 Automatic Operation	25 thru 55 Manual Control	60 and LATER Automatic Operation
REFUELING	AIR	UNMODIFIED Receiver	MODIFIED Probe	Receiver	
	GROUND	SINGLE POINT		25 and 30 Individual Tanks	35 thru 70 Single Point
BRAKES	NORM	HYDRAULICALLY POWERED		HYDRAULICALLY BOOSTED	
	EMERG	PNEUMATIC HYDRAULIC ACCUMULATOR		UNBOOSTED	
		SOME AIRCRAFT Anti-Skid Brake System			
FUEL FILTER DEICING		UNMODIFIED Removed	MODIFIED Operative	Removed	
GUN SIGHT		UNMODIFIED N-3C	MODIFIED N-9	APG-30	
EJECTION SEAT		UNMODIFIED Cartridge Catapult	MODIFIED Rocket Catapult and Seat—Man Separator	UNMODIFIED Cartridge Catapult	MODIFIED Rocket Catapult and Seat—Man Separator
		5 thru 25 Manual Height Adjust	30 thru 45 Electric Height Adjust	25 thru 50 Manual Height Adjust	55 thru 70 Electric Height Adjust
COMMAND RADIO		5 thru 30 AN/ARC-33	35 and LATER AN/ARC-34 With Standby Battery	25 thru 55 AN/ARC-33	60 and LATER AN/ARC-34 With Standby Battery

Figure 1—3.

Differences in aircraft configuration may also be brought about by Modification through compliance with Technical Orders. OPERATING PERSONNEL MUST BECOME FAMILIAR WITH SUCH TECHNICAL ORDERS. When a modification is applicable to all aircraft the words **MODIFIED** or **UNMODIFIED** will appear where the code appears. Aircraft modified by block number are identified by the prefix letter **M**, bracketed with the applicable code number or numbers, i.e. (**M 5**) or (**M 5 10**). Unmodified aircraft of particular block numbers are identified by the prefix letter **U**, bracketed with the applicable code number (**U 5**) or (**U 5 10**).

ENGINE.

The aircraft is powered by an axial-flow jet-propulsion engine, AF Model J65-3, -7 or -7D. The J65-3 engine is rated at 7,220 pounds thrust while the J65-7 and -7D is rated at 7,800 pounds thrust. Air is ducted to the engine from air intakes in the inboard leading edge of each wing and compressed progressively through the 13-stage compressor section of the engine and discharged through a two stage-turbine. The air inlet to the engine compressor is provided with a retractable screen to prevent foreign objects from being drawn into the compressor section and damaging the engine during engine operation. A combustion starter, ignition and fuel priming systems are provided for starting the engine. Supplementary thrust for take-off is supplied by externally mounted ATO units.

ENGINE FUEL CONTROL SYSTEM.

Fuel flow to the engine is supplied from the aircraft fuel system, controlled by the throttle and regulated and distributed by the fuel control system. The fuel control system (figure 1–8) consists of an engine driven booster fuel pump, a main dual element gear type pump, a speed-density fuel control unit, six flow dividers, 36 full nozzles and two fuel primers. The integral dual element pump is driven through the accessory gear box, and incorporates shear sections for each of the gear elements. In the event one pump element fails, it will not affect the operation of the remaining pump element, and sufficient fuel will be available for 100 per cent RPM operation at sea level on a 37.8°C (100°F) day. The emergency fuel system, which must be selected manually, by-passes the RPM-density regulating provisions of the fuel control.

Main Fuel Control.

The fuel control is basically an engine-speed governing control which by-passes main-fuel-pump output when necessary so as to maintain a fixed engine RPM for a given throttle position. The control prevents overspeeding beyond the maximum governed speed, preserves constant engine speed by compensating for changes in air density with changes in altitude, provides increased fuel during acceleration at such a rate as to increase the acceleration while avoiding surge and preventing excessive exhaust gas temperatures, limits fuel flow during engine deceleration in such a manner as to increase deceleration rate while avoiding flame-outs, and provides a means for selection of emergency operation whenever a failure occurs in the main-fuel-control system. As engine RPM tends to exceed prescribed limits the speed governor metering valve acts to restrict further fuel flow. A metering valve is also actuated by altitude and air temperature sensing bellows to provide correct fuel flow for these variables. A fuel control by-pass valve which permits excessive fuel to return to the inlet side of the fuel pump is also provided. This by-pass valve operates from two sources. From one source, it protects against excessive compressor pressure rise by sensing inlet and outlet compressor pressures which are balanced against a spring-load. When compressor pressure rise is excessive, the fuel control by-pass opens to permit fuel "runaround," reducing fuel flow to the engine. From the other source the fuel control by-pass will also open to permit excess fuel "runaround" to compensate for changes in engine RPM temperature and ambient air pressure. In this instance the fuel control by-pass valve is actuated by a differential pressure diaphragm which senses fuel pump output pressure on one side of the diaphragm and metered fuel pressure on the other side of the diaphragm and adjusts fuel flow to maintain constant RPM despite changes in altitude, air pressure and speed of the aircraft.

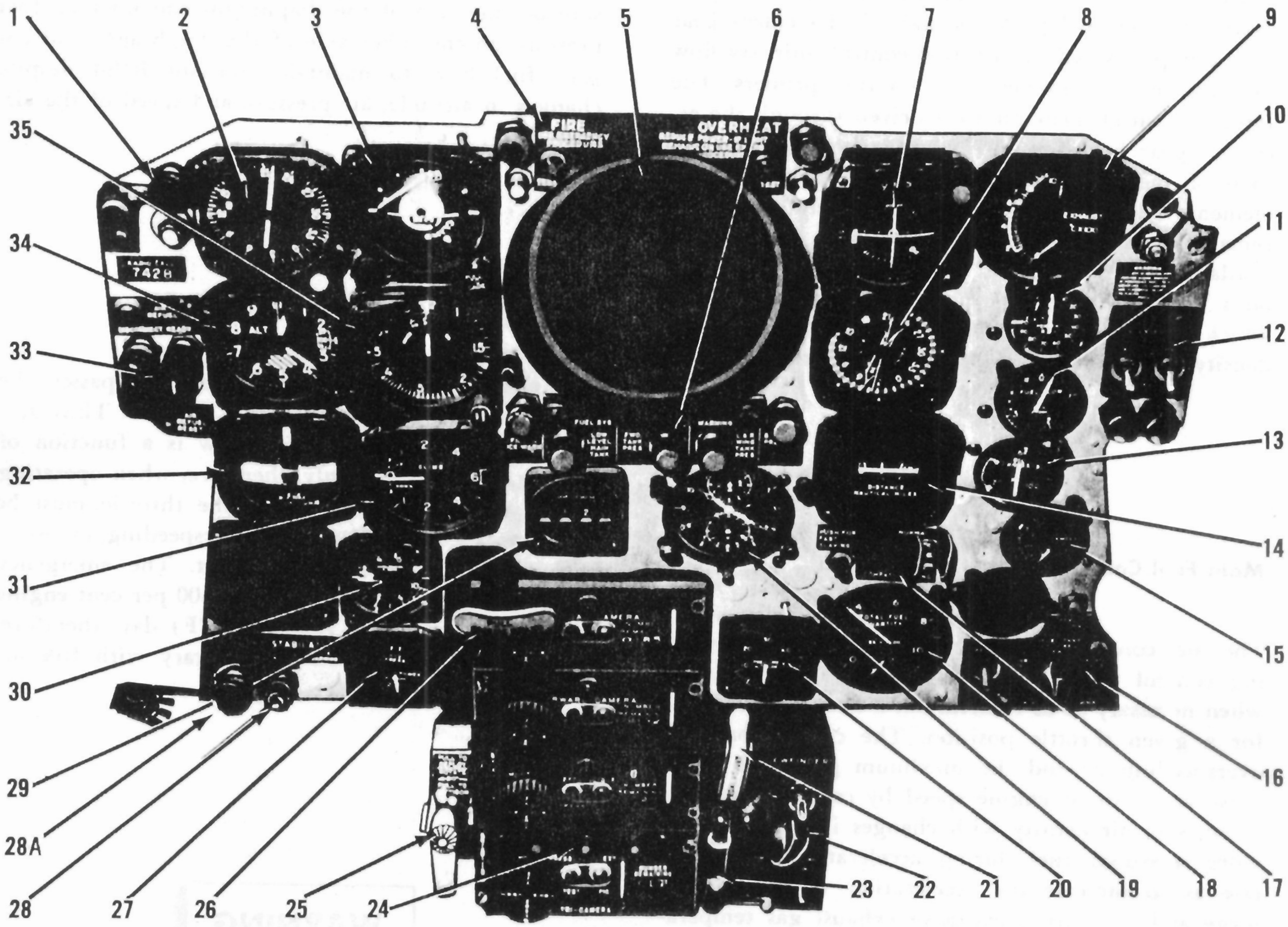
Emergency Fuel Control.

The emergency fuel control merely by-passes the normal fuel control compensating features. Thus during emergency operation fuel flow is a function of throttle control travel only, therefore, when operating on the emergency fuel system, the throttle must be moved with caution to prevent overspeeding, excessive exhaust temperatures, or flame-out. The emergency fuel control is adjusted to provide 100 per cent engine RPM at sea level on a 37.8°C (100°F) day; therefore, available full throttle RPM will vary with free air temperature and altitude.

WARNING

The emergency system is designed for in-flight failure of the normal system and provides a constant fuel flow for a given throttle setting. As altitude increases, fuel flow in excess of that required to maintain normal engine operation will produce engine overspeed conditions. Therefore, if the pilot should elect to go from the normal to the emergency system above 6,000 feet, he must throttle back before the transfer is made to avoid engine overspeeding or flame-out, caused by excessively rich mixtures. Below 6,000 feet with the RPM no less than 85 per cent transfer to the emergency system can be made without reducing the throttle.

MAIN INSTRUMENT PANEL
(TYPICAL)



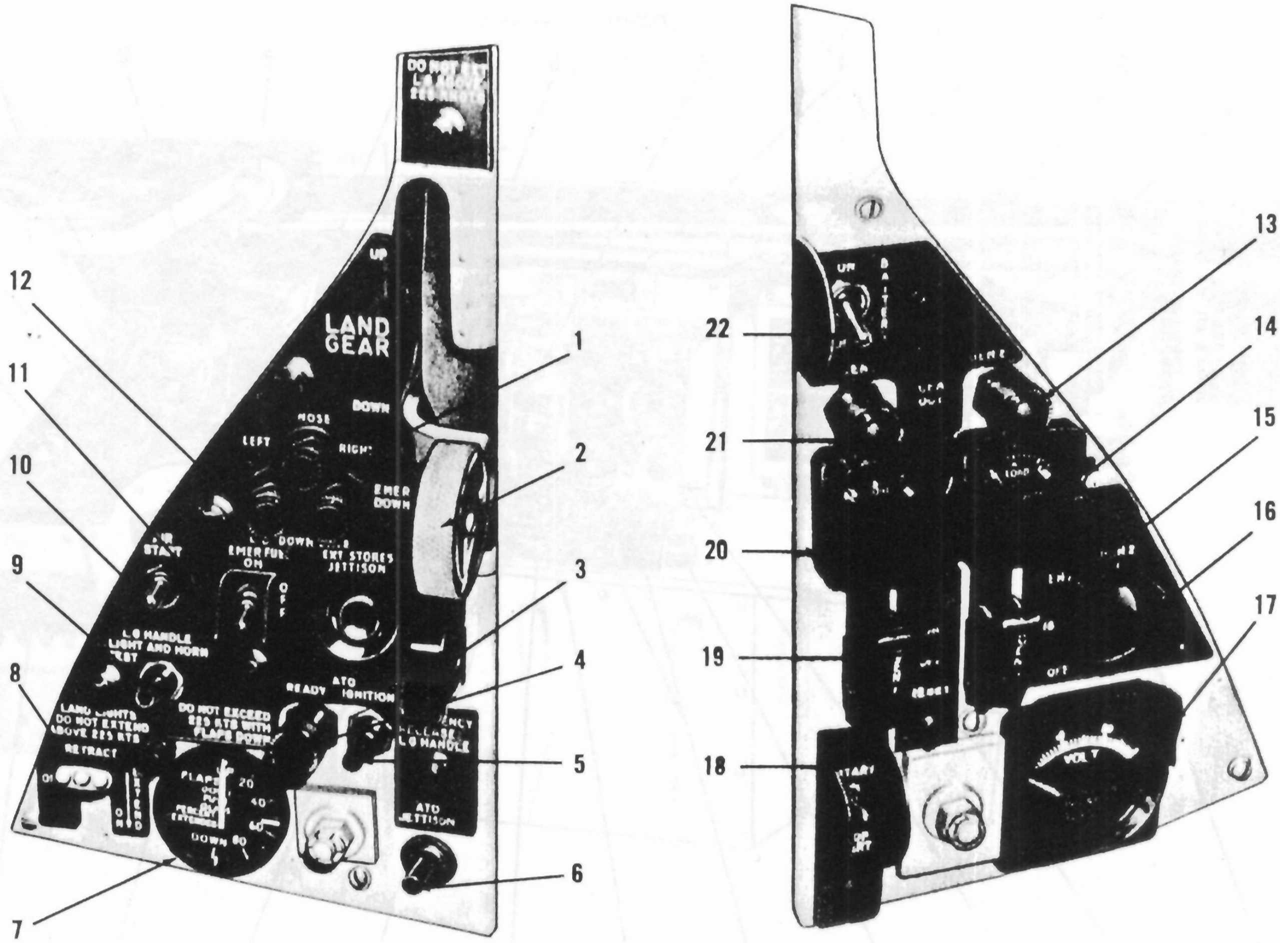
1. Canopy Open Indicator Light
2. Heading Indicator
3. Attitude Indicator
4. Engine Fire and Overheat Warning (See figure 1-26)
5. Viewfinder
6. Fuel System Warning Lights (See figure 1-6)
7. Course Indicator
8. Radio-magnetic Indicator
9. Exhaust Gas Temperature Indicator
10. Tachometer
11. Fuel Flow Indicator
12. Inverter Control Panel
13. Oil Pressure Gage
14. Tacan Range Indicator
15. Camera Compartment Temperature Indicator
16. Fuel Quantity Selector
17. Directional Indicator Control Panel
18. Fuel Quantity Indicator
19. Clock
20. Cockpit Altimeter
21. Rudder Pedal Adjusting Control
22. Viewfinder Controls (See figure 4-21)
23. Surface Control Lock
24. Camera Station Control Panels (See figure 4-21)
25. Drag Chute Handle (See figure 1-25)
26. Emergency Brake Handle
27. Utility Power Hydraulic Pressure Gage
28. Mechanical Advantage Shifter Indicator

28A. Emergency Fuel Indicator Light

29. Remote Channel Indicator
30. Accelerometer
31. Vertical Velocity Indicator
32. Turn and Slip Indicator
33. Air Refuel Controls (See figure 4-15)
34. Altimeter
35. Airspeed and Mach Number Indicator (MODIFIED)

Figure 1-4

AUXILIARY INSTRUMENT PANELS TYPICAL

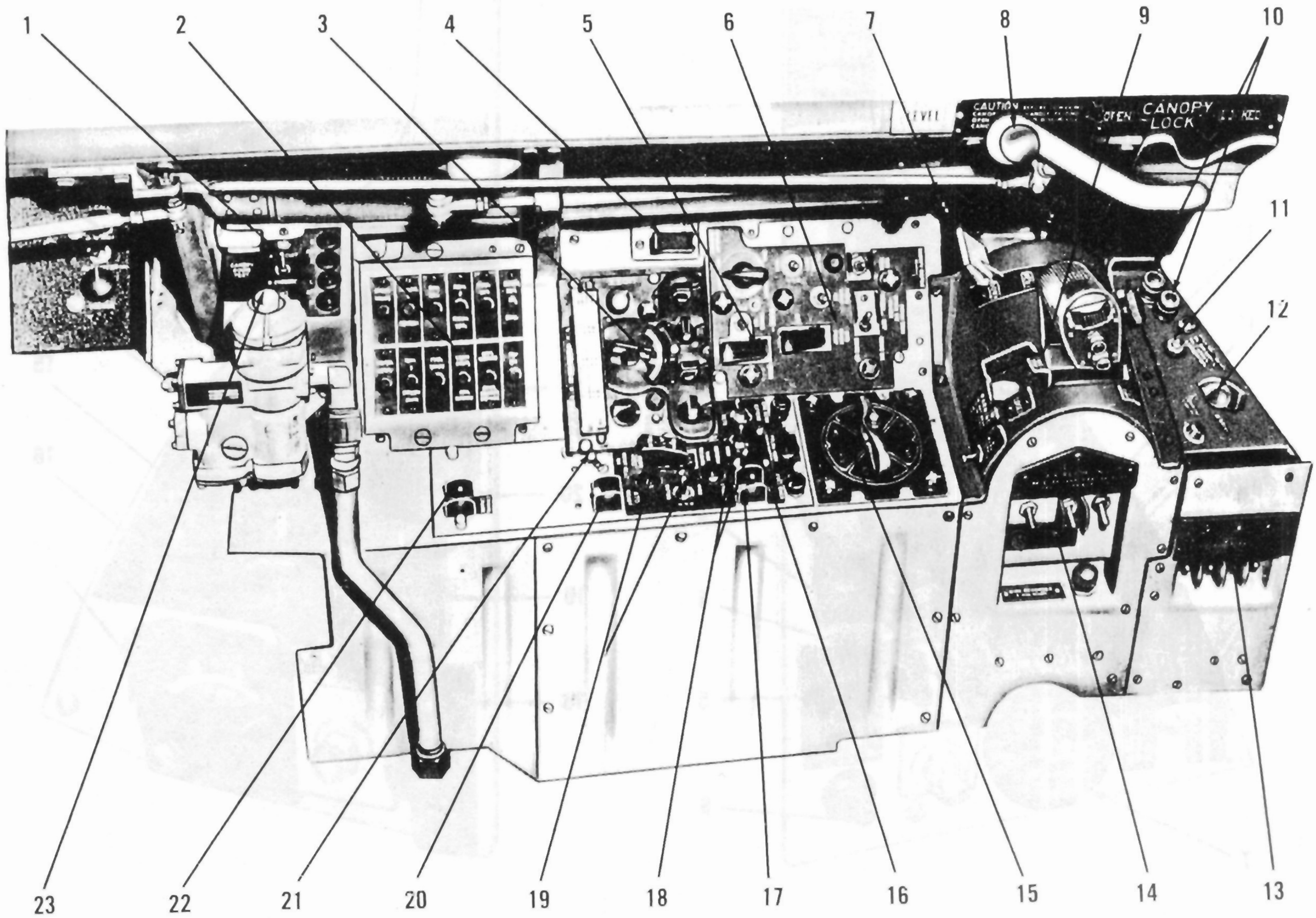


1	Landing Gear Position Indicator Lights	12	External Stores Jettison Switch
2	Landing Gear Selector Handle	13	No. 2 Generator-Out Indicator Light
3	Emergency Landing Gear Release Switch	14	No. 2 Loadmeter
4	ATO Ready Indicator Light	15	No. 2 Generator Switch
5	ATO Ignition Button	16	Voltmeter Selector Switch
6	ATO Jettison Button	17	Voltmeter
7	Wing Flap Position Indicator	18	Starter Switch
8	Landing Light Switch	19	No. 1 Generator Switch
9	Landing Gear Warning Test Button	20	No. 1 Loadmeter
10	Air Start Switch	21	No. 1 Generator-Out Indicator Light
11	Emergency Fuel Switch*	22	Battery Switch

*Aircraft not incorporating [811] have the EMERG FUEL SWITCH located on the left console.

Figure 1-5

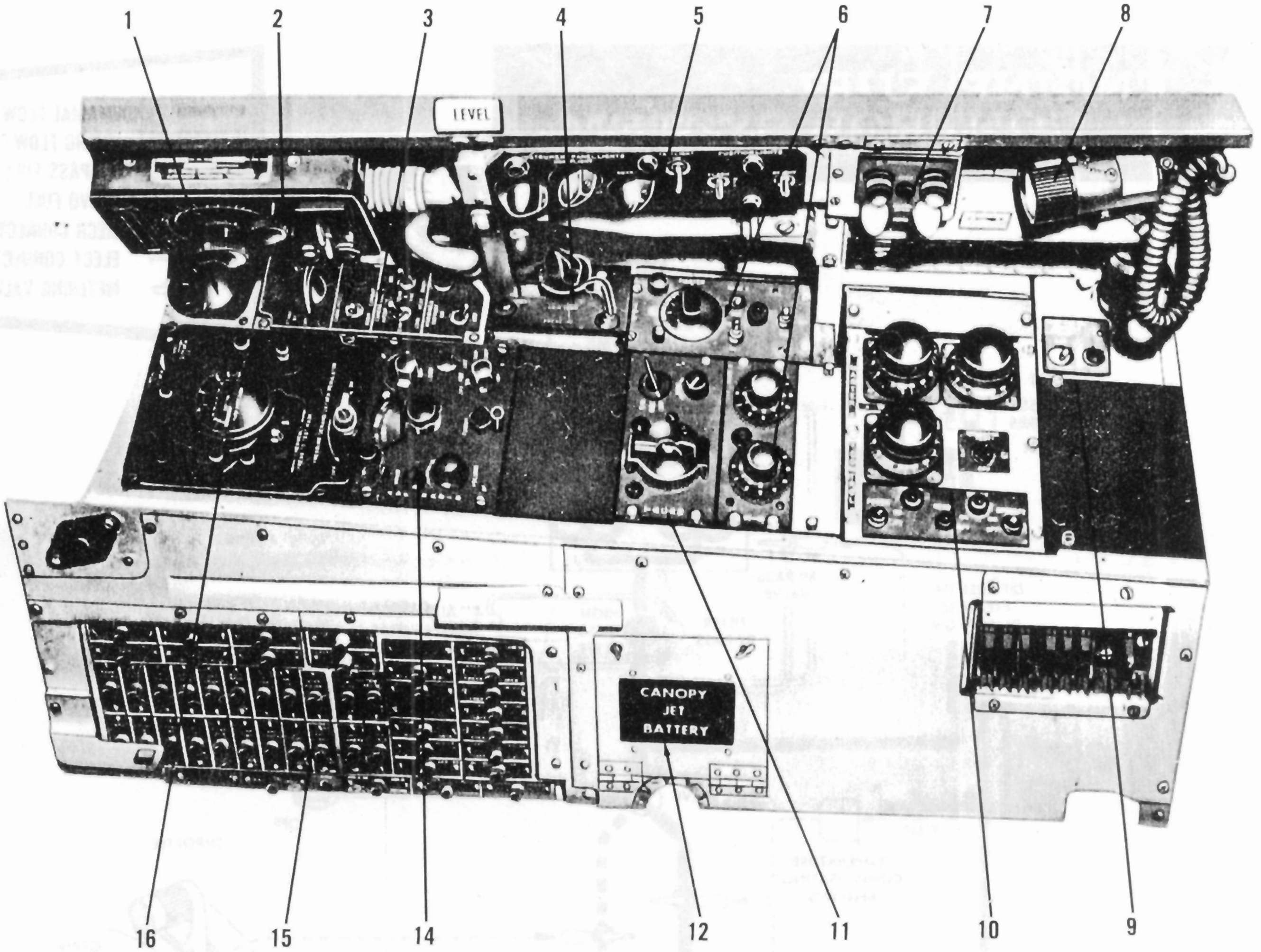
LEFT CONSOLE (TYPICAL)



1. Canopy Squib Test Panel (See figure 1-28)
2. Circuit Breaker Panel (See figure 1-19)
3. Command Radio (See figure 4-6)
4. Spoiler Shutoff Switch
5. Gun Controls (See figure 4-19)
6. Flight Control Panel (See figure 1-24)
7. Canopy Dry Air Switch
8. Canopy Lock Lever (See figure 1-27)
9. Throttle Quadrant (See figure 1-10)
10. Rudder & Aileron Neutral Trim Indicator Lights
11. Emergency Bus Circuit Breaker Test Lights
12. Console Lights Rheostat
13. Fuel Shutoff Valves Switches (See figure 1-15)
14. Air Refueling Controls (See figure 4-15)
15. Fuel System Control Panel (See figure 1-8)
16. Engine Screen Switch
17. Canopy Manual Jettison Switch (MODIFIED BY | 811 | See figure 1-27)
18. Pylon Tank Air Pressure Switches (See figure 1-9)
19. Pylon Jettison Switches
20. ATO Ready Switch (See figure 1-12)
21. Pitot Heat Switch
22. Engine Crank Switch
23. Anti-G Valve

Figure 1-6

RIGHT CONSOLE (TYPICAL)



1. Liquid Oxygen Quantity Gage
2. C-4 Control Panel (See figure 4-21)
3. Cockpit Temperature and Pressure Control (See figure 4-3)
4. Defroster Control
5. Interior & Exterior Lights Control Panel (See figure 4-10)
6. IFF/SIF Control Panels (See figure 4-7)
7. Photo Door Emergency Control Panel (See figure 4-21)
8. Cockpit Light
9. Latitude and Departure Switch
10. Master Camera Control (See figure 4-21)
11. Tacan Control Panel (See figure 4-9)
12. Canopy Jettison Standby Battery
13. Autopilot Control Panel (not shown) (See figure 4-16)
14. Automatic Direction Finder Control Panel
15. Circuit Breaker Panel (See figure 1-19)
16. Oxygen Regulator (See figure 4-11)

Figure 1-7

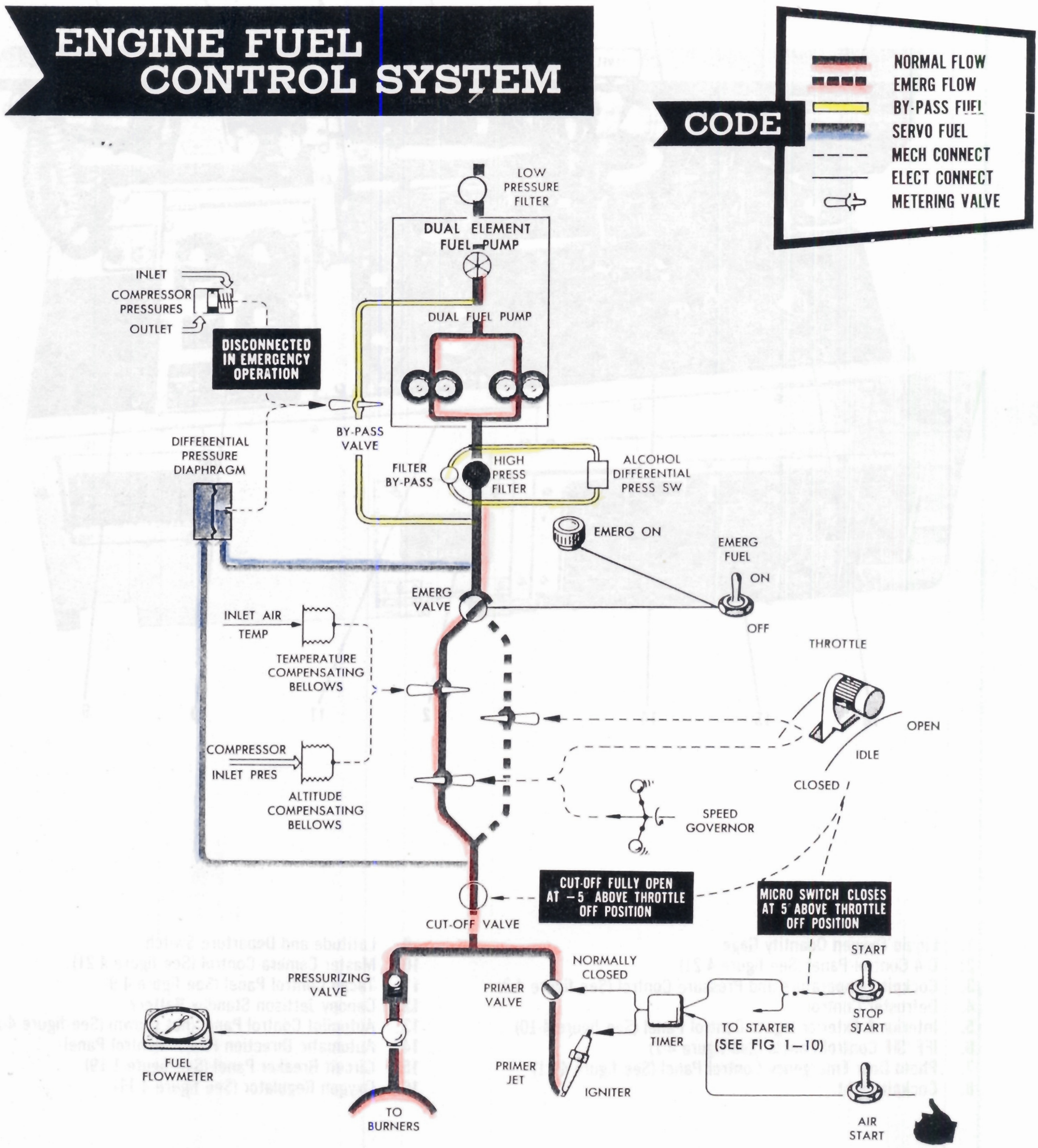

ENGINE FUEL CONTROL SYSTEM
CODE
NORMAL FLOW
EMERG FLOW
BY-PASS FUEL
SERVO FUEL
MECH CONNECT
ELECT CONNECT
METERING VALVE
LOW PRESSURE FILTER
DUAL ELEMENT FUEL PUMP
INLET
COMPRESSOR PRESSURES
OUTLET
DUAL FUEL PUMP
DISCONNECTED IN EMERGENCY OPERATION
DIFFERENTIAL PRESSURE DIAPHRAGM
BY-PASS VALVE
FILTER BY-PASS
HIGH PRESS FILTER
ALCOHOL DIFFERENTIAL PRESS SW
EMERG ON
EMERG FUEL
ON
OFF
EMERG VALVE
INLET AIR TEMP
TEMPERATURE COMPENSATING BELLOWS
THROTTLE
OPEN
IDLE
CLOSED
COMPRESSOR INLET PRES
ALTITUDE COMPENSATING BELLOWS
SPEED GOVERNOR
CUT-OFF FULLY OPEN AT -5° ABOVE THROTTLE OFF POSITION
MICRO SWITCH CLOSES AT 5° ABOVE THROTTLE OFF POSITION
CUT-OFF VALVE
START
NORMALLY CLOSED
PRESSURIZING VALVE
PRIMER VALVE
STOP
START
TO STARTER (SEE FIG 1-10)
TIMER
FUEL FLOWMETER
PRIMER JET
IGNITER
TO BURNERS
AIR START

Figure 1—8

FUEL SYSTEM *Control Panel* (TYPICAL)

Note

On aircraft with [811] complied with, the emergency fuel switch is located on the left auxiliary instrument panel.

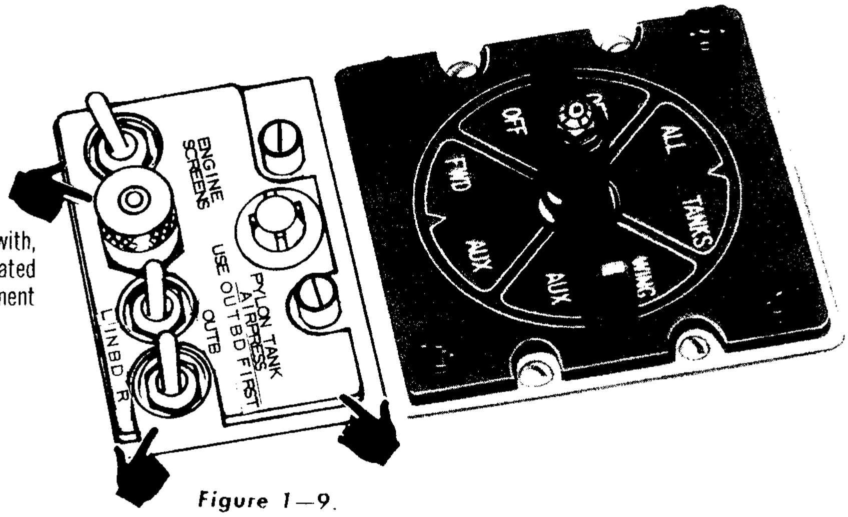


Figure 1—9.

Emergency Fuel Switch.

The emergency fuel switch (figure 1—5) is a two position switch, placarded EMER FUEL ON, and OFF. The ON position powered by the DC primary bus, energizes the fuel transfer solenoid valve which directs the flow of fuel through the emergency system. The OFF position deenergizes the fuel transfer solenoid valve, and since it is spring-loaded to the normal position, the fuel flow is directed through the normal system.

Note

- On aircraft modified by [811] the emergency fuel switch is located on the left auxiliary instrument panel.
- On aircraft not modified by [811] the emergency fuel switch is located on the left console.

WARNING

On 5 thru 25 aircraft the ON position is aft, and on 30 and later aircraft, the ON position is forward.

Emergency-On Warning Light.

The emergency-on warning light (28A, figure 1—4) is a red press to test light on the fuel system warning panel placarded EMER FUEL and powered by the DC primary bus. Illumination of the light indicates the emergency fuel system has been selected.

THROTTLE.

Engine thrust is selected by the throttle (figure 1—10). The extreme positions of throttle travel are placarded CLOSED and OPEN. In the CLOSED (full aft) position fuel flow through the fuel control is mechanically cut-off and the engine ignition and priming circuits from the starter switch are opened. The throttle position does not affect the air start button or engine crank switch circuits. When the throttle lever is five degrees or more forward of the CLOSED position, the fuel cut-off is fully open and the ignition and priming circuits to the start switch closed, arming the switch. The throttle quadrant is provided with an IDLE STOP which prevents inadvertent return to the CLOSED position. In order to return the throttle lever to CLOSED, the IDLE STOP must be depressed. From the IDLE STOP to OPEN (full forward) the full range of engine operation is available. The throttle grip incorporates the speed brake switch, microphone button, and an air refueling nozzle disconnect button. Complete description of these controls is provided under appropriate headings.

THROTTLE QUADRANT

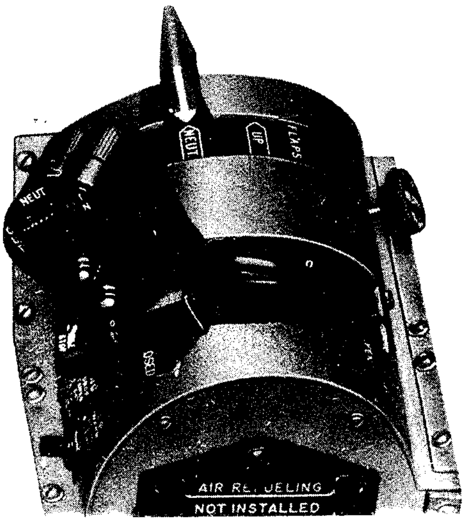


Figure 1—10.

PRIMING SYSTEM.

The engine fuel system prevents fuel from passing to the fuel distributors until a system pressure of approximately 100 PSI is attained at the fuel control pressurizing valve. Fuel pressure for the priming system, during a normal start on the aircraft's battery, is provided by the engine driven booster pump and the main fuel pump, since the fuel tank booster pumps are energized from the secondary bus, and power to these pumps is not available from the aircraft's battery. The rapid accelerating starter will turn the engine up to a high enough RPM for the engine driven booster pump and main booster fuel pump to deliver sufficient fuel pressure for engine starting. If a start is made using an external power source the secondary bus is energized and the fuel tank booster pumps will supply pressure to the engine driven booster pump and the main fuel pump. Fuel is supplied through a normally closed solenoid-operated primer valve to two primer jets around the engine, where it is ignited by igniters. The primer valve is energized to open from the primary bus through an automatic timing device. The timer opens the primer valve for a period of 15 (±3) seconds when the starter switch is actuated and the throttle is in the idle or above position. The timer is also energized to open the primer valve for 15 (±3) seconds if the air start switch is depressed, regardless of the throttle position; however, fuel will not flow to the engine unless the fuel cut-off is opened by having the throttle five degrees or more forward of the CLOSED position. In either case, the timer circuit can be interrupted so as to close the primer valve by actuating the starter switch to the STOP START position. Once the timer cycle has started, the 15 (±3) second limit must elapse before another 15 (±3) second cycle can be started, unless the timer circuit is interrupted by placing the starter switch in the STOP START position.

IGNITION.

The fuel-air mixture in the engine is ignited by an automatic ignition system which incorporates two igniters. Once the engine is started the ignition system is no longer used, as burning in the combustion chamber is continuous. Power for ignition is supplied by two high tension vibrators which derive their power from the primary bus. The system incorporates a timer device which supplies ignition continuously for 15 (±3) seconds; controls are also provided for interrupting the 15 (±3) second ignition cycle. The ignition system is operated by the start or air start switches which are discussed under starting system.

STARTING SYSTEM.

The starting system (figure 1—11) consists of an ignition and priming system with an automatic timer, a starting turbine for torque and the necessary controls. The starter is a self-contained combustion type turbine that drives the engine through reduction gears up to approximately 24 percent RPM in 3 to 3½ seconds. Sufficient air for one start is supplied by an air storage bottle. The air is replenished automatically by the pneumatic compressure after start. Sufficient fuel for one start is supplied by the starter fuel accumulator. The fuel is replenished by the aircraft fuel system when the booster pumps are operated. The fuel and air mixture is fed to the starter and ignited. When the turbine reaches its maximum RPM a centrifugal switch cuts off the fuel and air supply. If for any reason the centrifugal cut-out switch fails to operate, the pressure switch will deenergize the starter circuit when the pressure in the combustion chamber falls below the pressure switch cut-off setting (73 to 78 PSI) due to the exhaustion of the fuel. The engine ignition and primer operate concurrently with the starter (or without the starter for air start). The

ENGINE STARTER SYSTEM

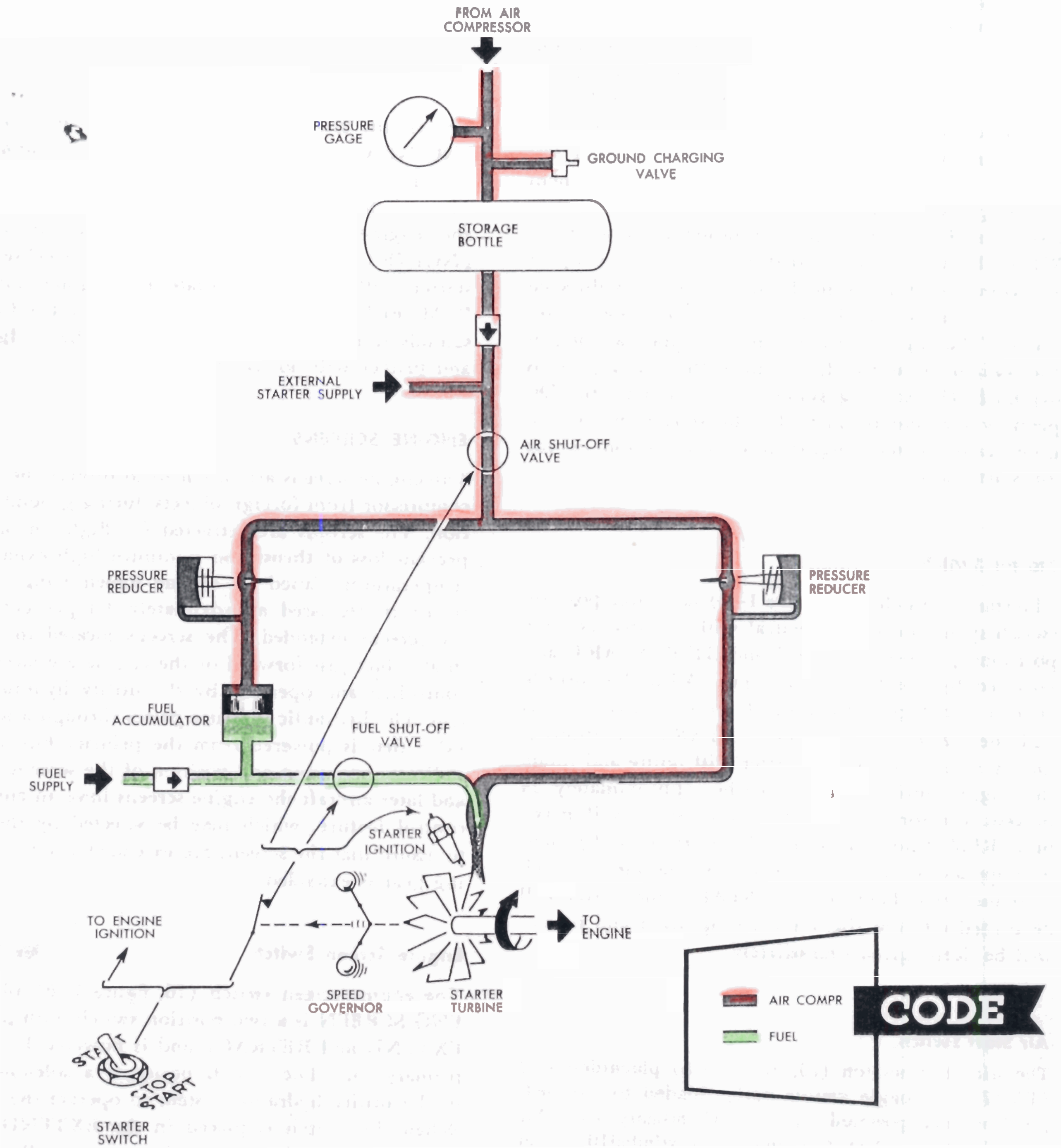


Figure 1—11

engine fuel control system prevents fuel from flowing to the engine until a system pressure of approximately 100 PSI is attained at the fuel control pressurizing valve. Fuel is supplied through the normally closed primer valve to two primer jets in the combustion chamber. The rapid accelerating starter will rotate the engine fast enough for the engine driven fuel booster pump and main dual element fuel pump to deliver sufficient fuel through the primer for starting. Ignition is provided by two igniters in the combustion chamber powered through individual high tension vibrators. Both the ignitors and the primer valve are energized through an automatic timer which permits operation for 15 (±3) seconds. An engine rotor test switch is located in the forward engine compartment and is powered by the DC primary bus. The rotor test is used to check engine rotation at slow speed. When the switch is actuated a solenoid valve opens and compressed air from the starter bottle is directed to the combustion chamber to operate the starter slowly. The engine will continue to rotate as long as the switch is actuated, or until the air supply is depleted. The starting system is powered by the DC primary bus and operated by the starter switch, engine crank switch, engine rotor test button or the air start button.

Starter Switch.

The starter switch (18, figure 1–5) is a three position switch spring-loaded to neutral with the two extreme positions placarded START and STOP START and powered by the DC primary bus. With the throttle at least five degrees forward of the CLOSED position and the starter switch held to START for approximately three seconds, the starter will ignite and rotate the engine until the RPM reaches approximately 24 percent (or for approximately four seconds if maximum RPM is not reached) and the igniter and primer will operate for 15 (±3) seconds. If the starter switch is momentarily held to STOP START, the starter will be turned off, and the automatic ignition and primer will be deenergized immediately.

Air Start Switch.

The air start switch (10, figure 1-5) placarded AIR START is a toggle switch spring-loaded to the OFF position and powered by the DC primary bus. Air start is used when the engine is windmilling and starting torque not required. When the switch is depressed momentarily an automatic timer energizes the igniters and opens the primer valve for 15 (±3) seconds. The timer cannot be recycled by actuating the air start switch until its 15 (±3) second period runs out, or the starter switch is positioned to STOP START. Ignition is available with the throttle in any position, however fuel will not be available when the throttle is closed, even though the primer valve may be energized and opened. A ground check of the ignition system may be made by depressing the air start switch when the throttle is closed.

Engine Crank Switch.

The engine crank switch (22, figure 1–6) placarded ENGINE CRANK is spring-loaded to the unmarked OFF position and powered by the DC primary bus. The switch is used for inspection of starter operation and engine rotation. When the switch is held to ENGINE CRANK while the throttle is CLOSED, the starter will ignite and rotate the engine until the RPM reaches approximately 24 per cent (or for four seconds if maximum RPM is not reached). Ignition and primer will not be supplied.

ENGINE SCREENS.

The engine screens are provided to protect the engine compressor from foreign objects during ground operation. The screens are retracted for flight in order to prevent loss of thrust and minimize high exhaust gas temperatures caused by engine screen icing. Engine thrust is decreased approximately 4.4 per cent with the screens extended. The screens located in the air intake duct just forward of the engine are electrically controlled and operated by the utility hydraulic system. The hydraulic pressure passes through a solenoid valve that is powered from the primary bus. A light indicates operation and position of the screens. On 40 and later aircraft the engine screens have an automatic control feature, which may be selected by the pilot, to assure that the screens are extended when the landing gear is extended.

Engine Screen Switch. 5 thru 35

The engine screen switch (16, figure 1–6) placarded ENG SCREEN is a two position switch with positions EXTEND and RETRACT and is powered by the DC primary bus. The switch positions a solenoid valve in the utility hydraulic system to operate the screens. When the switch is placed in the EXTEND or the RETRACT position the engine screen will move to the respective position and the solenoid valve will automatically return to neutral. The switch is left in the selected position after the screens have cycled.

Engine Screen Switch. 40 and LATER

The engine screen switch (16, figure 1–6) placarded ENG SCREEN is a three position switch with positions EXT, RETR and AUTO and is powered by the DC primary bus. The switch positions a solenoid valve in the hydraulic system to operate the screens. When the switch is placed in the RETR or EXT positions the engine screens will move to the respective positions and the solenoid valve will automatically return to neutral. The switch is left in the selected position after the screen has cycled. In the AUTO position, engine screen operation is automatic and the screens will automatically extend and remain extended as long as the main landing gear is in the extended position. The engine screens will also retract automatically after a time delay of 3½ (±1) minutes when the landing gear reaches the up and locked position. The time delay feature is provided so that enough altitude is gained before the screens retract after takeoff so that emergency procedures can be initiated in the event of engine failure caused by debris entering the engine compressor section. If manual selection is desired, the switch is placed in the desired position to retract or extend the screens.

Engine Screen Warning Light.

The engine screen warning light (figure 1-9) will illuminate whenever the engine screen switch is repositioned and will remain illuminated until both screens have completed their cycle to either the fully extended or retracted position if power is available from the primary bus.

ENGINE INSTRUMENTS.

The engine tachometer and the exhaust gas temperature indicator are self generated electrical instruments which do not require power from the aircraft electrical system. The fuel quantity indicator, the fuel flow indicator and the oil pressure indicator are operated from the AC power circuit and will operate from either the main or alternate inverters.

ASSISTED TAKE-OFF

Figure 1–12.

ASSIST TAKEOFF SYSTEM.

Provision is made for the installation of four solid fuel ATO units on an expendable adapter which is suspended from retractable hooks on the underside of the fuselage (figure 1–12). The units are used to supply additional thrust for high gross weight or shorter run takeoffs. Each unit is rated at 1,000 pounds thrust and once ignited, burns continuously for 15 seconds. After ignition takes place the units cannot be turned off or the adapter cannot be jettisoned while the units are producing thrust. Power to ignite the units is supplied from the DC primary bus. The adapter is jettisoned and the hooks on the fuselage are retracted by means of hydraulic pressure which is controlled by an electrically operated shut-off valve powered from the DC primary bus.

AIRCRAFT FUEL SYSTEM

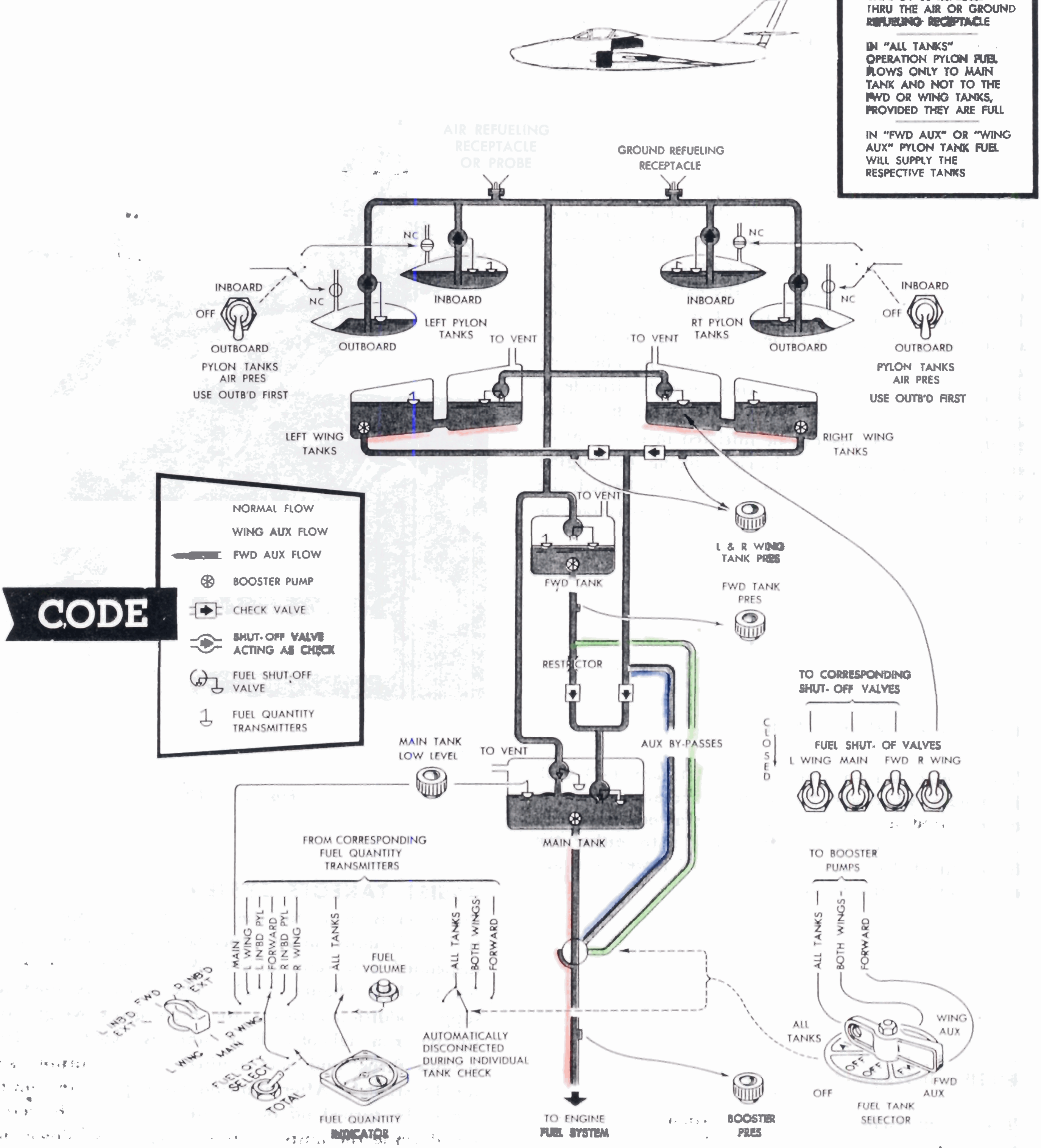


Figure 1—13

ATO READY SWITCH.

The ATO ready switch (figure 1–12) is provided as a safety switch to prevent accidental firing of the ATO units. It is a switch type circuit breaker with two positions marked OFF and ATO READY. The ATO READY position arms the ATO ignition system and illuminates the ATO ready indicator light (figure 1–12) which is placarded ATO READY.

ATO IGNITION BUTTON.

The ATO ignition button (figure 1–12) is a push button type switch marked ATO IGNITION and is powered by the DC primary bus. When depressed and held a circuit is completed to the ignition posts of the ATO units if the ATO ready switch is in the ATO READY position.

ATO JETTISON BUTTON.

The ATO unit adapter, suspended from retractable hooks under the fuselage, is hydraulically jettisoned after the ATO units have burned out. The jettisoning system is controlled by the ATO jettison button (figure 1–12) and is powered by DC primary bus. The button is a push button type jettison switch which energizes the ATO jettison solenoid shut-off valve to allow hydraulic pressure to release the adapter and retract the hooks into the fuselage.

OIL SYSTEM.

Lubricating oil is contained in an oil tank which is mounted on the upper left side of the engine compressor housing. The oil tank capacity is 6.0 U.S. gallons which includes expansion space and 4.27 gallons usable oil. Oil, under pressure, is delivered to the front bearings and accessory drives where it is scavenged and returned to the oil tank. Two metering pumps, in the oil pump, supply oil to the center and rear engine bearings. This oil is not scavenged but is vented overboard as a mist through a vent located on the underside of the fuselage. Oil consumption is approximately one quart per hour. An AC operated oil pressure indicator (13, figure 1–4) on the main instrument panel indicates oil pressure if the main or alternate inverter is operating. The oil tank is designed to supply sufficient oil to engine during inverted flight for a period of approximately one minute. Oil grade and specifications are noted in the servicing diagram figure 1–32.

AIRCRAFT FUEL SYSTEM.

The aircraft fuel system (figure 1–13), is designed to provide automatic fuel transfer during normal operation without attention from the pilot. The aircraft is equipped with four internal, self-sealing fuel tanks; a main tank installed behind the pilot, a forward tank under the cockpit floor and interconnected tanks in each wing. In addition, jettisonable external tanks can be carried on each of four pylons as follows:

- INBOARD – Any type I or II tanks.
- OUTBOARD – Only Type II, 230 GAL tanks.

Type I are 230 or 450 GAL tanks with fins.

Type II are 230 or 450 GAL tanks without fins.

Fuel from the external tanks is transferred to the main tank before wing or forward tank fuel is used. Normally tanks empty in the following order; external tanks, wing tanks, forward tank and the main tank. A restrictor is incorporated in the forward tank outlet to adjust fuel flow so as to automatically maintain the required CG location; the wing tanks are emptied at the same time or before the forward tank. Transfer of fuel from the external to the main tank is accomplished by means of air pressure which is manually controlled. Each internal tank incorporates a float which controls a shut-off valve in the transfer line to the tank. Fuel will flow to the tank until it is full then the float will automatically close the shut-off valve in the transfer line to that tank. These shut-off valves may be closed at any time to prevent fuel flow to any of the internal tanks. Internal tank fuel is pumped into the main tank by booster pumps which operate automatically according to the position of the fuel tank selector. The main tank is kept full until all external and internal fuel is used. A booster pump in the main tank supplies the engine fuel system. The pilot may vary the normal selector to allow a direct flow of fuel to the engine from either the internal wing or forward tanks. If the wing or forward tank is selected and the main tank shut-off valve is open, fuel will flow to the main tank until it is full then all fuel will flow from the selected tank to the engine. The system is provided with a flowmeter which indicates the rate of fuel flow to the engine. Also a fuel quantity indicator which indicates the amount of fuel remaining in the aircraft except for unmodified Type II external tanks. Provisions for ground refueling of

FUEL QUANTITY DATA

TANK	NO.	USABLE FUEL GAL	USABLE FUEL POUND	FULLY SERVICED GAL	FULLY SERVICED POUND
Main	1	193.0	1254.5	193.4	1257.1
Forward	1	117.0	760.5	117.6	764.4
L. Wing	1	102.0	663.0	104.2	677.3
R. Wing	1	133.0	864.5	135.8	882.7
L. Inboard Pylon—Type I	1	222.0	1443.0	225.0	1462.5
R. Inboard Pylon—Type I	1	222.0	1443.0	225.0	1462.5
L. Inboard Pylon Type I Alt	1	450.0	2925.0	450.0	2925.0
R. Inboard Pylon Type I Alt	1	450.0	2925.0	450.0	2925.0
L. Outboard Pylon Type II	1	222.0	1443.0	225.0	1462.5
R. Outboard Pylon Type II	1	222.0	1443.0	225.0	1462.5

USABLE FUEL TOTALS	GAL	POUNDS
Maximum Internal	545.0	3542.5
Maximum Internal plus Inboard Pylons	989.0 1445.0 Alt	6428.5 9392.5
Maximum Internal plus Inboard and Outboard Pylons	1433.0	9314.5

(NOTE) Weight based on fuel at 6.5 pounds per gallon

Figure 1—14.

the aircraft, from a single point on all aircraft, are made through a refueling receptacle on the underside of the right wing, or an air refueling receptacle on the left wing of 5 thru 35 aircraft not modified by [557]. On 40 and later aircraft and 5 thru 35 aircraft modified by [557] the air refueling receptacle is deactivated and a fixed probe extending from the left wing is used for air refueling. The single point refueling equipment is designed to operate similar to the air refueling system. The refueling truck must be equipped with a single-point nozzle and must be capable of delivering fuel at 500 GPM at a pressure of 50 PSI. The aircraft is provided with an air refueling system which is covered in Section IV.

FUEL SPECIFICATION AND GRADE.

Recommended fuel specification and grade are noted on the servicing diagram, figure 1—32.

ELECTRIC FUEL BOOSTER PUMPS.

An electric booster pump is provided in the main fuel tank to supply fuel to the engine fuel control system. Electric booster pumps in the wing and forward tanks normally transfer fuel from these tanks to the main tank but also may be used to supply the engine fuel control system directly by proper positioning of the fuel tank selector. On 5 thru 25 aircraft when the fuel tank selector is in ALL TANKS (normal) position the main tank booster pump is powered from the

DC primary bus. The forward and wing tank booster pumps are powered from the NO. 2 DC secondary bus, and will not operate in the event of failure of either generator. When the fuel tank selector is positioned to supply fuel directly to the engine from the wing or forward tanks, the respective booster pump is powered from the DC primary bus. On (30) and later aircraft with the fuel tank selector in ALL TANKS the main tank booster pump is powered from the the NO. 1 secondary bus, and the wing and forward tank booster pumps are powered from the NO. 2 secondary bus. The wing and forward tank booster pumps will not operate if one generator fails and the main tank booster pump will not operate if both generators fail. When the fuel tank selector is positioned to supply fuel directly to the engine from the wing or forward tanks, the respective booster pump is powered from the NO. 1 secondary bus. At altitudes below 6,000 feet, full engine RPM may be maintained with a failed booster pump as the fuel can be recovered by direct suction of the engine driven booster pump. Satisfactory engine operation up to maximum range power settings will result up to 20,000 feet under the most severe conditions when using JP-4 fuel. However, if booster pump failure is experienced, above 20,000 feet with JP-4 fuel and engine operation is satisfactory (no RPM drop of excessive RPM fluctuation) flight may be continued without reducing altitude. It must be remembered that prolonged flight at high altitudes, without booster pump operation, will shorten the engine fuel pump life due to the low inlet pressures. Fuel from the wing or forward tanks cannot be transferred to the main fuel tank without the aid of the booster pumps in the respective tanks, however, fuel may be fed directly to the engine from the forward tank but at slightly lower altitudes than would be available when feeding from the main tank. It is possible to feed in this manner from the wing tanks but operation under these conditions is not recommended since fuel in the tank with the failed pump cannot be recovered after the other tank is empty. There are no direct indicators to show when a booster pump is not operating, however, booster pump failure may be suspected as noted under fuel system indicators.

FUEL SHUT-OFF VALVE SWITCHES

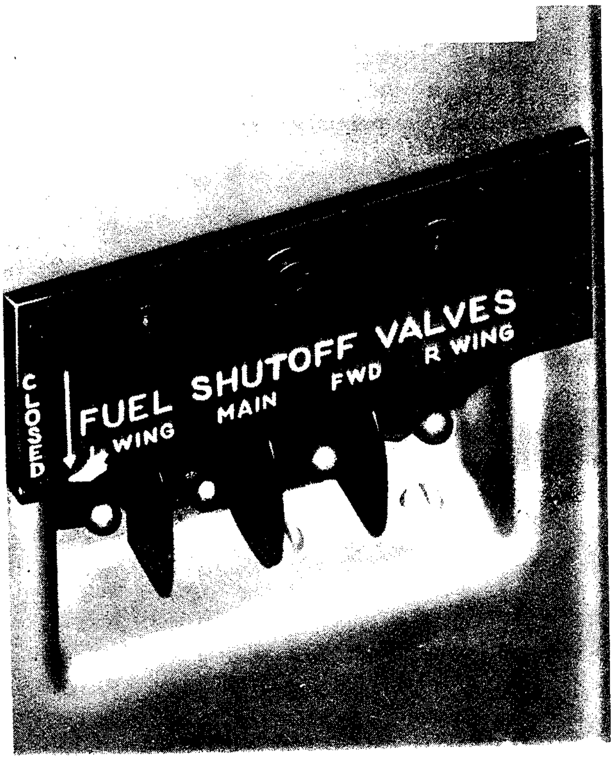


Figure 1—15.

FUEL SHUT-OFF VALVE SWITCHES.

Shut-off switches (13, figure 1–6 and figure 1–15) are provided to close the fuel shut-off valves in the internal tanks so that fuel will not be transferred from the external to the internal or from the wing and forward to the main tank. The same fuel lines are used for transfer of fuel from the external to the internal tanks as is used to supply all tanks during air refueling. The fuel shut-off valves provide a means of controlling fuel flow to the individual internal tanks by closing a valve at the fuel line entrance to the tank. Each of the four fuel shut-off valves is controlled by a two-position toggle switch. The switches are marked L WING, MAIN, FWD and R WING with an arrow indicating the CLOSED position. During normal operation the switches are left in the up position, which allows fuel to transfer from the external to the internal tanks and also allows all tanks to be refueled from the refueling receptacle. By placing the fuel shut-off valve in the CLOSED position the respective fuel tank will not receive fuel by transfer or air refueling. When the main fuel shut-off switch is in the CLOSED position both valves in the main tank are closed and fuel from neither internal nor external tanks can be transferred. This system is provided to isolate each of the internal tanks, in the event of battle damage to the tank, or failure of the booster pump in the tank. The fuel shut-off valves are powered by the primary bus.

FUEL TANK SELECTOR.

The fuel tank selector (figure 1—9) is a rotary control having four positions. The selector mechanically positions the fuel tank selector valve and indexes a rotary switch which opens and closes circuits to the internal tank booster pumps and to the fuel quantity indicator so that readings on the indicator will reflect the tank(s) being used. The four positions are as follows:

All Tanks.

The ALL TANKS (normal) position turns on all booster pumps and channels all internal fuel into the main tank. The fuel quantity indicator will read fuel in all tanks except outboard external tanks and unmodified Type II inboard external tanks. The main tank booster pump supplies fuel to the engine driven booster pump. As the fuel in the main tank drops to a predetermined level, a float in the tank opens a shut-off valve in the transfer line from external tanks and allows fuel from the external tanks to transfer to the main tank. When external tanks have emptied, fuel from the wing and forward tanks transfers to the main tank simultaneously. To maintain a favorable CG travel the flow is such that approximately 0 to 20 gallons will still be in the forward tank after the wing tanks have emptied. The remaining forward and main tank fuel is then used. The fuel quantity indicator will indicate fuel remaining in internal and Type I or modified Type II inboard external tanks.

Wing Aux.

The WING AUX position turns on the wing tank booster pumps; routes wing tank fuel directly to the engine driven booster pump and completes a circuit to the fuel quantity indicator so that the reading indicates fuel in the wing and Type I inboard external tanks only. No other booster pump will operate with the fuel tank selector in this position. When operating in WING AUX position with the fuel shut-off switches open fuel will feed to the engine and continue to transfer to the main tank until either the pylon tank float valve in the main tank closes if there is external fuel, or the internal tank float valve in the main tank closes if there is no external fuel. All remaining fuel in the external and wing tanks will then be available to the engine fuel system through the auxiliary feed lines.

CAUTION

When operating with the fuel tank selector in the WING AUX position, avoid uncoordinated maneuvers, steep descents or rapid maneuvers, since these maneuvers may uncover the wing tank fuel outlets thereby causing a flame-out.

Fwd Aux.

The FWD AUX position closes a circuit to turn on the forward tank booster pump; indexes the selector valve so that forward tank fuel is fed directly to the engine driven booster pump of the engine fuel system and completes a circuit to the fuel quantity indicator so that the reading will indicate fuel in the forward and Type I or modified Type II inboard external tanks only. No other electric booster pump will operate with the fuel tank selector in this position. When operating in the FWD AUX position with the fuel shut-off switches open, fuel will feed to the engine and continue to transfer to the main tank until either the pylon tank float valve in the main tank closes if there is external fuel, or the internal tank float valve in the main tank closes if there is no external fuel. All remaining fuel in the external and forward tanks will then be available to the engine fuel system through the auxiliary feed lines.

CAUTION

When operating with the fuel tank selector in the FWD AUX position with low fuel quantity remaining, avoid uncoordinated turns or nose down attitudes as these maneuvers may result in loss of fuel supply to the forward tank booster pump.

Off.

The OFF position turns off the internal tank booster pumps and shuts off the fuel supply to the engine fuel system. The fuel quantity indicator will show the amount of fuel remaining in the internal tanks plus the fuel in Type I inboard external tanks.

PYLON TANKS AIR PRESSURE SWITCHES.

Fuel from the external tanks is transferred to the main tank by means of air pressure. The pylon tank air pressure switches (figure 1–9) control the flow of air for pressurizing the external tanks. The switches are placarded PYLON TANK AIR PRESS USE OUTBD FIRST. There are two three-position toggle switches marked L and R with positions of OUTBD, OFF and INBD. When in the INBD or OUTBD position, electrical power is directed to open the selected solenoid valve permitting air from the engine compressor to pressurize the respective tanks. In the OFF position the air pressure and vent ports are closed. The vent port is automatically opened during the air refueling cycle so as to permit fuel flow into the tanks. The air pressure port is spring-loaded to the closed position. Power from the primary bus keeps the port open, therefore, in the event of primary bus power failure the external fuel will not transfer.

CAUTION

On 5 thru 35 aircraft not modified by [557] the air refueling receiver switch must be in the CLOSED position in order to pressurize the pylon tanks. On 40 and later aircraft and 5 thru 35 aircraft modified by [557] the probe READY switch must be down (closed) in order to pressurize the pylon tanks.

JETTISON SYSTEM.

Note

The speed limitations for jettison of external stores is presented in Section V.

External Stores Jettison Button.

The external stores jettison button (12, figure 1–5) on the left auxiliary instrument panel is a recessed push button type switch placarded EXT STORES JETTISON and is powered by the emergency bus. On 5 thru 25 aircraft when the button is depressed, all outboard and inboard pylon stores together with the pylons will be jettisoned. On 30 and later aircraft only the pylon stores will be jettisoned while the pylons will be retained. On 45 and later aircraft and 5 thru 40 aircraft modified by [505] an automatic time delay feature prevents both inboard tanks from being released simultaneously. The jettison button must be held depressed for a minimum of 0.4 seconds to allow the automatic delay circuit to drop the left inboard tank first and after a short delay to drop the right inboard tank.

EXTERNAL STORES Control Panel

External Stores Arming Switch.

The external stores arming switch (19, figure 1-6) selects the set of pylons or tanks to be jettisoned and is used in conjunction with the pylon or tank jettison switch. It is a three-position switch placarded EXT STORES with positions placarded INBD, OFF and OUTBD. When positoned to either the OUTBD or INBD position, the respective tanks or pylons will be jettisoned as selected by the jettison switch if the primary bus is energized.

Pylon or Tank Jettison Switch.

The pylon or tank jettison switch (19, figure 1-6) is provided so that the tanks or pylons can be selected when jettisoning external stores. It is a three-position guarded switch placarded TANK RELEASE, PYLON JETT and a center off position which is unmarked. When the switch is momentarily held in TANK RELEASE or PYLON JETT position, either the outboard or inboard units will jettison as selected by the external stores arming switch, if the primary bus is energized. On 45 and later aircraft and 5 thru 40 aircraft modified by 505 an automatic time delay feature is incorporated in the circuit to prevent simultaneous release of the inboard pylon tanks as they may collide and possibly damage the wings or fuselage. The switch must be held in the jettison position for a minimum of 0.4 seconds, to allow the automatic delay circuit to drop left inboard tank first and after a short delay to drop the right inboard tank.

WING PUMP PRESSURE WARNING LIGHT.

The wing pump pressure warning light (figure 1-16) is a press-to-test type light placarded WARNING L & R WING TANK PRES and is powered by the primary bus. Illumination of the light indicates one or both wing tanks are empty, or one or both booster pumps is inoperative. The light will illuminate when the pressure in the line from either tank is below approximately nine PSI. In the event of a failed booster pump in one wing, fuel from that wing will not transfer to the main tank. However, the fuel can be utilized with the fuel tank selector in WING AUX operation.

CAUTION

During normal operation, the wing pump pressure warning light will illuminate when one or both wing tanks are empty. However, in WING AUX operation, the engine is being fed directly from the wing tanks and at the first flicker or flash of the warning light the FWD AUX position on the fuel tank selector must be selected immediately (if fuel remains in the forward tank) to assure against a flame-out.

FORWARD PUMP PRESSURE WARNING LIGHT.

The forward pump pressure warning light (figure 1-16) is an amber press-to-test type light placarded FWD PUMP PRESS and is powered by the primary bus. Illumination of the light indicates the forward tank is empty, or the booster pump is inoperative. The light will illuminate if the pressure in the fuel line from the forward tank is below approximately nine PSI. In the event of a booster pump failure the forward tank fuel cannot be transferred to the main tank by gravity since the main tank is higher than the forward. However, the fuel can be utilized in FWD AUX operation.

FUEL SYSTEM

Warning Lights

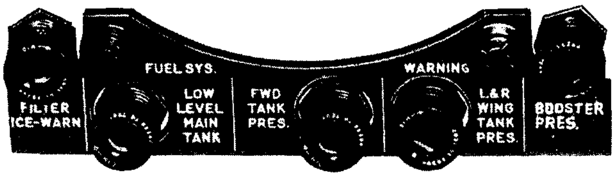


Figure 1—16.

MAIN TANK LOW LEVEL WARNING LIGHT.

The main tank low level warning light (figure 1—16) is a red press-to-test type light placarded LOW LEVEL MAIN TANK and is powered by the primary bus. Illumination of the light indicates the fuel level in the main tank is below 975 (± 145) pounds. The light will remain on until the main tank is refueled or the primary bus is deenergized.

BOOST PRESSURE WARNING LIGHT.

The boost pressure warning light (figure 1—16) is a red press-to-test type light placarded BOOSTER PRESS and is powered by the primary bus. Illumination of the light indicates the pressure in the main fuel supply line between the fuel tank selector valve and the engine driven centrifugal booster pump is below approximately nine PSI.

Note

It is permissible for the booster pressure warning light to flicker when the emergency fuel system is switched ON or OFF and flicker or remain on momentarily during burst acceleration from any throttle setting. This is due to a momentary increase in fuel flow demanded by the engine above the output of the pump resulting in a decrease in pressure in the engine feed line causing the pressure switch to actuate the light.

FUEL QUANTITY INDICATOR.

The fuel quantity indicator (figure 1—17) is an AC powered electrical unit which operates from either the main or alternate inverter. It reads the quantity of fuel remaining in any or all internal tanks in the aircraft and Type I, or modified Type II inboard external tanks in units of pounds. Fuel weight is calibrated by the combination of a float unit in each tank and a unit installed in the main tank which compensates for the density and temperature of fuel in the tanks. This combination assures a true indication of the weight of fuel in the aircraft within a tolerance of approximately 4 percent when operating with the fuel tank selector in the ALL TANKS positions; however, the indicator will read differently each time the aircraft is serviced with fuel of a different density. A fuel volume switch is provided to change the electrical circuit so that full fuel volume is indicated on the same dial. The fuel volume reading is necessary in that it provides a means of indicating that the aircraft is fully serviced regardless of fuel density. Three reference points are positioned on the dial to indicate full fuel for three different configurations.

Normal Reading.

Normally the reading on the large scale multiplied by 1,000 plus the reading on the small scale multiplied by 100 indicates the fuel remaining in the aircraft in pounds. With the fuel tank selector in the ALL TANKS or OFF position and the fuel quantity check switch in TOTAL the fuel quantity indicator will read the total weight of internal and Type I or modified Type II inboard external fuel remaining in the aircraft. With the fuel tank selector in the WING AUX or FWD AUX position and the fuel quantity check switch in TOTAL the quantity indicator will read the weight of fuel remaining in the respective tank plus the fuel remaining in the Class I inboard external tanks.

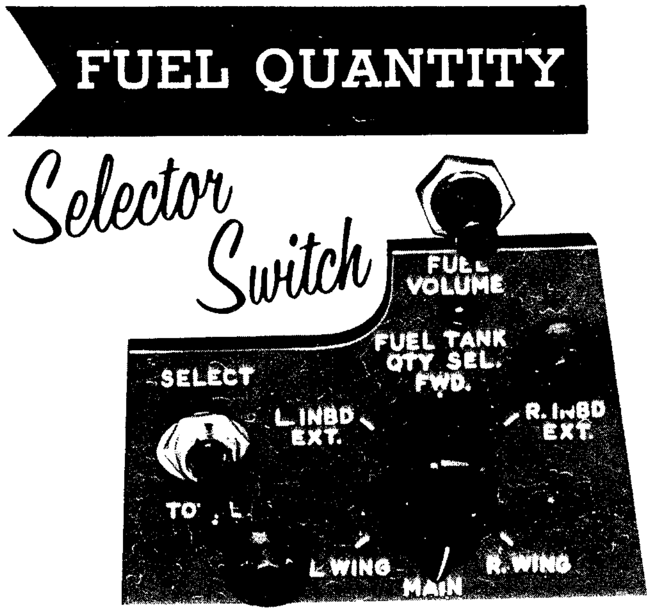


Figure 1—17.

WARNING

On 5 and 10 aircraft with {267} not complied with; due to an inherent error in the system, the fuel quantity indicator reads approximately 25 per cent low when operating with the fuel tank selector in the WING AUX or FWD AUX position.

Fuel Volume Switch.

The fuel volume switch is a spring-loaded pushbutton type switch placarded FUEL VOLUME (figure 1—17) and is provided to change the fuel quantity indicator AC circuit so that fuel volume instead of fuel weight is read on the fuel quantity indicator. This information is valuable when refueling on the ground or in the air as the fuel capacity of the aircraft, in gallons, remains constant. There are three dots on the indicator, one at 4,195 LB corresponds to full internal fuel only, the second dot at 7,551 LB corresponds to full internal fuel plus two 230 GAL Type I inboard external tanks, the third dot at 10,761 LB corresponds to full internal fuel plus two 450 GAL Type I inboard external tanks. When the fuel volume switch is depressed with the fuel tank selector in the ALL TANKS position and the aircraft fully serviced the pointer will stop at the dot corresponding to the aircraft configuration. Validity of these readings would be affected if outboard external tanks or unmodified Type II inboard tanks are carried since these do not have fuel quantity transmitters.

Fuel Tank Quantity Selector Switch.

The fuel tank quantity selector switch (figure 1—17) is provided so that the fuel quantity in individual tanks can be determined. It is a rotary switch placarded FUEL TANK QTY SEL with the following positions; FWD., R. INBD EXT., R. WING, MAIN, L. WING, and L. INBD EXT. When positioned to any one of these positions, the quantity of fuel in the respective tank will register on the fuel quantity indicator when the fuel quantity check switch is placed in the SELECT position regardless of the position of the fuel tank selector. Unmodified Type II inboard external tanks will not register fuel remaining as these tanks are not equipped with fuel quantity transmitters.

CAUTION

On 5 aircraft with {267} not complied with; the fuel quantity remaining readings for individual tanks, obtained by use of the fuel tank quantity selector switch, will not add up to the total fuel remaining reading obtained with the fuel tank selector in the ALL TANKS position and the fuel quantity check switch in TOTAL due to the inherent tolerances in the calibrating system. However, the total fuel quantity remaining will be correct to within approximately 4 per cent.

Note

On (5) and (10) aircraft with [267] not complied with; if the fuel quantity selector switch is positioned to the EXT R INBD position without pylon tanks installed all internal fuel remaining will be indicated on the fuel quantity indicator when the fuel quantity check switch is positioned to the SELECT position. If the quantity selector switch is positioned to EXT L INB'D position under the same conditions the quantity indicator will read zero.

Fuel Quantity Check Switch.

The fuel quantity check switch (figure 1—17) is used when it is desired to ascertain the quantity of fuel in the individual tanks. The switch has two positions SELECT and TOTAL. When placed in the SELECT position the quantity of fuel remaining in the tank, indicated by the position of the fuel tank quantity selector switch, will register on the fuel quantity indicator. The TOTAL position will register fuel remaining in tanks as indicated by the fuel tank selector.

FUEL FLOW INDICATOR.

The fuel flow indicator (11, figure 1—4) registers the flow of fuel to the engine in units of pounds per hour. The indicator is AC powered from either the main or alternate inverter. It is operated by a fuel flow meter which is located in the fuel system between the pressurizing valve and the engine burners. All fuel flowing to the engine will be recorded whether from the normal or auxiliary systems. The fuel flow indicator is accurate to approximately 1 percent on the high side to 3 percent on the low side.

ELECTRICAL POWER SUPPLY SYSTEM.

The aircraft is equipped with both AC and DC electrical power systems (figure 1—18). All circuits are protected by either circuit breakers or fuses which are accessible from the cockpit, except the emergency bus circuit breaker in the right wing stub. A standby 4.5 volt dry cell battery is installed as an emergency source of power for canopy jettisoning if the main electrical system fails. On (35) and later aircraft a 26.25 volt, wet cell, silver zinc battery is provided as an emergency source of power for the command radio. Refer to Section VII for electrical load chart.

CIRCUIT BREAKER PANELS.

Circuit breaker panels (figure 1—19) are provided to protect the various electrical circuits in the aircraft. The circuit breakers are of the push button type and are pushed in to reset.

DC SYSTEM.

The 28-volt DC system is powered from two engine-driven generators; a 400-ampere (NO. 1) generator and a 200-ampere (NO. 2) generator. A 24-volt, 36-ampere-hour battery is installed as a standby source of DC power. The system also incorporates an external power receptacle for the accommodation of an external power cart. Electrical power is distributed through a four-bus system consisting of: an emergency bus, a primary bus, a NO. 1 secondary bus and a NO. 2 secondary bus. The emergency bus services emergency equipment and remains energized if the emergency bus circuit breaker is closed, regardless of the battery switch position or generator operation. The primary bus services equipment essential to flight and is energized by; the battery, either generator, or the external power receptacle. The secondary system is divided into two busses; i.e., NO. 1 and NO. 2 which service equipment not essential for flight. In the event of failure of either generator, power will still be available for some equipment. The NO. 1 secondary bus is energized if either or both generators are operating or if external power is provided. The NO. 2 secondary bus is energized only if both generators are operating or if external power is provided. Therefore, in the event of failure of both generators in flight, all equipment not essential to flight will automatically cut out since both secondary busses will cease to be energized and battery power will be conserved for the primary bus equipment; i.e., equipment essential to flight. An

ELECTRICAL POWER SYSTEM SUPPLY

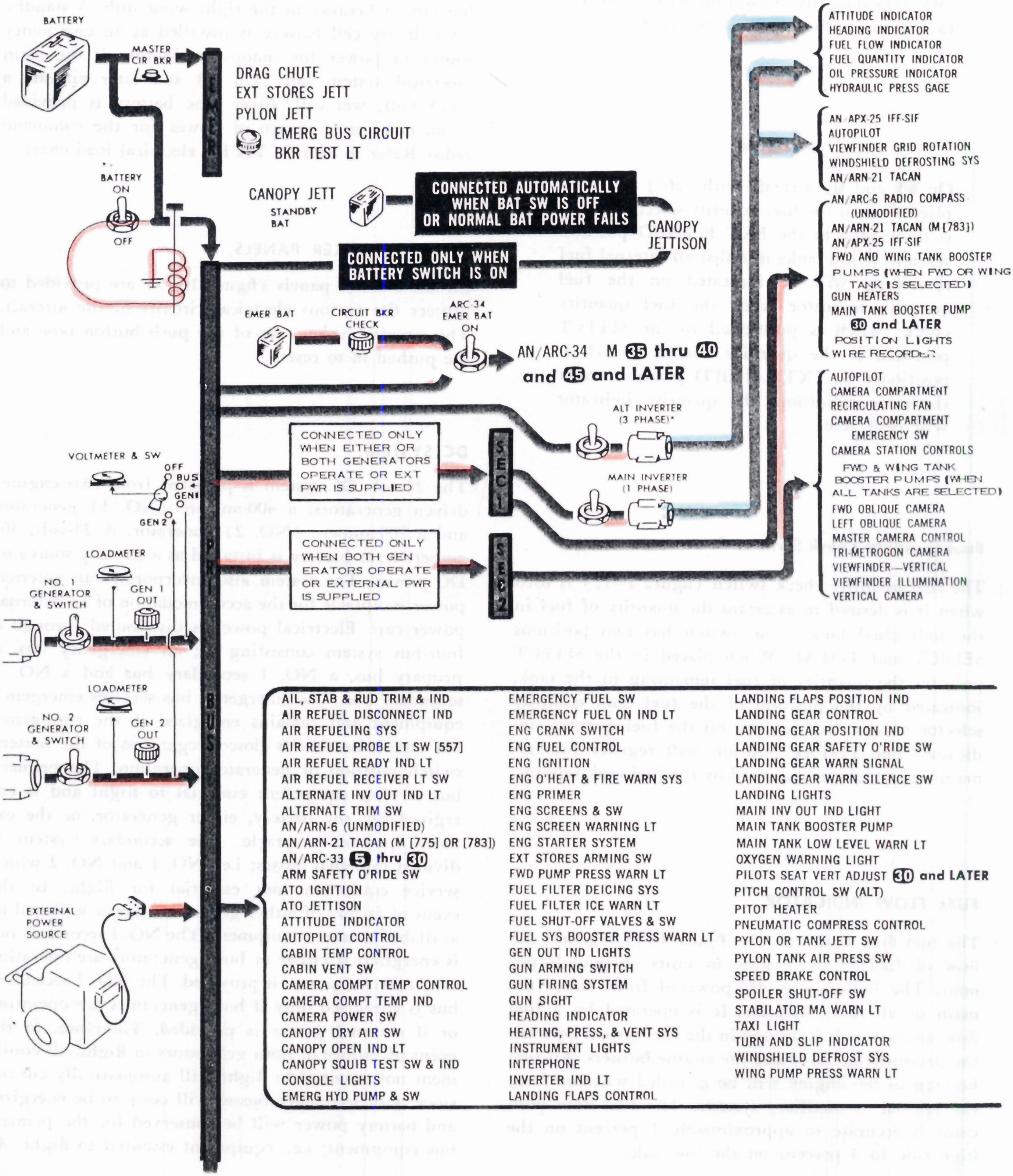


Figure 1-18

emergency bus circuit breaker is provided in the system to de-energize the emergency bus when ground personnel are working on the aircraft. External power must be used for operational checks if the engine is not operating because the secondary busses cannot be energized by the aircraft's battery.

Emergency (Battery) Bus Circuit Breaker Test Light.

The emergency (battery) bus circuit breaker test light (11, figure 1–6) is a green press to test type light located on the left console. The light is placarded PRESS TO CHECK LT. IND. POWER FOR PYLON & CANOPY JETT. When depressed and illuminated it indicates that the emergency bus circuit breaker located in the right wing stub is closed and the emergency bus is energized. The light is powered by the emergency bus.

Battery Switch.

The battery switch (22, figure 1–5) is a two-position on-off switch. The ON position connects the aircraft's battery to the primary bus through a relay which obtains its power directly from the battery. Approximately 18-volts are required to close the relay and it will remain closed down to approximately seven volts. Therefore, unless approximately 18-volts is available from the battery, the relay will not close. In the OFF position the aircraft's battery is disconnected from the primary bus.

Generator Switches.

Two guarded generator switches (15, 19, figure 1–5) marked GEN 1 and GEN 2 are provided to reset the generator field controls if the generator has cut out due to over-voltage. Each switch has three positions: ON, OFF and RESET and is guarded in the ON position with a cover-type guard. If the generator voltage is too high or too low, the generator is automatically disconnected from the electrical system. If the generator has cut out due to over-voltage, the respective generator switch can be held in the RESET position for a few seconds to reset the generator field control relay and then returned to the ON position. If the generator does not operate after positioning the respective generator switch to the RESET and returning it to the ON position, the OFF position is selected to disconnect the generator from the electrical system. The ON position connects the generators to the electrical system whenever the generator speed is high enough (approximately 30 per cent engine RPM) to close the reverse current relay.

Voltmeter Selector Switch.

The voltmeter selector switch (16, figure 1–5) is provided so that the voltage output of each generator or the primary bus may be selected and read individually, since they are all indicated on the one voltmeter. The switch is a rotary type with four positions: OFF, BUS, GEN 1 and GEN 2. When positioned to the GEN 1 or GEN 2 position, the voltmeter will indicate the voltage output of the respective generator. The BUS position indicates the voltage on the primary bus while the OFF position disconnects the voltmeter from the electrical system.

Voltmeter.

The voltmeter (17, figure 1-5) indicates the voltage output of the primary bus, the NO. 1 generator or the NO. 2 generator depending on the position of the voltmeter selector switch.

Loadmeters.

The loadmeters (figure 1–5) are placarded LOAD and indicate the load being drawn from the generator from zero to 100 per cent, with provisions for an additional 25 per cent reading to indicate over-load and a minus 10 per cent reading to indicate discharge. The normal loadmeter reading is approximately 0.5 (+ 0.1).

Generator-Out Warning Lights.

The two generator-out warning lights (figure 1–5) are red press-to-test-type lights placarded GEN OUT, GEN 1, and GEN 2 and are powered by the DC primary bus. Illumination of a light (1 or 2) indicates the respective generator is disconnected from the primary bus. If one generator is out, the other will supply power to the primary bus. If both generators are disconnected from the primary bus, this results in both secondary busses becoming deenergized and the aircraft battery alone powering the primary bus.

AC SYSTEM.

The AC system is energized by a single-phase and a three-phase inverter. Both inverters are rated at 115-volts and 400 cycles. The single-phase (main) inverter is powered by the NO. 1 secondary bus while the three-phase (alternate) inverter is powered by the primary bus. In the event of failure of one of the two DC generators, the AC system will not be affected. If both DC generators fail, only the three-phase (alternate) inverter will operate from the primary bus and only the necessary flight instruments will be supplied

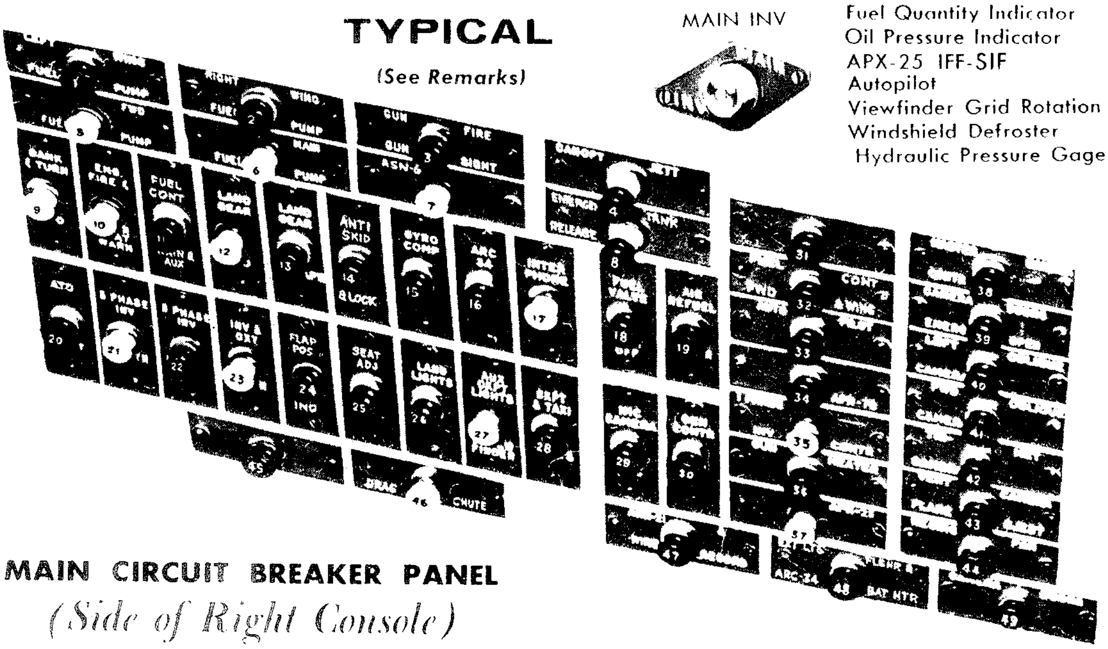
CIRCUIT BREAKER PANELS

TYPICAL

(See Remarks)

MAIN INV

Attitude Indicator
Heading Indicator
Fuel Flow Indicator
Fuel Quantity Indicator
Oil Pressure Indicator
APX-25 IFF-SIF
Autopilot
Viewfinder Grid Rotation
Windshield Defroster
Hydraulic Pressure Gage



MAIN CIRCUIT BREAKER PANEL
(Side of Right Console)

CIRCUIT BREAKER	OPERATING UNIT AND/OR CONTROL
1 LEFT WING FUEL PUMP	Left Wing Fuel Tank Pump
2 RIGHT WING FUEL PUMP	Right Wing Fuel Tank Pump
*3 GUN FIRE GUN SIGHT	Gun Firing Solenoids, Gun Arming Control Gun Firing Control, Gun Sight & Control
*4 CANOPY JETT	Canopy Jettison (24-volt battery)
5 FWD FUEL PUMP	Fwd Fuel Tank Pump
6 MAIN FUEL PUMP	Main Fuel Tank Pump
*7 ASN-6	Ground Position Indicator
*8 EMERGENCY TANK REL	Pylon Tank Jettison, External Stores Jettison, Pylon Jettison
9 BANK & TURN IND	Turn and Slip Indicator
10 ENG FIRE & OVERHEAT WARN	Engine Compartment Fire Warning System Aft Section Overheat Warning System
*11 FUEL CONT MAIN & AUX	Main & Wing Tank Fuel Pumps Control
12 LAND GEAR IND	Landing Gear Position Indicators
13 LAND GEAR UNSAFE	Gear Unsafe Warning Light & Test Switch Gear Warning Horn & Silencing Switch
*14 ANTI-SKID & LG LOCK HDLE	Anti Skid Control, Landing Gear Selector Lock Emergency Override Switch Armament Safety Override Switch, Hyd Boost Brakes
15 GYRO COMP	Heading Indicator Slaving Control
*16 ARC-34	COMMAND RADIO
17 INTERPHONE	Interphone
18 FUEL VALVE SHUT OFF	Primary & Secondary Shut-off for: Air Refueling Ground Single Point Refueling Battle Damage
19 AIR REFUEL CONT	Air Refueling Control Pylon Tank Air Pressure Control

REMARKS

*3 5 and 10

CAUTION
*4 4.5 Volt Standby Battery is connected when circuit breaker is out or battery switch is off.

*7 ASN-6 (Deleted by [783])
5 and 10
GUN SIGHT
5 thru 25
Gun Arming Control
Gun Firing Control
Gun Sight & Control

*8 Pylon Jettison Squibs

*11 5 thru 25
WING PUMP RELAY
Left & Right Wing Tank Fuel Pumps Control

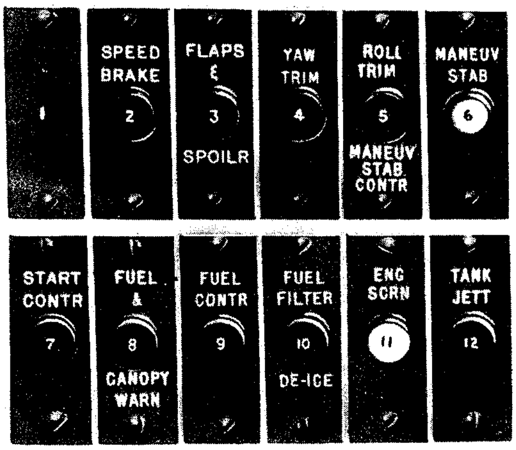
*14 5 thru 25
LAND GEAR ARM SAFETY
Landing Gear Selector & Emergency Release
Armament Safety & Override Switches
Hyd Boost Brakes

*16 5 thru 30
ARC-33
Command Radio

Figure 1—19 (Sheet 1 of 4)

CIRCUIT BREAKER	OPERATING UNIT AND/OR CONTROL	REMARKS
20 ATO JETT	Jato Ready Light Autopilot (Modified) Jato Ignition Jato Hook Retract Control	
21 3 PHASE INV CONTR	3 Phase (Alternate) Inverter Control (ON-OFF Switch)	
22 3 PHASE INV	Attitude Indicator Heading Indicator Fuel Flow Indicator Fuel Quantity Indicator Oil Pressure Indicator (Hydraulic Pressure Gage—Tandem Aircraft)	
23 INV & OXY WARN	Inverter Out Warning Control Oxygen Warning Light	
*24 FLAP POS IND	Flap Position Indicator System Emergency Brake Warning Light	*24 35 **AND LATER**
*25 SEAT ADJ	Seat Height Adjustment & Control	*25 5 thru 30 TEMP CONT Cabin Heating System & Control Cabin Pressurization Control Windshield Defrost—Electric Camera Compartment Air Conditioning (35 and Later Seat Height Adjustment & Control)
26 LAND LIGHTS	Landing Lights Extend & Retract Actuators	
27 AUX CKPT LTS & VIEW FINDER	Auxiliary Panel and Cockpit Lights & Control C-4 Light Viewfinder & Controls	
28 CKPT & TAXI LIGHTS	Instrument & Placard Lights Taxi Light	
29 IMC CAMERA MAG	Image Motion Compensating Magazine	
30 GEN CONT	1 & 2 Generator Control Generator Warning Lights	
*31 APX-6 DET	APX-6 Detonators	*31 30 **AND LATER** or T.O. [783] Compliance Deleted
*32 FUEL CONT FWD & WING	Fwd & Wing Fuel Pumps Relay Control	*32 5 thru 25 FUEL PUMP ALL ON Main, Fwd & Wing Fuel Pumps Control Relay
33 AUTO PILOT	Auto Pilot	
*34 APW-11	APW-11 Ground Support Radar	*34 5 thru 30 Modified by [783] Deleted
35 1 PHASE INV CONT	1 Phase (MAIN) Inverter Control (ON-OFF Switch)	
36 GUN HEATER	Gun Heaters and Control	*37 UNMODIFIED BY [783] APX - 6
*37 APX-25 IFF- SIF	Identification Radar	
38 MASTER CAMERA CONT	Master Camera Control	*45 5 thru 25 ONLY
39 CAMERA DOORS EMER OPEN	Emergency Pneumatic Control for: Viewfinder Door Prime Vertical Camera Pod Ejector Doors	*46 5 thru 30 FLAP POS IND Flap Position Indicator System Emergency Brake Warning Light
40 LEFT OBLIQUE CAMERA	Left Oblique Camera Control	
41 FWD OBLIQUE CAMERA	Fwd Oblique Camera Control	
42 TRI-MET CAMERA	Tri-Met Camera Control	
43 VERT CAMERA FLARE EJECT	Vertical Camera Control Flare Eject	*47 UNMODIFIED [783] ARN-6 & WIRE RECORD Radio Compass Wire Recorder
44 RECIRC FAN	Recirculating Fan (Camera Comp)	
*45 EMERG HYD PUMP	Emergency Hydraulic Pump	
*46 DRAG CHUTE	Drag Chute Deploy Solenoid & Control Drag Chute Jettison Solenoid & Control	
*47 WIRE RECORD & ARN-21	TACAN Wire Recorder	
*48 EXT LTS FLSHR & ARC-34 BAT HTR	ARC-34 Emergency Battery Heater	*48 5 thru 30 EXTER LIGHTS & FLASHER Position Lights & Flasher Unit
49 CAMERA DOORS	Camera Doors	

Figure 1—19 (Sheet 2 of 4)



AUXILIARY
CIRCUIT BREAKER PANEL

5 thru 25

(Location-Top of Left Console)

CIRCUIT BREAKER	OPERATING UNIT AND/OR CONTROL	REMARKS
1		
* 2 SPEED BRAKE	Speed Brake Control Pneumatic System Control	* **25 and LATER** FLAPS & SPOILER Flap Selector & Control Spoiler Control
3 FLAPS	Flap Selector & Control	
4 YAW TRIM	Rudder Trim & Indicator Light MA Shifter Actuator	
5 ROLL TRIM & MANEUV STAB CONT	Aileron Trim Actuator & Control Pitch Trim Actuator & Control Stabilator Electric Actuator Hydraulic Transfer Valves (Hydraulic Dump Valve—Tandem Aircraft)	
6 MANEUV STAB	Emergency Hydraulic Pump Control (Stabilator Electric Actuator Control—Tandem Aircraft)	
7 START CONT	Engine Start-Stop Start Engine Crank Rotor Test Air Start	
8 FUEL & CANOPY WARN	Fuel Pressure Warning Lights Canopy Open Warning Light Canopy Seal Inflation Control	
9 FUEL CONT	Fuel Control Emergency System Control and Indicator Light	
10 FUEL FILTER DE-ICE	Complete Fuel Filter De-Icing System	
11 ENG SCRN	Engine Screens Control & Warning Light	
12 TANK JETT	Pylon Tank Jettison	

Figure 1—19 (Sheet 3 of 4)

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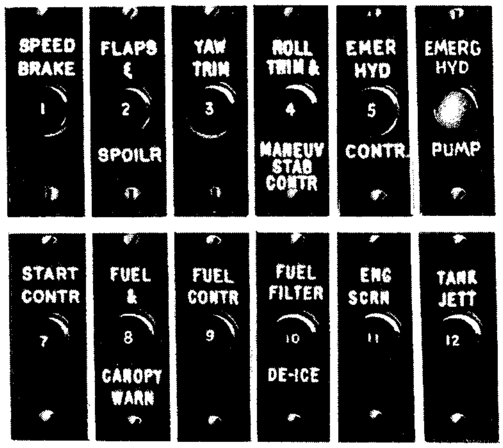


AUXILIARY CIRCUIT BREAKER PANEL

30 and LATER

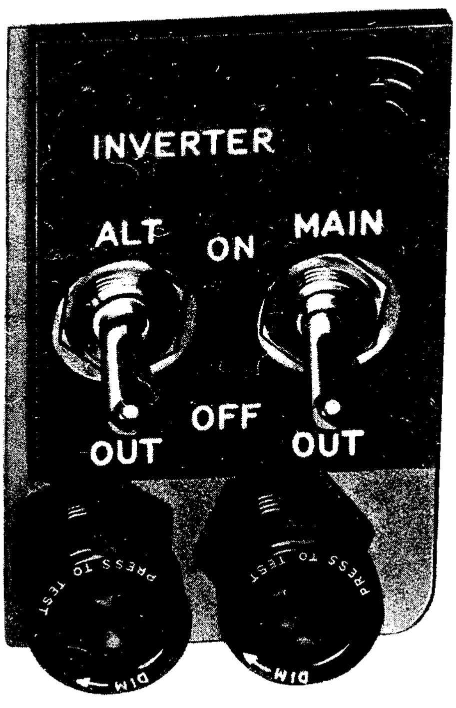
(Location-Top of Left Console)

CIRCUIT BREAKER	OPERATING UNIT AND/OR CONTROL	REMARKS
1 SPEED BRAKE PNEU CONTR	Speed Brake Control Pneumatic System Control	
2 FLAPS & SPOILER	Flap Selector & Control Spoiler Control	
3 YAW TRIM	Rudder Trim & Indicator Light MA Shifter Control	
4 ROLL TRIM & MANEUV STAB	Aileron Trim Control Pitch Trim Actuator & Control	
5 EMERG HYD CONT	Emergency Hyd Pump Control	
6 EMERG HYD PUMP	Emergency Hyd Pump (Stabilator Electric Actuator Cont—Non-Tandem)	
7 START CONT	Engine Start-Stop Start Engine Crank Rotor Test Air Start	
8 FUEL & CANOPY WARN	Fuel Pressure Warning Lights Canopy Open Warning Light Canopy Seal Inflation Control	
9 FUEL CONTR	Fuel Control Emergency System Control and Indicator Light	
*10 LAND GEAR ARM SAFETY	Landing Gear Selector Armament Safety Override	*10 5 thru 35 NOT MODIFIED BY [562] FUEL FILTER DE-ICE Complete Fuel De-Icing System
*11 ENG SCREEN & FUEL FILTER DE-ICE	Engine Screen Control Complete Fuel Filter De-Ice System	*11 5 thru 35 NOT MODIFIED BY [562] ENG SCRN Engine Screen Control & Warning Light
12 TANK JETT	PYlon Tank Jettison	

Figure 1-19 (Sheet 4 of 4)

power (if the battery switch is ON). During normal operation (main inverter) an automatic power transfer system is incorporated for the purpose of converting single-phase power into three-phase power for the flight instruments. During normal operation only the single-phase (main) inverter is in operation. The three-phase (alternate) inverter is maintained as a standby source of power for the flight instruments. A switch for control of each inverter, and a warning light for indication of failure of each inverter are provided in the cockpit. The alternate inverter must be manually selected.

INVERTER *Control Panel*



Inverter Switches.

The inverter switches (12, figure 1-4) are two-position toggle switches placarded MAIN and ALT. Each switch has two placarded positions, ON and OFF. With either switch in the OFF position, the respective inverter will be turned off. The MAIN (1 PHASE) switch connects power from the NO. 1 secondary bus to the single-phase inverter while the ALT (3 PHASE) switch connects primary bus power to the 3-phase inverter. The fuel quantity indicator may oscillate, if both inverter switches are on.

Inverter Failure Warning Lights.

The inverter failure warning lights (12, figure 1-4) on the main instrument panel are red press-to-test type lights placarded ALT OUT and MAIN OUT and are powered by the DC primary bus. Illumination of a light indicates the failure of the respective inverter or the inverter is not ON. If the main inverter switch is then placed OFF both lights will illuminate. Illumination of the MAIN OUT light when the MAIN inverter switch is ON indicates the MAIN inverter has failed. Illumination of the ALT OUT light when the ALT inverter switch is ON indicates the ALT (alternate) inverter has failed. When switching an inverter on, a few seconds are required for the respective light to go out.

Cockpit AC Fuse Box.

The following AC circuits are protected by individual fuses located in the AC fuse box on the right side of the cockpit above the aft end of the console. A placard inside the fuse box door identifies each fuse and indicates the correct amperage.

WINDSHIELD DEFROST & TEMPERATURE CONTROL

IMC CAMERA MAGAZINE

VIEW FINDER

AUTO PILOT

*FUEL LEVEL INDICATOR (2 fuses)

*OIL PRESSURE INDICATOR

AN/APX-25

1 PHASE INVERTER WARNING

POWER CONVERTER

INSTRUMENT TRANSFER (2 fuses)

*ATTITUDE INDICATOR

*HYDRAULIC PRESSURE INDICATOR

*HEADING INDICATOR (2 fuses)

Note

During normal operation all circuits above are powered by the main inverter. If the main inverter fails and the ALT INVERTER switch is placed at ON, only those circuits preceded by an asterisk (*) will be powered by the alternate inverter.

PNEUMATIC POWER SUPPLY SYSTEM

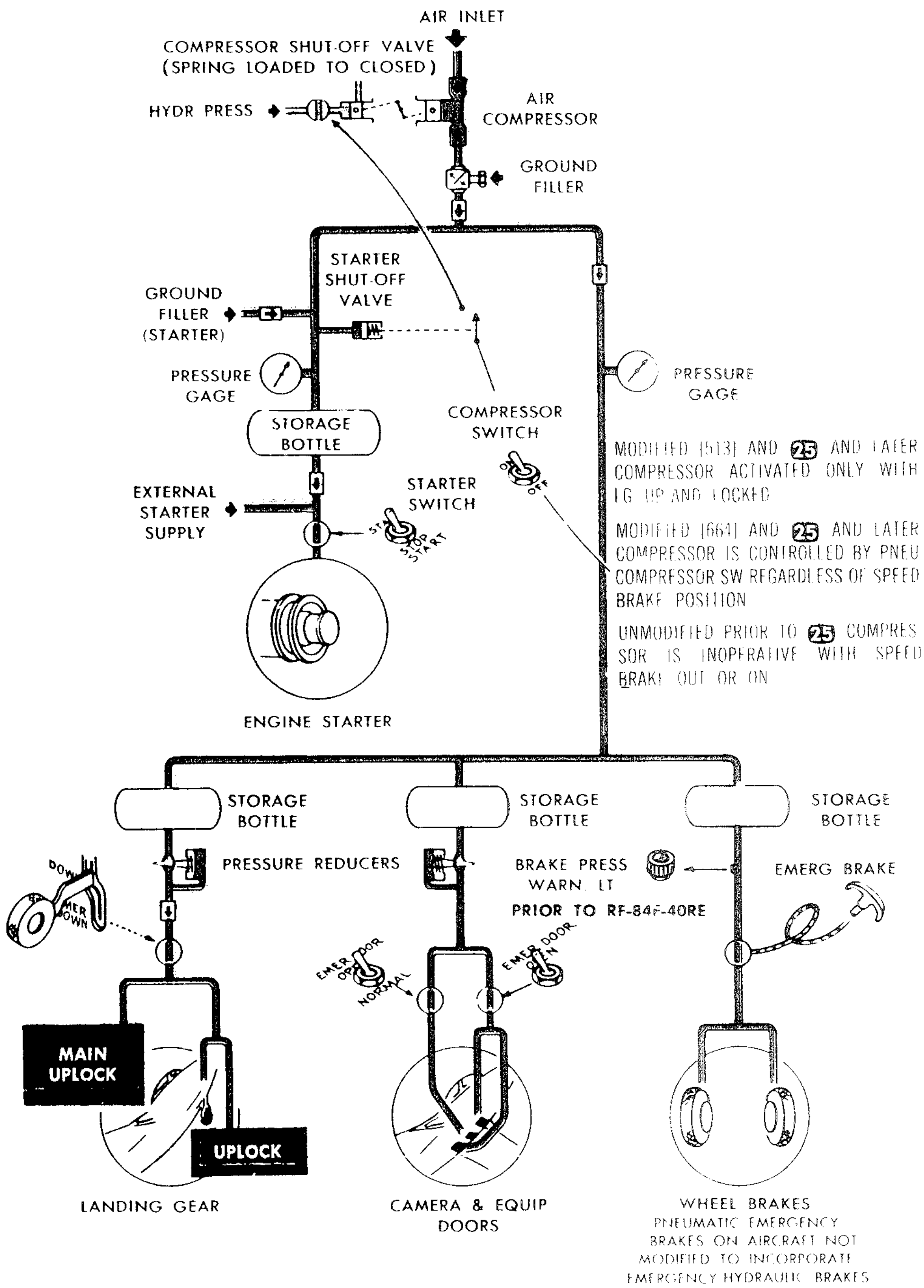


Figure 1—20

PNEUMATIC POWER SUPPLY SYSTEM.

The pneumatic power supply system (figure 1–20) is provided to supply air pressure to recharge the engine starter air storage bottle and to provide an emergency means of extending the landing gear and an emergency means of opening the camera doors. Air is bled from the engine compressor, then pressurized by a hydraulically operated compressor and stored in four storage bottles. The system is fully automatic; the hydraulic pressure from the utility hydraulic system to the compressor motor is shut off when air pressure in the storage bottles reaches 3,000 PSI. Manual shut-off is also provided. All bottles are charged simultaneously, in approximately 30 minutes. When all bottles are charged the hydraulic shut-off valve is closed shutting off the compressor. The hydraulic shut-off valve is electrically controlled with primary bus power which is available only when the speed brakes are closed on 5 thru 20 aircraft up to serial NO. 52-7304 not modified by [664]. However, on aircraft modified by [664] and 25 and later aircraft from serial NO. 52-7305, this feature has been eliminated and the compressor switch alone controls the compressor. On aircraft modified by [513] and 25 and later aircraft, the pneumatic compressor will be operative only when the landing gear is up and locked. This system is provided to cut out the hydraulic motor of the pneumatic compressor during periods when there is a greater than normal demand for pressure in the utility hydraulic system. The automatic cut-off switch in the left wheel well can be actuated on the ground to permit compressor operation with the landing gear extended.

CAUTION

The compressor should not be operated for more than 30 minutes continuously to prevent overheating and possible compressor explosion. Therefore, if the storage bottles take an excessive amount of time to refill, the compressor should be shut off.

Note

There is an automatic safety feature which will stop the storage bottle charging cycle in the event of compressor overheating. The compressor will automatically resume operation when the temperature reduces. Therefore, charging time may be more than the normal time required to charge the system. However, there is no indication by which the pilot can determine this condition.

PNEUMATIC COMPRESSOR SWITCH.

The pneumatic compressor switch (figure 1–9) is provided so that the compressor can be turned off during take-offs and landings or at any time when a demand is made for hydraulic pressure from the utility system. The switch has two positions and is marked PNEU COMP ON and OFF on 45 and later aircraft and PNEUMATIC COMPRESSOR and OFF on 5 thru 40 aircraft. The ON position, on 5 thru 20 aircraft opens the spring-loaded to closed hydraulic shut-off valve that controls the flow of hydraulic fluid to the pneumatic compressor drive motor, if the speed brake switch is in the IN position. However, on some 5 thru 20 aircraft, the pneumatic compressor is not affected by speed brake operation and is controlled solely by the pneumatic compressor switch. The pneumatic compressor will be automatically turned off and remain inoperative on aircraft modified by [513] and 25 and later aircraft when the landing gear is extended. However, the automatic cut-off switch can be manually actuated on the ground to permit compressor operation with the landing gear extended.

HYDRAULIC POWER SUPPLY SYSTEM.

The hydraulic power supply system consists of three systems; utility, power and emergency (figures 1–21 and 1–22). The utility and power systems are complete and independent of each other in that each system has its own reservoir, pressure pump, pressure indication and plumbing. The emergency system is simply a second source of pressure for the power system. The utility and power system are powered by individual, variable delivery, engine driven pumps. Each pump instantly and positively adjusts delivery of its output to the demands of its system and pressure is supplied only as required by the hydraulic system. When pressure is not required the suction ports of the pumps' pistons are closed. As pressure is demanded by the power or utility system, the suction ports of the respective pump open to supply the pump with hydraulic fluid from its reservoir and meet the pressure demands of its system. The utility system has a pressurized zero-G type reservoir and an accumulator ground charged with 500 PSI of air. The power system has a pressurized reservoir and an accumulator ground charged with 500 PSI of air, to maintain a pressurized fluid reserve in the system to meet extreme operational demands. The emergency system is powered by an electrically driven pump powered by the primary bus. The emergency hydraulic pump is supplied with hydraulic fluid from the power system reservoir. All three systems incorporate accumulators to satisfy immediate greater than normal demand. The power system is used exclusively for operation of one side of the tandem actuators for the stabilizer and ailerons. The utility system not only powers one side of the stabilizer and aileron actuators but in addition provides power for the spoilers, rudder, wing flaps, speed brakes, power wheel brakes, landing gear, air refueling system (on 5 thru 35 aircraft not modified by [557]), engine duct screens, pneumatic compressor and ATO release. Each of these systems are described under applicable headings. Operation of the hydraulic systems is automatic and only the movement of the selected switch or control is necessary to actuate the selected system. During normal operation both utility and power hydraulic pressures are supplied to the aileron, and stabilator tandem actuators through separate hydraulic lines so that if one hydraulic system fails, the actuators will still be powered by the engine driven pump of the other system. Should both the power and utility system supply fail, the pilot must manually switch to the emergency hydraulic pump to supply pressure to the surface control tandem actuators through the power system hydraulic lines. If the utility system fails the rudder will be controlled through mechanical linkage and the power brakes, flaps, spoilers, speed brakes, engine duct screens, pneumatic compressor, air refueling equipment, camera doors and ATO release will be inoperative, even though the power or emergency hydraulic pressure is available.

A compressed air system is provided for emergency extension of the landing gear and operation of the camera equipment doors.

HYDRAULIC PRESSURE GAGE.

A single hydraulic pressure gage (27, figure 1—4) provided with two indicator needles, indicates hydraulic pressure for the power and utility systems. The face of the gage is divided into two vertical half sections, with one half marked UTILITY for utility system pressure and the other half marked POWER for the power system pressure. Each half section of the gage face is marked to show hydraulic pressure ranging from 0 to 2,000 PSI. Normal operating pressure, of the system, is approximately 1,500 PSI for both the utility and power system, depending on the amount of equipment operating. Emergency system pressure is indicated on the POWER half of the gage face. When the emergency pump is operating, the POWER indicator needle will show an emergency operating pressure of approximately 1,500 PSI in the power system. The hydraulic pressure gage is powered by the AC electrical system and will operate only when the ALT or MAIN inverter switch is ON and the selected inverter is operating.

FLIGHT CONTROL SYSTEM.

The primary flight control system uses a movable horizontal stabilator for pitch control, ailerons and spoilers for roll control and a rudder for yaw control. The stabilator is fully powered hydraulically and cannot be operated without hydraulic pressure. The ailerons are hydraulically boosted and although manual operation is possible without hydraulic pressure, a high-force is required due to friction and airloads. Manual operation of the rudder is adequate without hydraulic pressure. The stabilator and ailerons utilize tandem actuators which are powered by both the power and utility hydraulic system. The power hydraulic system powers the power side of the stabilizer and aileron tandem actuators exclusively. The emergency hydraulic system is available as an alternate source of hydraulic pressure for the power hydraulic system. The utility hydraulic system powers the utility side of the stabilizer and aileron tandem actuators in addition to the spoiler actuator, rudder actuator, wing flaps, speed brakes, wheel brakes, landing gear, air refueling system, engine duct screens, pneumatic compressor and ATO release. If the utility system pressure should fail, the ailerons and stabilator will actuate from the power system. The rudder will be controlled only through mechanical linkage and the spoilers will be inoperative as these surfaces are actuated by utility pressure only. If the power system pressure should fail, all surfaces will operate with utility system pressure. In the event of failure of both the utility and power hydraulic systems the emergency hydraulic system must be manually selected for stabilator and aileron operation. Pressure gages are provided to indicate pressure in each of the systems.

STABILATOR.

The stabilator is operated by a fully powered hydraulic actuator controlled by a mechanical linkage attached to the control stick. There is a direct mechanical linkage between the control stick and stabilator. However, due to the surface hinge moments, direct manual control is not possible without hydraulic power. Since the powered systems are irreversible (i.e., air loads on the surfaces are not felt on the controls) control feel is simulated by an artificial feel unit in each of the primary control systems. These units vary control feel in proportion to stick or rudder pedal deflection only and will not be affected by aircraft speed or altitude. The artificial feel unit is a spring capsule designed to give the pilot a sense of control feel by increasing the control forces as the controls are moved. The units are also used when trimming the aircraft about

UTILITY HYDRAULIC POWER SUPPLY

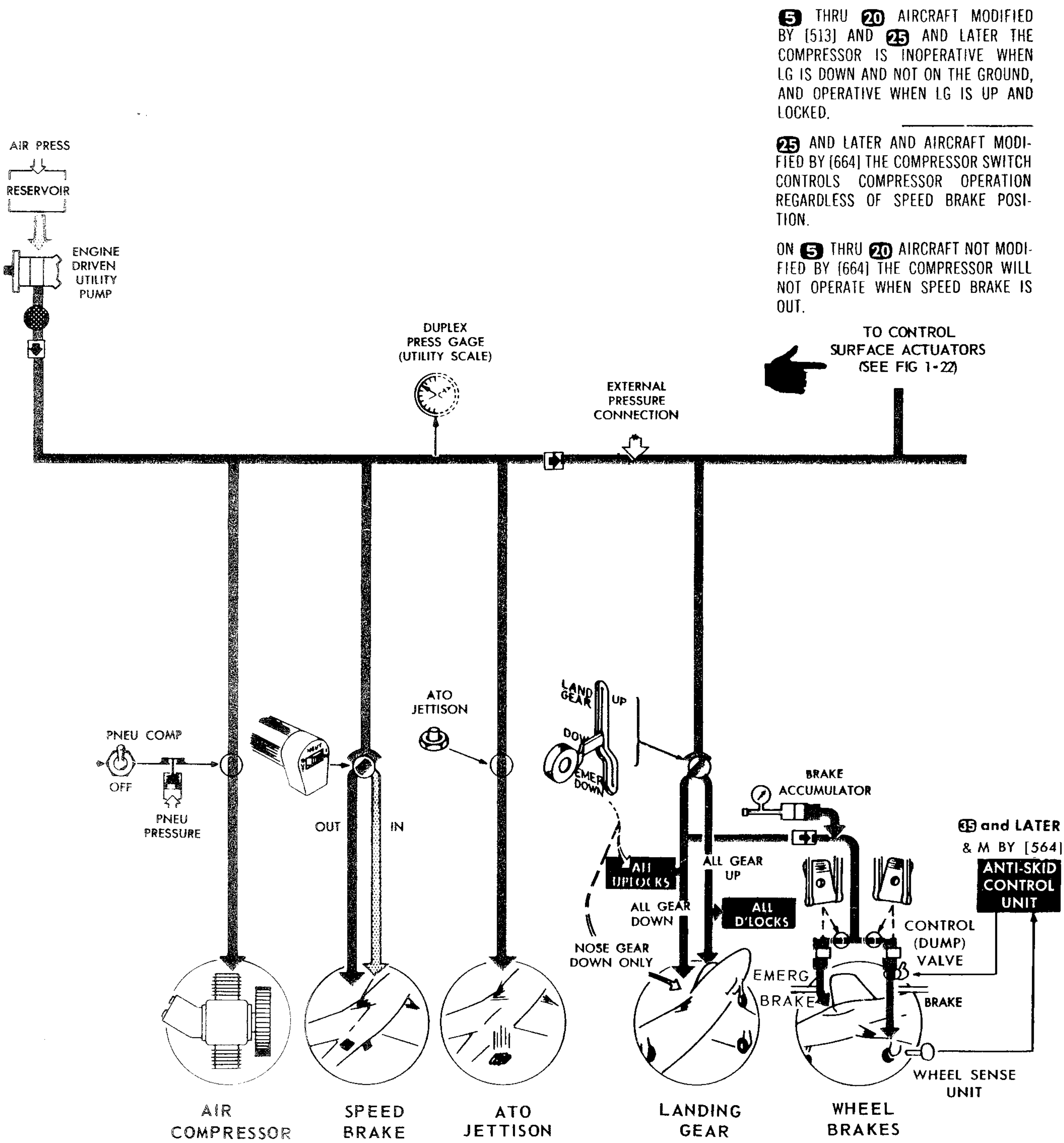
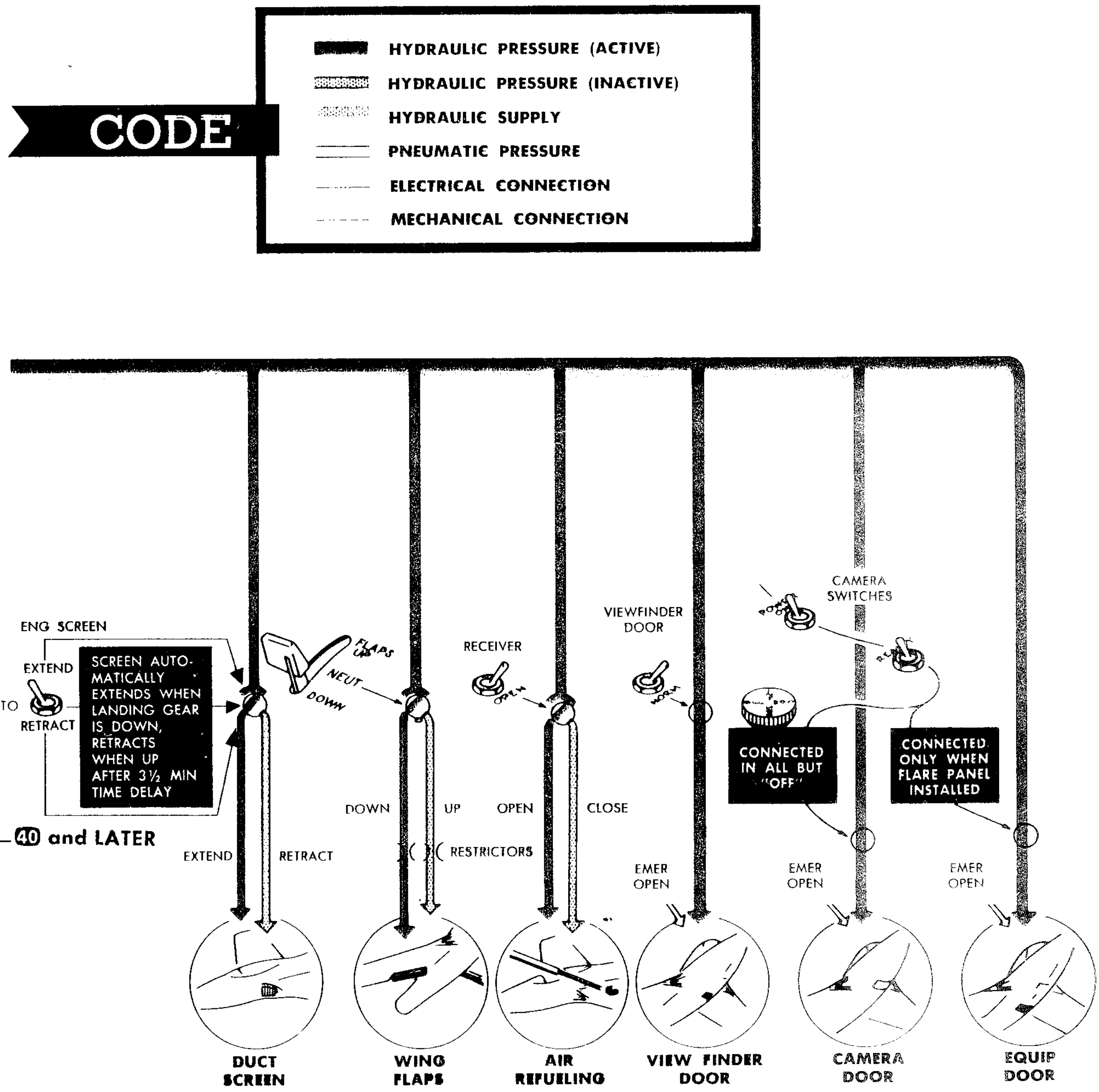


Figure 1—21



CODE
HYDRAULIC PRESSURE (ACTIVE)
HYDRAULIC PRESSURE (INACTIVE)
HYDRAULIC SUPPLY
PNEUMATIC PRESSURE
ELECTRICAL CONNECTION
MECHANICAL CONNECTION
ENG SCREEN
EXTEND
AUTO
RETRACT
SCREEN AUTOMATICALLY EXTENDS WHEN LANDING GEAR IS DOWN, RETRACTS WHEN UP AFTER 3½ MIN TIME DELAY
40 and LATER
EXTEND
RETRACT
FLAPS UP
NEUT
DOWN
DOWN
UP
RESTRICTORS
RECEIVER
OPEN
OPEN
CLOSE
VIEWFINDER DOOR
EMER OPEN
CAMERA SWITCHES
CONNECTED IN ALL BUT "OFF"
CONNECTED ONLY WHEN FLARE PANEL INSTALLED
EMER OPEN
EMER OPEN
DUCT SCREEN
WING FLAPS
AIR REFUELING
VIEW FINDER DOOR
CAMERA DOOR
EQUIP DOOR

the roll, yaw and pitch axes. This is accomplished with an actuator powered from the primary bus which repositions the spring capsule so that when the controls are moved to trim the aircraft, the spring capsule is moved to a no-load position and the aircraft will be trimmed about the axes with no load on the cockpit controls. A control stick damper (pitch only) is incorporated to restrict the speed at which the control stick can be moved to assist in preventing over-control and porpoising. The damper is factory set and cannot be adjusted for individual pilot feel during flight. A mechanical advantage shifter is incorporated in the stabilator linkage to decrease the control stick sensitivity when the landing gear is retracted. When the landing gear is not up and locked, the stabilator moves through an arc of approximately 16½ degrees, leading edge down (aircraft nose up) to approximately 4½ degrees, leading edge up (aircraft nose down). This range of movement during high speed flight could cause over-controlling and porpoising. Therefore, when the landing gear reaches the up and locked position, a circuit is completed to an electric actuator which cycles the mechanical advantage shifter and reduces the control stick to control surface ratio from 1:1 to 2:1. The change in ratio reduces the travel of the stabilator to an arc of approximately nine degrees leading edge down (airplane nose up) to 3½ degrees leading edge up (airplane nose down). Control stick travel remains approximately the same. Therefore, stick travel per degree of stabilator travel is increased to reduce control sensitivity by permitting finer control adjustment. The change in ratio is automatic and cannot be controlled by the pilot. When the right main landing gear outer door is up and locked, the stabilator control system begins to shift to the 2:1 ratio. Approximately eight seconds after the landing gear is up and locked, the mechanical advantage actuator has completed its cycle to change the stabilator controls into the 2:1 ratio. This requires a slight aft movement of the control stick which may feel to the pilot like a small nose down trim change. Conversely, when the landing gear is extended and the mechanical advantage ratio changes to 1:1, the opposite effect is noticeable. An indicator light is provided to indicate when the mechanical advantage actuator is shifting or is not in the proper ratio. A force of approximately 25 pounds is required to move the control stick to either extreme position while approximately three pounds are required to move it out of the neutral position. There are no controllable trim tabs provided.

CONTROL STICK.

The control stick (figure 1—23) is conventional and incorporates a hand grip with the following controls: trim switch, trigger, "roger" button for the radar beacon installation, extra picture switch camera external operate switch and the autopilot release switch. These switches are discussed under the applicable systems.

RUDDER PEDALS.

The rudder pedals can be folded down for pilot comfort. Depressing the release lever on the inboard side of each pedal allows the foot pad of the pedal to fold toward the pilot. This allows the pilot to rest his heel on the base of the pedal and still maintain rudder control with his heel. Brakes cannot be applied with the pedals in the folded position. Brake application is accomplished by toe action in the normal manner when the pedals are in the normal position.

RUDDER PEDAL ADJUSTING CONTROL.

The rudder pedal adjusting control (figure 1—4) adjusts both rudder pedals simultaneously to the desired fore and aft position. The control is marked PEDAL ADJUST.

TRIM SWITCH (stick grip).

The trim switch (figure 1—23) on the stick grip is a five position switch, spring-loaded to the center or off position and is powered by the primary bus when the alternate trim switch is in the NORM position. The switch trims the aircraft about the pitch and roll axis. The fore and aft positions of the switch adjust the artificial feel unit in the stabilizer control to a selected no load position. The left and right lateral positions of the switch adjust the artificial feel device in the aileron control system so as to change the neutral position of the spring capsule to a selected no load position.

FLIGHT CONTROLS HYDRAULIC SYSTEM

Schematic TANDEM AIRCRAFT

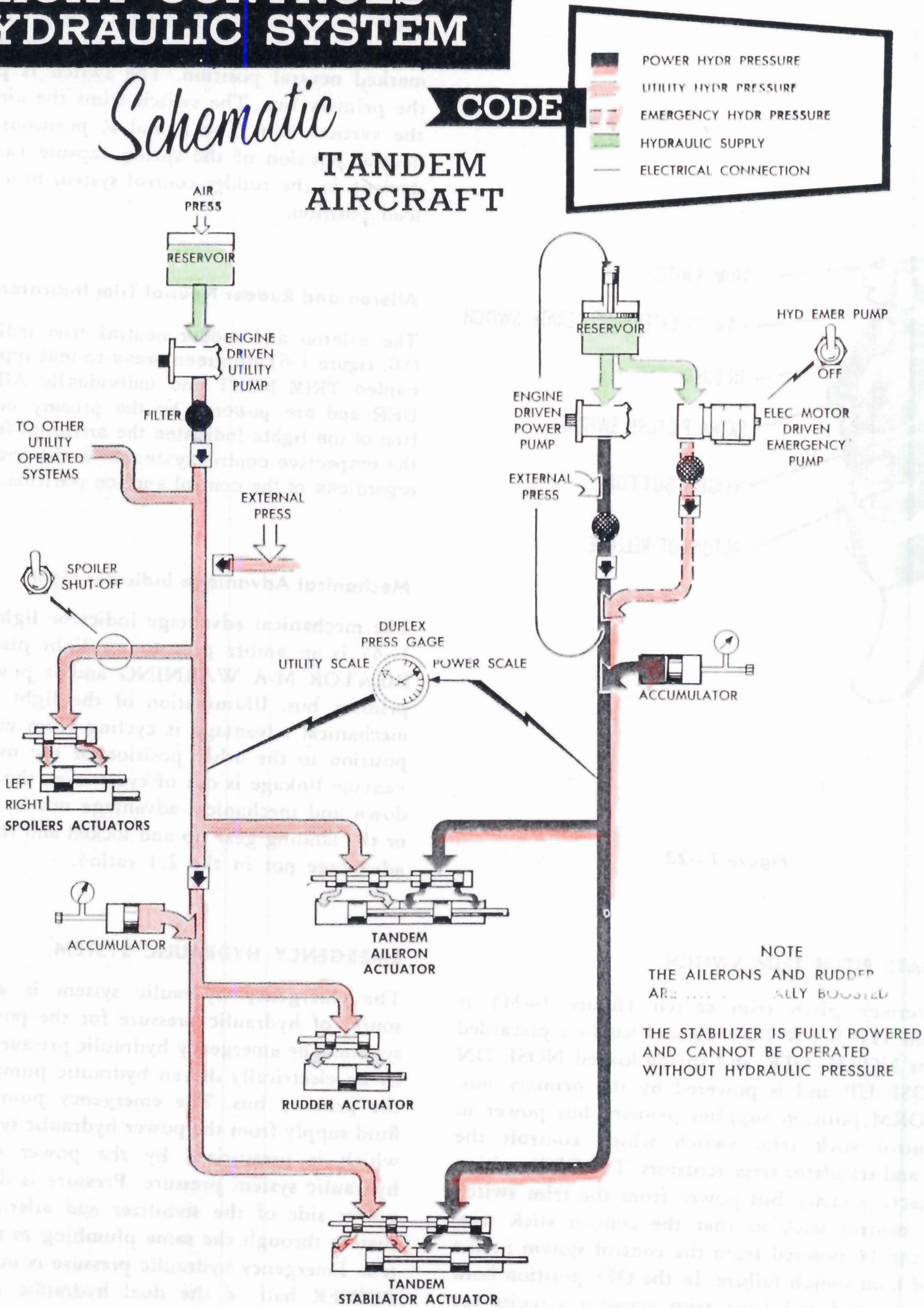


Figure 1—22

CONTROL STICK GRIP

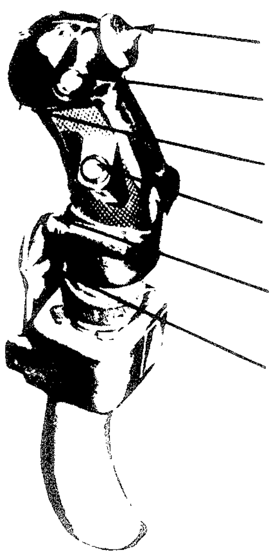


Figure 1—23.

ALTERNATE PITCH TRIM SWITCH.

The alternate pitch trim switch (figure 1—24) is placarded STICK SW POWER and has four placarded positions NORM, OFF, and spring-loaded NOSE DN and NOSE UP and is powered by the primary bus. The NORM position supplies primary bus power to the control stick trim switch which controls the aileron and stabilator trim actuators. The OFF position disconnects primary bus power from the trim switch on the control stick so that the control stick trim switch can be isolated from the control system in the event of trim switch failure. In the OFF position both the aileron and stabilator trim actuator circuits are inoperative. The NOSE DN and NOSE UP positions actuate the stabilator trim actuator to trim the aircraft in the pitch axis in the event of failure of the control stick trim switch.

RUDDER TRIM SWITCH.

The rudder trim switch (figure 1—24) placarded YAW is a three position toggle switch with positions L (left) and R (right) and spring-loaded to an unmarked neutral position. The switch is powered by the primary bus. The switch trims the aircraft about the vertical axis. The L and R positions adjust the neutral position of the spring capsule (artificial feel device) in the rudder control system to a selected no load position.

Aileron and Rudder Neutral Trim Indicator Lights.

The aileron and rudder neutral trim indicator lights (10, figure 1-6) are green press-to-test type lights placarded TRIM NEUT and individually AIL and RUDDER and are powered by the primary bus. Illumination of the lights indicates the artificial feel device in the respective control system is in the neutral position regardless of the control surface position.

Mechanical Advantage Indicator Light.

The mechanical advantage indicator light (28, figure 1—4) is an amber press-to-test light placarded STABILATOR M-A WARNING and is powered by the primary bus. Illumination of the light indicates the mechanical advantage is cycling from either extreme position to the other position or the mechanical advantage linkage is out of cycle (i.e., the landing gear down and mechanical advantage not in the 1:1 ratio or the landing gear up and locked and the mechanical advantage not in the 2:1 ratio).

EMERGENCY HYDRAULIC SYSTEM.

The emergency hydraulic system is an emergency source of hydraulic pressure for the power hydraulic system. The emergency hydraulic pressure is developed by an electrically driven hydraulic pump powered by the primary bus. The emergency pump receives its fluid supply from the power hydraulic system reservoir which is pressurized by the power or emergency hydraulic system pressure. Pressure is directed to the power side of the stabilizer and aileron tandem actuators through the same plumbing as the power system. Emergency hydraulic pressure is indicated on the POWER half of the dual hydraulic pressure gage. The system is manually selected by the pilot in the event of engine failure, or similar malfunction which would render the engine driven hydraulic pumps inoperative.

FLIGHT CONTROL SWITCHES

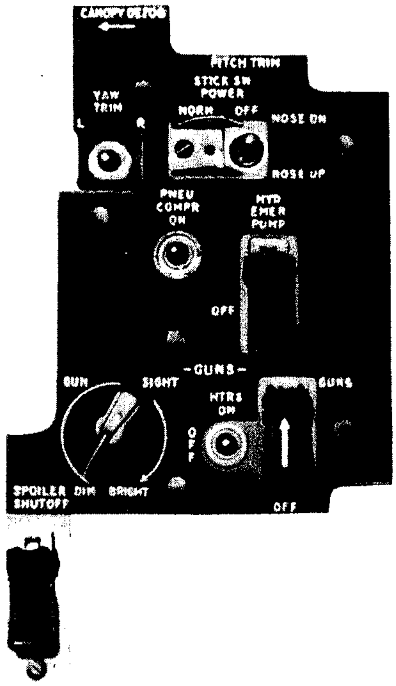


Figure 1-24

Emergency Hydraulic Pump Switch.

The emergency hydraulic pump switch (figure 1–24) is a two position toggle switch with positions placarded HYD EMER PUMP and OFF and is powered by the primary bus. The HYD EMER PUMP position energizes the pump and the OFF position deenergizes the pump.

Note

> With both generators inoperative and no other electrical equipment operating, the emergency hydraulic pump will operate from zero to approximately 15 minutes, depending on the condition of the aircraft battery. See figure 7–5, Electrical Load Chart.

WING FLAPS.

The wing flaps are partial span, plain, trailing edge type and extend from the aileron to the fuselage on each wing. The flaps are extended and retracted with hydraulic pressure and are controlled by an electrically operated selector valve powered from the primary bus. The flaps are locked in the up position by a mechanical lock in each of the actuating cylinders and are kept in the down position by hydraulic pressure. Intermediate positions of the flap may be selected and the position will be shown on the flap position indicator. A restrictor is installed in the flap hydraulic system so that seven to 10 seconds are required to retract or extend the flaps. The flaps will not extend, or if extended, will retract if the airspeed is above approximately 225 KIAS. The flaps are not mechanically connected but are hydraulically synchronized so that they will extend simultaneously in flight. On 5 thru 20 aircraft the flap system is synchronized to within a maximum of 10 degrees differential during full travel and a differential of 10 degrees at the 50 per cent position during ground operation. On 25 and later aircraft, this differential is a maximum of five degrees during full travel and a differential of six degrees at the 50 per cent position. However, with a cross-wind the differential may be greater. The flaps will synchronize due to air loads at a speed of approximately 80 knots. The flap position indicator may increase or decrease slightly when air loads are applied to the flaps and they become synchronized during the take-off run.

WING FLAP LEVER.

The wing flap lever (figure 1—10) has threee positions, UP, NEUT and DOWN. The UP position electrically positions the hydraulic selector valve to direct hydraulic pressure to the up side of the flap actuators. When the flaps are fully up and the mechanical locks in the actuating cylinders are locked, the flap selector valve automatically returns to the neutral position which relieves the pressure on the up side of the actuating cylinder. In the NEUT position, hydraulic pressure in the down side of the flap actuating cylinder is trapped while pressure in the up side is relieved. This position is used for partial flap extension. When the flaps are lowered to the desired position the landing flap lever is returned to NEUT. The trapped hydraulic fluid maintains the flap in the desired position. The DOWN position electrically positions the hydraulic selector valve to direct hydraulic pressure to the down side of the flap actuating cylinders where it is maintained until the lever is repositioned.

WING FLAP POSITION INDICATOR.

The wing flap position indicator (7, figure 1—5) powered by the primary bus shows the position of the landing flap in per cent of travel. A scale on the gage displays a range from zero to 100 per cent down.

SPEED BRAKES.

Speed brakes installed on each side of the aft fuselage are designed as drag increasing devices for use during "let down" and maneuvers. The speed brakes are powered by the utility hydraulic system and electrically controlled with a pilot operated switch powered by the primary bus. A solenoid type hydraulic selector valve directs fluid to the speed brake actuating cylinders to open or close, or hold the speed brakes in any intermediate position. In the event of utility hydraulic or electric failure, the speed brakes will not open, or if open will close under air loads. On 5 thru 20 aircraft not modified by {664} the pneumatic compressor will not operate unless the speed brakes are fully closed.

SPEED BRAKE SWITCH.

The speed brake switch (figure 1—10) is a three position sliding switch on the throttle control with positions placarded OUT, NEUT and IN and is powered by the primary bus. The switch positions a hydraulic control valve to port utility hydraulic pressure for speed brake operation. The OUT position indexes the control valve to apply pressure to open and hold the speed brakes open. The IN position indexes the control valve to close the speed brakes and hold them closed. The NEUT position indexes the control valve to trap hydraulic fluid in the hydraulic lines so that the speed brake will remain in any selected intermediate position. When the speed brakes are closed during flight, the switch should be left in the IN position as hydraulic pressure may bleed off sufficiently to allow the speed brakes to open slightly if the NEUT position is selected.

SPOILERS.

Spoilers are located on the upper surface of both wings, immediately forward of the flaps and are automatically actuated through the control stick lateral deflection for UP aileron only. The spoilers increase the roll performance (rate of roll) of the aircraft during high speed, low altitude flight. At the intermediate speeds the spoilers are less effective, but together with aileron displacement, rate of roll is more than adequate. At low speeds the spoilers are not effective, however, aileron effect is adequate for low speed flight. As the aileron moves up from zero to 10 degrees the corresponding spoiler moves from zero to approximately 45 degrees. Spoiler extension of approximately 45 degrees remains constant for the remainder of aileron travel. The spoilers are powered by the utility hydraulic system through a normally open shut-off valve powered by the primary bus. A switch is also provided in the cockpit so that the spoilers can be turned off during flight.

SPOILER SHUT-OFF SWITCH.

The spoiler shut-off switch (4, figure 1–6) is a two position toggle switch placarded SPOILER SHUT-OFF, guarded in the on position and powered by the primary bus. In the guarded (ON) position the normally open hydraulic valve is deenergized and utility hydraulic pressure is available for spoiler operation. When the guard is lifted and the switch positioned to OFF, the hydraulic valve is energized and closes, shutting off hydraulic fluid and deactivating the spoilers.

LANDING GEAR SYSTEM.

The retractable tricycle landing gear is hydraulically operated and electrically controlled by a landing gear selector valve powered by the primary bus. The valve is spring-loaded to the neutral position and in the event of primary bus failure hydraulic pressure will not be available for landing gear operation or boost brakes. Each strut is enclosed by fairing doors that are flush with the contour of the wing and fuselage when the gear is retracted and remain open when the landing gear is extended. The nose wheel shock strut is mechanically shrunk as it is retracted and automatically returns to its fully extended position when the gear is let down. Mechanical locks secure the three struts in the retracted or extended positions. Inadvertent retraction of the gear when the aircraft is on the ground is prevented by a solenoid lock which automatically prevents moving the landing gear selector handle. A switch is provided to override this safety system in emergencies. The main landing gear downlocks are spring-loaded to the locked position and are unlocked by hydraulic pressure when the struts are retracted. In an emergency the main gear uplocks are released by pneumatic pressure and the gears drop to the locked down position by gravity. During normal operation the nose gear is retracted and extended with hydraulic pressure. In an emergency the nose gear is unlocked and extended with pneumatic pressure. Ground safety locks are provided for maintenance purposes only. Landing gear position indicator lights are provided to inform the pilot of the position of the landing gear struts. A landing gear warning system is also provided. On 5 thru 25 aircraft an armament safety switch, incorporated on the main landing gear struts, prevents the guns from being fired as long as the weight of the aircraft keeps the shock struts compressed. On 30 and later aircraft the armament safety switch is incorporated in the landing gear selector handle. Also on 25 and later aircraft and aircraft modified by [513] a pneumatic compressor shut-off switch is actuated by the main landing gear uplock so as to cut the pneumatic compressor off when the landing gear is released from the up and locked position.

LANDING GEAR SELECTOR HANDLE.

The landing gear selector handle (2, figure 1–5) controls the extension and retraction of the landing gear for both the normal and emergency systems. The normal system control valves and indicators are energized from the primary bus. The valve is spring-loaded to the neutral position and in the event of primary bus failure, hydraulic pressure will not be available for landing gear operation or power brakes. The selector has a plastic wheel shaped knob which incorporates a red warning light. The selector handle has three positions; UP, DOWN and EMERG DOWN. The DOWN position electrically positions the hydraulic control valve to apply pressure to extend the landing gear and to energize the brake boosters. Hydraulic power is supplied to the brakes only when the landing gear selector handle is in the DOWN or EMERG DOWN position and primary bus power is available. An electrical solenoid prevents the landing gear selector from being moved from the DOWN position, with the shock struts compressed. This safety can be overridden if it is desirable to retract the gear before the weight is off the landing gear struts. The UP position electrically positions the hydraulic control valve so that hydraulic pressure releases the downlocks, retracts the landing gear and closes the fairing doors. Hydraulic pressure is maintained in the system as long as the selector handle is in the UP position. The EMERG DOWN position is used whenever the normal system fails to lower the gear. This position mechanically opens the pneumatic valve and compressed air unlocks all three gears and extends the nose gear. The main gears drop and lock by gravity. To position the landing gear selector handle from one position to another position, the handle must be pulled out toward the pilot first, then moved to the new position. When going from the DOWN to the EMERG DOWN position the handle must be pulled out toward the pilot first, then pushed inboard and down to the EMERG DOWN position. Once the EMERG DOWN position has been selected the handle cannot be returned to the DOWN position until the handle lock has been released by the ground crew and the system has been bled completely. An armament safety switch has been incorporated in the landing gear selector handle on 30 and later aircraft so that the armament system cannot be energized when the landing gear selector handle is in the DOWN or EMERG DOWN position.

CAUTION

Once the landing gear is extended by the emergency procedure, the emergency landing gear release switch will not retract the gear. In the event that only the main gear can be locked down with the main system, the possibility exists of back pressure in the return hydraulic lines unlocking the main landing gear whenever the nose gear is lowered by means of the pneumatic system. However, within a short time, the pneumatic system should reactuate the downlock cylinder, relocking the main gears in the down position. Therefore, the landing gear position indicators must be rechecked to ascertain whether all gears are down and locked after the pneumatic system has been used to lower any gears.

EMERGENCY LANDING GEAR RELEASE SWITCH.

The emergency landing gear release switch (3, figure 1–5) is a guarded toggle switch placarded EMERGENCY RELEASE LG HANDLE FOR GROUND RETRACT and is powered by the primary bus. The switch is provided as an override switch for the landing gear selector handle solenoid lock. When it is necessary to retract the landing gear, while the aircraft is on the ground, the cover guard is lifted and the emergency landing gear release switch is actuated. This energizes the landing gear selector handle solenoid lock which allows the selector handle to be moved to the UP position.

CAUTION

If the landing gear shock struts are overinflated, so as to be fully extended, it will be possible to move the landing gear selector to the UP position when the aircraft is on the ground.

LANDING GEAR POSITION INDICATORS.

The landing gear position indicators (1, figure 1–5) are green press-to-test type lights placarded LEFT, NOSE and RIGHT and are powered by the primary bus. Illumination of the lights indicates that the respective landing gear is down and locked.

LANDING GEAR WARNING SYSTEM.

Indication of an unsafe landing gear is presented to the pilot by illumination of the red warning light in the landing gear handle and the sounding of the MA-1 warning signal (beeper) on the interphone.

Note

On aircraft with {726} not complied with, a warning horn is installed instead of the MA-1 warning signal (beeper).

The warning light will illuminate and the signal will sound when any of the following conditions exist:

- When the landing gear handle is DOWN and any downlock is not fully engaged.
- When the landing gear handle is UP and any uplock is not fully engaged.
- When the landing gear is retracting or extending.
- When the landing gear is not down and locked and the throttle is below 82% RPM.
- On aircraft modified by {807} the signal will sound if the gear is not locked up or locked down regardless of the power setting.

The warning light and signal normally operate together, however, the warning signal may be silenced by depressing the landing gear warning silence button.

CAUTION

Do not use the warning silencing button while gear is in transit. If the warning horn is silenced during gear transit, and the gear fails to lock, the warning signal will stay silenced and will not provide the desired warning sound. If it is determined the gear will not actuate to a safe position, the silencing button may be depressed to stop the warning signal.

Note

The warning light in the landing gear handle may be dimmed by placing the console lights rheostat between the DIM and BRIGHT positions. However, when the rheostat is in the OFF position the landing gear selector handle light will illuminate brightly.

- Aircraft with [674] not complied with, the warning light does not incorporate the dimming feature.

Landing Gear Warning Silence Button.

The landing gear warning silence button (figure 1–13) is a push type switch placarded LG WARN HORN SILENCE. Depressing the button momentarily will silence the warning signal (beeper or horn). If the warning signal has been silenced, and the throttle is opened above the 82 percent RPM position, then closed again, the signal will start to sound again.

CAUTION

Do not use warning silence button while gear is in transit. If the warning sound is silenced during gear transit, and the gear fails to lock, the warning will stay silenced and will not provide the desired warning sound. If it is determined the gear will not actuate to a safe position, the silencing button may be depressed to stop the warning signal.

Landing Gear Warning Test Button.

The landing gear warning test button (9, figure 1–5) is the push-button-type switch placarded LG HANDLE LIGHT AND HORN TEST. When the test button is depressed and held, the red light in the landing-gear-selector handle will illuminate and the warning will sound if the throttle is below the 82 per cent RPM position, regardless of the landing gear position. This test is a functional test of the warning light and sounding device only.

BRAKE SYSTEMS.

WHEEL BRAKES.

The main landing gear wheels are provided with hydraulically operated power brakes. Hydraulic pressure, supplied by the utility system, is diverted to a power brake valve for each wheel when the landing gear selector handle is in the DOWN or EMERG DOWN position and primary bus power is available. The power brake valve is operated by depressing the toe of the respective rudder pedal. The valves individually meter fluid to the brakes. In the event of utility hydraulic pressure failure an emergency brake system is provided. Parking brakes are not provided.

CAUTION

In the event of primary bus failure or the landing gear armament safety circuit breaker open, or the landing gear handle not in the Down or Emerg Down detent; the springloaded landing gear selector valve will move to the neutral position and normal power brakes are lost. Emergency brake operation is available, if selected, but the emergency reservoir will not be replenished by the utility hydraulic system, regardless of utility hydraulic pressure.

Anti-Skid Brakes. *M and 35 and Later*

Aircraft prior to 35 modified by [562] and all 35 and later aircraft are equipped with a Hytrol antiskid brake system. This system assists the pilot during the landing operation by automatically preventing wheel skidding which provides greater braking efficiency and prevents tire blowouts and flat spots. It is the intent of the antiskid brakes to give maximum action braking by preventing wheel skid. When applying the antiskid brake at high speeds, it is normal that the pilot will receive a pulsing or cycling braking effect due to the mechanics of the brake. The pilot, when flying an aircraft with this brake system installed, will note a sensation similar to that which he would get by a rapid series of brake applications and releases. That is, when applying braking action to this system, it feels as if pressure has been applied and then while still applying brake pressure, one feels a sensation as if brakes had been released. This cycling effect gradually decreases in intensity as the aircraft speed is reduced. The system consists of a flywheel inertia skid detection mechanism mounted on each main gear axle, a spring-loaded to open solenoid shutoff valve installed in each brake control pressure supply line, and the necessary electrical components required to utilize the skid control signals of the skid detection mechanism. A switch is installed on each main landing gear scissor arm which cuts out the antiskid control circuit during the flight and energizes the hydraulic brake shutoff valves from the primary bus, to the closed position, thereby preventing operation of the brakes. The cutout circuit is controlled by a relay with a hold time of 2.5 seconds. Therefore, when the aircraft touches down and the scissor switches are opened, the holding relays maintain the cutout circuit for 2.5 seconds to permit the aircraft to enter a normal landing run. Rapid (above 25 RPM) acceleration or deceleration of either of the main wheels causes a circuit to energize or deenergize the hydraulic brake shutoff valves, to the brake cylinders. If during a landing roll, either, or both of the main wheels begin to skid, a circuit is completed to energize the shutoff valve which releases the brakes. When the skid is corrected the shutoff valve is deenergized, returning brake control to the pilot. Normal braking procedures may be used during the entire

DRAG CHUTE

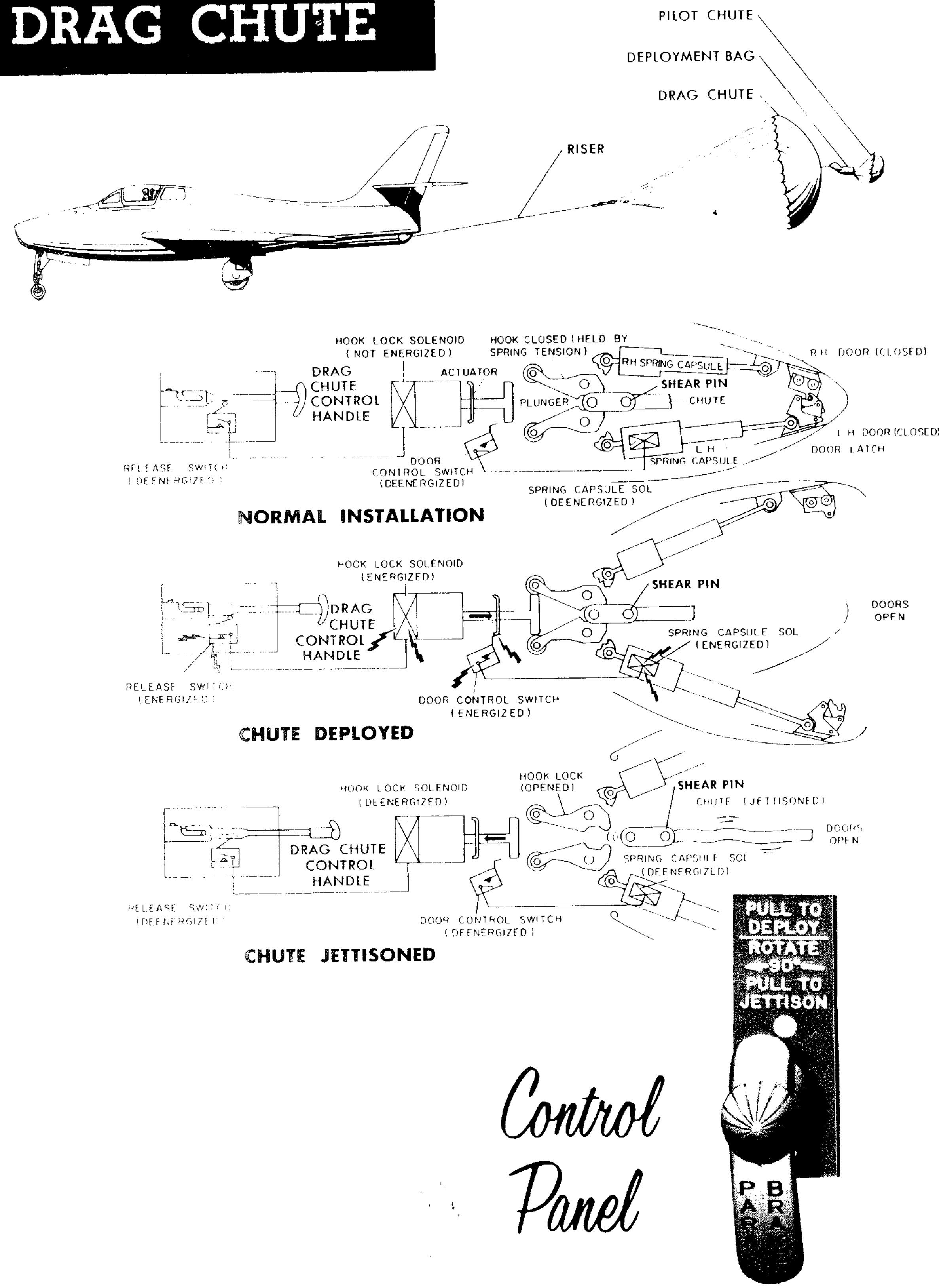


Figure 1—25

landing roll as the shutoff valves are spring-loaded to the open position and are electrically energized only when the landing gear wheels approach a skid condition. The shutoff valves are installed upstream of the emergency air brake shuttle valve to prevent any interference with the emergency brake system. If the antiskid circuit remains in the skid configuration for more than three seconds, an automatically timed relay breaks the circuit to the shutoff valves permitting normal brake operation.

Emergency Brake System (Hydraulic).

A hydraulic emergency system is provided for the emergency operation of the brakes. The emergency brake system consists of a hydraulic accumulator which is controlled by a by-pass valve. The by-pass valve is actuated by means of a cable connected to a control handle in the cockpit. To assure that the anti-skid system does not release the brakes during emergency operation, an anti-skid shut-off switch is mounted on the by-pass valve. The utility hydraulic system replenishes the emergency system automatically.

Emergency Brake Arming Handle.

The ARM EMER BRAKE handle is mounted in the cockpit between the instrument panel and the cockpit pedestal. Pulling the emergency brake arming handle with normal utility pressure merely cuts out the anti-skid feature and the normal brakes remain in the system. However, with utility hydraulic failure, actuation of the emergency brake arming handle provides a maximum of eight normal brake applications.

Note

When the emergency brake handle is pulled the anti-skid system is deactivated.

DRAG CHUTE.

A 16-foot drag chute is installed in a compartment in the ventral fin. When deployed, it extends, approximately 35 feet behind the aircraft and reduces landing roll, permitting the use of shorter runways and serves as an added safety factor for landing on slippery runways or in the event of brake system failure. The drag chute and riser cable are packed in a canvas bag and stowed in an aluminum container in the ventral fin. The compartment is equipped with doors, spring-loaded to the open position, and an electrically actuated latch lock and unlock mechanism controlled by a pull-type tee handle. A shear pin has been provided as a safeguard and will break, releasing the drag chute, in the event that the drag chute is accidentally released at speeds in excess of 220 knots. A properly packed chute will deploy at speeds as low as 60 KIAS in approximately four seconds. However, to insure consistent and proper operation, it is recommended that the chute be deployed above 75 KIAS. To keep the chute blossomed after landing roll, approximately 70 per cent RPM or sufficient thrust to attain an effective headwind of 10 knots is required. Experience has shown that 80 per cent RPM will cause scorching of the chute, when trailing directly behind the aircraft. Taxiing directly into strong headwinds should therefore be avoided. The drag chute should not be deployed before touchdown. Blossoming of the chute during landing flare-out will result in a faster rate of sink and a strong tendency to pitch nose down. Estimated data shows that chute deployment on a dry runway with 90-degree crosswinds in excess of 40 knots may cause the aircraft to leave the runway. Tests have been performed on dry runways in crosswinds up to 30 knots with excellent results. On icy runways, however, weathervaning, due to crosswinds as low as 10 knots, may result. Normally, the chute will be found to be of great value on icy runways. In the event of a go-around the drag chute must be jettisoned as the maximum airspeed the pilot can attain with the drag chute deployed is approximately 140 knots at 100 per cent RPM. The drag chute may be deployed or jettisoned if battery bus power is available.

Note

If the right-hand drag chute door alone opens in flight, the drag chute system will still function normally and no attempt to jettison the chute should be made. If the chute should inadvertently deploy at any speed, it will be automatically jettisoned without decelerating the aircraft, if the drag chute handle is in the full IN position.

Drag Chute Control Handle.

The drag chute control handle (figure 1—25) is a T-handle marked PULL TO DEPLOY: ROTATE 90°, PULL TO JETTISON. To deploy the drag chute, the handle is pulled aft approximately two inches, closing a microswitch which energizes the riser cable lock solenoid. The solenoid forces a plunger between the forward arms of a latch clamp locking the riser cable in the drag position. Movement of the plunger closes a second microswitch which energizes a solenoid in the left door spring capsule releasing the door latch and opening the door; the spring capsule on the right door is then free to force the right door to the open position. As the left door swings open, a cable attached to the door pulls the pilot chute rip pin, allowing the pilot chute and drag chute to be released in sequence. A button is provided in the T-handle which prevents the pilot from returning the handle to the full forward position, inadvertently jettisoning the chute when it is in the deployed position. To jettison the chute when it is in the deployed position, rotate the T-handle 90 degrees counterclockwise and pull aft a second time. This will electrically open the riser cable latch lock, releasing the riser cable and drag chute for jettisoning. The drag chute control circuit is powered by the emergency bus and will operate regardless of the position of the battery switch. After the drag chute is jettisoned, the button should be depressed and the handle returned to the full IN position to prevent unnecessary electrical drain on the battery.

Note

On aircraft not modified by {728} the button in the drag chute release handle is not provided.

CAUTION

To prevent inadvertent jettisoning of the drag chute, the control handle should not be rotated until jettisoning is required.

INSTRUMENTS — FLIGHT.

The turn needle in the turn and slip indicator, and the radio compass indicator are operated from the primary bus. The heading indicator and the attitude indicator are powered from the AC power circuit, and will operate from either the main or alternate inverters. The accelerometer, vertical velocity indicator, altimeter, airspeed indicator, the ball in the turn and slip indicator and the Mach indicator, installed in the instrument panel, do not require any electrical power. The static vent and the pitot pressure head are incorporated in a boom installed on the left wing tip. The pitot heater is described in Section IV.

ATTITUDE INDICATOR.

The attitude indicator, Type J-8 (3, figure 1–4) is a non-tumbling gyro and shows the attitude of the aircraft in relation to the earth's horizontal plane during any aircraft maneuver. It is an electrically driven instrument receiving power from the AC bus and will operate from either the main or alternate inverter. The portion of the sphere, which is visible to the pilot during level flight and in dives or climbs up to 27 degrees, is unmarked. Relative motion of the aircraft is indicated on the face of the instrument by movement of the horizontal bar with respect to the miniature aircraft in the center of the dial. Angular displacement of the horizontal bar, with respect to the miniature aircraft, indicates the degree of roll. The actual amount of roll is indicated by the position of the bank index relative to the 10-, 20-, 30-, 60- and 90-degree roll markings on the face of the instrument. When the aircraft exceeds 27 degrees of dive, the horizontal bar is held in its extreme (27 degree) position. At this point, the word DIVE, on the upper portion of the sphere, becomes visible. As the angle of dive increases graduations become visible on the sphere. These graduations are placed at the 70-, 75- and 80-degree intervals; the 85-degree dive indication is reached when the trim indicator coincides with the edge of the bull's eye. When the aircraft exceeds 27 degrees of climb, the horizontal bar is held in its extreme (27 degrees) downward position and any increase in climb is indicated on the sphere. The lower portion of the sphere is marked similarly to the upper with the word CLIMB substituted for DIVE. The horizontal bar may indicate a pitch and or a bank error in excess of 5 degrees after a loop or during a turn. The attitude indicator will immediately begin to correct these errors once true gravitational forces are sensed. This characteristic error is commonly called "sluggishness" or "lag" by pilots. In successive loops, the above described error may become increasingly greater and may cause the horizon bar to reach the limit of its movement. This is normal in successive loops and is not indicative of a defective instrument. The attitude indicator may be caged manually by means of a gyro centering device operated by pulling the cage knob. To cage the gyro, the PULL TO CAGE knob is drawn smoothly away from the face of the instrument. A momentary stop will be felt when the bank caging mechanism is engaged; as the cage knob is pulled further out, the pitch caging mechanism is engaged. As soon as the caging knob reaches the limit of its travel, it should be released quickly.

WARNING

A slight amount of pitch error in the indication of the 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 aircraft is flying straight and level. This error will be most noticeable at the time the aircraft 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.

CAUTION

A violent or hard pull on the caging knob when caging the attitude indicator may damage the instrument. Remember that the indicator cages to the attitude of the aircraft and not to the true vertical. Therefore, the instrument should never be caged to correct inflight errors, unless the aircraft is in straight and level flight by visual reference to a true horizon.

- A slight reduction in AC or DC power, or failure of certain electrical or mechanical components in the system, could cause loss of roll indication, and not cause the OFF flag to appear even though the instrument is not functioning properly. Therefore, periodically in flight, the attitude indications should be checked against other flight instruments, such as the standby magnetic compass, or the turn and slip, and vertical velocity indicators.

ALTIMETER.

The altimeter (34, figure 1–4) utilizes static pressure and provides the pilot with a constant indication of barometric altitude. Three pointers on the face of the instrument move over a scale graduated in 20 foot increments, with a major division every hundred feet from zero to one thousand feet. The larger pointer indicates hundreds of feet and makes one revolution for each 1,000 feet of altitude. An additional small pointer indicating tens of thousands of feet is painted on a black disc with an extension line terminating in a triangular section. This 10,000 foot pointer serves a second function as a warning indicator passing through a striped section which appears on the instrument face at altitudes below 16,000 feet. This type of altimeter offers improved readability and gives visual warning if an altitude of less than 16,000 feet is entered. The pointers and barometric scale can be set manually by turning the knob in the lower left corner of the instrument. To determine altimeter error, the pilot sets base altimeter setting on the barometric scale, then notes indicated altitude, which should be compatible with known field elevation.

CAUTION

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

ENGINE OVERHEAT and FIRE WARNING SYSTEM

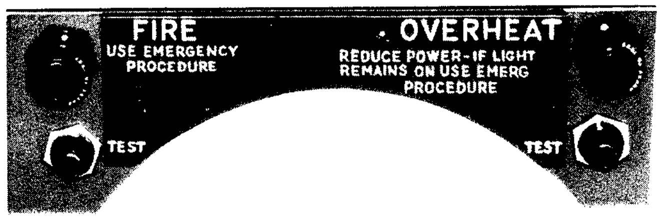


Figure 1–26

EMERGENCY EQUIPMENT.

ENGINE OVERHEAT WARNING SYSTEM.

The engine overheat warning system consisting of eight thermal switches located in the aft fuselage, and an amber press-to-test type warning light (figure 1–26) placarded OVERHEAT and powered by the primary bus. A button-type switch is installed adjacent to the warning light and is placarded PRESS TO TEST. Illumination of the light when the button is depressed indicates the circuits are complete and the system is functioning properly.

ENGINE FIRE WARNING SYSTEM.

An engine fire warning system, consisting of thermal switches in the forward section of the fuselage, and a red press-to-test type warning light (figure 1–25) placarded FIRE, and is powered by the primary bus. A button-type switch is installed adjacent to the warning light and is placarded PRESS TO TEST. Illumination of the light when the button is depressed, indicates the circuits are complete and the system is functioning properly.

ESCAPE SYSTEM.

The escape system extends the maximum and minimum airspeed and altitudes at which escape may be successfully accomplished, and requires a minimum of effort on the part of the pilot. In all cases of emergency exit in flight, it is recommended that escape be accomplished by means of the escape system. The system consists primarily of the jettisonable canopy, the ejection seat with automatic personal disconnects, automatic opening safety belt, seat man separator, and the automatic parachute with zero delay lanyard. These components of 'the escape system are explained in subsequent paragraphs under appropriate headings.

ESCAPE SYSTEM FUNCTIONS.

Canopy Jettison.

Raising both hand grips on the seat causes the following events to occur:

- Both armrests are moved to the ejection position.
- The shoulder harness control handle is moved to the locked position.
- The ejection trigger is moved to the cocked position, arming the seat for ejection.
- The canopy locks are opened, the canopy is opened and separated from the aircraft.

Seat Ejection.

Squeezing the ejection trigger(s) causes the following events to occur:

- The seat is ejected.
- All aircraft to seat connections are disconnected.
- The automatic safety belt initiator (incorporating a one second time delay) is actuated.
- After the seat has left the cockpit, the safety belt initiator fires, opening the safety belt and actuating the seat man separator.
- The open safety belt releases the shoulder harness straps but retains the parachute lanyard, and zero delay lanyard.
- The seat man separator strap assembly is drawn taut displacing the survival kit and pilot from the seat.
- As the pilot is separated from the seat, the parachute timer lanyard, still connected to the left half of the safety belt, will actuate the parachute timer. If above 14,000 feet, the timer will delay parachute deployment until 14,000 feet is reached, if below 14,000 feet the timer will deploy the parachute in one second. If, however, the zero delay lanyard is connected to the rip cord grip, it will override the timer and deploy the parachute immediately.

Note

Aircraft with {801} not complied with, the seat-man separator is not installed.

CANOPY.

The jettisonable canopy is a one piece bubble type enclosure made up of two layers of plexiglas with an air space between them. Heated and dried air, controlled by a manually operated switch, is circulated between the two layers to de-fog the canopy. The canopy is hinged to the fuselage in such a way that when it is manually opened, it moves up and aft in an arc so as to clear the pilot at all times. It is counter-balanced so that opening and closing is accomplished with a minimum effort. In the closed position, the canopy is sealed to the cockpit structure, for pressurization by rubber tubes that are automatically inflated by air pressure from the engine compressor. A warning light indicates when the canopy is not fully closed and locked. Controls for locking or unlocking the canopy from inside or outside are provided. The jettison control is designed to assure jettisoning the canopy before ejecting the seat.

Canopy Control Lever.

The canopy control lever (figure 1–27) operates a manual lock and has only two positions: OPEN and CLOSE. The CLOSE position mechanically locks the canopy to the fuselage structure. The OPEN position releases the canopy locks and the canopy can then be lifted to the fully open position. When fully open, a lock automatically engages on the side hinges to hold the canopy. These locks are released by pulling the canopy control lever aft, past the OPEN position. The operation assures that the canopy control lever will be in the OPEN position before closing the canopy. When the canopy is jettisoned, the mechanical locks between the canopy and fuselage are automatically opened.

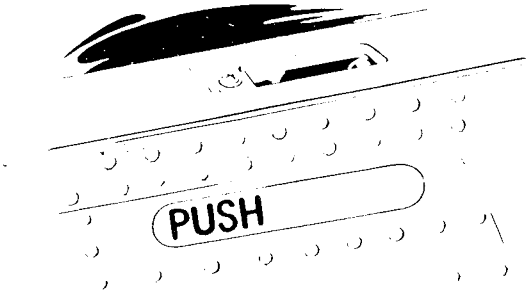
External Canopy Control.

The external canopy control (figure 1–27) on the left side of the fuselage below the canopy, is provided to lock or unlock the canopy from outside the cockpit. The control is a flush yellow lever with the forward end marked PUSH. When the forward end is pushed in, the aft end rotates out far enough to grasp with the hand. Moving the lever full forward unlocks the canopy and permits the canopy to be lifted open. To engage the lock, the lever is returned to the flush position. When the canopy is locked from the inside, the lever returns to the flush position.

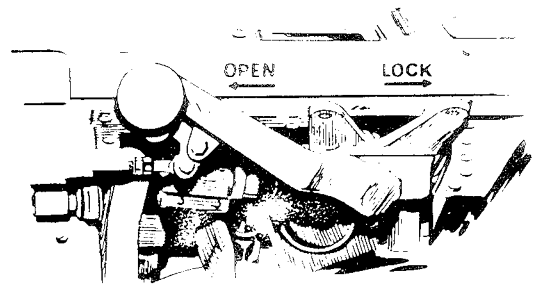
CANOPY CONTROLS

LOCK-UNLOCK CANOPY

EXTERIOR

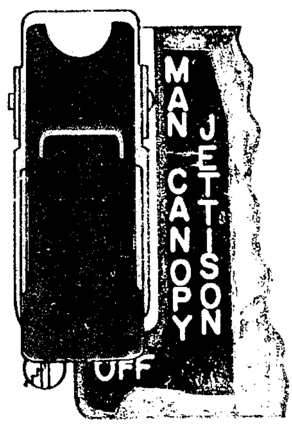


INTERIOR



JETTISON CANOPY WITHOUT ARMING EJECTION SEAT

The Manual Canopy Jettison Switch is located on the left console.

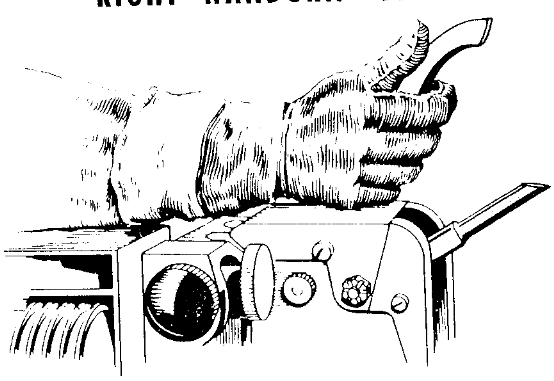


Aircraft with 811 not complied with, do not incorporate the Manual Canopy Jettison Switch.

JETTISON CANOPY AND ARM EJECTION SEAT

5 thru 25 Aircraft

RIGHT HANDGRIP ONLY



30 and LATER Aircraft

BOTH HANDGRIPS

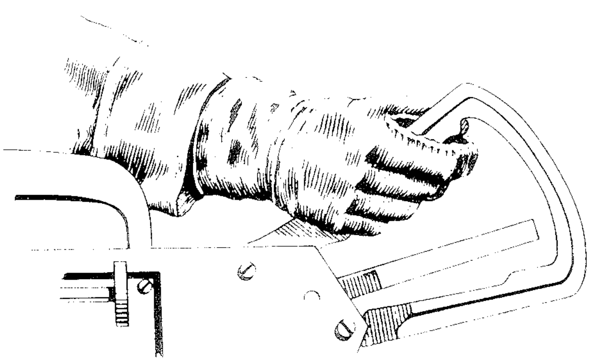


Figure 1-27

Manual Canopy Jettison Switch. **MODIFIED**

The manual canopy jettison switch (figure 1–27) placarded MAN CANOPY JETTISON is a two position toggle switch with positions ON and OFF and is powered by the primary bus if energized, or the standby canopy jettison battery. The switch is guarded in the OFF position, and the guard safetied with light breakaway safety wire. The switch arms the canopy jettison circuit without arming the ejection seat. Positioning the switch to ON arms the canopy jettison circuit. The canopy is manually unlocked and pushed up. The airstream will force it to the full open position. In the full open position the right canopy mounting arm actuates a microswitch which completes a circuit to the canopy jettison explosive squibs at the four hinge fittings. The explosive squibs are normally powered by the battery bus when the battery switch is in the ON position. In the event of battery failure, or when the battery switch is in the OFF position, the explosive squibs are powered by the canopy jettison battery, which is independent of the aircraft electrical system. The exploding squibs separate the canopy from the aircraft. In the event the squibs fail to fire, airstream pressure on the canopy will cause the canopy arms to break when they hit the up-stop. Holes in the canopy arms establish the breaking point.

WARNING

Aircraft with (763) not complied with do not have holes in the canopy arms. On these aircraft the canopy should be jettisoned at approximately 300 KIAS to assure canopy separation in the event squibs fail to fire.

- The present inspection requirement applicable to the 4.5 volt canopy jettison battery does not determine the ability of the battery to detonate the canopy squibs, therefore the aircraft battery switch should be left in the ON position until the canopy has been jettisoned.

Note

Aircraft with {811} not complied with do not have the manual canopy jettison switch installed.

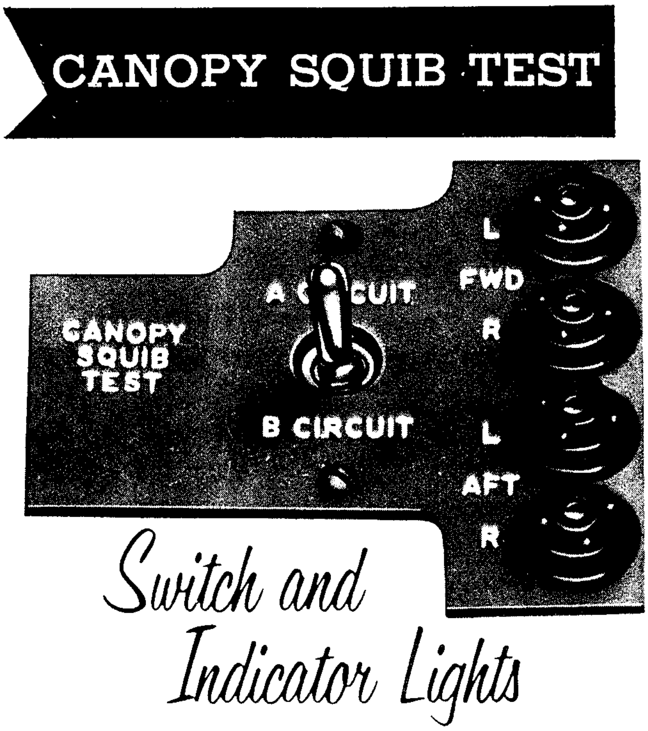


Figure 1—28.

Canopy Squib Test and Indicator Lights.

The canopy squib test switch (figure 1—28) is a three position spring-loaded toggle switch placarded CANOPY SQUIB TEST with momentary positions A CIRCUIT, B CIRCUIT and an unmarked neutral (OFF). Four adjacent lights are placarded FWD L and R, AFT L and R. Illumination of all the lights when the switch is held to A CIRCUIT indicates that the A circuit to all squibs is complete. Illumination of all the lights when the switch is held to B CIRCUIT indicates that the B circuit to all squibs is complete. Either the A or B circuit can detonate the squibs as they are dual circuits completely independent of each other. If any light does not illuminate during the test, the respective circuit, A or B, to that particular squib is open. The squib test must be made with the battery switch ON and the canopy fully open. In the full open position the right canopy mounting arm actuates a microswitch which completes a circuit to the canopy jettison explosive squibs at the four hinge fittings. Energizing the primary bus with the generator or external power has no effect on the squib test.

Canopy Open Warning Light.

The canopy open warning light (1, figure 1—4) is a red press-to-test type light placarded CANOPY OPEN and powered by the primary bus. Illumination of the light indicates the canopy is not fully closed and locked.

WINDSHIELD.

The windshield consists of three transparent panels, set in rubber and mounted in an aluminum frame. The center panel is laminated bullet-resistant glass with an electrical element incorporated between the laminations for de-icing and de-fogging. Each side panel is made up of two plexiglas panels with an air space between them for de-fogging. Heated and dried air is circulated between the two layers and is controlled by a manually operated switch. A defroster system, with a manual shut-off, directs hot air from the engine compressor to a perforated tube that directs the air over the inside of each side panel. For complete describtion of the defroster system, see Section IV.

EJECTION SEAT.

The ejection seat (figure 1—29) is an upward ejection type designed to eject the pilot clear of the aircraft in an emergency. A rocket catapult attached to the rear of the seat provides the propelling force to eject the seat. The seat incorporates the following; canopy jettison and seat ejection controls, shoulder harness and control, automatic opening safety belt with provisions for automatic parachute and zero delay lanyard, seat-man separator, pilot's personal lead quick disconnect, adjustable height and a bucket seat to accommodate an MD-1 survival kit. Aircraft with [801] not complied with, have a cartridge type catapult installed instead of a rocket catapult and the seat-man separator is not installed.

WARNING

An excessively compressible seat cushion increases chances of vertebra injury as the seat will gain considerable momentum before exerting impact on the pilot. On aircraft modified by [801] the MD-1 survival kit with its integral cushion will properly position the CG of the pilot in relation to the seat thrust line for ejection *only if* the packed height is 7.5 to 8.5 inches. Seat fillers consisting of large amounts of compressible mass which compresses to less than 5 inches on ejection will permit the CG to shift excessively during ejection.

EJECTION SEAT

5 thru 25 Aircraft

NOTE

Canopy Jettison Control and Seat Ejection Control is provided on right side of seat only.

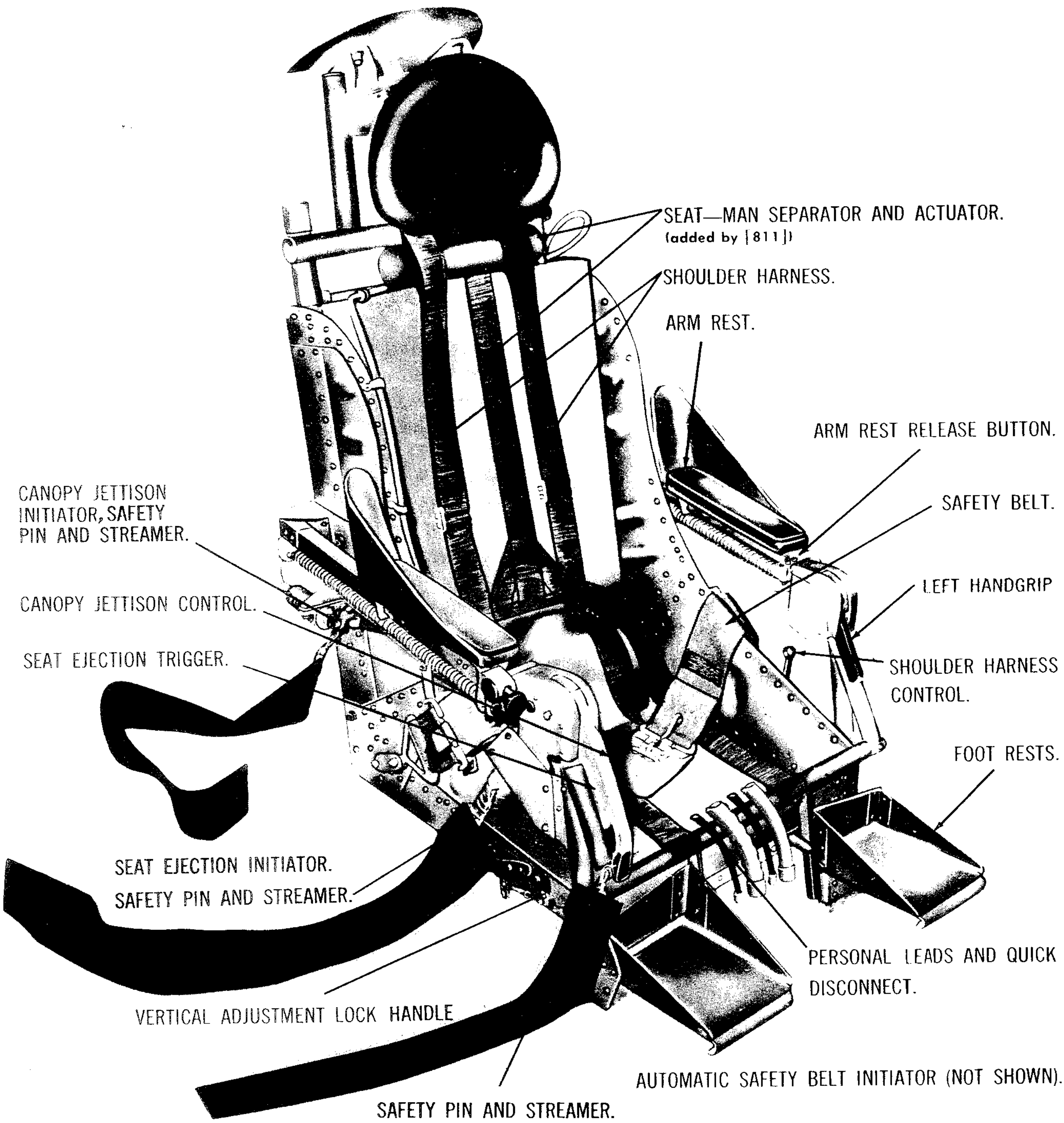
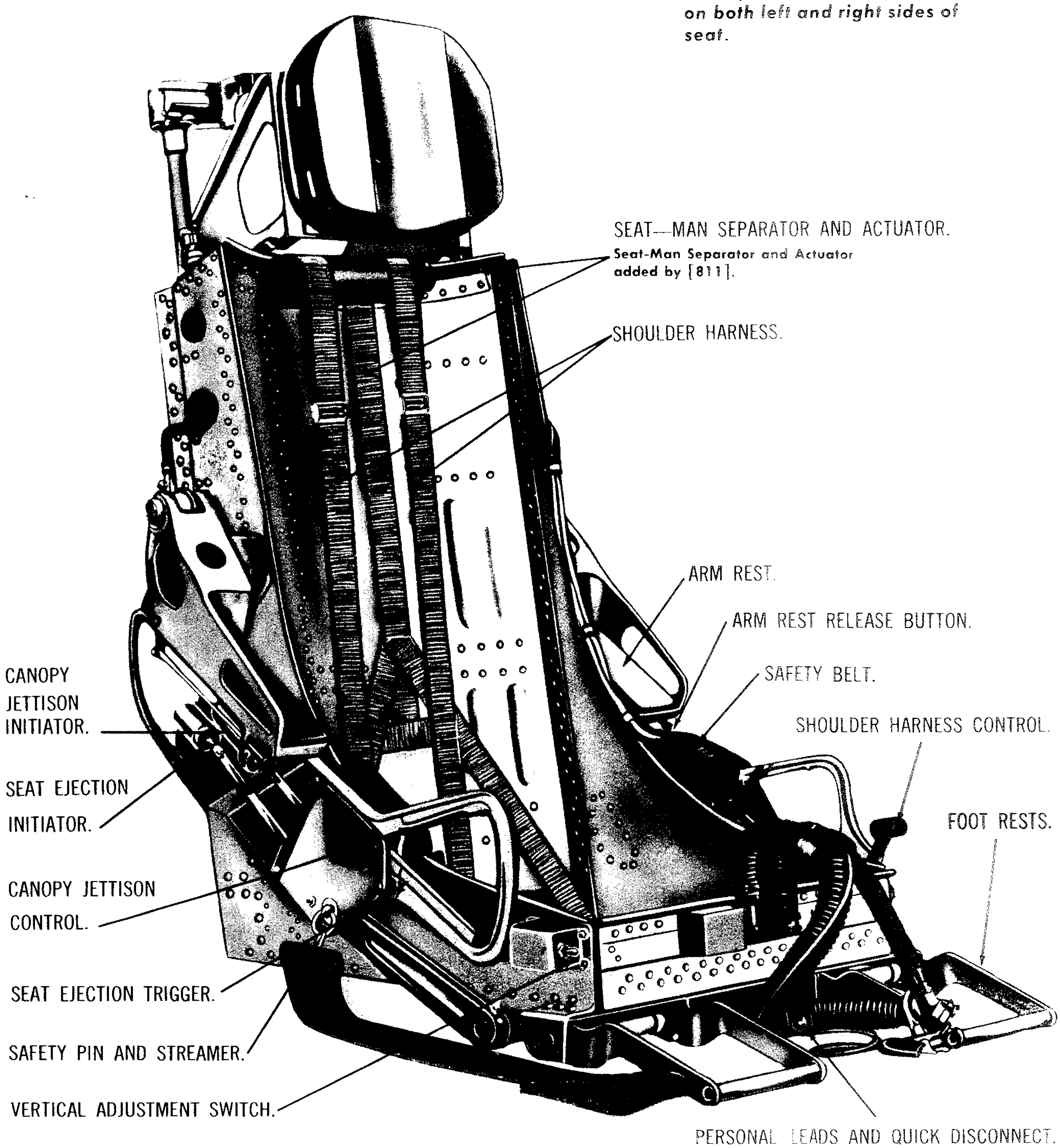


Figure 1-29

30 and LATER Aircraft

NOTE

Canopy jettison control and seat ejection control provided on both left and right sides of seat.



AUTOMATIC SAFETY BELT INITIATOR (NOT SHOWN).

Canopy Jettison Control.

On 5 thru 25 aircraft, the right handgrip (figure 1–27) marked CANOPY JETTISON, and guarded by a spring-loaded clip, is raised to jettison the canopy and arm the seat for ejection. On 30 and later aircraft, the left and right handgrips (figure 1–27) are interconnected and both are raised to jettison the canopy, arm the seat and lock the shoulder harness. Raising the canopy jettison control (handgrips), arms the squib circuit and fires the jettison initiator that develops a high pneumatic pressure which is carried through a flex hose to the canopy downlock release cylinder to unlock the canopy and actuates a second initiator. The pneumatic pressure from the second initiator is carried through a flex line to a piston type thruster which raises the canopy approximately four inches up into the airstream which will force it to the fully open position. In the full open position the right canopy mounting arm actuates a microswitch which completes a circuit to the canopy jettison explosive squibs at the four hinge fittings. The explosive squibs are normally powered by the battery bus when the battery switch is in the ON position. In the event of battery failure, or when the battery switch is in the OFF position, the explosive squibs are powered by the canopy jettison battery, which is independent of the aircraft electrical system. The exploding squibs separate the canopy from the aircraft. In the event the squibs fail to fire, airstream pressure on the canopy will cause the canopy arms to break when they hit the up-stop. Holes in the canopy arms establish the breaking point.

WARNING

- Aircraft with [763] not complied with do not have holes in the canopy arms. On these aircraft the canopy should be jettisoned at approximately 300 KIAS to assure canopy separation in the event squibs fail to fire.
- The present inspection requirement applicable to the 4.5 volt canopy jettison battery does not determine the ability of the battery to detonate the canopy squibs, therefore the aircraft battery switch should be left in the ON position until the canopy has been jettisoned.

The canopy jettison control is safetied by installing a pin with a red streamer attached. On 5 thru 25 the pin is installed through the control handle guard clip over the lower end of the control. On 30 and later aircraft the pin is installed through the control arm.

Left Handgrip. 5 thru 25

The left handgrip (figure 1–29) locks the shoulder harness and provides a handgrip to enable the pilot to keep his left arm from flailing during ejection. The handgrip locks in the raised position.

Shoulder Harness Control Handle.

A two position LOCKED-UNLOCKED shoulder harness control handle (figure 1–29) is located on the left side of the ejection seat. On 5 thru 25 the handle locks and is released for movement by pressing down on the top of the control handle. When the control handle is in the UNLOCKED position (full aft), the shoulder harness reel cable will extend to allow the pilot to lean forward in the cockpit and automatically retract as tension is released when the pilot moves back. However, the inertia reel will automatically lock, preventing further extension of the cable, when an impact force of 2 to 3 G's is encountered. When the reel is locked in this manner, it will remain locked until the control handle is moved to the locked position and then returned to the unlocked position. If the control is in the locked position (full forward) while the pilot is leaning forward, as he straightens up, the harness will retract with him, moving into successive locked positions as he moves back against the seat. The locked position is used when a crash landing or ditching is anticipated. This control position provides an added safety precaution over and above that of the automatic safety lock.

Note

- On 5 thru 25 aircraft, during the ejection sequence the left handgrip must be raised to lock the shoulder harness.
- On 30 and later aircraft, during ejection sequence, the shoulder harness locks automatically when the canopy jettison control is raised.

Vertical Adjustment Lock Handle. 5 thru 25

The vertical adjustment lock (figure 1–29) handle and indicator is provided for adjustment of seat height. Raising the lever disengages four locking pins and permits the pilot to raise or lower the seat to any of six positions. A spring assists the pilot when adjusting the seat upward. During the adjustment the footrests remain in against the floor. With the vertical adjustment lock handle in this position there is no "finger" space between the adjustment lock handle and the locking pin indicator. After the seat is at the desired height, the vertical adjustment lock handle is released and the seat and footrest locking pins automatically engage. If the locking pins are fully engaged there will be "finger" space (approximately ⅝ inch) between the vertical adjustment lock handle and the locking pin indicator. In the event that the locking pins are not fully engaged (indicated by no "finger" space), jiggle the seat to permit the pins to fully lock seat.

WARNING

After each vertical (height) adjustment of the seat insure that the lock handle and locking pin indicator are separated by approximately ⅝ inch (½ to ¾ inch). Failure of the handle and the locking pin indicator to separate as specified indicates one or more of the four seat locking pins has not locked. Failure of the top pins to lock in place can prevent successful seat ejection. Do not attempt to adjust vertical height of the manually operated seat while in flight.

Vertical Adjustment Switch. 30 and Later

The vertical adjustment switch (figure 1–29) is a three position type switch spring-loaded to the OFF position. When the switch is moved to the UP position, primary bus power is supplied to actuate an electric motor and screw jack mechanism to raise the seat. The seat-locking drive mechanism will remain in any selected position when the switch is released. Placing the switch in the DOWN position lowers the seat. Over-all travel of the seat is 3.75 inches.

Armrests.

The armrests (figure 1–29) are provided primarily to hold the pilot's arms from flailing during ejection. Both armrests move to the ejection position when both handgrips are raised. To provide better access to the left and right console the armrests may be moved from the normal (ejection) position.

On 5 thru 25 aircraft, the armrest is pulled back and engaged by a latch. To return it, an armrest release button (figure 1–29) is depressed, and a spring moves it forward.

On 30 and later aircraft, the armrest release button (figure 1–29) is depressed and the armrest pushed down. To return it, the armrest is pulled up.

Ejection Trigger.

When the seat handgrips (canopy jettison control) are raised and locked the seat ejection trigger is exposed and armed. Squeezing the ejection trigger to the handgrip mechanically fires an initiator that develops a high pneumatic pressure which is carried through a flex hose to fire the ejection catapult.

On 5 thru 25 aircraft, the ejection trigger is provided with the right handgrip only.

On 30 and later aircraft, ejection triggers are provided in both handgrips. Each trigger is connected to a separate initiator, either or both of which will fire the ejection catapult.

AUTOMATIC SAFETY BELT.

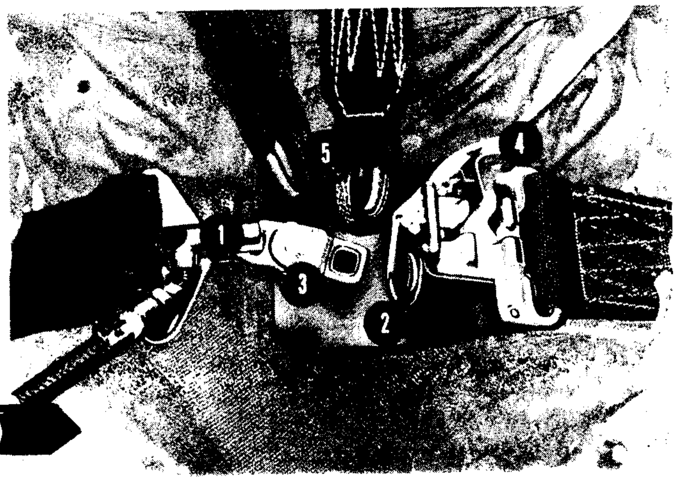
The type MA-5 or 6 automatic opening safety belt, an integral part of the escape system extends the altitudes at which escape may be successfully accomplished. In a low altitude ejection, use of the automatic system greatly reduces the time required for separation from the seat and deployment of the parachute, and consequently reduces the altitude required for safe ejection. The automatic belt has been thoroughly tested and is completely reliable. UNDER NO CIRCUMSTANCES SHOULD THE BELT BE MANUALLY OPENED BEFORE EJECTION REGARDLESS OF ALTITUDE. No matter how fast a pilot's reactions are, he cannot beat the automatic operation; besides, he may not remain conscious during an actual ejection. The belt is cartridge operated for automatic opening during seat ejection, but manually opened for normal operation. Automatic operation is accomplished during seat ejection by gas pressure from a separate automatically controlled initiator which supplies pressure through a length of high pressure hose that actuates a piston inside the belt, retracting the latch tongue, and releasing the belt swivel link. The link accommodates an anchor on a lanyard leading to the parachute automatic timer (figure 1–30). When the belt is manually opened the anchor is released automatically so that inadvertent actuation of the automatic parachute will not occur.

Automatic Operation.

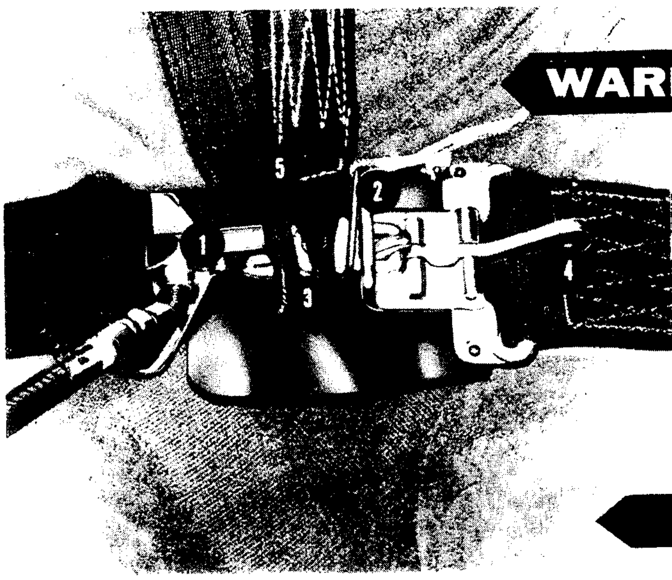
Automatic belt opening is accomplished as part of the seat ejection sequence and requires no additional effort on the part of the pilot. When the seat is ejected, a firing pin in the initiator is pulled. One second after the pin is pulled the initiator is fired. The time delay feature insures that the belt will not release until the seat and pilot are entirely clear of the aircraft. When an automatic parachute is used, an anchor attached to the parachute lanyard is installed over the lap belt swivel link. (See figure 1–30 for correct installation.) During automatic operation, the anchor remains fixed on the left half of the open safety belt. Thus the automatic rip cord release is actuated as the pilot separates from his seat.

AUTOMATIC SAFETY BELT TYPE MA-5 & 6

1. *Automatic Release*
2. *Parachute Lanyard Anchor*
3. *Swivel Link*
4. *Manual Release*
5. *Shoulder Harness Loops*



OPEN (MANUALLY)



WARNING

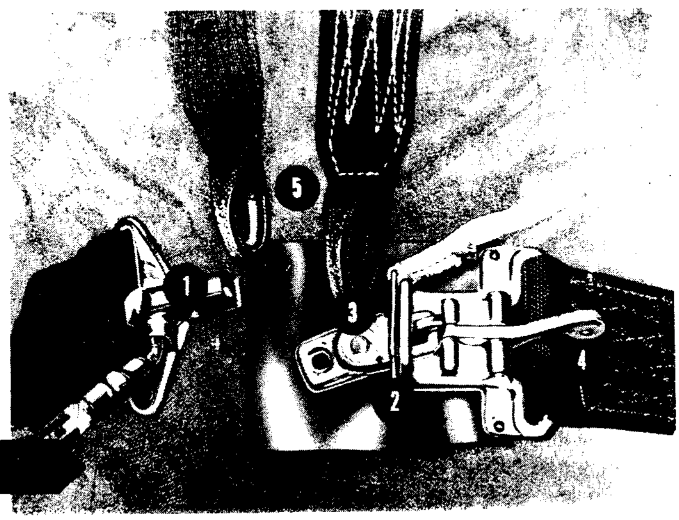
WARNING

Failure to install the shoulder harness loops and parachute lanyard anchor in correct sequence will prevent separation from the seat after ejection.

a. Place left and right shoulder harness loop on safety belt swivel link.
b. Place parachute lanyard anchor on safety belt swivel link and fasten safety belt.

LOCKED CONDITION

Swivel link (3) released from right side of belt.
Shoulder harness loops (5) released from swivel link (3).
Parachute lanyard anchor (2) retained by shoulder on swivel link (3).
Manual release lever (4) locked, holding swivel link to left side of belt.



AUTOMATICALLY OPENED

Figure 1-30

The automatic opening safety belt should never be opened manually before ejection for the following reasons:

- If the belt is opened manually the escape operation is prolonged.
- Manual opening of the automatic safety belt creates a hazard to survival during uncontrollable flight. The opening of the safety belt during uncontrollable flight will mean that the pilot cannot stay in his seat before ejection if negative G is incurred.
- Manual opening of the automatic safety belt creates a hazard to survival if the pilot decides to crash land the aircraft. The pilot will probably not be able to fasten the safety belt and shoulder harness as he will be using both hands to control and crash land the aircraft.
- The manual opening of the automatic safety belt will eliminate the automatic feature of the automatic opening parachute (if worn), unless the pilot manually arms the parachute by pulling the arming ball.
- Tail clearance is reduced by immediate separation of the pilot from the seat, which is likely to occur when the belt is opened before ejection.
- At high speeds the peak deceleration, due to air loads on the seat and pilot together, approaches the limits of human endurance. Since the deceleration of a pilot alone is greater than that of the pilot and seat together, immediate separation at extremely high speeds could result in severe injury to the pilot.
- Immediate separation of the pilot and seat at extremely high speeds could result in advertent opening of the parachute due to the pack being blown open. In this event, fatal injuries will probably be incurred, because of the extremely high opening shock of parachute at this speed, or because of serious damage to the parachute itself upon opening at high speeds.

Manual Operation.

Refer to figure 1—30 for proper sequence of opening and closing the safety belt, shoulder harness, and the automatic parachute lanyard.

SEAT-MAN SEPARATOR. MODIFIED

The seat-man separator (figure 1—29) operates automatically as a part of the ejection sequence and requires no additional effort on the part of the pilot. The system consists of a web-strap assembly shaped like an inverted Y, and a cartridge operated actuator. Two straps attached to the forward edge of the bucket seat are **routed under the survival kit** to the yoke from which a single strap is routed up the face of the seat to the actuator near the headrest. When the seat is ejected, a firing pin in the automatic safety belt initiator is pulled. One second after the pin is pulled the initiator is fired developing a high pneumatic pressure. The high pneumatic pressure is carried through a flex hose to open the automatic safety belt and fire the cartridge in the seat-man separator actuator. Rotary action of the actuator draws the web assembly taut, effectively displacing the survival kit and separating the pilot from the seat.

Note

Aircraft with [801] not complied with, do not have the seat-man separator installed.

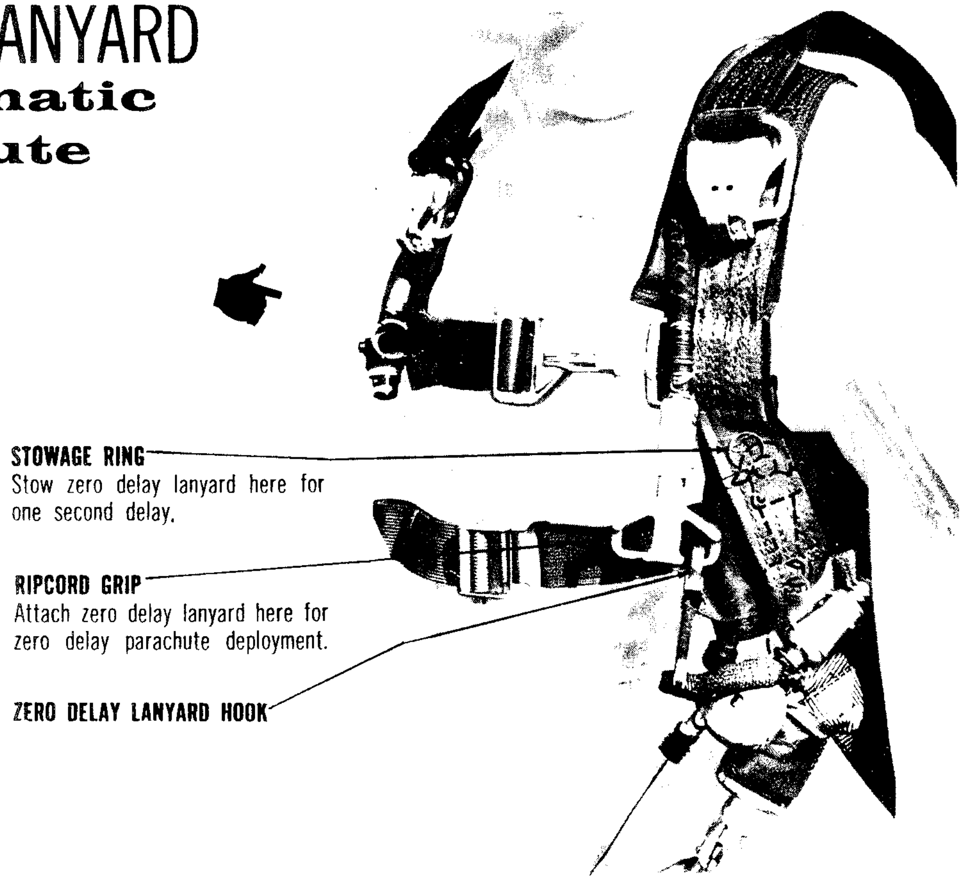
ZERO DELAY LANYARD.

In order to provide a low-altitude escape capability, a system incorporating a one second delay initiator for lap belt opening, and a zero parachute delay (one and zero system) is provided for ejection seat escape systems. This system makes use of a lanyard, installed on the parachute harness (figure 1—31), that connects the parachute arming lanyard to the parachute rip cord grip. At very low altitudes and airspeeds, this zero delay lanyard must be connected, thus providing parachute actuation immediately after separation from the ejection seat. At high altitudes and airspeeds, the lanyard must be disconnected from the rip cord grip, thus allowing automatic parachute operation. A ring, attached to the parachute harness, is provided for stowage of the lanyard hook when it is not connected to the parachute rip cord grip. *This hooking and unhooking of the lanyard must be done manually by the pilot.*

WARNING

- If the zero delay lanyard is left connected while flying at high altitude, and ejection becomes necessary, the parachute will be deployed at an altitude where sufficient oxygen is not available to permit safe parachute descent.
- If the zero delay lanyard is left connected while flying at high speed and ejection becomes necessary, the time delay feature of the parachute will be overriden and the parachute will be deployed at high speed, result-

ZERO DELAY LANYARD
for automatic parachute



CAUTION

The emergency minimum ejection altitudes specified for one-second safety belt and zero second parachute setting apply when the lanyard is attached to the ripcord grip and safety belt.

WARNING

The lanyard must be disconnected whenever operating at high altitude or airspeeds in order that the safety-delay provided by the parachute timer-aneriod will not be overridden.

Figure 1—31.

ing in an extremely high opening shock, in which case injuries will probably be incurred and the parachute seriously damaged.

- If the automatic safety belt is opened manually, the parachute rip cord must be pulled manually.

Note

See figure 3—3 for zero delay lanyard engagement requirements which show the low altitude capabilities and speed limitations of zero delay lanyard-parachute combinations.

AUTOMATIC PARACHUTE.

The automatic parachute extends the altitudes at which escape may be successfully accomplished. In a high altitude ejection (above 14,000 feet) use of the automatic parachute avoids deployment at an altitude where sufficient oxygen would not be available to permit safe parachute descent. The aneroid action of the automatic parachute release will delay the opening of the parachute until the pilot falls free to a safe altitude. For ejection below 14,000 feet, one second after separation from the ejection seat parachute deployment will commence. For low altitude ejection use of the zero delay lanyard will open the parachute immediately after separation from the seat.

WARNING

If the automatic safety belt is opened manually, the parachute rip cord must be pulled manually.

AUXILIARY EQUIPMENT.

Section IV of this manual contains information on the following auxiliary equipment: heating, pressurization, ventilation systems; anti-icing and deicing systems; communications and associated electronic equipment; lighting equipment; oxygen system; air refueling system; automatic pilot; navigation equipment, armament equipment, photographic and miscellaneous equipment.

SERVICING *Diagram*

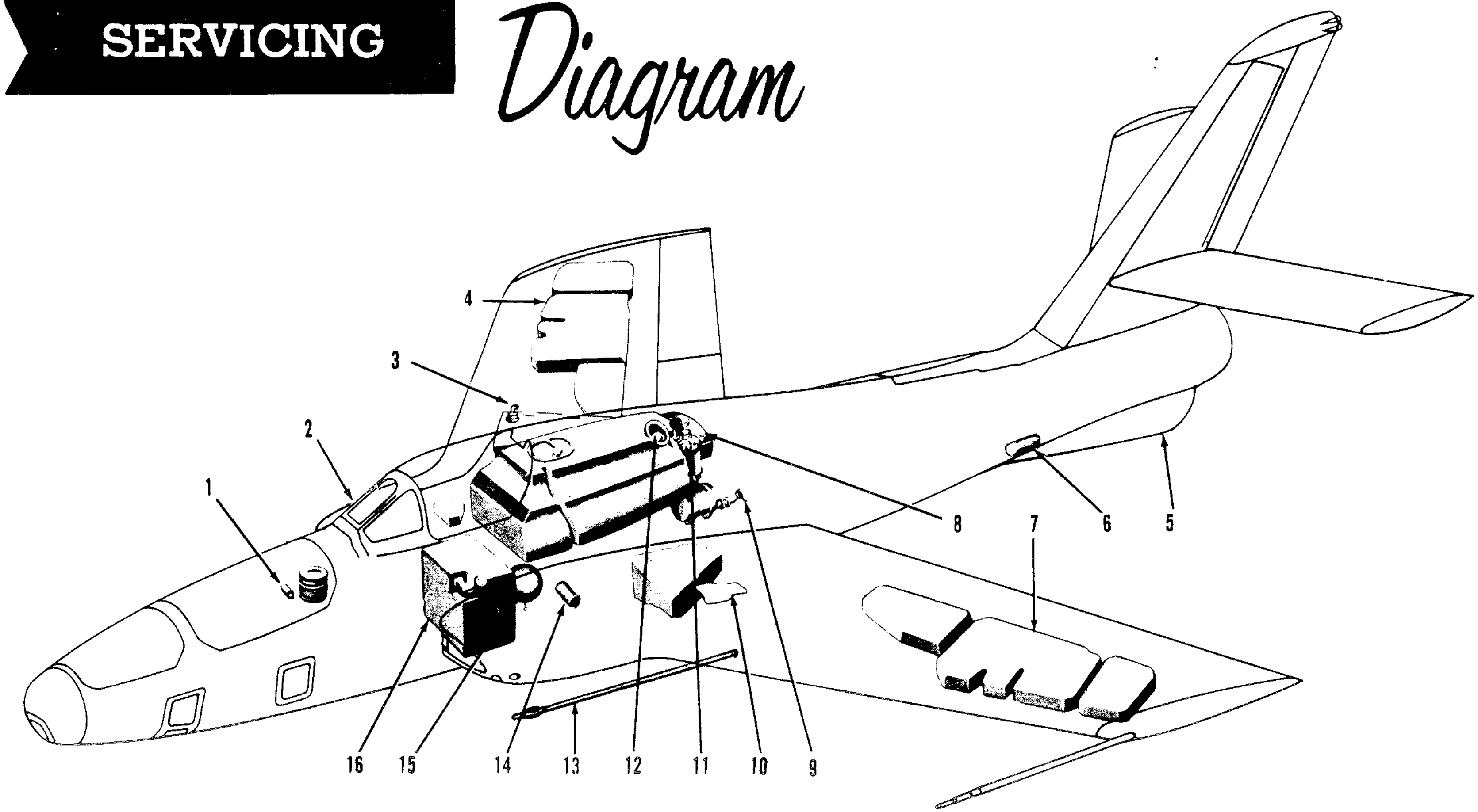
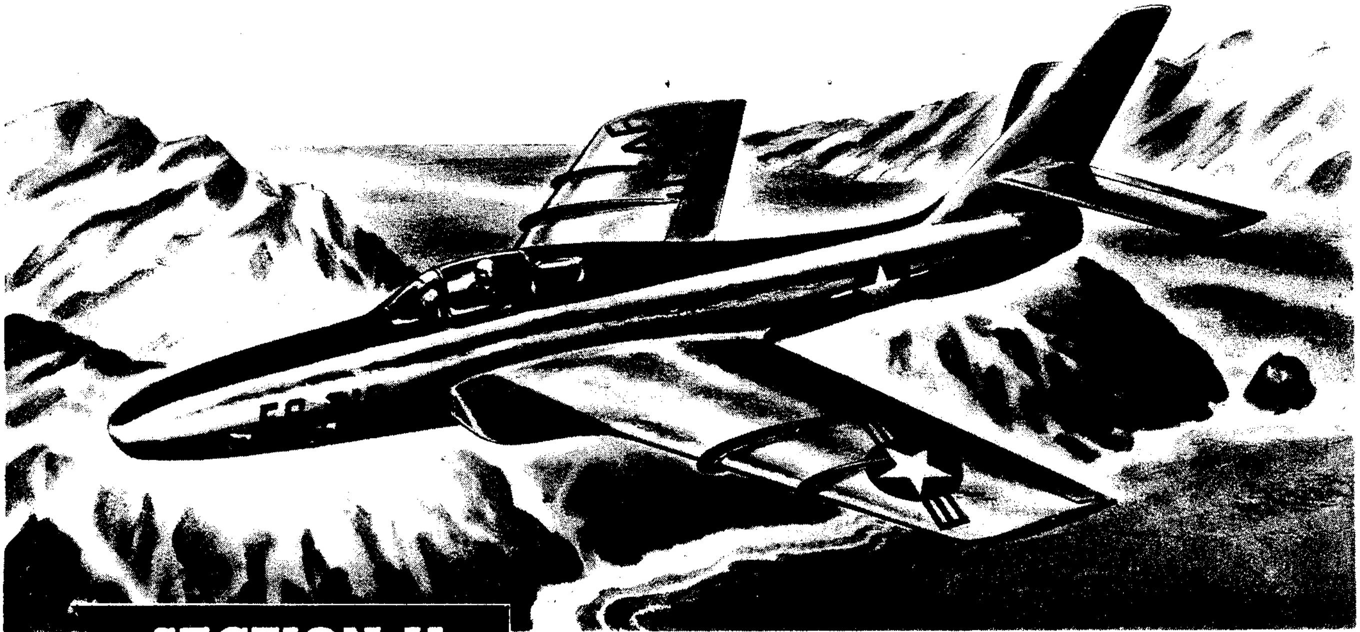


Figure 1—32

1. *Oxygen Filler Valve*
2. *External Power Supply*
3. *Single-Point Refueling*
4. *Wing Tank*
5. *Drag Chute Compartment*
6. *Emergency Hydraulic Reservoir*
7. *Wing Tank*
8. *Alcohol Tank*
9. *Oil Tank*
10. *Air Refueling Receiver (UNMODIFIED)*
11. *Utility Hydraulic Reservoir*
12. *Main Tank*
13. *Air Refueling Boom (MODIFIED)*
14. *Power Hydraulic Reservoir*
15. *Starter Air Bottle*
16. *Forward Tank*

SPECIFICATIONS

Fuel	MIL-J-5624 (NATO SYMBOL F-40)
Alternate Fuel (JP-5)	MIL-J-5624 (NATO SYMBOL F-44)
Emergency Fuel	MIL-G-5572(AVGAS, Lowest Octane Available, No Oil Mix Required)
Emergency Fuel	Unleaded AVGAS (NATO SYMBOL F-14)
Oil (Turbine)	MIL-L-7808 (NATO SYMBOL D-148)
Hydraulic Fluid	MIL-H-5606 (NATO SYMBOL H-515)
Oxygen	MIL-O-27210 (Replaces BB-0-925)



SECTION II
Normal Procedures

TABLE OF CONTENTS

PREPARATION FOR FLIGHT.

FLIGHT RESTRICTIONS.

Refer to Section V of this manual to determine the flight restrictions imposed on the aircraft.

FLIGHT PLANNING.

Determine cruise control data such as fuel required, airspeeds, power settings, etc., as necessary to complete the proposed mission from data contained in the Appendix of the manual.

TAKEOFF AND LANDING DATA CARD.

Complete information required on the Takeoff Data Card and the Landing Data Card. Illustration of these cards will be found in the Appendix of the manual.

WEIGHT AND BALANCE.

Check takeoff and anticipated landing gross weight and balance. Consult Manual of Weight and Balance Data, T.O. 1-1B-40 for loading procedure. Make sure weight and balance clearance (Form 365F) is satisfactory. Check to see that total weight of fuel, oil, armament, oxygen, and special equipment carried is suitable to the mission to be performed. Refer to Section V for weight limitations for various aircraft configurations.

ENTRANCE TO AIRCRAFT.

Place a ladder against the left side of the aircraft at the cockpit. No external grips or steps are provided. Open the canopy, using the external canopy control to unlock the canopy, then raise it manually using handgrip on canopy skirt.

PREFLIGHT CHECK.

BEFORE EXTERIOR INSPECTION.

1. Form 781 — Check.

   Check for release of aircraft, status and engine preoil requirements.

   Note

   The engine will be preoiled through the micro pump fittings whenever the oil pump or pressure lines have been disconnected, the oil pump or tank drained, prior to initial run of a new or overhauled engine, or if the engine has been idle for seven days.

   - Engines that do not incorporate overhaul change NO. 144 (center main bearing cooling tube external and internal silver plated washers) will be preoiled if not operated more than 30 minutes in the past 24 hours.

2. Applicable publications — Check aboard and current.

3. Canopy jettison control safety pin — Check installed.

4. Canopy jettison, seat ejection and automatic lap belt initiator safety pins — Check removed.

   Note

   If initiator pins are installed, consult maintenance personnel regarding status of ejection system before occupying ejection seat.

5. Rocket Seat Quick Disconnect — CHECKED.
6. Armament controls — Check safe.

   a. Gun arming switch — OFF.

   b. External stores arming switch — OFF.

   c. Pylon or tank jettison switch — OFF (guard down).

   d. Ejection seat cannon plug — Secure 5 thru 25.

EXTERIOR INSPECTION.

See figure 2—1.

INTERIOR CHECK — ALL FLIGHTS.

1. Enter cockpit, personal equipment — Adjust and connect, zero delay lanyard — hook up to rip cord grip.

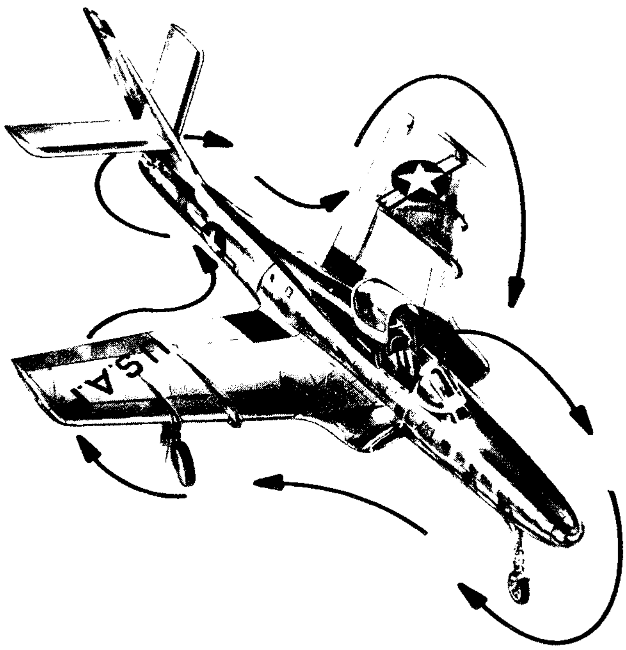
   a. Connect left and right survival kit parachute attaching straps. Make sure kit fasteners are properly positioned and pulled tight.

   b. Place right and left shoulder harness loops on safety belt swivel link.

   c. Place parachute lanyard anchor on safety belt swivel link and fasten safety belt.

   d. Leave shoulder harness control handle unlocked and adjust safety belt and shoulder harness.

   e. Connect personal leads (oxygen, radio, G-suit).



EXTERIOR INSPECTION

NOTE

At bases where ground personnel are not completely familiar with your aircraft make sure that the post flight and preflight are accomplished in accordance with the Technical Manual of Inspection Requirements, T. O. 1F-84(R)F-6 or WC 1F-84(R)F-6 PEPO.

Starting at the aft canopy make a complete exterior inspection. Check all surfaces for cracks, distortion, loose or missing rivets, signs of fuel, oil and hydraulic leaks; check that all access panels are secured and accomplish the following specific check.

1. AFT CANOPY

a. Pneumatic compressor compartment.
   (1) Check pressure 2500-3000 psi.
   (2) Close and secure door.
b. Hydraulic fluid reservoir – check gage and door secured.
c. Oil tank – check cap installed.

Figure 2-1 (Sheet 1 of 2)

2. FORWARD FUSELAGE

a. Ducts and screens — Condition and cleanliness
b. Static ports — Open
c. Brake accumulator — 450 - 550 PSI
d. General area — Check for fuel or hydraulic leaks
e. Camera ports — Windows clean and secure
f. Nose gear check
  (1) Strut — Correct extension
  (2) Tire — Condition, inflation, slippage mark
  (3) Ground wire — condition
  (4) Fairing doors — condition and security
g. Oxygen filler valve — cover secured.
h. All access doors and hatches — Secured.

3. RIGHT WING

a. Pylon jettison pressure — 1600 (±100) PSI
b. Pylon tank jettison pressure — 2000 (±100) PSI
c. Right main gear check
  (1) Tire — Condition, inflation,
  (2) Tire valve — Correct short type valve stem with flush cap.
d. Wheel well check — External tank ground refuel switch cover secured and ground refuel panel cover secured.

4. AFT FUSELAGE

a. Utility hydraulic system accumulator — 500 - 600 PSI
b. Power hydraulic system accumulator — 500 - 600 PSI
c. Hydraulic drain ports (3 on each side) — check for leaks
d. External tank fuel caps—Secure

5. DRAG CHUTE COMPARTMENT

a. Riser cable end fitting — In cable hook jaws
b. Ground handling pin — removed, pip pin — properly installed on left door
c. Check for moisture, fuel, oil or hydraulic fluid.
d. Secure door.

6. LEFT WING

a. Pitot static boom — cover removed, condition
b. Left main gear check
  (1) Tire — Condition, inflation, slippage mark
  (2) Tire valve — Correct short type valve stem with flush cap.
c. Pylon jettison pressure — 1600 (±100) PSI
d. Pylon tank jettison pressure — 2000 (±100) PSI

Figure 2—1 (Sheet 2 of 2)

WARNING

Failure to attach personal equipment correctly will prevent separation from the seat after ejection. Refer to figures 1–30, 1–31 and 4–13.

- When an Anti-G suit is worn, route the hose under the lap belt to prevent possible inadvertent actuation of the lap belt manual release should ejection become necessary.
- If G-suit is not used, make sure hose is properly stowed to prevent interference with control movement.

2. Seat vertical adjustment lock handle–Locked (5/8-inch gap 5 thru 25).

WARNING

After each vertical (height) adjustment of the seat on aircraft 5 thru 25, insure that the lock handle and locking pin indicator are separated by approximately 5/8 inch (½ to ¾ inch). Failure of the handle and the latch to separate as specified indicates one or more of the four seat locking pins has not locked. Failure of the top pins to lock in place can prevent successful seat ejection.

3. Oxygen system check – Complete, diluter lever – as required.

Refer to Section IV for oxygen system check.

For protection against carbon monoxide gas contamination when taxiing or run-up directly behind another aircraft or during run-up with tail into the wind, the following precautionary procedures will be used:

a. Before starting engine, don oxygen mask and place diluter lever at 100% OXYGEN position.

b. Use 100 percent oxygen during ground operation and takeoff when monoxide gas contamination is suspected.

c. Use NORMAL OXYGEN position when contamination is no longer suspected.

WARNING

- The oxygen diluter lever must be returned to the NORMAL OXYGEN position as soon as possible, because the use of the 100 percent oxygen throughout a long mission will so deplete the oxygen supply as to be hazardous to the pilot.
- Arrange oxygen hose so as not to interfere with full stick travel.

Note

The liquid oxygen gage should read between four and four-and-one-half liters when the oxygen system is fully charged. Do not be alarmed that the gage does not read five liters, since it is impossible to charge the liquid oxygen converter to five liters. Use the oxygen duration chart to determine your oxygen duration for the indicated supply.

4. Trim switch – Not sticking.

Check for security of mounting on the control stick. Operate the trim switch in all four ON positions and note that it automatically returns to the OFF position when released. If the switch sticks in any of the ON positions, enter this fact with a red cross on Form 781 and do not fly the aircraft.

CAUTION

Do not twist the grip as such action may cause the grip to become less secure.

Left Console.

1. All circuit breakers – IN.
2. Engine screen switch – AUTO or EXTEND.
3. Alternate trim switch – NORMAL.
4. Wing flap lever – Neutral.
5. Speed brake switch – IN.
6. Throttle – CLOSED.
7. Air refueling switches – DOWN.

8. Fuel shut-off valve switches — UP.
9. Emergency bus circuit breaker test light — Test.
10. Landing gear handle—Normal down and in detent.

Center Panel.

1. Altimeter, accelerometer, clock — Set.

CAUTION

Be sure the 10,000 foot pointer on the altimeter is set correctly.

2. Emergency brake handle — In.
3. Drag chute handle — In.
4. Viewfinder emergency door switch — NORM.
5. All other center panel switches — OFF.
6. Generator switches — ON.
7. Voltmeter selector — BUS.

Right Console.

1. All circuit breakers — IN.
2. Cabin pressure and temperature switches — As desired.
3. Camera compartment temperature switch—AUTOMATIC.

   Refer to section IV for manual operation.
4. Camera compartment temperature shut-off switch — ON.
5. Windshield defroster switch — ON.
6. All other right console switches — OFF or Normal.

INTERIOR CHECK — NIGHT FLIGHTS.

1. All lights — Check.
   a. Console lights — Check.
   b. Instrument panel lights — Check.
   c. Position lights — Check.
   d. Landing lights — Check.
   e. Taxi light — Check.
   f. Air refueling receptacle or probe light — Check.
   g. Auxiliary hand light — Check.

BEFORE STARTING ENGINE.

1. External power or battery switch — ON.
2. Warning lights — Check.
3. Altitude Indicator—Caged (hold until inverter is turned on).

Note

Use the alternate inverter if starting on battery and the main inverter if using external power.

4. Applicable inverter—ON.
5. Fuel quantity — Check.
6. Exhaust and intake areas — Check clear.

STARTING THE ENGINE.

Note

When making a battery start, the wing and forward pump pressure, and booster pressure warning lights will remain illuminated until the RPM reaches the generator cut-in speed, and the secondary bus is energized.

1. Fuel tank selector — ALL TANKS.

Note

If fuel tank selector binds turn the engine over by using the engine crank switch, or hold pressure against the fuel tank selector toward any on position while making a normal start. As the engine picks up speed the selector will free up and if positioned to a tank containing fuel, a normal start will follow. The engine-driven fuel pump depends upon engine fuel for its lubrication. Therefore, operating this pump without fuel may cause pump damage.

2. Throttle — IDLE.

Note

Index the starter switch to START immediately after the throttle is moved to IDLE.

3. Starter switch — HOLD to START.

   Approximately three seconds are required to energize the combustion starter, engine primer and ignition timer.

CAUTION

Observe EGT gage to ascertain engine light off and to assure that maximum EGT will not be exceeded. EGT may be controlled by retarding the throttle toward OFF not to exceed half way.

WARNING

The starter switch shall not be actuated more than one time during any starting cycle.

Note

If maximum EGT is anticipated the following procedure is recommended as an alternative for steps 2 and 3.

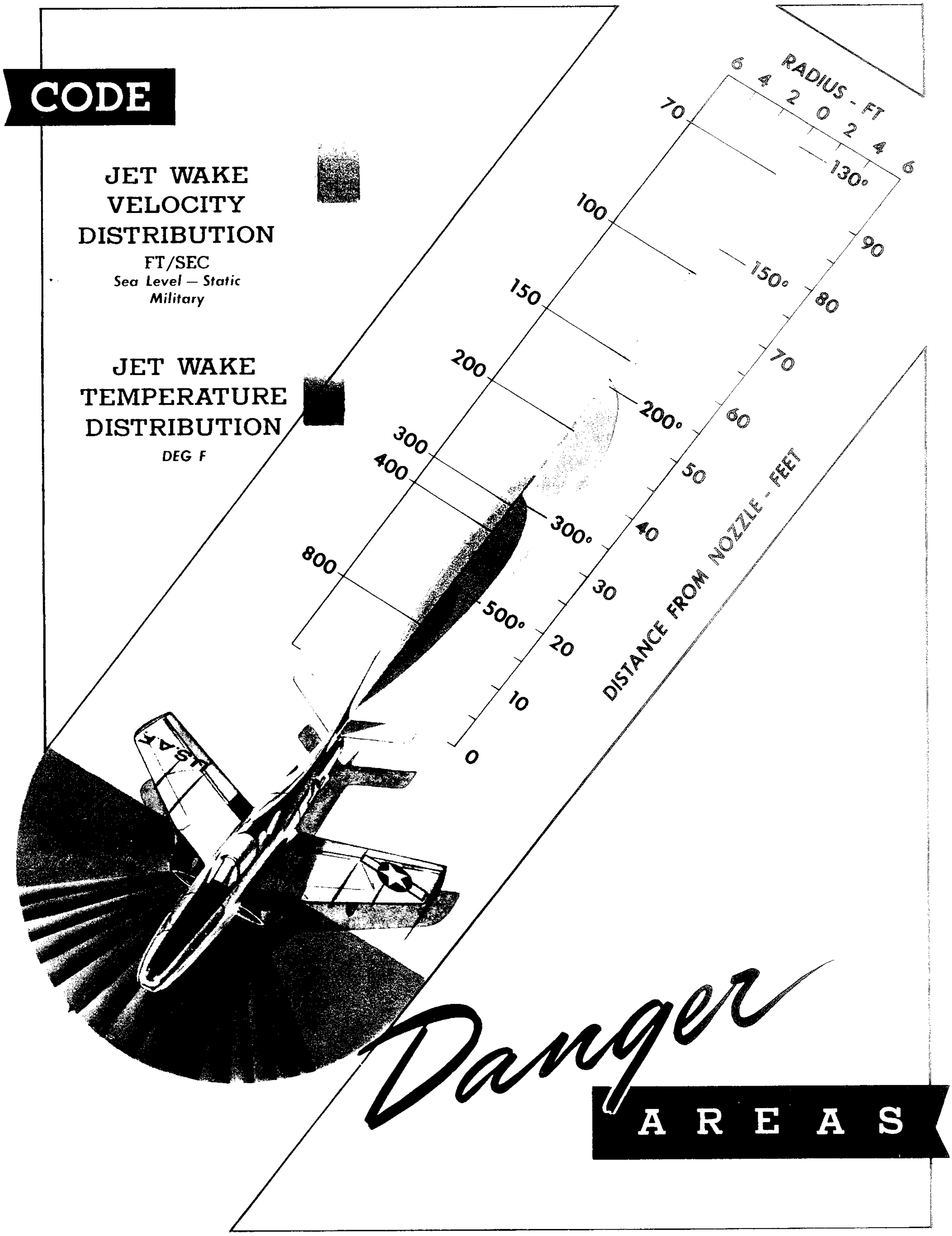

CODE
JET WAKE VELOCITY DISTRIBUTION
FT/SEC
Sea Level — Static
Military
JET WAKE TEMPERATURE DISTRIBUTION
DEG F
RADIUS - FT
6 4 2 0 2 4 6
70
100
150
200
300
400
800
130°
150°
200°
300°
500°
90
80
70
60
50
40
30
20
10
0
DISTANCE FROM NOZZLE - FEET
U.S.A.F.
Danger
AREAS

Figure 2—2

a. Throttle—CLOSED.

b. Engine crank switch—ENGINE CRANK and simultaneously depress air start switch.

Note

The engine starter is very efficient in that it will normally cause the engine to attain maximum starting RPM (approximately 2,000 RPM) in a maximum of 3.5 seconds. In order to assure a satisfactory light off, the throttle must be in the idle detent when engine speed is increasing through the 15-18 percent RPM range.

c. When RPM reaches 15-18 percent, throttle—IDLE.

4. As RPM rises through 30 percent RPM or 300 EGT, whichever comes first, starter switch—STOP—START.

As the RPM rises through 30 percent RPM, place the starter switch to STOP-START for approximately 2 seconds to assure that the primer valve is closed and the circuit to the engine ignition is broken.

5. Oil pressure—Check.

When engine speed has stabilized at idle (36 to 42 percent RPM for J65-3 engines and 42 to 48 percent RPM for J65-7 engines), check that the oil pressure is within limits. If there is no oil pressure after 60 seconds, or if pressure drops after a few minutes ground operation, shut down and check for blown lines or for congealed oil.

6. Engine instruments—Within limits.

7. Main inverter—ON (ALT inverter—OFF).

Note

When the ambient air temperature exceeds 32 C (90 F) maintain idle RPM exhaust gas temperature within limits by manually advancing the throttle to a higher RPM.

CAUTION

During static ground operation, the engine should not be operated in the 60-82 percent RPM range except during acceleration or deceleration through this range.

8. If external power used—Disconnect and battery switch ON.

FALSE START PROCEDURE.

An unsuccessful start will result if the exhaust gas temperature reaches maximum allowable for start, or if the EGT does not commence to rise before the engine has accelerated and then decelerated to 17 percent RPM, or if idle RPM is not obtained within 60 seconds after energizing start switch.

CAUTION

Before attempting a restart, place starter switch in STOP START position momentarily, close throttle and allow the engine to stop. Be sure all fuel has completely drained from the combustion chamber by observing the drain valve. Restart only if "hot start" conditions have not been exceeded. If the engine fails to start after one attempt, the starter air storage bottle must be recharged and external power must be used to recharge the starter fuel accumulator. If the engine fails to start after two starting attempts, a 45 minute cooling period is required before making a third attempt. Unlimited single starts at 30 minute intervals may be made thereafter.

WARNING

If a start is attempted and combustion does not occur in the starter, fuel can drain into the pants duct area and possibly into the generator ducts causing a fire hazard. When a false start of the combustion starter occurs, actuate the rotor test switch to scavenge the starter with air. Make a thorough inspection of the compressor inlet for latent fuel, which should be removed before the next start. Wait for completion of engine fuel system drainage before attempting another start. If a start is attempted and the attempt is accompanied by a loud noise for approximately ½ second indicating that combustion has taken place in the starter, and no engine RPM is indicated, the starter has become disengaged from the engine. Additional start attempts will not remedy this situation. Another start should not be attempted until the starter has been removed from the engine and the condition responsible for the unlatching has been corrected.

ENGINE GROUND OPERATION.

WARMUP.

No warmup period is required. Takeoff may be ma immediately if engine instruments show normal rea ings. However, some engines will not accelera properly when they are cold and will therefore requ a short warmup period in order that the accelerati limits (See Section V) are met.

CAUTION

During static ground operation, the engine will not be operated in the 60-82 per cent RPM range except during acceleration or deceleration through this range.

Note

In the event condensation in the form of mist or snow is blown into the cockpit from the air conditioning-outlets and is found to be objectionable to the pilot, the cabin vent switch can be positioned to RAM, or a higher temperature selected on the cabin temperature selector.

- In the event warmup or ground operation is accomplished approximately 8-10 minutes of steady operation at 100 percent RPM or maximum EGT is normally required to obtain stabilized EGT readings after starting a cold engine. A stabilized operation temperature is one that will remain constant for three individual readings taken at 30 second intervals. In many instances it is necessary to monitor the throttle to preclude prolonged overtemperature operation. Therefore, when possible, stabilized operating temperatures should be achieved by slowly advancing the throttle to 100 percent RPM or maximum EGT whichever occurs first. In the event EGT limits are reached prior to 100 percent RPM, the throttle may be advanced slowly as temperature decreases.

RPM AND FUEL FLOW FLUCTUATION.

Refer to Section V for RPM and fuel flow fluctuation limits.

BEFORE TAXIING.

1. Communications and navigation equipment—ON.
2. IFF SIF—STANDBY.
3. Hydraulic pressure—Check.
   Static pressure should be 1400-1600 PSI.
4. Speed brakes—Check operation then IN.
5. Flaps—Full down then return to take off (50 percent).
6. Loadmeter—Approximately 0.25 to 0.5.
7. Voltmeter—Check all positions within range.
8. Pitot heat—check climatic.
   Check operation of pitot heater by turning heater on and feeling pitot head for heating (assisted by crew chief).
9. Canopy dry air switch—Climatic.
10. Rudder trim—Check.
11. Stick trim—Check.
    Actuate and check operation of the stabilator and ailerons. Check aileron trim light ON, and stabilator aligned.
12. Communication, navigation, and camera equipment—Check operation.
    a. Command radio—Check necessary frequencies and guard frequencies.
    b. Radio compass—Check all positions then turn to first desired position (Unmodified).

c. TACAN—Check operation (Modified).

d. Camera switches—ON, if applicable. Check camera operation.

13. Seat safety pins—Remove, show to crew chief and stow.

14. Landing gear safety clips (nose, left and right gears) removed, display for pilot and stow.

15. Brakes—Check operation

Check for firm and positive brake pedal feel. Individually depress brake pedals and note fluctuation of utility hydraulic pressure each time a brake pedal is depressed.

16. Chocks—Removed

TAXIING.

Remove chocks, release brakes and increase power until aircraft starts to move. Once the aircraft is moving, taxi at the lowest practicable RPM to conserve fuel and avoid damage from tailpipe blast. Brakes are required for steering as the rudder is ineffective at low speeds and nose wheel steering is not provided. The brakes are quite sensitive and care must be used at all times, if jerky operation is to be avoided. When the aircraft is fully loaded with external stores, a much higher RPM is necessary to start the aircraft rolling. When heavily loaded, the turning radius must be slightly increased to prevent excessive side loads on the struts and tires. Limit taxiing to a minimum as the aircraft range is decreased by the high rate of fuel consumed during taxiing. Fuel consumed during taxiing is approximately 15 to 25 pounds per minute.

CAUTION

During taxiing, avoid engine operation in the 60-82 percent RPM range.

- Do not taxi with canopy open when wind gusts or taxi speeds will exceed 40 knots.
- Do not rest arms on canopy rails when canopy is open, as injury will result if the canopy unexpectedly closes during taxiing.
- Avoid excessive use of brakes during extended taxiing. Expander tubes may become overheated and fail.

1. Flight instruments—Check operation.
2. Windshield defrosting—As required.

BEFORE TAKING RUNWAY.

1. Recheck — Pins, lanyard, trim, flaps and speed brakes.
2. Canopy — Closed and locked.
3. Engine screens — As required.

   Extended screens reduce takeoff thrust approximately 4.4 percent.
4. Pitot heat switch — Climatic.
5. Fuel quantity — Check (1200 lbs min) and set fuel quantity selector to MAIN.
6. Flight controls — Check for free and correct movement.

RUNWAY CHECK.

1. Flight instruments — Check.
   a. Heading indicator — Check with runway.
   b. Attitude indicator — Check setting along with indices.
2. Engine acceleration and overspeed — Check.

   Advance throttle in one second from 47 percent (IDLE) RPM to 100 percent RPM. The engine should accelerate to 100 percent RPM (and not exceed 101 percent) in 15 seconds at sea level within EGT limits. Refer to Sections V and VII.

   Note

   This check is required only prior to first flight of the day. It may be performed prior to any flight if the pilot thinks it necessary.
3. Engine instruments and hydraulic pressure gage — Check.
4. Ato ready switch — ATO READY (for ATO takeoff).
5. IFF/SIF — NORMAL (mode and code as briefed).

TAKEOFF.

1. Throttle — Military Thrust.
2. Release brakes.
   a. Maintain directional control, by minimum use of brakes until rudder effective speed of approximately 60 KIAS.
   b. If using ATO, ATO switch — depress when IAS reaches value determined from the Appendix.
   c. Check aircraft acceleration, by checking IAS at predetermined points on runway.
   d. Leave control stick in neutral, until rotation speed is reached.
3. Go-No-Go speed — Check.
   a. At rotation speed (5 knots below lift-off), use necessary stick travel at such a rate that it will assume the pitch angle for takeoff at the recommended lift-off speed.

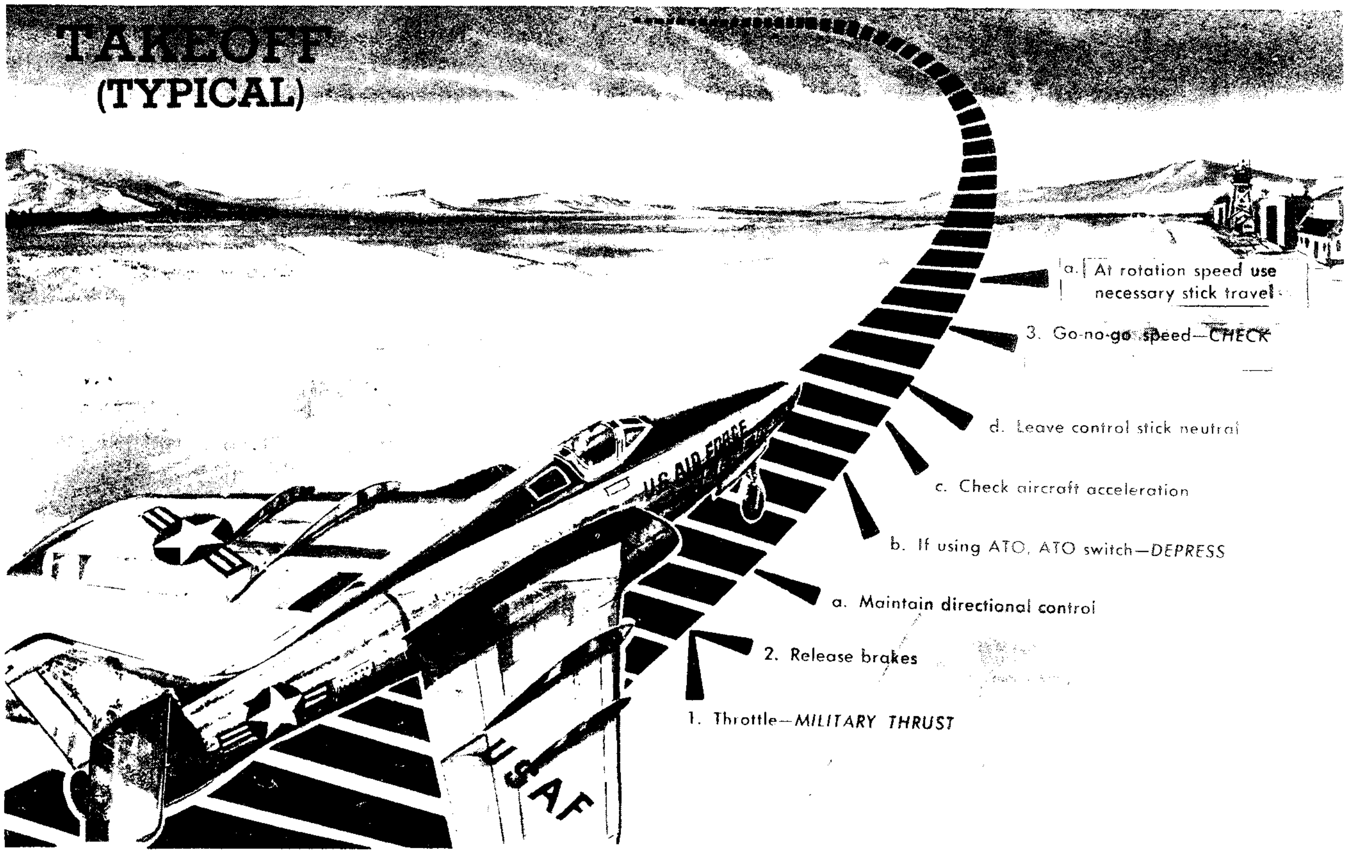


Figure 2—3.

WARNING

- Make certain aircraft is rotated at correct speed.
- Early rotation increases drag.
- Late rotation increases takeoff run.
- Do not rotate to an excessive nose high attitude.

See figure 2—3 for typical takeoff.

ASYMMETRICAL TAKEOFF.

Asymmetric takeoffs can be accomplished without difficulty provided aileron boost is available. In the event of complete hydraulic failure it is imperative that the stores be jettisoned *immediately* as the aileron stick forces will be excessive. The following procedure is recommended for an asymmetric takeoff.

a. Takeoff speed should be approximately 10 knots higher than the recommended takeoff speed for corresponding gross weight as noted in the Appendix.

b. Use brakes in conjunction with the rudder to maintain directional control until the rudder becomes effective at approximately 60 knots.

c. As much as 1/2 aileron travel may be required to maintain a level attitude.

d. Takeoff distances will be increased approximately 15 percent from that shown in the Appendix due to the use of brakes and higher takeoff speed.

ASSISTED TAKEOFF.

The effect of ATO on aircraft trim is slight as the units are installed near the fuselage center line. No special technique is required. Takeoff performance will depend on the speed at which the ATO units are fired during the takeoff run. Refer to the Appendix for the ATO cut-in speed for a two or a four ATO unit takeoff.

FAILURE OF ASSIST TAKEOFF UNITS.

In the event of failure of one or more of the ATO units during takeoff, rapid change of trim will not be necessary as the ATO units are close to the center line of the aircraft. Takeoff distance will be increased as noted in the Appendix.

AFTER TAKEOFF — CLIMB.

1. Landing gear selector handle — UP.

   The landing gear will retract in approximately 10 seconds at normal temperatures. Do not exceed 190 KIAS until landing gear is up and locked.

   When definitely airborne move landing gear selector to the UP position, feel for definite engagement in the detent. Check landing gear position indicators and warning light.

CAUTION

Do not use Warning Silencing Button while gear is in transit. If the warning sound is silenced during gear transit, and the gear fails to lock up, the warning will stay silenced and will not provide the desired warning sound. If it is determined the gear will not actuate to a safe position, the silencing button may be depressed to stop the warning signal.

2. Wing flap lever — UP at 190 KIAS MIN.

   After all obstacles are cleared, retract flaps.

3. ATO jettison switch — Depress, ATO ready switch — OFF (if applicable).

   Jettison ATO units when at a safe altitude, by depressing ATO jettison switch.

- Leave throttle in military thrust until the aircraft reaches a safe altitude observing the time limit specified.

4. Engine screen switch — As desired.
5. Pneumatic compressor switch — ON (unmodified aircraft).
6. Armament and gun heater switches — As desired.

Note

Leave pylon tank air pressure switches in the OFF position until the forward tanks fuel level drops. This procedure will purge the fuel transfer lines of air.

7. Pylon tank air pressure switches — OUTB'D or INB'D.

   Check external tank feeding (Refer to Section VII).

8. Engine and electrical indicators — Normal readings.

Note

Engine RPM droop during climb is sometimes encountered. Engine RPM droop when encountered, usually starts appearing at 10,000 feet and increases with altitude. An RPM droop of 1½ percent RPM at high altitude is considered acceptable.

9. Oxygen diluter lever — NORMAL OXYGEN.
10. IFF/SIF — Check.

    If positive operation of the normal mode of IFF/SIF has not been established during departure with an air traffic control facility, a check should be made with such a facility as soon after takeoff as flight conditions permit.

Note

For aircraft operating on Radar Controlled Airways or through Positive Controlled Airspace this check must be made before entering a radar advisory area. If the IFF/SIF is inoperative, consult the appropriate FLIP publications.

11. Zero delay lanyard — Disconnect.

    Refer to zero delay lanyard engagement requirements chart in figure 3—3 for altitude at whic to disconnect.

12. Defroster Control—ON.
13. Altimeter—Reset in accordance with FLIP.

EN ROUTE.

- Autopilot — Engage if desired (if installed).

  Trim aircraft for wing-level flight, desired pitch trim and directional heading.

  Hold stick firmly to prevent any malfunction from resulting in an abrupt maneuver and engage autopilot switch.

WARNING

Do not engage autopilot above 320 KIAS below 27,500 feet.

- Do not engage autopilot when close to the ground or near other aircraft.

- Pneumatic compressor switch — OFF after 30 minutes.

CAUTION

The compressor should not be operated for more than 30 minutes continuously to prevent overheating and possible compressor explosion. Therefore, if the storage bottles take an excessive amount of time to refill, the compressor should be shut off.

- Fuel check, every 30 minutes.
- Oxygen – Check.
- Pressurization – Check.
- Heading indicator – Check.
- Engine instruments – Check.

CAUTION

The throttle may have to be retarded to keep the exhaust gas temperature within specified limits. If necessary to throttle back below 35,000 feet altitude in order to maintain stabilized exhaust gas temperature within limits a notation should be made of this fact on the Form 781.

FLIGHT CHARACTERISTICS.

Refer to Section VI for detailed information on the aircraft flight characteristics.

DESCENT.

Note

When descending, consideration must be given to keeping the pylon tanks pressurized to prevent them from collapsing. Refer to PYLON TANK PRESSURIZATION in Section VII.

1. PITOT heat switch–Climatic.

Note

Due to the large mass of glass in the bullet-proof windshield, it is essential that the defroster be turned on during climb to altitude.

Since a descent often cannot be anticipated 30 minutes in advance, the defroster should be turned on whenever an altitude of 20,000 feet is reached, and left on for the duration of the flight. During the ground support mission the defroster should be turned on immediately after takeoff. If it is found that the defroster air is excessively uncomfortable, the defroster valve should be closed down to some intermediate position, but should not be turned off, unless during low altitude summer operations.

2. Pneumatic compressor switch–OFF.
3. Zero delay lanyard–CONNECTED.
4. Oxygen–100 percent OXYGEN if desired.
5. IFF/SIF–Check

   Within one hour prior to estimated time of landing, a positive IFF/SIF check should be made with air-traffic control facility.
6. Altimeter–Reset in accordance with FLIP.
7. Defroster control–as required.

BEFORE LANDING.

See figure 2–4 for typical landing pattern.

- Enter initial approach pattern at 300 KIAS.

  Pitch trim may be used throughout the pattern to lighten stick loads.
- Speed brake switch – OUT on break.
- Throttle – Retard momentarily to check warning system.
- Reduce speed to 225 KIAS
- Maintain break altitude and 225 KIAS until gear down.

LANDING PATTERN (TYPICAL)

- AIRSPEEDS GIVEN ARE BASED ON LANDING GROSS WEIGHT OF 16,000 POUNDS.
- AIRSPEEDS FOR OTHER GROSS WEIGHTS WILL BE FOUND IN THE APPENDIX.
- ADD 5 KNOTS TO THE BASE LEG AND FINAL APPROACH SPEEDS FOR EACH 1,000 POUNDS OF FUEL ABOVE 1,000 POUNDS.

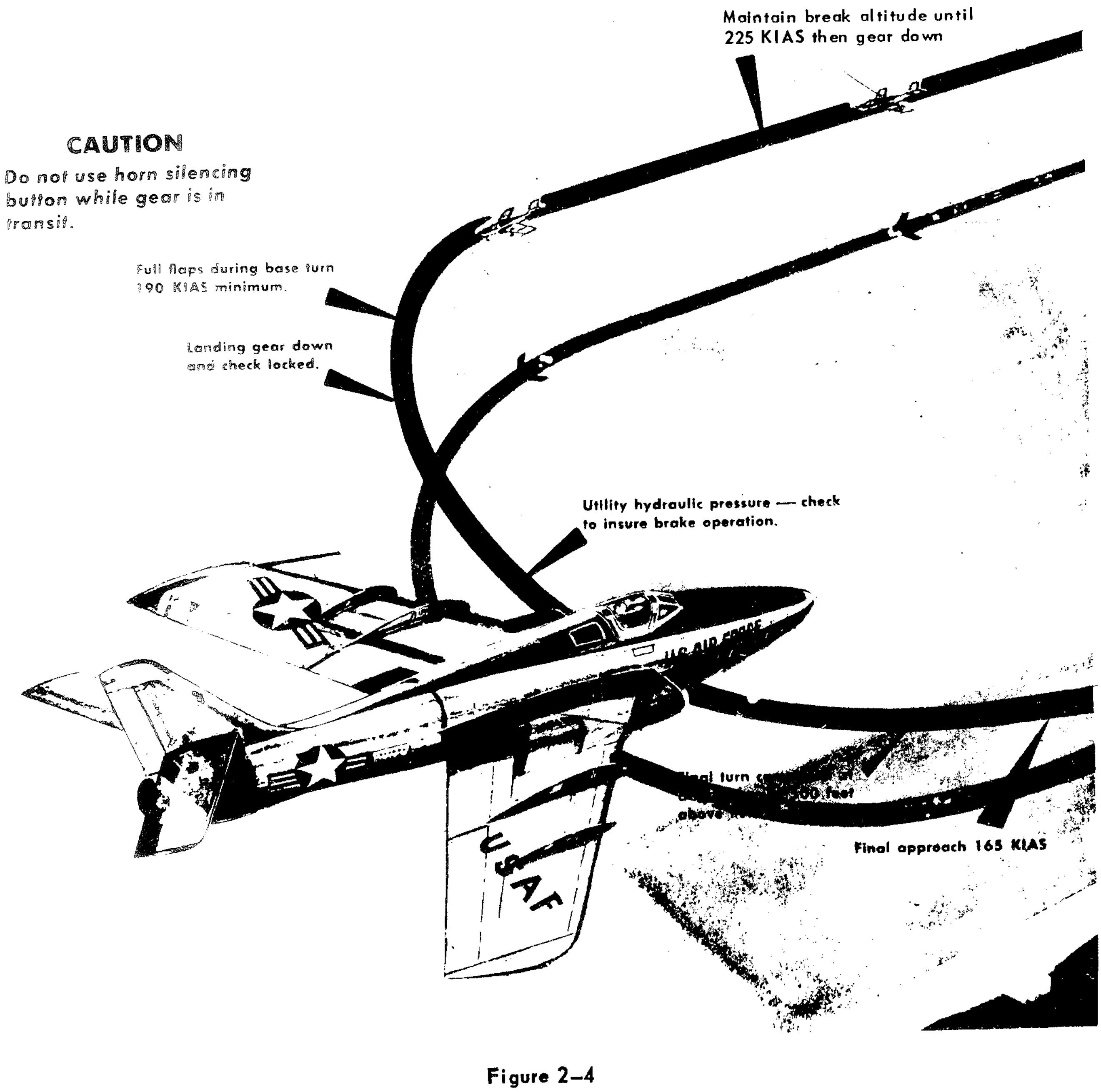
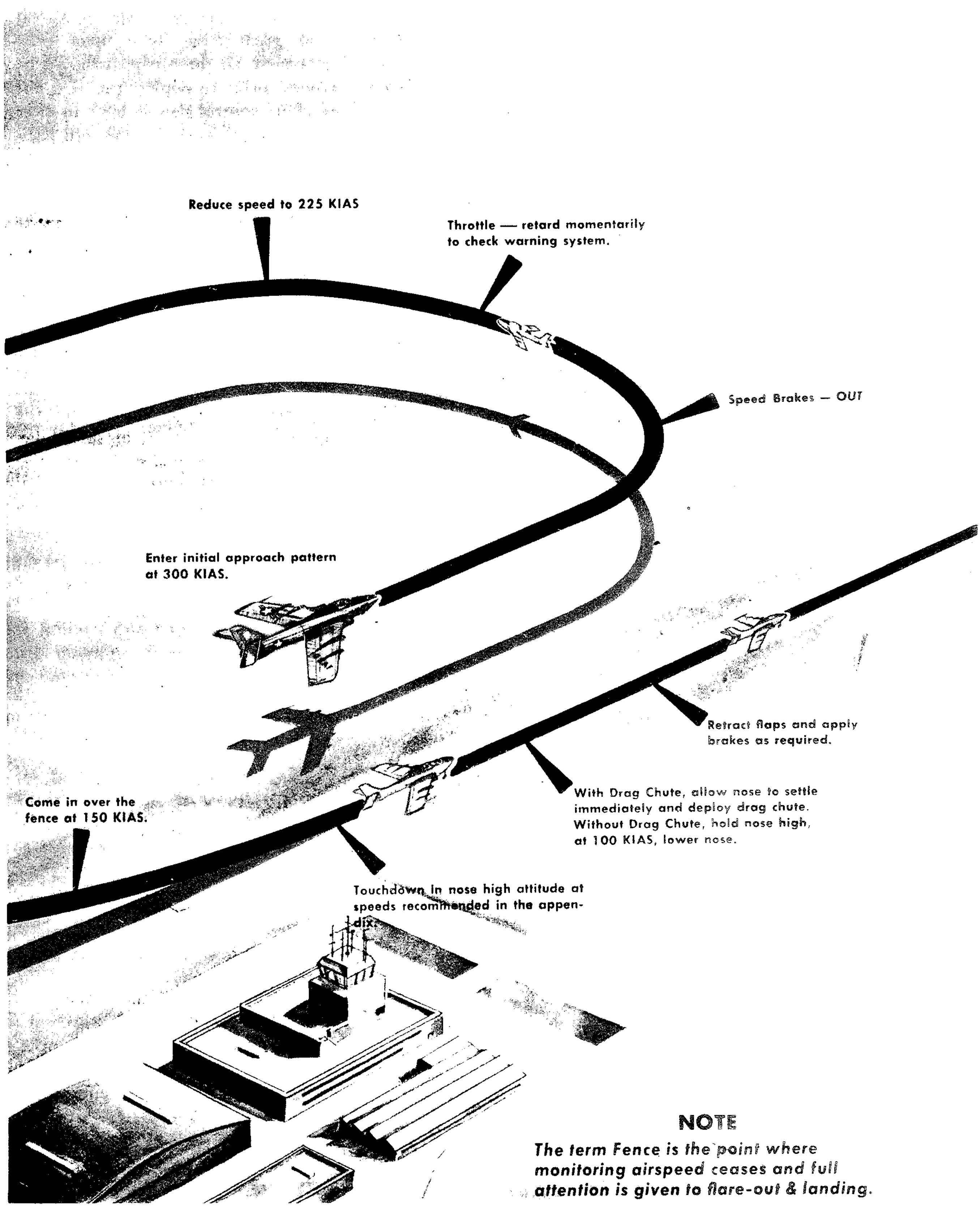


Figure 2–4


Reduce speed to 225 KIAS
Throttle — retard momentarily to check warning system.
Speed Brakes — OUT
Enter initial approach pattern at 300 KIAS.
Retract flaps and apply brakes as required.
With Drag Chute, allow nose to settle immediately and deploy drag chute. Without Drag Chute, hold nose high, at 100 KIAS, lower nose.
Come in over the fence at 150 KIAS.
Touchdown in nose high attitude at speeds recommended in the appendix.
NOTE
The term Fence is the point where monitoring airspeed ceases and full attention is given to flare-out & landing.

1. Compute final approach and touchdown airspeeds.
2. Speed Brakes – OUT.
3. Landing gear – Down, and locked – Check.

   Landing gear will extend and lock in approximately six seconds at normal temperature. The taxi light may be used as an indicator that nose gear is down and locked (observed by mobile control).

CAUTION

Do not use Warning Silencing Button while gear is in transit. If the warning sound is silenced during gear transit, and the gear fails to lock down, the horn or signal will stay silenced and will not provide the desired warning sound. If it is determined the gear cannot be locked down, the silencing button may be depressed to stop the warning signal.

4. Flaps–Down during base turn, maintain 190 KIAS.
5. Flap Handle–Neutral (after confirming that flaps are fully extended).
6. Utility hydraulic pressure–Check to insure brake operation.

LANDING.

Note

Landing with fuel in the external tanks requires that a good landing technique be employed to prevent wrinkling or buckling the wings during such landings.

It is recommended that the drag chute be jettisoned prior to stopping to prevent damage to the chute through contact with the runway or prior to using high power because engine exhaust may burn the shroud lines.

NORMAL LANDING.

Refer to the Appendix for approach and landing speeds. Ground roll distances in the Appendix are based on maximum braking.

- Touchdown in a nose high attitude.
- a. Without drag chute, apply enough aft stick to hold nose high for aerodynamic drag. (Don't drag tail.) Lower nose at 100 knots.

  b. With drag chute, allow nose to settle to ground immediately and deploy drag chute.

CAUTION

Due to the faster sink rate and a strong tendency to pitch nose down upon drag chute deployment, all three wheels should be on the ground prior to deployment. Do not push drag chute control handle back in after deploying drag chute as this action will jettison the drag chute.

- Do not apply hard brakes until nose wheel has contacted the ground so as to minimize landing load on the nose wheel strut.
- With nose wheel on ground, retract flaps and apply brakes as required.

Note

Directional control is good throughout the entire landing roll. However, at speeds below approximately 50 knots, brakes are also required for directional control.

MINIMUM DISTANCE LANDING ROLL ON DRY RUNWAY – WITH OR WITHOUT DRAG CHUTE.

Refer to Appendix for approach and landing speeds and ground roll distances. For a minimum distance landing roll the following procedure is recommended:

a. Turn on to the final approach should be made further out than normal.

b. During final approach, reduce speed to precomputed final approach speed for the aircraft gross weight as shown in the Appendix and adjust power so that a 200 to 300 FT/MIN rate of descent can be held.

c. Immediately prior to touchdown, reduce power to IDLE and touchdown at recommended touchdown speed for the aircraft gross weight as shown in the Appendix.

d. On touchdown allow the nose wheel to settle to the ground immediately and deploy the drag chute, if installed and raise wing flaps. Apply braking while gradually applying aft stick. Continue to increase braking so as to hold the nose wheel on the ground as aft stick is applied. Optimum braking is obtained when full aft stick is reached with the nose wheel still on the runway.

LANDING ON WET OR ICY RUNWAYS.

The procedure for landing on wet or icy runways is covered in Section IX.

LANDING WITH EXTERNAL LOAD.

The procedure for landing with external load is the same as for normal landing except the speed will be higher because of the higher stalling speed due to the greater weight. Caution must be exercised when landing with fuel in the pylon tanks because the load applied to the wing structure may cause wrinkles in the wings unless very smooth landings are made.

CROSS WIND LANDING.

The procedure for cross wind landing is the same as for normal landing. However, if the drift appears excessive, the upwind wing may be lowered just before contact. During landing roll, the aircraft can be held in a straight path with the rudder until speed reduces to approximately 70 knots. At the lower speeds, direction is controlled by the use of the brakes. Refer to crosswind landing charts in the Appendix of this manual.

EMERGENCY LANDING.

Refer to Section III for procedure in event of an emergency landing.

TOUCH AND GO LANDING.

Touch-and-go landings introduce a significant element of danger because of the many rapid actions which must be executed while rolling on the runway at high speed or while flying in the immediate proximity to the ground. Therefore touch-and-go landings should be made only when authorized and directed by the major command concerned. The following procedure is recommended:

a. Throttle — Increase to 100 percent RPM while retracting speed brakes.

b. Instruments — Cross check for proper indications.

Throttle should be moved smoothly to avoid overtemperature, overspeed, etc.

c. Airspeed — Accelerate to proper airspeed (approximately 140 knots with no external stores, before attempting lift-off).

d. Landing gear — Retract when safely airborne and raise wing flaps between 190 and 225 KIAS.

e. Landing lights — For night go-arounds, retract landing lights as soon as practicable.

GO-AROUND.

The decision to go-around should be made as soon as possible. See figure 2–5 for a typical Go-Around.

1. Throttle — Open smoothly to MILITARY THRUST and simultaneously return speed brake switch to IN.
2. Drag chute — Jettison if deployed.
3. Landing gear — Retract when definite rate of climb is established.

CAUTION

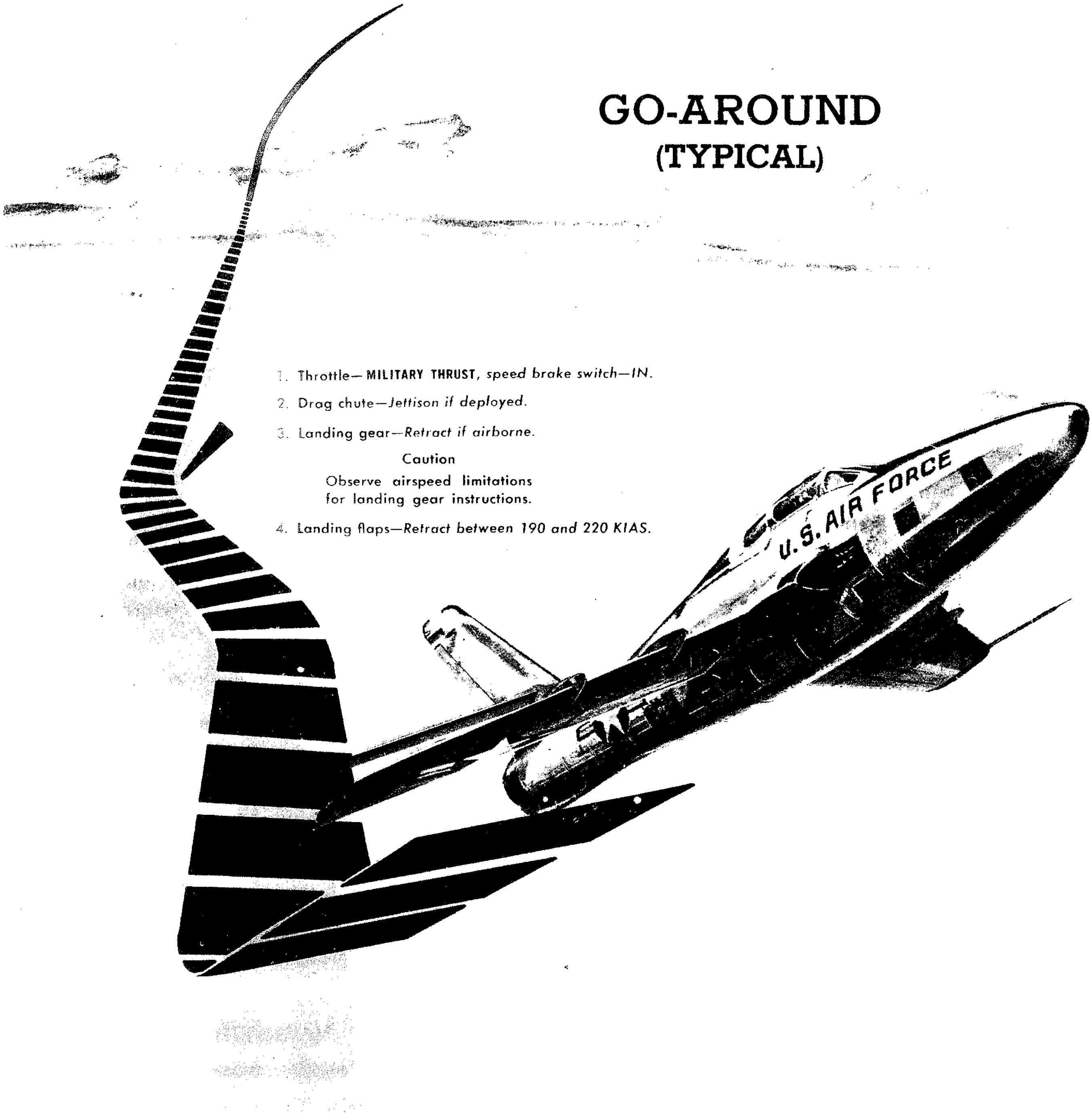
Observe airspeed limitations for landing gear retraction. (190 KIAS).

4. Landing flaps — Retract between 190 and 225 KIAS.

WARNING

Approximately 300 pounds of fuel are required for a go-around.

GO-AROUND
(TYPICAL)



1. Throttle—**MILITARY THRUST**, *speed brake switch—IN*.
2. Drag chute—*Jettison if deployed.*
3. Landing gear—*Retract if airborne.*

   Caution

   Observe airspeed limitations for landing gear instructions.

4. Landing flaps—*Retract between 190 and 220 KIAS.*

Note

THE DECISION TO GO-AROUND SHOULD BE MADE AS SOON AS POSSIBLE.

Figure 2—5

AFTER LANDING.

1. Drag chute — Jettison while rolling straight (if used).
2. Stop aircraft off runway.
3. Pitot heat switch — OFF.
4. Pylon tank air pressure switches — OFF.
5. Engine screen switch — EXTEND.
6. Speed brake switch — IN.
7. Wing flap lever — NEUTRAL.
8. Drag chute handle — IN (if used).

   Reset drag chute handle to preclude discharging battery.
9. Windshield and canopy defroster — OFF.
10. IFF SIF and navigation radios — OFF.

Note

Turn IFF SIF off as soon after landing as possible. This will eliminate signals from taxiing or parked aircraft which would block the controller's scope and interfere with control of airborne aircraft.

11. Seat pins — IN.
12. Taxi to parking area.

CAUTION

Do not open canopy when wind gusts or taxi speeds will exceed 40 knots.

ENGINE SHUTDOWN.

1. Main inverter switch — OFF.
2. Alternate inverter switch — OFF.
3. Command radio — OFF.
4. Engine screen switch — RETRACT.
5. Wheel chocks — In place.
6. Throttle — CLOSED.

Note

If the engine RPM has not exceeded 60 percent RPM during taxiing the engine can be shut down immediately, otherwise idle the engine for one minute prior to shutdown.

5. Engine run down time — Check.
6. After engine stops, fuel selector — OFF.

   The fuel selector is turned off after the engine has stopped to prevent it from sticking in the OFF position.
7. All switches except generators — OFF.

CAUTION

Turning the battery switch off before the engine stops may allow hydraulic fluid to leak past the landing gear selector valve and release the landing gear downlocks.

IF ENGINE FAILS TO STOP WHEN THROTTLE IS CLOSED.

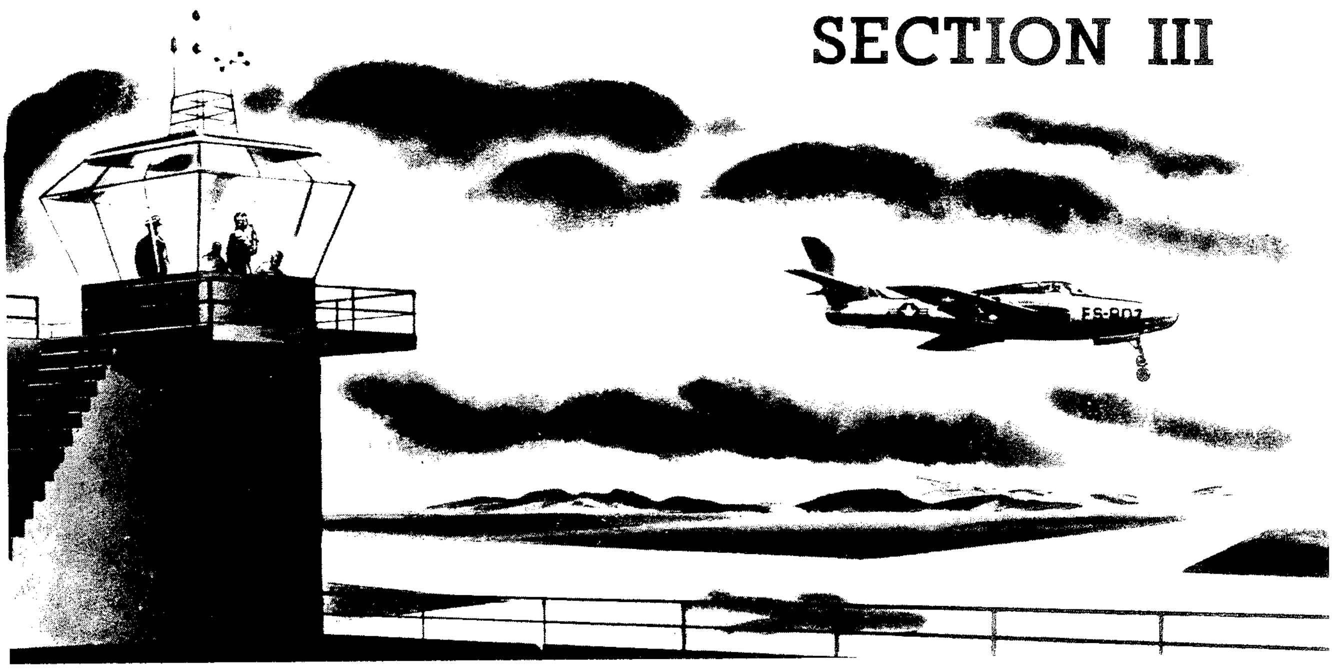
If closing the throttle does not stop the engine, place the fuel tank selector in the OFF position. The OFF position shuts off the fuel supply to the engine and it will stop without danger of fire. When this procedure is used make an entry in the Form 781 in order to have maintenance personnel check the engine driven fuel pump for damage and the fuel tank selector for binding.

BEFORE LEAVING THE AIRCRAFT.

1. Fill out form 781.

   Make appropriate entries covering any limits in the Flight Manual that have been exceeded during the 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 takeoff, long and fast landings and long taxi runs at high speeds, etc.

SECTION III



Emergency Procedures

TABLE OF CONTENTS

ABORT.

On most takeoff emergencies the pilot is faced with the immediate decision between aborting or continuing the takeoff. When the decision is made to abort, certain factors about the runway and aircraft known to the pilot have a direct bearing on the success of the abort and must be considered by the pilot, and his procedure accomplished accordingly. These runway and aircraft factors are:

Length of runway and gradient.

Condition of surface (dry, wet, ice or snow).

Condition or type of runway over-run area.

Availability of and type of barrier.

Aircraft configuration (gross weight and external stores).

The time the decision to abort is made and the particular failure will further compromise the procedure as follows:

Early decision to abort – Low speed, maximum runway available.

Late decision to abort – High speed, minimum runway available.

Availability of brakes.

Availability of drag chute.

The decision to jettison or retain external stores must be made by the pilot. Jettisoning full pylon tanks on the runway is hazardous. If it is impossible to stay on the runway, full pylon tanks should be jettisoned prior to leaving the prepared surface. Successful barrier engagements can be made with tanks retained. Certain stores may be jettisoned to reduce gross weight, or establish a configuration that will improve barrier engagement probability. Barrier engagements can successfully be accomplished at any ground speeds from a minimum of approximately 35 knots up to a maximum of 130 knots, but more positive engagement can be assured if the barrier is contacted at a speed as close to minimum as possible. Off center engagements are successful and may be safely accomplished. If target flags are attached to the barrier webbing adapter, it is advisable to avoid hitting the webbing right at the target flag. It is possible for the flag to foul the webbing and prevent engagement. The landing gear should be left extended (regardless of barrier availability) as it will absorb much of the initial shock. There are no recorded cases of aircraft flipping or tumbling with the gear extended.

ABORT/BARRIER ENGAGEMENT PROCEDURE.

1. **Throttle – IDLE** (For fire CLOSED and emergency brake handle – PULL).
2. **Drag chute – Deploy**
3. **External stores – Jettison** if necessary
4. Braking action – Aerodynamic – maximum safe breaking. Excessive use of the brakes might result in tire blowout. A blowout could result in a loss of directional control and failure to stay on the runway.
5. Engage barrier squarely even if this results in barrier contact at one side of the runway.
6. Immediately prior to engaging barrier or leaving runway:
   a. Throttle – CLOSED, and emergency brake-handle – PULL.
   b. Fuel tank selector – OFF.
   c. Shoulder harness – LOCK.
   d. Battery and generator switch – OFF.
7. When aircraft has stopped, evacuate cockpit.

WARNING

Use extreme care during evacuation of cockpit to avoid possible actuation of the ejection seat.

ZOOM-UP MANEUVER.

In the event of engine failure, fire or overheat during takeoff when it is too late to abort, or during flight at low altitude, and with sufficient airspeed available, the aircraft should be pulled up (zoom-up) to exchange airspeed for an increase in altitude. This will allow more time for accomplishing subsequent emergency procedures (air start, establishing forced landing pattern, ejection, etc.). Maximum altitude gain can be achieved by jettisoning external loads prior to zoom-up. The further up the climbing flight path that external loads are jettisoned, the less additional altitude will be gained. The point at which climb should be terminated will depend on whether the pilot intends to eject or whether he intends to continue attempting air starts, establish forced landing pattern, etc. In any event, it is recommended that:

Air start be attempted immediately upon detection of engine flame-out and repeated as many times as possible during the zoom-up.

If the decision is to continue attempting air starts, the climb should be terminated prior to the airspeed dropping below best glide speed in order that engine windmilling RPM will not drop below the minimum required for air start.

If the decision is to eject, the aircraft should be allowed to climb as far as possible. Ejection should be accomplished while the nose of the aircraft is above the horizon but prior to reaching a stall or sink.

ENGINE FAILURE.

Engine failure is defined as a loss of thrust, caused by complete engine failure or a flame-out. Complete engine failure, which rarely occurs, is internal damage to the engine, such as engine seizure or explosion. Engine instruments and operating characteristics often give indications of impending complete engine failure. Reduction of thrust and mimimizing G forces will generally prolong the engine operating time prior to complete engine failure. Flame-outs are generally due to improper fuel scheduling caused by fuel system malfünction or improper fuel system management. Fuel system warning lights and the fuel flow indicator often give indications of an impending flame-out. Appropriate emergency operating procedures may prevent a flame-out. Because of the many variables associated with an engine failure it is impossible to establish a predetermined set of rules which would provide a ready made procedure applicable to all circumstances. Also because of the many variables encountered, the final decision must remain with the pilot as to which procedure he will follow. The pilot's analysis of the condition of the aircraft, type of emergency and his proficiency are of prime importance in determining a course of action.

FACTORS AFFECTING DECISION TO ATTEMPT AIR START, FORCED LANDING OR EJECTION.

Note

Normally the pilot will be concerned with AIR START, EJECTION, or FORCED LANDING. The basic factors associated with these procedures are presented as a guide to assist the pilot in selecting the best procedure or combination of procedures for his particular circumstances.

Factors Affecting Air Start.

Air start is not possible and should not be attempted following complete engine failure (engine seizure or explosion).

Immediate air starts may be possible at any altitude if RPM has not decreased appreciably, however air start probability increases below 20,000 feet.

Continuous air starts should not normally be attempted above 20,000 feet (to conserve battery for starts below 20,000 feet, or possible forced landing).

225 KIAS (maximum glide speed) is the best speed for air start and should provide an indicated engine RPM between 17 and 21 percent. Gliding flight characteristics are normal and rapid trim changes not necessary.

The starter switch should not be used to increase engine RPM. Damage to the aircraft and starter may result.

Fuel supply must be available for air start.

If a flame-out is caused by an acceleration or deceleration, sustained inverted flight or a similar reason, air start can be made without changing fuel tank selection.

The fuel boost pressure warning lights will not isolate the fuel system failure when engine RPM is below approximately 30 percent (generator cut in speed) as they will all be illuminated regardless of fuel supply or condition of pumps.

The fuel quantity indicator and fuel flow meter and hydraulic pressure gages will operate on the alternate inverter with the battery switch on.

If it is suspected that the main tank is empty, an auxiliary fuel flow must be selected.

Do not turn fuel tank selector to OFF (Pressure build-up may make it difficult to select any other position).

If the fuel system boost pressure warning light is off when a flame-out occurs this is an indication of an engine fuel control failure.

If a flame-out is caused by an engine fuel control failure the emergency fuel control system must be selected.

If below 6,000 feet and RPM has not dropped below 85 percent, the emergency fuel system may be selected immediately.

If above 6,000 feet, or if RPM is below 85 percent, the throttle must be retarded to IDLE, then the emergency fuel system may be selected.

The air start switch opens the primer valve and energizes the ignition timer for 15 (+3) seconds. When successive air start attempts are made, the starter switch should be momentarily positioned to STOP START before actuating the air start switch. (The ignition timer will not recycle even though the air start switch is actuated until its 15 (+3) second period runs out or the circuit is interrupted by STOP START.)

When a flame-out occurs at high altitude it is imperative to determine if the reverse current relay has failed in order to conserve the battery. A failed relay is indicated by any one of the following:

Fuel booster pressure warning lights out.

Voltmeter indicating any value other than zero.

Generator out indicator lights not illuminated.

Main inverter continuing to operate.

If any of the above indications exist, accomplish the following:

a. Battery switch — OFF.

b. Booster pump circuit breakers — PULL OUT.

c. All equipment powered by secondary busses — OFF.

d. ALT INVERTER switch — ON.

The tachometer and EGT gage are independent of the aircraft electrical system. The tachometer will probably give the first indication of a successful air start as the EGT gage has a relative slow response rate.

After an air start is made, momentarily position the starter switch to STOP START to close th primer valve and deenergize the ignition timer. When operating on the emergency fuel system advance the throttle slowly and avoid rapid accelerations. The emergency fuel system does not incorporate provisions for acceleration or overspeed control. Small throttle movements result in large thrust changes. Monitor EGT and RPM at all times. Altitude permitting, EGT may be controlled or limited by diving to increase ram air flow. Complete flight on the emergency fuel system, do not switch back to normal.

Factors Affecting Forced Landing.

Forced landings are dangerous and should be attempted only under ideal conditions, EJECTION IS NORMALLY THE BEST COURSE OF ACTION. The following instructions are presented as a guide.

Forced landings should not be attempted unless the engine is windmilling sufficiently to maintain full power and utility hydraulic pressures unless altitude is too low for ejection or unless the landing can be made in a short period of time. The battery cannot be relied on to last long enough to guarantee emergency hydraulic operation of flight controls.

Forced landings should not be attempted by pilots who are not proficient in simulated forced landing approaches in this aircraft.

Forced landings should not be attempted at night, dusk or dawn, regardless of weather or field lighting.

Forced landings should not be attempted unless ideal weather conditions exist.

Cloud cover, ceiling, visibility, turbulence, surface winds, etc. must not impede in any manner the establishment of proper forced landing pattern.

Forced landings should not be attempted where approaches have obstacles or are over heavily populated areas.

Forced landings should not be attempted on other than a prepared or designed suitable surface of at least 8,000 feet.

Forced landings should be made with the landing gear extended.

Investigation has shown that even landing on reasonably rough terrain the extended gear absorbs the initial impact and results in less pilot injury and aircraft damage.

Air start attempts should be completed before high key is reached so that complete attention may be devoted to accomplishing a successful forced landing pattern. Further air start attempts down to low key may be made, provided that primary attention is devoted to proper execution of the forced landing pattern.

Do not continue attempting air starts after low key is reached as successful completion of forced landing requires complete attention.

This does not preclude air start attempts when flame-out occurs below low key.

Forced landings should not be attempted when a satisfactory "High Key" or "Low Key" cannot be achieved.

Forced landing should not be attempted if at any time during the approach, conditions do not appear ideal for successful completion of the landing. Eject no later than base leg altitude (2,000 feet)

Factors Affecting Ejection.

The following information should be observed when ejection must be accomplished:

UNDER LEVEL FLIGHT CONDITIONS, EJECT AT LEAST 2000 FEET ABOVE THE TERRAIN WHENEVER POSSIBLE.

WARNING

Do not delay ejection below 2000 feet above the terrain in futile attempts to start the engine or for other reasons that may commit you to an unsafe ejection or a dangerous flameout landing. Accident statistics emphatically show a progressive decrease in successful ejections as altitude decreases below 2000 feet above the terrain.

UNDER SPIN OR DIVE CONDITIONS, EJECT AT LEAST 10,000 FEET ABOVE THE TERRAIN WHENEVER POSSIBLE.

Ejection should not be delayed when the aircraft is in a descending attitude and cannot be leveled out.

Attempt to slow the airplane as much as practical prior to ejection by trading airspeed for altitude.

If the airplane is not controllable, ejection must be accomplished at whatever speed exists, as this offers the only opportunity for survival. At sea level wind blast will exert minor forces on the body up to about 525 knots ICAS, appreciable forces from about 525 to 600 knots IAS and excessive forces above about 600 knots IAS. As altitude is increased these speed ranges will be proportionately lower.

The automatic safety belt must not be opened manually before ejection, regardless of altitude. If the automatic seat belt is opened manually, the automatic opening feature of the parachute is eliminated and seat separation may be too rapid at high speeds.

Note

Improper routing of personal leads may cause inadvertent opening of the lap belt latch during ejection. Care must be taken to insure that flight clothing, such as sleeves, will not catch and release the lap belt during ejection.

Low Altitude Ejection

During any low altitude ejection, the chances for successful ejection can be greatly increased by zooming the airplane (if airspeed permits) to exchange airspeed for altitude. Ejection should be accomplished while the airplane is in a positive climb. This will result in a more nearly vertical trajectory for the seat and crew member thus providing more altitude and time for seat separation and parachute deployment.

Emergency minimum altitudes for ejection are:

MODIFIED
[801] complied with (rocket catapult and seat-man separator) F-1B parachute timer set at one second.
Airspeed 100 to 300 KIAS, Level Flight.

PARACHUTE	*ZERO DELAY LANYARD*	*ALTITUDE*
B-5 Pack, C-9 Canopy	Connected	0 FEET

UNMODIFIED
[801] not complied with (explosive catapult) no seat-man separator) F-1B parachute timer set at one second.
Airspeed 140 to 300 KIAS, Level Flight.

PARACHUTE	*ZERO DELAY LANYARD*	*ALTITUDE*
B-4 or B-5 Pack C-9 Canopy	Connected	100 FEET
	Stowed	200 FEET
B-5 Pack C-11 Canopy	Connected	150 FEET
	Stowed	200 FEET

WARNING

The above altitudes were determined through extensive sled tests and are based on altitude above terrain on initiation of seat ejection (i.e., time seat is fired). These figures do not provide any safety factor for such matters as equipment malfunction, delays in separating from the seat, etc. These figures are quoted only to show the minimum altitude you must go up to in the event of low altitude emergencies such as a fire on takeoff. They shall not be used as the basis for delaying ejection when above 2,000 feet since accident statistics show progressive decrease in successful ejection as altitude decreases below 2,000 feet. Therefore, whenever possible, eject above 2,000 feet.

After Ejection.

Immediately after ejection, attempt to manually open the seat belt as a precaution against the belt failing to open automatically.

As soon as the belt releases, a determined effort must be made to separate from the seat to obtain full parachute deployment at maximum terrain clearances. This is extremely important for low altitude ejections. (For airplanes with man-seat separators this is a precautionary measure in the event the separator fails to function).

If the seat belt has been opened manually during ejections above 14,000 feet, immediately pull parachute arming lanyard to permit automatic opening of the parachute at the preset altitude.

ENGINE FAILURE DURING TAKEOFF.

If Not Committed.

1. **ABORT.**

   Refer to Abort/Barrier engagement procedure.

If Committed.

1. **Emergency fuel system — Select.**

   Emergency on indicator light should illuminate.
2. **Air start button — Actuate** (Every 15 seconds, starter switch — STOP START, air start button — Actuate.)
3. **External stores — Jettison**, if necessary.
4. **Zoom-Up.**
5. Fuel tank selector — as desired.
6. If thrust is regained — land as soon as possible using precautionary landing.

   If thrust is not regained — EJECT or make forced landing.

CAUTION

When operating on the emergency fuel system advance the throttle slowly and avoid rapid accelerations. The emergency fuel system does not incorporate provisions for acceleration or overspeed control. Small throttle movements result in large thrust changes. Monitor EGT and RPM at all times. Altitude permitting, EGT may be controlled or limited by diving to increase ram air flow. Complete flight on the emergency fuel system, do not switch back to normal.

Forced Landing Immediately After Becoming Airborne.

1. **Landing gear selector — EMERGENCY DOWN.**
2. **External stores — Jettison.**
3. **Emergency hydraulic pump — ON.**
4. **Fuel tank selector — OFF.**
5. Shoulder harness — Lock.
6. Land straight ahead (change course only to miss obstacles).
7. Drag chute — Deploy after touchdown.
8. Battery and generator switches — OFF on ground contact.

ENGINE FAILURE AT LOW ALTITUDE.

1. Zoom-up and make decision to air start, force land or eject.

   See zoom-up maneuver in this section.

ENGINE FAILURE DURING FLIGHT.

1. **Throttle — CLOSED.**
2. Landing gear — UP, wing flaps — UP, speed brakes — IN.
3. Unnecessary electrical equipment — OFF.
4. Establish glide — 225 KIAS.

   225 KIAS is maximum glide and best air start speed.
5. ALT INVERTER switch — ON.
6. Make decision to air start, make forced landing or eject.

   Refer to factors affecting decision to attempt air start, forced landing or ejection.

AIR START.

Note

Factors affecting the pilot's decision to AIR START are covered under ENGINE FAILURE in this section.

For an air start accomplish as many of the following steps as necessary.

1. Fuel tank selector — As required.

   If fuel control failure is suspected, emergency fuel system — Select.

   The emergency on indicator light should illuminate.
2. **Air start button — Actuate.**

   The primer valve will open and ignition timer will be energized for 15 (±3) seconds.
3. **Throttle — Immediately move to IDLE or above, but not to exceed ⅓ open.**

   Do not delay opening the throttle. The ignition timer will already be operating, but fuel will not be available with throttle closed.
4. Monitor tachometer, EGT gage and fuel flow indicator for indications of start.
5. If start is unsuccessful, emergency fuel system — Select.

   The emergency on indicator light should illuminate.
6. Continue attempting air start. Every 15 seconds starter switch — STOP START, air start button — Depress.
7. When successful air start is made, starter switch — STOP START.

   The STOP START position closes the primer valve.

CAUTION

When operating on the emergency fuel system, advance the throttle slowly and avoid rapid accelerations. The emergency fuel system does not incorporate provisions for acceleration or overspeed control. Small throttle movements result in large thrust changes. Monitor EGT and RPM at all times. Altitude permitting, EGT may be controlled or limited by diving to increase ram air flow. Complete flight on the emergency fuel system, do not switch back to normal.

THRUST LOSS OR ENGINE SURGING.

Thrust loss or engine surging may be caused by failure of the fuel control or icing of the fuel control inlet pressure sensing probe in the number 2 engine strut (forward of the compressor blades). For the above condition proceed as follows:

1. **If above 6,000 feet, or if RPM is below 85 percent, throttle — IDLE.**

   If below 6,000 feet, or if RPM has not dropped below 85 percent, the throttle need not be retarded to select the emergency fuel system.
2. **Emergency fuel system — Select.**
3. Fuel filter de-ice-switch — MANUAL (if installed and operative).
4. Throttle — Adjust as required and complete flight on emergency system.

CAUTION

When operating on the emergency fuel system, advance the throttle slowly and avoid rapid accelerations.

5. Fuel tank selector — As required.

   Assure that fuel supply is available to engine.
6. Land as soon as possible.

OIL SYSTEM FAILURE.

LOSS OF ENGINE OIL PRESSURE.

If an oil system malfunction (as evidenced by high or low oil pressure or excessively low oil quantity) has caused prolonged oil starvation of engine bearings, the result will be a progressive bearing failure and subsequent engine seizure. This progression of bearing failure starts slowly and will normally continue at a slow rate up to a certain point at which the progression of failure accelerates rapidly to complete bearing failure. 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. A good possibility exists that the engine may operate for 30 minutes after experiencing a complete loss of lubricating oil. 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 of a successful ejection or power-off landing by shutting down the engine. Since the end result of oil starvation is engine seizure, the following procedures should be observed in an attempt to forestall engine seizure as long as possible.

AT FIRST INDICATION OF OIL SYSTEM MALFUNCTION:

1. Thrust — Maintain constant thrust between 85 and 93 percent.

   High thrust settings should be maintained. Upon detection of an oil system malfunction (as evidenced by the oil pressure gage indication) a constant thrust setting between 85 and 93 percent 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 (throttle movement should be minimized). 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 — Jettison if desired.
3. G forces — Minimize. Avoid all abrupt maneuvers causing high G forces.
4. Land as soon as possible, using precautionary landing in the event a complete engine failure occurs.

MAXIMUM GLIDE DISTANCE AND RATE OF SINK

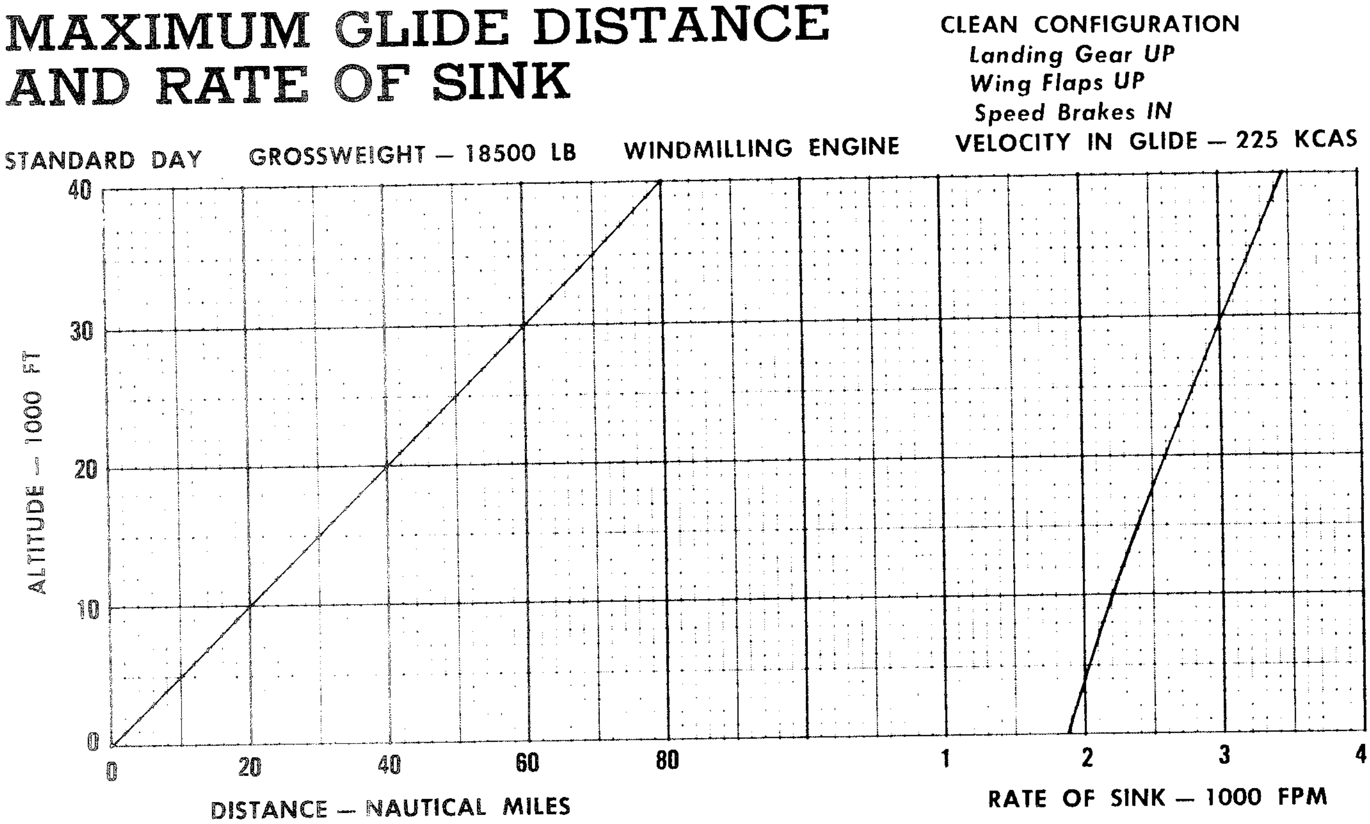


Figure 3—1.

MAXIMUM GLIDE WITH DEAD ENGINE.

For maximum glide distance, trim aircraft to maintain recommended glide speed with gear and flaps UP, speed brakes IN. (See figure 3—1.)

FORCED LANDING.

Note

> Forced landings are dangerous and should be attempted only under ideal conditions, EJECTION IS NORMALLY THE BEST COURSE OF ACTION. Factors effecting the pilot's decision to EJECT or attempt a FORCED LANDING are presented under "Engine Failure" in this section.

The forced landing pattern shown in figure 3—2 is ideal in that the assumption is made that sufficient altitude is available to glide to high key altitude. The ideal high key point is 8,000 feet (6,000 to 10,000 feet recommended) above terrain to one side of the approach end of the runway. The glide to high key is accomplished with aircraft clean (i.e., landing gear and wing flaps up, speed brakes IN and external stores jettisoned). The turn from high key should be initiated when the aircraft crosses high key (abreast the approach end of the runway). The landing configuration (gear down and flaps as desired, speed brakes in) is established at high key provided altitude is sufficient to permit the increased rate of descent. The earlier this configuration is established in the pattern, the more it will permit adjusting the pattern while there is still altitude to work with. It will avoid trim change and sudden increase in rate of descent and distraction from the touchdown point during the more critical phase of the pattern. Appropriate speed should be maintained throughout the pattern and position and altitude regulated by bank angle. A helpful technique is to check that 25 percent of the altitude is lost during each 90 degrees of the pattern after the gear is extended. This technique requires that if a 90 degree check point is reached at too high an altitude, the bank is shallowed out. If the altitude is too low, the turn is tightened to reach the next 90 degrees check point at the desired altitude. Excessive bank angles increase rate of descent and should be avoided. The preceding comments and procedure apply to the situation where sufficient altitude is available to permit attaining high key. If the decision is made to land when high key cannot be attained, pick up the pattern at the highest altitude possible, but no lower than low key. The decision for establishing the landing configuration would then be determined by how well you are doing in establishing the prescribed pattern.

FORCED LANDING

To Achieve High Key

1. Throttle—*CLOSED.*
2. Landing Gear—*UP*, Wing Flaps—*UP*, Speed Brakes—*IN.*
3. External Stores—*Jettisoned.*
4. Airspeed—*225 KIAS.*
5. Fuel Tank Selector—*OFF.*
6. Pneumatic Compressor Switch—*OFF.*
7. Shoulder Harness—*LOCK.*

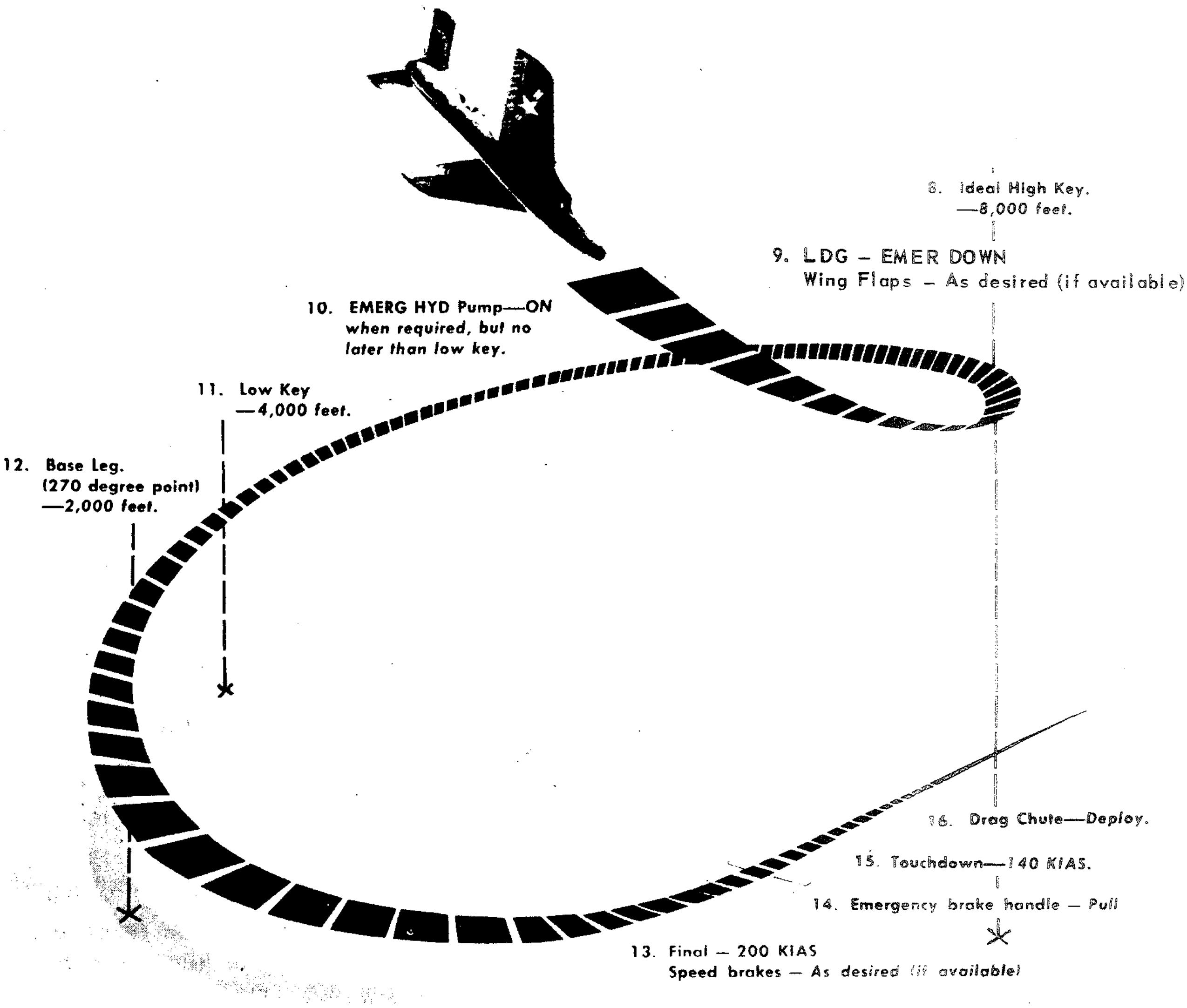


Figure 3–2

FORCED LANDING PROCEDURE.

1. Throttle — CLOSED.
2. Landing gear — UP, wing flaps — UP, speed brakes — IN.
3. External stores — Jettisoned.
4. Airspeed — 225 KIAS.
5. Fuel tank selector — OFF.
6. Pneumatic compressor switch — OFF.
7. Shoulder harness — LOCK.
8. Ideal high key — 8,000 feet (6,000 to 10,000 feet recommended).
9. Landing gear (at high key) — EMERGENCY DOWN, wing flaps — As desired (if available). Establish landing configuration provided pattern is assured and altitude is satisfactory.

   Establishing the landing configuration will increase the rate of descent from approximately 2,500 feet per minute to approximately 4,000 feet per minute. Landing gear extension requires approximately 7 to 8 seconds. Full wing flap extension requires approximately 14 seconds.

10. Emergency hydraulic pump switch — ON when required, but no later than low key.

    Do not turn pump on when a demand is on the system (to avoid possible hydraulic surge accompanied by erratic control system function).

11. Low key — 4,000 feet (3,000 to 5,000 recommended).
12. Base leg (270 degree point) — 2,000 feet (1,500 to 2,500 feet recommended).
13. Airspeed on final — 200 KIAS. Speed brakes as desired (if available).
14. Emergency brake handle — Pull.
15. Airspeed on touchdown — 140 KIAS.
16. After touchdown, drag chute — Deploy.

PRECAUTIONARY LANDING.

a. When engine failure is a possibility, a precautionary landing pattern should be made. The thrust-drag configuration shown in the chart below should be used. The precautionary landing pattern should be intercepted between 6000 and 10,000 feet AGL. In the event of actual engine failure, the speed brakes should be retracted immediately and a forced landing made.

b. When engine failure is not anticipated, a normal landing should be made, except where specific emergency procedures apply. However, for failures such as airframe structure or flight control malfunctions, a straight-in approach may be made.

Wing flaps — 68 percent Speed brakes — OUT	
ALTITUDE (feet)	*THRUST (percent RPM)*
above 15,000	88
15,000	83
10,000	83
HIGH KEY	83
LOW KEY	82
SEA LEVEL	82

With the above configuration and thrust settings, the actual altitudes and air speeds of the forced landing pattern shown in figure 3—2 may be flown. The landing gear should be extended as required by the pattern.

Note

When initiating a go-around from simulating a forced landing pattern, allow sufficient time for engine acceleration to MILITARY THRUST.

FIRE.

Illumination of the overheat warning light indicates an overheat condition; or possible fire in the aft section. Illumination of the fire warning light indicates a fire in the forward engine section. Some false warnings have occurred; therefore, it is preferable, under unfavorable flight conditions, to take precautionary measures before completely shutting down the engine whenever a fire warning or overheat light appears. In checking for other indications of fire, the following should be considered.

Excessive RPM, EGT, or fuel flow.

Smoke in cockpit.

Engine noise.

Verification from another aircraft.

FIRE WHILE STARTING THE ENGINE.

1. **Throttle — CLOSED.**
2. **Fuel tank selector — OFF.**
3. **Battery — OFF** and disconnect external power.
4. Evacuate cockpit.

WARNING

Use extreme care during evacuation of cockpit to avoid possible actuation of the ejection seat.

FIRE OR OVERHEAT WARNING LIGHT.

The exact procedure to follow is dependent on each set of circumstances and depends on altitude, airspeed, length of runway and overrun remaining, location of populated areas, etc. As a guide in making a decision, the following procedures are recommended:

ON TAKEOFF.

If Not Committed.

1. **Abort.**
   See abort barrier engagement in this section.

If Committed.

1. **External stores — Jettison.**
2. **Maintain MAX THRUST** and **Zoom-up.**
3. **Throttle — Retard, Check for fire.**
   Note EGT and reduce to within limits.
4. **If fire exists, Eject,** or make forced landing.
5. If no fire exists — Land as soon as possible.
   Use minimum thrust and throttle movement.

DURING FLIGHT.

1. **Throttle — IDLE, check for fire.**
   Note EGT and reduce to within limits.
2. **If fire exists.**
   a. **Throttle — OFF.**
   b. **Fuel Tank selector — OFF.**
   c. **Eject** or make forced landing.
3. If no fire exists — Land as soon as possible.
   Use minimum thrust and throttle movement.

ELECTRICAL FIRE.

Circuit breakers and fuses protect most of the electrical circuits and tend to isolate overloads that would possibly result in a fire. If, however, a circuit breaker or fuse fails to effect a disconnect of an overloaded circuit, the wire will overheat causing the insulation to smoke until the wire separates. If such a circuit is in the cockpit, it will be evident by smoke and or fumes. Refer to the procedure for Smoke Elimination in this section.

SMOKE ELIMINATION.

If smoke or fumes enter the cockpit, the cause may be a fire, a fuel leak in the engine compartment, a failed refrigeration turbine or a failed bearing seal.

1. Oxygen regulator diluter lever — 100% OXYGEN.
2. If refrigeration turbine failure is suspected.
   a. Cabin temperature control — position to highest point.
3. If electrical fire is suspected.
   a. All nonessential electrical equipment — OFF.
   b. Both inverters — OFF.
   c. Both generators — OFF.
   d. If smoke persists, battery switch — OFF.

Note

A generator and the main inverter must be ON for IFF operations. Battery (or emergency battery if installed) must be on for command radio operation.

4. If smoke persists or becomes more intense.
   a. Cabin vent selector — RAM.
   The intake distribution valve will close and the ram air and dump valves will open. The intake of air through the ram air valve will scavenge the air in the cockpit through the open dump valve, and rapidly dissipate the fumes and smoke.
   b. If necessary, canopy-Jettison.
   Jettison canopy without arming seat if possible. See canopy jettison in this section.

HOT COCKPIT PROCEDURE.

If cockpit temperature becomes intolerable, accomplish as many of the following steps as necessary.

1. a. Cabin vent switch — RAM 35 and EARLIER.
   b. Cabin temperature control — RAM 40 and LATER.
2. Defroster control — OFF.
3. Cabin temperature control switch — Position HEAT then COOL RAM 35 and EARLIER.
4. If heat becomes too intense, Canopy-Jettison.

   Jettison canopy without arming seat if possible. See canopy pettison in this section.

TAKEOFF AND LANDING EMERGENCIES.

An extensive study of injuries and damages resulting from emergency landings on unprepared surfaces has revealed that a greater injury hazard is presented whenever emergency landings are made with the wheels retracted, regardless of terrain. Increased air-speed or nose-high angle at impact during landings with the wheels retracted is common practice and contributes greatly to the severity of pilot injury and aircraft damage. This nose-high attitude causes the aircraft to literally "slap" the ground on impact, subjecting the pilot to possible spinal injury. Less injury and less aircraft damage resulted when the landings were made with the gear extended because of the absorption of initial shock by the extended landing gear. There were no cases recorded where the aircraft flipped or tumbled end-over-end when an emergency landing was attempted with the gear extended. Emergency landings on unprepared surfaces shall be made with the landing gear extended. This also applies to landing short on the runway, or when it is impossible to stop on the remaining runway during either takeoff or landing. This is not to be construed as prohibiting pilots from landing on empty external tanks on prepared surfaces under certain conditions in order to minimize damage to the aircraft. Refer to abort in this section for additional information on runway overrun.

Note

For certain landing emergencies it is recommended that foam be applied to the runway starting 1,000 feet from the approach end and laid as far as possible depending on amount of time and foam available.

LANDING WITH WHEELS RETRACTED.

In the event it is impossible to get landing gear down, the following procedure is recommended:

1. Jettison all stores except pylons and empty pylon fuel tanks.

   With pylons and empty fuel tanks, or pylons alone, the aircraft will be subject to less damage. If time is critical, jettison all stores.
2. Shoulder harness — LOCK, seat belt — Tighten.
3. Make normal approach.

   Use wing flaps and speed brakes.
4. Throttle — OFF on touchdown.
5. Drag chute — Deploy.
6. Battery and generator switches — OFF.
7. Evacuate cockpit.

WARNING

Use extreme care during evacuation of cockpit to avoid possible actuation of the ejection seat.

LANDING WITH MAIN GEARS DOWN AND NOSE GEAR UP OR UNLOCKED.

If the nose gear cannot be extended or fails to indicate down and locked proceed as follows:

Note

Illumination of the taxi light (observed by mobile control) may be used as an indication that the nose gear is down and locked.

1. Jettison all stores except pylons and empty pylon fuel tanks.

   With pylons and empty fuel tanks, or pylons alone, the aircraft will be subject to less damage. If time is critical, jettison all stores.
2. Shoulder harness—LOCK, seat belt—Tighten.
3. Landing gear handle—EMERGENCY DOWN.
4. Emergency brake handle—PULL.
5. Make normal approach with touchdown near end of runway. Use wing flaps and speed-brakes.
6. Throttle—Closed, Fuel Tank Selector—OFF at touchdown.
7. Lower nose gently to runway slightly above 100 KIAS.
8. Avoid using brakes, unless necessary, except for directional control.
9. Battery and generator switches—OFF.
10. STOP aircraft straight ahead.

WARNING

Use extreme care during evacuation of cockpit to avoid possible actuation of the ejection seat.

LANDING WITH MAIN GEAR UP OR UNLOCKED.

1. Jettison all stores except pylons and empty pylon fuel tanks.

   With pylons and empty fuel tanks, or pylons alone the aircraft will be subject to less damage. If time is critical, jettison all stores.
2. Shoulder harness – LOCK, safety belt – Tighten.
3. Make normal approach.
4. On touchdown, Throttle – CLOSED, fuel tank selector – OFF.
5. Drag chute – Deploy.
6. Battery and generator switches – OFF.
7. Evacuate cockpit.

WARNING

Use extreme care during evacuation of cockpit to avoid possible actuation of the ejection seat.

MAIN/NOSE GEAR TIRE FAILURE ON TAKEOFF.

If tire failure occurs at or near rotation (nose wheel lift off) speed, the pilot may elect to continue takeoff, and accept landing with a flat tire. Controlling the aircraft with a flat nose gear tire is much easier at light gross weight. In the event the pilot elects to abort, proceed as follows:

1. USE ABORT/BARRIER ENGAGEMENT PROCEDURE.

CAUTION

A strong possibility exists that brake failure may occur on the side with the blown tire. Fragments of the disintegrating tire may sever the brake hydraulic line, making brake pressure to that wheel unavailable.

Caution must be exercised when braking the wheel opposite a blown tire. As the aircraft decelerates, the possibility becomes progressively stronger of entering a sideways slide which will impose side loads on the main landing gear. Experience has shown that on properly prepared runway shoulders, less damage will occur when leaving the runway straight ahead than will occur when asymmetrical braking causes the aircraft to slide sideways off the runway.

LANDING WITH FLAT TIRE.

1. Use up excess fuel.
2. Make normal landing.
   a. For flat nose wheel tire, hold nose wheel off as long as possible.
   b. For one flat main tire, land on side of runway nearest inflated tire.
   c. For both flat main tires, land in center of runway, use brakes sparingly and with caution.
3. Drag chute – Deploy after touchdown.

LANDING WITH UTILITY HYDRAULIC SYSTEM FAILURE.

1. External stores – Jettison if barrier engagement is anticipated.
2. Landing gear handle – EMERGENCY DOWN.

   Hold handle in EMERGENCY DOWN until all gears indicate down and unlocked.
3. Use long flat final approach to minimize control surface deflections.

   Refer to Appendix for approach and touchdown speeds.
4. Emergency hydraulic pump switch – ON.
5. Emergency brake handle – Pull.
6. On ground contact.
   a. Throttle – CLOSED.
   b. Drag chute – Deploy.
   c. Brakes – As required.

      Approximately twice the toe pressure will be required for the brakes to be as effective as when utility pressure is available.

LANDING WITH UNBALANCED EXTERNAL LOAD.

Landing with an unbalanced inboard pylon configuration can be accomplished without any special technique, provided hydraulic power is available for aileron control. However, landing with an unbalanced outboard pylon configuration should only be accomplished if the pilot is sufficiently familiar with the aircraft and all other landing factors are normal. With hydraulic pressure available, there is sufficient lateral control to maintain level flight for the critical asymmetric outboard configuration of one full 230 gallon tank if the landing is made at a minimum speed of 180 knots. If the pilot has any doubt about making the landing it is recommended that the outboard stores be jettisoned.

LANDING AFTER THROTTLE LINKAGE FAILURE.

On aircraft equipped with J65-7D engines, failure of the linkage between cockpit throttle and engine fuel control will result in the fuel control lever automatically seeking the throttle position which gives 90 percent engine RPM. The following procedure is recommended if throttle linkage failure occurs: Continue flight to a suitable landing area (at high altitudes and heavy gross weights descent to a lower altitude may be necessary in order to maintain level flight). Make a normal penetration from 20,000 feet (all aircraft configurations) at 325 KIAS with speed brakes out. Level off 1,000 feet above initial approach altitude and lower 1/2 flaps at 250 KIAS. An airspeed of 220 to 230 KIAS should maintain level flight. Enter downwind leg with speed brakes OUT, half flaps, and 220 to 230 KIAS. Enter base leg approximately 12 miles from touchdown point, lower landing gear, retract speed brakes and maintain 220 KIAS. After turning final, extend full flaps and maintain 195 to 200 KIAS. When 8 to 10 miles from touchdown, use speed brakes as necessary to establish a 500 to 600 FPM rate of descent at 185 to 195 KIAS. Before touchdown, after landing is assured, turn Fuel Tank Selector Switch OFF.

WARNING

When the fuel tank selector switch is turned OFF, the effective thrust decreases rapidly and is not comparable with thrust reduction by throttle action. Use emergency braking procedures. After touchdown, deploy drag chute if available. Dragging the tail will shorten the roll if drag chute is not available. If barrier is available, aim for its center. Lock shoulder harness immediately prior to engaging barrier or leaving runway and turn Battery and Generator Switch OFF.

WARNING

Should throttle linkage separation require an aborted take-off, turn Fuel Tank Selector Switch OFF and follow Abort/Barrier Engagement Procedure.

EJECTION PROCEDURE

EJECTION ALTITUDES

DIVE OR SPIN CONDITION–Eject at least 10,000 feet above terrain.

LEVEL FLIGHT–Eject at least 2,000 feet above terrain.

EMERGENCY MINIMUM ALTITUDES FOR EJECTION-Refer to Factors Affecting Ejection in this section.

ZERO DELAY LANYARD ENGAGEMENT REQUIREMENTS

The zero delay parachute lanyard will be connected and disconnected as follows:

a. Connect prior to take-off.

b. Leave connected at all times below 10,000 feet pressure altitude including flights in which 10,000 feet may be temporarily exceeded.

c. Disconnect when passing through 10,000 feet pressure altitude when this altitude will be exceeded for prolonged periods.

Note

In operating above terrain over 8000 feet high, the zero delay lanyard should remain connected until the aircraft is at least 2000 feet above the terrain and should be connected at least 2000 feet above the terrain on descent.

d. Connect prior to initial penetration, or at 10,000 feet pressure altitude during enroute descent.

BEFORE EJECTION

if circumstances permit

- *Head aircraft away from populated area.*
- *Zero delay lanyard–Connected as required.*
- *Loose equipment stow and pull visor down (if applicable).*
- *At high altitude; bailout bottle ball handle–pu Pull.*
- *Throttle–OFF*

WARNING

- ***Do not manually open the automatic safety belt***
- ***Reduce speed to below 500 KIAS***

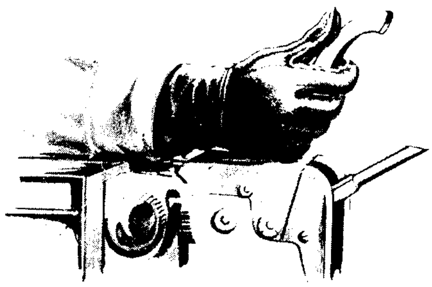
LOW ALTITUDE EJECTION

Zoom up; pull aircraft into a straight climb with wings level. Eject prior to reaching a stall or sink while nose of aircraft is above horizon.

EJECTION

1 RAISE ARMRESTS (Battery Switch ON)

Raise both handgrips to jettison canopy and lock shoulder harness.



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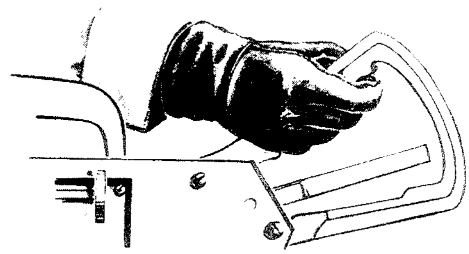


30 and LATER

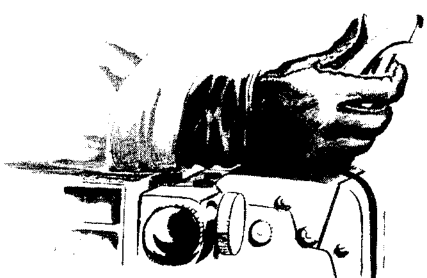
- **IF CANOPY FAILS TO JETTISON,** Canopy Jettison Switch—**ON** (if installed) unlock canopy and raise into airstream.
- As a last resort, eject through canopy if it is closed, and preferably locked. If canopy is open and bow in path of ejection, **DO NOT EJECT,** make a manual bail-out.

● ASSUME EJECTION POSITION

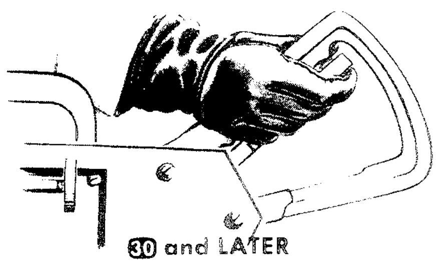
Sit erect with head hard against headrest, chin tucked in, arms on armrests, and feet in footrests.

2 SQUEEZE TRIGGER

Squeeze ejection trigger(s) (Right hand 5 thru 25, Both hands 30 and LATER)



5 thru 25



30 and LATER

IF SEAT FAILS TO EJECT — ***Make a manual bail-out***

AFTER EJECTION

1. Immediately after ejection — Attempt to manually open the safety belt. It will be impossible to beat the automatic operation unless it has failed.
2. As soon as the belt is open — Make a determined effort to separate from the seat to obtain full parachute deployment at maximum terrain clearance. This is extremely important for low altitude ejection.
3. If you opened the safety belt manually, the automatic feature of the parachute is eliminated therefore;
   a. If below 14,000 feet — Pull the ripcord grip.
   b. If above 14,000 feet — Pull the arming lanyard.
4. All ejections below 14,000 feet — Pull the ripcord grip. This is a precaution since the parachute should deploy automatically.

EMERGENCY ENTRANCE

TO OPEN CANOPY

a. Depress forward end of exterior canopy control.
b. Pull aft end of exterior canopy control forward.
c. Lift canopy.

IF CANOPY CANNOT BE OPENED, it should be jettisoned by the pilot. STAND CLEAR OF TURTLE DECK AREA.

WARNING

Ground crews must remain alert to the possibility of the pilot jettisoning the canopy.

If pilot jettisons canopy the seat will be armed for ejection and the ejection trigger(s) exposed.

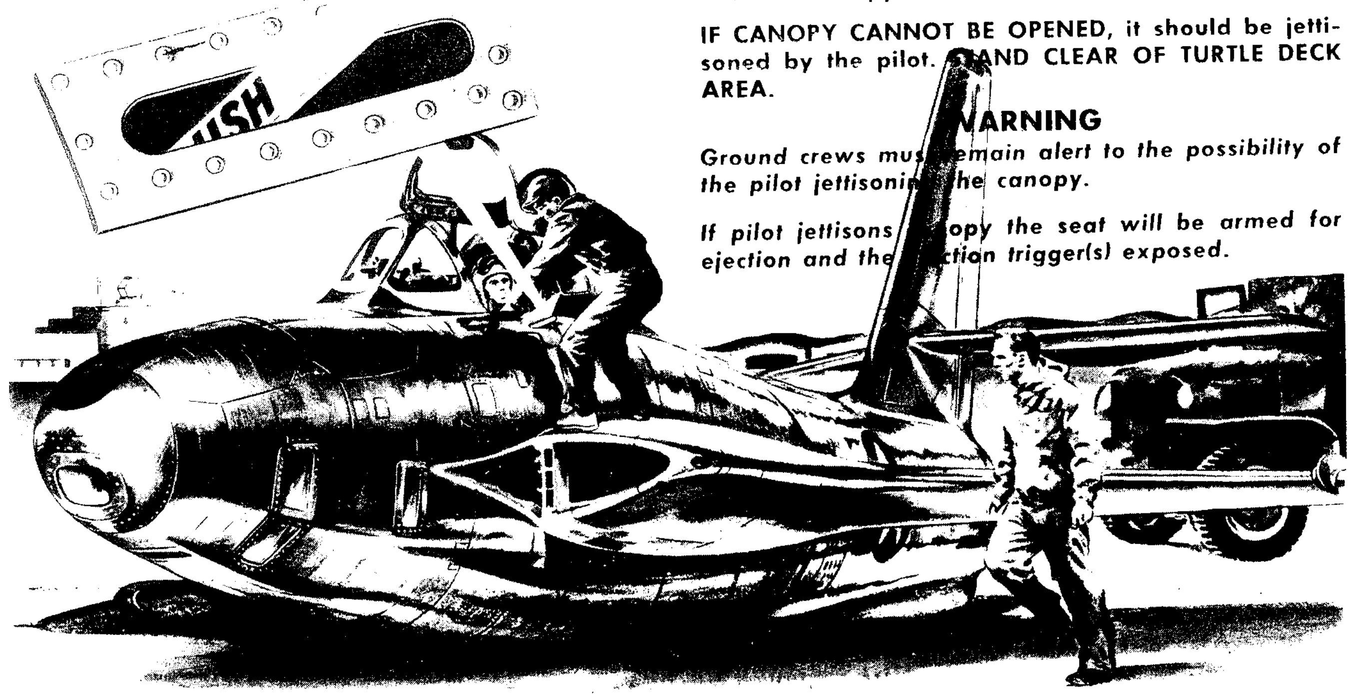


Figure 3—4.

EJECTION.

Note

Factors affecting the pilot's decision to EJECT are covered under ENGINE FAILURE in this section.

See figure 3—3 for ejection procedure.

MANUAL BAILOUT PROCEDURE.

In the event the seat fails to eject when the ejection trigger (both triggers on 30 and later) are squeezed, proceed as follows:

1. Safety belt — Release.
2. Bail out bottle — Actuate.
3. Personal equipment leads — Disconnect.
4. Trim — Nose down.
5. Invert aircraft, release stick and push free.
   Keep positive G load until inverted, then quickly release stick and push free.

Note

If aircraft is not controllable, slow down as much as possible and bailout over the side.

6. a. For bailout above 14,000 feet — Delay pulling rip cord grip until 14,000 feet is reached.
   b. For bailout at low altitude — Pull rip cord grip as soon as clear of aircraft.

EMERGENCY ENTRANCE.

See figure 3—4.

DITCHING.

Bailout is preferred to ditching. High speed aircraft tend to skip on contact, therefore, the pilot must continue to fly the aircraft after touchdown until aircraft settles. If a regular wave or swell pattern exists aim the touchdown parallel to the waves and attempt to land on the crest or on the falling side of the wave. More often, the sea surface will be irregular with two or more wave or swell patterns intermingled. In this case, the best compromise is to head into whatever wind may be blowing. Examine the sea to find areas where the intermingling waves cancel out. Aim touchdown for one of these calmer areas. Make touchdown in a nose high attitude in the same attitude as for a normal landing. Make the softest possible landing. Do not stall the aircraft at the time of contact. In the event of low altitude or any other reason where bailout is impracticable, ditch the aircraft as follows:

1. IFF SIF – EMERGENCY.
2. Oxygen – 100% OXYGEN.
3. External stores – Jettison (all).
4. Wing flaps – 50 to 75 percent.
5. Shoulder harness – LOCK, seat belt – Tighten.
6. Unbuckle parachute and disconnect anti-G and all communication and electrical connections.
7. Helmet visor – Down.
8. Canopy – Jettison.
   a. Canopy manual jettison switch – actuate (If installed).
   b. Canopy – Unlock manually and raise into slipstream.

WARNING

If it is necessary to jettison the canopy by use of the jettison control (seat handle) be careful not to squeeze the seat ejection control accidentally.

9. Throttle – OFF at normal touchdown attitude.
10. Release safety belt, disconnect oxygen and evacuate cockpit.

FUEL SYSTEM FAILURE.

Aircraft fuel system failure is indicated by premature illumination of the wing or forward pump pressure warning lights or the fuel system booster pressure warning light. The only indication of an engine fuel system component failure will be a loss of RPM, EGT, fuel flow or flame-out. (See figure 3–5.)

EXTERNAL TANKS FAIL TO FEED, OR STOP FEEDING.

In the event external tanks fail to feed, or stop feeding, accomplish as many of the following steps as necessary.

1. Pylon tank air pressure switches – Cycle and reposition correctly.
2. Receiver door – Cycle and close (5 thru 35 not modified by [557]).
3. AIR REFUEL CONT circuit breaker – Check IN.
4. Fuel shut-off valves – Cycle and position to OPEN (up).
5. Execute a series of zooms and dives.
6. Descend to lower altitude.

MAIN TANK BOOSTER PUMP FAILURE.

Main tank booster pump failure is indicated by illumination of the fuel system booster pressure warning light (generator out indicator light not illuminated) when operating with the fuel tank selector in the ALL TANKS position. If a main tank booster pump failure is accompanied by power loss (RPM drop or excessive RPM fluctuation), proceed as follows to recover all fuel.

CAUTION

Engine operation with the fuel system booster pressure warning light illuminated, especially at the higher altitudes should be continued with caution to prevent loss of thrust. Aerobatics or rapid changes in altitude should be avoided.

Note

Main tank booster pump failure will not cause an RPM drop under ordinary conditions below 6,000 feet altitude, as the fuel in the main tank can be recovered by direct suction of the engine-driven booster pump. Engine operation up to maximum range power setting, with an inoperative main booster pump and a maximum fuel temperature of 26.6 C (80 F), will be satisfactory up to 20,000 feet when operating on JP-4 fuel under the most severe conditions. However, satisfactory engine operation may be maintained above these altitudes depending on the fuel type and fuel temperature.

Note

In order to prevent fuel transfer from the remaining tanks to the main tank, place the fuel shutoff valve switch placarded MAIN in the CLOSED position.

1. Fuel tank selector – FWD AUX.

CAUTION

When operating with the fuel tank selector in WING AUX position, avoid uncoordinated maneuvers, steep descents or rapid maneuvers since these maneuvers may uncover the wing tank fuel outlets thereby causing a flame-out.

2. When wing fuel pump pressure warning light illuminates, fuel tank selector – ALL TANKS for remainder of flight.

CAUTION

With 700 lbs of fuel or less remaining in the main tank, sustained uncoordinated or climbing turns, accelerations, and nose high attitudes can result in a flame-out. A properly sealed main tank pump compartment will contain sufficient fuel to insure satisfactory engine operation, for the following periods, when performing any of the above maneuvers.

Main Tank Fuel Quantity	*Operating Time*
700 pounds	1.0 minute
400 pounds	0.5 minute

WING TANK BOOSTER PUMP FAILURE.

Premature illumination of the wing pump pressure warning light indicates failure of one, or both, wing tank booster pumps. Fuel in the wing tank with an inoperative booster pump will not be transferred to the main tank. The engine can be operated at the maximum range power setting with the fuel tank selector in WING AUX and without wing tank booster pump pressure up to an altitude of approximately 20,000 feet with JP-4 fuel. Fuel remaining in one wing tank cannot be recovered if the other wing tank is empty, as the engine-driven booster pump will suck air from the empty tank. If a wing booster pump failure is suspected, proceed as follows to recover the most fuel.

1. Both left and right wing fuel pump circuit breakers – Pull out.

   Operating pump will stop and tank will not empty.

2. Left wing and right wing fuel shutoff valve switches – CLOSE.
3. Plan remainder of flight so as to land as soon as possible.
4. Fuel tank selector – ALL TANKS at optimum altitude until the main tank low level warning light illuminates, or if range is critical, until the fuel quantity indicator shows the main tank to be almost empty.
5. Descend to an altitude below 20,000 feet with JP-4 fuel and turn fuel tank selector to WING AUX.

CAUTION

When operating with the fuel tank selector in the WING AUX position, avoid uncoordinated maneuvers, steep descents or rapid maneuvers since these maneuvers may uncover the wing tank fuel outlets thereby causing a flame-out. Should the booster pressure light come on, pull both left and right wing tank booster pump circuit breakers.

FORWARD TANK BOOSTER PUMP FAILURE.

Premature illumination of the forward pump pressure warning light indicates failure of the forward tank booster pump. Fuel in the forward tank can be recovered up to an altitude of approximately 20,000 feet with JP-4 fuel by operating with the fuel tank selector in the FWD AUX position. In event of a forward tank booster pump failure, proceed as follows to recover all fuel.

1. Fuel tank selector – ALL TANKS at optimum altitude until the main tank low level warning light illuminates, or if range is critical, until the fuel quantity indicator shows the main tank to be almost empty.
2. Plan remainder of flight so as to land as soon as possible.
3. Descend to an altitude below 20,000 feet with JP-4 fuel and turn fuel tank selector to FWD AUX.

CAUTION

When turning fuel tank selector from ALL TANKS to FWD AUX do not hesitate when in the WING AUX position as the wing tanks will be empty and a flame-out will result. When operating with the fuel tank selector in the FWD AUX position with low fuel quantity remaining, avoid uncoordinated turns or nose down attitudes as these maneuvers may result in loss of fuel supply to the forward tank booster pump, thereby causing a flame-out.

EMERGENCY JETTISONING.

Note

The speed limitations for jettison of external stores is presented in Section V.

The jettison limits are provided to prevent damage to the aircraft. It is impossible to predict the extent of damage which may occur if the external loads are released outside the established limits because of the number of factors involved. Depending on the emergency, it may be advisable to jettison the external load outside the release limits and risk some damage to the aircraft in order to increase the probability of being able to accomplish subsequent emergency procedures. However, when jettisoning external loads, consideration must also be given to several factors such as: sufficient airspeed to allow time for pilot reaction and jettisoning external load; terrain where external load will fall (populated areas, friendly or enemy territory, etc.); full or empty drop tanks; controllability of aircraft if one or more stores fail to release resulting in a dangerous asymmetrical condition at low altitude. In any event, the decision to jettison or retain external loads must be made by the pilot on the basis of his evaluation of the above factors and conditions existing at the time of the emergency.

JETTISON ALL STORES.

1. External stores jettison button – Depress and hold for at least 0.4 second. All stores will be jettisoned and pylons will be retained.

JETTISON INBOARD OR OUTBOARD PYLONS OR TANKS.

1. External stores arming switch – INBD or OUTBD.
2. Pylon or tank jettison switch – PYLON JETT or TANK RELEASE.

   On (45) and later aircraft hold switch actuated for 0.4 seconds.

ELECTRICAL POWER SUPPLY SYSTEM FAILURES.

Note

For equipment that will be inoperative in the event of electrical failure, see figure 3–5.

INVERTER FAILURE.

Inverter failure is noted if either the main or alternate inverter out indicator lights illuminate. Position the main inverter switch to OFF and the alternate inverter switch to ON if the main inverter light illuminates. If the alternate inverter indicator light illuminates, both inverters have failed.

ELECTRICAL POWER SUPPLY SYSTEM *Failures*

ESSENTIAL CONTROLS OR EQUIPMENT	ALTERNATE INVERTER FAILURE	MAIN INVERTER FAILURE	GENERATOR FAILURE (BOTH)	COMPLETE ELECTRICAL FAILURE
ATTITUDE INDICATOR HEADING INDICATOR FUEL QUANTITY IND FUEL FLOW INDICATOR HYDRAULIC PRESS. GAGE OIL PRESS. GAGE	POWER AVAILABLE WITH MAIN INVERTER ON	POWER AVAILABLE WITH BATTERY SWITCH ON AND ALTERNATE INVERTER ON		
GENERATOR OUT LIGHTS INVERTER INDICATOR LIGHTS TURN & SLIP INDICATOR EMERG HYD PUMP EMERG FUEL SYST SEL FUEL PRESS. WARN. LIGHT AIR START LAND. GEAR SELECTOR LAND. GEAR WARN. SYSTEM MECH ADV SHIFTER & IND WING FLAPS & POSITION IND SPEED BRAKES AN/ARC-33 (5) thru (30) MAIN BOOST PUMP (5) thru (25)	POWER AVAILABLE		POWER AVAILABLE WITH BATTERY SWITCH ON	
DRAG CHUTE EXTERNAL STORES JETTISON				
MAIN BOOST PUMP (30) & LATER WING & FORWARD BOOST PUMPS				
AN/ARC-34 (35) & LATER CANOPY JETTISON	INDIVIDUAL STAND-BY BATTERIES			
EGT INDICATOR TACHOMETER	INDEPENDENT OF AIRCRAFT ELECTRICAL SYSTEM			

Figure 3—5

CAUTION

In the event of failure of the main inverter the directional indicator, the oil pressure gage, hydraulic pressure gage, the fuel flow indicator and the attitude indicator, will tend to remain at their last reading. The above instruments will return to normal operation when the MAIN inverter switch is placed OFF and the ALT inverter switch is placed ON.

1. Land as soon as possible.

SINGLE GENERATOR FAILURE.

In the event either number 1 or 2 generator fails, the electrical load on the one operating generator will be reduced automatically by the shut down of certain electrical equipment. The automatic pilot, camera system and fuel booster pumps in two of the three internal tanks will shut off. The position of the fuel tank selector will determine in which tank the booster pump is operative, i.e.

ALL TANKS – Main tank booster pump

WING AUX – Wing tanks booster pumps (2)

FWD AUX – Forward tank booster pump

Generator 1 has sufficient capacity to operate all other electrical equipment.

Generator 2 does not have sufficient capacity to operate all other electrical equipment, therefor the load must be further reduced. This may be accomplished by switching the main inverter OFF and the alternate inverter ON.

Generator NO. 2 Failure.

If generator NO. 2 fails as indicated by the GEN 2 indicator light illuminating accomplish the following:

1. NO. 2 genertor switch – RESET. Hold in RESET for a few seconds, then return to ON. Lights should go out and voltmeter should read normal voltage.
2. If unsuccessful, GEN NO. 2 switch – OFF.

Note

If the NO. 2 generator-out indicator light remains illuminated and the voltmeter and loadmeter readings are normal, a possibility exists that the wire connecting the generator reverse current relay to the coil of the generator ON relay has broken. In this event, leave the NO. 2 generator switch in the ON position to permit the generator to remain in the system. However, it should be noted that with the broken wire, the secondary NO. 2 bus will be dropped out of the system and equipment on this bus will be inoperative.

3. Land as soon as possible.

WARNING

A failed generator is subject to partial disintegration and parts may be directed into the engine compressor or turbine sections possibly resulting in compressor and or turbine failure, fire, or flame-out.

Generator NO. 1 Failure.

If generator NO. 1 fails as indicated by the GEN 1 ingdicator light illuminating accomplish the following:

1. ALT INVERTER switch – ON (MAIN INVERTER switch – OFF.)

   See figure 3–5 for instruments that are operative and figure 7–5 the electrical load chart.
2. Reduce electrical load.
3. GEN 1 switch – RESET. Hold in RESET for a few seconds then return to ON.

   GEN 1 OUT indicator light should go out and voltmeter should read normal voltage.
4. If unsuccessful – Turn GEN 1 switch to OFF and land as soon as possible.

Note

If the NO. 1 generator out indicator light, remains illuminated, and the voltmeter and loadmeter readings are normal, a possibility exists that the wire connecting the generator reverse current relay to the coil of the generator ON relay has broken. In this event leave the NO. 1 generator switch in the ON position to permit the generator to remain in the system. However, it should be noted that with the broken wire, the secondary NO. 2 bus will be dropped out of the system and equipment on this bus will be inoperative.

5. If both generators are available–MAIN inverter ON (ALT inverter OFF). Resume normal operations.

WARNING

A failed generator is subject to partial disintegration and parts may be directed into the engine compressor or turbine sections possibly resulting in compressor and or turbine failure, fire, or flame-out.

BOTH GENERATORS FAIL.

If both generators fail as indicated by illumination of GEN 1 and GEN 2 OUT indicator lights, accomplish the following:

1. ALT INVERTER – ON (MAIN INVERTER – OFF.)

   See figure 3–5 for instruments that are operative and figure 7–5 the electrical load chart.

2. Reduce electrical load immediately.

CAUTION

No attempt should be made to change channels on the command radio, as 27 volts are required for the change. If change of channels is attempted, the set may hang up between channels.

3. Throttle – Idle (if practical).
4. Both generators – RESET. Hold in RESET for a few seconds, then return to on. Either or both lights should go out and voltmeter should read normal voltage.
5. If unsuccessful – Turn generators – OFF.
6. Land as soon as possible.

WARNING

Failed generators are subject to partial disintegration and parts may be directed into the engine compressor or turbine sections possibly resulting in compressor and or turbine failure, fire, or flame out.

7. If one generator is available – Follow procedures under single generator failure.
8. If both generators are available – MAIN INVERTER – ON (ALT INVERTER – OFF). Resume normal operations.

Fuel System Operation During Generator Failure.

In the event of failure of one or both generators, the fuel warning lights will operate normally. When operating the fuel system in the ALL TANKS position, the main tank boost pump is powered from the primary bus on **5** THRU **25** aircraft and from the NO. 1 secondary on **30** and later aircraft. The wing and forward boost pumps are powered from the NO. 2 secondary on all aircraft. The main tank is kept full by transfer of fuel from the external, wing, and forward tank, fuel until this fuel is used up. However, if operation is maintained with the fuel tank selector in the ALL TANKS position after a generator failure, the wing, and forward, tanks will not transfer fuel to the main tank as the NO. 2 secondary bus will not be energized. This will lower the fuel level in the main tank after external fuel is depleted. If the fuel tank selector is then switched to the WING AUX, or FWD AUX position, fuel will then be supplied to the engine, and also be transferred to the main tank as the boost pumps will be powered from the primary bus on **5** THRU **25** aircraft and from the NO. 1 secondary on **30** and later aircraft. The output of the wing or forward tank boost pumps will not be sufficient to supply both the engine and the transfer to the main tank and still build up enough pressure to put out both the wing, or forward pump pressure warning lights, and the fuel system boost pressure warning light. However, sufficient fuel will be supplied to the engine even though the fuel system boost pressure warning light is illuminated. Switching back to ALL TANKS will put out the fuel system boost pressure warning light. If the fuel shut-off switch for the MAIN tank is placed on the CLOSED position preventing fuel transfer to the main tank both the wing or forward and the fuel system boost pressure warning lights would be out when operating in WING AUX or FWD AUX respectively, when the main tank is partially full. Operate the fuel system in the same manner as for complete electrical failure except for external tanks, which will be used if the respective air pressure switches are in the INB'D or OUTB'D position.

CAUTION

Engine operation with the fuel system booster pressure warning light illuminated, especially at the higher altitudes should be continued with caution to prevent loss of thrust. Aerobatics or rapid changes in altitude should be avoided.

Note

Fuel in the pylon tanks can be recovered as the air pressure solenoid shut-off valves are energized by the primary bus.

COMPLETE ELECTRICAL FAILURE.

With complete electrical failure, the emergency hydraulic pump is inoperative, the landing gear will have to be extended by emergency procedure, flaps and speed brakes cannot be selected, the drag chute is not available, and only emergency brakes with accumulator brake pressure are available. (See figure 3-5.)

Equipment Available with Complete Electrical Failure.

With complete electrical failure, the following systems and instrument readings are available. (See figure 3—5.)

a. Tachometer.

b. Exhaust gas temperature indicator.

c. Canopy jettison.

d. AN ARC-34 radio. (35 and Later)

e. Emergency brakes.

Fuel System Operation with Complete Electrical Failure.

With complete electrical failure, the booster pumps and fuel system indicators will be inoperative and it will not be possible to select the emergency fuel system. At altitudes below 20,000 feet with JP-4 fuel satisfactory engine operation up to maximum range power settings can be maintained with fuel temperatures up to 26.6°C (80°F). Operate fuel system as follows.

1. Fuel tank selector — FWD AUX, until forward tank fuel is nearly depleted.
2. Fuel tank selector — WING AUX until wing tank fuel is nearly depleted.
3. Fuel tank selector — ALL TANKS for remainder of fuel in main tank.

WARNING

It will be necessary to time the operation on each tank and estimate remaining quantity to prevent running tanks empty. Do not permit FWD or WING AUX to empty as there is no electrical power available for an air start.

- With complete electrical failure, fuel in the pylon tanks cannot be recovered because the solenoid operated air shutoff valves will be closed, fuel warning lights are inoperative and fuel consumption from the fuel tanks will have to be estimated. Land as soon as possible.

HYDRAULIC POWER SUPPLY SYSTEM FAILURE.

In the event of utility system pressure failure, all hydraulic systems except the aileron and stabilator will be inoperative. These two control systems will continue to be actuated by the engine-driven power system pump. The rudder will be actuated through the mechanical linkage from the rudder pedals to the rudder. The landing gear can be extended by means of pneumatic pressure. In the event of power hydraulic system failure all hydraulic systems will remain operative. If both utility and power systems are inoperative due to hydraulic pressure failures, the emergency hydraulic system must be selected for aileron and stabilator system operation. A failure in the power hydraulic tubing or reservoir, however, will not be corrected by emergency hydraulic system operation.

UTILITY HYDRAULIC SYSTEM FAILURE.

The power hydraulic system will provide adequate pressure for continued flight. The rudder will be controlled by mechanical linkage and the spoilers will be inoperative. The rudder will buffet slightly at high speed with this type of failure as the rudder system is not irreversible when operating without hydraulic pressure. A landing should be made as soon as possible. When preparing to land it is recommended as an added safety precaution that the emergency hydraulic system be turned ON.

POWER HYDRAULIC SYSTEM FAILURE.

In the event of a line failure in the power hydraulic system, the emergency hydraulic system will be inoperative as these two systems use common lines. If a line failure is suspected in the power hydraulic system, it can be checked by turning the emergency pump on and noting the pressure recorded on the power hydraulic system gage. If the gage indicates that the emergency hydraulic pump has restored operating pressure to the power system, the failure is probably due to a power hydraulic pump malfunction. If pressure is not regained a line failure can be suspected. Turn emergency pump off after ascertaining where failure has occurred. The utility hydraulic system will provide adequate pressure for continued flight. However, a landing should be made as soon as possible. When preparing to land it is recommended that the emergency hydraulic system be turned ON if the failure of the power hydraulic system was caused by the power pump. If the failure is such that the emergency system is inoperative, the pneumatic compressor and spoilers should be turned off.

UTILITY AND POWER HYDRAULIC SYSTEM FAILURE.

An engine flame-out will cause both the utility and power hydraulic system pressure to drop considerably. However, the aircraft can be controlled adequately with hydraulic pressure and flow developed by a windmilling engine. Duration of the emergency hydraulic pump when operating on the aircraft's battery will be zero to 15 minutes depending on condition of battery and equipment operating. The emergency hydraulic system should be turned on prior to landing to supplement the windmilling engine-driven pumps.

WARNING

If both utility and power hydraulic pressure gages read zero and engine is operating normally the flight controls must not be moved until the emergency system is turned ON and hydraulic pressure is regained as shown on the power hydraulic pressure gage. If hydraulic pressure is not regained the surfaces will tend to creep in the direction of applied air loads. If hydraulic pressure is not regained abandon aircraft after ascertaining that gage pressure indications are realistic and that hydraulic failure has occurred. This can be accomplished by slightly operating the ailerons to determine if hydraulic pressure is available.

Note

The utility and power hydraulic pressure gages are powered by the inverter. In the event of an A-C power failure or engine failure gages will be inoperative and remain at the last pressure reading unless the MAIN INVERTER switch is positioned to OFF and the ALT INVERTER switch is positioned to ON.

FLIGHT CONTROLS SYSTEM FAILURE.

HIGH STICK FORCES.

There is a possibility of an electrical malfunction in the autopilot amplifier box causing excessively high stick forces due to autopilot engagement. In the event such a condition should be encountered during flight, the pilot should turn the ALT INVERTER switch to the ON position to eliminate the autopilot from the system.

TRIM SWITCH FAILURE.

In the event of trim switch failure, which may cause a "runaway" trim or no trim the following procedure is recommended to isolate the stick grip trim switch from the circuit:

1. **Alternate trim switch (PITCH TRIM) — OFF.**
2. **Alternate trim switch — NOSE UP** or **NOSE DN** position to trim aircraft, about pitch axis. Trim about the roll axis cannot be controlled.

/ING FLAPS SYSTEM FAILURE.

In the event of utility system failure it will not be possible to lower the wing flaps. In the event of primary bus failure, the flap position indicator will not function and it will not be possible to operate the flap system as the selector valve will remain in the neutral position.

SPEED BRAKE FAILURE.

In the event of hydraulic failure or primary bus power failure the speed brake cannot be opened. However, if the speed brakes are open when the failure occurs the air loads on the brakes will close them.

LANDING GEAR SYSTEM FAILURE.

LANDING GEAR RETRACTION WHILE ON THE GROUND.

it becomes necessary to retract the landing gear nile the weight of the aircraft is on the main struts proceed as follows:

1. Emergency ground retract switch — Actuate.
2. Landing gear selector handle — UP.

Note

The landing gear will retract only if hydraulic pressure and primary bus power are available.

LANDING GEAR EXTENSION WHILE IN THE AIR.

In the event that the left or right main gear does not extend in flight by means of the main system and utility hydraulic pressure is available the following procedure is recommended:

1. Cycle the landing gear while pulling negative G's or while shaking the aircraft.

   Pulling positive G's will normally aggravate this condition since the weight of the wheel rests on the inner landing gear door.

Note

The gear cannot be retracted, if desired, once the EMERG DOWN position is selected.

In the event that only the main gear can be locked down with the normal hydraulic system, the possibility exists of back pressure in the return hydraulic lines unlocking the main landing gear whenever the nose gear is lowered by means of the pneumatic system. The main gear will relock itself through normal hydraulic pressure with the landing gear selector handle left in the EMERG DOWN position. Therefore, the landing gear position indicators must be checked to ascertain whether all gears are down and locked after the pneumatic system has been used in the lowering of any gear.

2. If the landing gear will not extend by normal procedures, indicated by the landing gear indicator lights, warning horn or low hydraulic pressure, proceed as follows:

   a. Reduce speed to 225 knots IAS or below.

   b. Landing gear selector handle — EMERG DOWN.

Note

If the landing gear does not lock down (indicated by the landing gear indicator lights) hold the landing gear selector handle in the EMERG. DOWN position for a few seconds.

- Once the landing gear is extended by positioning the landing gear selector to the EMERG DOWN position you are committed to a wheels down landing since the selector handle can only be released from the EMERG DOWN position by use of a screwdriver. The possibility of blowing up the hydraulic reservoir due to back pressures when retracting the landing gear also exists.

   c. If necessary, yaw the aircraft to lock the main gear.

   d. Landing gear indicator lights — Check to ascertain that all gears are locked down.

Note

Observe the landing gear indicator lights when yawing the aircraft to determine when the spring-loaded downlocks on the main gear engaged. The nose gear is extended by air pressure and the spring-loaded nose gear downlock is locked when the strut is fully extended as indicated by the nose gear indicator light.

Note

The position of the nose gear may be observed through the view finder.

BRAKE FAILURE.

1. Use up the fuel until minimum fuel remains (1000 lbs.).
2. Emergency brake handle – Pull.
3. Throttle – CLOSED on touchdown.
4. If left brake is out, land on left side of runway.
5. If right brake is out, land on right side of runway.
6. If both brakes are out, land as short as possible at lowest safe speed, wing flaps DOWN, and speed brakes OUT.
7. Drag chute – Deploy after touchdown if installed.
8. Dragging the tail during landing will shorten the roll, if drag chute is not installed.
9. Prepare to engage barrier.

EMERGENCY BRAKE.

In the event normal brake failure is suspected, loss of utility hydraulic pressure or loss of braking action, actuation of the emergency brake handle engages the emergency brake accumulator.

Note

When utility hydraulic pressure is available after the emergency brake handle is actuated the brakes will function normally with unlimited applications, but will not retain the anti-skid feature of the normal brake system. If utility hydraulic pressure is lost, the emergency brake accumulator will allow a maximum of eight normal brake applications.

1. Maintain rudder directional control.
2. Actuate emergency brake handle.

CAUTION

When emergency brake system is utilized, stop aircraft straight ahead and do not taxi. Shut engine down and insure that landing gear down lock pins are installed and wheel chocks are in place.

ANTI-G SUIT FAILURE TO DEFLATE.

In the event the Anti-G suit fails to deflate after a maneuver.

1. Disconnect G-suit hose by pulling on the hose.

CANOPY MANUAL JETTISON.

Normally the canopy is jettisoned automatically as part of the ejection sequence by actuating the canopy jettison control (right handgrip on 5 thru 25 or either handgrip on 30 and later). This procedure also arms the ejection seat. The canopy may be jettisoned manually and the seat remain unarmed as follows:

1. Canopy manual jettison switch – ON (if installed), unlock canopy manually and raise intc airstream.

Canopy manual jettison speeds, if practical.

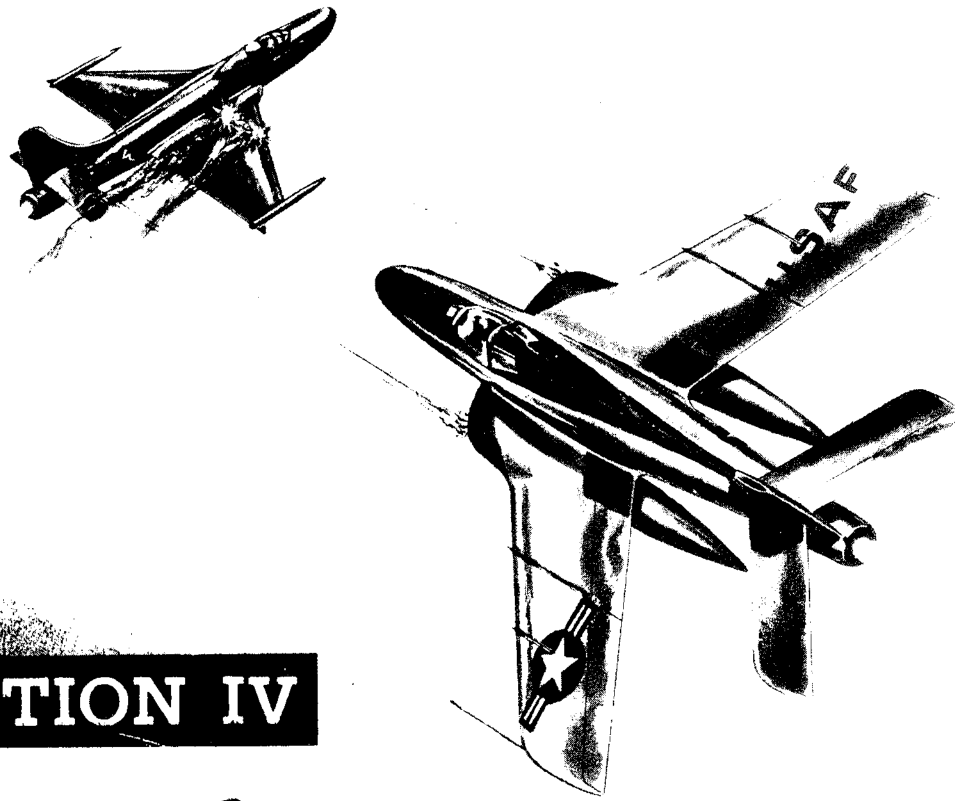
- 130 KIAS minimum with [763] complied with.
- 300 KIAS minimum with [763] not complied witl
- Jettison canopy at altitude and attitude to permi ejection in the event the canopy damages the en pennage.

WARNING

To detonate the canopy squibs the battery switch should be ON. The inspection requirements for the canopy jettison battery do not determine its ability to detonate the squibs.

When the canopy is raised the airstream will force it to the fully open position. In the full open position the right canopy mounting arm actuates a microswitch which completes a circuit to the explosive squibs if the canopy jettison switch is ON. The explosive squibs separate the canopy from the aircraft. If the canopy jettison switch is not installed, not used, or the squibs fail to fire, the airstream on the canopy will cause the canopy arms to break where they hit the upstop. Holes in the canopy arms establish the breaking point.

For runway emergencies, if the canopy is opened manually, the canopy lock should be engaged when the canopy is lifted to prevent it from slamming full closed and possible jamming.



SECTION IV

Auxiliary Equipment

TABLE OF CONTENTS

HEATING, PRESSURIZING AND VENTILATING SYSTEMS.

COCKPIT AIR CONDITIONING.

Pressurization, heating and ventilating are combined into an air conditioning system (figure 4–1) which is controlled electrically with power from the primary bus. When the canopy is closed the cockpit is sealed by an automatically inflated rubber seal. Air for pressurization, heating, ventilating and canopy seal inflation is obtained from the compressor section of the engine. Cockpit temperature is controlled by diverting a portion of hot air from the engine compressor section through a turbo-refrigerator, for cooling, before it enters the cockpit. Air enters the cockpit through two foot registers, a register behind the pilot's seat, and through an outlet on the right side directly opposite the pilot's chest. Pressurization is maintained for two schedules, normal or combat, by the pressure regulator, which releases air from the cockpit through a variable opening, designed to maintain the proper pressure differential and rate of change of cockpit air from the normal schedule. (See figure 4–2.) From sea level to 12,500 feet altitude the cockpit is unpressurized; from 12,500 to 31,000 feet altitude normal pressure remains equivalent to atmospheric pressure at 12,500 feet; above 31,000 feet a constant normal pressure differential of 5.0 PSI is maintained between cockpit and outside atmosphere. For the combat schedule; the cockpit is unpressurized from sea level to 12,500 feet; from 12,500 to 21,200 feet altitude cockpit pressure remains equivalent to atmospheric pressure at 12,500 feet; above 21,000 feet a constant pressure differential of 2.75 PSI is maintained between cockpit and outside atmosphere. The cockpit altimeter (20, figure 1–4) indicates the equivalent cockpit altitude. The dump valve operates automatically to relieve excessive cockpit pressure and can also be opened to dump cockpit pressure if necessary. Outside ventilating air is available to the cockpit only if pressurization is shut off.

5 THRU 15

Cabin Vent Switch.

Cockpit pressurization is selected by the cabin vent switch (figure 4–3) a rotary switch having three positions marked RAM, NORM PRESS and COMBAT PRESS. The RAM position opens the ram air valve and dump valves, allowing a flow of ram air through the cockpit and closes the engine air shut-off valve, stopping the flow of engine air to the cockpit. Cockpit temperature cannot be selected when operating in the RAM position. The NORM PRESS position closes the ram inlet and dump valves and positions the pressure regulator to maintain a maximum pressure differential between the cockpit and outside atmosphere of 5.0 PSI. The COMBAT PRESS position closes the ram inlet and dump valves and positions the pressure regulator to maintain a maximum pressure differential between the cockpit and outside atmosphere of 2.75 PSI. This position is provided for pilot safety to minimize danger resulting from sudden decompression, if battle damage causes a sudden loss of cockpit pressure during combat. The cabin vent switch is powered from the primary bus and is normally left in the NORM position.

5 THRU 15

Cabin Temperature Control.

Cockpit temperature is adjusted for pilot comfort by using the cabin temperature control (figure 4–3) which is a three-position switch marked COOL, off and HEAT, spring-loaded to the off position. By holding the cabin temperature control in the COOL or HEAT position the cockpit temperature can be adjusted to the individual needs if primary bus power is available and the cabin vent switch is not in RAM position.

20 and LATER

Cabin Temperature Control.

The cabin temperature control (figure 4–3), is a mechanically operated rotary control. The control is coaxially installed with the defroster control. The range for cockpit pressurization and temperature control is marked HEAT and COOL. The position of the temperature control, in relation to its total range from HEAT to COOL, gives a visual indication of the temperature selected. Temperature is changed by controlling the mixture of air through the mixing valve. The temperature control has a third or RAM AIR position. This position opens the ram air valve and the dump valve allowing a flow of ram air through the cockpit and closes the engine air shut-off valve stopping the flow of engine air to the cockpit.

Note

The cockpit cannot be pressurized when the temperature control is in the RAM AIR position.

Side Air Outlet Shut-off.

The side air outlet shut-off is located on the right side of the cockpit below the interior and exterior lights control panel. The quantity of hot or cold air going into the cockpit through this outlet may be varied by manually sliding the shut-off over the air outlet.

AIR CONDITIONING SYSTEM

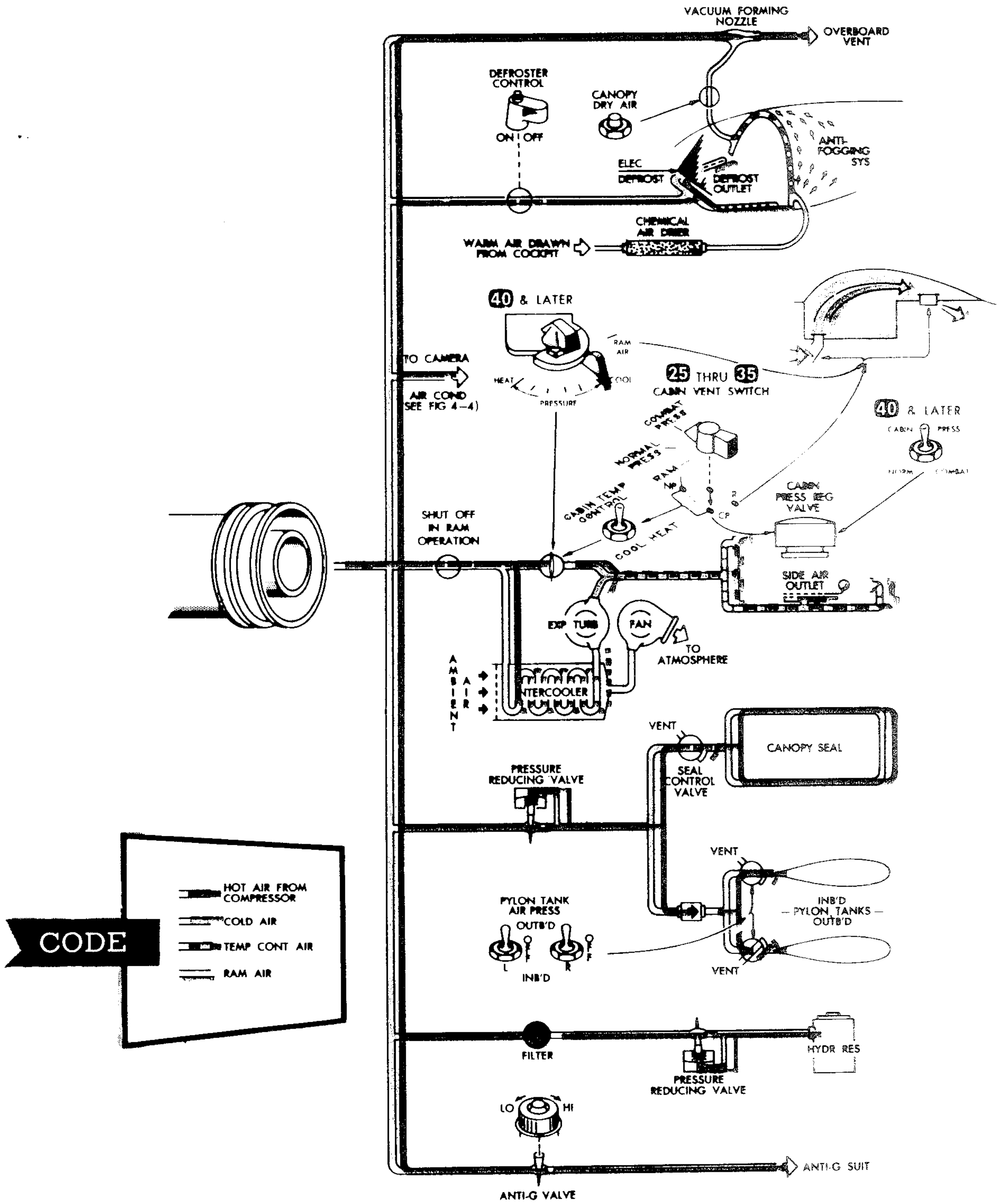


Figure 4—1

COCKPIT PRESSURE

SCHEDULE

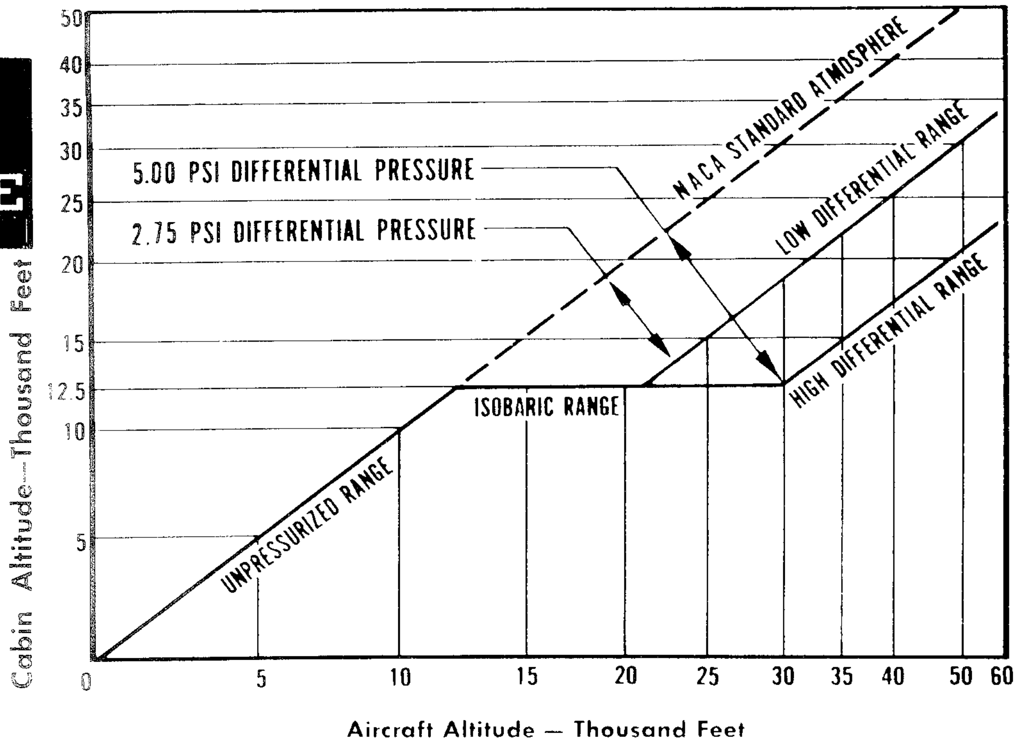


Figure 4—2.

CAMERA COMPARTMENT AIR CONDITIONING.

The camera compartment, in the forward fuselage, is air conditioned (figure 4—4) and controls, (both automatic and manual) are provided to maintain the compartment at approximately 18.3 to 29.3 degrees centigrade (65 to 85 degrees Fahrenheit). The temperature is maintained during all conditions of level flight and during or following a rapid descent from altitude after the initial 15 minutes of system operation. Compartment temperature is controlled by diverting a portion of hot air from the engine compressor section, through another turbo-refrigerator for cooling before it enters the compartment. A pressure relief valve is provided to insure against excessive pressure differentials. The camera windows in the compartment are kept free of frost and fog under all conditions of flight, including descents into warm humid atmosphere, by means of a recirculation unit and a chemical drier. This unit consists of a fan which circulates the warm air in the compartment through a heat exchanger, which keeps the air at the prescribed temperature. During rapid descents heated air enters the compartment through a chemical drier to equalize the pressure in the compartment. The recirculating unit operates automatically during a picture run. Camera compartment temperature is indicated by a temperature gage on the instrument panel. This air conditioning system is powered from the primary bus, while the recirculating fan is energized by the NO. 2 secondary bus.

Shut-off Switch.

The compartment temperature control shut-off switch is a two-position switch (figure 4—3) placarded SHUT OFF, ON and OFF on 5 and 10 aircraft and ON, and SHUT OFF on 15 and later aircraft and is powered by the NO. 2 DC secondary bus. The ON position opens the air-conditioning shut-off valve allowing air from the engine compressor section to enter the air conditioning system. It also turns on the camera vacuum system and energizes the recirculating fan to circulate camera compartment air through the heat exchanger.

Note

The shut-off switch must be in the ON position for operation of the air conditioning system, the recirculating system or the camera vacuum system.

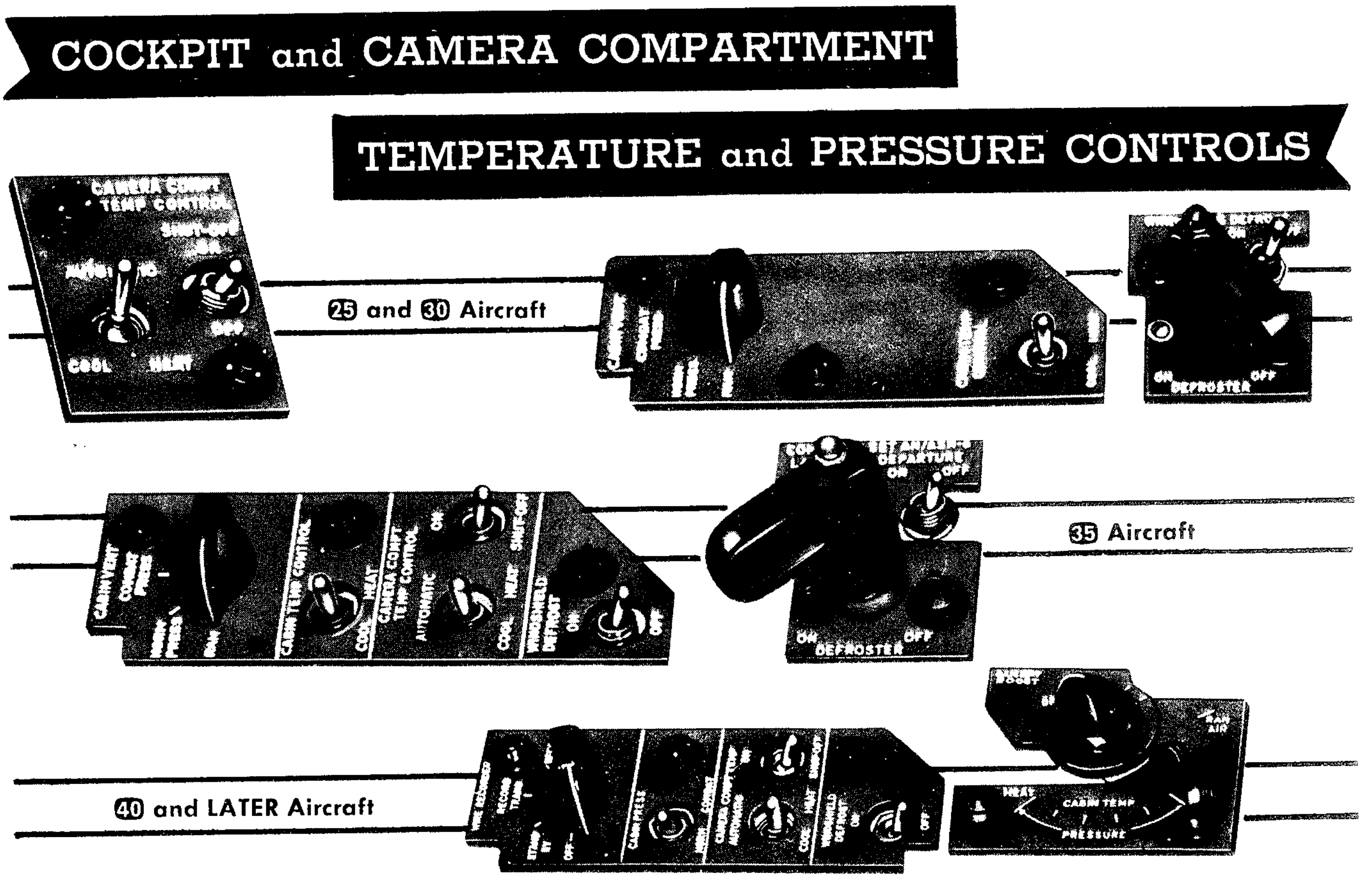


Figure 4—3.

Camera Compartment Temperature Control.

The camera compartment temperature is automatically or manually selected by the temperature control switch (figure 4—3) which has four positions; AUTOMATIC, COOL, HEAT and off and is powered by the DC primary bus. With the temperature control in the AUTOMATIC position, the camera compartment air conditioning system will automatically maintain a temperature range of 18.3 to 29.3°C (65 to 85°F) in the camera compartment, except when the recirculating unit is in operation for extended periods of time. If the automatic temperature control fails, the momentary COOL and HEAT positions, provide for manual control, and are used to obtain desired compartment temperature. By holding the temperature control in the COOL or HEAT position, the air mixing valve is motorized to close or open respectively. To stop the rotation, the control is released and automatically returns to off, stopping the positioning of the air mixing valve. By trial and error, the position of the air mixing valve is obtained to maintain proper compartment temperatures. The air mixing valve is powered from the primary bus.

Camera Power Switch.

The camera power switch (figure 4—21) is an ON-OFF switch and is described in detail under camera equipment in this Section. However, when the camera power switch is in the OFF position, two recirculating valves are positioned so that the air conditioning system operates. When positioned to the ON position, the recirculating valves are positioned so that heated air from the air conditioning system is directed through the heat exchanger. The recirculating valves are actuated by the primary bus.

Note

The air conditioning shut-off switch must be in the ON position for the recirculating system to operate.

CAUTION

Camera power switch should be on for 15 minutes and camera compartment temperature within limits (18.3°C to 29.3°C) before photography is attempted.

CAMERA COMPARTMENT AIR-CONDITIONING *System*

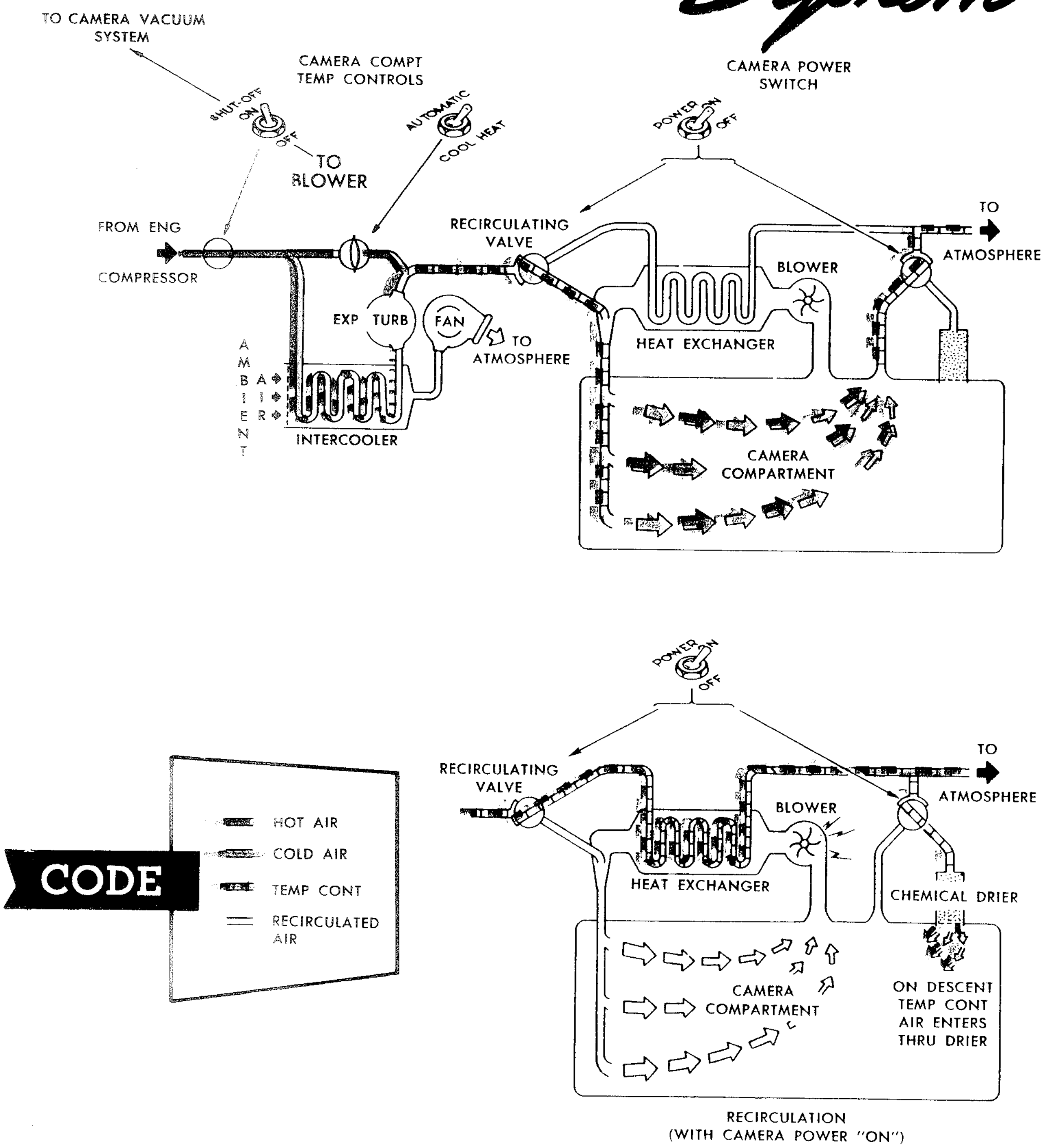


Figure 4—4

Camera Compartment Temperature Indicator.

Temperature in the camera compartment is shown on the camera compartment temperature indicator (15, figure 1–4). The indicator is powered by the primary bus.

ANTI-ICING AND DE-ICING SYSTEMS.

DEFROSTING SYSTEM.

The center windshield panel is designed and built so that an electrical current, passing through an electrical conductive coating between laminations, keeps the windshield panel at a temperature of 43°C (110°F). Both AC and DC power are required for operation; the AC being supplied only by the single-phase inverter and the DC by the primary bus. The side panels on the windshield and the side panels on the aft canopy are defrosted with hot air supplied from the engine compressor to the perforated defroster tubing assembly. The hot air defroster is available whenever the engine is running, regardless of the position of the cabin vent switch.

Windshield Defrost Switch.

The windshield defrost switch (figure 4–3) located on the right console, is a two-position switch placarded WINDSHIELD DEFROST, with positions placarded ON and OFF. The OFF position opens the AC power circuit which in turn automatically opens the DC power circuit through a relay. The ON position energizes the windshield defroster circuit if the DC primary and NO. 1 secondary busses are energized, and the MAIN inverter switch is in the ON position.

Defroster Control.

The defroster control (figure 4–3) is a manually operated shut-off valve that controls the flow of hot air from the engine compressor section to the defroster tubes. The control is placarded DEFROSTER and has two positions: ON and OFF. The ON position supplies hot air to the windshield side panels and the aft canopy side panels, and will operate regardless of the position of the cabin vent switch. On 20 and later aircraft the defroster control is mounted co-axially with the cabin temperature control on the right console. There are two unmarked detent positions between the ON and OFF positions.

CAUTION

Defroster operation should be kept to a minimum during ground operation to prevent distortion of the windshield side panels.

Note

Due to the large mass of glass in the bullet-proof windshield, it is essential that the defroster be turned on at least 30 minutes before a descent from altitude is undertaken. Since a descent often cannot be anticipated 30 minutes in advance, the defroster should be turned on whenever an altitude of 20,000 FT is reached, and left on for the duration of the flight. During a ground support mission, the defroster should be turned on immediately after take-off. If it is found that the defroster air is excessively uncomfortable, the defroster valve should be closed down to some intermediate position, but should not be turned off.

ANTI-FOGGING SYSTEM.

The canopy and the windshield side panels are de-fogged by air, taken from the cockpit, passed through a chemical drier, then through the space between the inner and outer surfaces of the panels and vented overboard. The drier must be checked periodically to make sure that the chemical is still active. Operation is controlled manually through a shut-off valve.

Canopy Dry Air Switch (Anti-Fogging).

The canopy dry air switch (7, figure 1–6) controls the flow of heated and chemically dried air, taken from the cockpit and directed between the inner and outer surfaces of the side windshield and the canopy for anti-fogging purposes. The switch is a momentary push button type marked CANOPY DRY AIR. When depressed, the switch mechanically opens a valve which allows heated and dried air to pass to the space between the inner and outer spaces of the canopy and side windshields. On 5 thru 35 aircraft the switch must be held for the system to operate. Actuating the switch for periods longer than necessary for clearing the canopy and side windshields of fog is unnecessary as continued use will only replace dry air with more dry air. On (40) and later aircraft, an automatic time delay has been incorporated so that it is only necessary to depress the switch momentarily to actuate the system. The shut-off valve will then automatically close after a period of 15 to 30 seconds have elapsed.

PITOT HEATER.

The pitot tube, installed in boom on the left wing tip, is electrically heated from the primary bus to keep it ice free.

Pitot Heater Switch.

The pitot heater switch (21, figure 1–6), has two positions OFF and PITOT HEAT. The PITOT HEAT position turns on the heater element which heats the pitot tube and keeps it free of ice.

CAUTION

The pitot heater should not be used on the ground as serious overheating may occur.

FUEL FILTER DE-ICING SYSTEM.

A fuel filter de-icing system is installed in some aircraft to remove ice that may collect in the engine fuel filter. Alcohol is injected into the fuel line upstream of the fuel filter by means of an electric pump which is inoperative if the alcohol tank is empty or in the event of primary bus failure. Automatic or manual operation may be selected. The alcohol supply is sufficient for approximately two minutes of continuous injection. An icing condition is indicated by a warning light. Specifications of the de-icing fluid used is noted in the servicing diagram (figure 1–31).

CAUTION

On aircraft where the fuel de-icing system has been reactivated by [R521], the low pressure fuel filter icing conditions will be indicated to the pilot and/or ground personnel by gradual reduction in engine RPM only. The filter icing light will not indicate this condition. When a gradual reduction in engine RPM is experienced, the fuel filter de-icing switch should be placed in the manual position until engine RPM again increases to normal operating requirements. When engine RPM is normal, place the de-icing switch in the OFF position.

Fuel Filter De-Icing Switch.

The fuel filter de-icing switch located on the left console aft of the fuel control panel is a three-position switch placarded AUTO, OFF, and MANUAL, and is powered by the DC primary bus. In the AUTO position the injection of alcohol starts automatically as soon as a predetermined pressure drop across the engine high pressure fuel filter is sensed and ceases as soon as the pressure drop across the fuel filter returns to normal. In the MANUAL position, a continuous flow of alcohol is injected into the fuel filter. The MANUAL position is used in the event of failure of the automatic system.

Note

If the fuel filter de-icing switch is positioned to MANUAL with the throttle closed and the fuel tank selector in the OFF position, the alcohol will build up enough pressure in the fuel line to prevent turning the fuel tank selector to any other position.

Fuel Filter Ice Warning Light.

The fuel filter ice warning light (figure 1–16) is an amber light marked FILTER ICE-WARN. The light will illuminate automatically when the fuel pressure drop across the engine high pressure fuel filter is indicative of icing and will go out when the pressure returns to normal.

CAUTION

The ice warning light will not indicate an icing condition in the low pressure fuel filter.

COMMUNICATION and ASSOCIATED ELECTRONIC EQUIPMENT

TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION OF CONTROLS
Command Set	AN/ARC-33 OR AN/ARC-34	Two-way voice communication	30 miles at 1000 feet altitude to 135 miles to 10,000 feet altitude	Left Console
Interphone	AN/AIC-10	Improved radio reception and ground maintenance station	Within airplane	Left Console
Radio Compass	AN/ARN-6	Radio Navigation	250 miles	Right Console
IFF/SIF Set	AN/APX-25	Automatic Identification	100 miles up to 50,000 feet altitude	Right Console
Radar Set	AN/APW-11	Ground Support	—	Right Console Provisions only
Tacan	AN/ARN-21	To provide an improved tactical air navigational system.	—	Control panel on right console; Course, bearing, distance and heading indicators on main instrument panel.
Wire Recorder	25X	Record pilot's comment and radio transmission	—	Right Console

Figure 4—5.

COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT.

Communication and associated electronic equipment consists of radio sets to provide aircraft-to-aircraft and aircraft-to-ground communication; radar sets for aircraft identification and ground support and navigation equipment for guidance during flight. All sets are remotely controlled from the cockpit with visual indicators on the console and instrument panel. Antennas are concealed within the aircraft and therefore are protected from air loads during high speed flight and are kept free of ice and dirt.

OPERATION OF COMMUNICATION EQUIPMENT.

Insert microphone plug and headset plug into two extensions on the front of the pilot's seat. The primary bus must be energized for command set operation. Each set is described individually in the following paragraphs:

Note

The Command Radio Set and the Radio Compass will operate from the aircraft batteries as power is supplied to these sets from the primary bus. When the main inverter switch is ON the IFF/SIF Set is energized by the main inverter, on modified and later aircraft, and by the alternate inverter on unmodified aircraft. On all aircraft the IFF SIF Set is inoperative when the main inverter switch is OFF.

INTERPHONE.

The AN/AIC-10 interphone set is provided so that type M-32/AIC oxygen mask microphone and type H-75/AIC headset or their equivalents can be used. This microphone and headset combination makes possible voice communication of high intelligibility in jet aircraft at all altitudes and under severe ambient noise conditions. Units of the interphone control, on the aft end of the left console, are used to connect the headset and microphone to the communications equipment. The interphone volume control affects the headset volume of communication equipment. It is possible to adjust to volume control in flight. An interphone station is provided in the camera compartment in the forward fuselage for use of ground maintenance personnel. The interphone is powered by the primary bus.

COMMAND RADIO CONTROL PANEL
AND REMOTE CHANNEL INDICATOR

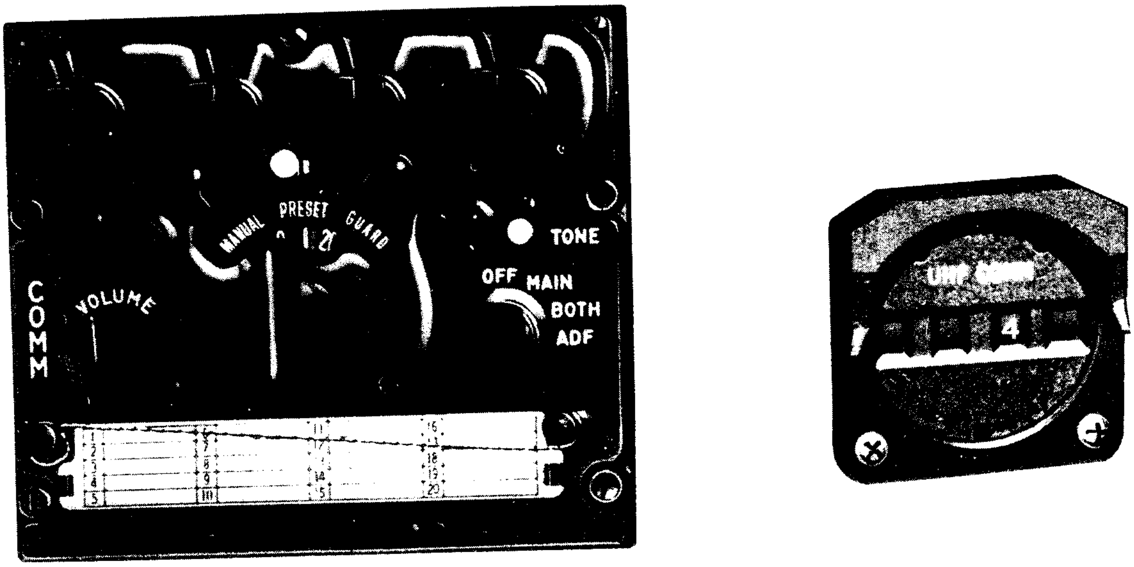


Figure 4—6.

COMMAND SET.

Radio set AN/ARC-33 or AN/ARC-34, is a remotely controlled receiver-transmitter operating in the 225.0 to 399.9 mc band. A total of 1,750 crystal controlled receive-transmit channels, in increments of one-tenth of a megacycle, are within tuning range of the radio set and may be manually selected by the pilot. Any 20 channels within the radio's total frequency range may be preset in any order for quick selection by the rotary type preset channel selector switch on the radio control panel. Four knobs are provided for manual selection of an operating frequency, so that manual operation does not disturb the preset channel arrangement. Receiver and transmitter tuning is automatically completed after a channel change. Two receivers, a main, and a guard receiver which is constantly tuned to the guard frequency are provided. The guard receiver may be operated along with the main receiver, thus making it possible to monitor the guard channel for command or emergency communication, while still carrying on communication on another channel. Receiver operation is manually selected by a rotary type function switch. Continuous tone transmission is also provided through a tone button which may be used for voice transmission, or as a key for code signals. All radio operating controls (figure 4—6) are located on the left console with the exception of the microphone press-to-talk button, which is located on the throttle. The pilot can monitor his own transmitted signals through a side-tone circuit which feeds from the transmitter into his headset. The transmitter and the main receiver are tuned to the same frequency. The radio will operate if the primary bus is energized. On 35 thru 45 (to Serial NO. 53-7633) aircraft modified by [697] and 45 and later aircraft (from Serial NO. 53-7634) a standby emergency battery may be switched on if the electrical system does not produce sufficient voltage for AN/ARC-34 radio operation. An on-off type switch, which is guarded in the off position by a safety-wired cover guard, is provided to switch the radio to emergency battery power. A circuit breaker press-to-test light is mounted adjacent to the standby battery switch and is used to test the standby battery circuit. The indicator light will illuminate if the circuit is functioning regardless of the position of the standby battery switch.

Note

Aircraft prior to 35 modified by [786] have an AN/ARC-33 radio with an AN/ARC-34 type control panel. These aircraft do not have the command radio standby battery.

Function Switch.

The function switch (figure 4–6), a rotary, four-position selector switch marked OFF-MAIN-BOTH-ADF is used to turn the command radio on or off, or select the type of receiver operation desired. The OFF position shuts off power to the transmitter, main and guard receivers. In the MAIN position the radio receives on the main receiver and both the main receiver and transmitter are tuned to the preset or manually set up channel selected while the guard receiver remains inoperative. When the function switch is in the BOTH position the radio will receive signals simultaneously from the guard receiver which is constantly tuned to the guard channel, and from the main receiver, which with the transmitter will operate on either a preset or manually selected frequency depending upon the type of channel selection used. The ADF position is for an automatic direction finder which is not installed.

Manual-Preset-Guard Selector Switch.

The lever type, three-position selector switch (figure 4–6) marked MANUAL-PRESET-GUARD is used to select the method of channel selection or to switch both the main receiver and the transmitter to the guard channel frequency. A MANUAL-PRESET-GUARD switch indicator window is provided and the type of selection as set up by the switch is indicated, while the other positions are visible, but covered by a green shutter. In the MANUAL position, the four tuning knobs at the top of the control panel are used permitting manual selection of any one of the 1,750 frequencies in the radio's tuning range for transmit-receive operation. The PRESET and GUARD positions on the MANUAL-PRESET-GUARD switch indicator window will be covered, as will the preset channel indicator window. The indicator windows above each of the four manual tuning knobs are open and provide a direct reading in megacycles and tenths of a megacycle of each frequency manual selected by the knobs. In the PRESET position, the preset channel selector switch is used and transmit-receive operation may be carried on any one of the 20 preset channels as selected by the preset channel selector switch and indicated in the preset channel indicator. The manual tuning knob windows and the MANUAL and GUARD positions on the MANUAL-PRESET-GUARD switch indicator windows will be blanked out. In the GUARD position, the main receiver and transmitter are switched to the guard channel frequency and the guard receiver is inoperative even if the function switch is in the BOTH position. The manual tuning indicator window, the preset channel indicator window and the PRESET and MANUAL positions in the MANUAL-PRESET-GUARD indicator window will be covered.

Note

On aircraft prior to 35 with [786] not complied with the AN/ARC-33 radio set has no provisions for manually selecting a specific frequency. Only the 20 preset channels (frequencies) can be selected.

Tone Push Button.

The spring-loaded push button marked TONE (figure 4–6) switches the radio over from receive to tone transmission on the manually selected or preset frequency that the radio is operating on as long as the depressed position is held. A 1,020 cycle tone is continuous during voice transmission as long as the tone button is depressed. The tone button may also be used as a key for code transmission. The microphone press-to-talk button, regardless of its position has no effect on the operation or function of the tone button.

Note

Tone transmission is usually perceptible over a slightly greater distance and through greater interference than voice transmission. Therefore it is especially adaptable in an emergency for code key operation, when interference or jamming conditions make voice transmission impractical, or to serve as a radio marked signal for direction finding.

Volume Control.

The volume control (figure 4–6) marked VOLUME regulates the headset volume for signals received on both the main and guard receivers. Volume control range is predetermined so that the signal volume may not be reduced below a preset level.

Microphone Press-to-Talk Button.

The microphone press-to-talk button (figure 1–10) located on the engine throttle control, switches the radio set from receive to transmit operation for voice modulation when held in the depressed position. When the button is released the radio returns to receive operation. The microphone press-to-talk button, regardless of its position, has no effect on the function or operation of the tone push button.

Emergency Radio Battery Switch.

On aircraft equipped with the AN/ARC-34 radio, an emergency radio battery and emergency radio battery switch are provided for emergency radio operation. The emergency radio battery switch is located on the left console and is safety-wired and guarded in the off position by a cover-type guard, while the on position is marked ON. The emergency battery is a 26.25 volt wet cell, silver zinc battery located below the aircraft battery, and is connected to the radio through a circuit breaker in the battery well.

Note

On 35 thru 45 aircraft not modified by [697] and on all earlier aircraft the emergency battery is not provided.

Remote Frequency and Channel Indicator.

Aircraft modified in accordance with [786] have a remote frequency and channel indicator installed on the main instrument panel. The indicator enables the pilot to see what frequency or channel he is selecting on the modified AN/ARC-33 or AN/ARC-34 radio without looking down at the console.

Emergency Radio Battery Circuit Breaker Check Light.

The emergency battery circuit breaker check light is a press-to-test light mounted adjacent to the AN/ARC-34 radio emergency battery switch and is provided to check the emergency radio battery circuit breaker in the battery well. The check light will function regardless of the position of the emergency radio battery switch.

Starting.

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.

1. Place the function switch in the BOTH position.
2. Turn MANUAL-PRESET-GUARD selector switch to PRESET position.
3. Rotate preset channel selector knob until desired channel number appears in preset channel indicator window. Allow approximately one minute for equipment warm-up and automatic channel tuning adjustment cycle. At the end of the warm-up period the equipment will be in the standby condition ready to receive signals on the preset command and fixed guard frequencies simultaneously. During transmission periods the pilot should receive his own signals on a side-tone received by his headset. A little receiver noise may or may not be heard during non-transmission periods.

Note

When selecting a new preset channel there will be a delay of four seconds before the automatic tuning cycle adjusts the set for operation on the newly selected frequency.

4. Adjust the volume control for a comfortable signal level in the headset.
5. Before transmission, check that equipment is functioning and tune-up is complete by listening for voice side-tone, or tone signal when tone button is depressed.

Stopping.

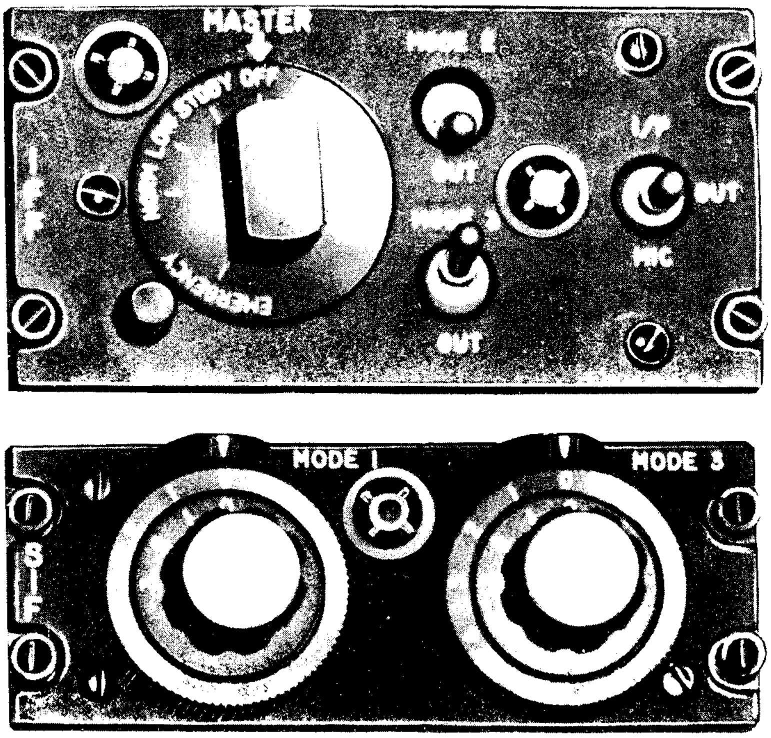
1. Place the function switch in the OFF position.

Note

Once the radio set has been turned off, the one minute warm-up period must be completed before radio transmission can be resumed.

Operating Precautions.

1. Transmit only when the channel is clear to prevent confusion and loss of messages. The guard frequencies should not be used for transmission unless the message is urgent.
2. Use both receivers for general reception unless signals from one make the desired signals from the other unusable.
3. It is possible to set the preset channel buttons or manual frequency selector knobs for frequencies below 225.0 megacycles. Since this is below the operating frequency range, the automatic tuning mechanism cannot accomplish tune-up. It will operate for approximately 120 seconds and will then be turned off automatically by a protective relay. Under these conditions there will be no side-tone if transmission is attempted and operation of the frequency selection knobs will have no effect. To restore operation, select a channel within the 225.0 to 399.9 megacycle range. Turn the function switch to the OFF position and then to BOTH. After approximately one minute the radio equipment will operate in a normal manner.



IFF/SIF

AN/APX-25

CONTROL PANELS

Figure 4—7.

IFF/SIF SET — AN/APX-25.

The AN/APX-25 radar identification set (IFF/SIF) enables the aircraft in which it is installed to identify itself automatically as friendly, whenever it is challenged by the proper signals from other appropriate radar recognition equipment at ground bases, or in other aircraft. The set has two supplementary purposes: (1) It enables specific friendly aircraft to identify themselves apart from numerous other friendly aircraft; (2) provides means for transmitting a special coded signal called the "emergency reply." In operation the AN/APX-25 set receives coded interrogation signals and transmits coded reply signals to the source of the challenging signals where the reply codes are displayed, together with associated radar information (targets, etc.), on the radar indicators. When a radar target is accompanied by a proper reply code from the IFF set, the target is considered friendly. Three modes of operation are provided for response to interrogation signals: mode 1, mode 2, and mode 3 which are used for security, personal, and traffic identifications, respectively. The IFF set provides for two methods of reply coding: Mark X and Mark X SIF. An internal selector switch (set up by ground personnel only) permits the IFF set to be operated in the Mark X, or Mark X SIF, configuration. The Mark X configuration provides for use of the IFF (transponder) control panel only, and selection of reply coding is limited to the one code reply combination preset into the equipment. When using the Mark X SIF configuration, a SIF (selective identification feature) control panel is used in conjunction with the IFF control panel, providing for elaboration of the reply coding through the many code combinations available with the SIF control panel. The radar identification set is powered by the AC and DC secondary busses.

Note

> In Mark X operation the SIF (selective identification feature) control panel is eliminated, rendering the code selector dials inoperative. However, the set will still operate in all three IFF modes providing limited preset interrogation and response signals.

Identification Radar Control Panel.

Two radar control panels (figure 4—7) marked SIF and IFF are located on the right console. The IFF or transponder control panel contains two mode switches, an identification of position (I P) switch, and a fire position master switch whose positions are OFF, STDBY, LOW, NORM, and EMERGENCY. In the STDBY position, the system is inoperative but ready for instant use. In the LOW position, the system operates in partial sensitivity and replies only in the presence of strong interrogation. In the NORM position, the system operates at full sensitivity which provides maximum performance. In the EMERGENCY position, the system replies to all modes of interrogation with a special coded signal to indicate an emergency. The mode 2 switch placarded MODE 2 and OUT, is used by the pilot for personal identification. The mode 3 switch, placarded MODE 3 and OUT is used by the pilot for traffic identification. The identification of position (I P) switch, placarded I P, OUT and MIC, is used by the pilot upon request to provide momentary identification of position when held in the I P position. When placed in the MIC position, the identification of position signals are transmitted when the microphone button is held depressed. The SIF control panel contains two, concentric, rotary, code selector switches which are used to select the specified code signals to be used in mode 1 or mode 3 operation when in Mark X SIF operation. The specified coded signals to be used in mode 2 are preset on the ground and cannot be changed in flight. The rotary code selector switches are marked MODE 1 and MODE 3, and each contain inner and outer knobs for selection of specified code signals. The inner and outer knobs of mode 3 and the outer knob of mode 1, selector switches, are marked 0 thru 7 consecutively, as the knobs are turned clockwise. The inner knob of mode 1 code selector switch is marked 0 through 3, consecutively.

Operation of Identification Radar.

1. Rotate master switch to STDBY, to maintain equipment inoperative but ready for instant use.
2. Rotate master switch to NORM to place equipment in operation.

Note

The LOW position of the master switch should not be used except upon proper authorization. Mode 1, the security identification feature, is in operation when the master switch on the IFF control panel is in NORM.

3. Set mode 2 and mode 3 switches OUT unless otherwise directed.
4. Set mode 1 and mode 3 code selector switches on the SIF control panel as directed (when operating in the Mark X SIF configuration).
5. For emergency operation, press dial stop and rotate master switch to EMERGENCY, so that the set will automatically transmit a special coded distress signal in response to interrogation.
6. Rotate master switch to OFF to turn set off.

RADIO COMPASS

CONTROL PANEL

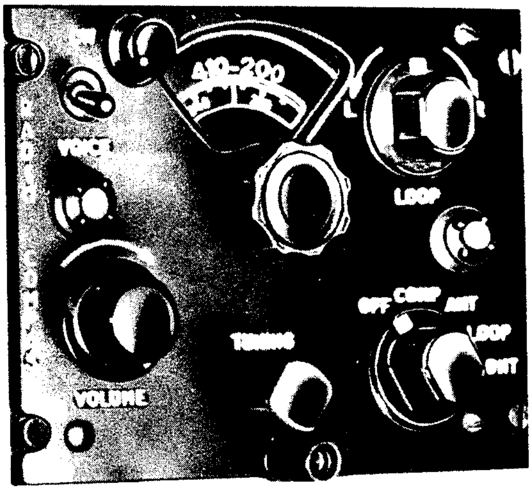


Figure 4—8.

RADIO COMPASS — AN/ARN-6.

The radio compass AN ARN-6 (figure 4—8) is an airborne navigational instrument. There are four bands covering a frequency range of 100 to 1,750. The radio compass is capable of providing the following:

1. Automatic visual bearing indication of the direction of arrival of radio signals and simultaneous aural reception of corresponding sound.
2. Aural reception of radio signals, using a non-directional antenna.
3. Aural-null directional indicators of the arrival of radio signals using a loop antenna.

Note

The AN ARN-6 radio compass is deleted on aircraft modified by [783].

Starting.

1. Turn the function switch to COMP, ANT or LOOP position.

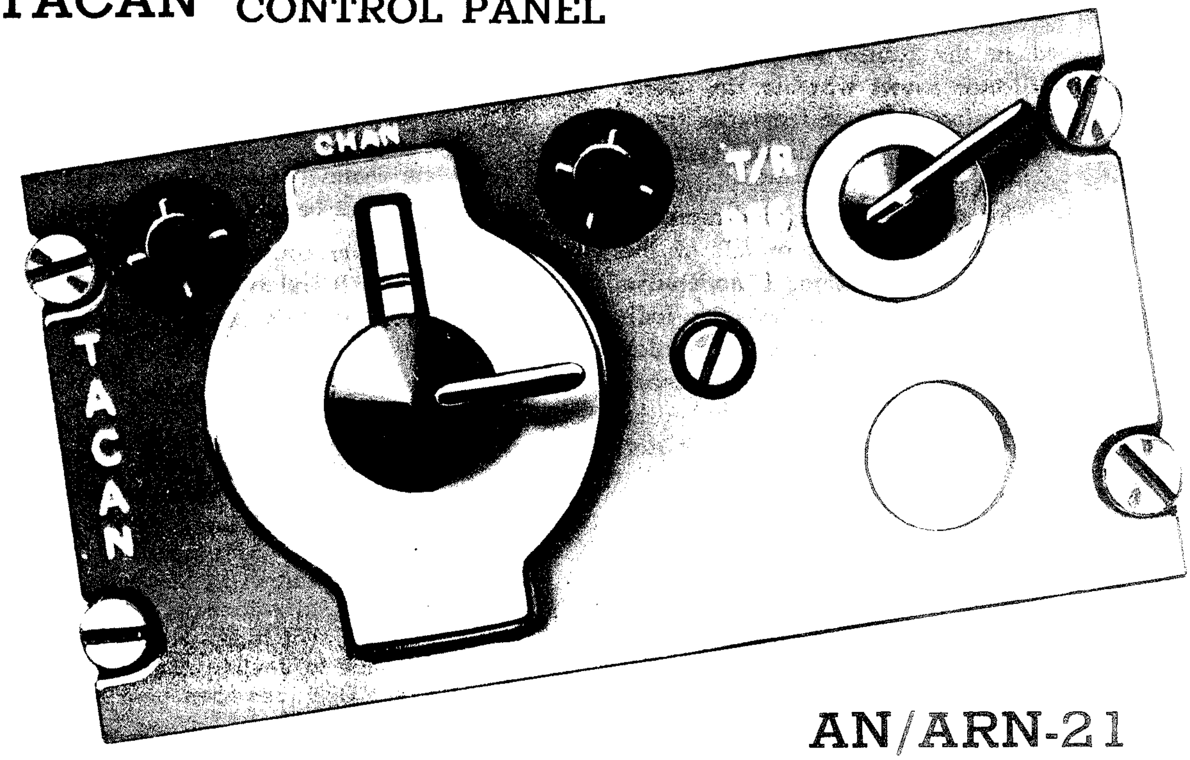
Note

The function switch position marked CONT on the control panel is not used on this installation.

Stopping.

1. Rotate function switch to OFF.

TACAN CONTROL PANEL



AN/ARN-21

Figure 4—9.

TACAN — AN/ARN-21.

The TACAN set is a tactical air navigation system which provides cockpit displays of distance and bearing to a suitable TACAN surface navigation beacon. Slant range of the set is line-of-sight up to approximately 195 nautical miles. The Morse code which identifies the surface beacon is audible in the headset. Major components of the set are a receiver-transmitter, indicator coupler, and a control panel. Distance, bearing, and course signals from the receiver-transmitter are sent to the indicator coupler, which processes the signals and couples them to the various cockpit indicators. The set is powered by the DC primary bus and the main inverter.

TACAN Controls and Indicators.

Channel Selector Knob. The channel selector knob (figure 4—9) selects the desired navigation beacon channel. The left-hank knob selects the tens and hundred figure of the beacon channel number; and the right-hand knob selects the unit figure of the beacon channel. Combinations of channel settings may be made from 00 to 129, but the equipment operates on channels 01 thru 126 only. The channel frequencies selected are shown on a window marked CHAN.

Note

Allow about 12 seconds after channel selection for the bearing indicator and the distance (range) indicators to stabilize.

Power Control Switch. The power control switch (figure 4—9) is a three-position switch marked OFF, REC and T R. The OFF position deenergizes all of the ARN-21 equipment. The REC position places the receiver portion of the equipment into operation so that only bearing information is furnished by the radio set. In the T R position, the airborne equipment transmits a distance-measuring pulse, and interrogator, a corresponding reply pulse from the ground beacon, to furnish distance information in addition to bearing information.

Volume Control. A volume control (figure 4—9) marked VOL is provided for adjusting the volume of the audio identification signal received from the beacon. Clockwise rotation increases the volume.

Course Indicator. The course indicator (7, figure 1—4) has a course set knob, a course selector window, TO-FROM indicator, a course deviation indicator, a glide slope indicator and a heading pointer. The course set knob permits selection of a desired course which is displayed in the course selector window. The TO-FROM indicator shows whether the course selected will take the aircraft to or from the transmitting station, the course deviation indicator (vertical bar) shows the position of the aircraft in relation to the desired course. The heading pointer indicates the number of degrees of heading, right or left of the course selected. The glide slope indicator is inoperative and not used in TACAN. The course deviation indicator is provided with a course warning flag which is visible at anytime a received signal is unreliable, or the system is OFF. The marker beacon light, in the upper right corner of the course indicator, is not operational.

Radio Magnetic Indicator. The radio magnetic indicator (8, figure 1—4) is located on the instrument panel. The indicator has two bearing pointers marked 1 and 2, and a rotating compass card. The compass card is rotated by signals from the J-2 directional indicator (when operating) and provides magnetic heading of the aircraft displayed against a fixed top index at the 12 o'clock position on the indicator. The number 1 pointer is normally driven by the AN/ARN-6 radio compass which is removed from the aircraft when [775] or [783] are complied with. The double barred (NO. 2) pointer shows the magnetic bearing to the TACAN surface beacon from the aircraft. The indicator operates when the TACAN function switch is at either REC or T/R. If the bearing signal is lost or weak, the bearing pointer spins clockwise until the signal is picked up again. During a channel change or when the equipment is first turned on, the bearing-pointer may falsely lock on momentarily to a bearing, but as the correct data is fed into the system, the pointer will swing to the correct bearing.

Range Indicator. The range indicator (14, figure 1—4) shows the slant range distance in nautical miles from the aircraft to surface beacon. This indicator operates only when the TACAN function switch is at T/R. When the indicator is not operating or when the channel is being changed, a red bar drops across the numbers displayed on the indicator. If a return signal from the surface beacon is lost because of interference or because the aircraft is beyond the 195 mile range of the equipment, a memory circuit retains the lost distance for about 10 seconds; if the signal is still unreliable then the red bar drops across the numbers. When the aircraft is back within range the range indicator corrects itself and the red bar disappears automatically. There will be a momentary false indication when the equipment is first turned on or when changing channels. However wait a few seconds to ensure that the indication can be relied upon.

Operation of TACAN (AN/ARN-21).

1. Rotate channel selector knob on TACAN control panel to surface beacon channel desired.
2. Move power control switch from OFF to either REC or T/R and allow about 2 minutes for warm-up, or until radio magnetic indicator stops spinning.

Note

Occasionally TACAN equipment will "lock-on" a false bearing which will be 40° or a multiple of 40° in error. These errors can be on either side of the correct bearing. When using TACAN, cross check for false "lock-on" with radar, VOR, DF, or other available means. A False lock-on can usually be corrected by switching to another channel and then back to desired channel or by turning the set OFF and then back ON to recycle the search mode.

Note

This deficiency does not affect the DME display provided by the TACAN equipment. If, during an emergency, the size and direction of error of a false "lock-on" can be determined, TACAN can be utilized if compensation is made for the bearing error.

3. Adjust volume control to desired level.
4. To home on surface beacon:
   a. Turn aircraft until NO. 2 pointer is aligned with the top index on radio magnetic indicator.
   b. Rotate course set knob on course indicator to select desired course. The course deviation indicator will indicate if the aircraft is on course or to the left or right of the selected course to the TACAN beacon.

Note

If the power control switch is at T/R, the range indicator shows a reduction in mileage as the aircraft approaches the surface beacon, and shows an increase in mileage as the aircraft flies away from the beacon.

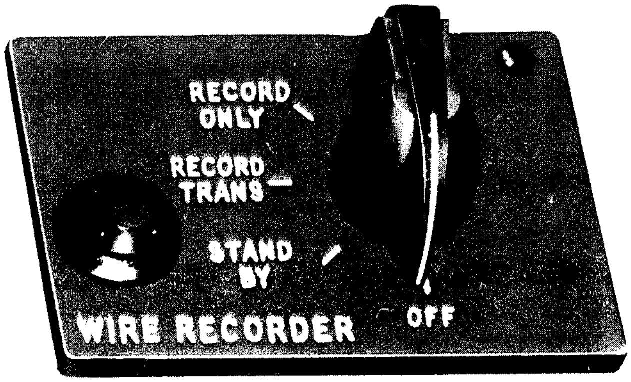
5. To turn equipment off position power control switch to OFF

RADAR SET — AN/APW-11A.

An AN/APW-11A type radar set is installed in some aircraft with AN/ARC-34 command radios. These radar sets, have a higher classification than this handbook. Refer to applicable T.O.'s for detailed information.

WIRE RECORDER

Control Panel



WIRE RECORDER.

The wire recorder is provided so that visual observation or transmitted information can be recorded for future reference and study without the necessity of making notes. The recording wire is sufficient for one hour of total elapsed recording time and is powered by the NO. 1 DC secondary bus. An indicator light is provided to show when a limited amount of recording time remains. The recorded wire spool can only be removed and played back when on the ground.

Wire Recorder Selector Switch.

Operation of the wire recorder is initiated by the rotary wire recorder selector switch located forward of the temperature and pressure control panel on the right console. The switch has four positions; OFF, STANDBY, RECORD TRANS and RECORD ONLY. When in the OFF position, the wire recorder is not energized. The STANDBY position warms up the recorder so that it is ready for operation when needed. When in the RECORD TRANS position, anything that is transmitted when the microphone button on the control stick is depressed will also be recorded. The RECORD ONLY position records only what the microphone picks up when the microphone button is depressed, but with wire recorder selector switch in this position the command equipment will not transmit.

Wire Recorder Indicator Light.

The wire recorder indicator light is a green light marked WIRE RECORDER 5 MIN WARN. When the light illuminates, it indicates that there is only 5 minutes of elapsed recording time left on the recorder.

LIGHTING EQUIPMENT.

INSTRUMENT LIGHTS.

Individual instrument ring lights are installed on each instrument and illuminate the instruments with a red light. Red instrument flood lighting is provided by a spot light mounted on each side of the canopy frame to illuminate placards and can also be used as auxiliary illumination.

Instrument Panel Light Switches.

The instrument panel lights are controlled by three rheostat switches (figure 4–10). The flight instrument light rheostat is marked FLIGHT, and controls the individual lights on all the flight instruments. The non-flight instrument light rheostat is marked NON-FLIGHT, and controls the individual lights on all non-flight instruments. The red auxiliary light rheostat is marked AUXILIARY, and controls the two spot lights mounted on the canopy frame. All rheostats have two positions OFF and BRIGHT. The FLIGHT, NON-FLIGHT and AUXILIARY lights are powered from the DC primary bus.

CONSOLE AND INSTRUMENT PANEL LIGHTS.

The left console is lighted by three incandescent lights covered with red filters and mounted above the console. The placards on the instrument panel and on the right console are lighted with incandescent lights used in conjunction with plastic panels. These lights are energized by the DC secondary bus and are controlled by a rheostat switch (12, figure 1–6) marked CONSOLE LIGHTS with three positions, OFF, DIM and BRIGHT. When placed between the DIM and BRIGHT positions on aircraft modified by [674] the landing gear selector handle light brightness is controlled.

INTERIOR AND EXTERIOR LIGHTS

Control Panel

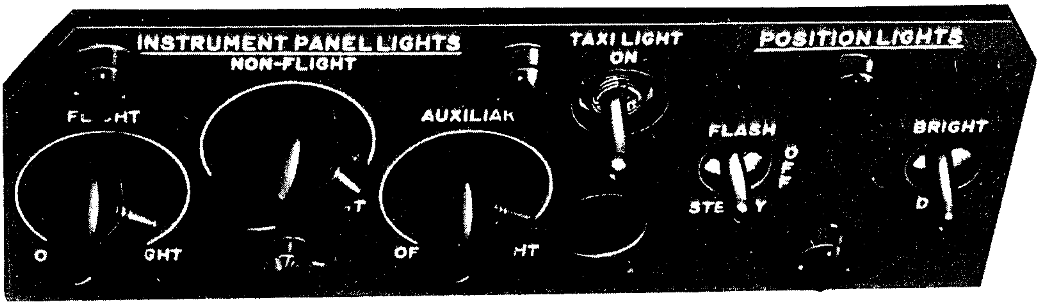


Figure 4—10.

COCKPIT LIGHT.

A type C4A cockpit light (8, figure 1—7) is mounted on the right side of the cockpit. The light is provided with an extension cord and it may be removed from its mounting bracket to be used as a portable light. The light is powered by the DC primary bus and controlled by a rheostat switch located on the light. The rheostat switch controls the intensity of the light for continuous illumination. A push-button type switch on the light may be used for intermittent light use. The light is equipped with a red filter which may be removed and the light used as a white spot light.

LANDING LIGHTS.

A landing light, mounted on an electrically retracted door is installed on each wing tip. The door and light are both actuated with power from the DC primary bus and controlled by a landing light switch.

Landing Lights Switch.

The landing light switch (8, figure 1—5) has three positions EXTEND & ON, OFF and RETRACT. The EXTEND & ON position opens the doors and turns the landing lights on. The OFF position will turn the lights off if the doors are open but the doors will not retract. The RETRACT position retracts the landing lights to the fully closed position.

TAXI LIGHTS.

The taxi light is mounted on the nose wheel strut and is powered by the DC primary bus and controlled by an ON-OFF switch (figure 4—10) located above the right console. A switch in the nose wheel well will automatically put the light out when the nose wheel is retracted.

Note

The taxi light will illuminate only when the nose gear is down and locked. Therefore, when viewed from the ground, it may be used as an indication of a safe nose gear condition.

POSITION LIGHTS.

Position lights consist of a red light on the left wing tip, a green light on the right wing tip, a yellow and a white light on the tail, and a white light on the top and the bottom of the fuselage. All lights are powered from the NO. 1 secondary bus and controlled by a position light switch (figure 4–10), having three positions, STEADY, OFF and FLASH. In the STEADY position the wing, tail, and fuselage lights will provide continuous illumination. In the FLASH position the wing and white tail lights will flash alternately with the yellow tail light, and the fuselage lights will be steady. The intensity of the position lights is controlled by a DIM-BRIGHT switch (figure 4–10) adjacent to the position light switch.

Note

On those aircraft having the new flasher installed, the amber tail light has been replaced with a white light and operation in the Flash Position changed to simultaneous flashing.

OXYGEN SYSTEM.

LIQUID OXYGEN SYSTEM.

The liquid oxygen system consists of a five liter capacity vacuum insulated container, buildup coils, check valves, relief valves and quantity gage. Liquid oxygen is stored in the vacuum container and passes into the buildup coils. Here it evaporates into gaseous oxygen and passes into the oxygen regulator at approximately 70 PSI. The oxygen system from the regulator to the pilot's oxygen mask is identical with previous aircraft. Excessive pressures in the system between the vacuum container and the regulator are relieved through the relief valves and vented overboard. A buildup and vent valve is provided for servicing the oxygen system. It is recommended that this valve be left in the vent position when the aircraft is parked as less loss of oxygen will occur with the valve in the vent position than in the buildup position.

Note

The liquid oxygen quantity gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read 5 liters, since it is impossible to charge the liquid oxygen converter to 5 liters. Use the oxygen duration chart to determine your oxygen duration for the indicated supply.

CAUTION

When installing the canopy cover, care should be exercised to avoid damage to the oxygen filler cover door and the buildup and vent valve when the valve is left in the vent position.

REGULATOR.

An automatic pressure breathing diluter-demand oxygen regulator (figure 4–11) is installed on the right console and includes a low pressure gage and a flow indicator. The regulator automatically supplies the proper mixture of oxygen and air at all altitudes with provisions for positive pressure breathing at high altitudes. Either Type D-1 or Type D-2 automatic pressure demand regulators may be installed. The Type D-2 regulator differs from the late Type D-1 regulator in that the slight positive pressure from 8,000 to 28,000 feet has been removed and a better panel and instrument lighting system has been included. The slight positive pressure has been removed to conserve oxygen. It will be noted in the oxygen duration chart that an increasing number of manhours of oxygen is available above 25,000 feet altitude when operating with the diluter lever in the NORMAL OXYGEN position. This is caused because with increasing altitude, the volume of an equivalent mass of air at sea level increases, the regulator attempts to maintain a constant mass of flow of oxygen to the lungs by increasing the oxygen flow from the oxygen system and decreasing accordingly the amount of air mixed with the oxygen. Beyond the altitude at which 100 per cent oxygen is being used, further expansion of the gas will occur and, unless a pressurized system is used, the lungs cannot expand sufficiently to absorb the normal oxygen consumption. Therefore, even with a pressurized system though not as soon, an altitude will be reached beyond which, less and less mass will be absorbed because of the continually expanding gas.

OXYGEN SYSTEM CONTROLS

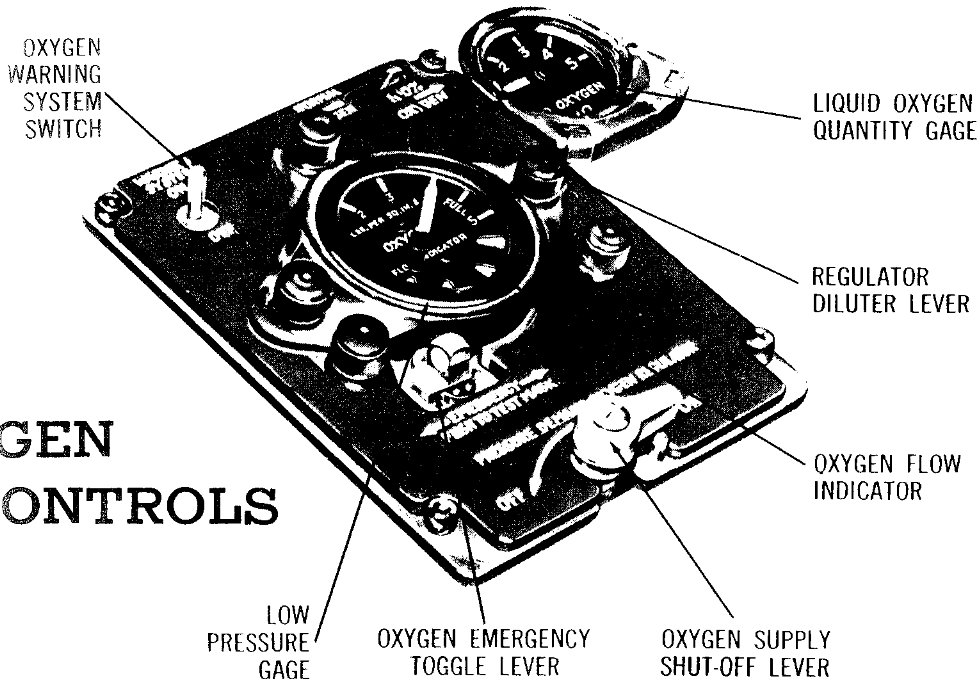


Figure 4—11.

Regulator Diluter Lever.

The regulator diluter lever (figure 4—11) has two positions, NORMAL OXYGEN and 100% OXYGEN. When the lever is in the NORMAL OXYGEN position, the regulator unit will function to provide automatic mixing of air and oxygen in sea level concentration at all altitudes. In the event that the regulator malfunctions, a pressure relief valve in the regulator unit will protect the pilot from excessive pressure. When the lever is in the 100% OXYGEN position, the automatic air-oxygen mixing feature is by-passed and 100 per cent oxygen is supplied regardless of altitude.

Oxygen Supply Shutoff Lever.

An ON-OFF oxygen supply shutoff lever (figure 4—11) is located on the aft end of the regulator. When the lever is in the ON position, system oxygen is supplied to the regulator unit. When the lever is in the OFF position the oxygen supply to the regulator is shut off. However, the supply valve is lockwired to the ON position. Soft copper wire is used so that the supply may be turned off by the pilot in an emergency

Oxygen Emergency Toggle Lever.

The oxygen emergency toggle lever (figure 4—11) is provided for emergency operation or to supply maximum pressure for leakage test of the oxygen mask. The emergency toggle lever should remain in the center position at all times, unless an unscheduled pressure increase is required. Moving the toggle lever either to the left or right of its center position to the EMERGENCY position, provides continuous positive pressure to the mask for emergency use. When the toggle lever is depressed in the center position, it provides positive pressure to test the mask for leaks.

CAUTION

When positive pressures are required, it is mandatory that the oxygen mask be well fitted to the face. Unless special precautions are taken to insure no leakage, then continued use of positive pressure under these conditions will result in the rapid depletion of the oxygen supply. In aircraft which employs liquid oxygen, this condition would result in extremly cold oxygen flowing to the mask.

CREW MEMBER OXYGEN DURATION HOURS

Upper figures indicate Diluter Lever 100% Oxygen.

Lower figures indicate Diluter Lever Normal Oxygen.

1 Crew Member
1 Type A-3 Converter

CABIN ALTITUDE FEET	GAGE QUANTITY — LITERS 5	4	3	2	1	BELOW 1
40,000	*24.0* **24.0**	*19.2* **19.2**	*14.4* **14.4**	*9.6* **9.6**	*4.8* **4.8**	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
35,000	*24.0* **24.0**	*19.2* **19.2**	*14.4* **14.4**	*9.6* **9.6**	*4.8* **4.8**	
30,000	*17.3* **17.8**	*13.8* **14.2**	*10.4* **10.6**	*6.9* **7.1**	*3.5* **3.5**	
25,000	*13.3* **16.8**	*10.6* **13.4**	*8.0* **10.0**	*5.3* **6.7**	*2.7* **3.4**	
20,000	*10.1* **19.0**	*8.1* **15.2**	*6.1* **11.4**	*4.0* **7.6**	*2.0* **3.8**	
15,000	*7.8* **23.0**	*6.2* **18.4**	*4.6* **13.8**	*3.1* **9.2**	*1.5* **4.6**	
10,000	*6.6* **23.0**	*5.3* **18.4**	*4.0* **13.8**	*2.6* **9.2**	*1.3* **4.6**	

Figure 4—12.

Low Pressure Gage.

The low pressure gage (figure 4—11) incorporated in the oxygen regulator unit records the oxygen pressure being supplied to the regulator. The gage scale reads from 0 to 500 PSI and the operating pressure is approximately 70 PSI. Tolerance and operating conditions can cause this pressure to vary between 65 and 115 PSI in the liquid oxygen system.

OXYGEN QUANTITY GAGE.

An oxygen quanity gage (figure 4—11) is installed on the right console to record the amount of liquid oxygen remaining in the vacuum container. The gage is calibrated to read from 0 to 5 liters. The quantity gage may fluctuate approximately 1 liter while taking deep breaths. Fluctuation and erratic indications may be expected during maneuvers and while flying in rough air. Excessive engine vibration, etc. may also cause fluctuation. After the vent and buildup valve is positioned in the BUILDUP position, the gage will register erratic, false indications for approximately 30 minutes. A full converter will then indicate only approximately 4½ liters due to the vapor loss resulting from heat generated during servicing. Use the oxygen duration chart to determine your oxygen duration for the indicated supply.

OXYGEN FLOW INDICATOR.

A blinker type oxygen flow indicator (figure 4—11) is incorporated in the pressure demand regulator unit. Black and luminescent segments alternately appear through four slots in the indicator face with each breath taken through the oxygen mask.

OXYGEN WARNING SYSTEM.

The oxygen warning system consisting of a warning light on the main instrument panel and an ON-OFF switch on the oxygen regulator panel has been deactivated.

PRESSURE DEMAND OXYGEN MASKS.

Only the type A-13, A-13A or MS22001 pressure demand oxygen mask will be used with the automatic pressure demand oxygen regulator. These masks can be identified by the presence of a gray anodized aluminum exhalation valve (pressure-compensating), which is located in the mask directly below the chin position. Pressure demand masks, when used at altitude, will occasionally produce a distinct vibration in the mask that can be identified by a "wheezing" sound. This condition may be overlooked, in that operational qualities are not disturbed in any manner.

If a blocking condition (cannot exhale) occurs during flight, a "sharp" exhalation will usually correct the difficulty. In the event a "sharp" exhalation does not relieve the blocking condition, the mask may be lifted off the face momentarily at the chin section. Extreme caution must be exercised in using this procedure, since the danger of hypoxia increases rapidly above 30,000 feet cockpit altitude. Oxygen masks other than those specified above will not be used with automatic pressure demand oxygen regulators. Use of unauthorized oxygen masks results in rapid depletion of the aircraft oxygen supply and pressure breathing required at altitude will be lost.

USE OF AUTOMATIC PRESSURE DEMAND REGULATOR.

1. The diluter lever will always be set at the NORMAL OXYGEN position, except in cases where noxious gases are suspected, or pre-breathing of oxygen is deemed necessary. These exceptions are rare, and if the diluter lever is placed in the 100% OXYGEN position, extreme care must be exercised in monitoring the oxygen supply.

2. Turn oxygen supply shutoff lever to the ON position if not already safety wired in the ON position.

3. Press oxygen emergency toggle lever straight in to test mask for leakage at any altitude. Place lever to right or left of the normal off position to provide an increased flow of oxygen only in case of emergency. If emergency use is necessary, however, extreme caution must be used to prevent rapid loss of system pressure through the emergency valve.

4. As breathing through the mask is started, the oxygen flow indicator should start functioning. The proper flow of oxygen will be automatically maintained by the regulator unit.

5. At approximately 10,000 feet (cockpit pressure) the D-1 regulator will supply a "SAFETY PRESSURE" or continuous flow if the pilot's oxygen mask is off or loose fitting. This pressure increases with altitude and will occur in both the NORMAL and 100% OXYGEN positions. The D-2 oxygen regulator has the same characteristics as the D-1 except that the "SAFETY PRESSURE" comes on at approximately 30,000 feet (cockpit pressure).

6. Blow into the mask if the "SAFETY PRESSURE" or flow occurs at cockpit altitudes below those shown above; if the leakage stops the regulator is satisfactory. Numerous instances of the presence of carbon monoxide in the flight comparement of jet aircraft have been suspected. Some of these instances have been brought to light through accident investigations. There are various possibilities by which carbon monoxide may enter the compartment during ground operation; however, as yet, neither the exact concentration nor the exact sources have been determined, except as indicated below. Consequently, the following instructions should be compiled with. If the subject aircraft is to be operated under possible conditions of carbon monoxide contamination, such as during "runup" or taxiing directly behind another operating jet aircraft or during "runup" with its tail into the wind the following procedure shall be used:

1. Before starting engine, don oxygen mask, connect hose to oxygen regulator, and place diluter lever at the 100% OXYGEN position.

2. Whenever contamination is suspected, 100 per cent oxygen will be used during ground operation and take-off.

3. After contamination is no longer suspected, place the diluter lever to the oxygen regulator at the NORMAL OXYGEN position.

WARNING

The oxygen diluter lever must be returned to the NORMAL OXYGEN position as soon as possible because the use of 100 per cent oxygen throughout a long mission will so deplete the oxygen supply as to be hazardous to the pilot.

Note

See figure 4-12 for oxygen system duration.

OXYGEN SYSTEM CHECK.

All Flights.

With the liquid oxygen system, the liquid quantity gage should indicate between 4 and 5 liters and the pressure gage on the regulator should read approximately 70 PSI. Tolerance and operating conditions can cause this pressure to vary between 65 and 115 PSI. There should be no evidence of steady venting from the oxygen vent port.

OXYGEN MASK CONNECTION

PROPER ATTACHMENT OF THE OXYGEN HOSE CONNECTION is extremely important to assure that the oxygen hose does not —

a. become accidently disconnected during flight causing a loss of oxygen supply.

b. prevent quick separation from the seat during ejection

c. flail during ejection causing pilot injury

MC-3A

1. Attach oxygen mask hose (male connector) to parachute harness chest strap by wrapping mask connector tie-down strap underneath and up behind harness chest strap twice, then snapping.

WARNING

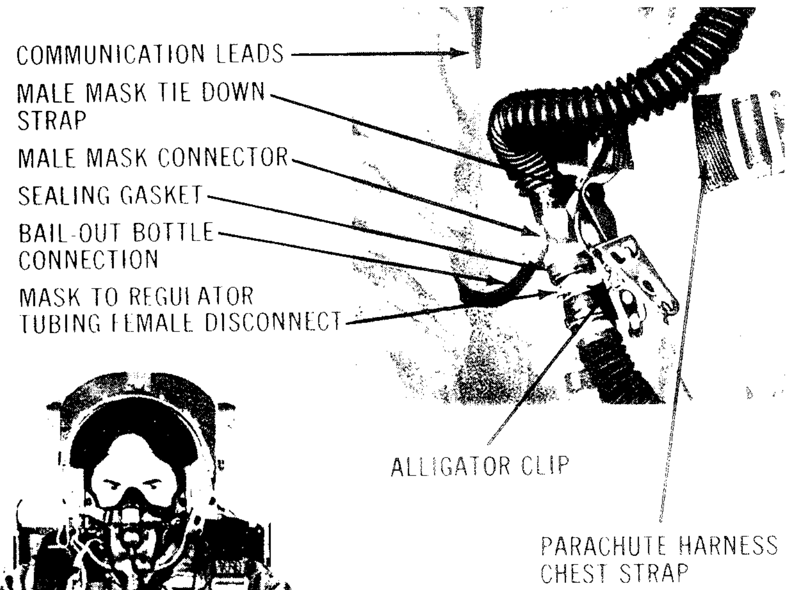
- *Failure to double-loop tie-down strap around chest strap may permit the tie-down strap to slip into and open the chest strap during ejection.*
- *Do not wrap the tie-down strap around the chest strap snap.*

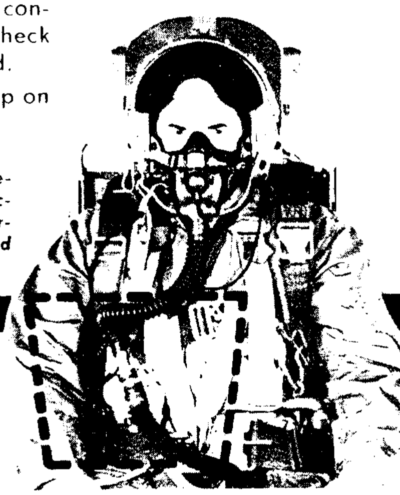
2. Attach the mask to regulator hose female disconnect to the mask male connector. Listen for click and visually check that sealing gasket is only half exposed.

3. Fasten alligator clip as close to snap on tie-down strap as possible.

WARNING

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





CRU-8/P

1. Insert connector into the mounting plate attached to the parachute harness. Check that the connector is firmly attached and that the lockpin is locked.

2. Insert male bayonet connector, on the end of the oxygen mask hose, into the female receiving port of the CRU-8/P connector. Turn bayonet connector to lock prongs into the recess in the lip of receiving port.

3. Couple the seat oxygen hose to the lower port of the connector.

4. Snap the strap attached to the seat oxygen hose onto the CRU-8/P connector.

WARNING

The CRU-8/P system should not have an alligator clip on the seat hose. If one is installed, it shall not be used.

5. Attach the bail-out bottle hose to the swiveling port of the connector by inserting the male coupling of the bail-out bottle hose and turning it clockwise against the spring-loaded collar.

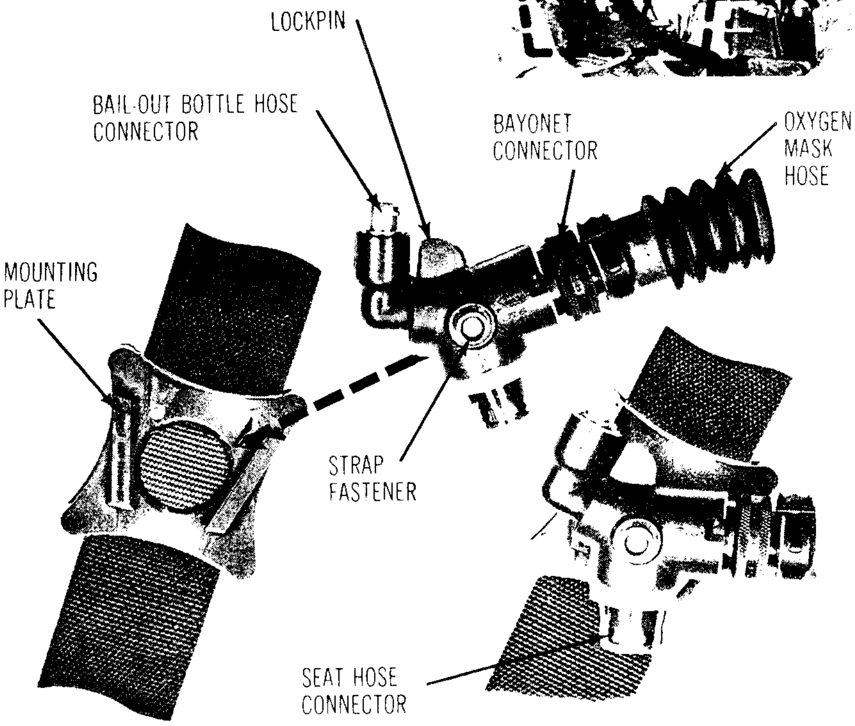


Figure 4—13

Oxygen Hose Connection.

WARNING

PROPER ATTACHMENT OF THE OXYGEN HOSE CONNECTION is extremely important to assure that the oxygen hose does not —

a. Become accidentaly disconnected during flight causing a loss of oxygen supply.

b. Prevent quick separation from the seat during ejection.

c. Flail during ejection causing pilot injury.

MC-3A Connector — UNMODIFIED

1. Attach oxygen mask hose (male connector) to parachute harness chest strap by wrapping mask connector tie-down strap underneath and up behind harness chest strap twice, then snapping.

WARNING

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

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

2. Attach the mask to regulator hose female disconnect to the mask male connector. Listen for click and visually check that sealing gasket is only half exposed.

3. Fasten alligator clip as close to snap on tie-down strap as possible.

WARNING

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

CRU-8/P Connector. — MODIFIED

1. Insert connector into connector mounting plate attached to parachute harness. Check that connector is firmly attached and that lockpin is locked.

2. Insert male bayonet connector, on end of oxygen mask hose, into female receiving port of connector, and turn connector to lock its prongs into recess in lip of receiving port.

3. Couple mask to regulator hose to lower part of connector.

4. Attach bail-out bottle hose to swivel port of connector by inserting male coupling of bail-out bottle hose and turning it clockwise against spring-loaded collar.

EMERGENCY OPERATION.

1. Should symptoms of hypoxia be suspected, or if smoke or fumes should enter the cockpit, immediately place the emergency toggle lever in the emergency position. After determining that sufficient oxygen is being received, revert to 100 per cent oxygen, by placing regulator diluter lever in the 100% OXYGEN position and turning the emergency toggle lever off. If it is then ascertained that the 100 per cent oxygen position provides sufficient oxygen, check the oxygen equipment to determine if the normal setting of the regulator diluter lever may be used. If so, place the diluter lever in the NORMAL OXYGEN position.

WARNING

No attempt should be made for a normal oxygen supply setting if smoke and fumes are present in the cockpit. In the event that the system is in the EMERGENCY or 100 per cent oxygen position, extreme care must be exercised in monitoring the oxygen supply.

2. In the event of accidential loss of cockpit pressure, no action is required if oxygen is being used, as the regulator unit will automatically compensate for the increased cockpit altitude.

3. If the oxygen regulator should become inoperative, pull the cord of the H-2 emergency oxygen cylinder, and descend to a cockpit altitude not requiring oxygen.

AIR REFUELING SYSTEMS.

The aircraft is equipped with a receiver type or boom-probe type air refueling system (figure 4—14). On aircraft modified by [R557] a boom-probe type air refueling system is installed to accept fuel from a drogue equipped tanker aircraft. The aircraft is refueled through a probe on the end of a boom extending from the leading edge of the left wing. On aircraft not modified by [R557] a receiver type air refueling system is installed to accept fuel from a boom equipped tanker aircraft. The aircraft is refueled through a receiver on the left wing upper surface. The receiver is concealed by a hydraulically operated flush type door. Common to both air refueling systems are the fuel shut-off valve switches to control fuel flow to the individual internal tanks or to isolate tanks in the event of battle damage. Refer to FUEL SHUT-OFF VALVE switches below for details on the operation of these switches.

CAUTION

Do not perform inflight refueling with MA-2 nozzles, Parker PN 1325-556659 and Shultz PN 11-257-1.

FUEL SHUT-OFF VALVE SWITCHES.

The same fuel lines are used for transfer of fuel from the external to the internal tanks as are used to supply all tanks during air refueling. The fuel shut-off switches (figure 4—15) provide a means of controlling fuel flow to the individual internal tanks by closing valves at the fuel line entrance to each tank. Each of the four fuel shut-off valves is controlled by a two-position toggle switch. The switches are marked L WING, MAIN, FWD and R WING with an arrow indicating the CLOSED position. During normal operation the switches are left in the up position, which allows fuel to transfer from the external to the internal tanks and also allows all tanks to be refueled from the refueling receptacle or the refueling probe. By placing a fuel shut-off valve in the CLOSED position the respective fuel tank will not receive fuel by transfer or air refueling although a slight internal leakage is normal. When the main fuel shut-off switch is in the CLOSED position both valves in the main tank are closed and fuel from neither internal nor external tanks can be transferred. This system is provided to isolate each of the internal tanks, in the event of battle damage to the tank, or failure of the booster pump in the tank. The fuel shut-off valves are actuated by the DC primary bus.

RECEIVER AIR REFUELING SYSTEM.

The aircraft is equipped with an air refueling system (figure 4—14) which enables the aircraft to be refueled in the air from a tanker using a flying boom. The receiver is located on the left wing upper surface and is concealed by a flush type door, which is hydraulically operated. During the refueling operation, the engine is operated with the fuel tank selector in the ALL TANKS position, so that the engine is fed from the main tank. The forward, wing, main and pylon tanks are filled during the refueling cycle. Once the receiver door in the wing is open and the tanker's boom is inserted in the nozzle, refueling sequence is accomplished electrically through an amplifier, which is powered from the primary bus. When the tanks are full, the fuel flow in the refueling lines is reduced to an amount equal to the engine consumption and the fuel pressure in the lines increases. These changes are noted in the tanker aircraft and a disconnect is effected. An automatic disconnect will be accomplished if the fuel pressure is excessive in the refueling lines, if rough air causes excessive tension on nozzle, or by any uncontrolled or intentional change in flight attitude of the receiver aircraft wherein a conical angle of 15 degrees from the normal is exceeded. Provision is made so that the forward, main or wing tanks can be isloated, if damaged, from the refueling system. In the event the main tank is damaged and is shut off during refueling, operation can be accomplished with the fuel tank selector in the WING AUX or FWD AUX position. On 5 thru 25 aircraft modified by [501] and all 30 and later aircraft an air refueling amplifier override switch (figure 4—15) provides an emergency method of controlling hook-up and release of the tanker boom in the event of amplifier failure or malfunction.

Receiver Switch. RECEIVER

The air refueling receiver switch (figure 4—15) is a two-position toggle switch marked OPEN and an unmarked position. The OPEN position unlocks and opens the receiver door and retracts the nozzle latches hydraulically, opens the pylon tank vent valves and cuts off power to the tank-pressurization circuit, and also supplies power from the DC primary bus to the refueling amplifier, an electronic device that automatically controls the air refueling cycle by control

AIR REFUELING SYSTEM

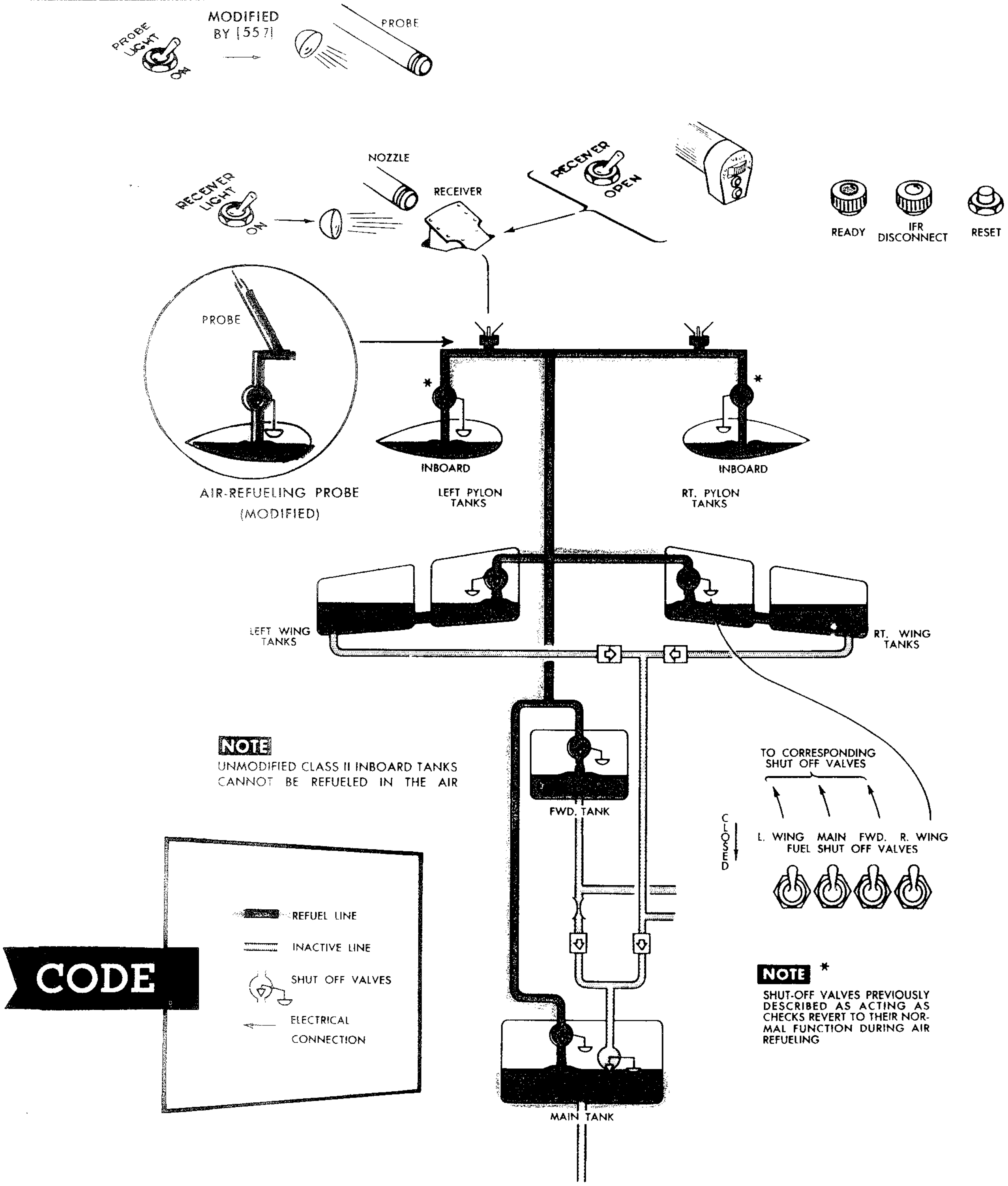


Figure 4-14

signals from the tanker aircraft received through the signal coil in the nozzle. The ready indicator light will illuminate when the receiver door is opened. The unmarked position closes and locks the receiver door hydraulically, closes the pylon tank vent valves, restores power to the tank-pressurization circuit and disconnects the power supply to the refueling amplifier.

Nozzle Disconnect Switch. **RECEIVER**

The gun-bomb-rocket sight caging switch (figure 1—10) on the throttle control, is used as the nozzle disconnect switch when refueling in the air. The caging switch is in the air refueling circuit only when the receiver switch is in the OPEN position. The switch is depressed to AIR REFUEL DISCONNECT if it is desired to end the refueling cycle before the fuel tanks are full. Depressing the nozzle disconnect switch illuminates the disconnect indicator light, causes the tanker pumps to shut down, closes the fuel valves, signals the tanker operator that a disconnect has been made and releases the refueling boom from the receiver nozzle.

Amplifier Override Switch. **RECEIVER**

An air refueling amplifier override switch (figure 4—15) is provided as an emergency method of controlling hook-up and release of the tanker boom in the event of amplifier failure or malfunction on 30 and 35 aircraft, and 5 thru 25 aircraft modified by [501]. The amplifier override switch is a two-position switch marked NORMAL and AMPLIFIER OVERRIDE. During the normal refueling cycle, the override switch is in the NORMAL position and air refueling system power and actuating signals function automatically through the amplifier and the entire air refueling sequence is an automatic operation after contact is made. If the amplifier should fail with all amplifier relays in the de-energized position a normal hook-up can be made but the start signal will not be transmitted to the tanker and the ready indicator light will remain illuminated after contact is made. When this occurs the override switch should be actuated to the AMPLIFIER OVERRIDE position to ready the circuit for disconnect. Disconnect will be accomplished by depressing the disconnect switch on the throttle. If relays NO. 1 and NO. 2 in the amplifier are stuck in the energized position the ready light will not illuminate. If the thyraton tube and remaining relays in the amplifier are functioning normally an immediate disconnect indication will occur. The override switch must be in the AMPLIFIER OVERRIDE position, in this event, to accomplish a hook-up, and automatic disconnect is not possible. Disconnect must be accomplished by depressing the disconnect switch on the throttle.

Reset Switch. **RECEIVER**

The air refueling reset switch (figure 4—15) is a push button switch spring-loaded to the off position. If at any time during a refueling cycle the aircraft becomes disconnected and the disconnect indicator light illuminates the aircraft is made ready for refueling again by depressing the reset switch to the RESET position. The refueling system can also be made ready for refueling by closing then reopening the receiver door.

Note

> Holding the reset button in the depressed position during nozzle contact in a refueling operation will cut off power to the refueling amplifier and cause a disconnect.

Receiver Light Switch. **RECEIVER**

The air refueling receiver light switch (figure 4—15) is a toggle switch with an ON and an unmarked position. The ON position supplies power from the DC primary bus to illuminate a flood light on the side of the aircraft which lights up the receiver nozzle to aid the boom operator in the tanker during night refueling operations. The light can also be used as a formation light at night.

Ready Indicator. **RECEIVER**

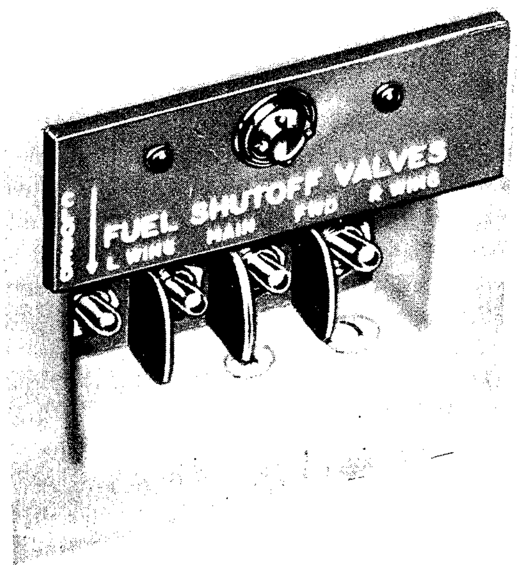
The ready indicator light (figure 4—15) powered by the primary bus is a green light marked READY and when illuminated indicates that the receiver nozzle is open, power is supplied to the refueling amplifier and the amplifier is ready for the refueling cycle. The ready indicator light will go out when contact is made.

Disconnect Indicator. **RECEIVER**

The disconnect indicator (figure 4—15) powered by the primary bus is an amber light marked AIR REFUEL DISCONNECT. Illumination indicates that the boom nozzle has been disconnected from the receiver. If the disconnect indicator is illuminated the reset switch must be depressed or the receiver door closed and reopened before the refueling system will be ready to make another refueling cycle.

AIR REFUELING CONTROLS

5 thru 30 Aircraft
MODIFIED BY [557] & ALL
40 & LATER AIRCRAFT



NOTE

AMPLIFIER OVERRIDE SWITCH
ON 5 THRU 25 AIRCRAFT MODIFIED
BY [501] & ALL 30 & 35 AIRCRAFT

Figure 4-15

BOOM PROBE REFUELING SYSTEM.

A boom-probe air refueling system is installed to accept fuel from a drogue equipped tanker aircraft which will automatically distribute it to all internal and external tanks. The system consists of a refueling probe with an engaging nozzle extending from the leading edge of the left wing, a series of tubes connected together inside the wing and fuselage to direct fuel to the various external and internal tanks, a spotlight in the wing, to illuminate the probe, and a ready, and probe light switch in the cockpit.

The probe-drogue air refueling system is put into operation automatically when the probe assembly nozzle and the tanker drogue (receptacle) are fully engaged and the ready switch is placed in the up position to vent the external fuel tanks in the aircraft. On contact, the fuel begins flowing through the probe and refueling transfer system and then through the flapper valve to the internal and external tanks. The distribution of fuel and operation of the float operated valves and switches is the same as with the boom-receptacle air refueling system. Upon completion of the probe drogue air refueling operation, the fuel flow is automatically stopped by the back-up pressure of fuel. The pilot of the probe aircraft then throttles back to reduce speed and the probe assembly releases from the drogue.

If required, the pilot can at any time accomplish a forceable disconnect by reducing speed. The locking mechanism will release under the drag load of the receiving airplane. If a night refueling operation is required, the light switch on the left console is placed in the up position to illuminate the probe by the spotlight in the leading edge of the left wing.

Ready Switch. **BOOM-PROBE**

The ready switch (figure 4—15) is a two-position toggle switch marked READY, and an unmarked position. The READY position vents the external fuel tanks, permitting the fuel to flow through the probe and refueling transfer system to the internal and external tanks. The unmarked position closes the vent.

Probe Light Switch. **BOOM-PROBE**

The probe light switch (figure 4—15) on the left console is a toggle switch with an ON and an unmarked position. The ON position supplies power from the primary bus to illuminate the probe by spotlight in the leading edge of the left wing, to aid the tanker drogue operator during night refueling operations.

AIR REFUELING SYSTEM OPERATION.

Refer to Air Refueling Technical Order (T.O.1-1C-1 and supplements).

AUTOMATIC PILOT.

An MB-2 automatic pilot is installed in some aircraft. The autopilot will automatically hold the aircraft on any predetermined course and altitude, change course at will with an exact coordinated turn, maintain the aircraft at a selected altitude and pitch trim in laterally straight level flight or at a desired angle of climb or dive up to 45 degrees, and hold a fixed radius in climbing turns and dives. Changes in airspeed are automatically compensated for and control surface displacement is regulated in relation to airspeed and altitude. Automatic control originates in an AC powered gyroscopic unit which includes a vertical and a directional gyro as references. The vertical gyro establishes a flight reference about the lateral and longitudinal axis of the aircraft while the directional gyro establishes a reference for the aircraft's directional course in relation to the earth's magnetic field. Error signals are transmitted from both the vertical and directional gyros to the autopilot system for any deviation of the aircraft in flight. The autopilot compensates and corrects for these errors through the flight controls. An automatic altitude control senses changes in pressure and when engaged will maintain the aircraft at a constant altitude. The automatic pilot can be easily overpowered manually at any time by the human pilot or it can be immediately disconnected by means of the autopilot release switch. If the autopilot is engaged with the aircraft in a wing-level climb or dive, the aircraft will continue on course in the climb or dive. However, if the aircraft is in a turn when the autopilot is engaged, a rapid roll to level attitude will take place as there is no roll synchronization.

INVERTER SWITCHES.

The inverter switches (12, figure 1—4) are two-position switches placarded MAIN, ON and OFF; and ALT, ON and OFF. The MAIN inverter switch must be ON for autopilot operation. These switches are described in detail in Section I.

FLIGHT CONTROLLER.

All control functions of the autopilot are centered about the flight controller (figure 4—16) which contains the autopilot engaging switch, the roll trim and pitch trim wheels and the turn knob. An automatic interlocking system is provided which prevents the autopilot from being turned on until it is warmed up, or if the "turn" knob is out of the neutral detent position. There is also an electrical circuit provided to automatically disengage the autopilot if the emergency hydraulic pressure pump switch is turned on.

Autopilot Trim Indicator.

The trim indicators (figure 4—16) operate only when the autopilot is engaged. The three indicators (Heading, Roll, and Pitch) indicate control surface displacement during flight with the autopilot operating.

Autopilot Engaging Switch.

The autopilot engaging switch (figure 4—16) is a rotary two-position switch marked OFF and ON. The ON position locks the autopilot to the flight control system. The engaging switch cannot be placed in the ON position unless the MAIN INVERTER switch is ON and power is supplied to the autopilot for approximately three minutes allowing sufficient time for the gyro to reach full RPM and stabilize. It is also impossible to select the ON position if the turn knob is out of the neutral detent position or the emergency hydraulic pump switch is ON. The engaging switch will automatically return to the OFF position if the AC or DC power supply fails, if the main inverter switch is positioned to the OFF position, if the emergency hydraulic pump switch is turned ON, or if the autopilot release switch on the control stick is depressed.

Pitch Trim Wheel.

The pitch trim wheel (figure 4—16), located on the left and right side of the flight controller and marked DN and UP, controls the pitch attitude of the aircraft. If either pitch trim wheel is rotated aft for nose up and forward for nose down trim, the aircraft will maintain the selected attitude. The pitch trim is limited to a climb or dive angle of approximately 45 ($\pm$5) degrees from the horizontal plane. The wheels may be rotated as rapidly as required, consistent with structural limitations and pilot comfort.

Roll Trim Wheel.

The roll trim wheel (figure 4—16) marked ROLL TRIM, controls the lateral trim of the aircraft, if the roll trim wheel is rotated clockwise for right wing down or counterclockwise for left wing down, the aircraft will maintain a selected trim. The roll trim is limited to approximately $\pm$5 degrees from level flight.

Turn Knob.

The turn knob (figure 4—16) is a rotary switch marked LIFT TO TURN and has two extreme positions to the left and right of neutral marked L and R. If the turn knob is in the neutral detent position, it must be lifted before it can be turned to the L or R positions. When rotated to the L or R position, the aircraft will make a coordinated turn to the left or right. The aircraft will lock onto the directional heading it is taking at the time the turn knob is returned to the detent position. The bank angle is governed by the degree of right or left rotation of the turn knob from the neutral position. The bank angle is held to a maximum of 45 ($\pm$5) degrees from wing level flight position. The autopilot cannot be engaged if the turn knob is out of the neutral detent position.

Note

When flying an autopilot controlled turn, hold the flight controller turn knob out of detent until the aircraft returns to wings-level attitude on the new course. Placing the turn knob in the detent position too soon will cause overshoot and skid condition until return to the "lock-on" course.

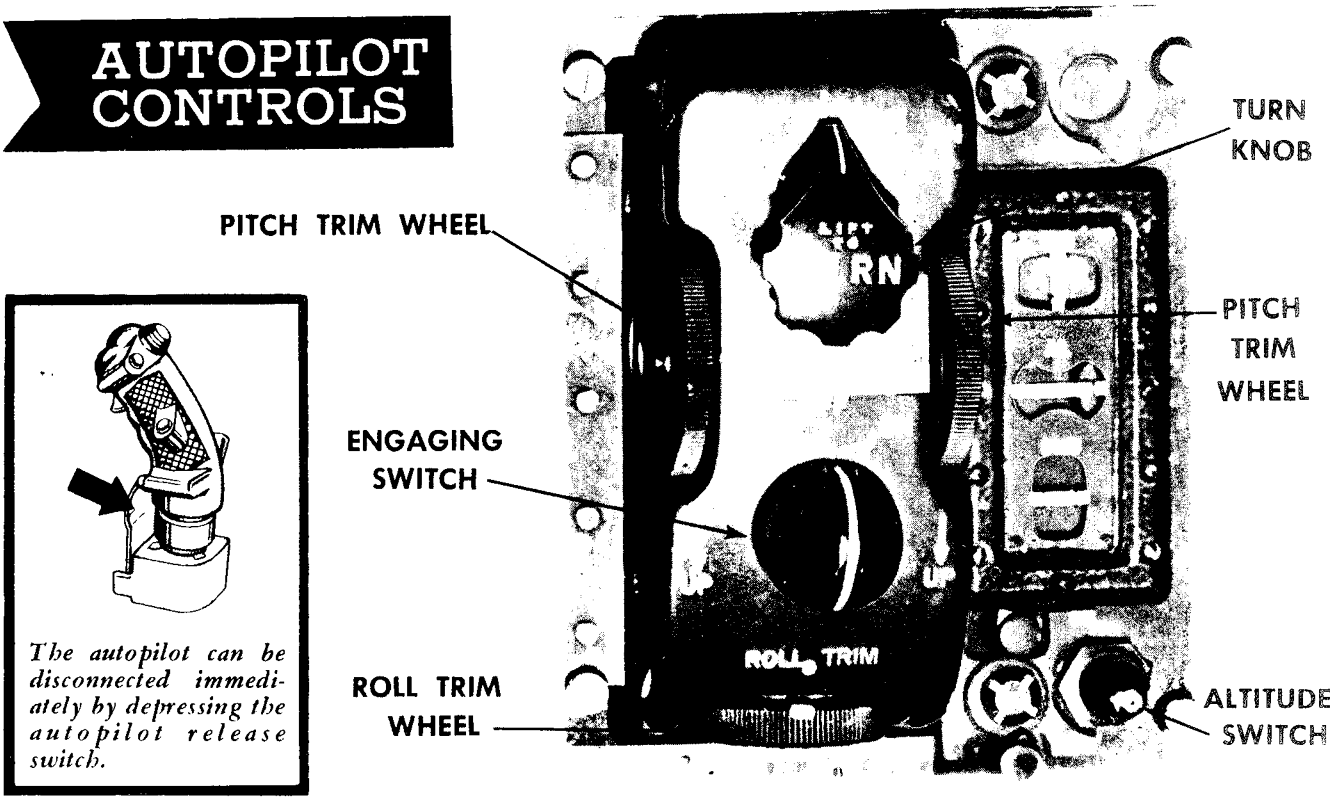


Figure 4-16.

AUTOPILOT RELEASE SWITCH.

An autopilot release switch (figure 4—16) is installed on the forward side of the control stick. This is a spring-loaded switch and when depressed, automatically disengages the autopilot from the control system. The autopilot engaging switch, on the flight control, will automatically return to the OFF position when the release switch is depressed.

ALTITUDE CONTROL SWITCH.

The automatic control consists of a solenoid held toggle switch (figure 4—16), with placarded positions ALTITUDE ON and ALTITUDE OFF. When the switch is at ALTITUDE ON the automatic altitude control will hold the aircraft to within ±30 feet or ±10 per cent of indicated airspeed in feet, whichever is greater in level flight. If engaged in a climb or dive up to 1,500 FPM, the aircraft will be stabilized on engaged altitude within ±100 feet. During turns, the altitude should be held to within ±100 feet. The automatic altitude control can be switched off by placing the altitude control switch at ALTITUDE OFF or by disconnecting the autopilot from the control system.

GROUND TEST BEFORE TAXIING.

WARNING

- Autopilot should be given a thorough check before each flight.
- No aircraft with an autopilot write-up should be flown, unless the autopilot is electrically disconnected by pulling the two autopilot circuit breakers in the cockpit, bearing legend AUTOPILOT.

To insure proper operation while airborne, the autopilot should be given a thorough check prior to each flight as follows:

1. With flight controls in neutral and turn knob in detent, turn engaging switch to ON. When the autopilot is engaged on the ground, a slight oscillation of the control may be expected. This is normal and should not be encountered with air loads on the control surfaces when in flight. There should be no appreciable movement of the control stick or rudder pedals. A very slight momentary "kick" is permissible but there should be no movement to a new position. There may be lateral movement of the control stick.

Note

The engaging switch cannot be moved to the ON position unless the MAIN INVERTER switch is in the ON position, and the primary and secondary bus are energized for approximately 3 minutes.

2. Rotate roll trim wheel in each direction and check control stick for corresponding movement. Return ailerons to neutral by rotating roll trim wheel as required.

3. Lift turn knob out of neutral detent. There should be no appreciable movement of the control stick or rudder pedals. Rotate turn knob in each direction and check control stick for corresponding movement. There should be no perceptible rudder movement. Disengage the autopilot.

4. Trim control stick forward approximately two inches, wait five seconds then engage the autopilot. There should be no movement of the stick in pitch. Disengage autopilot. Trim control stick aft approximately two inches, wait five seconds then engage the autopilot. There should be no movement of stick in pitch.

WARNING

If abrupt stick movement occurs when the autopilot is engaged during this pitch synchronization check, the discrepancy must be corrected prior to flight or the autopilot deactivated by pulling the autopilot AC fuse, and DC circuit breaker.

Rotate pitch trim wheels in each direction and check control stick for corresponding movement. Return control stick to neutral using pitch trim wheels.

5. With autopilot still engaged, apply overpower momentarily to control stick in all directions and to rudder pedals in either direction. Positive opposition from the autopilot should be felt as overpower force is applied. Controls should return to the original position when overpower force is released.

CAUTION

The overpower check should be performed as rapidly as possible to avoid overheating the clutch mechanism. The clutch should not be forced to slip continuously for periods longer than 20 seconds out of every minute. In the pitch axis the overpower should be held only for a few seconds as the automatic trim will cause the trim to oppose the overpower, and if sustained will give full opposing trim.

6. Actuate altitude control switch to ON. Stick movement should not exceed ½ inch. Altitude control switch should remain ON.

7. Disengage the autopilot by depressing the release switch on the control stick. Autopilot engaging switch (on flight controller) should return to the OFF position. Manually check flight controls for correct operation and freedom of movement.

Note

As a safety feature, the automatic pilot will disengage if the emergency hydraulic pump switch is placed in the ON position. After making this check, manually recheck the flight controls for correct operation and freedom of movement.

WARNING

During preflight check out operations, automatic trim is in operation. At the conclusion of the preflight check out, the controls will be at an undetermined position that may include either extreme. Trim must be checked and reset before takeoff. Aileron and rudder neutral position can be determined by the aileron and rudder neutral trim indicator lights. The control stick should be centered for stabilator neutral.

INFLIGHT OPERATION.

Normal Engagement.

1. Trim the aircraft for wing level flight, desired pitch trim and directional heading. Maintain stabilized flight for a few seconds.

Note

Automatic synchronization in pitch and in yaw permits the autopilot to be engaged on any heading and in any pitch attitude up to 45 degrees from level flight, without aircraft attitude change. If engagement is made when the aircraft is not in a wing level attitude, the aircraft immediately will assume a near wing level attitude upon engagement.

2. Prepare to monitor the aircraft controls and to overpower the autopilot, if necessary, in case engagement should cause a sudden attitude change as a result of autopilot malfunction during manual flight.

3. Position the engaging switch to ON.

CAUTION

Do not engage the autopilot at speeds in excess of 320 KIAS at altitudes below 27,500 feet.

4. After engagement, momentarily check overpower of the autopilot in the yaw, roll and pitch axes. Aircraft should return to reference attitude. Control operations should be made at the flight controller while the autopilot is engaged.

CAUTION

The aircraft controls should be monitored in autopilot flight whenever in proximity to the ground or other aircraft.

5. Lift the turn knob and adjust the roll trim for wing level flight.

Note

A small wing down error will cause the aircraft to turn slightly until balanced by rudder deflection.

6. If necessary, correct the pitch attitude to that desired by means of the pitch trim wheels.

7. For desired change in course, turn knob will produce a coordinated turn up to 45 (±5) degree bank.

Engagement of Automatic Altitude Control.

Automatic altitude control is engaged by actuating the altitude control switch to ON. The altitude control switch should remain engaged in the ON position.

Note

Automatic altitude control can be engaged in dive or climb. The aircraft will return to and hold the altitude at which the switch is operated. Smoothest "lock-on" of altitude control is made with the aircraft in level flight at the desired altitude.

CAUTION

If the altitude control is engaged in a steep dive or climb, the clutch in the altitude unit may slip and the aircraft will level at an altitude below or above the desired altitude at some discomfort to the pilot.

Disengagement of Autopilot.

1. Allow the autopilot to fly the aircraft on a straight course for a short time following any autopilot controlled maneuvers. This will insure that the autopilot circuits have stabilized on the present altitude and heading.

2. Monitor the aircraft controls, prepare to take immediate corrective action if necessary.

3. Place the engaging switch in the OFF position.

Note

The autopilot can be disengaged by depressing the release switch on the control stick. The engaging switch will automatically return, to the OFF position.

Emergency Operation.

In the event any of the following malfunctions occur during flight, corrective action must be taken immediately.

a. Unusually heavy stick forces.

b. Aircraft difficult to control.

c. Jerky or erratic controls.

d. Inadvertent engagement.

WARNING

The autopilot may possibly be engaged due to short circuits, even though circuit breakers are pulled.

Corrective Action.

1. Immediately depress the autopilot release switch on the control stick or place the autopilot engaging switch in the OFF position.

2. MAIN INVERTER switch – OFF.

3. ALT INVERTER switch – ON.

4. In the event that G forces prevent the placing of the MAIN INVERTER switch OFF, place the emergency hydraulic pump switch to EMER PUMP or EMERG until G forces are relieved and the MAIN INVERTER switch is placed at OFF and the ALT inverter switch is placed at ON. Return emergency hydraulic pump switch OFF.

5. If one of the above actions does not correct the malfunction and the aircraft is under control, pull the AUTOPILOT circuit breaker located on the right circuit breaker panel.

WARNING

Autopilot may possibly be engaged due to short circuits, even though circuit breakers are pulled.

6. Place the ALT inverter switch OFF and the MAIN inverter switch ON.

7. Land the aircraft as soon as possible.

NAVIGATION EQUIPMENT.

HEADING INDICATOR.

A type J-2 heading indicator is installed in the aircraft which provides visual indication of the magnetic heading of the aircraft. The indication is read on an indicator (2, figure 1–4) whose operation is governed by a GYRO whose spin axis is stabilized in a horizontal plane by means of a leveling device and whose orientation in azimuth is slaved to the earth's magnetic meridian by a direction-sensing component, located at the left stabilizer. The indicator requires both AC and DC power. The DC power is supplied from the primary bus and the AC power is supplied by the main or alternate inverter. The gyro is free to operate within 85 degrees from level flight in dive and climb, and in right and left bank. At the limits, it strikes mechanical stops, which render the indications on the directional gyro control and the settable dial indicator inaccurate. After return to level flight, errors up to five degrees in heading may be introduced; but the gyro will recover its erect and slaved positions automatically, in a period of five minutes or less, and thereafter will again resume correct indications until the limits are again exceeded. The flux valve unit of the remote compass transmitter remains pendulous through 30 degrees on both sides of the vertical, in pitch and roll. When these limits are exceeded, or a coordinated turn is being executed, the vertical components of the earth's field are picked up which results in flash signals. Restoration of the aircraft to an attitude within these limits renders the flux valve unit pendulous again, and it automatically resumes correct sensing. A thermal switch in the amplifier provides fast slaving and leveling of the directional gyro, during the initial operation of the indicator.

Heading Indicator Slaving Switch.

The heading indicator slaving switch (figure 4–17) has two positions: NORMAL and CUTOUT. The NORMAL position supplies power to the heating, leveling and slaving systems. The CUTOUT position cuts off the power supply to the control field of the slaving torque motor, and is used when the horizontal lines of magnetic force dip excessively at the higher latitudes.

Heading Indicator Fast Slaving Switch.

The heading indicator fast slaving switch (figure 4–17) marked PUSH FOR FAST SLAVING, is pushed in momentarily to shorten the time required to restore the gyro to its erect and slaved position, after level flight is resumed, following maneuvers in which the gyro has hit the mechanical stops. Approximately three minutes of fast slaving is obtained by depressing the fast slaving switch. The fast slaving system is automatically energized when both AC and DC power are supplied.

CAUTION

The fast slaving switch shall not be operated more than once in ten minutes. More frequent use of the switch will damage the torque motors and make the indicator inoperative or inaccurate.

Note

After the fast slaving switch is pressed, a time delay circuit maintains the fast slave action for approximately three minutes. During this three minute interval, any maneuvering of the aircraft can induce errors into the equipment. At the completion of the three minutes, the system normally reverts to slow slave and the large errors which have

HEADING INDICATOR Controls

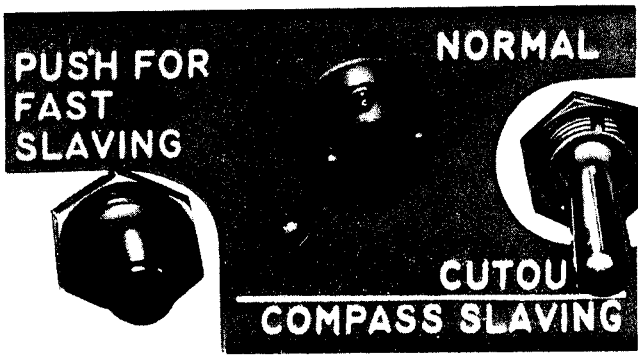


Figure 4—17.

been introduced will remain for a considerable time. Therefore, the fast slaving switch should not be used during flight, except when the aircraft can be maintained in straight flight for at least three minutes after the fast slaving switch is depressed.

Starting.

The indicator will operate if the engine is operating and the MAIN, or ALT inverter switch is ON, if the engine is inoperative and the battery switch is positioned to ON and the ALT INVERTER switch is ON, or if the primary and secondary busses are energized through the external power receptacle and the MAIN INVERTER switch is ON. Allow three minutes to elapse so that the gyro in the directional gyro control comes up to operating speed, levels and aligns the indication on the settable dial indicator with that sensed by the remote compass transmitter.

Operation.

Setting Heading Indicator.

By means of the compass card set knob on the heading indicator, set the compass card for the heading it is desired to fly. It is preferable to set the compass card against the top index of the indicator, although any index may be chosen.

Using the Heading Indicator — Straight Flight.

After the aircraft becomes airborne, the indicator is referred to in the same manner as a magnetic compass.

Using the Heading Indicator — In Turns.

Perfect 45, 90 and 180 degree turns can be executed by setting the compass card, with the overlapping pointer against the top index, then flying the aircraft to align the pointer with the 45 and 90 degree bezel indices on both sides of the top index, or with the index at 180 degrees. The final heading may be set against the top index by means of the compass card set knob. Another method is to set the compass card for the new heading against any bezel index, then flying the aircraft to align the pointer with that bezel index.

CLOCK

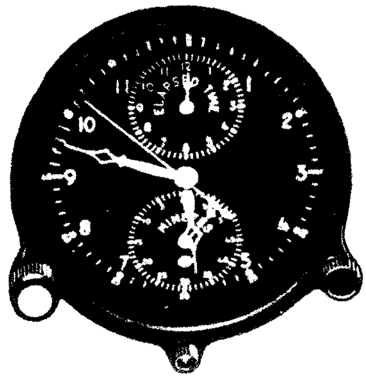


Figure 4—18.

CLOCK.

Unlike previously installed aircraft clocks, this clock (figure 4—18) includes elapsed timing, split-second timing, as well as an 8-day clock winding mechanism. Split-second timing is achieved as follows: Press the right-hand button to start split-second timing; the time traversed by the sweep, split-second needle will be indicated in minutes on the small inner dial, located at the bottom of the instrument face; press the button a second time to stop split-second timing and a third time to return the sweep, split-second needle to zero.

Elapsed time is determined by pressing the left-hand button to start, and reading time traversed on the small inner dial located on the upper portion of the instrument face. Pressing the left-hand button a second time stops elapsed time recording; and for a third time, returns the elapsed time hands to zero. The button on the lower center (6 o'clock position) of the instrument face is used to manipulate the elapsed time needles for adjustment of elapsed time for refueling, repair, or other conditions, so that an exact record of flying time can be indicated for any desired phase of flight. To assist in adjusting for exact elapsed time, indicators (small holes in the dials) are backed up by a black shield when not in use, or partially backed in black when recording of time has been temporarily arrested. The indicators are backed by a luminous shield when the elapsed time dials are operating. The lower center button is turned to the right when the elapsed time needle is to be arrested; in this event the indicator is partially black. By turning the button to the left elapsed time commences to be recorded and the indicator is luminous. When the upper indicator is partially black, the elapsed time hands have been temporarily stopped; and when completely black, the hands are back to zero. When both upper and lower indicators are completely luminous both elapsed time dials are operating.

PLOTTING BOARD.

A Batori plotting board is provided on modified and 35 and later aircraft. The purpose of the Batori plotting board is to provide the pilot with a navigational aid for flights of extreme duration. By means of an adapter the plotting board is mounted in a support assembly located on the right hand of the windshield bow. Stowage provisions consist of two floor mounted brackets on the right side of the cockpit and an elastic strap on the side of the right console just above the main circuit breaker panel. The board is stowed with the mounting adapter attached.

CAUTION

In order to preclude the possibility of the plotting board holding circuit breakers in the reset position when stowed, assure that it is stowed correctly in the floor mounted brackets.

BATORI COMPUTER.

Provisions are made on the right side of the cockpit for the Batori Computer, which is swivel mounted for ready reference. This device is easily accessible to the pilot for one-hand computation of fuel consumption, ground speed, ETA's, etc.

ARMAMENT EQUIPMENT.

The aircraft is equipped to carry two guns in each wing stub. A gun sight mounted in the front of the instrument panel is provided for sighting the guns. No provisions are made for carrying bombs or rockets.

GUN ARMING SWITCH.

The gun arming switch (5, figure 1–6) controls power to the gun sight and gun firing circuit and is powered by the DC primary bus. The switch has two positions: OFF and GUNS, and is guarded in the OFF position with a cover type guard. When in the OFF position, the gun sight is not energized and the guns cannot be fired. The GUNS position supplies primary bus power to the gun sight and energizes a relay so that the guns will fire when the stick trigger is actuated if the aircraft is airborne, on 5 thru 25 aircraft, or the landing gear selector handle is not in the DOWN or EMERG DOWN position on 30 and later aircraft.

GUN SIGHT.

The gun sight (figure 4–19) mounted in front of the instrument panel is a fixed reticle sight. Unmodified 5 thru 30 aircraft are equipped with a modified N-3C reflecting type sighthead with a 25 mil reticle. Modified 5 thru 30 and later aircraft are equipped with a N-9 reflecting type gunsight with a 50.5 mil reticle. The principle of operation of this sight is the apparent projection of the reticle image in space due to the action of a lens and reflector. Therefore, the reticle should be made to coincide optically with the target which is essentially the same as having the circle or cross lines superimposed on the target with an older type sight. The sight does not control fire power. Electical power is supplied from the primary bus.

GUN SIGHT DIMMER CONTROL.

The gun sight dimmer control (5, figure 1–6) controls the intensity of the sight reticle from DIM to BRIGHT.

GUNNERY EQUIPMENT.

Two caliber 0.50 machine guns are installed in each wing stub. Electric heaters are provided for the guns to keep them within operating temperatures. The maximum load of ammunition is 200 rounds per gun. Expended cases and links are conveyed overboard during flight. The guns are charged manually prior to take-off. On 5 thru 25 aircraft an armament-safety switch, mounted on the left landing gear scissors, prevents the guns from being fired when the weight of the aircraft is on the gear provided the struts are not overinflated. The armament safety circuit, on 30 and later aircraft, is incorporated in the landing gear selector handle and prevents the guns from being fired when the selector is

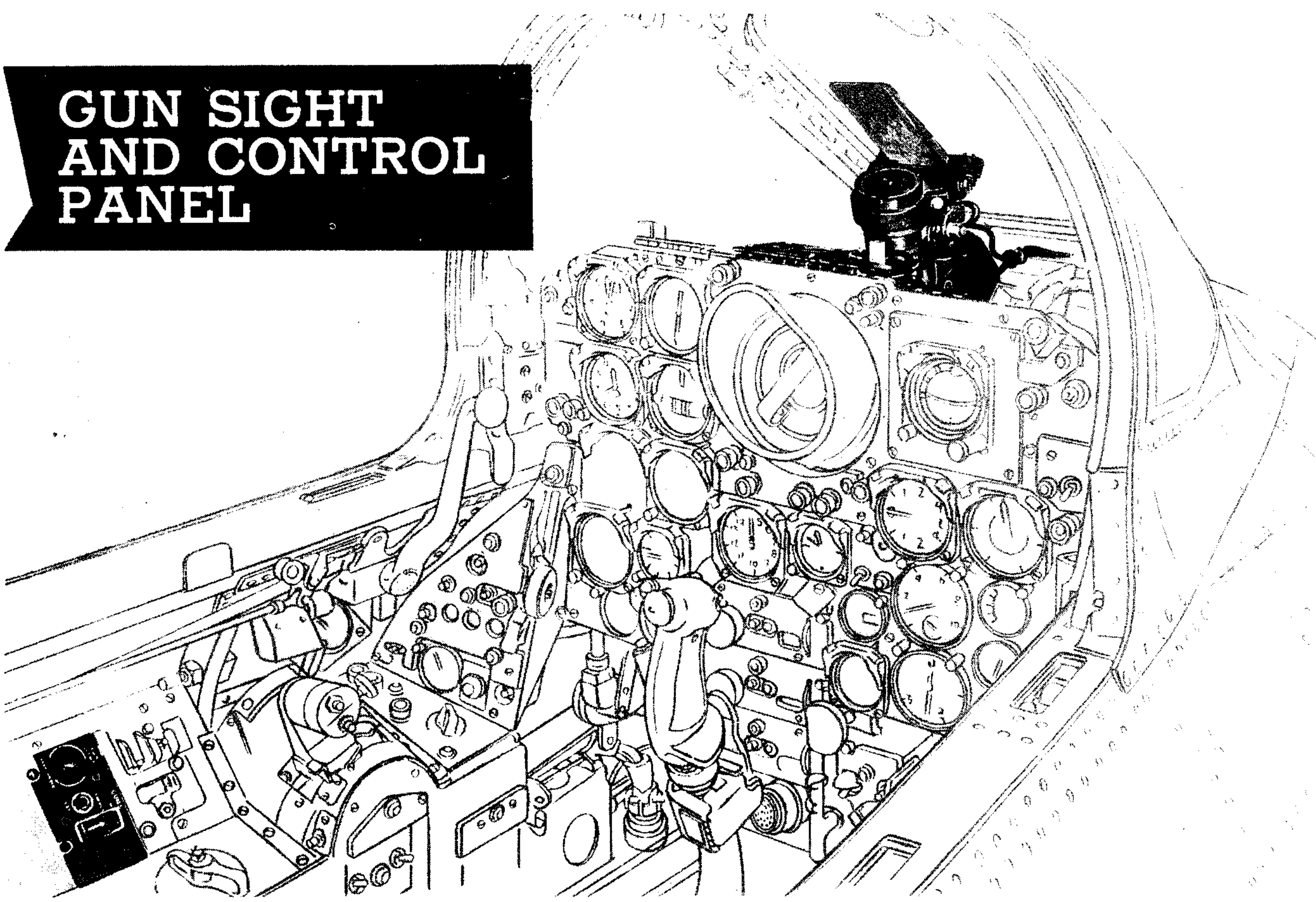


Figure 4—19.

in the DOWN or EMERG DOWN position. If for any reason the guns must be fired while the aircraft is on the ground, the safety circuit can be bypassed by removing the cover of the armament safety override switch located in the right wheel well and pulling the switch aft. The guns' heaters are powered from the NO. 1 secondary bus while the guns are fired by power from the primary bus.

Stick Trigger.

Depressing the stick trigger (figure 1—23) fires the guns, if the gun arming switch is in the GUNS position, primary DC bus power is available and the aircraft is airborne on 5 thru 25 aircraft, or the landing gear handle is not in the DOWN or EMERG DOWN position on 30 and later aircraft.

WARNING

If the landing gear shock struts, on 5 thru 25 aircraft, are overinflated so as to be fully extended it will be possible to fire the guns on the ground by depressing the stick trigger, if the gun arming switch is in the GUNS position and the primary bus is energized. This is also possible on 30 and later aircraft, if the landing gear selector handle is not in the DOWN or the EMERG DOWN position.

Note

The engine screens should be extended when the aircraft is engaged in gunnery to prevent engine damage from expended cartridge cases or ammunition links.

Gun Heater Switch.

The gun heater switch (5, figure 1—6) located on the left console is a circuit breaker type switch having two positions: OFF and HEATER. The HEATER position supplies power to the gun heaters from the DC secondary bus.

Gunfire Operation.

1. Gun heater switch — As required.
2. Gun arming switch — GUNS.
3. Reticle dimmer control — Set for desired brilliance.
4. Depress stick trigger to fire guns.

Note

To eject cartridge links after gunfire and prior to landing; rock wings while aircraft is in a nose high attitude.

PHOTOGRAPHIC EQUIPMENT.

The forward fuselage is divided into two compartments that are accessible from the outside through hinged covers. The forward compartment houses a forward oblique, forward vertical, and left and right mid-high oblique camera. The aft compartment will accommodate a left oblique and a prime vertical camera (figure 4–20). At each camera station there is a glass window hinged to the fuselage, which is opened from the outside for cleaning and maintenance. The prime vertical camera, viewfinder and a photo cell are enclosed by hydraulically operated, flush-type doors on the underside of the fuselage to protect the windows from dirt thrown back from the nose wheel during taxiing and take-off. An emergency pneumatic system is provided to open the doors, so that in the event of a hydraulic failure, the mission can still be completed. The camera equipment will accomplish high or low altitude, day photographic missions and low altitude night missions (figure 4–20). A photo cell which reacts to light from released photoflash cartridges, causes the film in the vertical camera to be automatically advanced during night missions. The photoflash cartridges are released from ejectors carried in pods on the outboard pylons. The camera equipment is mounted on removable mounts and is remotely operated from the cockpit with power from the NO. 2 DC secondary bus. The viewfinder is installed behind the aft camera compartment to furnish a terrain image to the instrument panel. A vacuum system is provided to hold the film against each camera's focal plane during exposure. The vacuum is available whenever the engine is running and the camera compartment air conditioning system is turned ON. There are no controls provided for the vacuum system. The camera compartment air temperature is automatically maintained by an air conditioning system. Power to all cameras is supplied through a master control while individual station controls select the mode of operation of each camera.

MASTER CONTROLS.

Power Switch and Indicator Light.

The power switch (figure 4–21) is a two-position switch placarded ON and OFF and is powered by the NO. 2 DC secondary bus. When ON it provides power to the camera motors, camera compartment recirculating system completes a power circuit to the ready switch and illuminates the POWER indicator light adjacent to the switch.

CAUTION

The camera compartment temperature shut-off switch must be in the ON position, and the camera compartment temperature stabilized to within limits 18.3 to 29.3°C (65 to 85°F) with the temperature control in AUTOMATIC position for 15 minutes, before the camera power switch is turned to the ON position.

Ready Switch and Indicator Light.

The ready switch (figure 4–21) is a two-position switch placarded ON and OFF and is powered by the NO. 2 secondary bus. The ON position supplies power to the intervalometers, the exposure indicator and counter, the operate switch, and also opens the prime vertical camera doors and illuminates the ready indicator light (figure 4–21). If night operating equipment is installed, the ON position will also open the photo cell and flash pod doors.

Note

The power switch must be in the ON position to energize the ready switch.

Operate Switch.

The operate switch (figure 4–21) is a two-position switch placarded ON and OFF. When in the ON position, power is applied to turn on the intervalometers and operates the cameras if the individual camera station controls are in the CONT or INTV positions. The ON position also readies the extra picture switch if the individual camera station controls are in the COMP position. The OFF position supplies power to the external operate switch on the control stick.

Intervalometers.

Three intervalometers (figure 4–21) are provided and are marked OBLIQUES, VERTICAL and TRI-CAMERA. The intervalometers provide time regulated operating impulses to the cameras selected by the individual station controls. The timer interval can be selected within a range of one-half to 60 seconds in increments of one-half second. The OBLIQUE intervalometer controls the forward oblique and the left oblique cameras, the VERTICAL intervalometer controls the prime vertical camera and the TRI-CAMERA intervalometer controls the forward vertical, and the left and right mid-high oblique cameras which form the trimetrogon camera installation.

CAUTION

Damage to the intervalometers will result if the time interval is changed while the cameras are operating.

CAMERA STATIONS

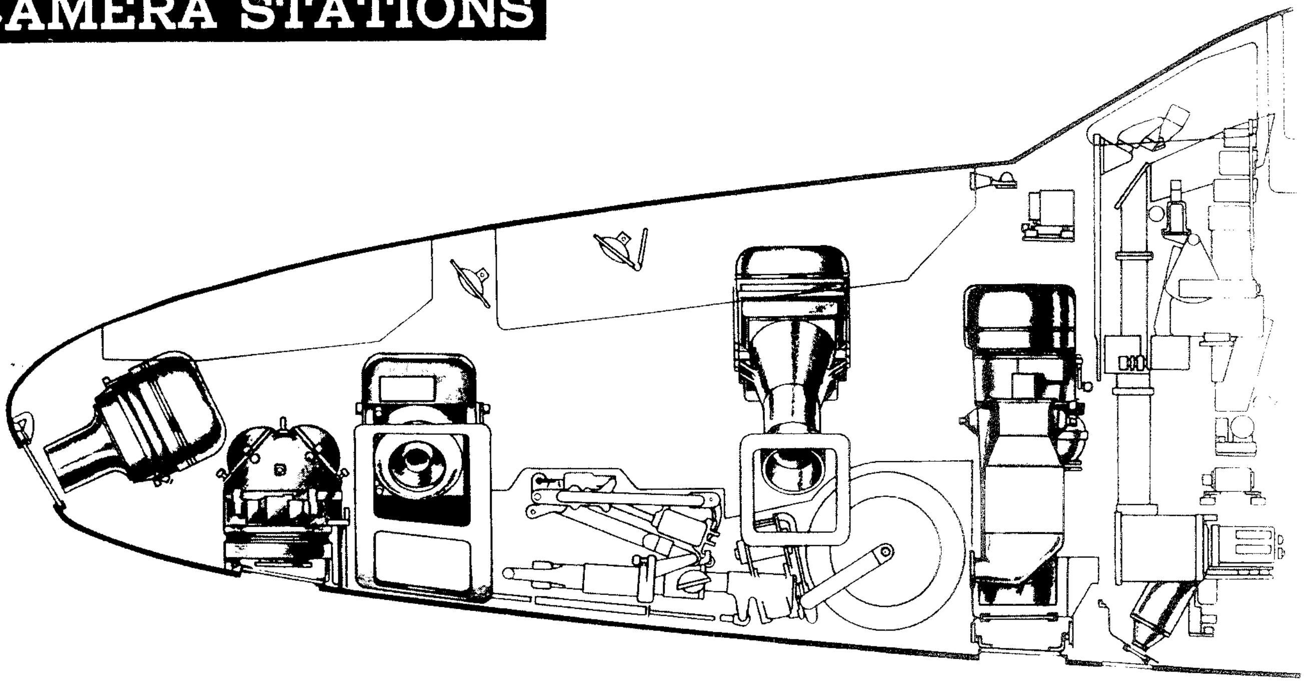
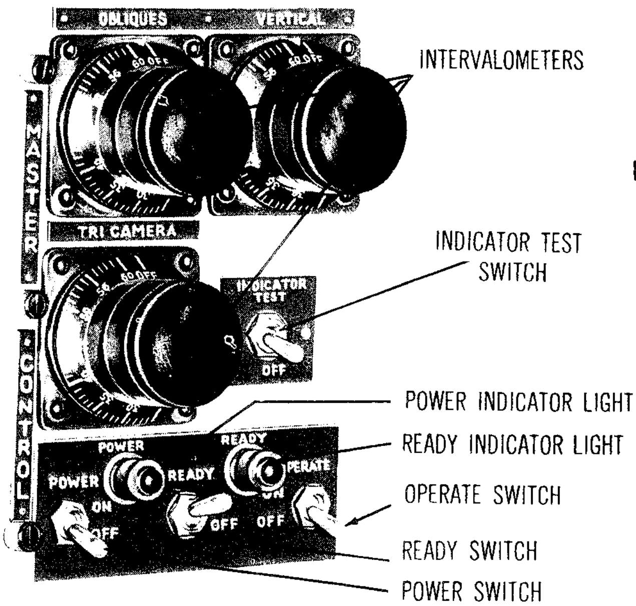


CHART OF Camera APPLICATIONS

	STA #1 FWD OBLIQUE			STA #3 FWD VERTICAL		STA #4 MID HIGH L & R OBLIQUE		STA #6 AFT HIGH L. OBLIQUE			STA #7 PRIME VERT.	
	CAMERA	MAGAZINE	ANGLE BELOW HORIZONTAL	CAMERA	MAGAZINE	CAMERA	MAGAZINE	CAMERA	MAGAZINE	ANGLE BELOW HORIZONTAL	CAMERA	MAGAZINE
HIGH ALTITUDE DAY MISSION	K-22A-6"	A-9A A-9B	25	T-11-6" K-17C-6"	A-9A A-9B	K-17C-6"	A-9A A-9B	K-22A-24"		15 TO 40	K-38-36"	A-8B
	K-22A-12"	A-9A A-9B	25	K-22A-12"	A-9A A-9B			K-22A-12"		25 TO 40	K-38-24"	A-8B
	K-38-36"	A-8B	12					K-22A-24"		15 TO 40	K-38-36"	A-8B
								K-22A-12"		25 TO 40	K-38-24"	A-8B
	KA-2	A-9B	25					KA-2		25 TO 40		
LOW ALTITUDE DAY MISSION	K-22A-6"	A-9A A-9B	25	K-17C-6"	A-9A A-9B	K-17C-6"	A-9A A-9B	K-22A-24"		15 TO 40	K-38-36"	A-8B
	K-22A-12"	A-9A A-9B	25	K-22A-12"	A-9A A-9B			K-22A-12"		25 TO 40	K-38-24"	A-8B
	K-22A-6"	A-9A A-9B	25	K-17C-6"	A-18	K-17C-6"	A-9A	K-22A-24"		15 TO 40	K-38-36"	A-8B
	K-22A-12"	A-5A	25				A-9B	K-22A-12"		25 TO 40	K-38-24"	A-8B
	K-22A-6"	A-9A A-9B	25	K-17C-12"	A-18	K-17C-6"	A-9A A-9B	K-22A-24"		15 TO 40	K-38-36"	A-8B
										25 TO 40	K-38-24"	A-8B
	K-22A-24"	A-9A A-9B	12					K-22A-24"		15 TO 40	K-38-36"	A-8B
								K-22A-12"		25 TO 40	K-38-36"	A-8B
	KA-2	A-9B	25					KA-2		25 TO 40		
LOW ALTITUDE NIGHT MISSION				K-37-12"	A-18							

Figure 4—20

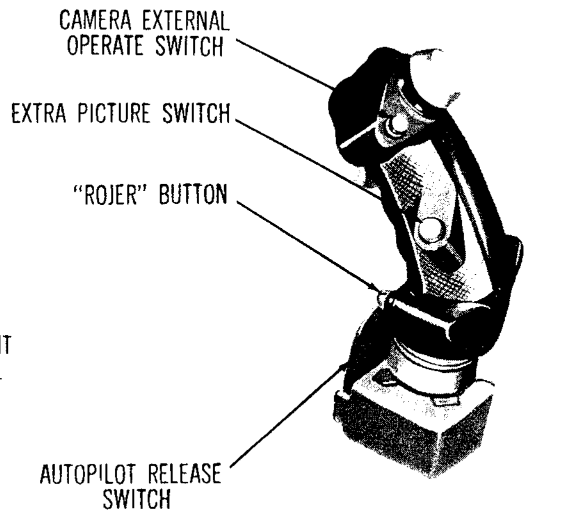
CAMERA CONTROLS



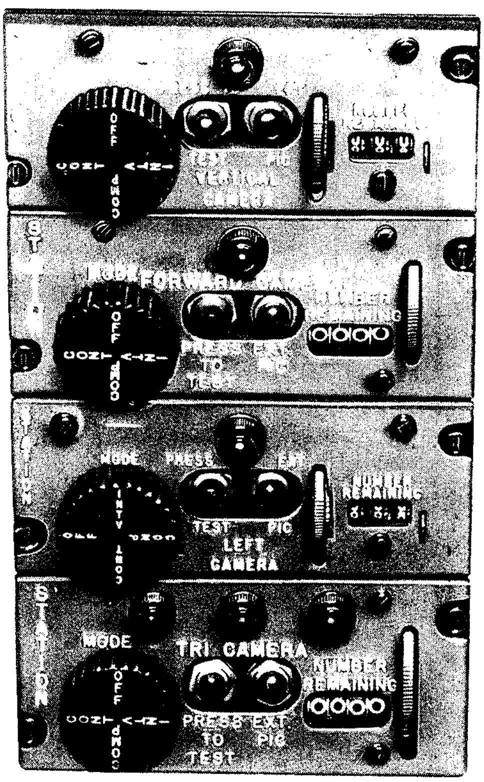
MASTER CAMERA CONTROLS

NOTE:

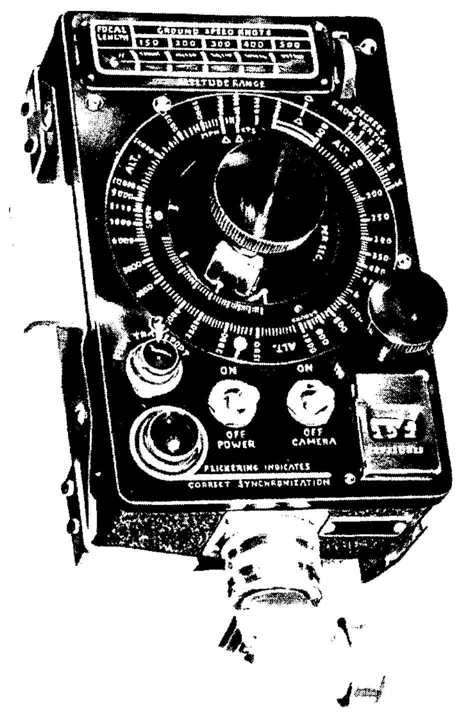
On aircraft 25 thru 45 not modified [549] the extra picture switch and the camera external operate switch are interchanged.



CONTROL STICK



CAMERA CONTROLS STATION



C-4 CONTROL PANEL

Figure 4—21 (Sheet 1 of 2)

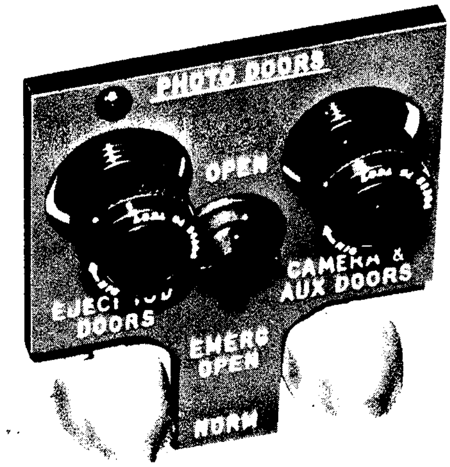
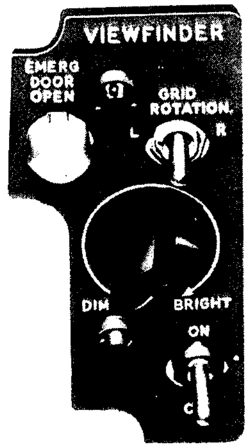
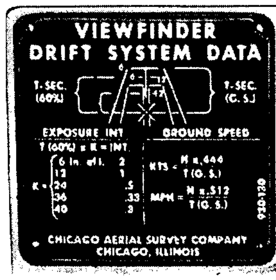


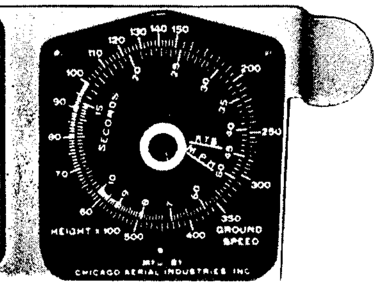
PHOTO DOORS EMERGENCY CONTROL PANEL



VIEWFINDER CONTROL PANEL



DATA PLATE



GROUND SPEED COMPUTER

Figure 4—21. (Sheet 2 of 2)

Indicator Test Switch.

The indictaor test switch (figure 4—21) is provided to test the power and ready indicator lights and is marked INDICATOR TEST and OFF. When placed in the INDICATOR TEST position, the power and ready indicator lights will illuminate, if the ready switch is in the ON position. All equipment powered by the ready switch will be energized.

External Operate Switch.

The external operate switch (figure 1—23) located on the control stick grip, duplicates the function of the operate switch but is more accessible. With the power and the ready switches ON and the operate switch OFF, depressing the external operate switch will operate the intervalometers and cameras, if the individual camera station controls are in the CONT or INTV position. The cameras will continue to operate until the external operate switch is again depressed or the operate switch is turned ON then OFF.

Extra Picture Switch.

When depressed, the extra picture switch (figure 1—23) located on the control stick grip, will cause all cameras whose individual controls are in the COMP or INTV positions to trip and recycle, thereby taking one picture.

CAMERA STATION CONTROLS.

Four camera station control panels (figure 4—21) are installed in the pedestal beneath the center of the instrument panel. The panels are marked TRI-CAMERA, VERTICAL, LEFT OBLIQUE and FWD OBLIQUE. The tri-camera control selects the mode of operation of the forward vertical and the left and right mid-high oblique cameras. The vertical control selects the mode of operation of the prime vertical camera. The left oblique selects the mode of operation of the left oblique camera. The forward oblique control selects the mode of operation of the forward oblique camera. The power switch and ready switch must be in the ON position, and either the operate or external operate switch actuated, for the respective cameras to operate through the camera station control panel. Each panel has the following controls:

VERTICAL CAMERA COVERAGE

VERTICAL CAMERA COVERAGE (feet) using 9x9 inch plate

ALTITUDE FEET	6 IN. LENS X	12 IN. LENS X	24 IN. LENS X
1000	1500	750	375
2000	3000	1500	750
3000	4500	2250	1125
4000	6000	3000	1500
5000	7500	3750	1875
6000	9000	4500	2250
7000	10500	5250	2625
8000	12000	6000	3000
9000	13500	6750	3375
10000	15000	7500	3750
15000	22500	11250	5625
20000	30000	15000	7500
25000	37500	18750	9375
30000	45000	22500	11250
35000	52500	26250	13125
40000	60000	30000	15000

Figure 4—22 (Sheet 2 of 2)

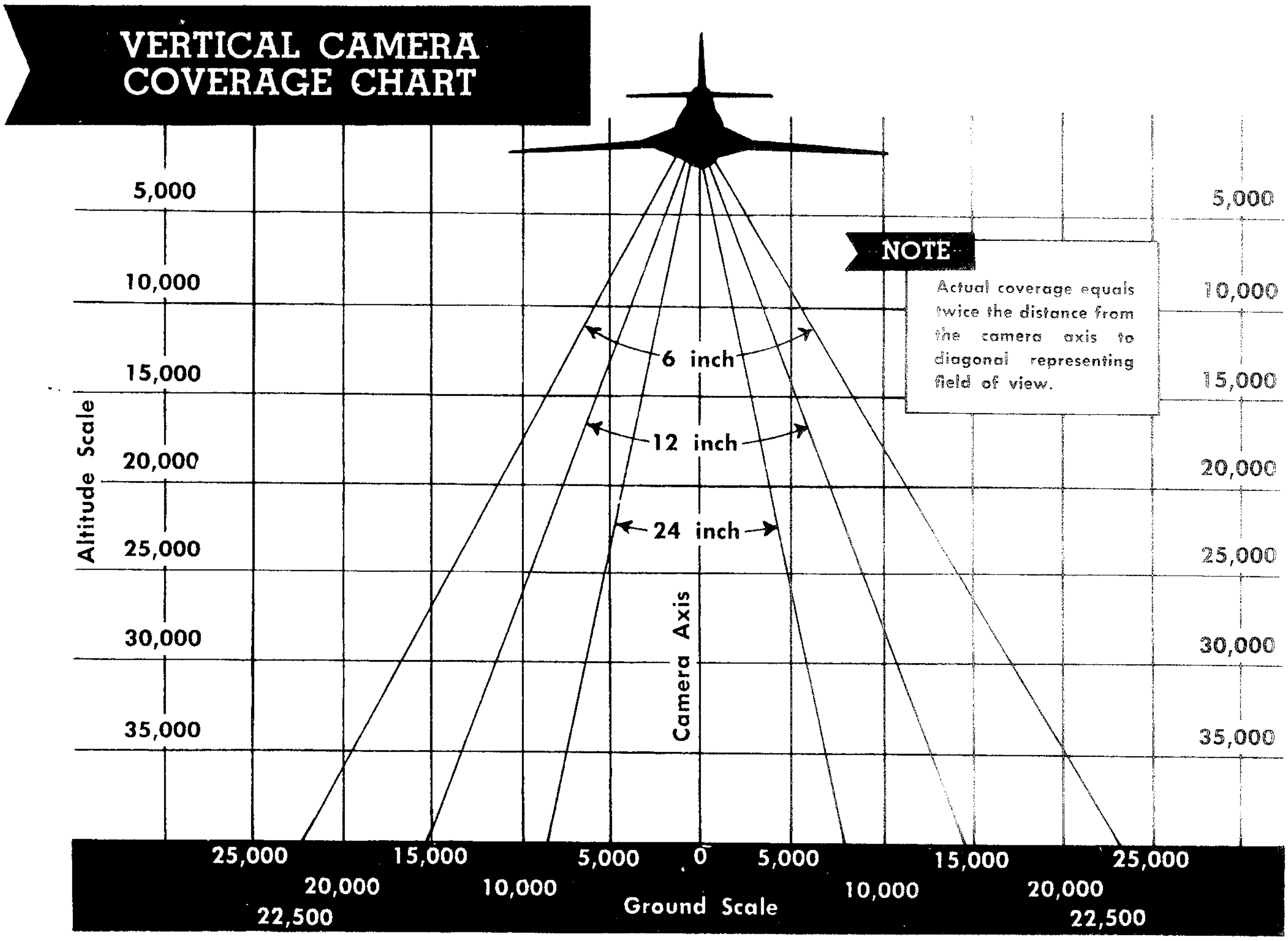


Figure 4—22. (Sheet 2 of 2)

Mode Selector Switch.

The mode selector switch (figure 4—21) selects the method of operation of the respective camera and is powered by the NO. 2 DC secondary bus. It is a rotary switch with the following positions: OFF, INTV, CONT and COMP. When in the INTV position, the camera will operate according to the impulse received from the respective intervalometer. The CONT position allows the camera to trip and recycle as rapidly as it can without any controllable time interval, as long as either operate switch is actuated. The COMP position is used, if it is desired to take individual pictures by using either extra picture switch.

Exposure Indicator Light.

The exposure indicator light (figure 4—21) is a green light which illuminates each time the camera is cycled. The tri-camera panel contains an exposure indicator light for each of the three trimetrogon cameras i.e., forward vertical, and left and right mid-high oblique.

Exposure Counter.

The exposure counter (figure 4—21) is a three-digit counter which indicates the number of exposures remaining in the respective cameras. A thumb-operated, knurled wheel protrudes through the control panel, and is used to reset the counter.

Extra Picture Switch.

The extra picture switch (figure 4—21) is a push button type switch marked EXT PIC, which trips and recycles the respective camera when depressed. If the extra picture switch is held in the depressed position, the camera will operate continuously.

Test Switch.

A push button test switch (figure 4—21), marked PRESS TO TEST, is provided to test the lamp in the exposure indicator light.

Camera Magazines.

Standard type magazines are installed on the various cameras for day missions. These magazines employ a roll of film which will produce 250 to 485 exposures, 9x9 or 9x18 inches in size. A type A-18 image motion compensating magazine can be used with the forward vertical camera. The film is exposed in the magazine, as it is moving at a synchronized speed with the terrain

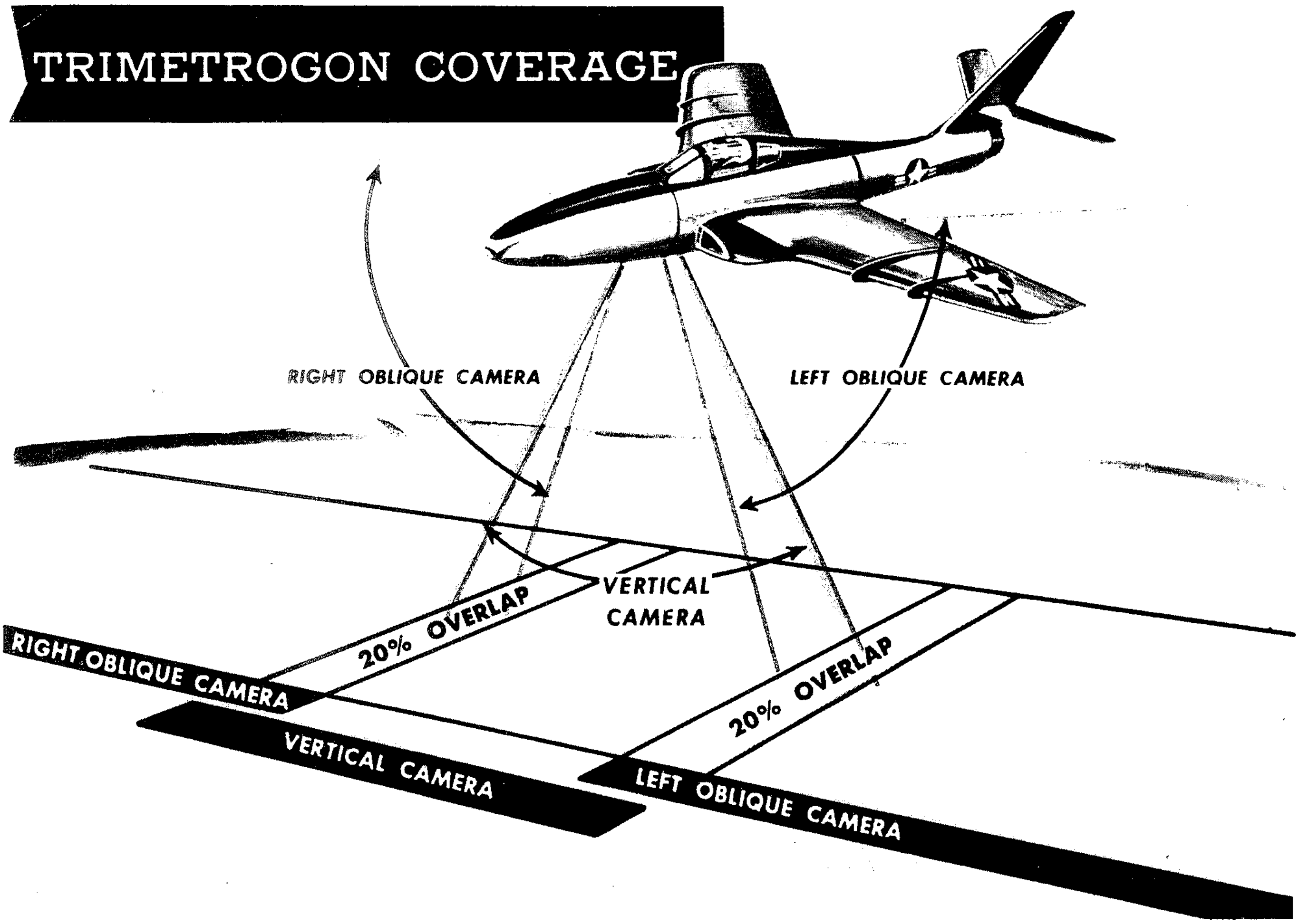


Figure 4—23.

image across the focal plane, thereby producing a sharper negative than with a stationary film. However additional controls are required so that the film speed can be adjusted to the speed of the terrain image. The A-18 magazine is controlled by a C-4 control panel.

C-4 CONTROL PANEL.

The C-4 control panel (figure 4—21) controls the film speed in the A-18 magazine. Film speed control is accomplished through a logarithmic potentiometer in the panel. The panel contains an altitude and ground speed adjusment knob, a focal length adjustment knob, an altitude range indicator, a power switch, a camera switch, two indicator lamps and an exposure counter. The adjustment knobs change the settings of the potentiometer. The altitude range indicator interprets the operating ranges of various focal length lenses in terms of altitude and ground speed. It has a knob which changes the readings for each focal length. The green lamp indicates when the film is in motion within the magazine, and the amber lamp flickers when the film speed is correctly synchronized for the control panel setting. The counter indicates the approximate number of exposures remaining in the magazine. A dial assembly in the center of the panel contains dials for altitude, ground speed, focal length, film speed and synchronizing angle. The dials provide a reading of the adjustments made in the control panel by the control knobs.

Note

Under normal operating conditions with the A-18 magazine, the master camera power switch need only be placed in the ON position three minutes, minimum, to allow the amplifier to warm-up prior to taking pictures.

PHOTO DOORS.

The prime vertical camera, the viewfinder and the photo cell are enclosed by hydraulically and spring-operated, flush-type doors on the underside of the fuselage. Hydraulic pressure is controlled by electrically operated valves. The doors are opened with hydraulic pressure and are closed by means of a spring and cable linkage. The prime vertical camera door opens when the power and ready switches are ON and the mode selector is not OFF. If night operating equipment is installed the photo cell doors will open. The viewfinder doors are opened independently. In the event of hydraulic failure, all doors can be opened with pressure from the pneumatic system, which is controlled electrically with power from the NO. 2 DC secondary bus.

CAMERA & AUX DOORS SWITCH AND INDICATOR LIGHT.

The camera and aux doors switch (figure 4—21) is used for emergency opening of the prime vertical camera and the photo cell doors. There are two positions: NORM and EMERG OPEN. The NORM position is an off position and the doors are operated by the camera controls. The EMERG OPEN position opens the doors with pneumatic pressure. Once the EMERG OPEN position is selected the switch cannot be returned to the NORM position, except by the maintenance crew. When the doors are opened, the camera and aux doors indicator light (figure 4—21) will illuminate, if the power and ready switches are ON.

VIEWFINDER — VERTICAL.

The vertical viewfinder (5, figure 1—4) is designed to furnish the pilot with a clear unobstructed view of the terrain below and ahead of the aircraft, for flight line navigation and the location of photographic targets. The fields of the camera and the lenses used with the viewfinder are also shown. The system covers an 85 degree included angle at a small scale, furnishing the observer with an extended view from near the forward horizon to a point 15 degrees behind the vertical. This system is fitted with a movable grid which contains a track line and a drift angle scale for the solution of drift problems. The grid has controlled illumination which may be used in night viewing or with dark terrain. The view image is unreversed. The viewfinder illumination is powered from the NO. 2 secondary bus, and the grid is adjusted using power from the main (single-phase) inverter.

Viewfinder Switch.

The viewfinder switch (figure 4—21) is an ON-OFF switch which illuminates the viewfinder grid and opens the viewfinder doors by energizing the hydraulic shut-off valve. The OFF position closes the hydraulic valve and the viewfinder doors will close by spring pressure.

Grid Rotation Switch.

The viewfinder grid can be rotated so that the track of an image can be brought on, or parallel to the center track line. Drift can then be read in degrees on the scale next to the pointer. Grid rotation is controlled by the grid rotation switch (figure 4—21) which is a three-position switch marked L and R and is spring-loaded to the unmarked off position.

Grid Dimmer Switch.

The viewfinder grid illumination is adjusted by a rotary switch with the two extreme positions marked DIM and BRIGHT.

Viewfinder Emergency Door Switch.

The viewfinder emergency door switch (figure 4—21) is provided to open the viewfinder doors in the event of hydraulic failure. The switch has two positions: EMERG DOOR OPEN and NORM. The NORM position is an off position and the doors are opened by actuating the viewfinder switch. The EMERG DOOR OPEN position opens the doors with pneumatic pressure by energizing the pneumatic valve. Once the switch is positioned to the EMERG DOOR OPEN position, it cannot be returned to the NORM position except by the maintenance crew.

LEFT OBLIQUE CAMERA SIGHT.

An optical sight for the left oblique camera is mounted on the interior of the left side of the canopy frame. The polarized sight contains a pattern of concentric rings when viewed on a plane parallel to the optical center of the sight. The optical axis of the sight may be rotated by means of the mounting ring so that the angle indicated on the mount calibration can parallel the angle of deflection of the left oblique camera. The unit can be stowed when not used by drawing the sight mount and arm back to the horizontal position.

Operation.

1. Draw the sight arm up to the vertical position.
2. Rotate the sight until the calibration inscribed on the inboard side of the mount equals the angle of deflection of the left oblique camera. Various aiming points within the format of the picture may be made by adjusting the sight, dependent upon camera focal length and depression angle.

EJECTOR PODS.

An ejector pod is installed on each outboard pylon. Each pod is capable of carrying four type A-6 or B-4 photoflash cartridge ejectors. The ejectors are for the purpose of releasing photoflash cartridges in order to take aerial photographs at night. The A-6 ejector has a capacity of 52 cartridges each or 208 cartridges per pod. The type B-4 ejector has a capacity of 20 cartridges or 80 per pod. Type A-6 and/or B-4 ejectors can be carried simultaneously in the left pod, while only A-6 or B-4 ejectors can be carried in the right pod. The control panel for the ejector cartridges is installed in the pedestal in place of the left oblique and forward oblique control panels. The cartridges, when installed in the pods, are enclosed by flush-type doors that are opened when the power and ready switch are in the ON position. A self-contained pneumatic system is built into each pod for emergency opening of the pod doors.

PHOTOFLASH CONTROLS.

The dual ejector control panel is installed in the pedestal below the instrument panel, in the place of left oblique and forward oblique camera station control panels. The panel incorporates an intervalometer for controlling the interval between releasing cartridges, a limiter which is set to the desired number of cartridges to be released, a selector switch to select which type cartridge is released, two counters, identified as A and B, and two indicator lights. The selector switch is marked EJECT A, OFF and EJECT B. One indicator light illuminates each time a cartridge is released. The other indicator will illuminate if the ejector racks are not properly locked in place.

EJECTOR DOORS SWITCH AND INDICATOR LIGHT.

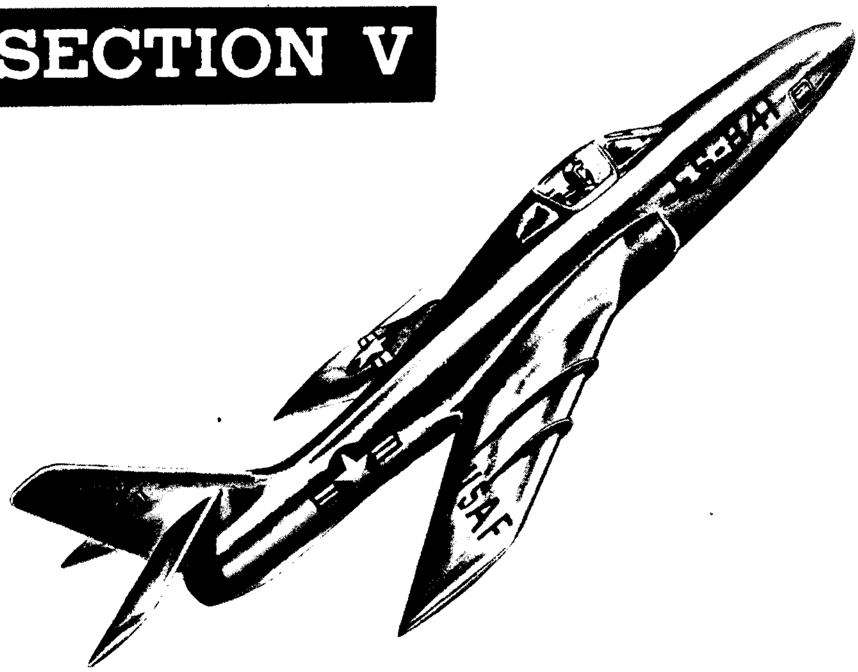
The pod ejector doors are electrically opened and closed when the power and ready switches are in the ON position, if the photoflash control panel is installed. However, in the event of electrical failure, the doors can be opened by pneumatic pressure. The ejector doors switch (figure 4—21) has two positions: NORM and EMERG OPEN. The NORM position is an off position and the doors are operated by the camera controls. The EMERG OPEN position energizes a pneumatic valve to open the doors with pneumatic pressure. The switch cannot be returned to the NORM position once it is placed in the EMERG OPEN position. When the doors are open, the green ejector doors indicator light will illuminate if the NO. 2 DC secondary bus is energized.

MISCELLANEOUS EQUIPMENT.

ANTI-G-SUIT PROVISIONS.

An air pressure outlet connection on the front of the ejection seat (figure 1—29) provides for the attachment of the air pressure intake tube of the pilot's anti-g suit. Air pressure for inflation of the anti-g-suit bladder is conducted from the engine compressor through an anti-g valve (23, figure 1—6) located on the left console which starts functioning when a force of 1.75 G is applied to the aircraft. A control marked HI and LO allows for adjustment of the rate of inflation of the anti-g suit. In the LO range the valve opens at 1.75 G and then allows 1 PSI of air pressure to pass to the suit for every increase of 1 G force thereafter. In the HI range the valve still opens at 1.75 G but delivers 1.5 PSI per G force thereafter. The suit will inflate in 0.2 to 2.0 seconds depending on the input pressure.

SECTION V



Operating Limitations

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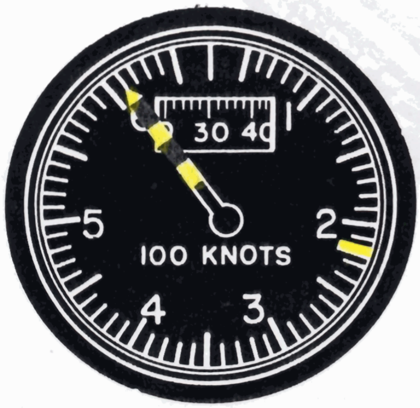
INSTRUMENT MARKINGS

FUEL GRADE-JP-4



UNMODIFIED

AIRSPEED INDICATOR

225 Knots—Landing gear or flaps DOWN

610 Knots—Maximum airspeed

UNMODIFIED

MACH INDICATOR

1.175 Design dive speed

MODIFIED

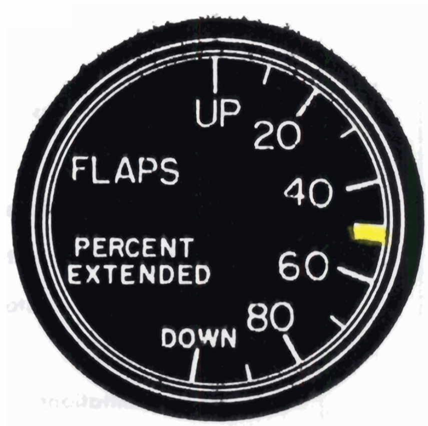
AIRSPEED MACH INDICATOR

225 Knots—Landing gear or flaps DOWN

610 Knots—Maximum airspeed

ACCELEROMETER

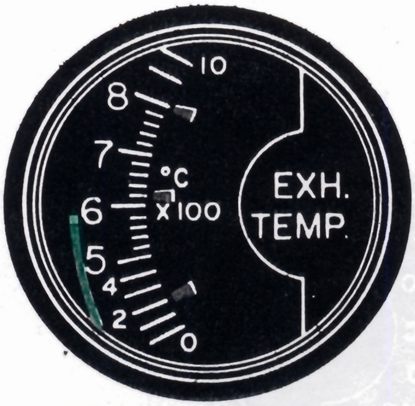
+8.67 G	Maximum below M0.9 clean
+7.0 G	Maximum above M0.9 clean or at any Mach No. with external stores.
—3.0 G	Maximum



WING FLAP INDICATOR

50% Take-off

Figure 5-1 (Sheet 1 of 3)



J65-3 ENGINE

EXHAUST GAS TEMPERATURE

- 200 to 585°C — Continuous operation
- 200°C — Minimum for flight
- 620°C — Maximum stabilized for flight (30 minutes)
- 800°C — Maximum during starting and acceleration

J65-7 ENGINE

EXHAUST GAS TEMPERATURE

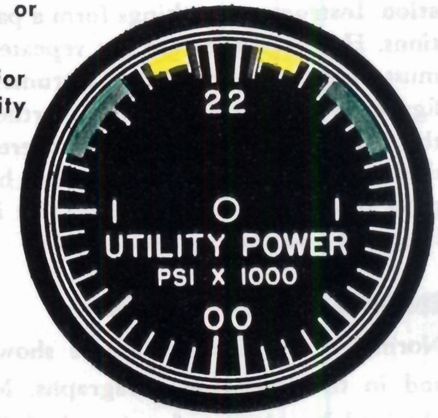
- 200 to 595°C — Continuous operation
- 200°C — Minimum for flight
- 650°C — Maximum stabilized for flight (30 minutes)
- 800°C — Maximum during starting and acceleration

TACHOMETER

- 83.0 to 96.5% RPM — Best cruise RPM
- 101.0% RPM — Maximum (Ground or flight 30 minutes)
- 60 to 82% RPM — Restricted range (For operational necessity only)

OIL PRESSURE

- 20 PSI — Minimum—Below 50% RPM
- 24 PSI — Minimum—50% RPM and above
- 40 PSI — Maximum
- 60 PSI — Maximum for cold start, initial run-up, take-off and climb



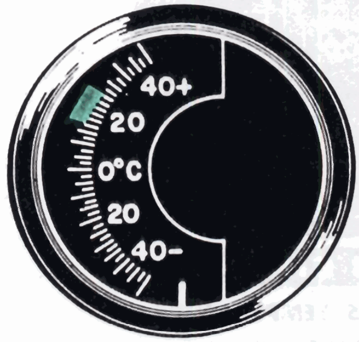
HYDRAULIC PRESSURE
UTILITY AND POWER SYSTEMS

- 1250 - 1600 PSI — Normal full flow
- 1400 - 1600 PSI — Static
- 1750 - 1900 PSI — Permissible 5 seconds max
- 1925 PSI — Maximum

Figure 5-1 (Sheet 2 of 3)

STARTER BOTTLE AIR PRESSURE

2000 - 3000 PSI Normal



CAMERA COMPARTMENT TEMPERATURE

18 - 30°C Normal operating range

Figure 5—1. (Sheet 3 of 3)

INTRODUCTION.

This section includes the engine and aircraft limitations that must be observed for safe and efficient operation. Instrument markings form a part of these limitations. However, they are not repeated in the text and must be referred to on the instrument marking page, figure 5—1. Where necessary, further explanation of the instrument markings are covered in the text of this section, under the appropriate heading. For complete restrictions carefully read the instrument marking page and the explanatory text.

ENGINE LIMITATIONS.

Normal engine limitations are shown in figure 5—1 and in the following paragraphs. Military thrust is obtained by placing the throttle full forward. Maximum continuous thrust (normal rated thrust) is defined as the thrust obtained at approximately four percent engine RPM below military thrust RPM. Maximum thrust is the same as military thrust since the engine is not equipped with afterburner.

OVER TEMPERATURE LIMITS.

During any start immediate action should be taken to abort start when it appears EGT will exceed the normal starting limit, 800°C. The EGT and duration of each hot start or acceleration in excess of 800°C will be recorded in the AFTO Form 781 and AFTO Form 44. In addition the following overheat conditions require corrective action as indicated:

Condition	*Corrective Action*
Any single start during which EGT exceeds 800°C (1,472°F)	Requires investigation to determine cause
After four starts or acceleration during which EGT exceeds 860°C (1580°F) for more than 10 seconds	Requires first stage turbine blade inspection
Any single start during which EGT exceeds 900°C (1,652°F)	Requires hot section inspection and turbine wheel replacement in accordance with applicable T.O.

TRANSIENT TEMPERATURE LIMITS.

For starting and accelerations, exhaust gas temperature should not exceed 800 degrees (1,472°F) or remain at its peak value for more than 10 seconds. Following starts and accelerations, the exhaust gas temperature will overshoot the stabilized temperature for a given throttle setting, and several minutes are required to reach the stabilized value. After peaking, the exhaust gas temperature will show a continuous decline until temperature is stabilized. Initial temperature decline rate from the peak will be quite rapid and will become slower and slower as stabilization is approached. A stabilized temperature is one that will remain constant for three individual readings taken at 30 second intervals. The exhaust gas temperature should not stabilize on any point above the maximum steady state value. Figure 7–1 graphically illustrates the EGT behavior following rapid acceleration of a cold J65-3 or -7 engine.

MAXIMUM STABILIZED EGT.

J-65-3 Engine

IDLE 36 to 42 percent RPM)	660°
Takeoff & during flight	620°

J-65-7 Engine

IDLE (42 to 48 percent RPM)	660°
Takeoff & during flight	650°

WARNING

Since the aircraft maintains a nose-high attitude during stalls, airflow to the engine can be critically low when the aircraft is in a stall. Therefore, the throttle should be retarded, and rapid throttle advancements should not be attempted, until the nose has been lowered and airspeed is definitely increasing. Should the throttle be advanced before the engine receives sufficient airflow, turbine overtemperature may result.

Note

When the ambient air temperature exceeds 32°C (90°F) maintain idle RPM exhaust temperature within limits by manually advancing the throttle to a higher RPM not to exceed 50 percent RPM.

OVERSPEED LIMITS.

During an acceleration the RPM may momentarily exceed 101 percent. If the RPM should exceed 103 percent, the throttle is to be retarded immediately to stabilize RPM at 100 percent or below. If the engine exceeds 106 percent RPM, it is necessary to shut down the engine as soon as possible and perform a hot section inspection and turbine rotor replacement as outlined in the applicable maintenance manual.

ENGINE ACCELERATION LIMITS

Refer to Section VII for discussion on engine acceleration. Engine acceleration time should be as follows with screens retracted.

Engine Model	*Type Fuel Control*	*Percent RPM*	*Accel. Time Seconds*
J65-3	TJ-L1	47 to 100	8-14
J65-3 or -7	TJ-L2	47 to 100	15 max

Engine Acceleration and Overspeed Check.

Advance throttle in one second from 47 percent (IDLE) RPM to 100 percent RPM. The engine should accelerate stall free to 100 percent RPM (and not exceed 101 percent) in 15 seconds at sea level within EGT limits.

Note

Acceleration time may be increased ½ second for each 1,000 feet increase in altitude.

- Rapid manipulation of the throttle may result in engine chugs and/or stalls.
- Some engines will not accelerate properly when they are cold and will therefore require a short warmup period in order that the acceleration limits are met.

If the engine acceleration limits are not met, repeat the acceleration check. If the acceleration limits are not met, after the second check, reject the engine and record in Form 781. If the EGT during an acceleration exceeds the maximum limit, retard the throttle slightly until temperature drops below the limiting value and then advance it slowly to avoid exceeding the limit. Avoid further rapid accelerations until the cause of the overtemperature has been determined. Record all operation above the acceleration limit. Refer to Section VII, Exhaust Gas Temperature Stabilization.

ENGINE RPM RESTRICTIONS.

Engine operation in the 60 to 82 percent RPM range can induce compressor blade stresses which may cause rotor blade failures in the number one, and seven, stages. Operating time in this range is cumulative and has a direct effect on blade life. When the restricted range is entered, it is not necessary to pass through the complete range before reducing RPM to a point below the restricted range. To preclude the possibility of compressor rotor blade failure caused by steady state operation in the 60 to 82 percent RPM range the following restrictions will be adhered to:

1. Static ground operation – Do not operate in the 60 to 82 percent RPM range, except during acceleration or deceleration through this range.
2. Taxiing – Avoid stabilized operation in the 60 to 82 percent RPM range.
3. In Flight – Operation in the 60 to 82 percent RPM range will be performed only as required by operational necessity.

Note

J-65-3A, -7C and 7D Engines do not have a restriction on engine operation in the 60-82 percent RPM range above 10,000 feet.

ENGINE FLUCTUATION.

Engine fluctuation (at any RPM) is permissible provided that the following is observed:

1. Fuel flow shall not vary a total of more than 500 pounds per hour.
2. Total RPM variation shall not vary more than 1½ percent.
3. Normal EGT limits are not exceeded.
4. Engine surging does not produce any perceptible movement (or change in movement) of the aircraft.

OIL PRESSURE LIMITS

Time and pressure limitations are based on the minimum oil pressure required to insure an adequate supply of oil to the center and rear main bearings. Minimum allowable oil pressure limits are 20 PSI with engine operating below 50 percent RPM and 24 PSI with engine operating at 50 percent RPM and above.

1. Do not operate engine with zero oil pressure longer than one minute after initiating start.
2. Operation below the minimum allowable oil pressure is permissible for a period of one minute or less.
3. Operation below the minimum allowable pressure for a period of over one minute but not exceeding two and one-half minutes requires an engine inspection.
4. Operation below the minimum allowable pressure for a period exceeding two and one-half minutes requires engine removal.

Note

The same time limits shall apply to windmilling engines with zero oil pressure.

ATO UNITS.

The following ATO units are approved for utilization without restrictions, except those prescribed in the Inspection, Handling and Storage Technical Order for each type unit.

14AS1000
14DS1000 MK4. Mod 2
14DS1000 M-8
15KS1000
M-15 (16NS1000 or T-60)

AIRSPEED LIMITATIONS.

Landing gear extension speed – 225 KIAS maximum.

Flap extension speed;

50 percent–260 KIAS maximum.
100 percent–225 KIAS maximum.

Landing light extension speed – 225 KIAS maximum.

Air refueling receptacle opening speed – 350 KIAS maximum.

Autopilot engagement speed – 425 KIAS maximum below 20,000 feet.

Drag chute deployment speed – 220 KIAS maximum.

At higher speeds shear pin will shear, if airborne, an excessive sink rate will result.

ATO units, jettison speed – Between 200 and 250 KIAS.

Jettison outside these limits may result in damage to the rear hooks and fuselage.

Canopy Speed Restriction.

Do not taxi with canopy open when speed or wind gusts exceed 40 knots.

ACCELERATION LIMITATIONS.

See figure 5–2.

ACCELERATION LIMITATIONS

SYMMETRIC FLIGHT SPEED LIMIT AND LOAD FACTOR LIMIT

These limits apply except when limited to lower values by buffet, or as noted in Section IV.

When external stores are used in combination, the lower limit shall apply.

When carrying tanks, max deceleration 4 G, safe rate of descent 5,000 feet per minute.

Max rate of descent 10,000 feet per minute.

Abrupt large displacements of the rudder should be avoided. Excessive sideslip angles can lead to dangerous snap maneuvers. External stores and/or extended speed brakes tend to further aggravate this condition.

Abrupt rolling pullouts are prohibited when carrying external stores due to shackle limitations (See Bomb Rack Limitations in this Section).

CONFIGURATION			MAXIMUM SPEED KIAS	MAXIMUM POSITIVE G BELOW 5,000 FT	MAXIMUM POSITIVE G ABOVE 5,000 FT	MAXIMUM NEGATIVE G
CLEAN—(ANY CAMERA LOADING, ANY AMOUNT OF FUEL OR AMMO)		Below M 0.9	610	+8.7	+7.0	−3.0 G
		Above M 0.9	610	+7.0	+7.0	
230 GAL TYPE I TANKS	INBD		610	+7.0	+6.0	
450 GAL TYPE I TANKS	INBD	MB-3 Shackles	610	+6.0	+5.0	
		S-3 Shackles	610	+4.5	+4.0 to 15,000 FT +3.0 above 15,000 FT	
230 GAL TYPE II TANKS	INBD		610	+4.0	+4.0	−2.0 G
	OUTBD	MA-4 Shackles	350	+4.0	+4.0	
		S-2A Shackles	350	+3.7	+3.7	
450 GAL TYPE II TANKS	INBD	MB-3 or S-2A Shackles	610	+4.0	+4.0	
FLARE PODS (ANY LOADING)	OUTBD	MA-4 Shackles	550	+7.0	+6.0	−3.0 G
		S-2A Shackles	550	+5.4	+5.4	

JETTISON LIMITATIONS

Jettison of stores above or below jettison speeds may result in stores striking aircraft and causing structural damage.

In case of an emergency, all types of tanks can be jettisoned when full or nearly full to lighten aircraft.

Stores not listed may be jettisoned at the same speeds provided they are of the same type and have similar contours and capacity.

250 KIAS may be considered as a general jettison speed for all stores.

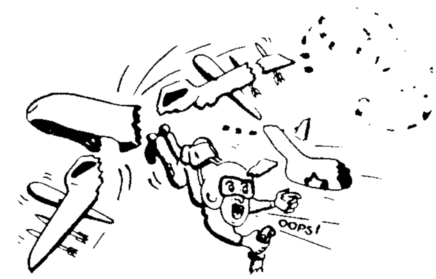
STORE	STATION	JETTISON SPEED KIAS	JETTISON ATTITUDE
PYLONS (CLEAN)		Up to **400** with revised gun installation	Level or pulling up
230 GAL TYPE I TANKS	INBOARD	Up to **300** any quantity of fuel Up to **400** with 0.3 second time delay	Dive, Level or Climb
450 GAL TYPE I TANKS	INBOARD	Up to **250** any quantity of fuel **300** optimum with 0.3 second time delay **400** maximum with 0.3 second time delay	Dive, Level or Climb
450 GAL TYPE II TANKS	INBOARD	**275**	Steady Climb
230 GAL TYPE IV TANKS	INBOARD	WHEN LESS THAN HALF FULL, DO NOT JETTISON When more than half full, jettison in emergency at lowest speed possible.	Steady nose high
230 GAL TYPE IV TANKS	OUTBOARD	**240** to **275** (Tail cone removed)	Steady climb
FLARE PODS	OUTBOARD	Release only in Emergency	

Figure 5—2

PROHIBITED MANEUVERS

Any maneuver resulting in negative acceleration of −1.0 G or more for 5 seconds is prohibited. Negative acceleration in excess of 5 seconds will result in flame-out.

Intentional spins — Prohibited



Snap rolls while carrying external stores are prohibited.

Abrupt large displacement of the rudder should be avoided with external stores installed. In this configuration excessive sideslip angles can lead to dangerous snap maneuvers. Extended speed brakes tend to further aggravate this condition.

Figure 5—3.

JETTISON EXTERNAL STORES.

See figure 5—2.

Note

Jettisoned tanks occasionally appear to "fly" rather than fall as expected. When jettisoning fuel tanks in a formation, aircraft should be in line abreast with at least 500 feet separation between aircraft. When jettisoning singly the pilot shall warn adjacent aircraft and allow ample time for their reaction. Pilots of adjacent aircraft will exercise extreme watchfulness and, if possible, take a position forward of and above aircraft jettisoning its tanks.

CAUTION

Jettisoning while in a turn or yaw will increase the tendency for tanks to "fly".

PYLON TANKS.

The Type IV, 225 gallon, plastic, external fuel tanks, due to its non-conductivity, is more seriously affected by lightning stroke. Therefore, these tanks should not be flown where lightning hazards exist.

BRAKE LIMITATIONS.

1. When the landing gear (wheels) remains in the slip stream for a series of successive landings, a minimum of 15 minutes should elapse between landings.

   This restriction does not apply to "touch and go" landings when no brake application is involved.

2. When the landing gear (wheels) is retracted into the wheel wells, a minimum of 30 minutes should elapse between landings.

   This restriction may be reduced to 5 minutes for "touch and go" landings when no brake application is involved.

PROHIBITED MANEUVERS.

See figure 5—3.

OPERATING LIMITATIONS.

See figure 5—4.

BOMB RACK LIMITATIONS.

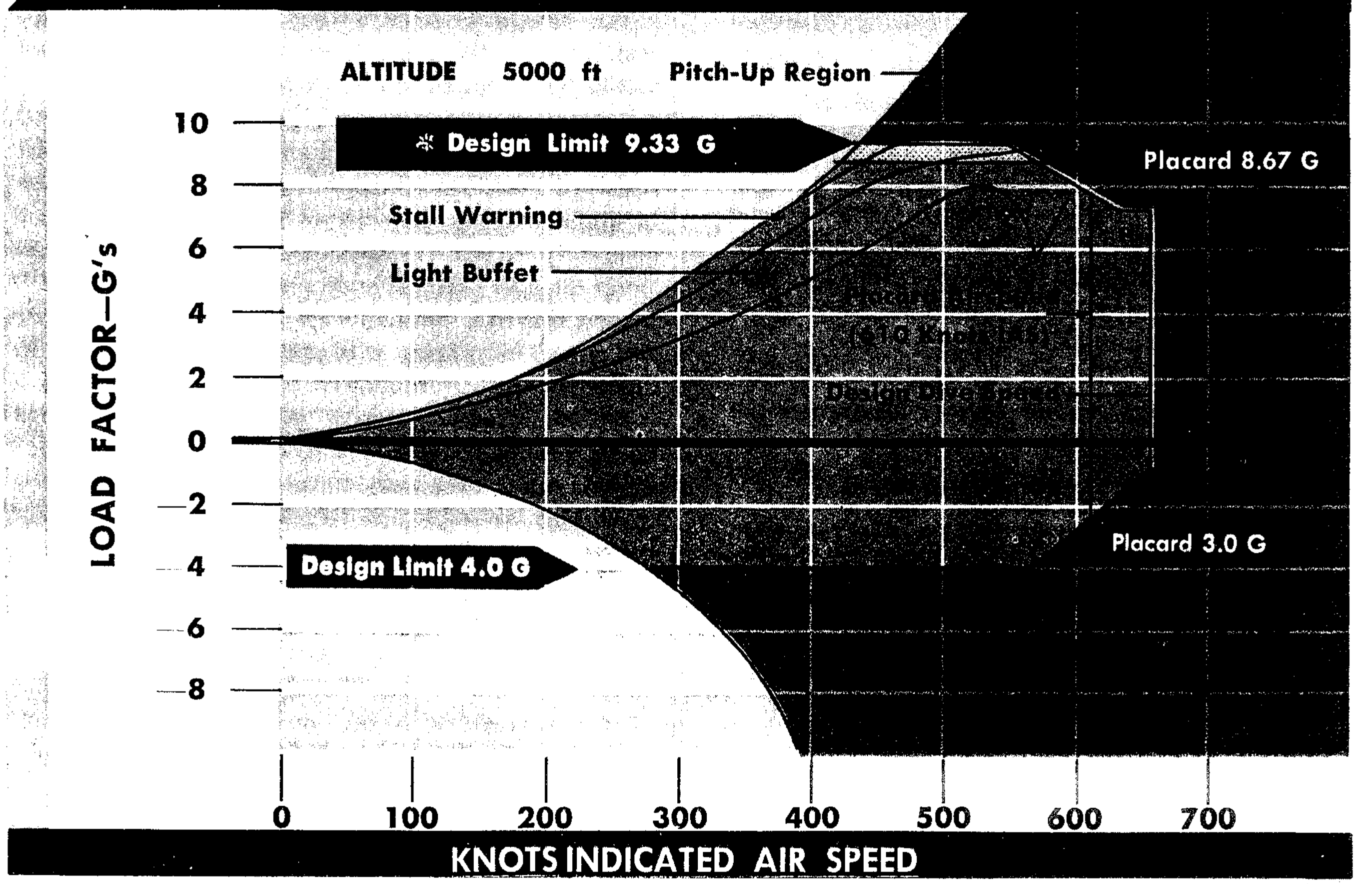
Recent accidents resulting from failure of the following bomb racks; S-2, S-2A, S-3, MA-4 and MA-4A, have indicated the necessity for maintaining closer con-

OPERATING LIMITS

DIAGRAM

GW — 19000 LB.

*See figure 5—2 for acceleration limits with external stores.



This operating limits diagram shows what speeds and G are possible and allowable for the aircraft. In the light buffet area a very mild buffet is experienced. As the G are increased the buffet intensity increases as denoted by the stall warning area. To keep from exceeding the design limit of 9.33 G it is important that the pilot recognize the heavy buffet region. Heavy buffet is an indication that pitch-up may occur. A more detailed description of the actual operating limits which must be observed by the pilot are given in figure 6—3, Maneuvering Flight Characteristics.

The green area represents the G loads and airspeeds which may be used by the pilot without damage to the aircraft. The aircraft can perform all the maneuvers required within these areas.

Figure 5—4

trol over these racks. Therefore, until the above mentioned bomb racks are replaced with racks of greater strength, it is necessary that the following instructions be observed to insure that appropriate inspections are accomplished. A notation should be made in the DD Form 781 whenever any of the following conditions are encountered with any of the above mentioned bomb racks installed.

1. Whenever a suspended fuel tank collapses during flight.
2. Whenever any of the following conditions are encountered with more than 1,000 pounds suspended from type S-2, S-2A, MA-4, MA-4A bomb racks, or more than 2,000 pounds suspended from type S-3 bomb racks.

   a. Whenever the aircraft exceeds 90 degrees per second rate of roll.
   b. Whenever the aircraft exceeds 4.5 G.
   c. Whenever any unusual incident occurs in flight or during ground operations which could cause damage to any part of the racks, such as rough taxiing or hard landings.

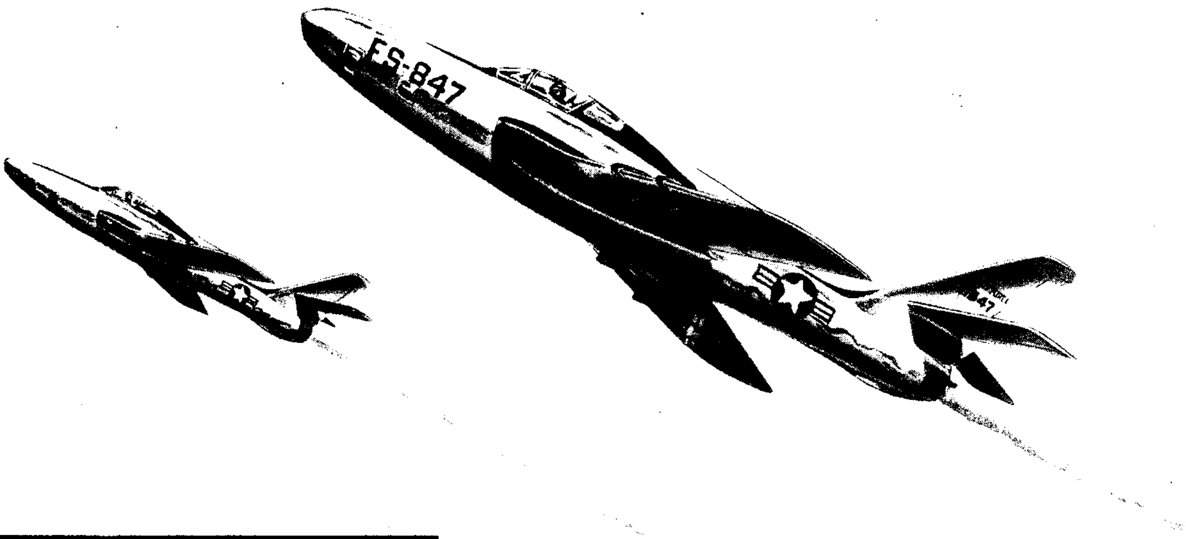
WEIGHT LIMITATIONS.

MAXIMUM GROSS WEIGHT.

The maximum gross weight of the aircraft has been limited to 27,000 pounds, plus the weight of the expendable assist takeoff units. The load limiting factor is determined by the main wheel tires which have a maximum loading of 13,500 pounds each. The normal gross weight of the aircraft is approximately 19,000 pounds. This includes pilot, all internal fuel, oil, all guns and ammunition. Therefore, a total of 8,000 pounds of external stores of any description, not including assist takeoff units, may be loaded on the aircraft to obtain the maximum gross weight. Each aircraft is equipped with a Manual of Weight and Balance, Data, T.O. 1-1B-40, which gives more accurate information as to basic weight and center of gravity positions of the individual aircraft. This Manual should be consulted before each flight. The center of gravity of the aircraft is very stable. All loadings will fall between the established most forward and most rearward positions. Only by some unusual condition in flight of using the expendable load can the center of gravity exceed its limits. If the loading of external stores includes external tanks and fuel on the outboard pylons, this fuel should always be used first, otherwise the center of gravity will exceed its most aft limit, due to taking the fuel from the internal tanks. If a four bomb load condition is used, drop outboard bombs first, if possible. This is not imperative, but for smooth center of gravity travel, is practical. All other conditions are stable and the center of gravity cannot get out of bounds.

CAUTION

On aircraft with fuel booster pumps modified in accordance with T.O. 6J10-3-10-502 dated 15 June 1954, the center of gravity will move approximately one percent forward of the maximum allowable forward CG position when special stores are carried on the inboard pylons. Furthermore, when operating under this condition, a large unsticking force will be required to raise the nose wheel during takeoff.



SECTION VI
Flight Characteristics

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GENERAL FLIGHT CHARACTERISTICS.

The aircraft is designed for high speed at all altitudes. The swept wing and tail surfaces reduce the effects of compressibility, therefore the high speed is limited only by the available thrust and the total drag of the aircraft. All flight controls are hydraulically operated to provide comfortable control forces and ease of maneuvering. Minor longitudinal trim changes occurring in level flight at high Mach Numbers are easily trimmed out. This, in combination with trimming of the aileron and rudder artificial "feel" systems permits complete trim throughout the speed range.

FLIGHT CONTROLS.

The ailerons and rudder are conventional hydraulically actuated flight controls. The elevators and stabilizer have been combined to form a single surface known as the stabilator. All flight controls are hydraulically actuated and irreversible. The stabilator and ailerons are equipped with tandem hydraulic actuators. The utility hydraulic system provides hydraulic fluid under pressure to the utility side of the stabilator and aileron tandem actuators and the rudder actuator in addition to the landing gear, wing flaps, speed brakes, spoilers and all other hydraulic operated systems. The power hydraulic system provides hydraulic fluid under pressure to the power side of the stabilator and aileron tandem actuators exclusively. Both the utility and power hydraulic systems are completely independent of each other and either system alone is capable of sustaining flight control. Since both hydraulic systems are powered by engine-driven hydraulic pumps, an emergency hydraulic system incorporating an electrically driven hydraulic pump is provided. The emergency pump provides hydraulic fluid under pressure to the power side of the stabilator and aileron tandem actuators. The stabilator is fully powered hydraulically and cannot be operated without hydraulic pressure. The ailerons are hydraulically boosted and although manual operation is possible, a high stick force is required due to friction and airloads. Manual operation of the rudder is adequate and it is a useful control for picking up a wing. Since the controls are irreversible and airloads on the surfaces cannot be felt, control feel is simulated by artificial feel units in each primary control. The artificial feel unit is a spring capsule designed to give the pilot a sense of control feel by increasing the force required to deflect the controls proportional to the amount of deflection. The feel devices also provide trim control by means of electric actuators which reposition the spring capsules to the selected no-load position. A mechanical advantage shifter is incorporated in the stabilator control system to allow for more control surface deflection in takeoff and landing than at high speeds. The shifting is done automatically as the landing gear is retracted or extended. After takeoff, when the gear retracts, the stabilator leading edge will move up (aircraft nose down trim change) while the shifter is operating at any fixed aft stick position. Since this movement is relatively slow and the normal aircraft nose up trim requirement is decreasing as the aircraft accelerates, only small corrections need be made by the pilot to compensate for the change in stabilator position. When the gear is extended for landing the procedure outlined in the preceding sentence will be reversed and a slight nose up trim change will be noticed. Failure of the shifter in the one to one ratio during flight makes the aircraft more sensitive in the pitch axis therefore, overcontrolling and porpoising is possible in high speed flight. Failure of the shifter in the 2 to 1 ratio in the landing configuration results in approximately nine degrees aircraft nose up stabilator being available for landing. Even with the aircraft at the forward critical loading (CG at 15 percent MAC) this type of malfunction presents no hazard as long as a touchdown speed of 140 KIAS or above is utilized. Therefore, the Flight Control Laboratory recommends that in the event of a 2 to 1 ratio malfunction in the landing configuration a touchdown speed of 140 KIAS be used for all loading conditions. In addition to the mechanical advantage indicator light which is illuminated while the ratio is changing or if the system is in the wrong ratio, the pilot can recognize when the mechanical advantage remains in the 2 to 1 ratio when the landing gear is extended by the absence of an aircraft trim change.

WARNING

Lateral control with unboosted ailerons is inadequate in high speed, low altitude conditions, under turbulent landing conditions, and with asymmetrical loads installed.

- Abrupt, large displacements of the rudder should be avoided. Excessive sideslip angles can lead to dangerous snap maneuvers. External stores, and/or extended speed brakes, tend to further aggravate this condition.

SPEED BRAKES

Retractable speed brakes on each side of the aft fuselage provide good deceleration from any speed with only minor trim changes. They may be positioned to any intermediate position between full open and closed to provide additional speed control. Buffeting due to the extension of these brakes is moderate and not limiting to the utility of the aircraft.

LEVEL FLIGHT CHARACTERISTICS.

LOW SPEED.

The low speed characteristics and handling qualities are good. Angles of attack at minimum flying speeds are somewhat higher than with a straight wing. The stabilizing tendency for the leading wing to rise, when the aircraft is yawed with the rudder, is more noticeable with swept wings than with a straight wing. This is known as "dihedral effect". This dihedral effect produces considerable rolling tendency when the aircraft is yawed, particularly at low speeds, and can be used very effectively to assist lateral control. Good control effectiveness about all control axes is provided down through the stall. The glide path becomes increasingly difficult to adjust at low speeds and therefore approach speeds must be governed accordingly. Extension or retraction of the landing flaps and gear causes only a slight trim change.

CRUISE AND HIGH SPEED.

The aircraft trims well throughout the entire speed range. Stability is good about all axes. At Mach Numbers between approximately 0.85 and 0.95 a slight nose down tendency is present. This trim change is scarcely noticeable. In unaccelerated flight the aircraft does not encounter any buffet throughout the speed range.

STALLS.

WARNING

Since the aircraft maintains a nose high attitude during stalls, airflow to the engine can be critically low when the aircraft is in a stall. Therefore, the throttle should be retarded, and rapid throttle advancements should not be attempted, until the nose has been lowered and airspeed is definitely increasing. Should the throttle be advanced before the engine receives sufficient airflow, turbine overtemperature may result.

STALLS UNACCELERATED.

A slight yawing tendency followed by light buffeting occurs in advance of the stall. A control softening or slight nosing up tendency occurs coincident with heavier buffet. This is easily controlled with stabilator. Lateral control is adequate during the stall and recovery and any roll off tendency is easily controllable. Rudder is very effective in bringing a wing up particularly if the aileron boost is inoperative with resultant high stick forces. Since the aircraft does not "fall through" until after the stall, but maintains a nose high attitude, practice stalls will be misleading. The aircraft appears to be flying at speeds well below those which are usable for landing but under these conditions the sinking speeds are excessively high. It is rather difficult to establish a clear cut stall speed. With increased experience it will become apparent to the pilot that the practical airspeeds for landing are governed by the sinking speeds of the aircraft and those speeds at which the ship begins to yaw slightly prior to the buffet. Figure 6–1 gives minimum flying speeds of the aircraft.

ACCELERATED STALLS – LOW SPEED.

As the stall is approached in turns the control force softens coincident with the onset of mild buffeting. At somewhat higher G, buffeting increases indicating that stall is imminent and a final moderate pitchup occurs at the stall.

ACCELERATED STALLS – HIGH SPEED.

At high speeds the characteristics are similar except that pulling up through the heavy buffet region (stall warning) will result in a severe pitchup which will increase G, and at altitudes below 25,000 feet could cause structural damage or failure. To avoid experiencing this pitchup, observe the accelerated stall warnings consisting of a slight nose up tendency and/or an increase in buffet severity. If the pitchup is experienced, correct immediately using forward stick. This phenomenon is described in more detail in a separate section on Accelerated Stall Pitchup.

STALL SPEEDS

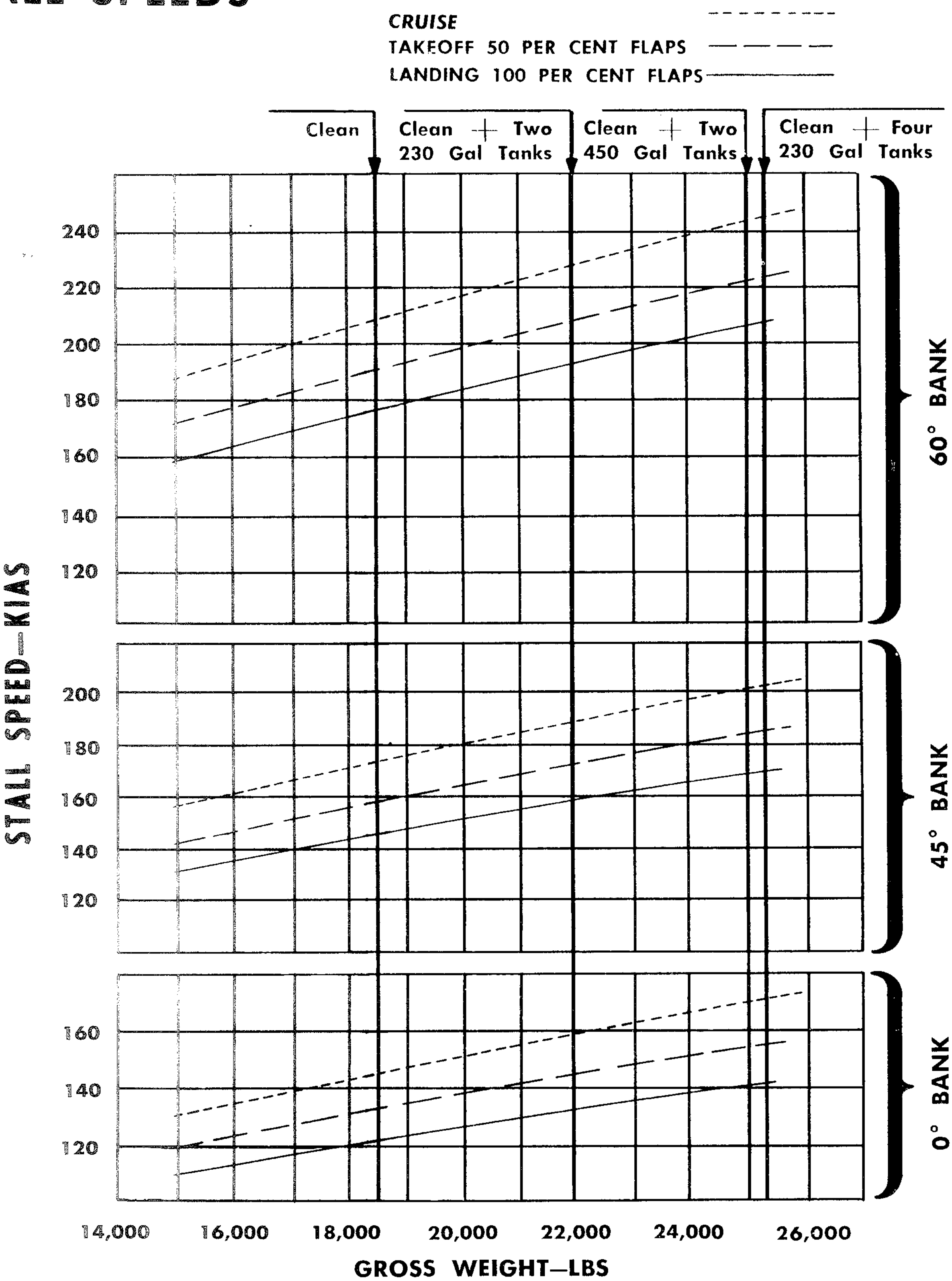


Figure 6—1

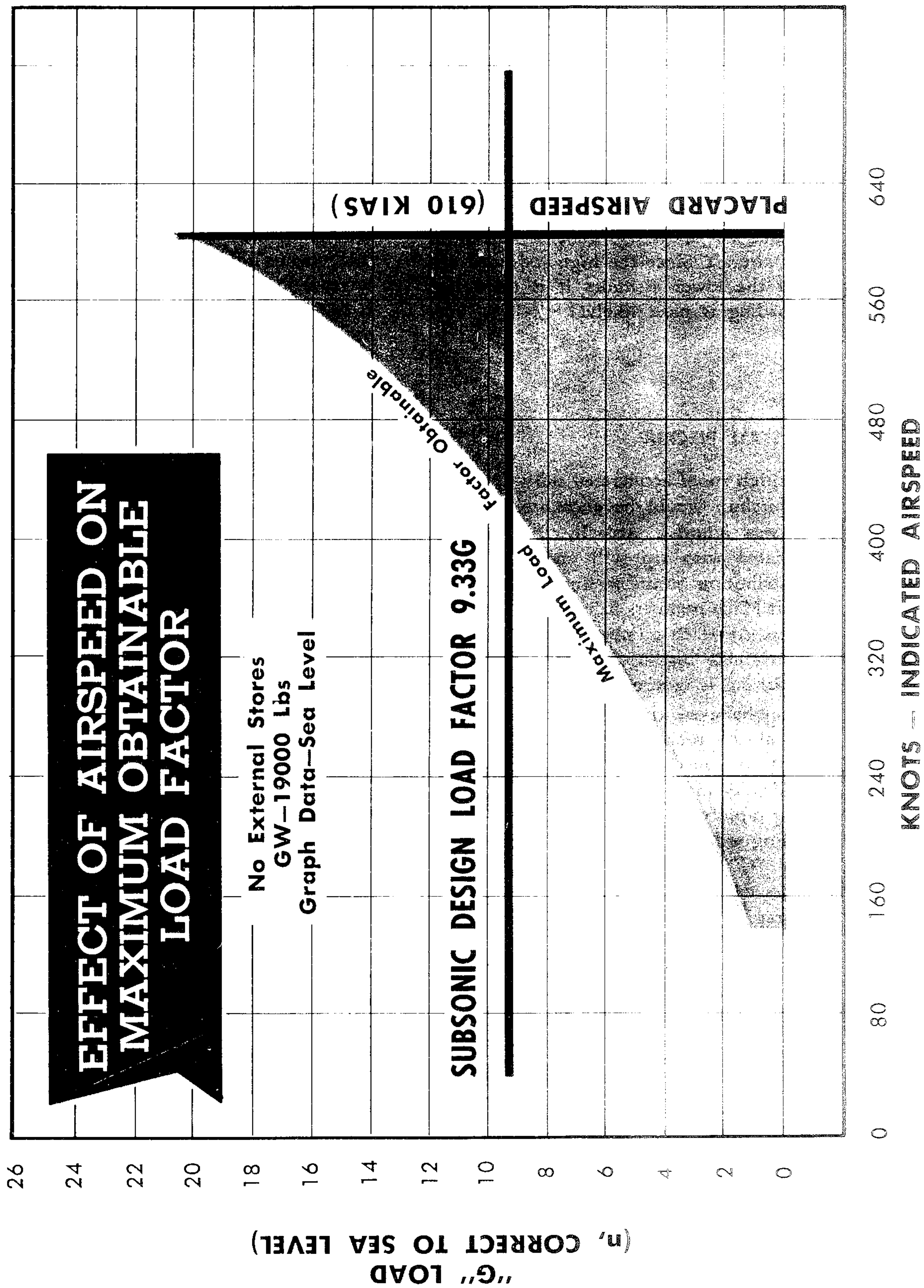

EFFECT OF AIRSPEED ON
MAXIMUM OBTAINABLE
LOAD FACTOR
No External Stores
GW—19000 Lbs
Graph Data—Sea Level
SUBSONIC DESIGN LOAD FACTOR 9.33G
PLACARD AIRSPEED (610 KIAS)
Factor Obtainable
Maximum Load
KNOTS — INDICATED AIRSPEED
0
80
160
240
320
400
480
560
640
"G" LOAD
(n, CORRECT TO SEA LEVEL)
26
24
22
20
18
16
14
12
10
8
6
4
2
0

Figure 6-2

MANEUVERING FLIGHT.

With the high speed capability of this aircraft, you will be operating during a major part of the flight time in the high speed region where the wing is capable of developing load factors which can destroy the aircraft. The G which the wing is capable of developing at various airspeeds are shown in figure 6—2. Note that the obtainable G increases with indicated airspeed and that above 430 knots it is possible to overload the aircraft. Since the high indicated airspeeds occur at the lower altitudes it is there that strutural overloading is most likely.

ACCELERATED STALL PITCHUP.

The section on high speed accelerated stalls explained that a severe pitchup would be encountered if the stall warning was ignored. Actually this pitchup, which is associated with tip stall and an inboard shift or lift distribution occurs at high angles of attack. However at Mach numbers below 0.7 the aircraft does not pitchup. At Mach numbers between 0.7 and 0.95, an uncontrollable pitchup will be encountered if the buffet warning is ignored.

This pitchup may be severe enough to break the wing. The pitchup is always preceded by a sudden increase in buffet which combined with a moderate control softening provides adequate warning. However, if "G" are applied very rapidly it is possible to pass through the buffet warning without time to correct. The maximum allowable "G" can be applied at high speeds and low altitudes without danger. However, the speed bleeds off very rapidly under high "G" and if the "G" is held on until the aircraft slows down through the heavy buffet the aircraft will pitchup uncontrollably to dangerous load factors. At high altitudes the aircraft will pitchup at correspondingly lower "G". Although the load factors imposed on the aircraft will not be too great, the rapid nose up movement could cause loss of control. Note that at low and medium altitudes the maximum hazard due to pitchup exists because there you may encounter it at just about the limit load factor and the uncontrollable "G" due to pitchup can break the aircraft. *Therefore*: Observe the accelerated stall warning, terminate high "G" maneuvers at or before heavy buffet. If pitchup is encountered, correct immediately using forward stick. Referring to figure 6—3 for example, it can be seen that at 5,000 ft altitude and a Mach Number of 0.90, 8.67G can be put on the aircraft. If this "G" is not reduced as the airspeed bleeds off, the buffet will become severe at M = 0.80 and at M = 0.78 the aircraft will pitch up to 10½ "G".

BUFFET BOUNDARY.

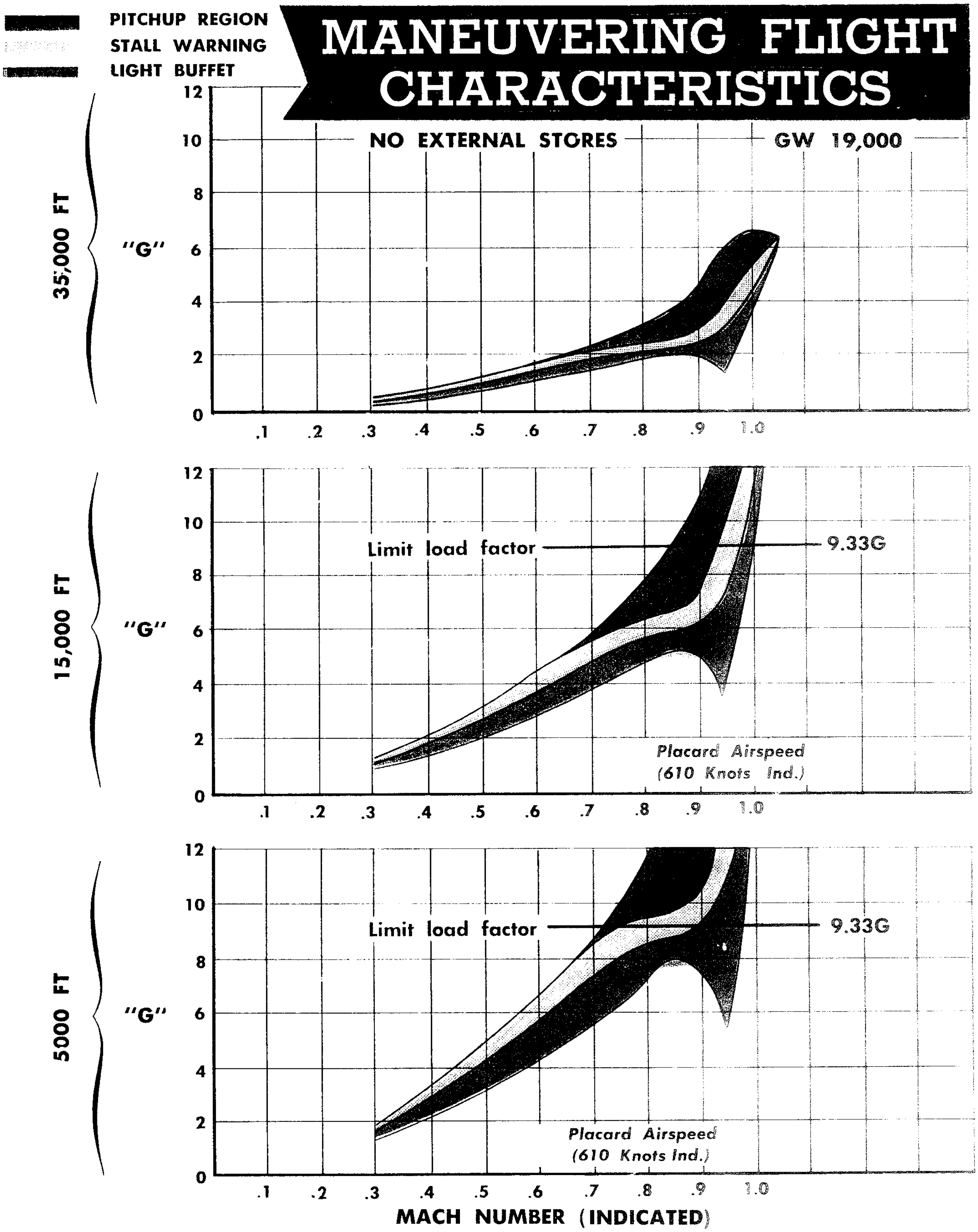
The buffet boundary occurs at relatively high "G's" particularly at the higher indicated speeds as shown in figure 6—3. A very mild buffet is experienced when the "G" is increased to the initial onset of buffet and this remains mild with further increase in "G" until an increase in the buffet intensity occurs at a higher G. It is the latter increase in intensity which constitutes the accelerated stall warning. This should normally be considered a stopping point to prevent the high Mach Number stall pitchup previously described. The mild buffeting at somewhat lower G's does not limit maneuvers.

DIVES.

The acceleration in a dive is very rapid, and high dive speeds can be obtained. The speeds reached in a 60 degree dive from 40,000 feet are shown in figure 6—4. Dives at high Mach Numbers close to the ground, are prohibited because of the large altitude loss during recovery as shown in figure 6—5. The high Mach Number dive characteristics are excellent. No buffet is experienced as the aircraft picks up speed. The transition from subsonic to supersonic velocity is sometimes accompanied by a very slight, wing dropping, and/or yawing. Both are easily controlled by the pilot. Supersonic characteristics are excellent but reduced control effectiveness should be anticipated. During a high speed dive, engine RPM may increase two or three percent from the original setting. This is caused by a lag in inlet temperature compensation in the fuel control resulting from the rapid change in the temperature. It may be necessary to control the engine speed in order to avoid exceeding the overspeed limit of 103.0 percent RPM.

CAUTION

If the angle of dive is not steep enough to accelerate to supersonic speed and the aircraft stays at approximately Mach 1.0 some aircraft will tend to roll at this transition speed. Recovery is easily accomplished by either increasing speed or slowing down by reducing power and extending speed brakes.



PITCHUP REGION
STALL WARNING
LIGHT BUFFET
MANEUVERING FLIGHT CHARACTERISTICS
NO EXTERNAL STORES
GW 19,000
35,000 FT
15,000 FT
5000 FT
"G"
Limit load factor
9.33G
Placard Airspeed (610 Knots Ind.)
0
2
4
6
8
10
12
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
MACH NUMBER (INDICATED)

Figure 6-3

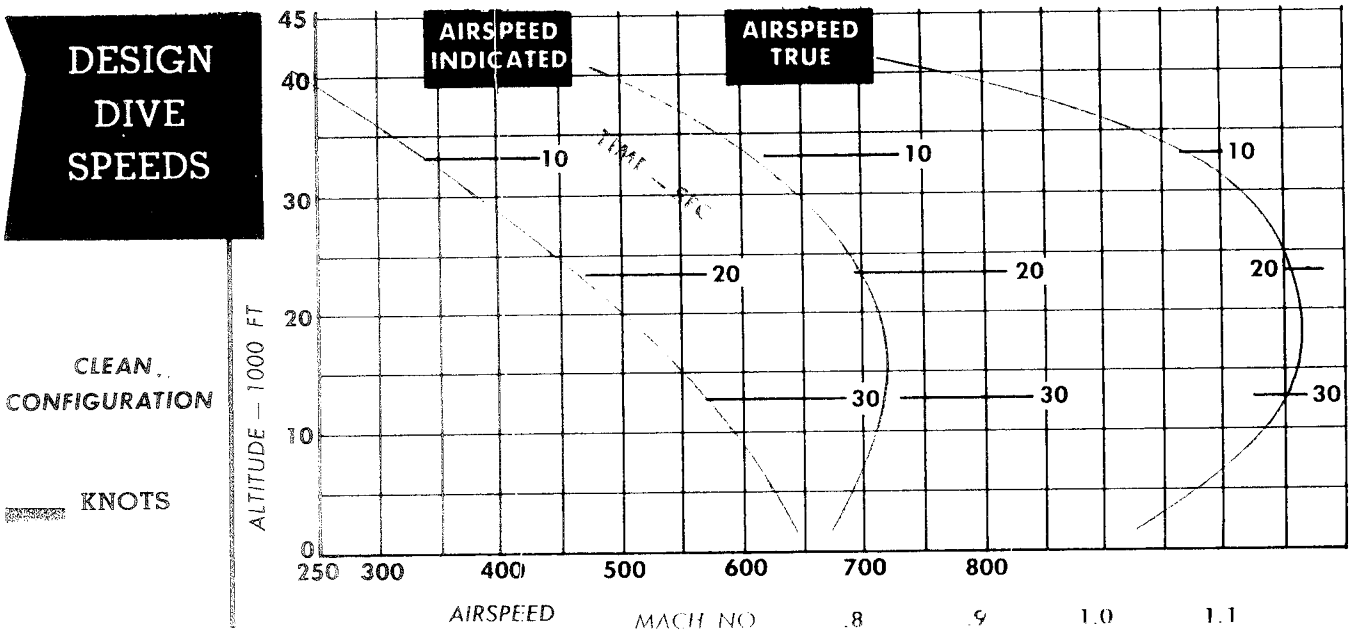


Figure 6—4

SPINS.

The aircraft has demonstrated consistently good spinning characteristics with no tendency toward flat or unrecoverable spins. When the aircraft is held in a spin the first three or four turns are oscillatory. These oscillations decrease in amplitude as the spin progresses until after approximately four turns the aircraft's longitudinal axis is at an angle of approximately 60 degrees relative to the ground. The aircraft loses altitude at the rate of 1,000 to 1,200 feet per turn taking approximately two seconds to complete one turn. Demonstration spins indicate that the best recoveries are effected when the corrective action is initiated during that part of the oscillation where the aircraft nose is descending and with speed brakes retracted.

CAUTION

Intentional spins are prohibited.

NORMAL SPIN RECOVERY.

To recover from a spin the following procedure is recommended:

1. Apply full back stick and approximately ⅓ ailerons with the spin, then abruptly apply full opposite rudder.

WARNING

Ailerons held against the spin will prevent recovery. It is imperative that the ailerons be held neutral or slightly with the spin and the stick full back during recovery.

2. When the spin rotation stops neutralize rudder and ailerons and when the pullout is well underway ease the stick forward slowly. Premature forward movement of the stick during recovery attempt will cause the nose to go under excessively which may result in an inverted spin.
3. If the spin is entered when carrying external stores, use the recommended spin recovery procedure. If the spin does not stop in three turns, jettison external stores, and repeat recommended recovery procedure.
4. If a spin is entered with asymmetric external stores it is recommended that the asymmetric store be jettisoned then recommended spin recovery procedure used. If the asymmetric store is outboard of the spin, the spin would be aggravated.

WARNING

If in a spin and still spinning without any indication of recovery at 10,000 feet above terrain immediate ejection is recommended.

EMERGENCY SPIN RECOVERY.

The results of aircraft spin tests have shown satisfactory spin recovery characteristics when proper techniques are followed. These techniques are considered adequate for recovery from most spins. In an emergency, however, if the normal spin recovery techniques have not been successful, the drag parachute may also

DIVE RECOVERY *Chart*

EXAMPLE:

If a 6.0g pullout from a 30 degree dive at 600 knots CAS is started at 25000 feet the altitude lost during dive recovery will be 2700 feet.

1. Enter chart at calibrated airspeed of aircraft (A) at start of pullout.
2. Move to the right to curve representing altitude (B) at start of pullout.
3. Sight vertically down to curve representing angle of dive (C).
4. Move to the right to curve representing the load factor to be held in the pullout (D).
5. Sight vertically and read altitude lost during pullout on scale (E).

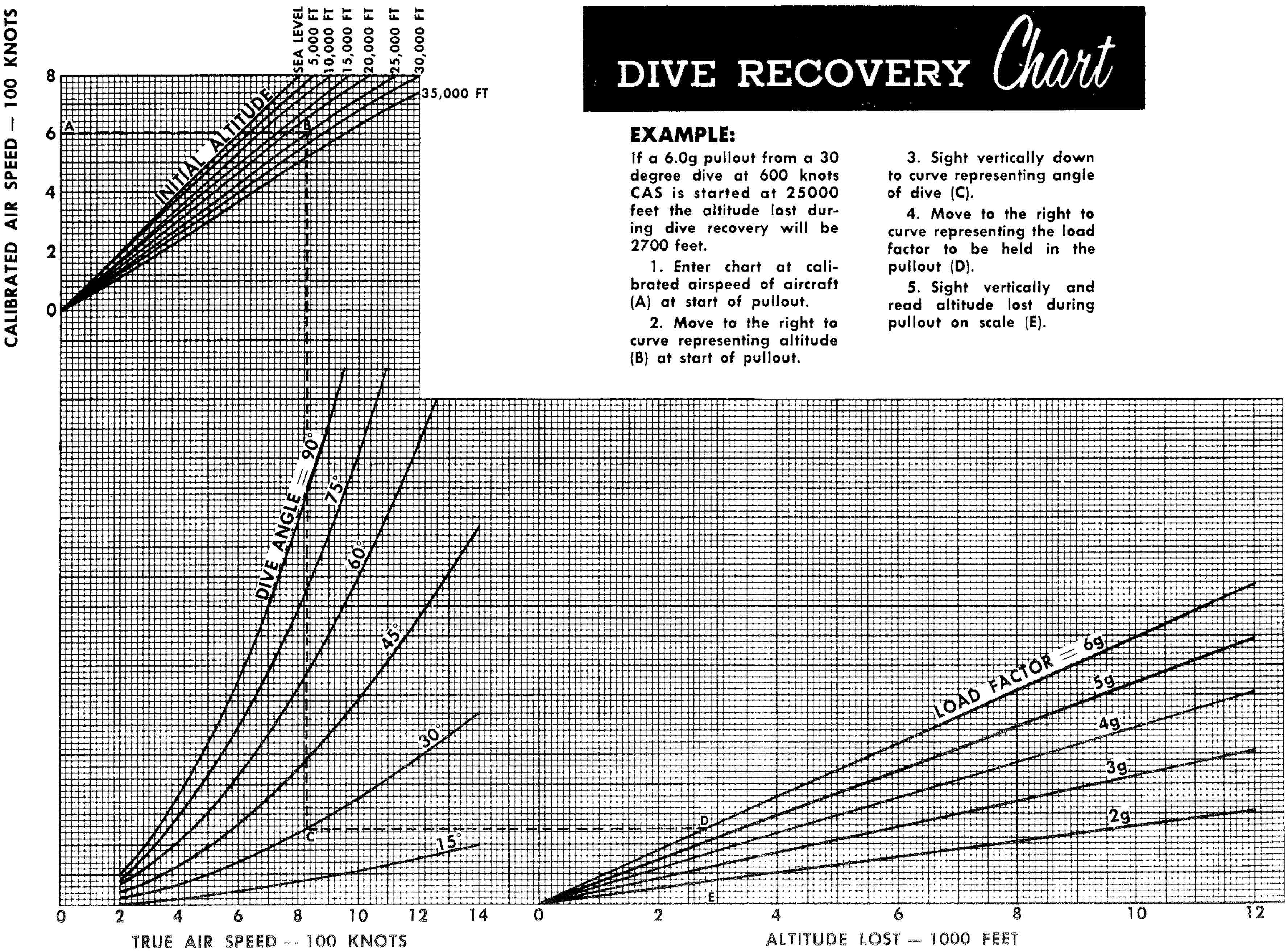


Figure 6—5

be used for spin recovery. Use of the drag parachute will probably result in some damage to the drag chute compartment doors because of the large riser angles. However, this should not preclude use of the drag chute in emergencies. After the spin rotation has been stopped and the airspeed starts increasing, jettison the chute in accordance with the procedure set forth in Section 1.

INVERTED SPINS.

As mentioned above, improper recovery procedure from a normal erect spin may cause the spin to become inverted. This is recognized by the negative Gs and the tendency for the pilot's hands and feet to pull away from the controls. The aircraft is easily recoverable from an inverted spin using the following recovery procedure.

1. Neutralize controls.
2. Apply full rudder opposite to the direction of rotation.

WARNING

The direction of rotation in an inverted spin is easily confused. Therefore, in the inverted spin if the recovery is not accomplished in two turns, the pilot has interpreted the spin direction improperly and the rudder should be reversed.

FLIGHT CHARACTERISTICS WITH EXTERNAL STORES.

Various combinations of external tanks, bombs, rockets and special stores can be flown to very high Mach Numbers with good flight characteristics. The aircraft with external stores is limited from higher speeds only by a slight directional and lateral instability which is easily recognized by the pilot as a "snaking" and roll wobbling effect. External stores tend to increase the severity of buffet during accelerated G maneuvers so that there is ample warning prior to any stall or pitchup.

WARNING

Abrupt, large displacements of the rudder should be avoided with external tanks installed. In this configuration excessive sideslip angles can lead to dangerous snap maneuvers. External stores, and/or extended speed brakes, tend to further aggravate this condition.

- With unsymmetrical loadings, such as the special store installed, the aircraft will tend to roll off at high speeds. If severe roll off is encountered, slow down by reducing power. Applying G to the aircraft will increase the roll off tendency.

ASYMMETRIC FLIGHT CHARACTERISTICS.

Flight tests on aircraft not incorporating spoilers have revealed that asymmetrical loading to a maximum of a 450-gallon tank on an inboard pylon is feasible provided that the following precautions are observed. The asymmetrical flight characteristics for those aircraft incorporating spoilers are improved at moderate to high speeds.

INBOARD PYLON.

One Full 450-Gallon Tank or 230-Gallon Tank or Special Store.

1. Since the aircraft will have a tendency to roll into the store on takeoff, lateral and rudder trim should be established to counteract this effect before takeoff. For the 450-GAL tank configuration approximately 2½ inches of lateral displacement will be required with the rudder trimmed against yaw to the light out position at 165 knots, takeoff speed. Wtih the 230-GAL tank or the external store configuration a somewhat less trim setting may be required.

2. During flight with gear and flaps up on non-spoiler aircraft, increase in lateral control is required with increasing speed, the amount of lateral control also dependent on the altitude and the size of the external store. At the higher airspeeds, above 520 KIAS, 5,000 feet pressure altitude, full 450-GAL tank configuration, available lateral control may run out, however, with spoilers, lateral control will be available at a higher speed. Speeds in excess of those requiring more than ½ stick deflection is not recommended as requirements may suddenly increase with increasing G or gust loads.

WARNING

Whenever loss of hydraulic pressure is encountered, it is recommended that the external store be jettisoned immediately in order to preclude loss of lateral control.

3. With the aircraft in the leading configuration lateral control diminishes with decreasing airspeed until at approximately 150 KIAS, lateral control becomes negligible. Therefore, approach speed should be maintained above 180 KIAS with touchdown speed at 165 KIAS.

OUTBOARD PYLON.

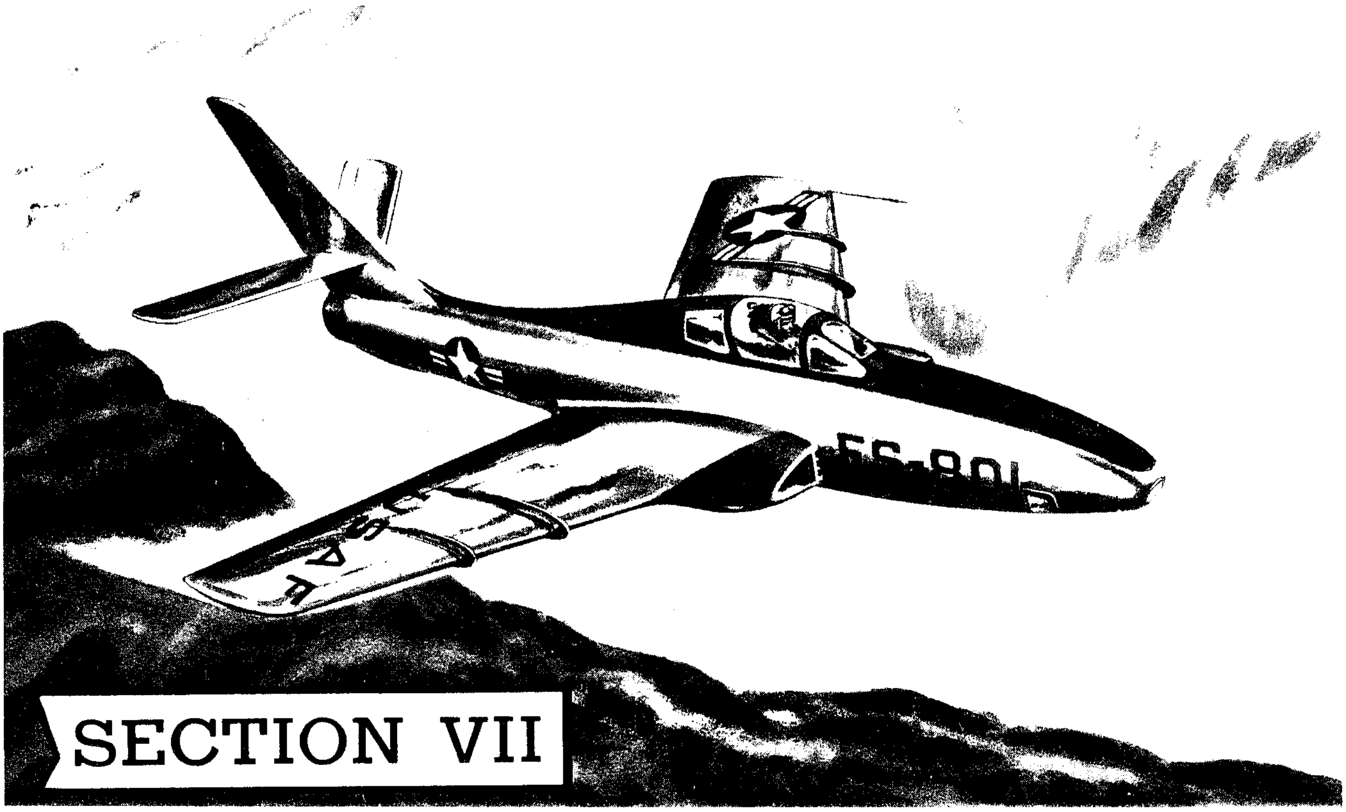
One Full 230-Gallon Tank.

1. Takeoff in this configuration is prohibited.

2. The aircraft should not be flown below 180 KIAS during any phase of flight. Flight tests have revealed that aileron correction between 250 and 460 KIAS at 5,000 feet pressure altitude should be less than that for the 450-GAL tank inboard configuration.

3. With the aircraft in the landing configuration lateral control diminishes with decreasing airspeed until below approximately 175 KIAS it becomes marginal. It is recommended, therefore, in the event the full tank cannot be jettisoned, landing be accomplished at the minimum final approach speed of 200 KIAS and minimum touchdown speed of 180 KIAS.

4. If on takeoff an outboard tank is inadvertently lost, the opposite tank should be jettisoned immediately since lateral control will be insufficient until an airspeed of from 185 to 200 KIAS is attained.



Systems Operation

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ENGINE.

FUEL CONTROL.

The essential purpose of the fuel control system is to regulate thrust as a function of throttle position. Each position of the throttle always corresponds to the same percentage of maximum available thrust. The control also provides protection against overspeed, overtemperature, compressor surge and combustion flame-out. The top speed is established as high as possible without exceeding the maximum allowable turbine stresses, in order to obtain the maximum possible thrust. A small increase in speed above maximum greatly increases the turbine stresses. Furthermore, increasing the speed increases the turbine inlet temperature, and this temperature rise reduces the maximum allowable stresses of the turbine materials. Consequently, any speed increase above maximum will drastically shorten the life of the engine, if not cause immediate failure. In addition to providing steady state speed regulation, the control provides the proper fuel flow during acceleration and deceleration. During an acceleration, the control increases the fuel flow sufficiently to produce the most rapid acceleration of which the engine is capable, without encountering compressor surge or exceeding the maximum temperature limits of the engine. During a deceleration, the control decreases the fuel flow sufficiently to produce a rapid deceleration, yet prevent loss of combustion, or flame-out. A compressor discharge pressure limiter is incorporated in the fuel control which automatically reduces the fuel flow to the engine, whenever the compressor discharge pressure exceeds its limit. This fuel flow reduction will cause a gradual reduction in RPM until the compressor discharge pressure returns to its limit. Reduction in fuel flow will occur at high airspeed, low altitude conditions and should be recognized by the pilot, to avoid unnecessary concern on his part regarding proper functioning of the fuel control. In order to maintain short acceleration times with increasing altitude, and to prevent loss of combustion during deceleration at altitude, it is necessary that the engine idle at progressively higher speeds as the altitude increases. In the event of failure of the main fuel control, the emergency fuel control system is provided which will permit safe operation of the engine. The essential requirement of the emergency system is that it be completely reliable. It must be as simple and as uncomplicated as possible. Therefore, the emergency fuel control consists of a simple throttle valve, and during emergency operation fuel flow is a function of throttle travel only. The compensating features for overspeed, overtemperature, compressor surge, and combustion flame-out have been eliminated to simplify the emergency system as much as possible.

ENGINE CHUGS AND STALLS.

When chugging is encountered during an acceleration there is a momentary RPM hesitation without an exhaust gas temperature rise. When a stall is encountered, the RPM hangs up and then starts dropping as exhaust gas temperature rapidly increases. The stall may or may not be accompanied by chugging. If a stall or severe chugging is encountered on the ground, recover by reducing throttle, shut down the engine and investigate. To recover from either a chug or a stall while in flight, reduce throttle. If the stall persists, increase airspeed by reducing altitude. After recovery advance throttle cautiously to desired power. Have this condition investigated as soon as possible.

EXHAUST GAS TEMPERATURE STABILIZATION.

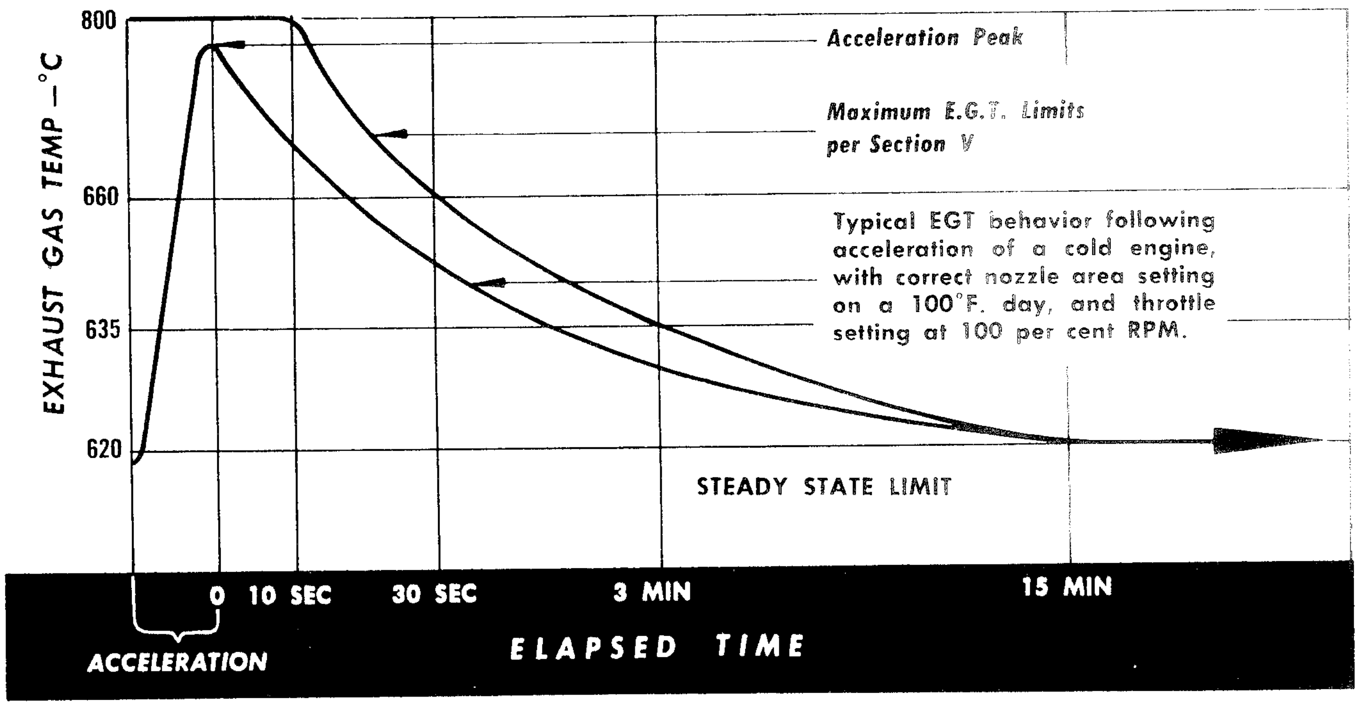
Approximately 8-10 minutes of steady operation [illegible] 100 percent RPM is normally required [illegible] obtain stabilized exhaust gas temperature readings after initiating the start of a cold engine. The [illegible] time required for stabilization will be directly affected by the length of time the engine is [illegible] speeds prior to advancing the throttle [illegible] RPM, and the ambient temperature [illegible] up. Assume that a start is performed [illegible] engine and the throttle is immediately [illegible] obtain 100 percent RPM. As the engine [illegible] the turbine casing expands more rapidly than the turbine wheel. The net effect is a temporary increase in the clearance between the rotor blade tips and the shrouds. When the clearance is at maximum, the turbine efficiency is poorest and the exhaust gas temperature is highest. As the engine is run at full RPM, the turbine wheel heats up and expands, the blade tip clearance diminishes, the turbine efficiency improves and the exhaust gas temperature declines until stabilization is achieved. This is graphically illustrated in figure 7-1 under extreme conditions, i.e. rapid acceleration (idle to 100 percent RPM) of a cold engine immediately following a start on a 100 degree F day. A cold engine is defined as an engine which has been inactive for a sufficient period of time to allow cooling to ambient temperature.

Note

- The criteria illustrated in figure 7-1 are to be used as a guide only and are applicable only to EGT behavior during ground operation of the engine under the conditions mentioned above.

EGT BEHAVIOR Following Cold Engine Acceleration

J65-3 Engine



J65-7 Engine

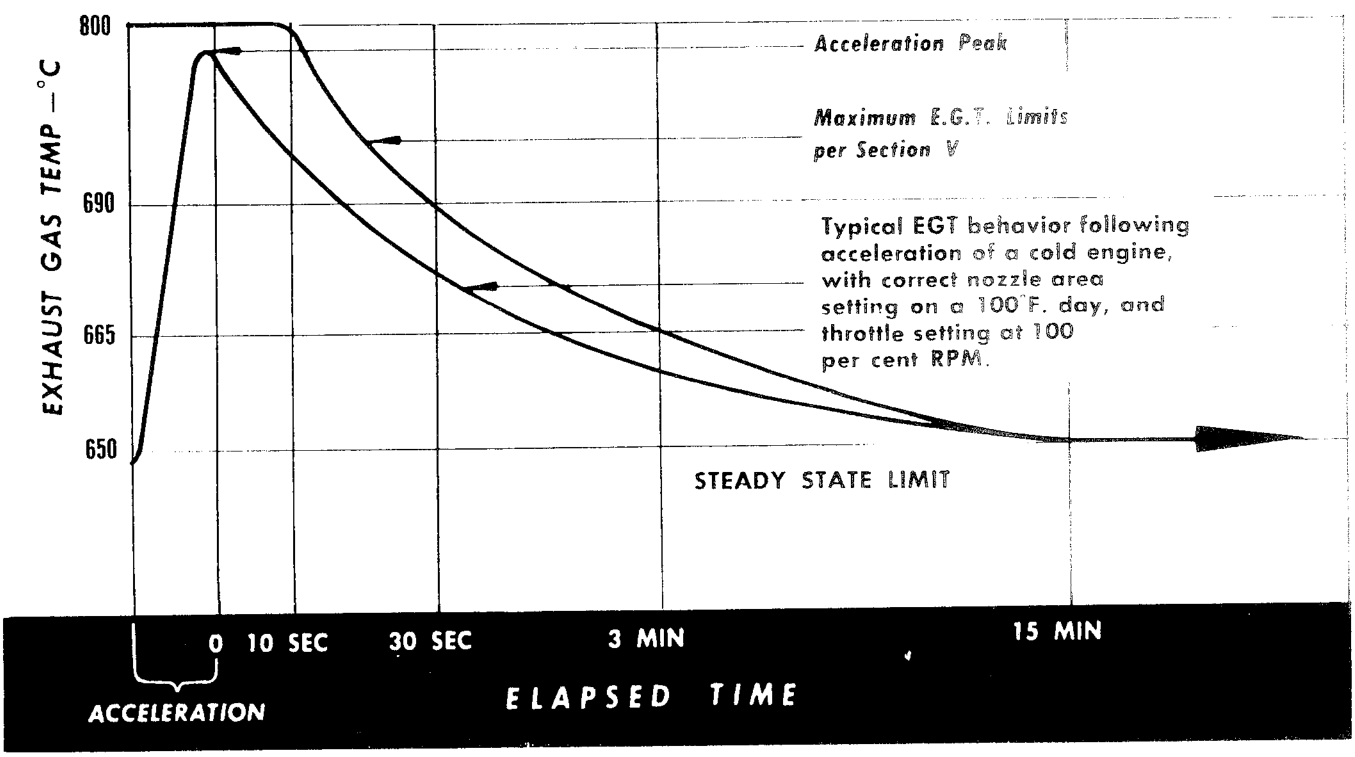


Figure 7—1

STABILIZED Exhaust Gas Temperature at 100% RPM After 15 Minutes Operation

THESE FIGURES ARE FOR GROUND OPERATION INFORMATION ONLY

J65-3 ENGINES

AMBIENT TEMP °C	AMBIENT TEMP °F	MAX EGT °C	MIN EGT °C
37.8	100	620	600
32.2	90	618	598
26.7	80	616	596
21.1	70	614	594
15.5	60	612	592
10.0	50	611	591
4.4	40	609	589
−1.1	30	607	587
−6.7	20	605	585
−12.2	10	603	583
−17.8	0	601	581
−23.3	−10	599	579
−28.9	−20	597	577
−33.3	−30	595	575
−40.0	−40	593	573
−45.5	−50	592	572
−51.1	−60	590	570
−53.9	−65	589	569

J65-7 ENGINES

AMBIENT TEMP °C	AMBIENT TEMP °F	MAX EGT °C	MIN EGT °C
37.8	100	650	630
32.2	90	649	629
26.7	80	648	628
21.1	70	647	627
15.5	60	645	625
10.0	50	644	624
4.4	40	643	623
−1.1	30	642	622
−6.7	20	641	621
−12.2	10	640	620
−17.8	0	639	619
−23.3	−10	638	618
−28.9	−20	637	617
−33.3	−30	636	616
−40.0	−40	635	615
−45.5	−50	634	614
−51.1	−60	633	613
−53.9	−65	632	612

Figure 7—2.

- Under the most extreme conditions (immediate takeoff following cold engine start on a 100°F day), 10-15 minutes may elapse before exhaust gas temperature stabilization is achieved.

Under normal pre-takeoff procedures approximately 5-8 minutes of engine operating time precedes the takeoff roll. Exhaust gas temperature stabilization time should be of slight duration depending upon the following variables; length of time the engine is operated and throttle manipulation during the period prior to takeoff, ambient temperature, tailpipe nozzle area, etc. The decline rate from peak temperature to stabilization should be more rapid than during ground operation due to the influence of airspeed on engine temperature. Under the most extreme conditions (immediate takeoff following cold engine start on a 100 degree F day), 10-15 minutes may elapse before exhaust gas temperature stabilization is achieved. In such a case, and after the aircraft is airborne, the exhaust gas temperature should be monitored, if necessary, to a point on or below the maximum steady state allowable (620 degree F, J65-3 and 650 degree C, J65-7). Power may be gradually increased as stabilization is approached. Also to be considered are the factors which influence exhaust gas temperature soaking effects. Lower ambient temperatures encountered as the aircraft increases in altitude and characteristics of fuel flow scheduling will account for part of the exhaust gas temperature sag at the medium altitudes. However, at altitudes of 35,000 feet and up, the exhaust gas temperature tends to rise due to the characteristics of the engine and fuel control combination. Both of these later manifestations may occur regardless of how long the engine has been running and should therefore not be confused with exhaust gas temperature stabilization behavior.

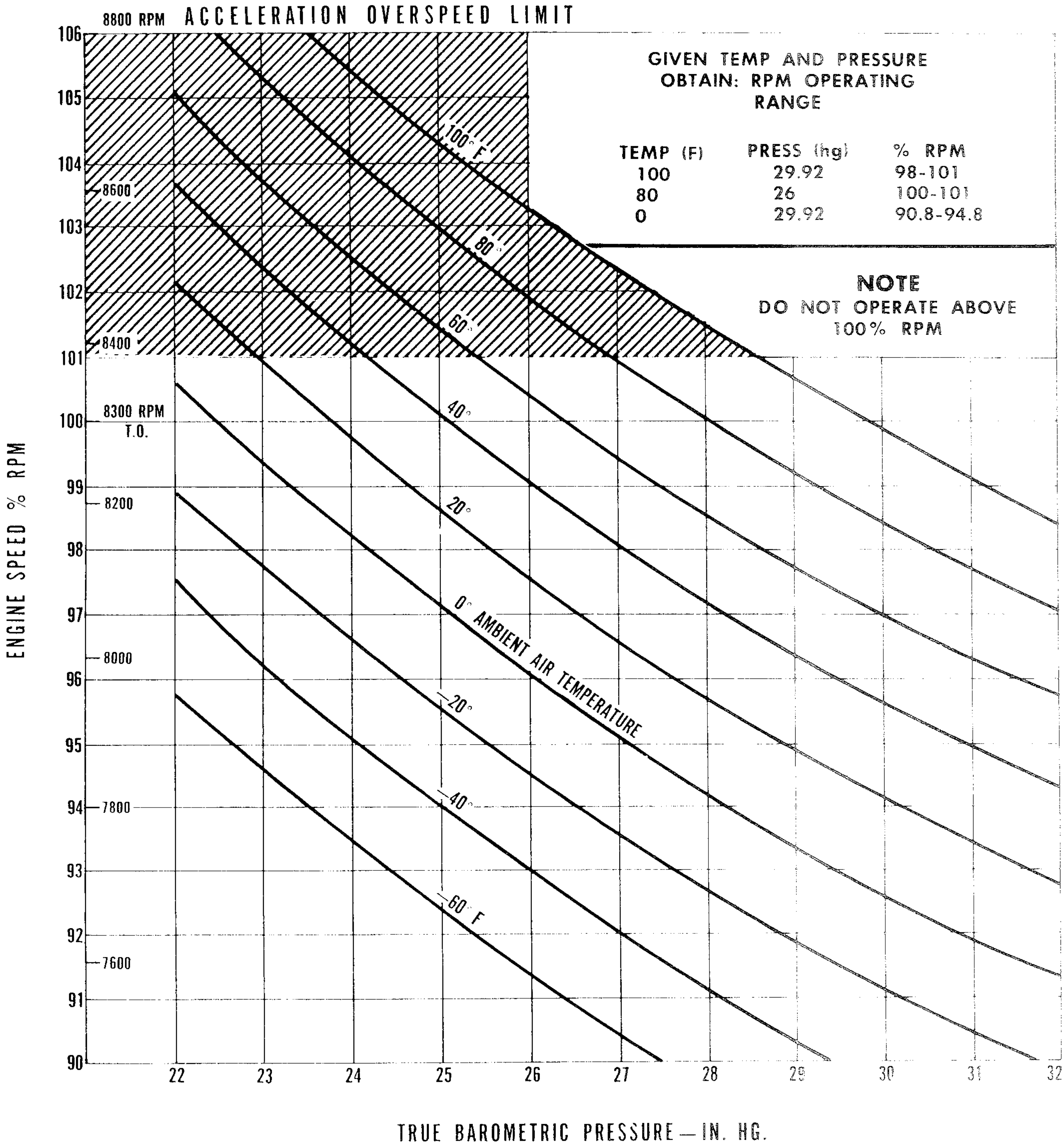
EMERGENCY FUEL SYSTEM CHECK.

1. With the engine on the normal fuel system, set the RPM at 90 percent and place the emergency fuel switch in the ON position. The emergency fuel system warning light should illuminate.
2. If the RPM changes, this indicates that the shift has been made to the emergency fuel system.
3. If there is no RPM change, advance the throttle cautiously to full throttle or 100 per cent whichever comes first. Less than 100 percent RPM at full throttle on emergency fuel system shows the control has shifted satisfactorily.

J65 EMERGENCY FUEL SYSTEM

ESTIMATED FULL THROTTLE RPM

RPM ACCURATE TO ± 2%



TEMP (F)	PRESS (hg)	% RPM
100	29.92	98-101
80	26	100-101
0	29.92	90.8-94.8

Figure 7-3

FUEL SYSTEM MANAGEMENT

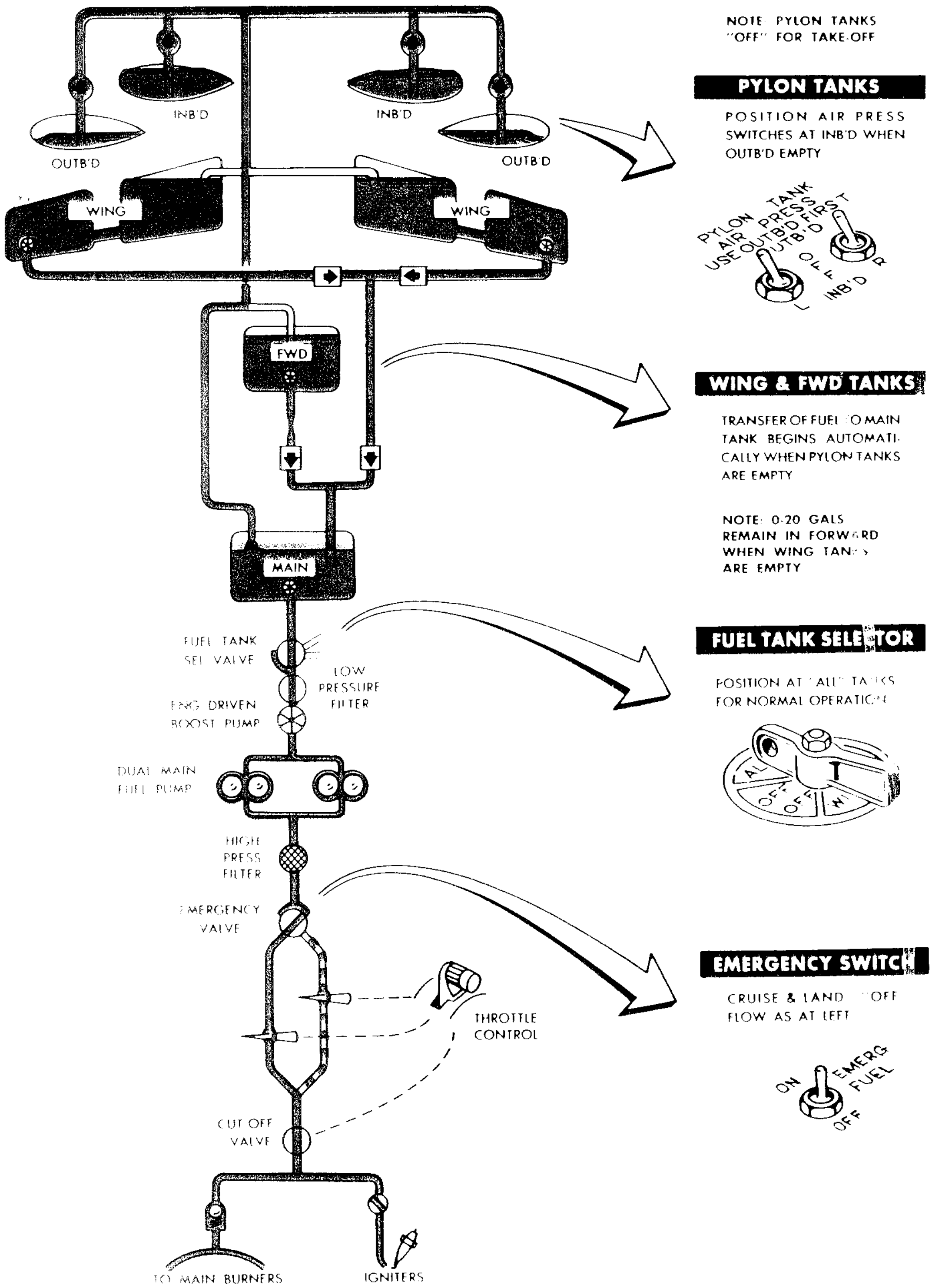


Figure 7—4.

Note

If there is no RPM change when shifting to emergency and 100 percent RPM was obtained either the throttle stop is not set correctly or the control is not shifting from the normal system. This condition should be investigated and corrected.

4. Retard throttle to IDLE and as RPM passes 60 percent place the emergency fuel switch in the OFF position. This returns the engine to the normal system.

ENGINE OIL PRESSURE.

Loss of engine oil pressure occurs when the aircraft is flown in a zero G attitude and does not return to normal for considerable time, even though the aircraft is returned to the normal flight attitude. The engine oil pressure will recover to normal in one minute or less, if the engine RPM is not decreased. The oil pressure will not recover in one minute, if the throttle is retarded when zero oil pressure is experienced. Therefore, it is recommended that engine RPM not be decreased when the oil pressure drops to zero during zero G maneuvers.

STARTING SYSTEM.

The starting system is automatic. However, the starter switch should be held for a minimum of one second to assure energizing the starter and engine ignition systems and the primer timer units. The starter turbine reaches a speed of 44,000 RPM in 3.0 to 3.5 seconds to turn the engine up to a self-sustaining speed of 2,000 RPM through a two-stage planetary-gear reduction with a ratio of 22:1. The starter is engaged to the engine with a spline drive with an overrunning clutch, and no jaw or advancement mechanism is required.

ENGINE SCREENS.

The engine screens are installed to prevent foreign objects from entering the engine compressor section during ground operation or during flight when firing the guns or other instances when foreign objects may present a hazard to flight. Performance is somewhat penalized during flight with the screens extended. Test data shows that the extension of the screens reduces the rate of climb between 300 and 475 FPM, depending on altitude, and increases the time to climb to 40,000 feet by 1.5 minutes. Maximum sea level static thrust is reduced by approximately 4.4 percent. Extension of the screens produced a negligible effect on level flight maximum speed, exhaust gas temperature, fuel used in the check climb and the acceleration of the aircraft and engine. Tests have also shown that the screens have excellent debris retention characteristics. However, there is a remote possibility that some forms of debris caught by the screens can peel off the screen edge and be sucked into the compressor during the retraction of the screen. Therefore, where possible, screens should not be retracted until flight conditions make it possible for the pilot to use the established emergency procedures in the event of extreme foreign object damage to the engine.

FUEL FILTER ICING.

The fuel filter ice warning light (if installed) will not illuminate if icing occurs in the low pressure fuel filter. This condition will be indicated to the pilot by a gradual reduction in engine RPM and fuel flow only. If fuel filter icing is suspected proceed as follows:

1. Aircraft with active fuel filter de-icing system:
   a. Fuel filter de-icing switch — MANUAL until engine RPM increases to normal operating requirements.
2. Aircraft with inactive fuel filter de-icing system:
   a. Descend to an altitude below freezing level if practicable.
   b. Proceed to nearest suitable landing field.
   c. Land as soon as external fuel is consumed.
   d. If drop in fuel flow and RPM occurs, land as soon as possible.

FUEL SYSTEM MANAGEMENT.

During normal operation fuel is transferred first from the outboard pylons, then the inboard pylons to the main tank so that the pylons can be dropped to reduce drag. Fuel is then transferred simultaneously from the wing and forward tanks to the main tank at varying flows so that the wing tanks will empty while there still remains zero to approximately 20 GAL in the forward tank. The fuel in the main tank is then consumed. Inasmuch as the normal fuel system is fully automatic, with the tank selector in the ALL TANKS position, the pilot does not have to select the various fuel flow patterns, except for manual selections of the external tanks air pressure. If it is necessary to operate on either the wing or forward auxiliary fuel flow patterns, the fuel tank selector should not be turned through the OFF position, as a flame-out may occur due to a lack of fuel supply. The emergency fuel switch must be in the OFF position for all normal operations.

Note

When operating on internal fuel alone, monitor the fuel level in the main tank by placing the fuel tank quantity selector switch in the main position and fuel quantity check switch in SELECT.

FUEL TANK SELECTOR.

If the fuel tank selector is turned to the OFF position, immediately after closing the throttle, the fuel tank selector becomes tightly bound on some aircraft and cannot be moved to any other position without extreme difficulty. This is caused by a fuel pressure build-up between the fuel tank selector valve and the engine driven pump as the engine coasts to a stop. If this condition exists prior to starting, free the fuel tank selector by holding pressure against the fuel tank selector toward any on position, and make a normal start. As the engine picks up speed, the selector will free and if positioned to a tank containing fuel, a normal start will follow.

OPERATION WITH VARIOUS FUELS.

Fuel in accordance with Spec MIL-J-5624, grade JP-4 is the recommended fuel for use in the aircraft. Continued use of jet fuel contaminated in excess of one percent of AVGAS, will cause failure of the vaporizing tubes and subsequent turbine blade damage resulting in a potential safety of flight hazard. The continued use of jet fuel contaminated with less than one percent of AVGAS is permissible.

PYLON TANK FEEDING.

All pilots will determine if pylon tanks are feeding immediately after takeoff and or immediately after air refueling. If pilots cannot determine beyond reasonable doubt that pylon tanks are feeding, the mission will be aborted. Check pylon tank feeding as follows:

CAUTION

It is recommended that when operating with Type I or modified Type II pylon tanks installed, the pilot will check individual tank readings at least every 15 minutes until external tanks are emptied. This is to assure that external fuel is transferring properly into the internal tanks.

Note

When operating on external tanks monitor the fuel level in the forward tank by placing the fuel tank quantity selector switch in the FWD position and the fuel quantity check switch in the FUEL QTY CHECK position except when flying with Type IV or unmodified Type II and operating at low level.

With Type I or Modified Type II 230-GAL Tanks.

1. Prior to takeoff check fuel of each internal and external tank. Note fuel level.
2. Takeoff will be accomplished with pylon tanks unpressurized.
3. Pylon tanks will be pressurized immediately after takeoff.
4. Fuel tank quantity selector switch will be used to check the feeding of each pylon tank and to cross-check the fuel load in each internal tank.

Note

It is recommended that the means of fuel feeding of each external tank be established within a maximum of 15 minutes after take-off.

With Type I 450-GAL Tanks.

1. Takeoff will be accomplished with pylon tanks unpressurized.

Note

Actuate air refueling receiver switch to OPEN momentarily (for receiver equipped aircraft) or ready switch momentarily to READY (for probe equipped aircraft) to depressurize tanks.

2. Pylon tanks will be pressurized immediately after takeoff.
3. Fuel tank quantity selector switch will be used to check the feeding of each pylon tank and to cross check the fuel load in each internal tank.

Note

It is recommended that the status of fuel feeding of each external tank be established within a maximum of 15 minutes after take-off.

With Unmodified Type II or IV 230-GAL Tanks.

These tanks do not afford the pilot an immediate indication of fuel feeding in the cockpit and should be checked for feeding as follows:

Note

This check should be complete within 20 minutes after takeoff.

1. Take-off will be accomplished with pylon tanks unpressurized.
2. Place the fuel tanks quantity selector switch in the FWD position.
3. The left outboard pylon tank (if installed) will be pressurized. After a definite indication that the forward tank has refilled, the left outboard pylon tank air pressure switch will be turned to the OFF position.
4. The right outboard tank (if installed) will be pressurized. After a definite indication that the forward tank has refilled, the right outboard pylon tank air pressure switch will be turned to the OFF position.
5. Proceed as in steps 3 and 4 with the left inboard and right inboard tanks, if applicable.
6. If certain all tanks are feeding, pressurize both outboard pylon tanks and when empty pressurize both inboard pylon tanks.

With Type II or IV 450-GAL Tanks.

These tanks do not afford the pilot an immediate indication of fuel feeding in the cockpit and should be checked as follows:

Note

This check should be complete within 20 minutes after takeoff.

1. When fuel permits, a ground check of the operation of each 450 GAL pylon tank will be accomplished by placing the fuel tanks quantity selector switch in the FWD position and checking to determine that the tank is feeding.
2. Takeoff will be accomplished with pylon tanks unpressurized.

Note

Actuate air refueling receiver switch to OPEN momentarily (for receiver equipped aircraft) or ready switch momentarily to READY (for probe equipped aircraft) to depressurize tanks.

3. Place the fuel tanks quantity selector switch in the FWD position.
4. The left inboard pylon tank (if installed) will be pressurized. After a definite indication that the forward tank has refilled the left inboard pylon tank air pressure switch will be turned to the OFF position.

Note

Actuate air refueling receiver switch to OPEN momentarily (for receiver equipped aircraft) or ready switch momentarily to READY (for probe equipped aircraft) to depressurize tanks.

5. The right inboard pylon tank (if installed) will be pressurized. After a definite indication that the forward tank has refilled, the left inboard tank pylon tank will again be pressurized.

IF PYLON TANK STOPS FEEDING.

When a pylon tank stops feeding, there may be a slight rumble accompanied by vibration from below the pilot's seat area. This condition is of short duration and is caused by air surges into the main tank or as the pressure reducing valve, which controls both the cockpit seal and pylon tank air pressure, fluctuates on stabilizing for the new required air flow.

PYLON TANK PRESSURIZATION.

Suction relief valves are not incorporated in Type II external tanks, therefore, the air pressure must be maintained to prevent the tanks from collapsing during high rates of descent. If inboard pylon tanks are retained, the pylon tanks air pressure switches should be left in the INB'D PYLON TANKS AIR PRESS position after the tanks are empty. However, if inboard and outboard tanks are both retained, descents must be made at a slow rate and the pylon tanks air pressure switch alternately placed in the INB'D PYLON TANKS AIR PRESS and OUTB'D PYLON TANKS AIR PRESS position at approximately every 1000 feet of descent in order to equalize the pressure differential and prevent the tanks from collapsing.

PYLON TANK AIR PRESSURE SWITCH.

When the aircraft is serviced on the ground, the feed lines from the wing tanks to the main tank may be filled with air. The trapped air is not bled out until the main tank level drops and allows the main tank shutoff valve, from the wings and forward tanks, to open. If the fuel tank selector is positioned to WING AUX before the wing transfer lines are filled with fuel, the engine will be momentarily starved and a flame-out will result. With the pylon tank air pressure switches OFF enough fuel will be consumed from the main tank during ground operation through takeoff to start transfer of fuel from the forward and wing tanks to the main tank. This should bleed the wing and forward transfer lines sufficiently.

ELECTRICAL LOAD CHART

COMPONENT Components requiring very little amperage are not listed	APPROXIMATE AMPERAGE REQUIRED	PERCENT OF TOTAL AMPERAGE AVAILABLE GEN 1 and 2 (600 AMPS)	GEN 1 ONLY (400 AMPS)	GEN 2 ONLY (200 AMPS)
Emergency Hydraulic Pump	* 90.	* 15.	* 22.5	* 45.
Main Inverter	68.5	11.4	18.75	37.5
Alternate Inverter	* 14.6	* 2.43	* 3.65	* 7.3
Main Tank Booster Pump	37.4	6.23	** 9.35	** 18.7
Forward Tank Booster Pump	30.4	5.6	** 7.6	** 15.2
Wing Tanks Booster Pumps (2)	65.2	10.86	** 16.3	** 32.6
Camera System	66.0	11.0	INOPERATIVE	
Communication Equipment	37.0	6.13	9.25	18.5
Pitot Heater	7.0	1.16	1.75	3.5
Flight Instruments	6.6	.93	1.4	2.8
Misc.	24.0	4.0	6.0	12.
TOTAL EXCLUDING EMERGENCY*	341.1	59.74%	67.25 to 75.95%	89.5 to 106.9%

*Emergency Use Only
**Fuel Booster Pumps will be inoperative in two tanks during single generator operation

Figure 7—5.

ELECTRICAL POWER SUPPLY SYSTEM.

The electrical load chart (figure 7—5) lists the major components which require comparatively high amperage while in operation so that a pilot can ascertain that the loadmeter reading is normal prior to takeoff. Components requiring very little amperage are not listed.

HYDRAULIC POWER SUPPLY SYSTEM.

EMERGENCY HYDRAULIC POWER.

The emergency hydraulic system can only be tested when the engine is not operating. With external power available or the battery switch ON and the instrument power switch in the ALT position check the utility and power hydraulic pressure gages for zero readings. If gages do not read zero, cycle controls to dissipate pressure. The emergency hydraulic system accumulator should have a charge of 500 to 600 PSI air pressure when the hydraulic pressure is zero. Place the emergency hydraulic pump switch in the HYD EMER PUMP position, the power hydraulic gage should indicate 1500 (+100) PSI. Move control stick full forward. With hydraulic pressure at 1500 (+100) PSI control stick should operate smoothly and steadily from full forward to full aft, then to full forward and then, the stick action may then become restricted during the next aft travel due to the emergency pump limitations. Rate of motion fore and aft must be equal. Return control stick to neutral and observe hydraulic pressure return to 1500 (+100) PSI. Operate ailerons through complete range at normal rates. Operation should be smooth and steady with no indication of system starvation or reduction in rate of operation. Rate of motion from side to side must be equal.

LANDING GEAR SYSTEM.

1. The nose wheel strut should have the correct extension. A bottomed or low strut will increase the nose wheel "unsticking speed" to above the takeoff speed. Consequently this will increase the takeoff speed and ground roll.

2. A vibration, felt after takeoff when the landing gear is retracted may be mistaken for engine roughness but could be caused by an unbalanced nose wheel. This vibration would be more noticeable after takeoff in the heavier configuration as the takeoff speed is higher. Vibration, caused by the nose wheel, will diminish as the wheel coasts to a stop.

USE OF LANDING WHEEL BRAKES.

It is absolutely necessary that airplane brakes be treated with respect. Consideration must also be given to airplanes equipped with wheel brake anti-skid systems. Although the anti-skid system will give consistently shorter landing distances on dry runways, it should not be used to its maximum potential to purposely make all landing rolls as short as possible. Generally, operating personnel stop the aircraft as quickly as possible regardless of the length of the runway, use the brakes consistently for speed up turns, and drag the brakes while taxiing. Brakes, themselves, can merely stop the wheel from turning, but stopping the aircraft is dependent on the friction of the tires on the runway. For this purpose, it is easiest to think in terms of coefficient of friction which is equal to the frictional force divided by the load on the wheel. It has been found that optimum braking occurs with approximately a 15 to 20 percent rolling skid, i.e. the wheel continues to rotate but has approximately 15 to 20 percent slippage on the surface so that the rotational speed is 80 to 85 percent of the speed which the wheel would have were it in free roll. As the amount of skid increases beyond this amount, the coefficient of friction decreases rapidly so that with a 75 percent skid the friction is approximately 60 percent of the optimum and, with a full skid, becomes even lower. There are two reasons for this loss in braking effectiveness with skidding. First, the immediate action is to scuff the rubber, tearing off little pieces which act almost like rollers under the tire. Second, the heat generated starts to melt the rubber and the molten rubber acts as a lubricant. NACA figures have shown that for an incipient skid with an approximate load of 10,000 pounds per wheel, the coefficient of friction on dry concrete is as high as 0.8, whereas the coefficient is of the order of 0.5 or less with a 75 percent skid. Therefore, if one wheel is locked during application of brakes, there is a very definite tendency for the aircraft to turn away from that wheel and further application of brake pressure will offer no corrective action. Since the coefficient of friction goes down when the wheel begins to skid, it is apparent that a wheel, once locked, will never free itself until brake pressure is reduced so that the braking effect on the wheel is less than the turning moment remaining with the reduced frictional force. To minimize brake wear, the following precautions should be observed insofar as is practicable.

1. Optimum approach and landing speeds should be used or ground roll distances will increase accordingly.

2. The full length of the runway should be used to take advantage of aerodynamic braking and to use the brakes as little and as lightly as possible when bringing the aircraft to a stop.

3. Use extreme care when applying brakes immediately after touchdown, or at any time when there is considerable lift on the wings, to prevent skidding the tires and causing flat spots. A heavy brake pressure can result in locking the wheels more easily if brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the aircraft is on the wheels. A wheel once locked in this manner 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.

4. Antiskid systems are intended to prevent skids at high speed under light wheel loads. Therefore, brakes equipped with an antiskid system may be applied immediately after touchdown, but this should be done only when definitely necessary. The antiskid system will function to prevent tire skidding if it is operating properly, however, it is not designed to perform as a completely automatic braking system. Continuous braking from the point of touchdown will result in considerable overworking of the antiskid system beyond design limits in addition to causing excessive wear and extreme heating of the brakes.

5. For short landing rolls, a single, smooth application of the brakes with constantly increasing pedal pressure is most desirable. This procedure applies equally well for operation on emergency braking systems.

6. If maximum braking is required after touchdown, lift should first be decreased as much as possible by raising the flaps and dropping the nose before applying brakes. This procedure will improve braking action by increasing the frictional force between the tires and the runway.

7. It is recommended that a minimum of 15 minutes elapse between landings where the landing gear remains extended in the slip stream, and a minimum of 30 minutes between landings where the landing gear has been retracted to allow for cooling if brakes are used for steering, crosswind taxiing operation, or a series of landings are performed. See Section V for brake limitations.

8. After the brakes have been used excessively for an emergency stop and are in the heated condition, the aircraft should not be taxied into a crowded parking area. Peak temperatures occur 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.

9. The brakes should not be dragged when taxiing.

LANDING TECHNIQUE WITH ANTISKID BRAKES.

Minimum Distance Landing Roll.

If minimum landing distance is required, the following procedure is recommended:

a. Apply brake pressure gradually until system cycling is felt. This cycling will cause some pitching of the airplane and/or some pulsing in the brake pedals as pressure is released at the brakes.

b. When cycling is noted, release some of the brake pedal force until cycling ceases.

c. When cycling ceases, gradually apply more pedal force until cycling starts again and then relieve.

Note

A greater amount of pedal force can be applied toward the end of the run as a greater weight is applied to the landing gear wheels due to loss of lift.

- When applying the antiskid brake at high speeds, it is normal that you will receive a pulsing or cycling braking effect due to the mechanism of the brake. When flying an airplane with this brake system installed, you will note a sensation similar to that which you would get by a rapid series of brake applications and releases. That is, when applying braking action, it feels first as if pressure has been applied and then while still applying brake pressure, you feel a sensation as if brakes had been released. This cycling effect gradually decreases in intensity as the airplane speed is reduced.

Normal Landing.

If minimum landing distance is not necessary, the brakes can be applied and held as required which will increase brake and tire life. In this case, system cycling will not be felt on the brake pedals as the condition exists only when the wheels approach the skid condition.

SECTION VIII

Crew Duties

NOT APPLICABLE TO THIS AIRCRAFT

SECTION IX

All Weather Operation

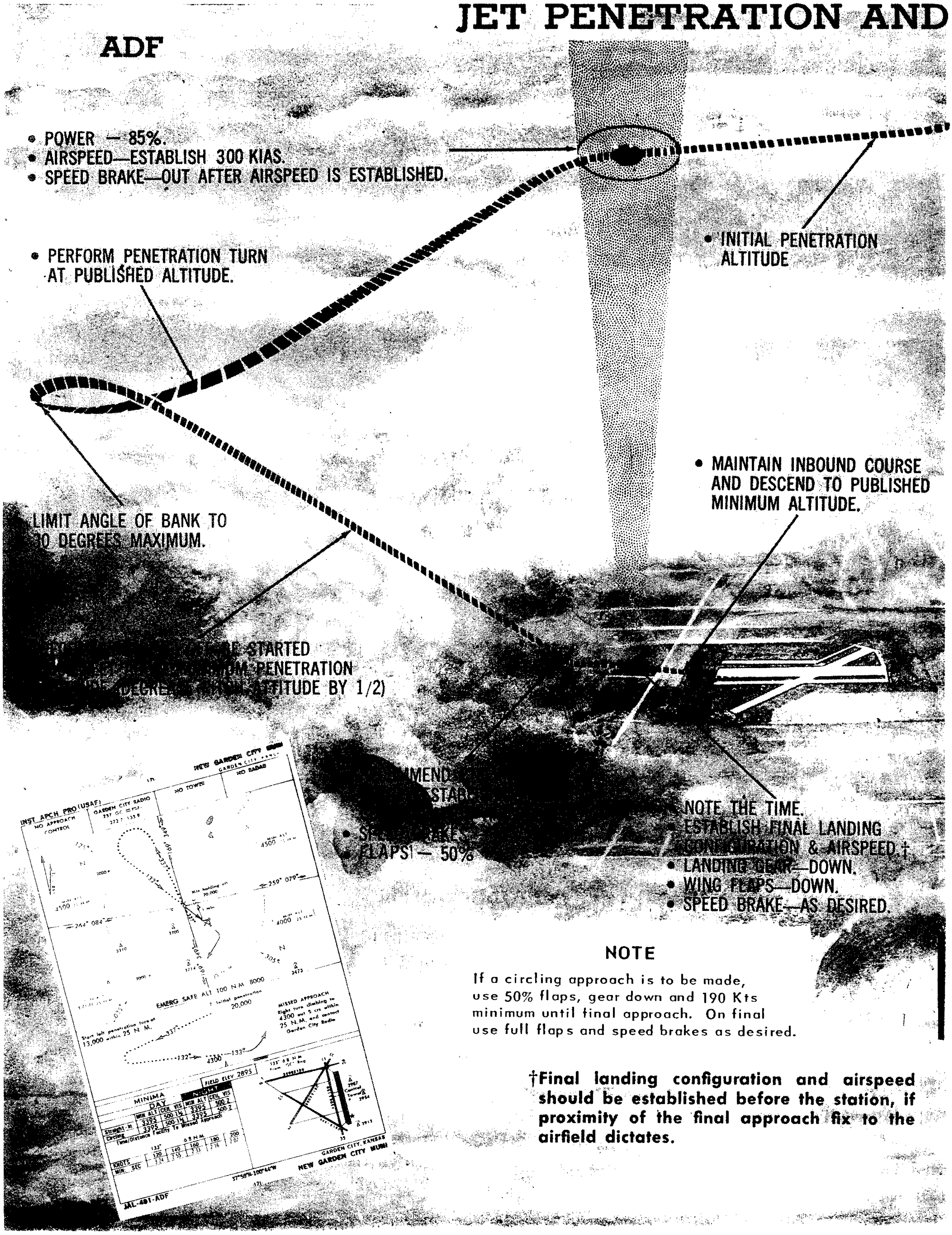
TABLE OF CONTENTS

INTRODUCTION.

This section contains special or unusual procedures for this aircraft. Instrument flight procedures will be accomplished as outlined in USAF Manuals, Instrument Flying Techniques and Theory of Instrument Flying.

OPERATION UNDER INSTRUMENT FLIGHT CONDITIONS.

For ease of handling, bank angles should be limited to approximately 30 degrees unless conditions dictate otherwise. At altitudes above 30,000 feet with an external load, it becomes necessary to limit the bank angle to 20 degrees to prevent loss of altitude during turns. All recommended airspeeds in this section should be adhered to during the final stages of any instrument procedure or approach in minimum weather conditions. During some stages of landing approaches, the aircraft is being operated on the flat portion of the power curve, that is, small changes in power give fairly large changes in airspeed. If airspeed is excessive, especially on the glide path in conditions of low ceiling and visibility, with a wet or icy runway, the distance required to stop will be increased.



NOTE

If a circling approach is to be made, use 50% flaps, gear down and 190 Kts minimum until final approach. On final use full flaps and speed brakes as desired.

†Final landing configuration and airspeed should be established before the station, if proximity of the final approach fix to the airfield dictates.

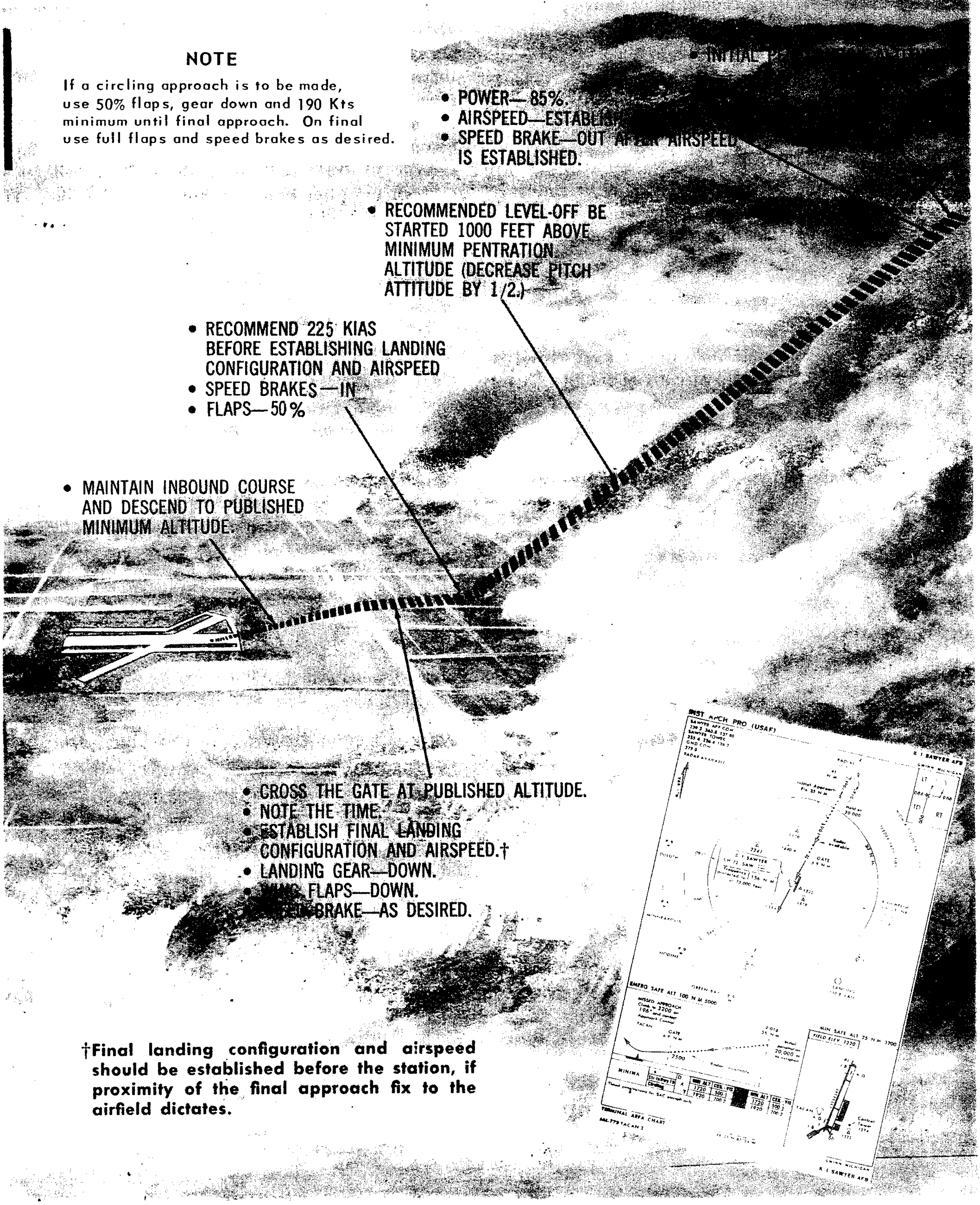
Figure 9-1

LOW APPROACH (TYPICAL)

TACAN

NOTE

If a circling approach is to be made, use 50% flaps, gear down and 190 Kts minimum until final approach. On final use full flaps and speed brakes as desired.



†Final landing configuration and airspeed should be established before the station, if proximity of the final approach fix to the airfield dictates.

INSTRUMENT TAKEOFF AND INITIAL CLIMB.

1. Line up visually with centerline of runway.
2. Heading indicator — Rotate compass card until runway is aligned with top index.
3. Attitude indicator — Adjust miniature aircraft by aligning it with horizon bar.
4. Release brakes.
5. Throttle — MILITARY thrust.
6. Use brakes for directional control until rudder becomes effective. Use heading indicator as the primary instrument for heading until the aircraft is airborne.

Note

> If miniature aircraft was lined up with horizon bar on attitude indicator at start of takeoff, placing miniature aircraft about two horizon bar widths above horizon bar gives a good takeoff and initial climb attitude.

7. Take-off at normal speeds (Refer to Appendix).
8. Maintain a constant nose-high, wing-level attitude, after leaving the ground.
9. Landing gear selector — UP when definitely airborne and the vertical velocity indicator and altimeter show definite ascent.
10. Wing flap control — UP at 190 KIAS minimum.
11. Climb to a safe altitude and trim for best climbing speed.

INSTRUMENT CLIMB.

1. Maintain takeoff attitude until all obstacles are cleared.
2. Accelerate gradually holding takeoff attitude. Intercept climb schedule at 5,000 to 10,000 feet.
3. Limit climbing turns to 30 degrees angle of bank.

DURING INSTRUMENT CRUISING FLIGHT.

Airspeeds between 250 and 350 KIAS provide the best handling qualities and allow time for diversions to accomplish other instrument flight functions such as operating the radio, navigating, etc. Above this range excessive concentration is necessary to maintain pitch attitude.

SPEED RANGE.

The aircraft handles satisfactorily throughout the speed range. The most desirable speed range for individual flights will be governed by the nature of the flight and on local conditions. For the most desirable speed under specific conditions refer to the Flight Operating Charts in the Appendix.

INSTRUMENT DESCENTS.

The optimum power for conservation of fuel during descents is IDLE. However, if descents are made in formation, the lead aircraft must maintain adequate thrust to afford wing ships flexibility of position, otherwise, wing aircraft may periodically be operating in the 60 to 82 percent RPM range. Descending turns become progressively more difficult as the bank angle exceeds 30 degrees. Prior to descending through an overcast use pitot heat and windshield defroster. The defroster should be turned on at least 30 minutes prior to starting descent.

CAUTION

> Monitor the altimeter closely during descents to prevent misreading.

IFR LOITERING AND HOLDING PATTERN.

For best loitering indicated airspeeds (fuelwise), which depend upon altitude and weight of the aircraft, refer to the Maximum Endurance Charts in the Appendix. These speeds should be considered as minimum. Normal holding airspeed is 250 KIAS. Approximately two to four percent RPM must be added to the level flight RPM to maintain airspeed during holding pattern turns.

INSTRUMENT APPROACHES.

All instrument approaches will be in accordance with USAF Manuals, Instrument Flying Techniques and Theory of Instrument Flying, except for recommended airspeeds.

INSTRUMENT LETDOWN.

Typical instrument letdown (TACAN and ADF) procedures are presented in figure 9–1.

RADAR APPROACH.

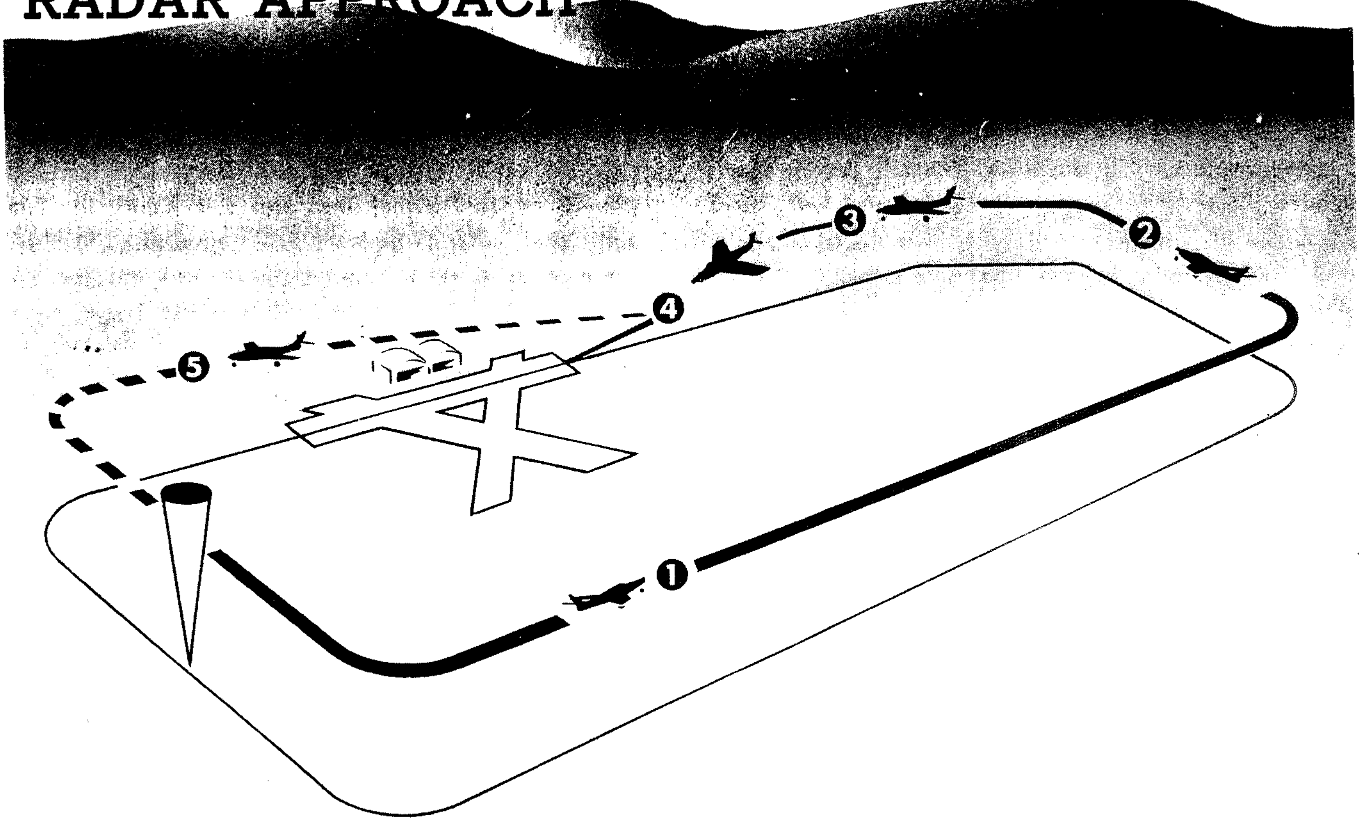
Typical radar approach procedure is presented in figure 9–2.

MISSED APPROACH TECHNIQUES.

Missed approaches after an instrument approach present problems that may be compounded by low ceilings, turbulance, spatial disorientation, precessing gyros, and pilot confusion. It is imperative that a pilot blend the steps into a well practiced pattern.

1. RPM–100%–96% if in formation.
2. Speed brakes–IN.
3. Gear–UP as climb is started.
4. Flaps–50% at 190 IAS.
5. Climb–1500–200 FPM at 225 IAS.
6. Retard throttle to 92%, maintain 225 IAS, and 1500–2000FPM climb.
7. Level Off–Lead by 150 feet, reducing power smoothly to 85%.
8. Execute established missed approach for particular base at 225 KIAS, 50% flaps and 85% RPM.

RADAR APPROACH



NOTES

1. This information applies to the approaches of a single airplane. For formation flights local SOP may dictate minor revisions to this procedure.

2. The data presented in the table is representative of a typical GCA Approach, based on an airplane gross weight of 16,500 pounds (1000 pounds of fuel).

3. Power settings should be changed accordingly, using 1000 pounds of fuel as a base, to increase or decrease final approach airspeed 5 knots for each fuel variation of 1000 pounds. For each increase or decrease of 5 knots an approximate change of 2% of power should give the required power settings.
(Power settings ±2% RPM to maintain IAS at all air temperatures).

Base leg, final approach and glide path airspeeds to be varied in relation to amount of fuel remaining.

	% POWER	KIAS	SPEED BRAKES	FLAPS	GEAR
1 DOWNWIND	85	225	IN	50%	UP
2 BASE LEG	88	190	IN	50%	DOWN
3 FINAL APPROACH	88	165	IN	100%	DOWN
4 GLIDE PATH	88	165	OUT	100%	DOWN
5 MISSED APPROACH	MAXIMUM	225	IN	50%	UP

TIME & FUEL REQUIRED	
TYPICAL RECTANGULAR RADAR APPROACH	374 LBS
TIME	9 MIN
MISSED APPROACH (Go around)	480 LBS
(No conservation factors included)	

Figure 9-2

FLIGHTS IN ICE, SNOW AND RAIN.

The only forward visibility in heavy precipitation is through curved panels of the windshield. Adequate fuel reserve should be allowed for missed radar approach, due to radar controller's difficulty in maintaining contact with the aircraft when precipitation echoes clutter the scopes. Icing has marked effect on wings of this aircraft, notably in reduced airspeed and rate of climb. A check of radiosonde information should be made at the point of departure and destination, and the flight should be planned at ice-free altitudes, due to absence of wing and tail de-icing.

Vision in Rain.

1. If mist or light rain is reported, vision will not be significantly affected during landing. Airflow over windshield will prevent accumulation of water leaving streaks or tracks with good vision between the streaks.

2. If moderate rain is reported, vision forward will be possible, but will be significantly impaired. The airflow over the windshield tends to carry the water away but streaks or tracks will cover a substantial portion of the windshield.

3. If heavy rain is reported, forward vision will be virtually impossible. The windshield will be covered with water.

LANDING ON WET RUNWAYS.

For landing on wet or slippery runways, refer to the Stopping Distance, Runway Correction Reading (RCR) Chart (figures A7-5, A7-6, and A7-7).

Procedure.

An extended traffic pattern is recommended when a wet runway landing is to be made. Prior to landing:

a. The pilot must assure himself sufficient landing pattern spacing so that jet wash from preceding aircraft is at a minimum.

b. The approach speeds for each landing must be computed before attempting a wet surface runway landing. Establish the final approach speed for each gross weight, in accordance with the Appendix. As a rule-of-thumb, starting with 160 knots IAS for 1,000 pounds of fuel remaining, add five knots for each additional 1,000 pounds of fuel remaining. Establish the base leg 1,500 feet above the runway altitude, IAS 190 knots, wheels down, 50 percent flaps, speed brakes in. Make the turn to final at 800-1,000 feet about five miles out, maintain power as required to hold 190 knots IAS, and complete the landing checks, full flaps and speed brakes out. When about two miles from touchdown, reduce the airspeed to the predetermined final approach speed and adjust the power so that a 200 to 300-foot rate of descent can be held to the flare-out point. Immediately prior to touchdown, the power should be reduced to idle. If the drag chute is not available, touchdown in a nose high attitude and apply enough aft stick to hold the nose high thereby obtaining the maximum aerodynamic braking. After the nose wheel contacts the runway raise flaps and use brakes intermittently. Do not drag the tail. If the drag chute is to be used, allow the nose wheel to contact the runway immediately and deploy the drag chute. Maintain directional control by use of the rudder and after the rudder becomes ineffective, use brakes for directional control. If the drag chute fails to deploy, use maximum aerodynamic braking by raising the nose to the optimum position. Leave the flaps extended and after the nose wheel contacts, use brakes intermittently.

c. When landing on a wet surface runway, the first 2,000 feet of roll is the most critical in that the aircraft has a "skimming" tendency until the lift of the wing has dissipated. If brakes are used during this period, they will tend to aggravate this condition, resulting in a severe yaw. It is virtually impossible to determine when one wheel has stopped rotating on a wet runway; therefore, the best assurance against blowing a tire is intermittent braking action, with equal pressure being applied to the brakes. If the aircraft starts to yaw, the pilot must not try to "catch" it by use of asymmetric braking; rather, he should release both brakes, and *after* the aircraft stabilizes, *use the rudder*, and, *only* a light application of brakes.

Note

Due to the wide gear of the aircraft and absence of nosewheel steering, the slightest asymmetric braking, either pilot-induced or caused by varied traction such as on a wet or icy runway, can establish yaw angles. This condition can occur with either standard or ice grip tires. If this occurs, the pilot must release the brakes and allow the aircraft to realign with the runway. This usually results in a longer landing roll.

GROUND HANDLING ON ICE.

Operation of the aircraft on ice is hazardous and should be attempted only when the mission is of the nature that such operation is necessary. Due caution must be exercised when landing or taxiing on ice. The aircraft is not equipped with nose wheel steering. Directional control can be maintained only with wheel brakes at taxi speeds and with brakes and rudder at speeds above rudder effectiveness. Touchdown should be made from a power approach at the minimum safe speed possible. Hold the nose wheel "off" as long as possible to obtain maximum aerodynamic drag. Braking after lowering the nose wheel must be made with caution to prevent sudden yawing and skidding. On ice it is very difficult to apply brakes without skidding the tires, due to the sensitive hydraulic actuated power brakes. It is also very difficult for the pilot to sense that the wheels are skidding. Landings on ice-covered runways should not be attempted unless the wind is within 10 degrees of runway heading. Using the drag chute when cross winds exceed 10 knots may cause weathervaning. Runway lengths required to bring the aircraft to a stop will be excessive and landings should not be attempted on runways of less than 10,000 feet unless a drag chute is available to aid deceleration after landing. The aircraft should be equipped with ice tires prior to operation on ice.

ENGINE ICING.

Axial-flow jet engines are seriously affected by icing. Ice forms on fixed inlet screens and compressor inlet guide vanes and restricts the flow of inlet air. This is indicated by a loss of thrust and a rapid rise in exhaust gas temperature. As the air flow decreases, the fuel-air ratio increases, which in turn raises the temperature of the gases going into the turbine. The fuel control attempts to correct any loss in engine RPM by adding more fuel, which aggravates the condition. Complete turbine failure from extreme overtemperature may occur in a matter of seconds after ice builds up in the engine inlet. Critical ice buildup on inlet screens can occur in less than one minute under severe conditions. With the inlet screens retracted, serious blocking of the air passages between the inlet guide vanes can still occur in four minutes or less. The idea that heating due to ram pressure at high speed will prevent icing is dangerous. Not enough heat is generated at subsonic velocities to prevent the formation of ice. The rate of engine icing for a given atmospheric icing intensity, with outside air temperature below freezing, is relatively constant up to an airspeed of approximately 250 KTAS. The rate of icing increases with increasing airspeed above 250 KTAS. Therefore, a reduction of airspeed to a safe minimum will reduce the rate of inlet icing. Serious inlet duct icing can occur without the formation of ice on the external aircraft surfaces. When jet aircraft fly at velocities below approximately 250 KTAS, and at high power settings, as in a climb, the intake air is sucked, instead of rammed into the engine compressor inlet. This suction causes a decrease of air temperature. Under these conditions, air at an ambient temperature above freezing may be reduced to subfreezing temperature as it enters the engine. Free moisture in the air may become super cooled and could cause engine icing, while no external surface icing would be evident. The maximum temperature drop which can occur is approximately 5°C (9°F). The greatest temperature drop occurs at high RPM on the ground and decreases with decreasing engine RPM and increasing airspeed.

CAUTION

Jet engine icing can occur without wing icing when the temperature is between minus 10°C (14°F) and 5°C (41°F) if fog is present or the dewpoint is within 4° (7°F) of the ambient temperature. When the above conditions exist, the engine screens should be retracted prior to takeoff.

An accumulation of ice on the leading edge of number two strut of the front main bearing support can block the fuel control aneroid sensing holes. This results in the fuel control losing its capability to compensate for ram air and causes a drop in engine RPM. Static altitude compensation will be maintained through the drain line and the engine RPM drop will be no lower than the particular altitude idle setting. The RPM will usually stabilize between 75-80 percent however the engine will not flame-out. Normal engine RPM and thrust can be maintained by properly switching to emergency fuel system. After leaving the icing area return to the normal fuel system.

A different hazard occurs if icing is encountered with the engine screens retracted. If any ice builds up on the nose of the aircraft it may erode or melt off, and blow back into the intake duct and compressor section. Damage to the engine will occur which will be similar to foreign object damage. This may or may not result in complete engine failure. The following procedure is recommended:

1. If the icing conditions cannot be avoided, check that engine screens are retracted prior to entering the icing area.

2. Attempt to leave the icing area as soon as possible.

3. Monitor EGT for indication of icing.

WARNING

If the throttle is not immediately retarded to maintain normal tailpipe temperatures, engine failure may result due to overheating of the turbine and exhaust system. This may occur very rapidy. Do not advance the throttle in an effort to maintain thrust, as this will aggravate the overheating condition and accelerate engine failure.

CAUTION

The engine does not have anti-icing provisions and therefore operation in icing conditions should be avoided whenever possible since axial flow turbo-jet engine operation is seriously affected by ice buildup. The rate of ice formation is often very rapid, resulting in a decrease of engine airflow accompanied by an excessive exhaust gas temperature. Icing of the number two strut of the front main bearing support can result in blocking of the fuel control aneroid sensing holes and cause a reduction of engine RPM. Normal RPM and engine thrust can be maintained by properly switching to the emergency fuel system.

4. Extend engine screens after icing has terminated. This will minimize engine damage caused by large chunks of ice being ingested into the engine.

TURBULENCE AND THUNDERSTORMS.

CAUTION

If at all possible, avoid flight through a thunderstorm to prevent damage from hail or icing conditions which are commonly encountered in thunderstorms.

The following factors, singly or in combination, have caused engine flame-outs.

a. Penetration of cumulus buildups with associated high liquid content.

b. Engine icing of inlet guide vanes.

c. Turbulence associated with penetration can result in angles of attack of plus nine degrees or more causing marginal engine performance.

d. Above 40,000 feet the surge margin of the engine is reduced and there is poor air distribution across the face of the compressor.

CAUTION

Flying in turbulence, or hail, may increase inlet distortion. At higher altitudes, this distortion can result in engine surge and possible flame-out. However, normal air starts may be accomplished.

Approaching the Storm.

It is imperative that you prepare the aircraft prior to entering a zone of turbulent air. If the storm cannot be seen, its proximity can be anticipated by heavy static on the radio compass. Prepare the aircraft as follows:

1. A safe comfortable pentration speed for the aircraft in severe turbulence is 275 KIAS.

CAUTION

Do not lower gear and flaps as they merely decrease the aerodynamic efficiency of the aircraft.

2. Pitot heater – ON.
3. Engine screen – Check RETRACT.
4. Safety belt – Tight (This is important).
5. Shoulder harness – Locked.
6. Turn radio volume down during severe static conditions.
7. At night, turn cockpit lights full bright to minimize blinding effect of lightning. In addition, turn on thunderstorm lights if aircraft is equipped with them.

CAUTION

After leaving the storm, extend engine screens to prevent ice formed on the leading edge of the intake ducts from going through the engine.

Note

Make every effort to avoid looking up from the instrument panel at lightning flashes. The blinding effect of lightning can be reduced by lowering the seat.

NIGHT FLYING.

Before takeoff, make certain that all lights function properly. The various rheostat controls afford selection of red lighting contract of the instruments, auxiliary panels and consoles. All cockpit lights should be adjusted to minimum intensity for normal operation to reduce canopy reflections and to permit rapid changeover to instrument flight.

When the position light switch is in the FLASH position, the flashing lights cause distracting reflections on the surrounding clouds. The pilot should be aware that a failure of both generators will cause a failure of the position lights; in addition will cause the forward and wing booster pumps to become inoperative, and the respective warning lights will illuminate.

If flight through thunderstorm is anticipated, adjust cockpit lights to brightest intensity to prevent momentary blindness from lightning flashes.

The landing lights are aimed for a normal nose-high landing and do not provide adequate runway illumination when a steep power off approach is made. The recommended approach should be made with power and the runway lights should be used as a primary reference for the final approach. The landing lights provide adequate runway illumination once round-out is accomplished.

COLD WEATHER PROCEDURES.

PREFLIGHT.

When the ambient temperature is 0°C (32°F) or lower, use a portable heater to blow hot air into the nose wheel well and starter bottle area. This heating will prevent possible malfunctioning of the starter system. Heat should also be directed toward the engine gear box. Place heater duct in the aircraft air inlet duct for a period of 10-15 minutes. This procedure is necessary to prevent the starter unit from being damaged due to ice seizure of the compressor rotator. Depress the engine rotor test switch for approximately one-half second and listen for audible sound of rotor freedom or observe indication on tachometer. To heat the cockpit, loosen the canopy cover and open the canopy enough so that the heater hose can be inserted into the cockpit. This heating will restore flexibility to the rubber cockpit seal and also will prevent cockpit switch malfunction due to moisture condensation. The drag chute and drag chute compartment should be inspected for moisture. Under certain atmospheric conditions the drag chute compartment is susceptible to condensation of moisture. Freezing of a damp drag chute can result in failure to deploy. Remove snow and ice from wings, fuselage, tail and landing gear mechanisms.

WARNING

Depending on the weight of snow and ice accumulated, takeoff distances and climb-out performance can be seriously affected. The roughness and distribution of the ice and snow could vary stall speeds and characteristics to an extremely dangerous degree. Loss of an engine shortly after takeoff is a serious enough problem without the added, and avoidable, hazard of snow and ice on the wings. In view of the unpredictable and unsafe effects of such a practice, the ice and snow must be removed before flight is attempted.

When conditions are such that mud, snow or slush will freeze, it is recommended that the mud guard be removed from the nose wheel. Use external power for operating and ground check all electrical and radio equipment. Hold battery use to a minimum prior to engine starting. Battery life is reduced to as little as one fifth rated power during extreme cold.

PRESTARTING.

Before the start, one should always manipulate the emergency fuel switch to insure proper operation; otherwise the valves may hang and cause unexpected switching later on. Start engine in normal manner using external power. If there is no oil pressure after 60 seconds running, or if pressure drops after a few minutes ground operation, shut down and check for blown lines or for congealed oil. After start, delay initial movement of controls for a few moments. This is done to permit as much hydraulic fluid as possible to circulate through the pumps. Never turn on electrical equipment except that absolutely needed, until generator shows positive reading. While still in parking area, check out all hydraulic, electrical communication, refueling, lights, defrosting and air conditioning systems and double check that engine screens are extended. Make a final recheck on wings and tail for snow or ice. During taxi-out, check area fore and aft for aircraft. Higher RPM necessary for initial movement and poor braking action on snow require more maneuvering space. Care should be exercised when using full, or near full engine power when aircraft is being run up on chocks as slippage of chocks occurs frequently. Operate wing flaps through several cycles. Turn pitot heater ON if icing conditions are anticipated. Turn heating (pressurizing system), and defroster system ON. Some engines will not accelerate properly when they are cold and will therefore require a short warm-up period in order that the acceleration limits are met. During the runup have aircraft thoroughly checked for leaks. This check should be performed at maximum practicable RPM.

Note

The engine is susceptible to engine hang-up whenever the temperature has dropped five degrees or more at −29°C (−20°F) and usually takes place in the 75 to 80 percent RPM range. Whenever this occurs, the Bendix fuel control may require an adjustment of the temperature compensating unit.

TAKE-OFF.

WARNING

Prior to takeoff make sure all instruments have warmed up sufficiently to insure normal operation. Check for sluggish instruments during taxiing.

Avoid taxiing through loose snow as it may get into brakes and freeze. Pack or remove loose snow from runway prior to takeoff. Spacing between elements or aircraft on takeoff should not be less than 30 seconds when operating on a snow-covered runway, for aircraft on the takeoff roll will leave a hanging wall of powdered snow which requires a few seconds to settle to the runway.

AFTER TAKE-OFF.

After takeoff from a snow or slush covered field, operate landing gear and flaps through several complete cycles to preclude their freezing in the UP position. Leave the gear up only after sufficient time has elapsed to remove all slush or moisture. During this time, hold IAS well below maximum permissible gear extension speed. Gear, flaps and all other hydraulic systems will take longer to complete their cycle in cold weather and all limit speeds must be observed.

Note

Landing gear retracting time is from nine seconds at −29°C (−20°F) to 12 seconds at −54°C (−65°F).

LANDING.

Note

In night landings, on snow or ice it is recommended that landing lights not be used for landing due to the intense "bounce" of light the pilot receives from the reflection of the lights on the snow and ice.

Note

Landing gear extension time is from eight seconds at −29°C (−20°F) to 8.5 seconds at −54°C (−65°F).

POST FLIGHT.

If lay-over of several days is expected, remove the battery. Further at temperatures below −29°C (−20°F) remove the battery if lay-over exceeds four hours. Leave canopy slightly open to prevent cracking of transparent areas due to differential contraction. Also air circulation retards frost formation in cockpit. Install wing, empennage and canopy covers and install dust plugs in air intake duct and in tailpipe.

SUMP DRAINAGE.

The fuel tank sump should be drained frequently of condensate. Under prolonged freezing conditions a small amount of ice or snow gets into the fuel tanks each time the aircraft is serviced. When there is sufficient rise in temperature due to placing the aircraft in a hangar or to warmer weather, these crystals melt, resulting in water in the system. Regular and frequent drainage especially under thawing conditions, is the best method of preventing ice in the fuel lines when the aircraft is again subjected to freezing weather. Keeping the tanks as full as possible when the aircraft is parked will also help to reduce moisture condensation.

HOT WEATHER AND DESERT PROCEDURES.

BEFORE ENTERING THE AIRCRAFT.

All metal surfaces exposed to the sun are burning hot to touch. Wear gloves to prevent burns. Make all possible ground checks before starting the engine. If operating in sandy country, ascertain that air filters, instrument filters, and oil filters have been cleaned for each flight. Check seals and tires to ascertain that they are not blistered or show other evidence of deterioration.

STARTING THE ENGINE.

Head the aircraft into the wind if practicable. Start in the normal manner. Do not run up engines to windward of other planes, personnel or ground installations.

Note

When the ambient air temperature exceeds 32°C (90°F) maintain idle RPM exhaust temperature within limits by manually advancing the throttle to a higher RPM but not to exceed 50 percent RPM.

TAKE-OFF.

If ground is sandy or dusty, avoid taking off in the wake of another aircraft. Place the cabin vent switch in the PRESSURE position unless high humidity causes cockpit to fill up with fog. If so take-off with cabin vent switch in RAM.

Note

Takeoff distances will be longer because the air is less dense during warm weather.

AFTER TAKE-OFF.

Do not climb the aircraft at less than flying speed specified in the climb chart.

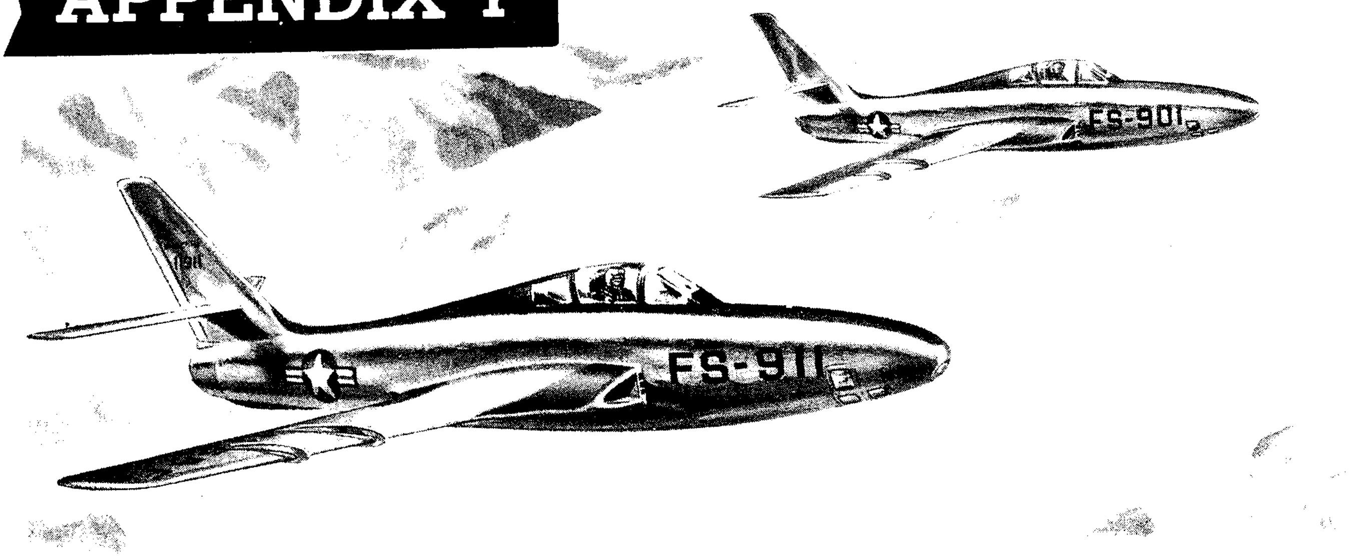
LANDING.

Because hot air is less dense than cold air, true stalling speed will be greater and additional distance will be required for landing.

PARKING.

If blowing sand is a hazard, close and cover all openings to keep sand out. Cover windshield and canopy to prevent sand scratches. Lay cover on canopy, do not slide. Keep canvas covers on the windshield and canopy whenever the aircraft is parked in the sun. If this is not done, the sun's heat will soften and distort the transparent plastic. Malfunctioning of instruments and communications equipment will also result. If blowing sand is not a hazard, keep canopy and selected access doors open to permit air circulation.

APPENDIX I



Performance Data

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Note

Aircraft equipped with J65-7 engines may demonstrate slightly better performance than those predicted for the J65-3 engine.

PART 1 INTRODUCTION

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INTRODUCTION.

The flight performance charts in this section provide the pilot with flight test data for flight planning purposes. Two types of charts are included: (1) profile type charts and (2) graphical charts. These charts are described in the following pages. The profile type charts are a supplement to the graphical data and facilitate flight planning by reducing the computations that must be made. These charts are based on the recommended climb and cruise settings shown on the profile for the particular load configuration of the aircraft. This type of presentation gives a direct indication of the fuel and time required to cover a given distance if the recommended settings are maintained. A decrease in weight has been accounted for as fuel is consumed. For cruise at Mach numbers other than those given on the profile charts, the graphical charts should be used for flight planning. For flight planning where accurate results are mandatory, the graphical data should be used. All charts are based on NACA Standard Day conditions. The take-off and landing distance charts contain the temperature correction in graphical form.

AIRSPEED CORRECTION.

Airspeed installation error is minor and may be considered negligible with the aircraft in any configuration. Therefore, calibrated airspeed (CAS) is considered equal to indicated airspeed (IAS). A Compressibility Correction Table (figure A1—1) is provided for computing true indicated airspeed (TIAS) from calibrated airspeed (CAS). Dividing the true indicated airspeed by the square root of relative density, ratio of ambient to standard sea level density obtained from figure A1—2, (Standard Altitude Table) provides true airspeed (TAS). Vectorially adding wind velocity to true airspeed provides ground speed.

SAMPLE PROBLEM.

For purposes of explaining the use of the Compressibility Correction Table, and the Standard Altitude Table, consider the aircraft flying at 25,000 feet and an airspeed indicator reading of 350 knots per hour. Since the aircraft is not equipped with an outside air temperature indicator, determine the ambient temperature at 25,000 feet from the Standard Altitude Table which will be —35°C.

Airspeed Indicator Reading	350 knots
Correction for Instrument Error (from instrument calibration card)	−2
Indicated Airspeed (IAS)	348 knots
Calibrated Airspeed (CAS)	348 knots
Correction for Compressibility Error (from Compressibility Correction Table)	−16 knots
Equivalent Airspeed (EAS)	332 knots
Correction for Air Density (from Standard Altitude Table)	× 1.49
True Airspeed (TAS)	495 knots

SPEED CONVERSION CHART.

The speed conversion chart (figure A1−3) reflects the Mach number and the true airspeed at any altitude for a given calibrated airspeed.

AVERAGE WEIGHTS OF AIRCRAFT AND STORES

REFER TO T.O. 1F-84(R)F-5 FOR WEIGHT AND BALANCE

Item	*Weight*
CLEAN AIRCRAFT	18870 LB
plus two 450 GAL. Type I Tanks With Fins (Full)	25505 LB
plus two 450 GAL. Type II Tanks (Full)	25414 LB
plus two 230 GAL. Type I Tanks With Fins (Full)	22406 LB
plus two 230 GAL. Type II Tanks (Full)	22313 LB
plus two 230 GAL. Type IV Tanks (Full)	22250 LB
plus two 450 GAL. and two 230 GAL. Type I Tanks With Fins (Full)	29041 LB
plus four 230 GAL. Type I Tanks With Fins (Full)	25797 LB
plus one 450 GAL. Type I Tank With Fins (Full)	22188 LB
plus one 450 GAL. Type II Tank (Full)	22142 LB
plus one 230 GAL. Type I Tank With Fins (Full)	20658 LB
plus one 230 GAL. Type II Tank (Full)	20614 LB

NOTE: Clean Weight of Aircraft Includes Pilot and Equipment, Fully Serviced Internal Fuel and Oil Tanks, Cameras, Full Complement of Ammunition and Four Guns.

AVERAGE WEIGHTS OF PYLONS AND EMPTY PYLON TANKS

Item	*Weight*
450 GAL. Type I Tank With Fins	241 LB
450 GAL. Type II Tank	195 LB
230 GAL. Type I Tank With Fins	173 LB
230 GAL. Type II Tank (Sutton)	127 LB
230 GAL. Type IV Tank (Royal Jet)	95 LB
Pylon S-2A	144 LB
Pylon S-3	152 LB
Outboard Pylon	80 LB
Special Store Pylon	167 LB

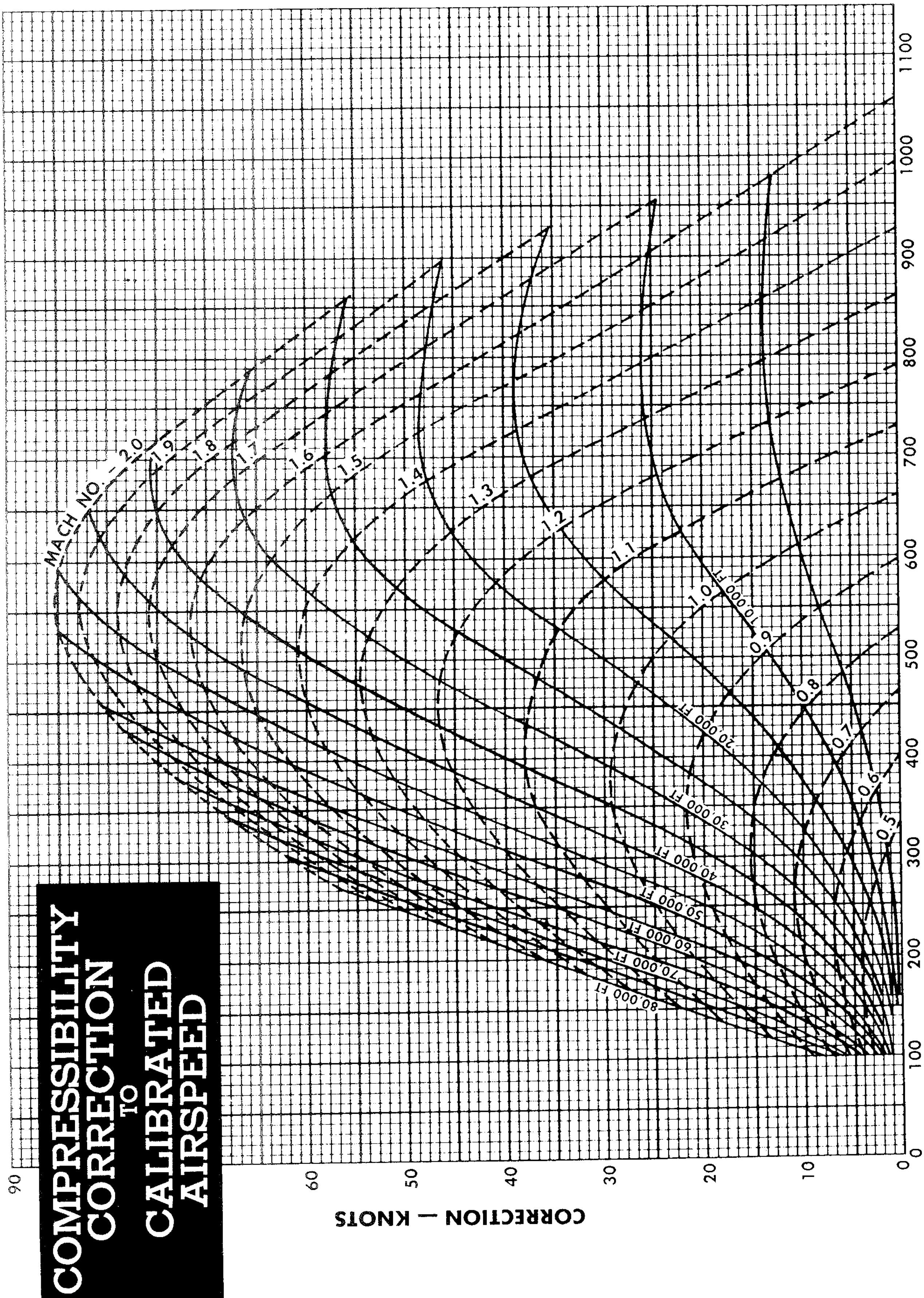

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED
CALIBRATED AIRSPEED — KNOTS
CORRECTION — KNOTS
MACH NO. — 2.0
1.9
1.8
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
10,000 FT
20,000 FT
30,000 FT
40,000 FT
50,000 FT
60,000 FT
70,000 FT
80,000 FT
0
100
200
300
400
500
600
700
800
900
1000
1100
10
20
30
40
50
60
90

Figure A1—1

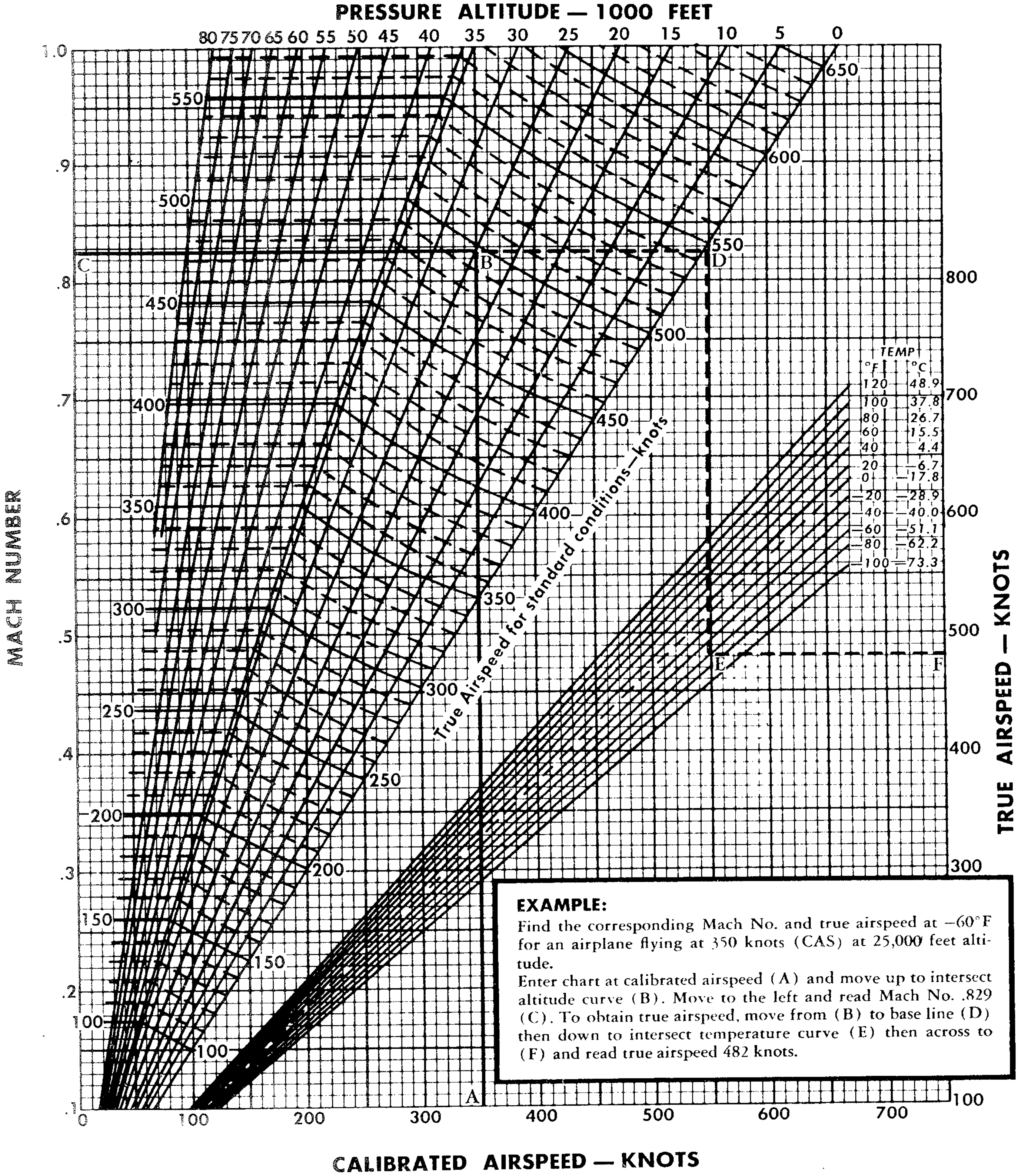
ICAO STANDARD ALTITUDE TABLE

Standard Sea Level Air:
T = 15° C. W = .07651 lb/cu. ft. ρ_o = .002378 slugs/cu. ft.
P = 29.921 in. of Hg. 1" of Hg. = 70.732 lb/sq. ft. = 0.4912 lb/sq. in.
This table is based on NACA Technical Report No. 218 a_o - 1116 ft./sec.

Altitude feet	Density Ratio ρ/ρ_o	$\frac{1}{\sqrt{\sigma}}$	Temperature Deg. C	Temperature Deg. F	Speed of Sound Ratio a/a_o	Pressure In. of Hg.	Pressure Ratio P/Po
0	1.0000	1.0000	15.000	59.000	1.0000	29.92	1.0000
1000	.9710	1.0148	13.019	55.434	.997	28.86	.9644
2000	.9428	1.0299	11.038	51.868	.993	27.82	.9298
3000	.9151	1.0454	9.056	48.301	.990	26.81	.8962
4000	.8881	1.0611	7.075	44.735	.986	25.84	.8636
5000	.8616	1.0773	5.094	41.169	.983	24.89	.8320
6000	.8358	1.0938	3.113	37.603	.979	23.98	.8013
7000	.8106	1.1107	1.132	34.037	.976	23.09	.7716
8000	.7859	1.1280	-0.850	30.471	.972	22.22	.7427
9000	.7619	1.1456	-2.831	26.904	.968	21.38	.7147
10000	.7384	1.1637	-4.812	23.338	.965	20.58	.6876
11000	.7154	1.1822	-6.793	19.772	.962	19.79	.6614
12000	.6931	1.2012	-8.774	16.206	.958	19.03	.6359
13000	.6712	1.2206	-10.756	12.640	.954	18.29	.6112
14000	.6499	1.2404	-12.737	9.074	.950	17.57	.5873
15000	.6291	1.2608	-14.718	5.507	.947	16.88	.5642
16000	.6088	1.2816	-16.699	1.941	.943	16.21	.5418
17000	.5891	1.3029	-18.680	-1.625	.940	15.56	.5202
18000	.5698	1.3247	-20.662	-5.191	.936	14.94	.4992
19000	.5509	1.3473	-22.643	-8.757	.932	14.33	.4790
20000	.5327	1.3701	-24.624	-12.323	.929	13.75	.4594
21000	.5148	1.3937	-26.605	-15.890	.925	13.18	.4405
22000	.4974	1.4179	-28.586	-19.456	.922	12.63	.4222
23000	.4805	1.4426	-30.568	-23.022	.917	12.10	.4045
24000	.4640	1.4681	-32.549	-26.588	.914	11.59	.3874
25000	.4480	1.4940	-34.530	-30.154	.910	11.10	.3709
26000	.4323	1.5209	-36.511	-33.720	.906	10.62	.3550
27000	.4171	1.5484	-38.493	-37.287	.903	10.16	.3397
28000	.4023	1.5768	-40.474	-40.853	.899	9.720	.3248
29000	.3879	1.6056	-42.455	-44.419	.895	9.293	.3106
30000	.3740	1.6352	-44.436	-47.985	.891	8.880	.2968
31000	.3603	1.6659	-46.417	-51.551	.887	8.483	.2834
32000	.3472	1.6971	-48.399	-55.117	.883	8.101	.2707
33000	.3343	1.7295	-50.379	-58.684	.879	7.732	.2583
34000	.3218	1.7628	-52.361	-62.250	.875	7.377	.2465
35000	.3098	1.7966	-54.342	-65.816	.871	7.036	.2352
36000	.2962	1.8374	-55.000	-67.000	.870	6.708	.2242
37000	.2824	1.8818	-55.000	-67.000	.870	6.395	.2137
38000	.2692	1.9273	-55.000	-67.000	.870	6.096	.2037
39000	.2566	1.9738	-55.000	-67.000	.870	5.812	.1943
40000	.2447	2.0215	-55.000	-67.000	.870	5.541	.1852
41000	.2332	2.0707	-55.000	-67.000	.870	5.283	.1765
42000	.2224	2.1207	-55.000	-67.000	.870	5.036	.1683
43000	.2120	2.1719	-55.000	-67.000	.870	4.802	.1605
44000	.2021	2.2244	-55.000	-67.000	.870	4.578	.1530
45000	.1926	2.2785	-55.000	-67.000	.870	4.364	.1458
46000	.1837	2.3332	-55.000	-67.000	.870	4.160	.1391
47000	.1751	2.3893	-55.000	-67.000	.870	3.966	.1325
48000	.1669	2.4478	-55.000	-67.000	.870	3.781	.1264
49000	.1591	2.5071	-55.000	-67.000	.870	3.604	.1205
50000	.1517	2.5675	-55.000	-67.000	.870	3.436	.1149

Figure **A1-2**

SPEED CONVERSION CHART



EXAMPLE:

Find the corresponding Mach No. and true airspeed at −60°F for an airplane flying at 350 knots (CAS) at 25,000 feet altitude.

Enter chart at calibrated airspeed (A) and move up to intersect altitude curve (B). Move to the left and read Mach No. .829 (C). To obtain true airspeed, move from (B) to base line (D) then down to intersect temperature curve (E) then across to (F) and read true airspeed 482 knots.

Figure A1—3

TEMPERATURE CONVERSION CHART

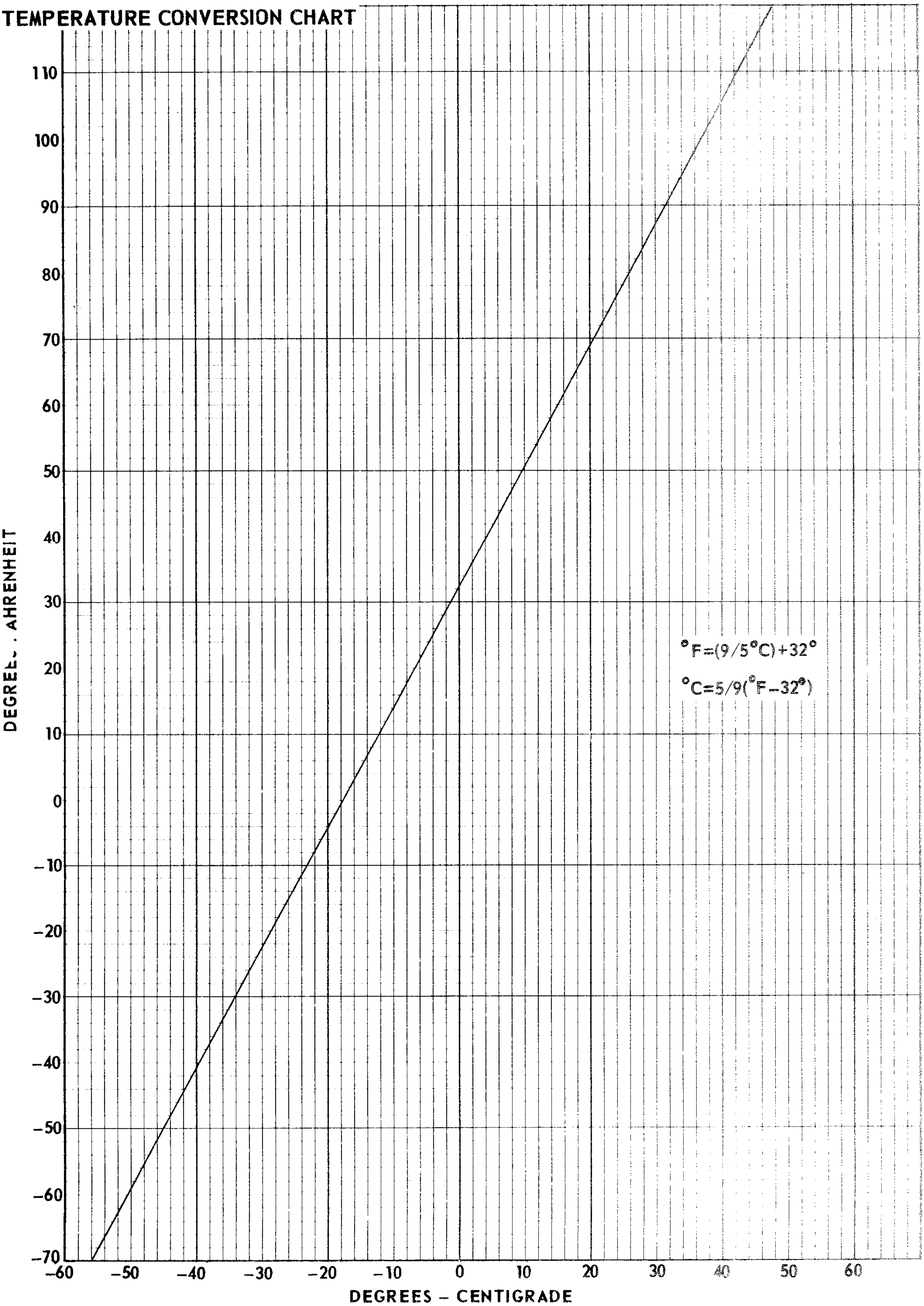


Figure A1-5.

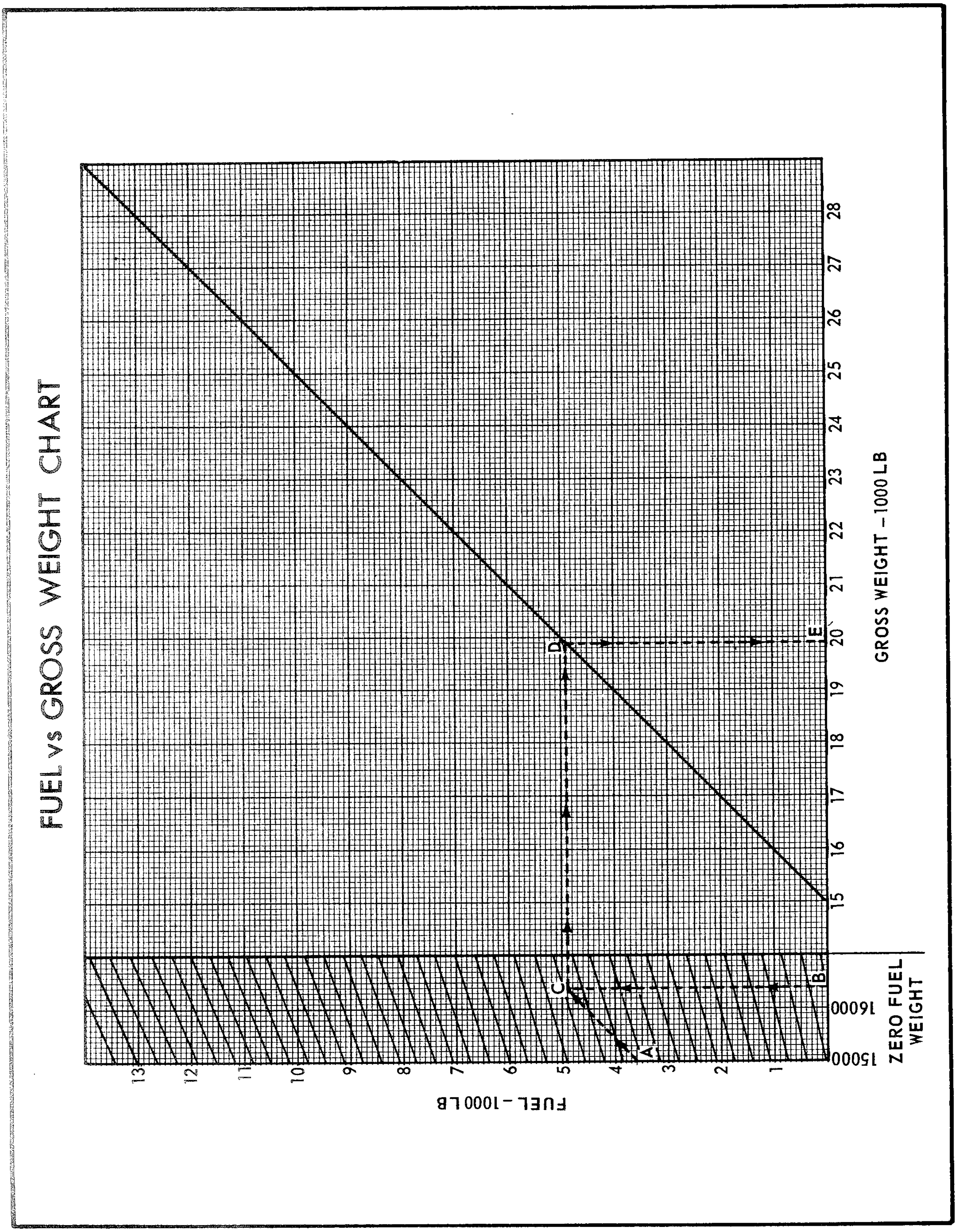

FUEL vs GROSS WEIGHT CHART
GROSS WEIGHT - 1000 LB
15
16
17
18
19
20
21
22
23
24
25
26
27
28
FUEL - 1000 LB
1
2
3
4
5
6
7
8
9
10
11
12
13
ZERO FUEL WEIGHT
15000
16000
A
B
C
D
E

Figure A1-5

PART 2 TAKEOFF

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TAKE-OFF PLANNING.

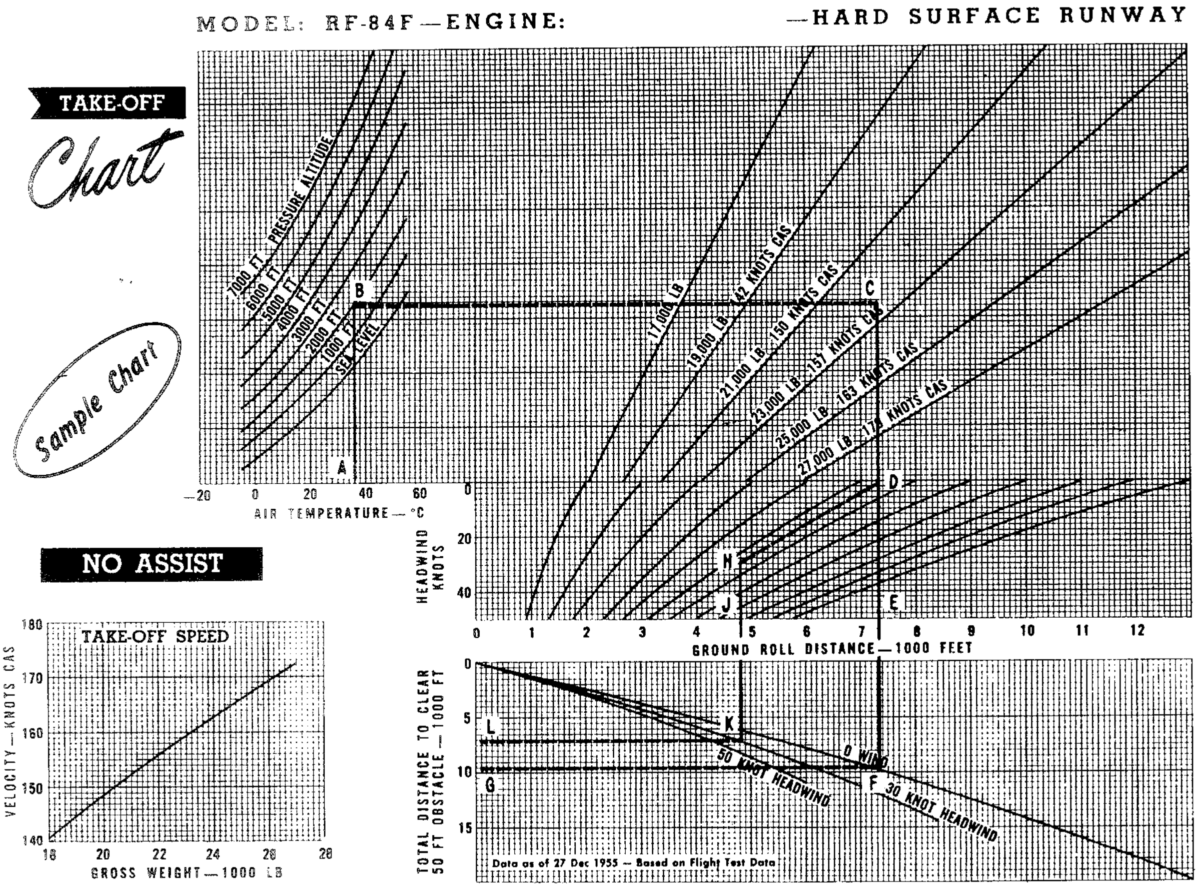
This part covers the information and charts to be used to obtain take-off speeds and distances. The terms used in the planning procedure are defined in the following paragraphs. Take-off performance is affected by a large number of variables, i.e., temperature, altitude, gross weight, and wind, as well as runway surface, use of brakes for directional control, and engine condition. Charts including these variables are provided for take-off distance, acceleration distance and speed, and stopping distance or refusal speed. Increases in any of these variables except wind tend to increase take-off ground roll to a point where, on a take-off in which normal techniques are used, the take-off may not be successfully made in the available runway length. The take-off chart shows distances for ground roll as well as total distance required to clear a 50-foot obstacle, take-off speeds, and rotation speeds. The velocity during take-off ground run chart shows the speed-distance relationship during the ground roll portion of take-off before rotation speed is reached. The refusal speed chart shows the combined distance traveled in acceleration to any given refusal speed and the distance required for a full stop.

RUNWAY DISTANCE MARKING SYSTEM.

The numbering and placement of runway distance markers reflects the distance remaining to the end of the runway in 1000-foot increments. These markers are placed alongside the runway, and the appropriate markers become the acceleration check distance marker and the go, no-go distance marker. In accordance with the marker system on a runway length in excess of the 1000-foot interval (10,500), one half the length over the exact thousands of feet [½ (10,500 − 10,000)] must be added to the distances shown on the markers to determine the actual distance remaining; i.e., at marker No. 6, the distance remaining would be 6250 feet.

TAKE-OFF DISTANCES.

Ground-run distances and total distances to clear a 50-foot obstacle with Military Thrust are plotted in the take-off distance charts. The distances shown are for normal take-off technique on a dry, hard-surface runway and may be used for any configuration if the gross weight at take-off is considered. The calibrated airspeeds (CAS) for take-off at various gross weights are also shown. Use of the charts is explained by a "chase-through" sample problem.



TAKE-OFF CHARTS.

Take-off charts are provided for no assist, two ato and four ato assist configurations. The charts take into account ambient temperature, pressure altitude and gross weight. Take-off speed, ground roll and distance to clear a 50 foot obstacle with or without a headwind are obtained from the charts. Ato cut-in speeds are plotted on separate charts which are to be used in conjunction with the corresponding take-off chart.

EXAMPLE: NO ASSIST.

Determine take-off speed, ground roll and distance to clear 50 foot obstacle for the following configuration:

Aircraft gross weight	22,300 LB
Pressure altitude	2,000 FT
Air Temperature	35°C (95°F)
Headwind	0 to 30 KN
Ato Units	0

Procedure:

Enter figure at air temperature (A) and move up to pressure altitude curve (B). Continue horizontally across to aircraft gross weight (C) and vertically to ground roll distance (E). Drop to 0 wind curve (F) and horizontal to distance to clear 50 foot obstacle (G). For a 30 knot headwind, drop from aircraft gross weight (C) to (D) then follow curve to headwind (H) then drop to ground roll distance (J). Continue from (J) vertically to 30 knot headwind curve (K) and horizontal to distance to clear 50 foot obstacle (L).

Conclusion:

Take-off speed (C) 154 knots CAS (154 knots IAS)
Ground roll distance (E) 7250 feet (zero wind)
Distance to clear 50 foot obstacle (G) 9650 feet (zero wind)
Ground roll distance (J) 4700 feet (30 knot headwind)
Distance to clear 50 foot obstacle (L) 7000 feet (30 knot headwind)

ACCELERATION CHECK DISTANCE.

The decision to continue or to abort a take-off is made at the go, no-go point; however, a preliminary check is made before reaching the go, no-go point to monitor the progress of the take-off. This check is made at the acceleration check point and is defined as the distance to the runway marker, which is 2000 feet short of the go, no-go point. This distance is established by subtracting 2000 feet from the go, no-go distance. The acceleration check speed is the minimum speed allowable at this check distance and is established by obtaining the speed for the distance 2000 feet short of the go, no-go distance.

REFUSAL SPEEDS.

The highest indicated airspeed to which the aircraft can accelerate and then stop in the available runway length is the refusal speed. The refusal speed chart is based on a Military Thrust acceleration to the refusal speed, and then heavy braking to a stop on a dry, hard-surface runway, with or without use of drag chute. The drag chute reduces the distance required to stop and should be used if possible. The ground-roll distance required to accelerate to refusal speed can be found on the takeoff speed versus distance charts.

ROTATION SPEED (NOSE WHEEL-OFF).

Rotation speed (nose wheel-off) is the airspeed at which stick travel is started. The stick should be moved aft at such a rate that the aircraft will be in proper lift off attitude when the recommended lift off speed is reached. Rotation speeds are shown on the various take-off ground roll charts.

TAKEOFF SPEED.

Takeoff speed is the airspeed at which the main wheels leave the runway.

GO, NO-GO DISTANCE.

Go, no-go distance is defined as the distance traveled in reaching the last 1000-foot runway marker short of the refusal distance. This is the point at which the final decision to continue or to abort the take-off is made. The speed attained at the go, no-go distance is defined as the go, no-go speed.

GO, NO-GO SPEED.

Go, no-go speed is obtained by takeoff speed versus distance charts with the go, no-go distance.

GO, NO-GO SPEED TOLERANCE.

Speed tolerance is the maximum speed deficiency that is acceptable at the go, no-go distance to ensure a safe take-off in 90 percent of the available runway. If acceleration is deficient to the extent that the speed tolerance is exceeded at the go, no-go distance, a serious malfunction of the aircraft is indicated and the take-off should be aborted.

TAKE-OFF ACCELERATION CHECK DATA.

Take-off ground roll distance curves, and speed vs distance curves, are presented to allow the pilot to determine speed at any distance down the runway (line speed) during the take-off ground roll. Stopping distance curves are presented to determine the minimum distance required to stop from any speed. Using this information, the runway marker (painted stripes, signs, etc.) at which acceleration is checked can be chosen so that the aircraft may be stopped on the remaining runway from normal predicted speed at that point should the take-off be aborted.

The purpose of line speed is to provide a means of checking take-off acceleration. A check on the speed at some point reasonably early in the take-off run will allow the pilot to properly monitor his take-off run. It is possible to predict two line speeds for a given line or check point on the runway. (a) normal line speed which assumes normal take-off ground roll distance and (b) minimum line speed which assumes that the entire runway length will be used for the take-off ground roll due to low acceleration. To minimize the number of aborts and assure full utilization of available runway length, the minimum line speed should be used for the acceleration check. If the observed speed at the check point is equal to or more than the minimum line speed, take-off acceleration is acceptable and the take-off should be continued. If it is less than the minimum line speed the take-off should be aborted. Since the check point is chosen so that the aircraft can be stopped on the remaining runway from the normal line speed, it will be less difficult to stop from a speed less than minimum line speed.

EXAMPLE:

Determine the GO, NO-GO DISTANCE, SPEED AND SPEED TOLERANCE for the following conditions:

Runway Length	9,000 ft
Gross Weight	22,300 lbs
Runway Temperature	35°C (95°F)
Pressure Altitude	2,000 ft
Dry Hard Surface Runway	
Zero Wind	
Take-off Distance	7,250 ft
Take-off Speed	158 KCAS (from figure A2-1)
MAX Refusal Speed	129 KCAS (from figure A2-7)

Procedure.

Using take-off speed obtained from figure A2-1, enter figure A2-4 at 158 knots CAS (A). Enter chart with ground roll distance 7,250 ft (B). Proceed downward from (A) until guide line intersects ground roll distance at (C). Enter chart with max refusal speed 129 knots CAS (D). Follow the curve down and to the left until it intersects the refusal speed 129 CAS (E). Read right to refusal distance 4,500 ft (F). Therefore, the GO-NO-GO distance will be 4,000 feet.

Enter figure A2-4 with take-off speed (A). Proceed down until intersection of 90% of available runway line 8,100 feet (H). Follow the curve down and to the left to 4,000 foot intersection (J). Read down to GO-NO-GO speed (minimum) 117 knots CAS (K).

To correct GO-NO-GO speeds for prevailing wind conditions add Δ air speed for headwind to GO-NO-GO speed subtract Δ airspeed for tailwind from GO-NO-GO speed.

Example:

Correct the above GO-NO-GO speed (minimum) of 117 knots CAS for a 20 knot headwind condition.

Enter correction chart on figure A2-4 with headwind component of 20 knots (L). Proceed up to intersect diagonal line (M). Read right for GO-NO-GO airspeed correction (N) Δ airspeed = 7 knots.

Corrected GO-NO-GO speed = 117 + 7 = 124 knots CAS.

TAKE-OFF AND LANDING CROSSWIND CHART.

The take-off and landing crosswind chart figure A2–8 determines if take-off and landings, at predicted speed for gross weight, are recommended for the direction and velocity of the existing crosswind.

TAKE-OFF DATA CARD.

Following is a sample of a take-off data card. For the data card form refer to Section II Abbreviated Check List. Information on this card should be filled out prior to each flight. This data is obtained by referring to the Take-off Chart and the Landing Distance Chart for the aircraft configuration.

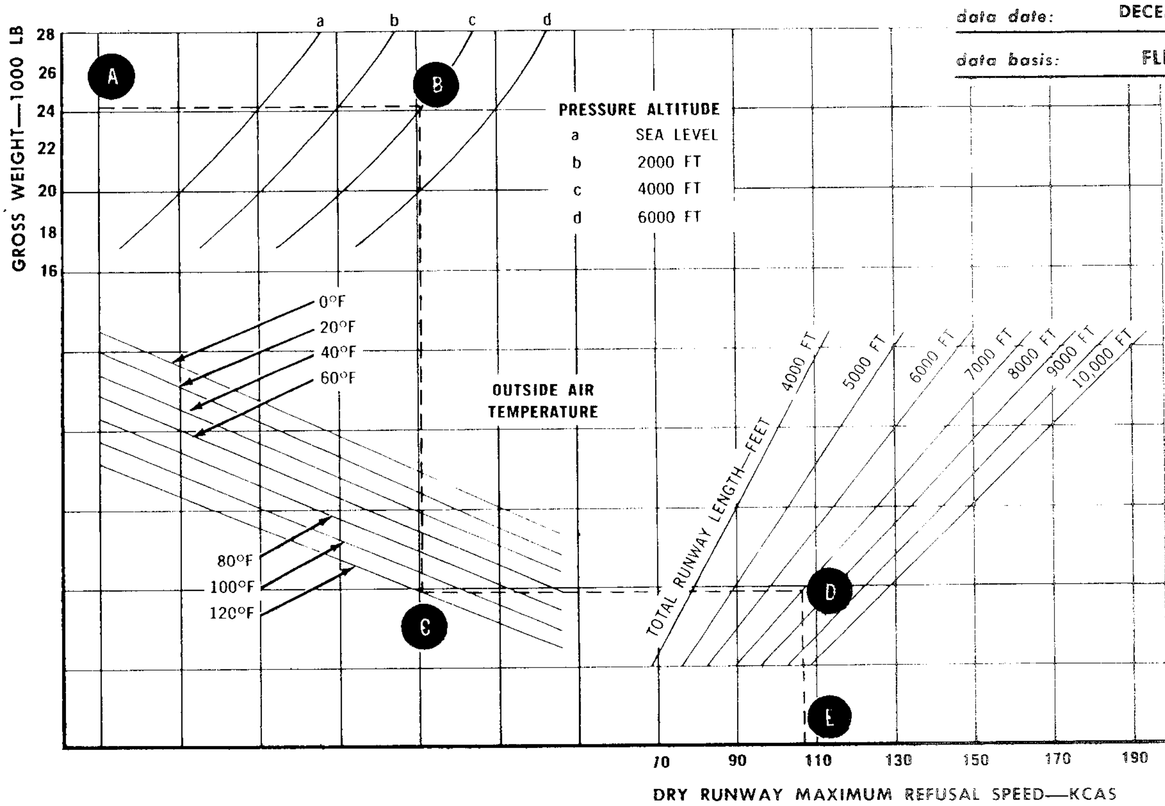
TAKE-OFF DATA CARD

Conditions	
Gross Weight	22,300 LB
Runway Length	9,000 FT
OAT	35°C (95°F)
Pressure Altitude	2,000 FT
Runway Gradient	0%
Wind	0 KN
Take-off	
Take-off Distance	7,250 FT
Take-off Speed	154 KN
Go-No-Go Distance	4,000 FT
Go-No-Go Speed (Minimum)	109 KN
Landing Immediately After Take-off	
Approach Speed	195 KN
Landing Ground Roll (without drag chute)	5,400 FT
Landing Ground Roll (with drag chute)	3,000 FT

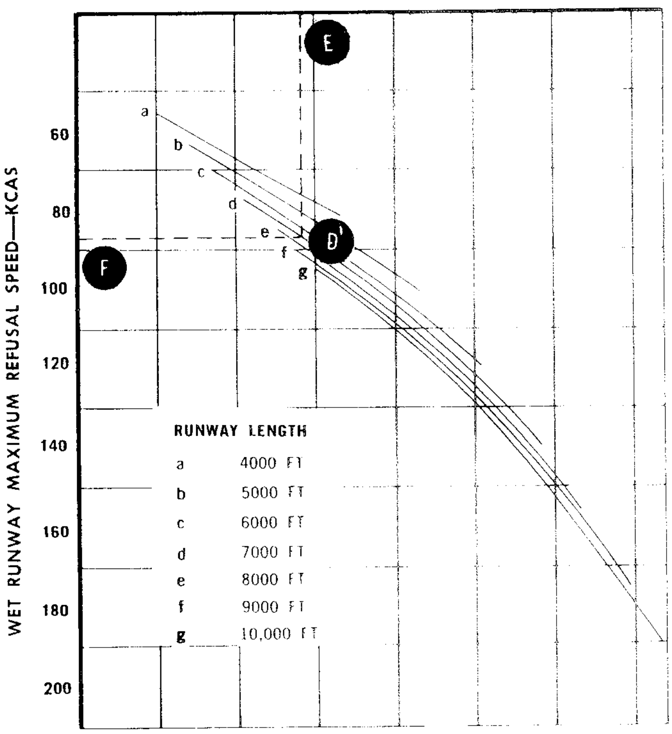
MAXIMUM REFUSAL SPEED

Military Thrust
With Drag Chute

model:	(R)F-84F
engine:	J65-7
fuel grade:	JP-4
fuel density:	6.5 LB/GAL
data date:	DECEMBER 1962
data basis:	FLIGHT TEST



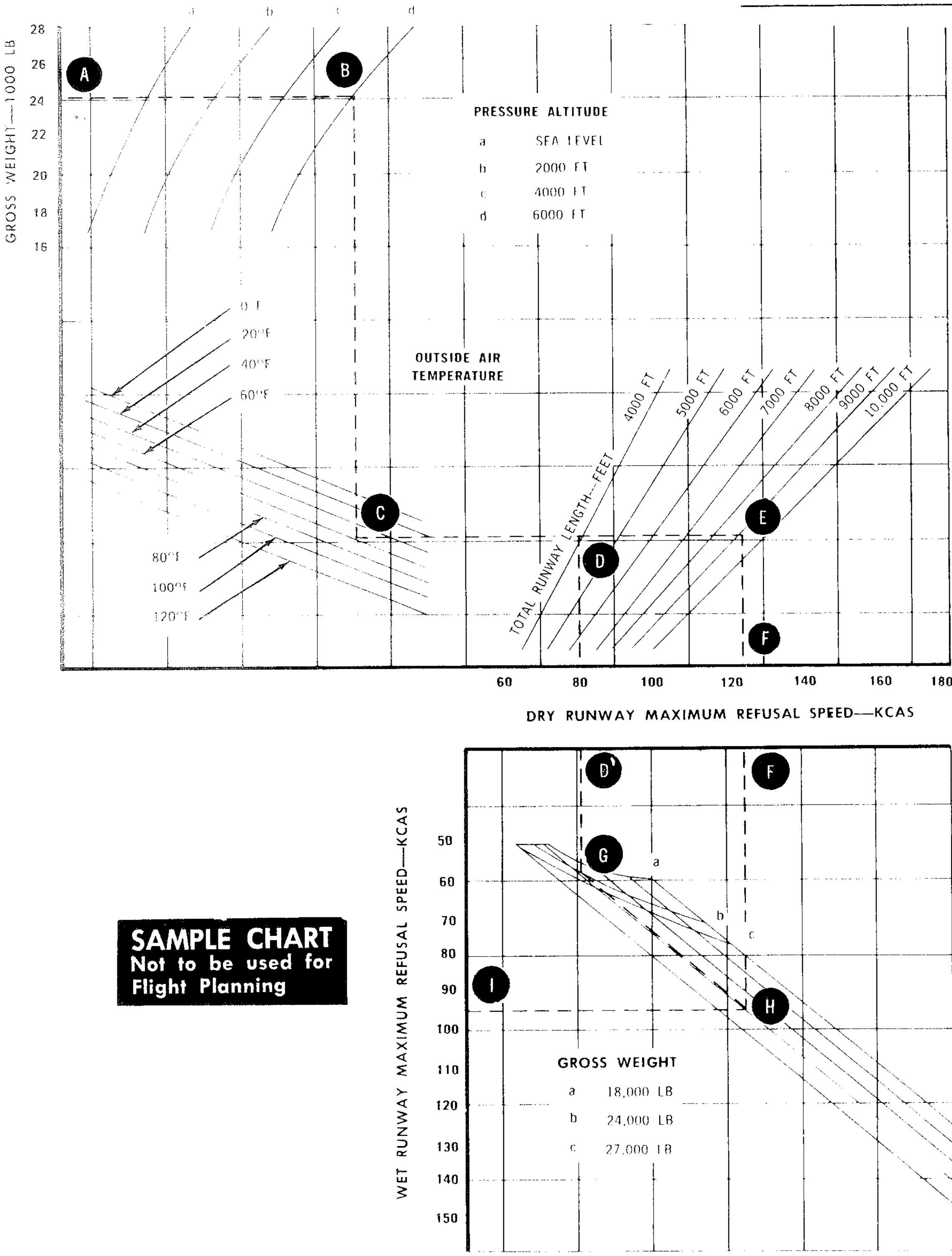
SAMPLE CHART
Not to be used for Flight Planning



MAXIMUM REFUSAL SPEED

Military Thrust
No Drag Chute

model:	**F-84F**
engine:	**J65-7**
fuel grade:	**JP-4**
fuel density:	**6.5 LB/GAL**
data date:	DECEMBER 1962
data basis:	**FLIGHT TEST**



SAMPLE CHART
Not to be used for Flight Planning

MAXIMUM REFUSAL SPEEDS.

These charts show the highest speed to which an aircraft can be accelerated and still be stopped on the remaining runway. They take into account gross weight, pressure altitude, temperature and actual runway lengths. The refusal speeds shown are presented for level, hard wet or dry runways, with or without drag chute deployed. These speeds were determined for the clean and externally loaded configurations.

USE

Example: 1 Drag Chute Deployed.

(a) Zero Wind

Determine the refusal speed for an aircraft when the runway length available for takeoff is 7000 feet. Prevailing conditions are as follows: gross weight 24,000 pounds; pressure altitude 4000 feet; temperature 120°F; dry runway and wet runway; drag chute deployed; zero wind; engine J65-7.

Enter chart at gross weight (A), (24,000 pounds); read directly across to pressure altitude (B), (4000 feet); then follow a vertical path downward to temperature (C), 120°F; read directly across to runway length available (D), (7000 feet); read along a vertical path from (D) to scale denoting dry runway, and read dry runway refusal speed (E), (106.5 KCAS). To obtain wet runway refusal speed enter correction chart at dry runway refusal speed (E), (106.5 KCAS); then follow a vertical path downward to runway length available (D^1) (7000 feet); read along a horizontal path to scale denoting wet runway, and read wet runway refusal speed (F), (87 KCAS).

(b) Prevailing Wind Condition

To correct for head wind or tailwind use the following procedure:

Correct for a tailwind by subtracting the tailwind (knots) from the wet or dry runway refusal speeds.

Correct for a head wind by adding the head wind (knots) to the wet or dry runway refusal speeds.

CONDITION

Correct wet runway refusal speed (F), (87 KCAS) from zero wind condition to 20 knot tailwind condition.

Corrected wet runway refusal speed (F) 87 KCAS for a 20 knot tailwind becomes (F^1) 67 KCAS.

Example: II Drag Chute Not Deployed.

(a) Zero Wind

Determine the refusal speed for an aircraft when the runway length available for takeoff is 9300 feet. Prevailing conditions are as follows: gross weight 24,000 pounds; pressure altitude 6000 feet; temperature 60°F (15.6°C); wet runway and dry runway; drag chute not deployed; zero wind, engine — J65-7.

Enter chart at gross weight (A), (24,000 pounds); read directly across to pressure altitude (B), (6000 feet); then follow a vertical path downward to temperature (C), 60 F; read directly across to base line (D); then continue across to runway length available (E), (9300 feet); read along a vertical path from (E) to scale denoting dry runway refusal speed and read dry runway refusal speed (F), (125.5 KCAS); to determine wet runway refusal speed read along a vertical path from (D) to prevailing, gross weight (A), (24,000 pounds) at G; from (F) read along a vertical path and from (G) follow parallel to guide line until vertical path from (F) is intercepted by line from (G) at (H); follow along a horizontal path to scale denoting wet runway refusal speed and read wet runway refusal speed (I), (95 KCAS).

(b) Prevailing Wind Condition

To correct for head wind or tailwind use procedure as stated in example I, Part (b).

CONDITION

Correct dry runway refusal speed (F), (125.5 KCAS) from zero wind condition to 10 knot head wind condition.

Corrected dry runway refusal speed (F) 125.5 KCAS for a 10 knot head wind becomes (F^1) 135.5 KCAS.

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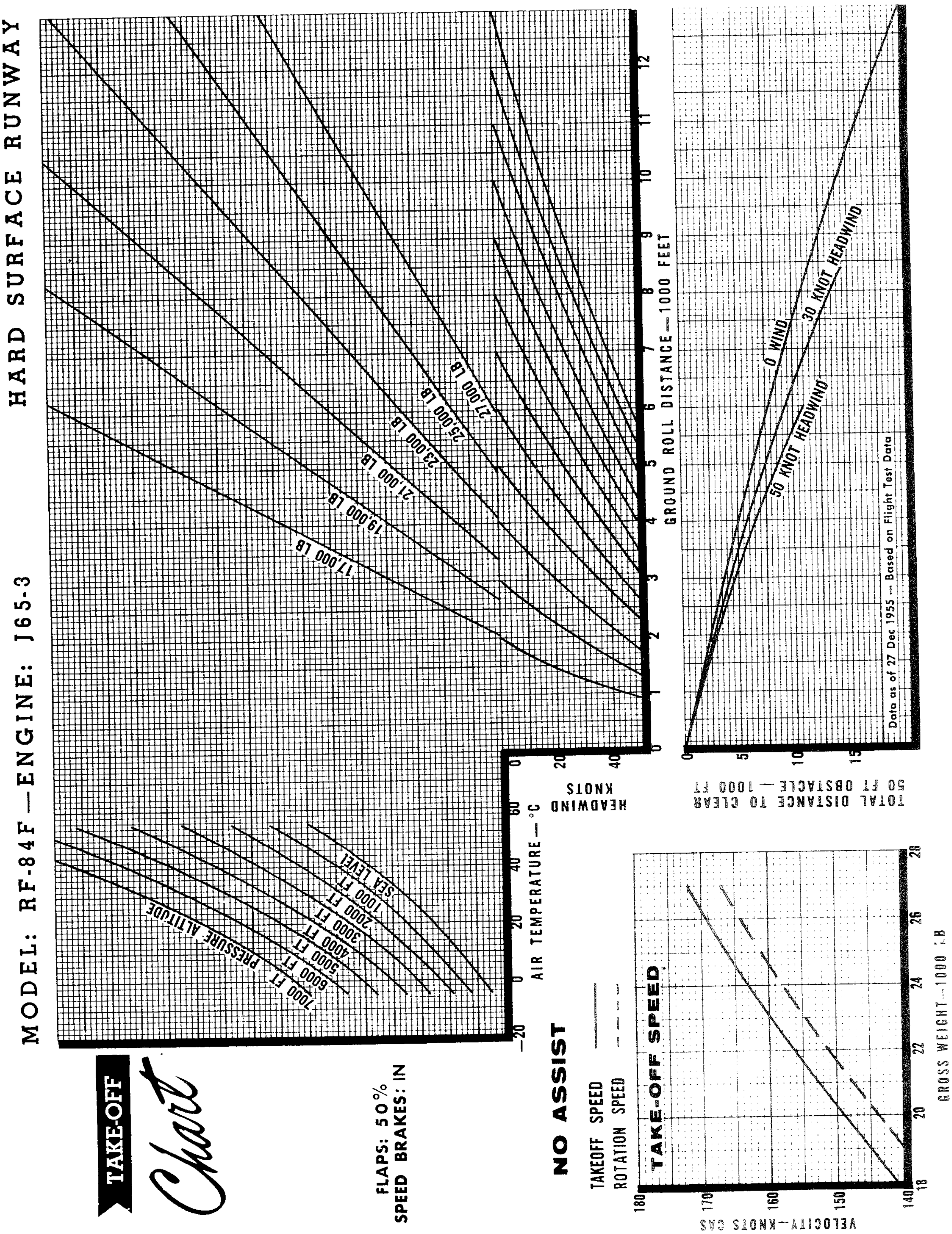

TAKE-OFF Chart
MODEL: RF-84F — ENGINE: J65-3
HARD SURFACE RUNWAY
FLAPS: 50%
SPEED BRAKES: IN
NO ASSIST
PRESSURE ALTITUDE
7000 FT
6000 FT
5000 FT
4000 FT
3000 FT
2000 FT
1000 FT
SEA LEVEL
AIR TEMPERATURE — °C
-20
0
20
40
60
HEADWIND KNOTS
0
20
40
17,000 LB
19,000 LB
21,000 LB
23,000 LB
25,000 LB
27,000 LB
GROUND ROLL DISTANCE — 1000 FEET
0
1
2
3
4
5
6
7
8
9
10
11
12
0 WIND
30 KNOT HEADWIND
50 KNOT HEADWIND
TOTAL DISTANCE TO CLEAR 50 FT OBSTACLE — 1000 FT
0
5
10
15
Data as of 27 Dec 1955 -- Based on Flight Test Data
TAKE-OFF SPEED
TAKEOFF SPEED
ROTATION SPEED
VELOCITY—KNOTS CAS
140
150
160
170
180
GROSS WEIGHT—1000 LB
18
20
22
24
26
28

Figure A2-1

MODEL: RF-84F — ENGINE: J65-W-3 HARD SURFACE RUNWAY

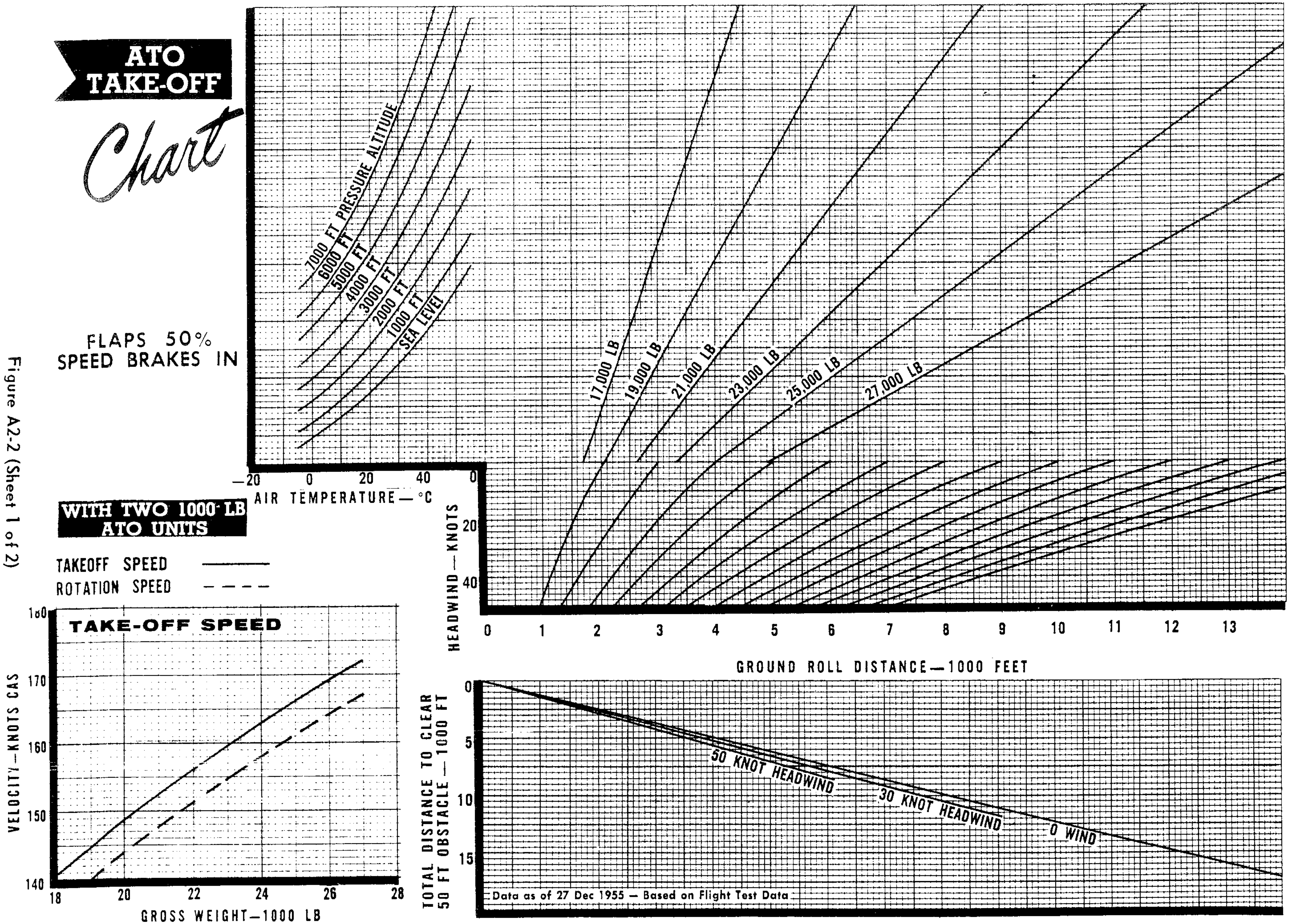


Figure A2-2 (Sheet 1 of 2)

SATO IGNITION SPEED Chart

MODEL: RF-84F
ENGINE: J65-3
HARD SURFACE RUNWAY

WITH TWO 1000 LB ATO UNITS

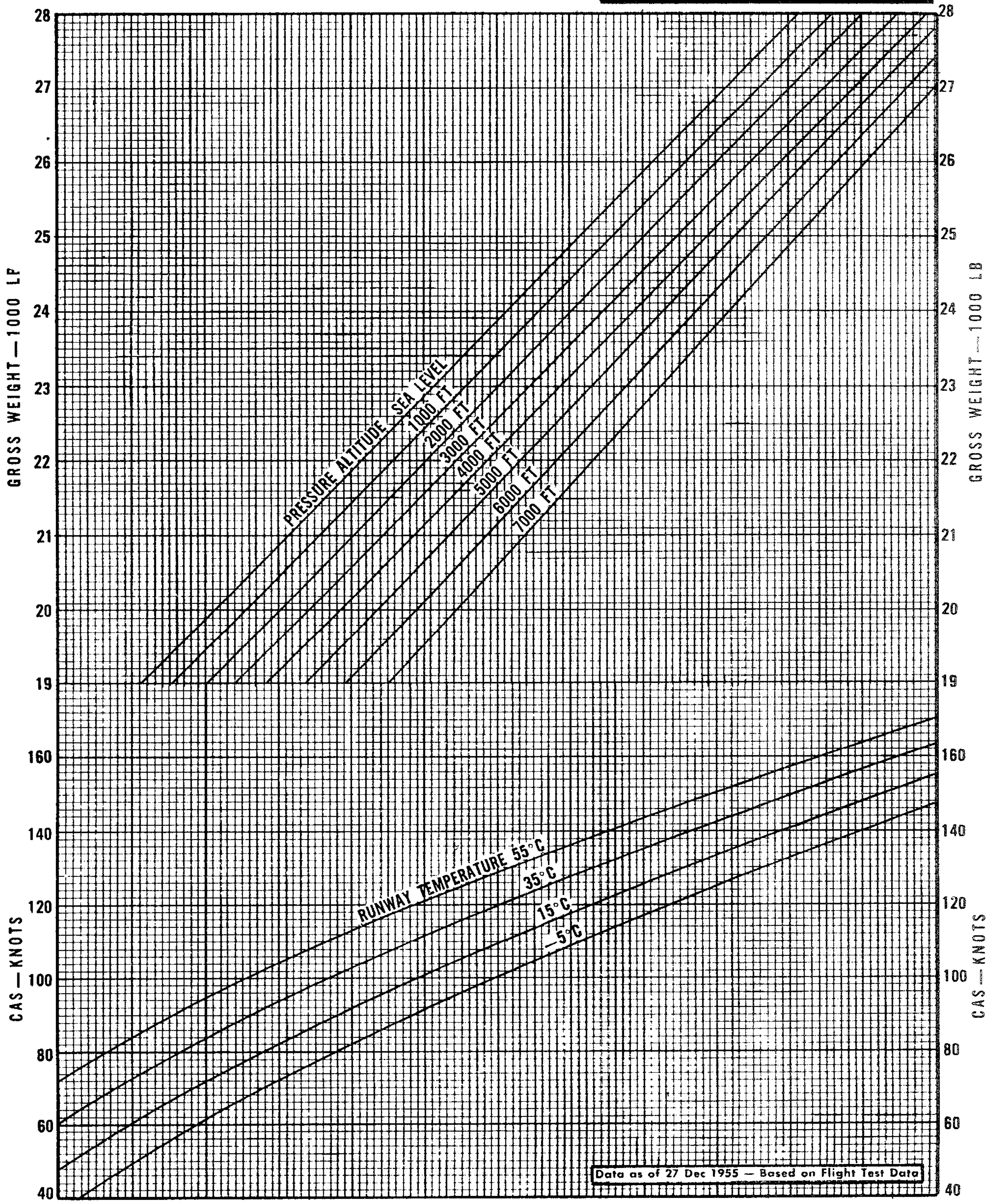


Figure A2—2 (Sheet 2 of 2)

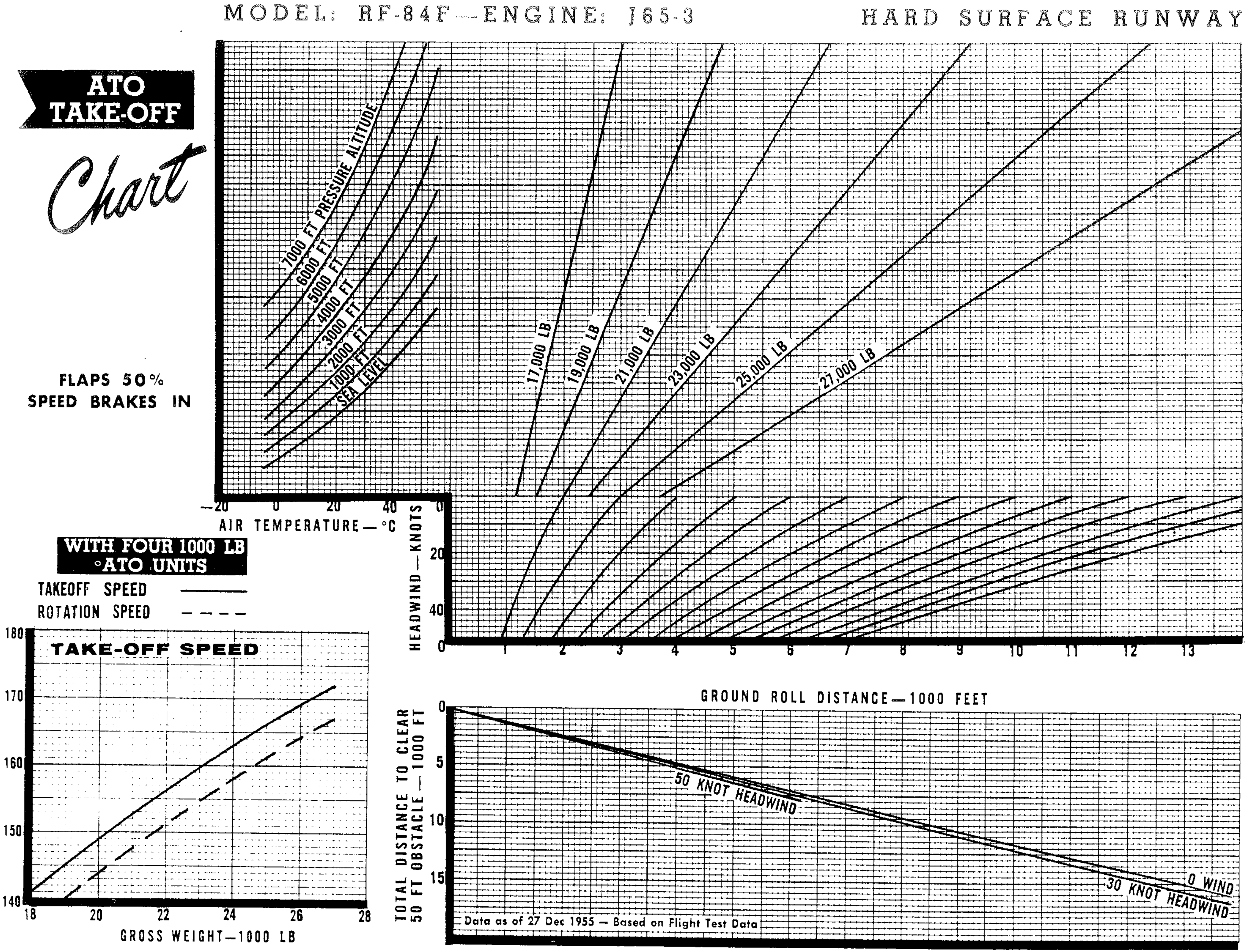


Figure A2-3 (Sheet 1 of 2)

ATO IGNITION SPEED Chart

MODEL: RF-84F
ENGINE: J65-3
HARD SURFACE RUNWAY

WITH FOUR 1000 LB ATO UNITS

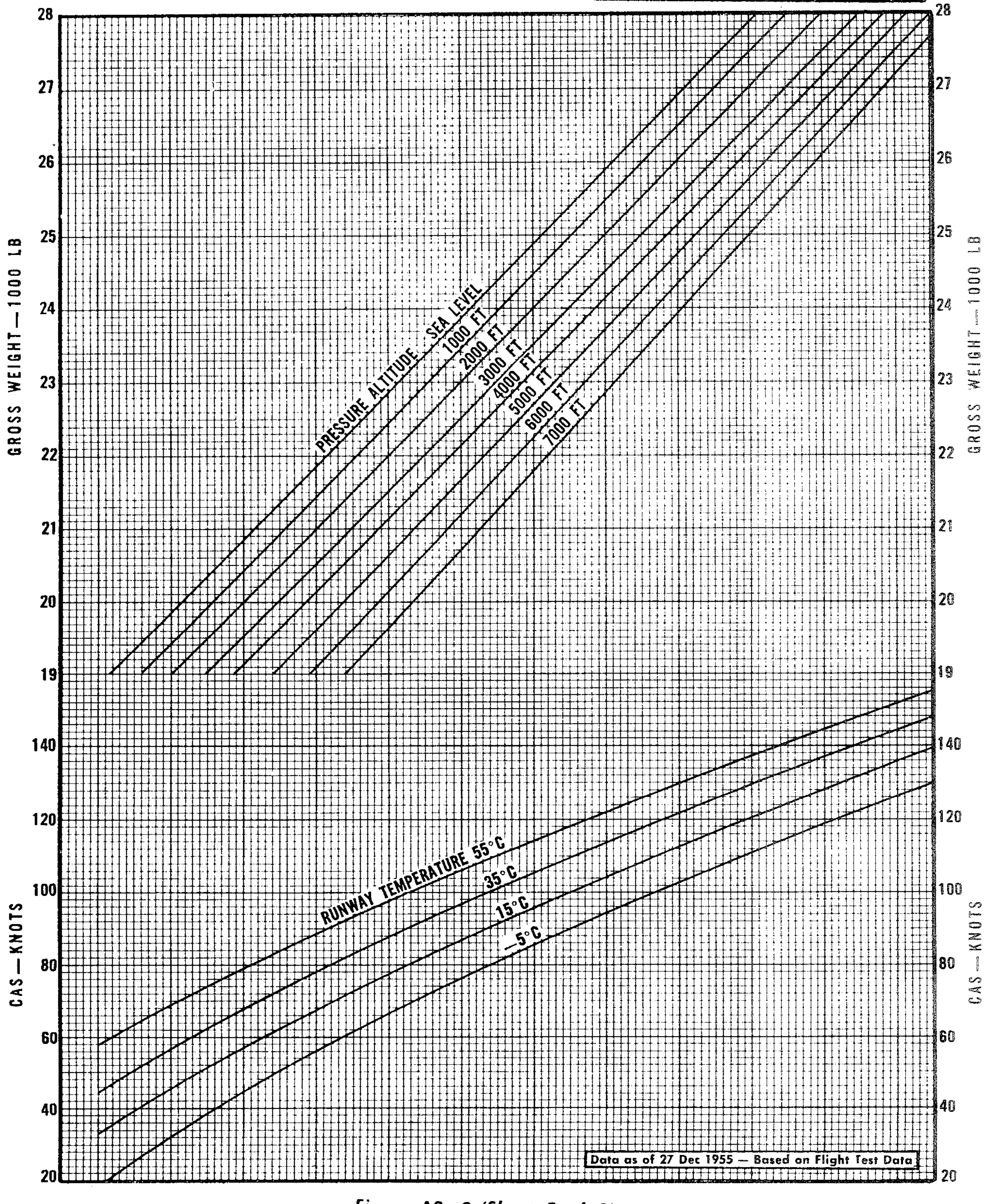


Figure A2—3 (Sheet 2 of 2)

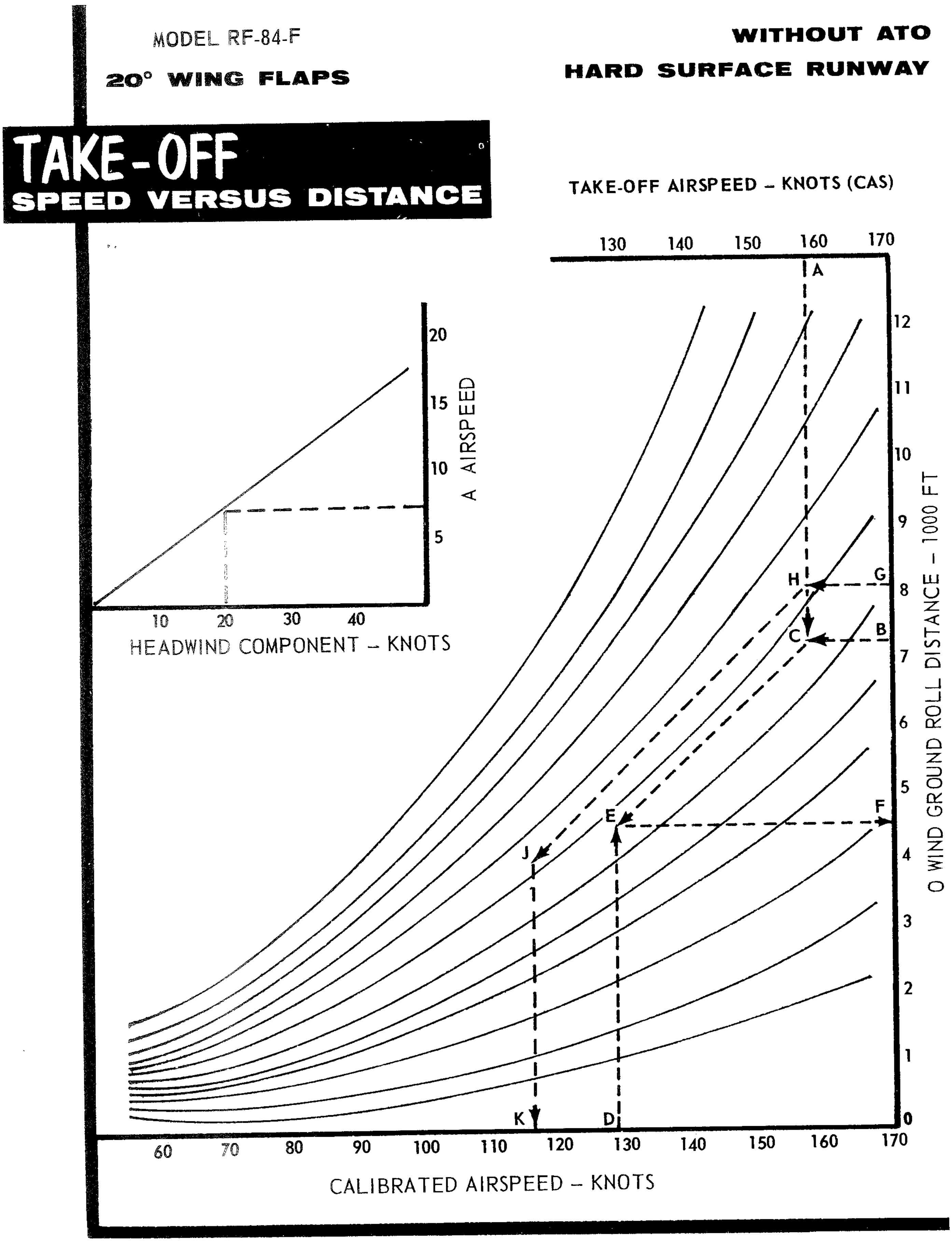

MODEL RF-84-F
20° WING FLAPS
WITHOUT ATO
HARD SURFACE RUNWAY
TAKE-OFF
SPEED VERSUS DISTANCE
TAKE-OFF AIRSPEED – KNOTS (CAS)
130 140 150 160 170
A AIRSPEED
20 15 10 5
10 20 30 40
HEADWIND COMPONENT – KNOTS
0 WIND GROUND ROLL DISTANCE – 1000 FT
12 11 10 9 8 7 6 5 4 3 2 1 0
A B C D E F G H J K
60 70 80 90 100 110 120 130 140 150 160 170
CALIBRATED AIRSPEED – KNOTS

Figure A2–4

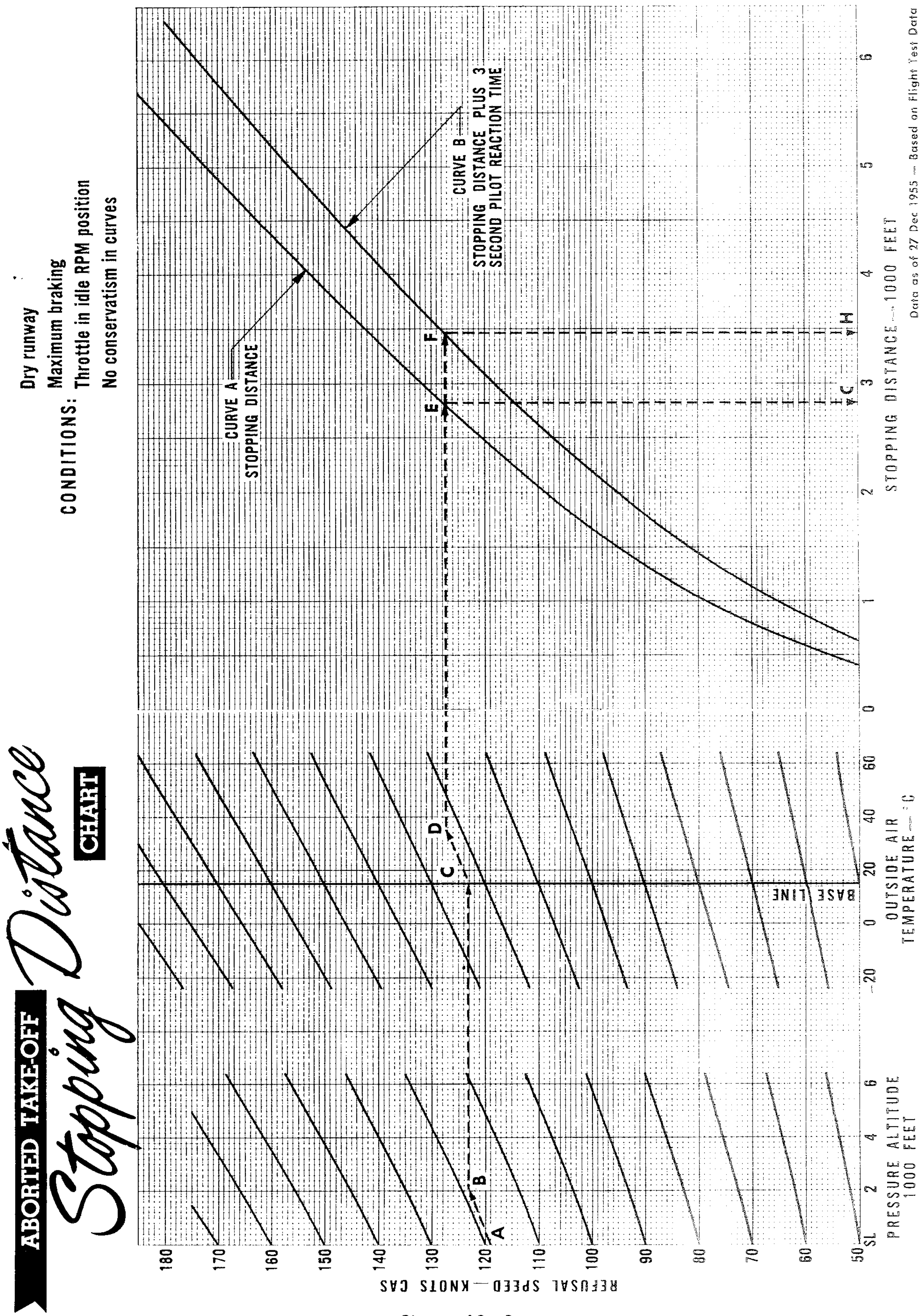

ABORTED TAKE-OFF
Stopping Distance
CHART
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY
CONDITIONS: Dry runway
Maximum braking
Throttle in idle RPM position
No conservatism in curves
CURVE A
STOPPING DISTANCE
CURVE B
STOPPING DISTANCE PLUS 3 SECOND PILOT REACTION TIME
BASE LINE
REFUSAL SPEED — KNOTS CAS
PRESSURE ALTITUDE 1000 FEET
OUTSIDE AIR TEMPERATURE — °C
STOPPING DISTANCE — 1000 FEET
Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A2—5

MAXIMUM REFUSAL SPEED

Military Thrust
With Drag Chute

model:	(R)F-84F
engine:	J65-3
fuel grade:	JP-4
fuel density:	6.5 LB/GAL
data date:	DECEMBER 1962
data basis:	FLIGHT TEST

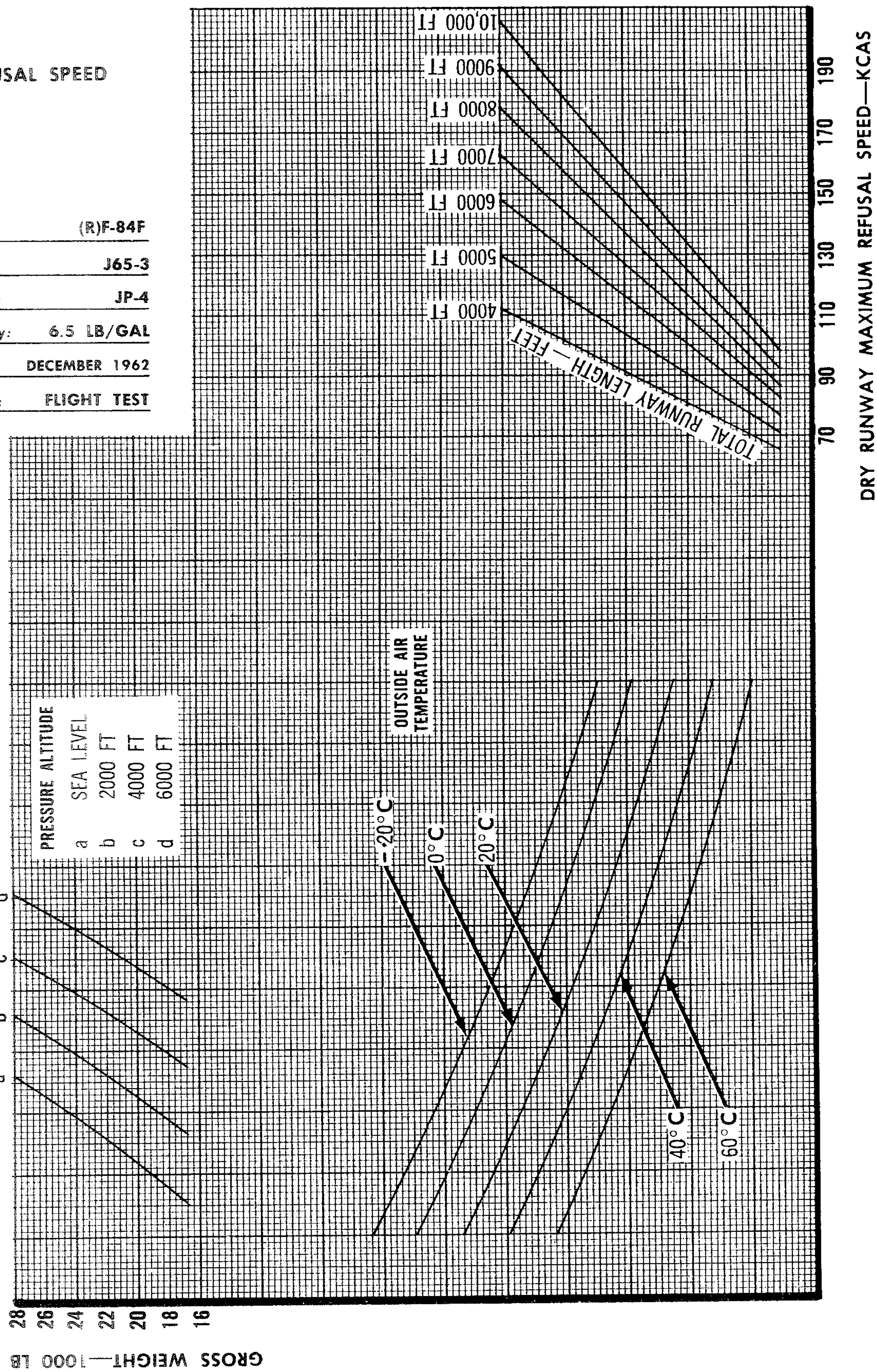


Figure A2—6 (Sheet 1 of 2)

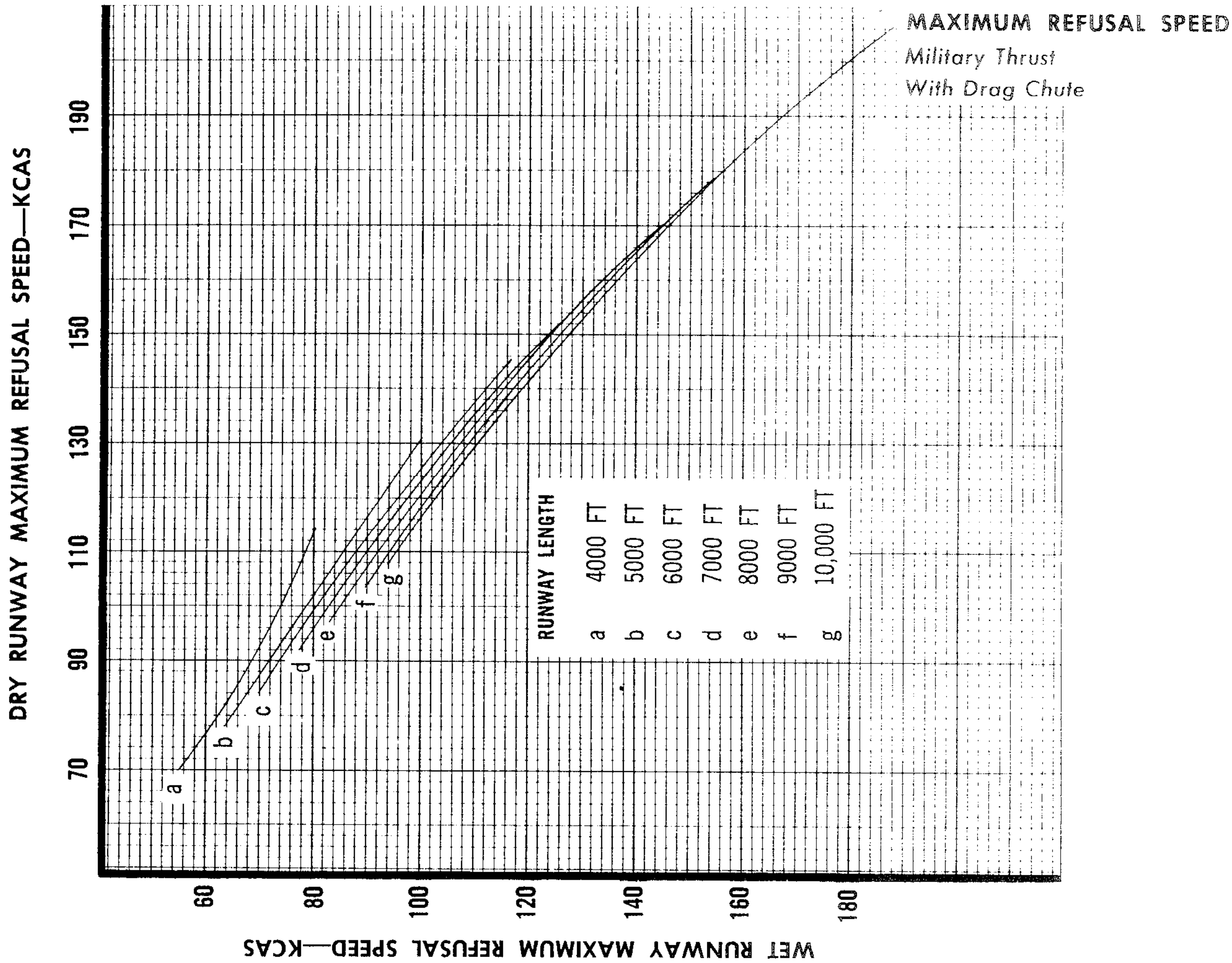


Figure A2—6 (Sheet 2 of 2)

MAXIMUM REFUSAL SPEED

Military Thrust
No Drag Chute

model:	(R)F-84F
engine:	J65-3
fuel grade:	JP-4
fuel density:	6.5 LB/GAL
data date:	DECEMBER 1962
data basis:	FLIGHT TEST

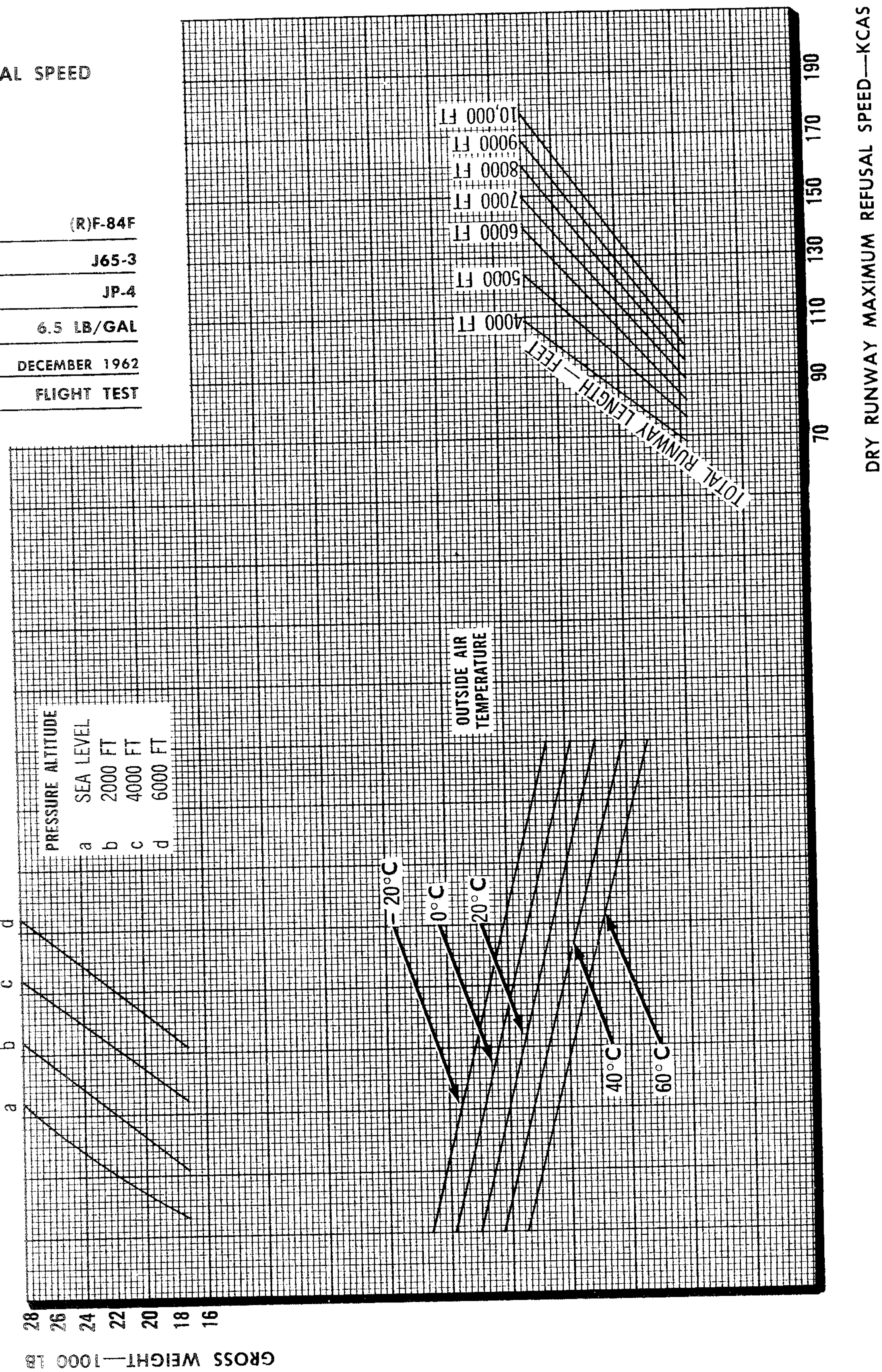


Figure A2—7 (Sheet 1 of 2)

MAXIMUM REFUSAL SPEED

Military Thrust

No Drag Chute

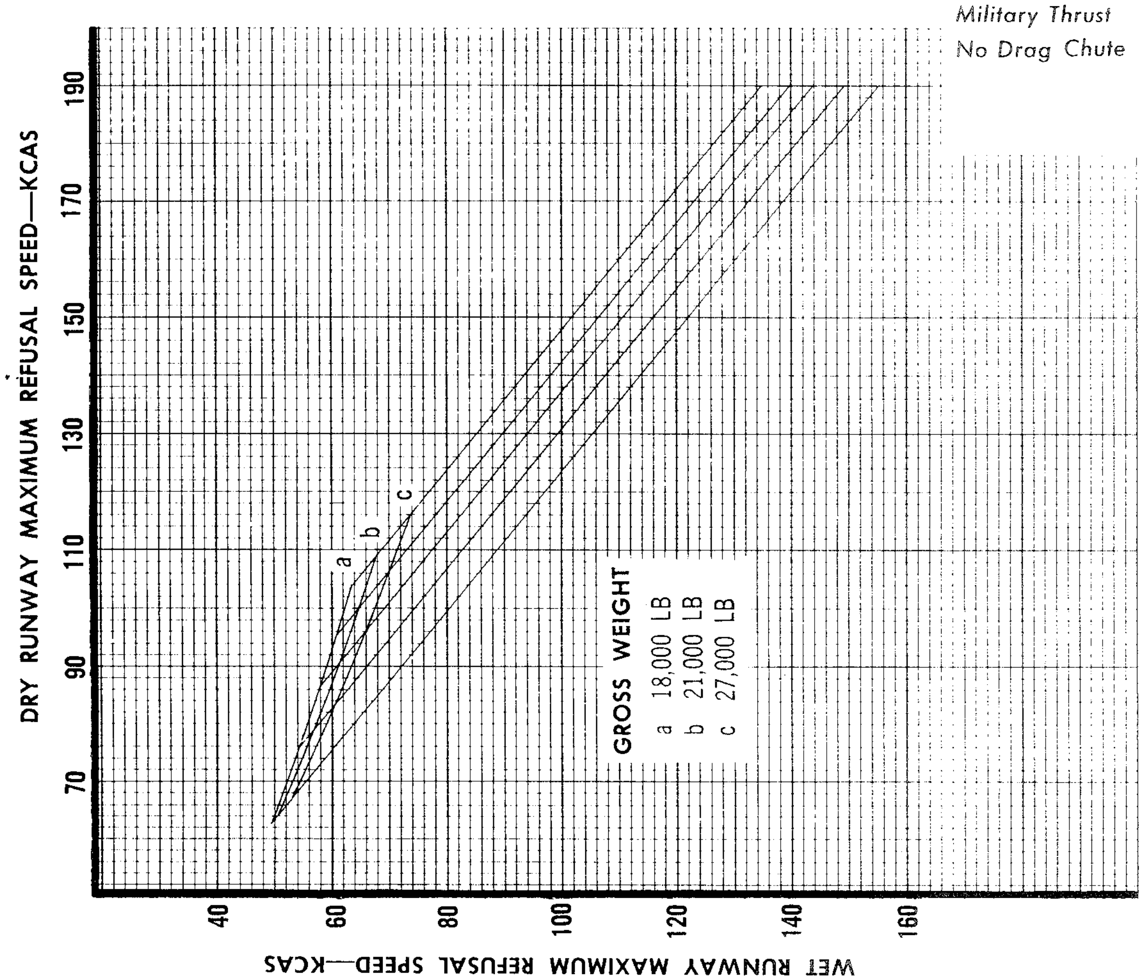


Figure A2—7 (Sheet 2 of 2)

TAKEOFF AND LANDING

CROSSWIND CHART

LANDING CONFIGURATION

DATA BASIS: FLIGHT TEST

DATA DATE: 1 JULY 1959

ENGINE J65

FUEL GRADE: JP-4

FUEL DENSITY: 6.5 LB/GAL

HOW TO USE CHART

GIVEN:

Crosswind 33 knots at 63 degrees to runway. Determine if takeoff speed of 135 knots IAS is recommended for clean aircraft.

SOLUTION:

a. Enter with the maximum gust velocity (other factors such as tire limit speed, nose gear limits, etc. should also be considered).

b. Determine coordinates of wind velocity and direction (a).

c. Proceed vertically to intersect grid representing 135 knots IAS (b).

d. This point is in the recommended area.

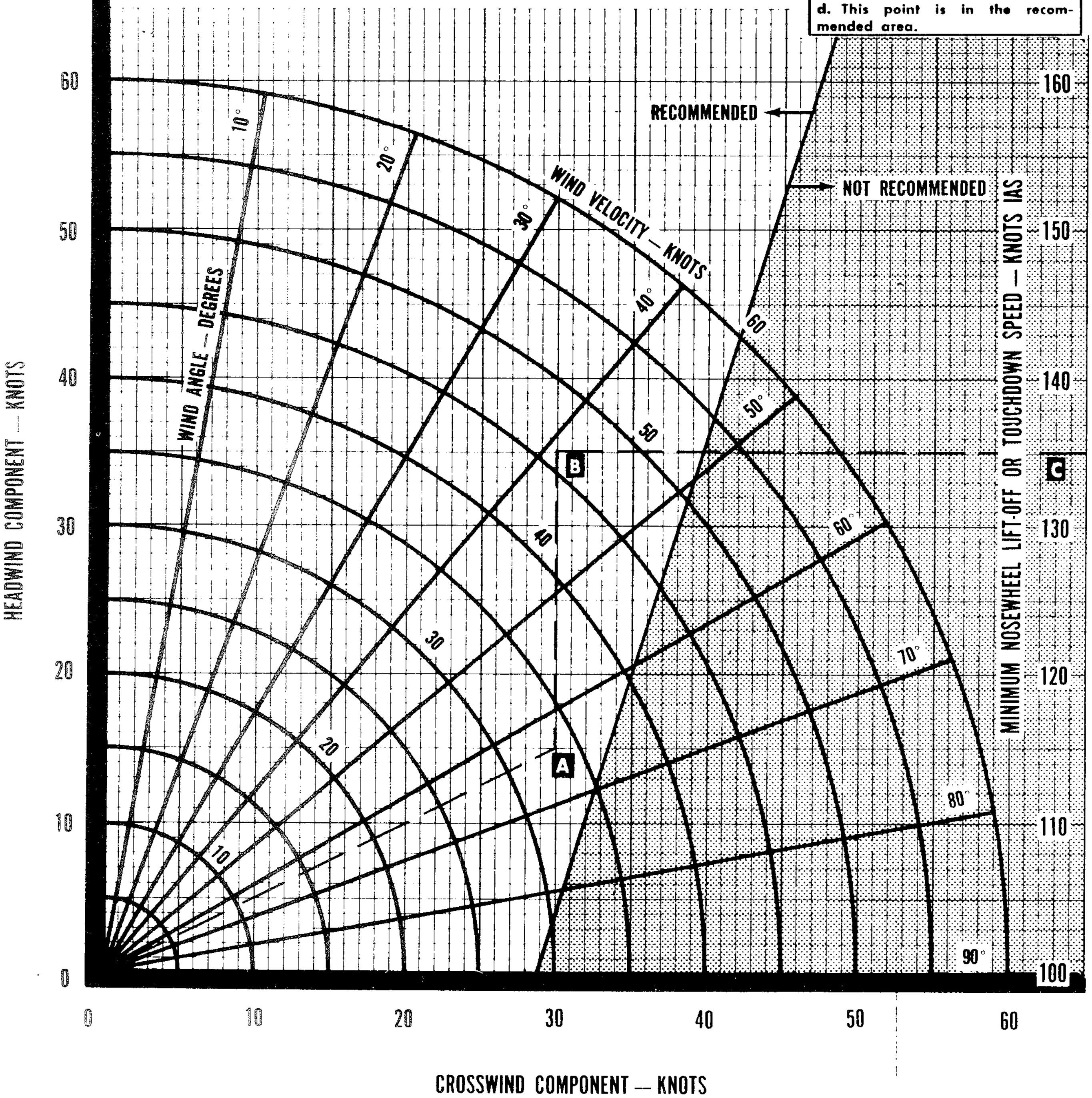


Figure A2—8 (Sheet 1 of 2)

TAKEOFF AND LANDING CROSSWIND CHART

LANDING CONFIGURATION WITH TWO WING TANKS INSTALLED

HOW TO USE CHART

GIVEN:
Crosswind 33 knots at 63 degrees to runway. Determine minimum takeoff speed for given conditions for aircraft with two tanks.

SOLUTION:
a. Enter with the maximum gust velocity (other factors such as tire limit speed, nose gear limits, etc. should also be considered).
b. Determine coordinates of wind velocity and direction (a).
c. Proceed vertically to intersect point between recommended and not-recommended areas (b).
d. Minimum speed 145.5 knots IAS.

DATA BASIS: FLIGHT TEST

DATA DATE: 1 JULY 1959

ENGINE J65

FUEL GRADE: JP-4

FUEL DENSITY: 6.5 LB/GAL

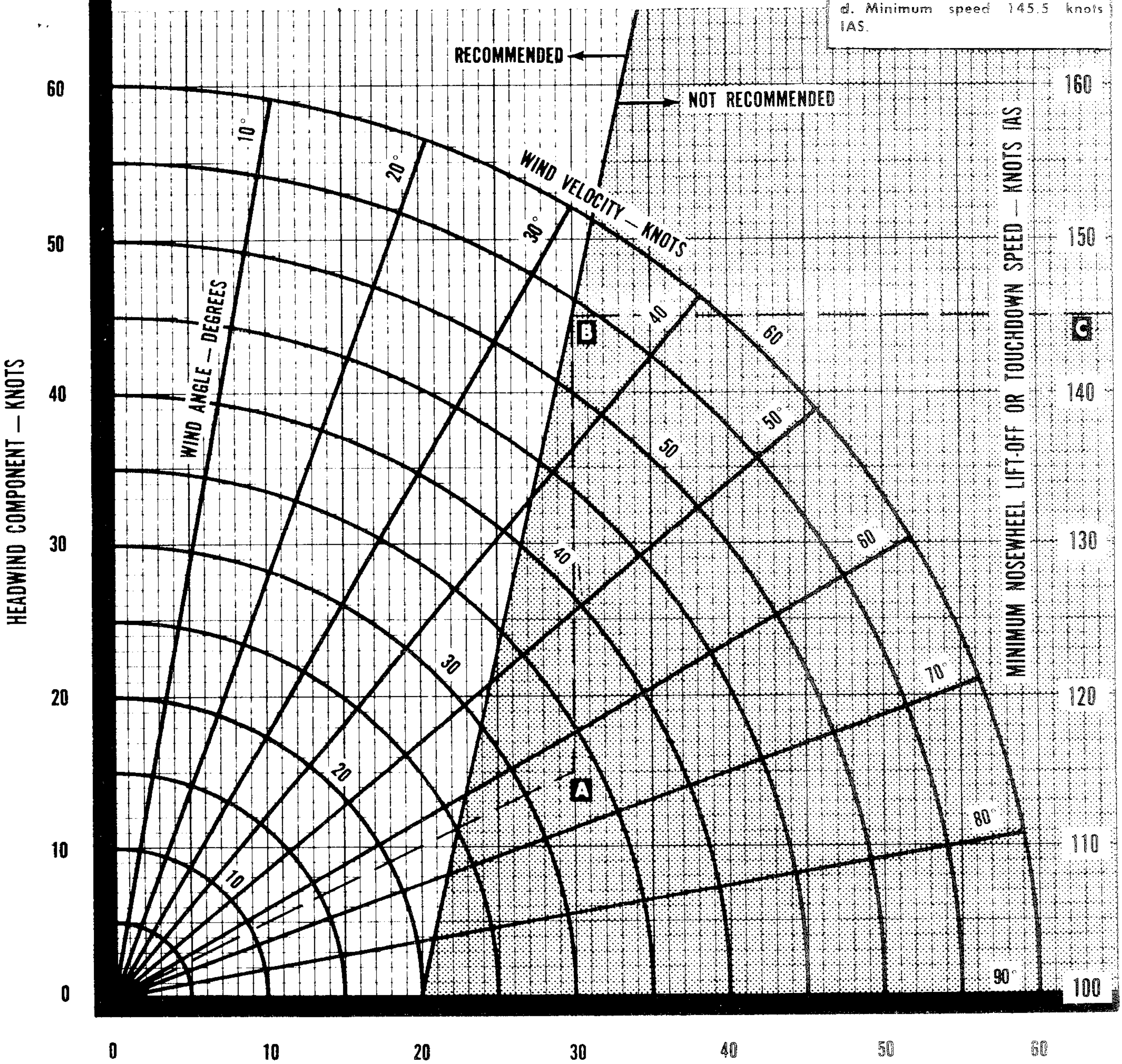


Figure A2—8 (Sheet 2 of 2)

PART 3 CLIMB

TABLE OF CONTENTS

PART 3 CLIMB

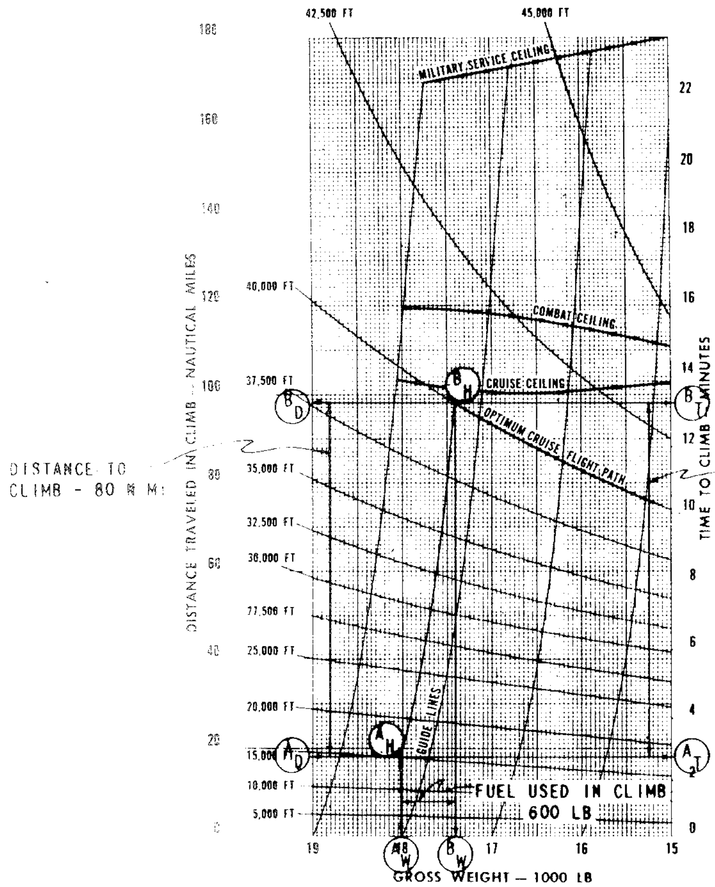
TABLE OF CONTENTS

CLIMB — GRAPHICAL

MODEL: RF-84F

STANDARD DAY

CLEAN AIRPLANE



Sample Chart

TIME TO CLIMB
10.5 MIN

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
45,000	220	.81
40,000	245	.81
35,000	275	.81
30,000	300	.79
25,000	315	.75
20,000	330	.71
15,000	340	.67
10,000	355	.64
5,000	365	.60
SEA LEVEL	375	.57

CLIMB SCHEDULE

Data as of 27 December 1955
Based on Flight Test Data

CLIMB.

DESCRIPTION.

Climb charts for Military, 98% Military, and Normal Thrust operation based on a recommended climb speed schedule, are shown for each configuration. Time and distance are plotted against gross weight with guide lines to show the reduction in gross weight during climb due to the fuel used. Service ceiling (100 FPM), Combat Ceiling (500 FPM), Cruise Ceiling (300 FPM at Normal Thrust) and optimum cruise flight path are superimposed on the graph.

USE.

To obtain the climb data desired, enter the proper climb chart at the gross weight and altitude at start of climb. Note the time and distance at this point. From this initial altitude point, trace a curve parallel to the guide lines until it intersects the desired altitude at end of climb. Note the time, distance, and gross weight at this intersection. The difference between the initial and final time is the time required to climb. The difference between initial and final values for distance and for gross weight gives, respectively, the distance traveled and fuel used to climb. Since time and distance are zero at sea level, the time required and distance traveled may be read directly for climbs starting at sea level. Fuel used, however, must still be determined by the difference in gross weights. The example shows the fuel used, distance traveled, and time to climb from 15,000 feet to 40,000 feet, using military thrust, clean aircraft, with a gross weight of 18,000 pounds at start of climb.

Note

Fuel used for start, take-off, taxi and acceleration to climb is not accounted for in the climb charts.

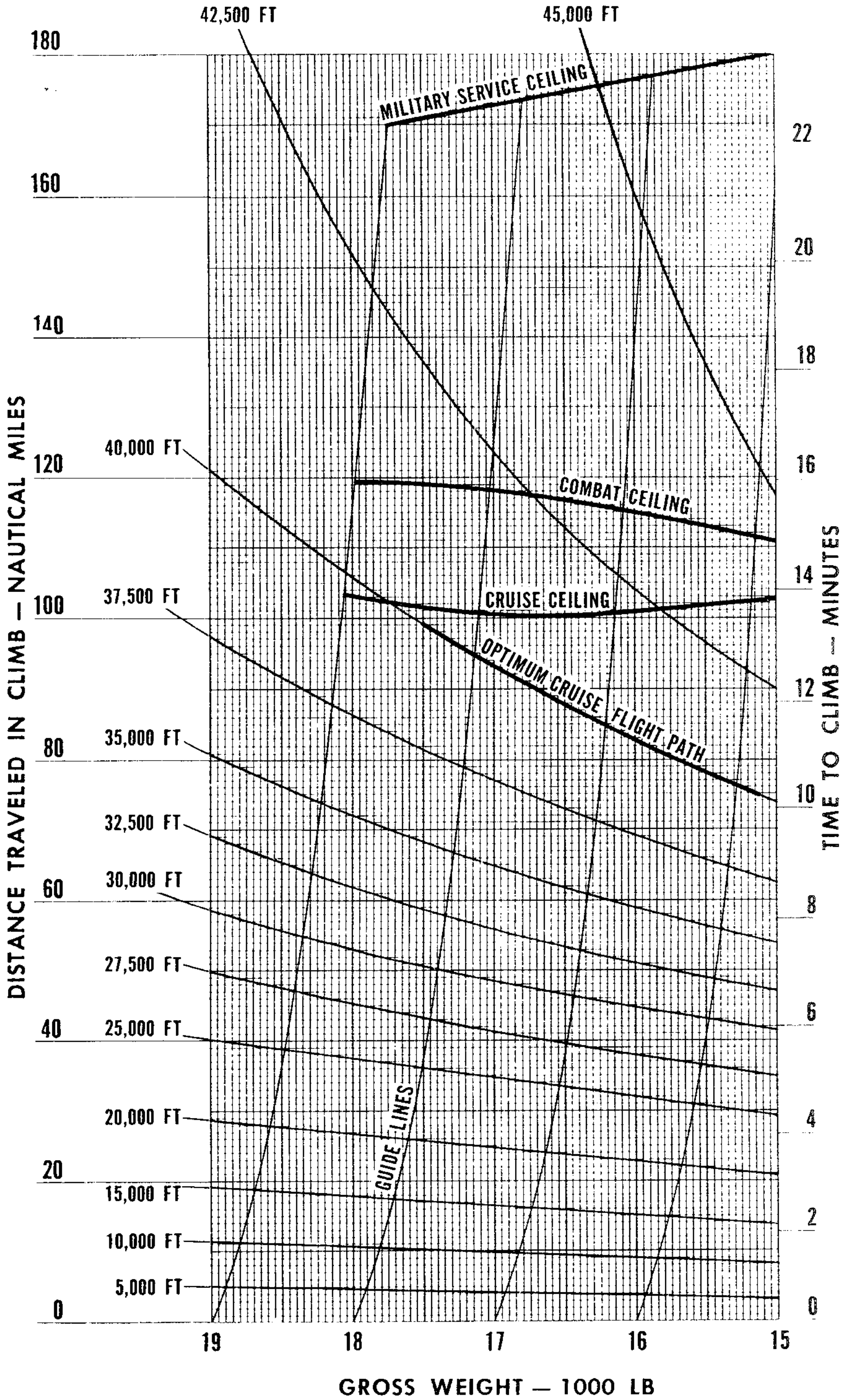
EXAMPLE:

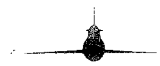
A_w is initial gross weight (18,000 LB).
A_h is initial altitude (15,000 FT).
A_d is initial distance (18 nautical miles).
A_t is initial time (2.8 MIN).
B_h is final altitude (40,000 FT).
B_w is final gross weight (17,400 LB).
B_d is final distance (498 nautical miles).
B_t is final time (13.3 MIN).
A_w-B_w is fuel used (600 LB).
B_d-A_d is distance traveled (80 nautical miles).
B_t-A_t is time to climb (10.5 MIN).

Military Power Climb

100% RPM

MODEL: RF-84F
ENGINE:
J65-3
STANDARD DAY





	CLIMB SPEEDS	
ALTITUDE FEET	CAS KNOTS	MACH NO
45,000	220	.81
40,000	245	.81
35,000	275	.81
30,000	300	.79
25,000	315	.75
20,000	330	.71
15,000	340	.67
10,000	355	.64
5,000	365	.60
SEA LEVEL	375	.57

Data as of 27 December 1955
Based on Flight Test Data

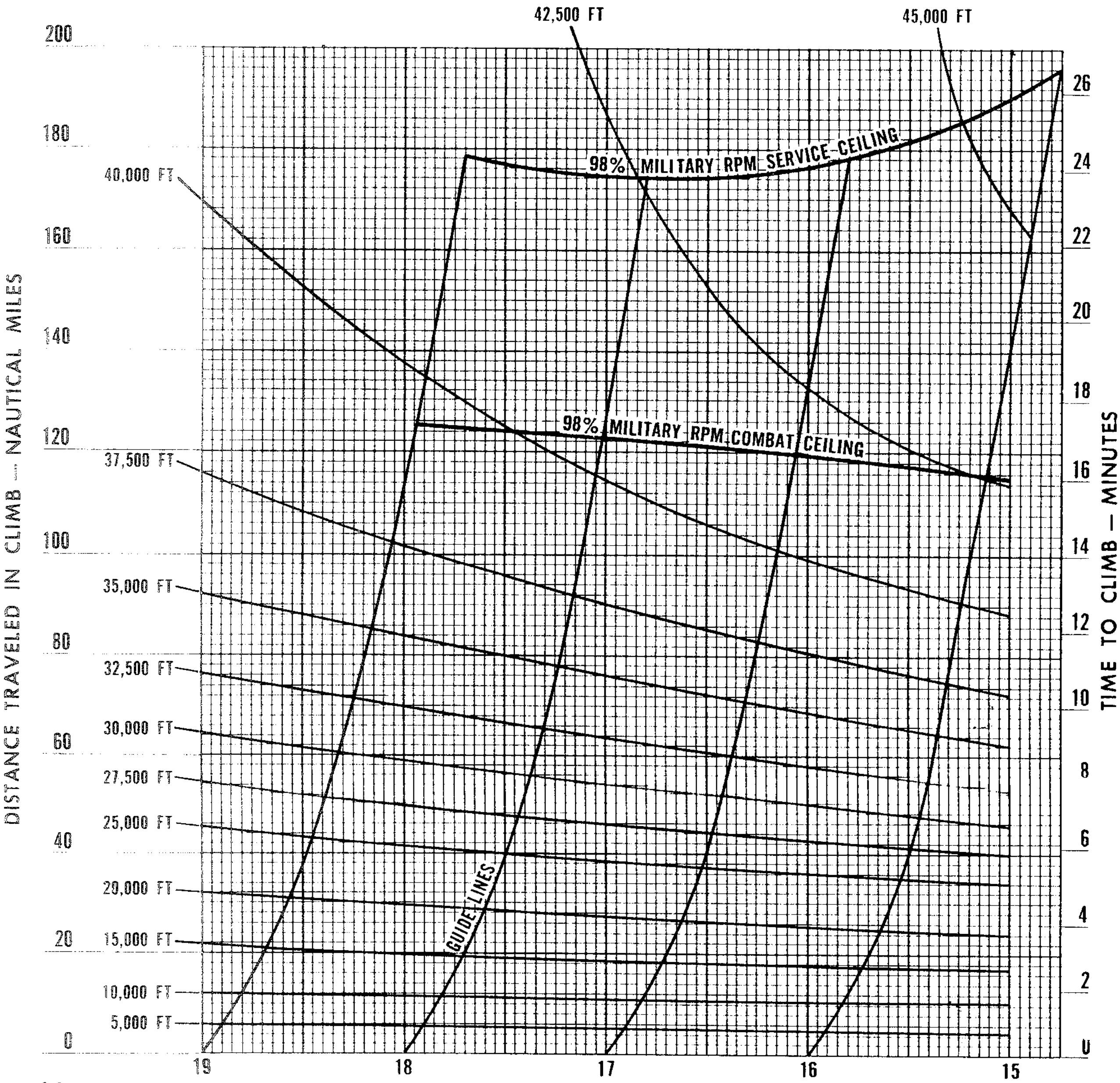
Figure A3—1

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
45,000	220	.81
40,000	245	.81
35,000	275	.81
30,000	285	.76
25,000	300	.72
20,000	315	.68
15,000	330	.65
10,000	335	.60
5,000	345	.57
SEA LEVEL	355	.54

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

98% Military RPM Climb

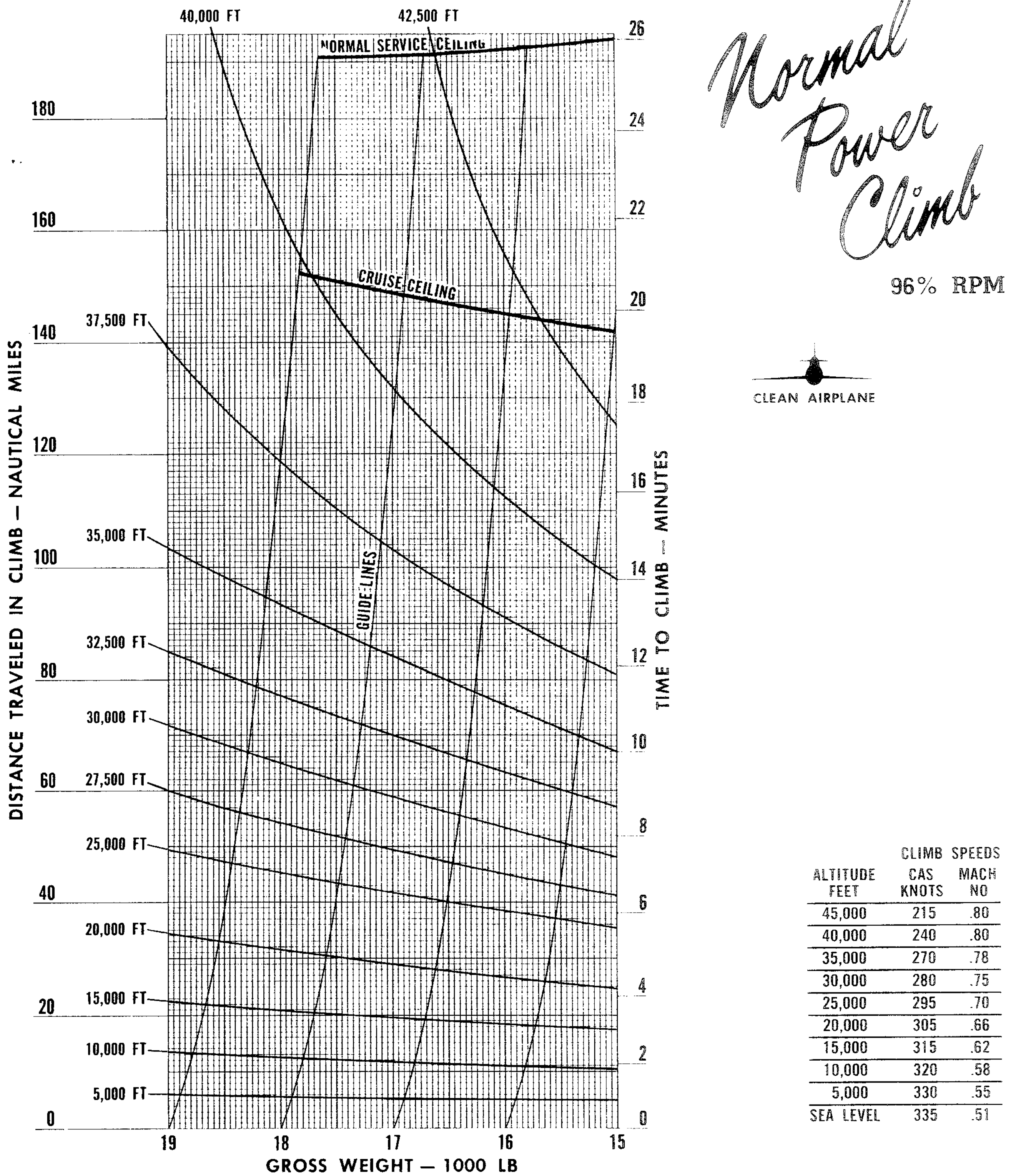
CLEAN AIRPLANE



Data as of 27 December 1955
Based on Flight Test Data

Figure A3—2

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY



ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
45,000	215	.80
40,000	240	.80
35,000	270	.78
30,000	280	.75
25,000	295	.70
20,000	305	.66
15,000	315	.62
10,000	320	.58
5,000	330	.55
SEA LEVEL	335	.51

Data as of 27 December 55
Based on Flight Test Data

Figure A3—3

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

100% RPM

Military Power Climb

CONFIGURATION
CLEAN I TWO 230 GAL CLASS I TANKS

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
45,000	210	.78
40,000	230	.78
35,000	260	.76
30,000	270	.72
20,000	295	.64
10,000	310	.56
SEA LEVEL	340	.51

Data as of 27 December 55
Based on Flight Test Data

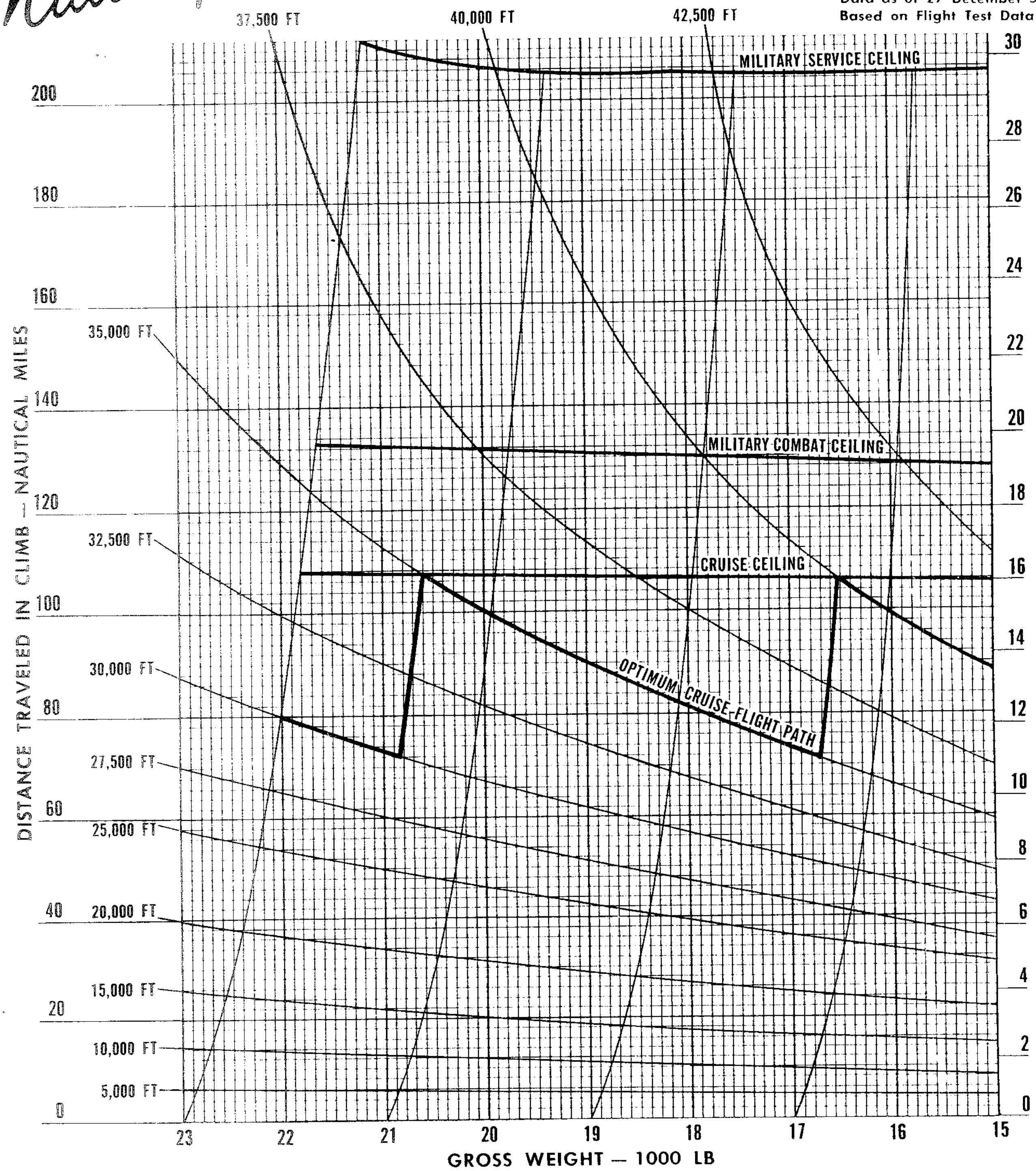


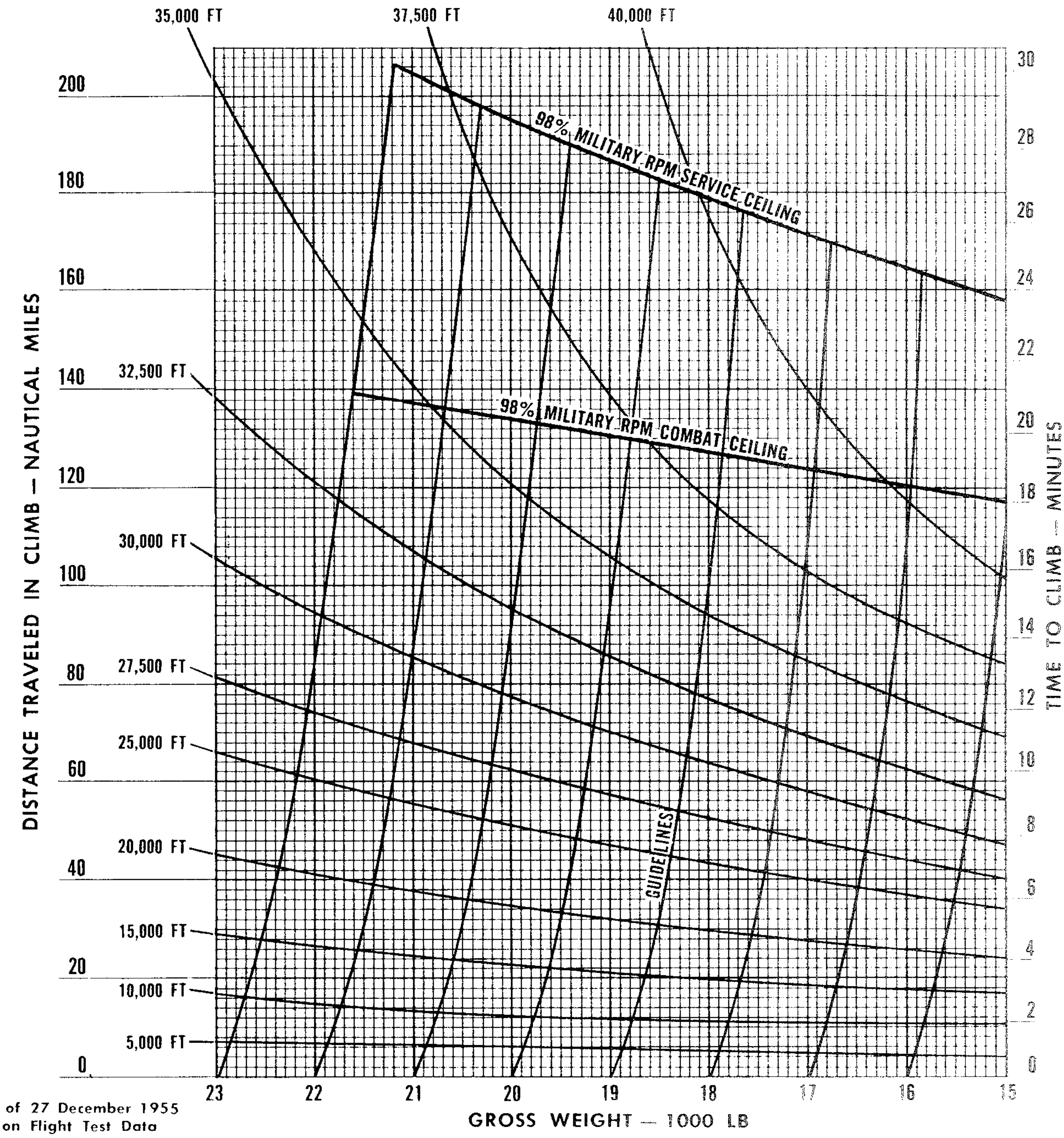
Figure A3—4

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION
CLEAN + TWO 230 GAL
CLASS I TANKS

98% Military RPM CLIMB

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
40,000	225	.75
35,000	250	.75
30,000	260	.70
25,000	270	.65
20,000	285	.62
15,000	295	.59
10,000	300	.54
5,000	305	.51
SEA LEVEL	315	.47



Data as of 27 December 1955
Based on Flight Test Data

Figure A3—5

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

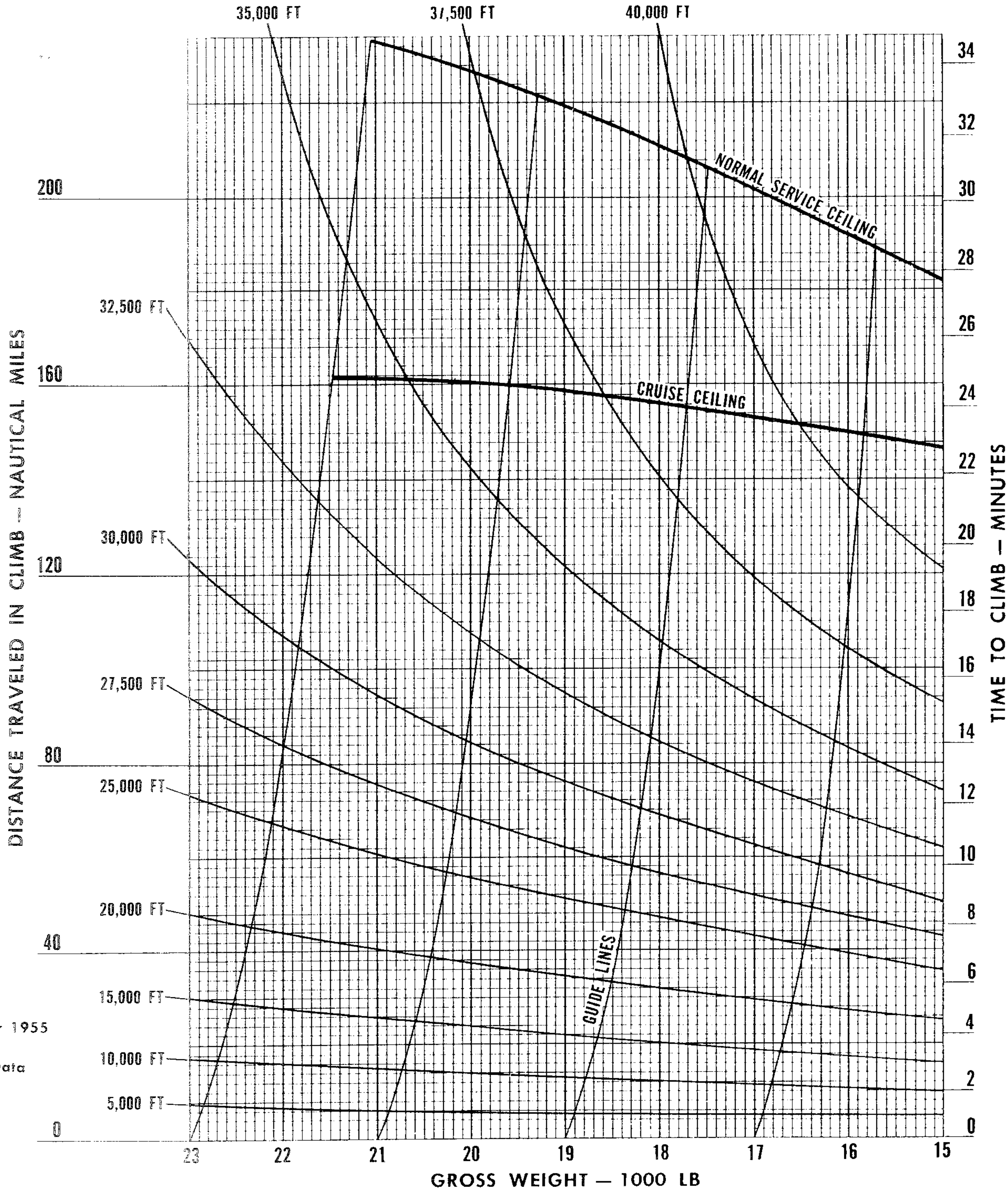
96% RPM

Normal Power Climb

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
45,000	205	.77
40,000	230	.77
35,000	250	.74
30,000	260	.69
20,000	275	.60
10,000	290	.52
SEA LEVEL	290	.44

CONFIGURATION

CLEAN + TWO 230 GAL CLASS I TANKS



Data as of 27 December 1955

Based on Flight Test Data

Figure A3—6

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

100% RPM

Military Power Climb

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
40,000	235	.78
35,000	265	.78
30,000	275	.73
25,000	285	.68
20,000	295	.64
15,000	305	.60
10,000	310	.56
5,000	315	.52
SEA LEVEL	320	.48

Data as of 27 December 1955
Based on Flight Test Data

CONFIGURATION
CLEAN + TWO 450 GAL
CLASS I TANKS

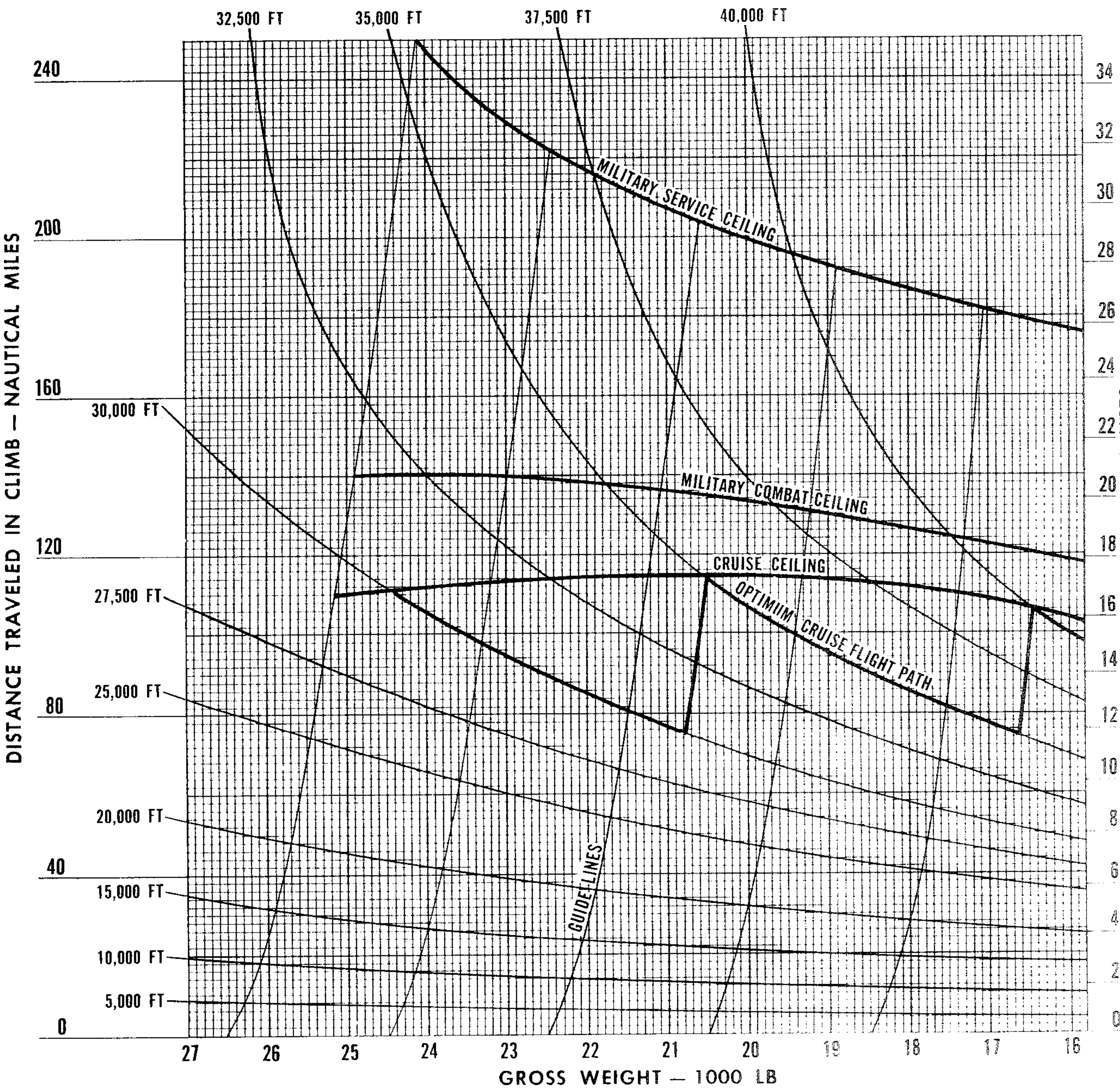
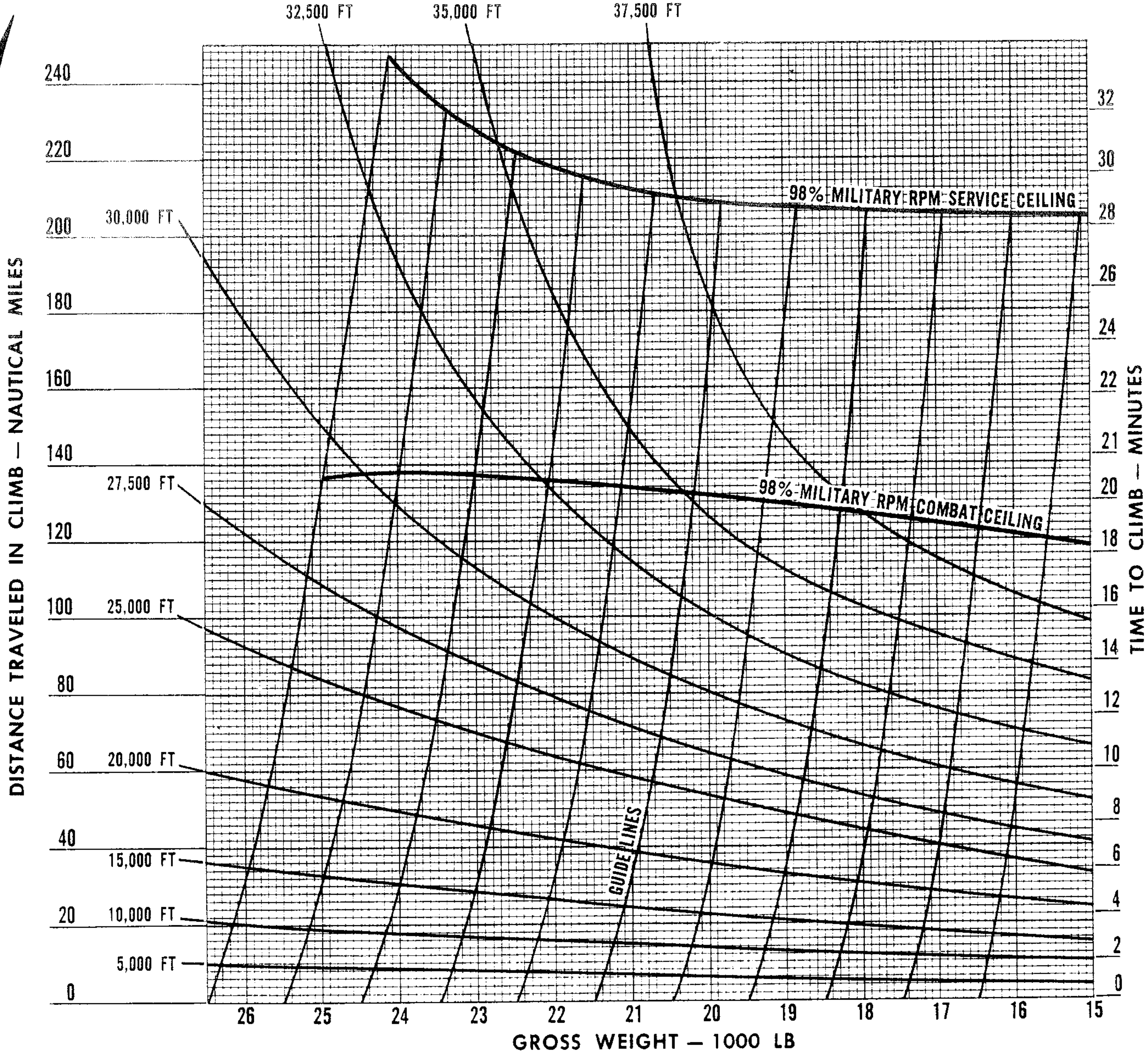


Figure A3—7

98% Military RPM CLIMB

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION
CLEAN + TWO 450 GAL
CLASS I TANKS



ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
35,000	250	.75
30,000	270	.71
25,000	280	.67
20,000	290	.63
15,000	295	.59
10,000	305	.55
5,000	310	.51
SEA LEVEL	315	.48

Data as of 27 December 1955
Based on Flight Test Data

Figure A3—8

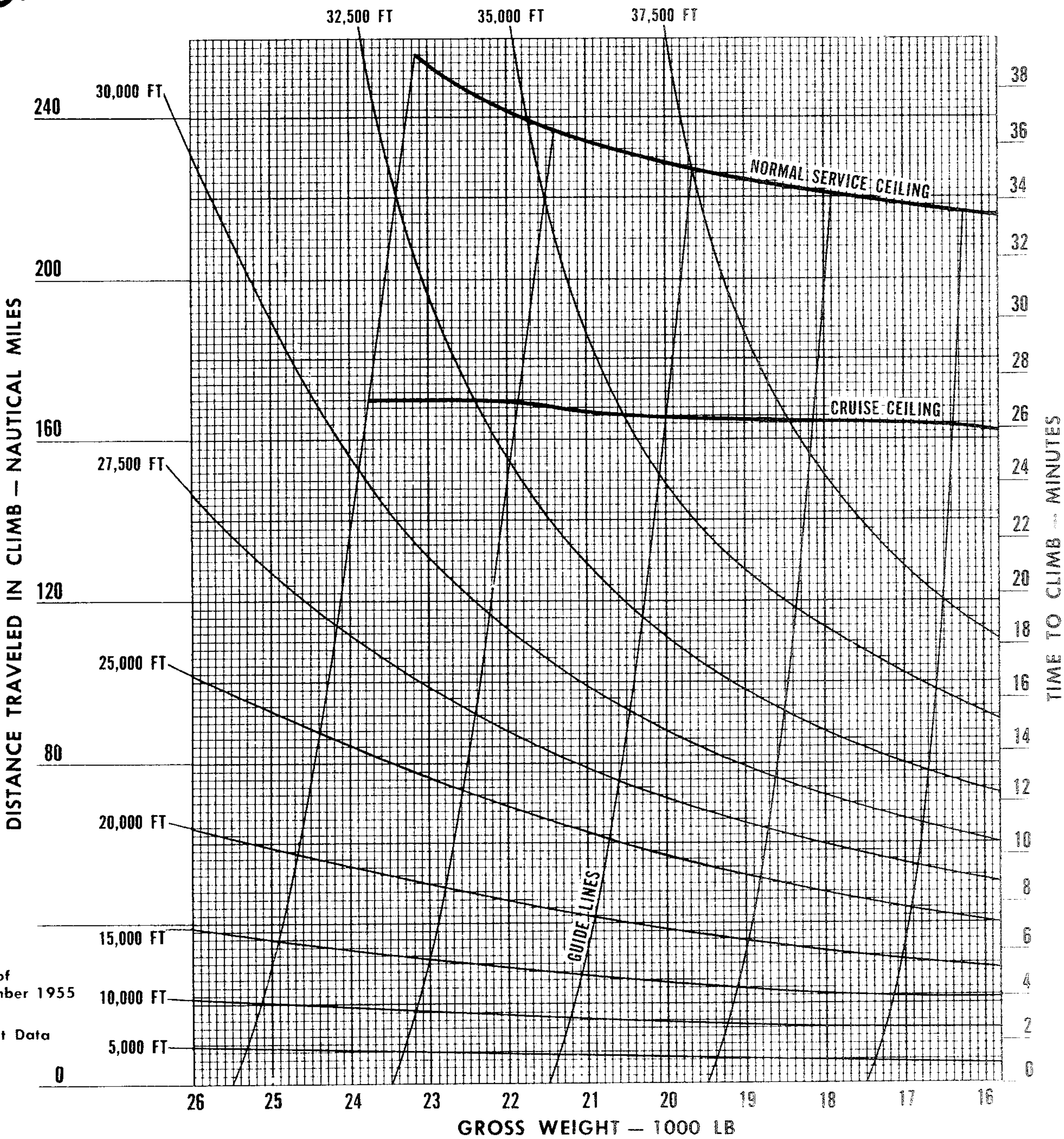
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

96% RPM

Normal Power Climb

CONFIGURATION
CLEAN + TWO 450 GAL CLASS I TANKS

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
35,000	245	.73
30,000	255	.69
25,000	265	.64
20,000	275	.60
15,000	285	.56
10,000	290	.52
5,000	295	.49
SEA LEVEL	300	.45



Data as of
27 December 1955
Based on
Flight Test Data

Figure A3—9

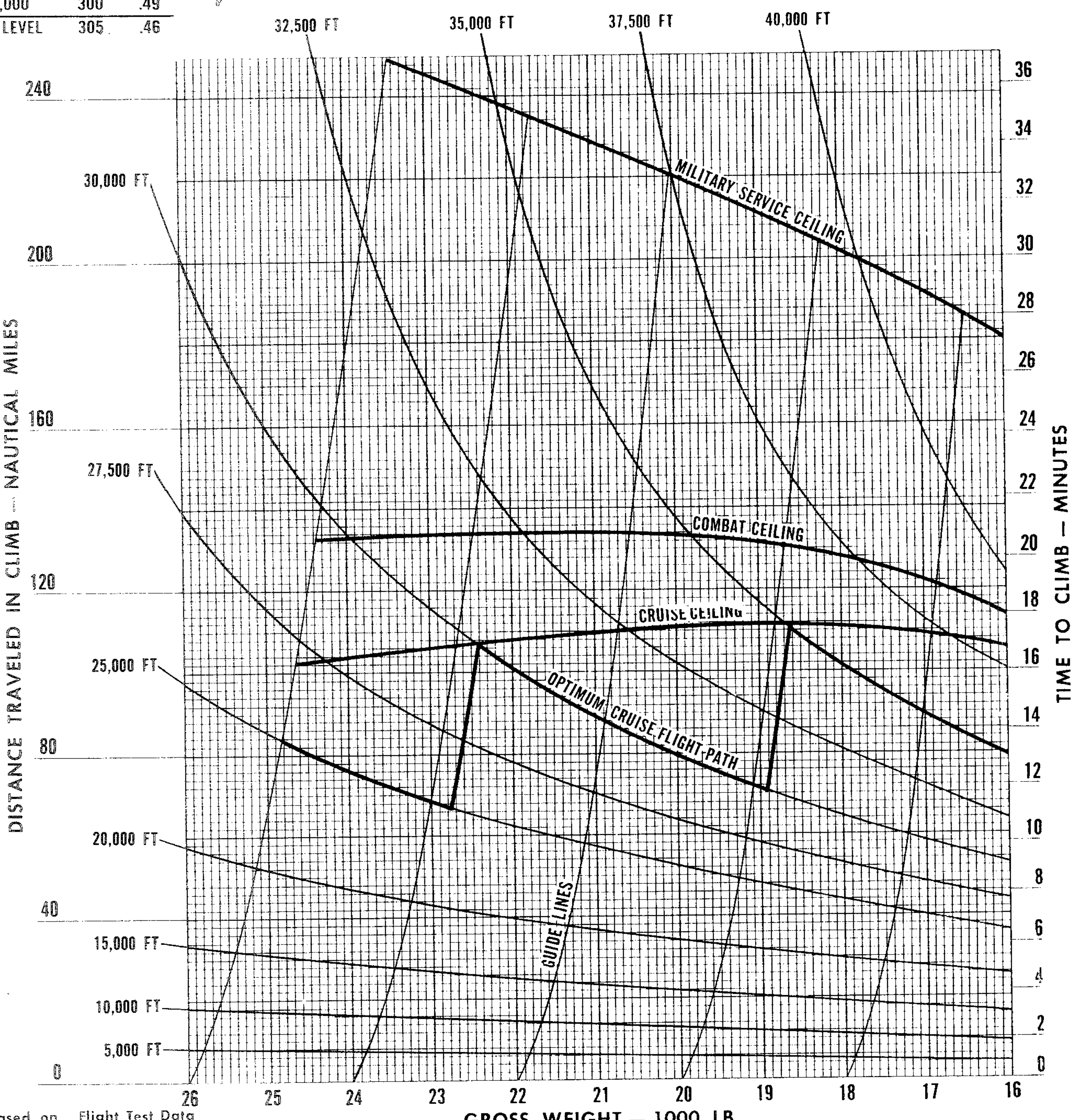
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
40,000	215	.72
35,000	240	.72
30,000	255	.68
25,000	265	.64
20,000	275	.60
15,000	280	.56
10,000	290	.53
5,000	300	.49
SEA LEVEL	305	.46

100% RPM

Military Power Climb

CONFIGURATION: CLEAN
TWO 230 GAL CLASS I TANKS
TWO 230 GAL CLASS II TANKS



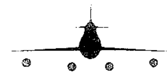
Based on Flight Test Data
Data as of 27 December 1955

Figure A3—10

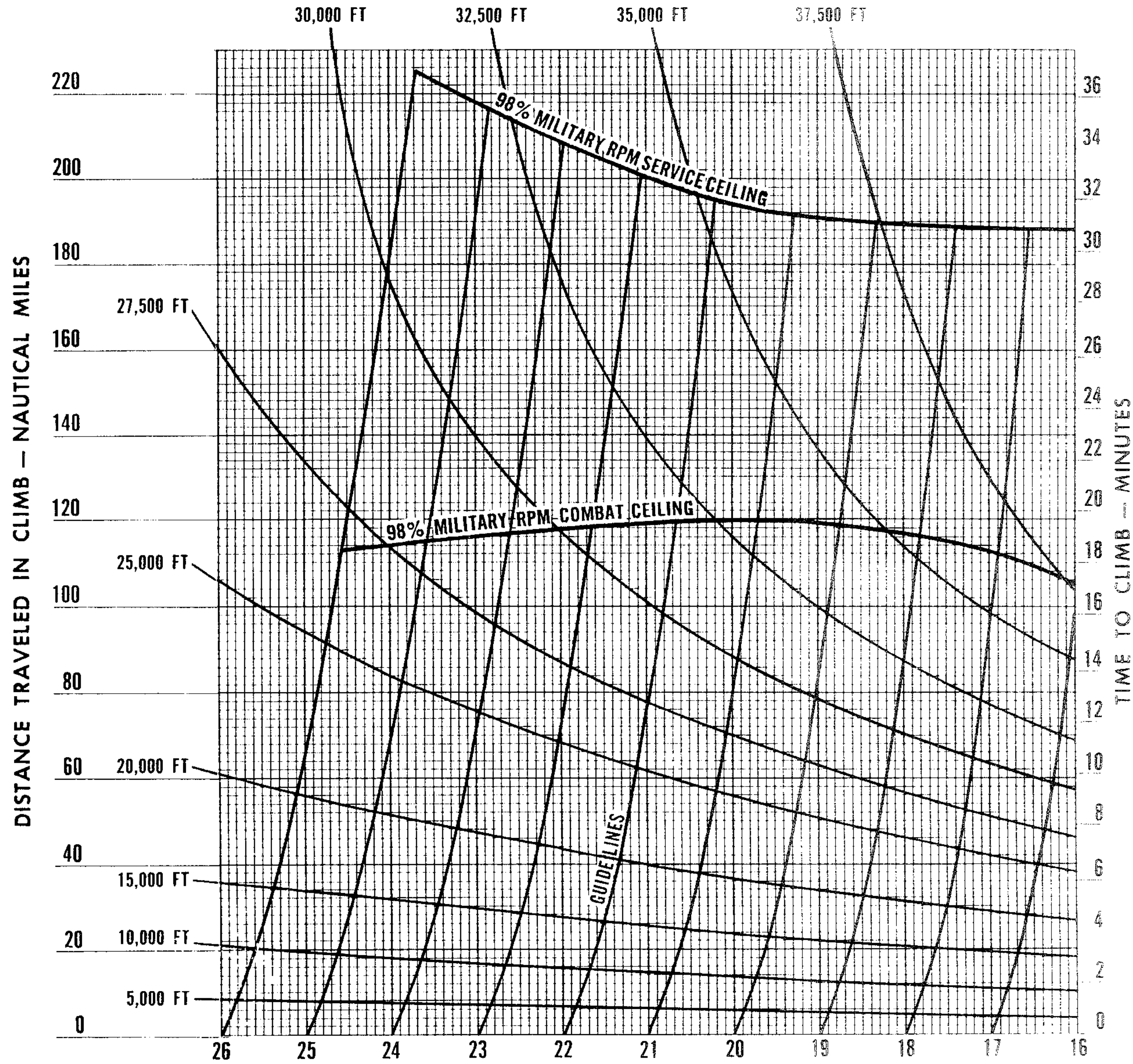
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

98% Military RPM CLIMB

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
35,000	230	.69
30,000	245	.65
25,000	255	.61
20,000	260	.57
15,000	270	.53
10,000	275	.50
5,000	280	.47
SEA LEVEL	285	.43



CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS + TWO 230 GAL CLASS II TANKS



Data as of 27 December 1955
Based on Flight Test Data

Figure A3—11

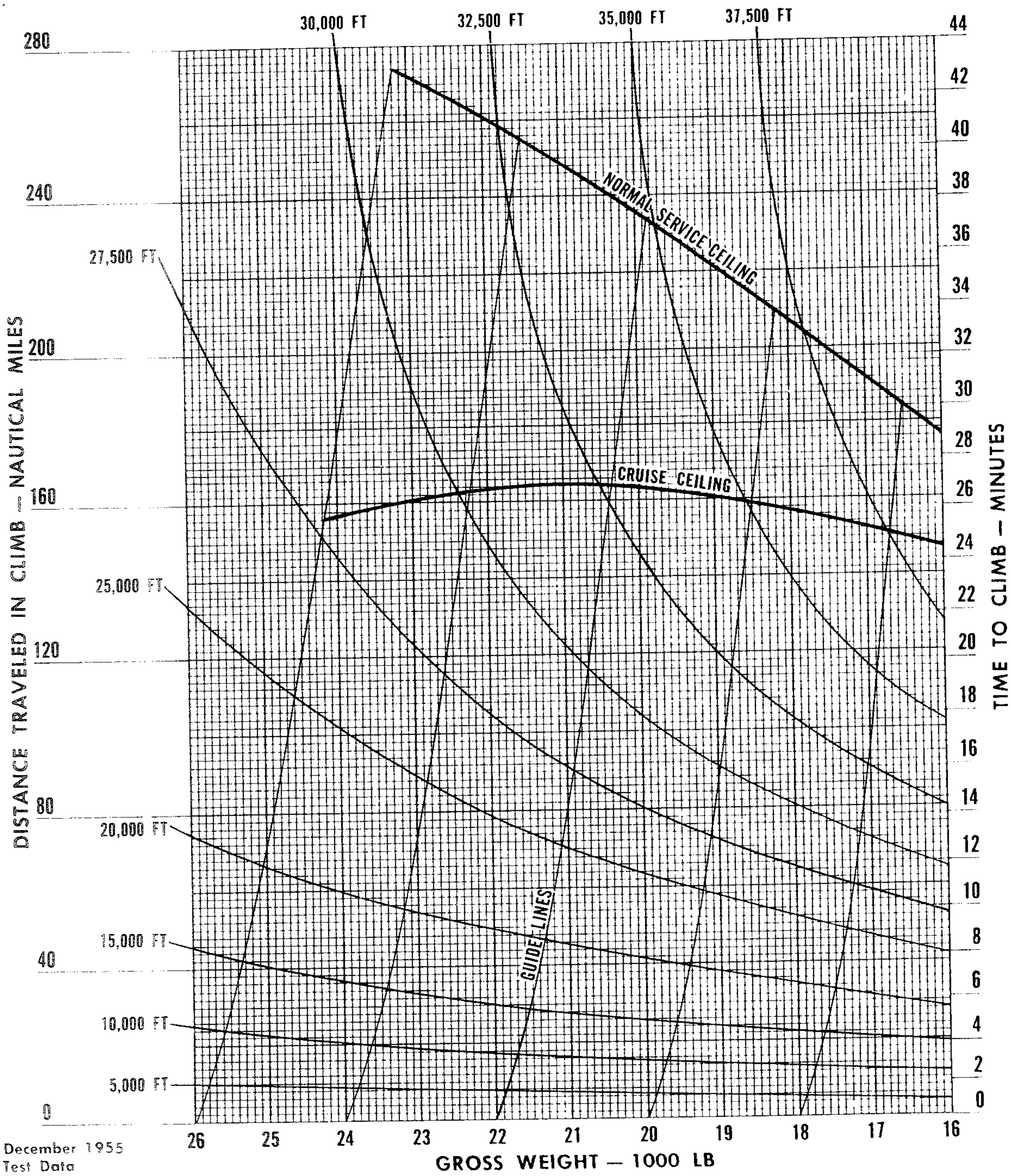
MODEL: RF-84F
ENGINE: J65-3
STANDARD DAY

Normal Power Climb

96% RPM

CONFIGURATION: CLEAN
TWO 230 GAL CLASS I TANKS
TWO 230 GAL CLASS II TANKS

ALTITUDE FEET	CLIMB SPEEDS CAS KNOTS	MACH NO
40,000	210	.70
35,000	230	.69
30,000	240	.65
25,000	250	.61
20,000	260	.57
15,000	265	.53
10,000	270	.49
5,000	275	.45
SEA LEVEL	280	.42



Data as of 27 December 1955
Based on Flight Test Data

Figure A3—12

PART 4 RANGE

TABLE OF CONTENTS

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.

NAUTICAL MILES PER POUND OF FUEL GRAPH.

Cruise data (no wind) throughout the speed range from maximum endurance to Military Thrust are shown on the Nautical Miles per Pound of Fuel graphs. Several weights for each configuration are given at altitudes of sea level, 15,000, 25,000, 30,000, 35,000, and 40,000 feet. Each graph includes specific range (nautical miles per pound), fuel flow, and RPM. Also included are curves of recommended cruise Mach number, maximum endurance, and Normal and Military Thrust. Specific range is plotted against Mach number.

Cruising range is calculated from the Nautical Miles per Pound of Fuel graphs on a fuel increment basis. The smaller the increment of fuel used in the calculation, the greater the accuracy of the range; therefore, if a high degree of accuracy is desired, several increments should be used.

To obtain the cruising range for an increment of fuel, use the following steps: (If several increments of fuel are used, repeat the steps shown for each increment. The sum of the individual ranges is the total cruising range.)

a. Select the proper graph for the aircraft configuration and altitude.

b. Determine the average weight of aircraft for the increment of fuel being considered.

c. Enter the graph at this average weight and the desired Mach number, or desired RPM, to obtain specific range (nautical miles per 100 pounds of fuel).

d. The specific range multiplied by the amount of fuel (pounds ÷ 1000) equals cruising range.

e. Determine the approximate fuel flow and RPM at the Mach number and average weight. When there is a wind to be considered, multiply the specific range found in step c. by the range factor (ground speed divided by true airspeed) to obtain the specific range for wind. Proceed with steps d. and e. to complete the problem.

MISSION PROFILE

MODEL: RF-84F

STANDARD DAY

CLEAN AIRPLANE

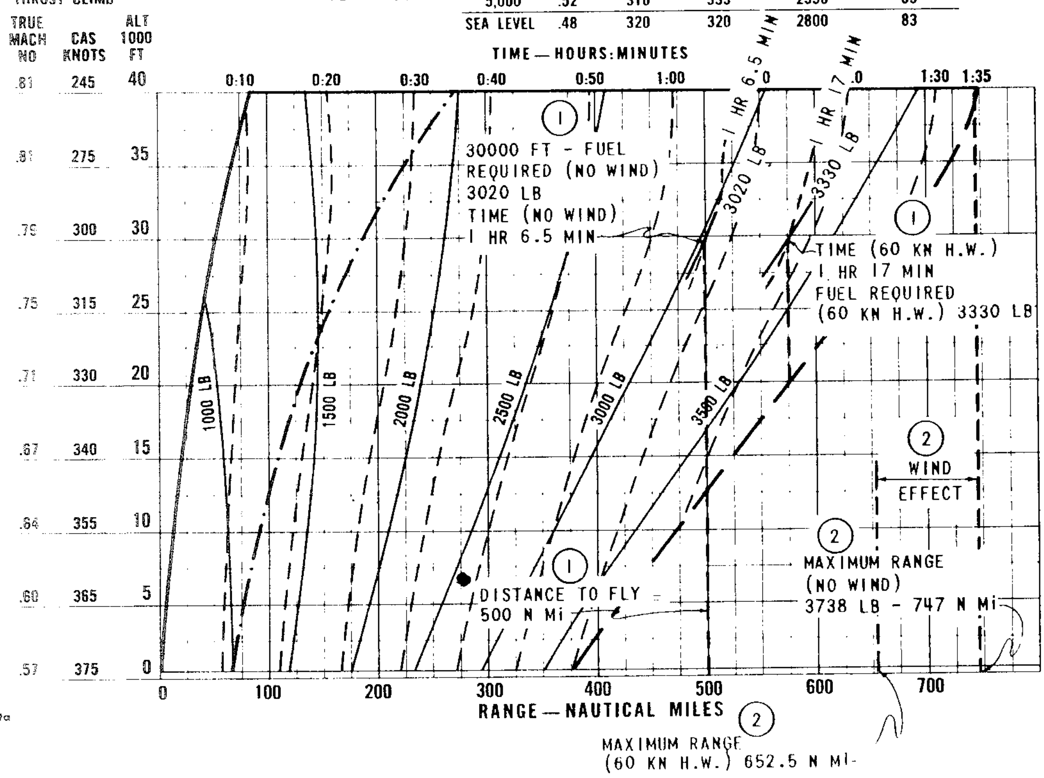
CRUISE SETTINGS 2

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.83	255	480	1800	95
35,000	.82	280	470	1850	90
30,000	.76	290	450	1950	88
25,000	.71	295	425	2050	87
20,000	.65	300	400	2100	85
15,000	.60	305	375	2250	85
10,000	.55	305	350	2400	84
5,000	.52	310	335	2550	83
SEA LEVEL	.48	320	320	2800	83

CRUISE SETTINGS 1

GROSS WEIGHT 15,112 — 18,850 LB

MILITARY THRUST CLIMB



Sample Chart

REMARKS

1. Fuel allowance for start, taxi and take-off (465 lb) included.
2. No allowance or reserve for loiter, descent or landing.
3. Use military thrust for climb. (see military thrust climb chart for detailed information)
4. Cruise at recommended Mach No.

CLIMB SPEED SCHEDULE

LEGEND

- ———— Optimum cruise flight path
- — - — Line of best range for constant altitude
- — — — Time (start, taxi and take-off not included)
- ———— Fuel consumed
- —— —— 3738 lb fuel consumed — Zero fuel remaining

Data as of 27 Dec 1955 — Based on Flight Test Data

DESCRIPTION.

These charts give the relationship of time, fuel, distance, and altitude to maximum range for no-wind conditions. This relationship is based on a mission sequence of take-off, military thrust climb, and maximum range cruise. The fuel curves include a 465 pound allowance for start, taxi, and take-off, and the fuel used in climb to each altitude, as well as the fuel required for maximum range cruise. The time lines include the time required to climb to cruise altitude but do not include the time to start, taxi, or take-off.

The line labeled "Initial Climb Path" shows the distance traveled during the Military Thrust climb from sea level to cruising altitude, using the climb speed schedule tabulated at the left of the chart.

As an aid to preflight planning, a line of best range for constant-altitude flight appears on the chart. This curve is not a flight path, but a plot of best cruise altitude against distance. For distances greater than those covered by the curve, use step climb procedure for maximum range.

A cruise table gives recommended Mach numbers and approximate operating conditions for cruise at constant altitude. (Cruise-at-constant-altitude data is given for each 5000 feet.)

USE.

The charts may be entered with one or more of the four range factors: time, fuel, distance, and altitude. By entering the chart with the known factors, the others may readily be determined. This is for a no-wind condition. To determine wind effect upon time, fuel, and distance, compute the average true airspeed (TAS)—distance ÷ time, no wind—and apply wind to TAS to obtain ground speed (G.S.). Then compute the time with wind (distance÷G.S.). Re-enter the profile at the cruising altitude and the computed time with wind to determine the fuel required with wind.

Sample Problem 1.

Using the example shown, find the fuel required, time, necessary speed, and power setting to cruise 500 nautical miles at 30,000 feet with a head wind of 60 knots in the clean configuration.

a. Enter at 500 nautical miles and 30,000 feet to obtain fuel required (no wind) — 3020 LB

b. Time (no wind) — 1 HR 6.5 MIN (1.11 HR)

c. Average TAS (500 ÷ 1.11) — 450 knots

d. Apply wind to obtain G.S. (450-60) — 390 knots

e. Calculate time with 60-knot wind (500 ÷ 390) — 1.28 HR (1 HR 17 MIN)

f. Re-enter at cruise altitude at the time with wind. Fuel required with wind — 3330 LB

g. Determine cruise speed from table — 0.76 Mach NO.

h. Determine cruise power setting from table — 88% RPM

Sample Problem 2.

Determine the maximum distance flyable, using clean aircraft with 3738 pounds of fuel and a 60-knot head wind.

a. Enter at 3738 pounds of fuel and obtain maximum air distance as step climb (no wind) — 747 (nautical miles)

b. Time (no wind) — 1 HR 35 MIN (1.58 hr)

c. Calculate average TAS (747 ÷ 1.58) — 473 knots

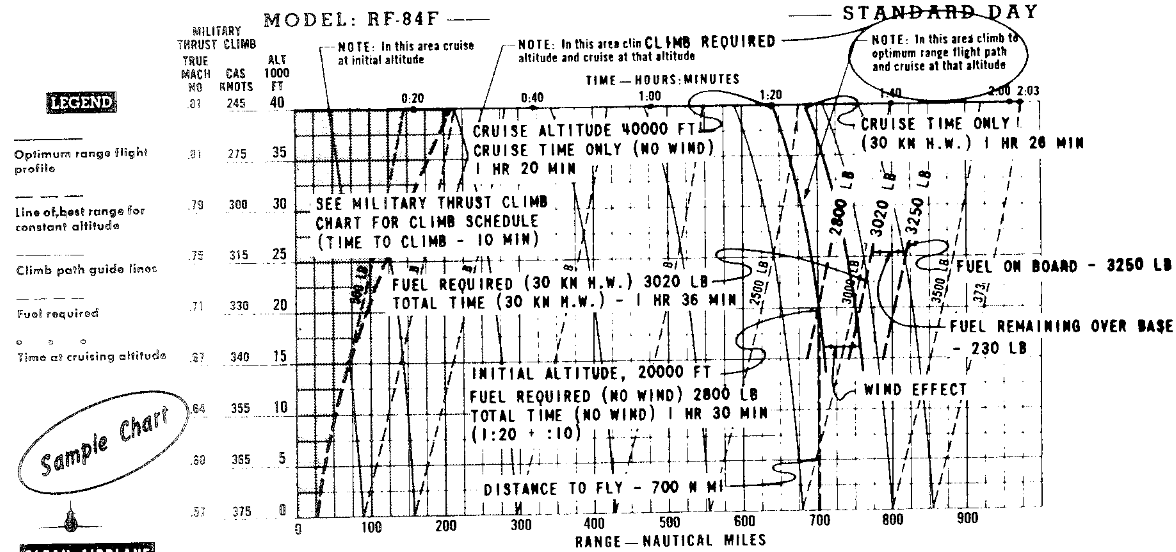
d. Apply wind to obtain G.S. (473-60) — 413 knots

e. Calculate distance with wind (1.58 × 413) — 652.5 nautical miles

f. Determine step-climb speed from table — 0.83 Mach NO.

g. Determine cruise power setting from table — 95% RPM

OPTIMUM RETURN PROFILE



CRUISE SETTINGS

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.83	255	480	1800	95
35,000	.82	280	470	1850	90
30,000	.76	290	450	1950	88
25,000	.71	295	425	2050	87
20,000	.65	300	400	2100	85
15,000	.60	305	375	2250	85
10,000	.55	305	350	2400	84
5,000	.52	310	335	2550	83
SEA LEVEL	.48	320	320	2800	83

GROSS WEIGHT
15,112 — 18,850 LB

REMARKS

1. Fuel required at any point below the line of best range for constant altitude flight or below the optimum range flight profile includes military power climb to flight altitude.
2. No allowance made for loiter, descent or landing.
3. Best cruise condition determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach No.

Data as of 27 Dec 1955 — Based on Flight Test Data

DESCRIPTION.

These profiles show the minimum fuel required for maximum distance (no wind) based on an optimum flight path from any starting point within the range of the aircraft configuration. The flight path required is indicated by the different shaded areas and the notes relative to them. The fuel curves are based on a military thrust climb to, and recommended cruise at, the optimum altitude. The military thrust climb speed schedule and recommended cruise settings are tabulated on each chart. No reserve for descent and landing has been included. The time shown at the optimum altitude is cruise time only; it does not include the time required for the climb to optimum altitude or any allowing for descent, loiter, and landing.

USE

The chart may be entered at the initial altitude with either the fuel on board, (to determine the distance available) or with the distance to be flown (to determine the fuel required). The shaded area in which the initial point falls establishes the cruising procedure to be used, as stated in the note relative to the area. The time required to fly the distance is the time at cruise altitude (obtain from profile), plus the time required to climb (obtained from graphical military thrust climb chart).

The effect of wind must be applied to obtain the actual fuel and time to fly the distance. A close approximation can be obtained by considering the head or tail wind for the time it requires to complete the flight (neglecting the difference in wind at the lower altitudes, since comparatively little time is spent during the climb phase).

From the example shown, determine the fuel and time required to return to a base 700 nautical miles away. The aircraft is at 20,000 feet with 3250 pounds of fuel on board in the clean configuration (gross weight = 18,362 pounds). A 30-knot head wind is assumed.

a. Enter profile at 700 nautical miles and 20,000 feet to establish starting point. Fuel required (no wind) 2800 pounds. b. In this area, note that a climb is required and a step-climb procedure followed. c. By following the climb guide lines, the cruise altitude is 40,000 feet. d. Cruise time (no wind) 1 HR 20 MIN. e. From the military thrust climb chart for clean configuration — time to climb = 10 MIN. f. Total time (no wind) (d + e) 1 HR 30 MIN. g. Average TAS (distance ÷ total time) 467 knots. h. Average ground speed (TAS − head wind) 437 knots. i. Total time (wind) (distance ÷ AVG GS) 1 hr 36 MIN. j. Cruise time (wind) (i − e) 1 HR 36 MIN. k. Using the cruise time (j) on the profile, backtrack down the climb path from the line of best range to 20,000 feet to obtain fuel required (wind), 3020 pounds. l. Fuel remaining over base at altitude (3250 − 3020) 230 pounds. m. Use the flight path originally determined at no wind.

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CLEAN AIRPLANE

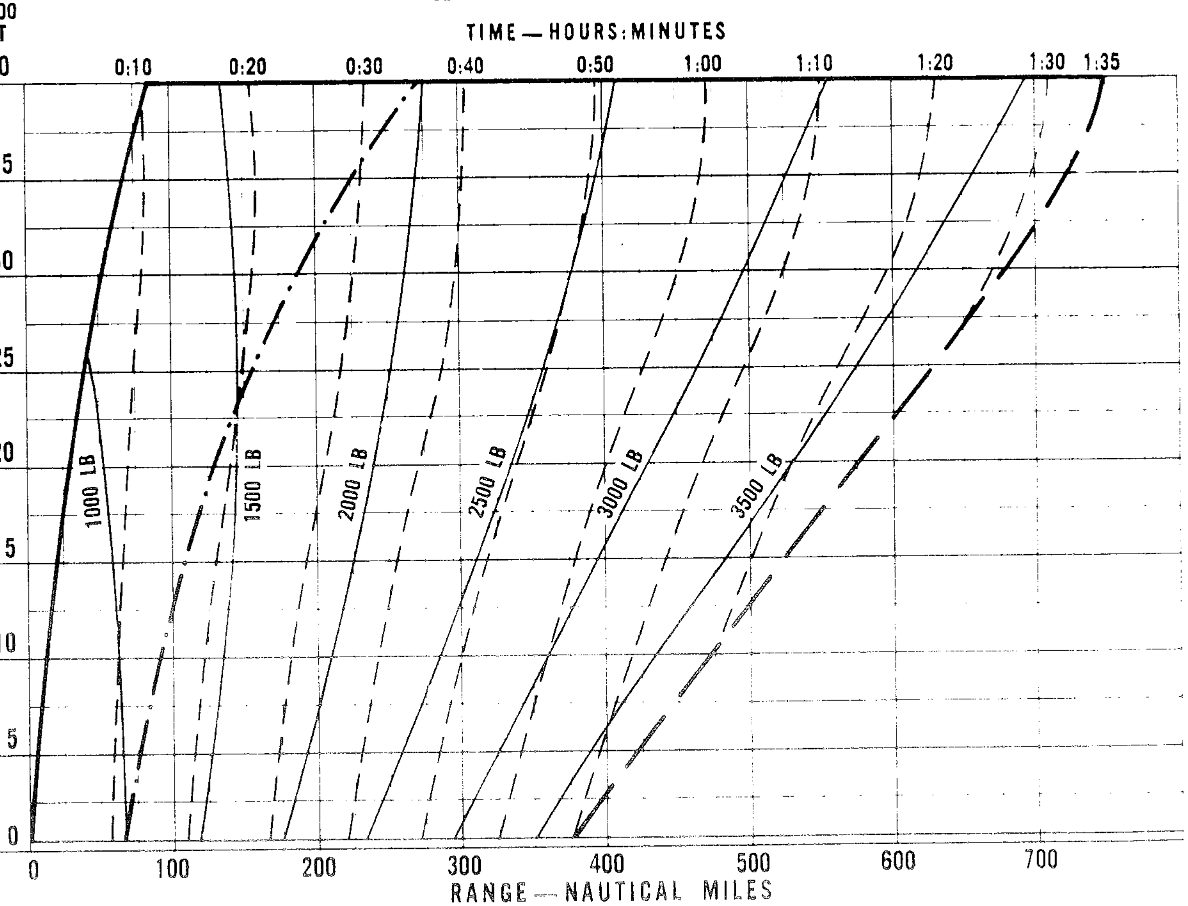
Mission Profile

GROSS WEIGHT
15,112 — 18,850 LB

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.83	255	480	1800	95
35,000	.82	280	470	1850	90
30,000	.76	290	450	1950	88
25,000	.71	295	425	2050	87
20,000	.65	300	400	2100	85
15,000	.60	305	375	2250	85
10,000	.55	305	350	2400	84
5,000	.52	310	335	2550	83
SEA LEVEL	.48	320	320	2800	83

MILITARY THRUST CLIMB

TRUE MACH NO	CAS KNOTS	ALT 1000 FT
.81	245	40
.81	275	35
.79	300	30
.75	315	25
.71	330	20
.67	340	15
.64	355	10
.60	365	5
.57	375	0



REMARKS

1. Fuel allowance for start, taxi and take-off (465 lb) included.
2. No allowance or reserve for loiter, descent or landing.
3. Use military thrust for climb. (see military thrust climb chart for detailed information)
4. Cruise at recommended Mach No.

LEGEND

- Optimum cruise flight path
- Line of best range for constant altitude
- Time (start, taxi and take-off not included)
- Fuel consumed
- 3738 lb fuel consumed — Zero fuel remaining

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A4—1

MODEL: RF-84F ENGINE: J65-3 — STANDARD DAY

LEGEND

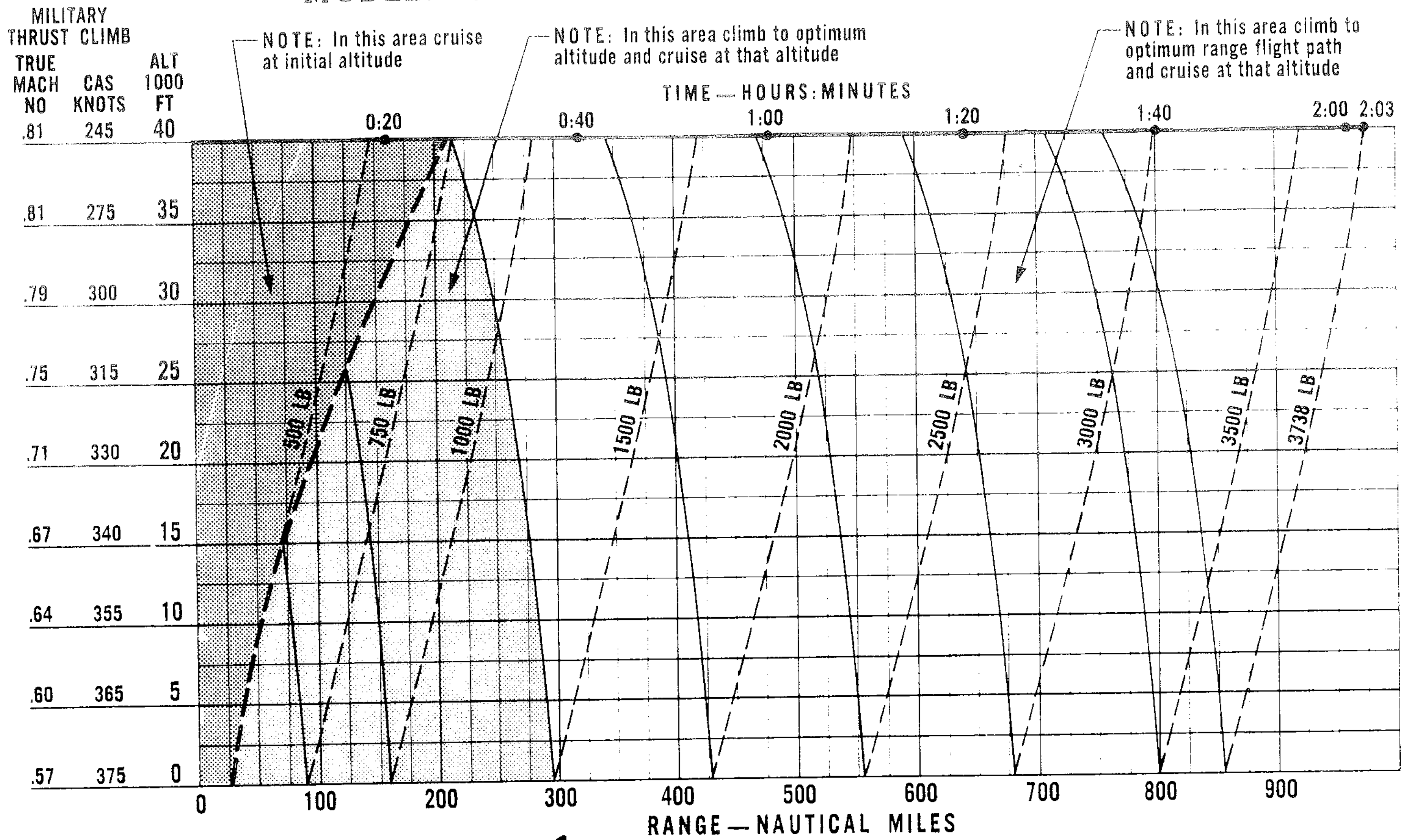
Optimum range flight profile

Line of best range for constant altitude

Climb path guide lines

Fuel required

Time at cruising altitude



CLEAN AIRPLANE

Optimum Return Profile

GROSS WEIGHT
15,112 — 18,850 LB

REMARKS

1. Fuel required at any point below the line of best range for constant altitude flight or below the optimum range flight profile includes military power climb to flight altitude.
2. No allowance made for loiter, descent or landing.
3. Best cruise condition determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach No.

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.83	255	480	1800	95
35,000	.82	280	470	1850	90
30,000	.76	290	450	1950	88
25,000	.71	295	425	2050	87
20,000	.65	300	400	2100	85
15,000	.60	305	375	2250	85
10,000	.55	305	350	2400	84
5,000	.52	310	335	2550	83
SEA LEVEL	.48	320	320	2800	83

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A4—2

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

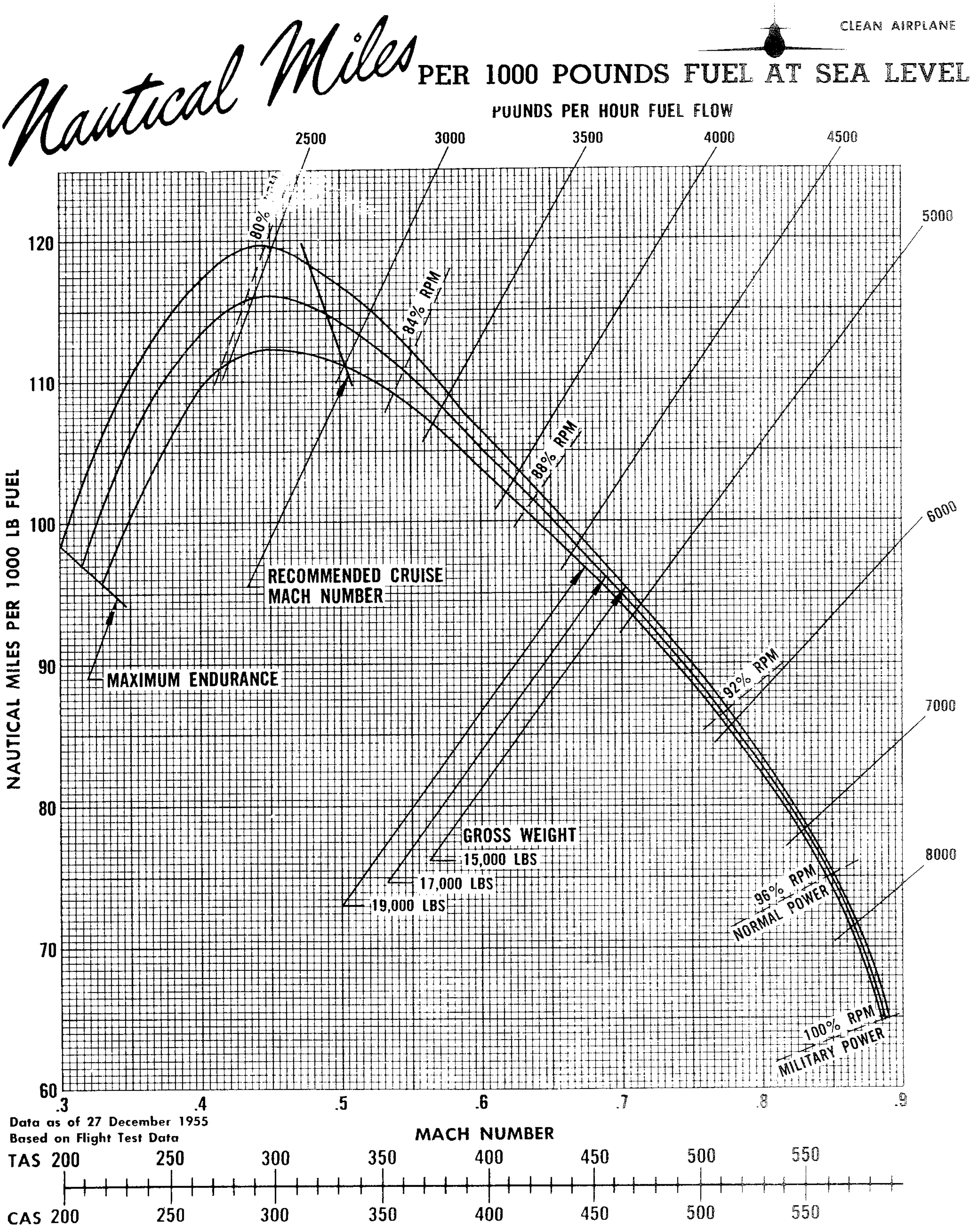
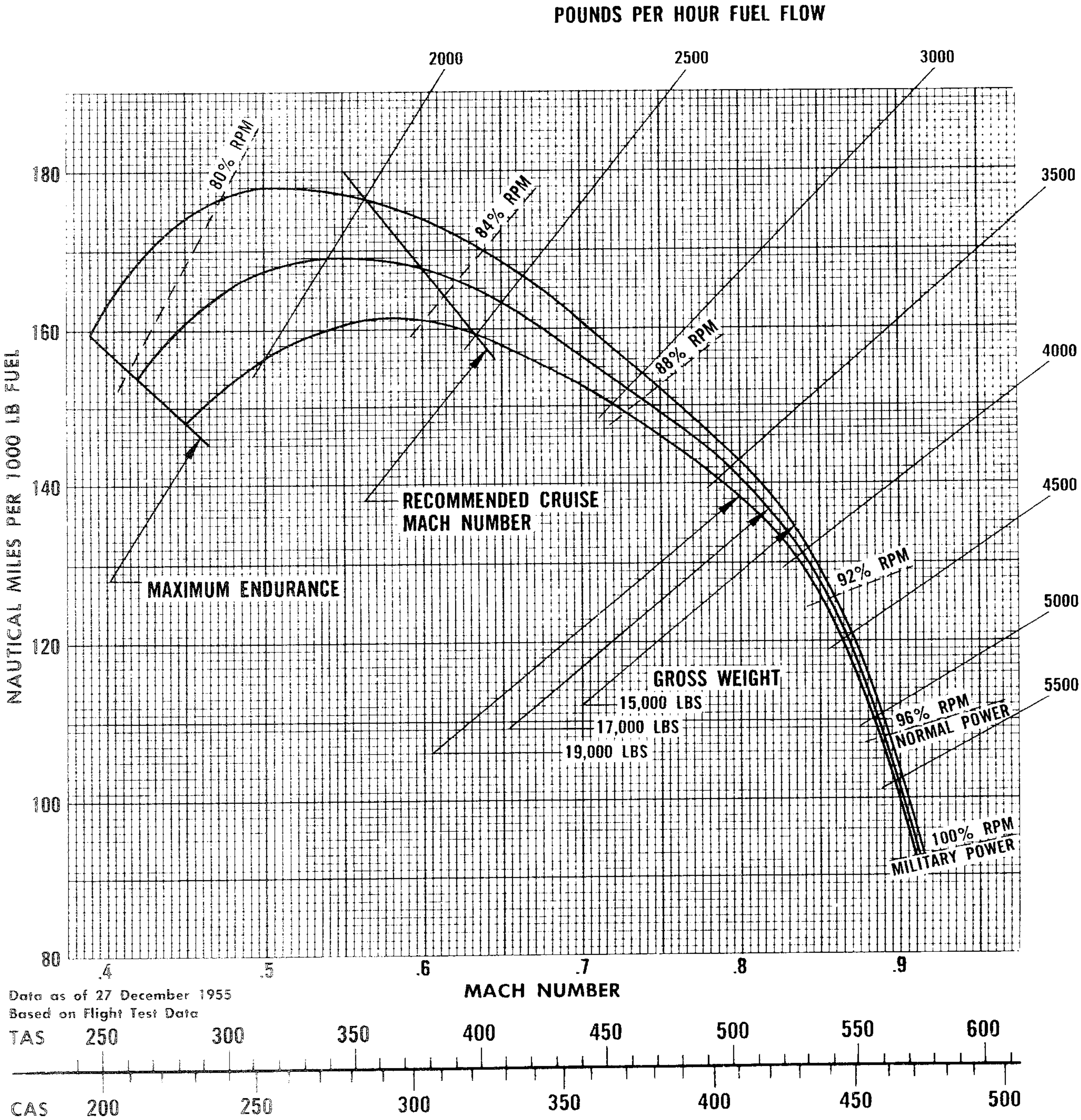


Figure A4—3 (Sheet 1 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CLEAN AIRPLANE

Nautical Miles PER 1000 POUNDS FUEL AT 15,000 FEET



Data as of 27 December 1955
Based on Flight Test Data

Figure A4—3 (Sheet 2 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 25,000 FEET

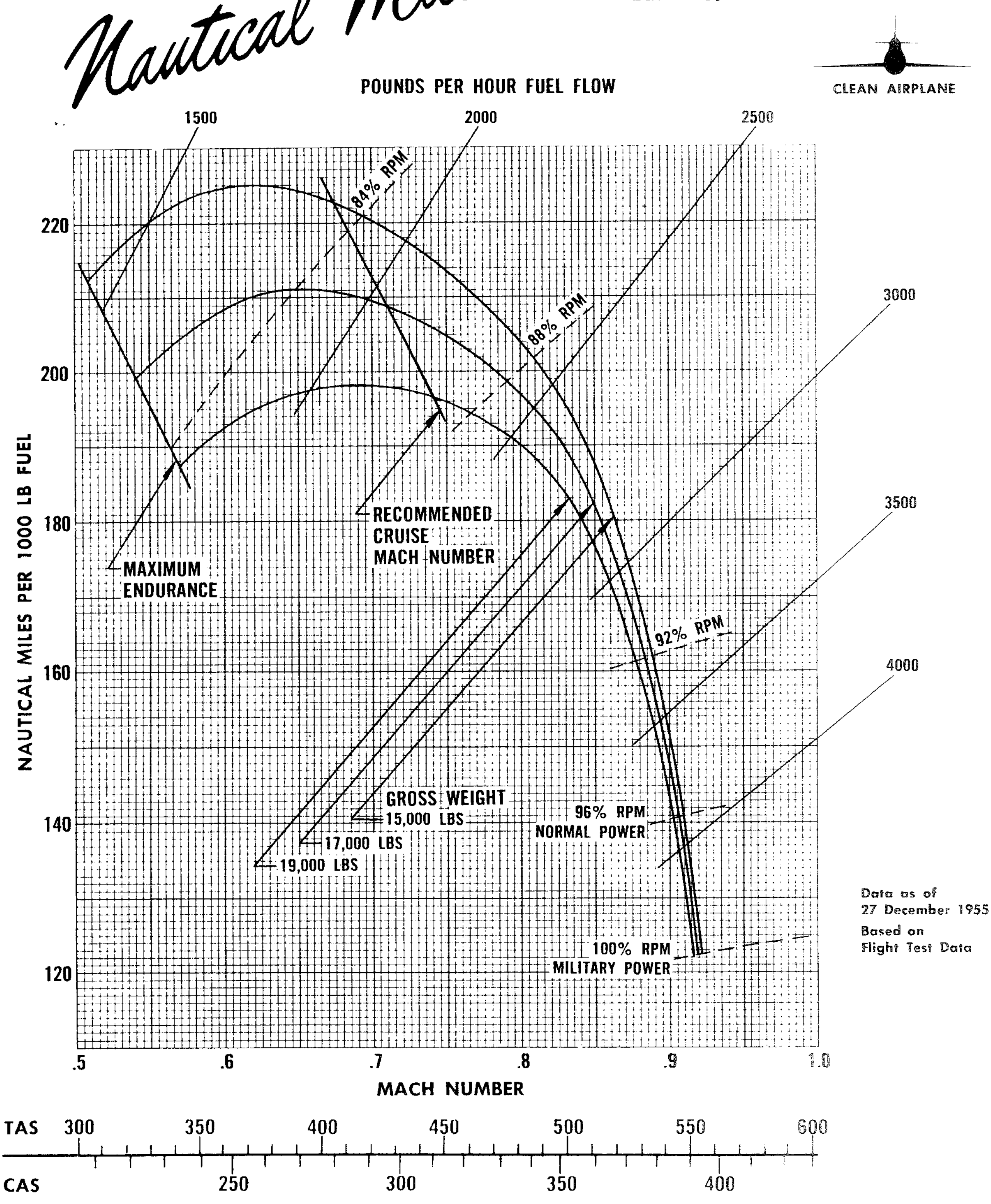
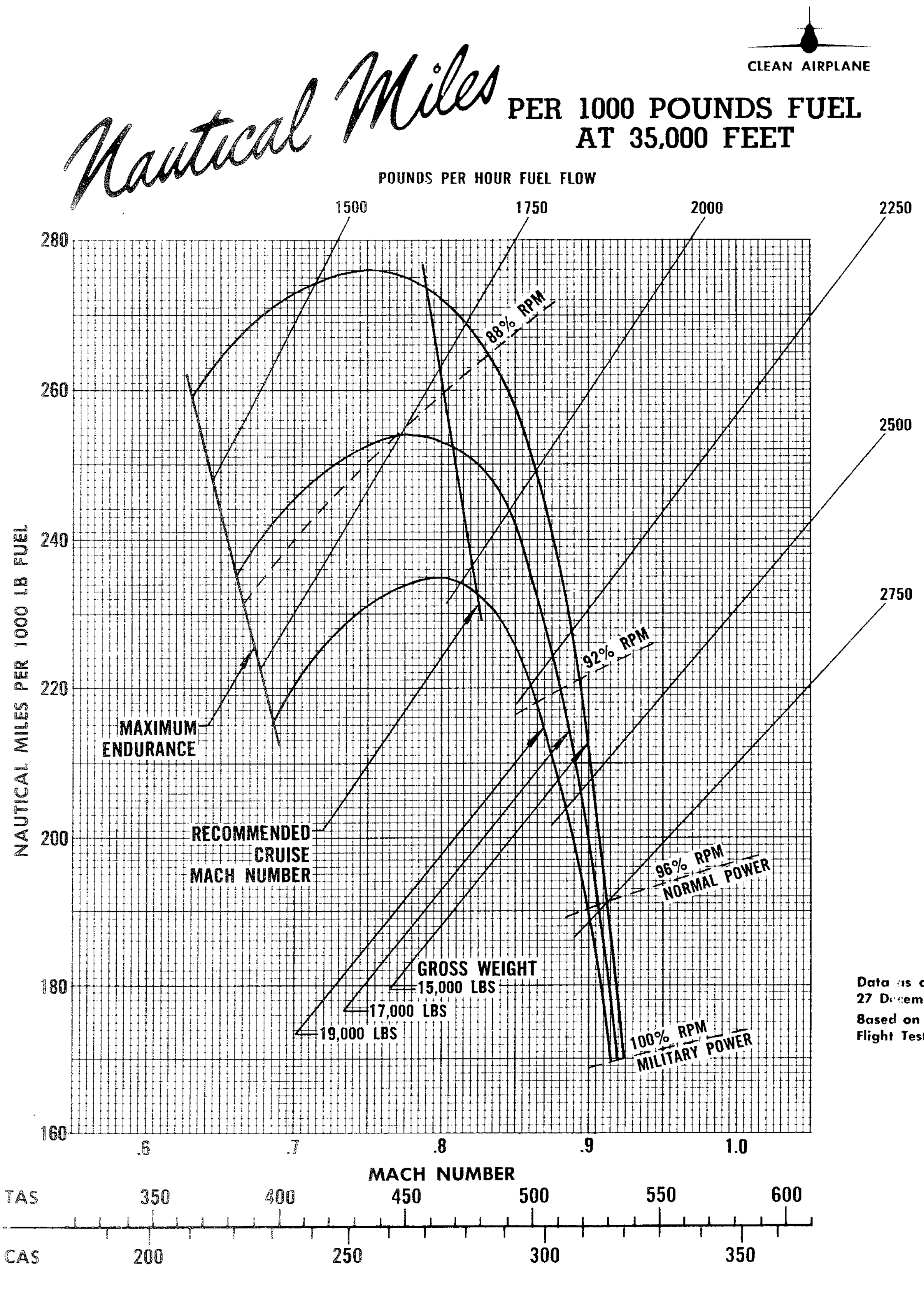


Figure A4—3 (Sheet 3 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 35,000 FEET

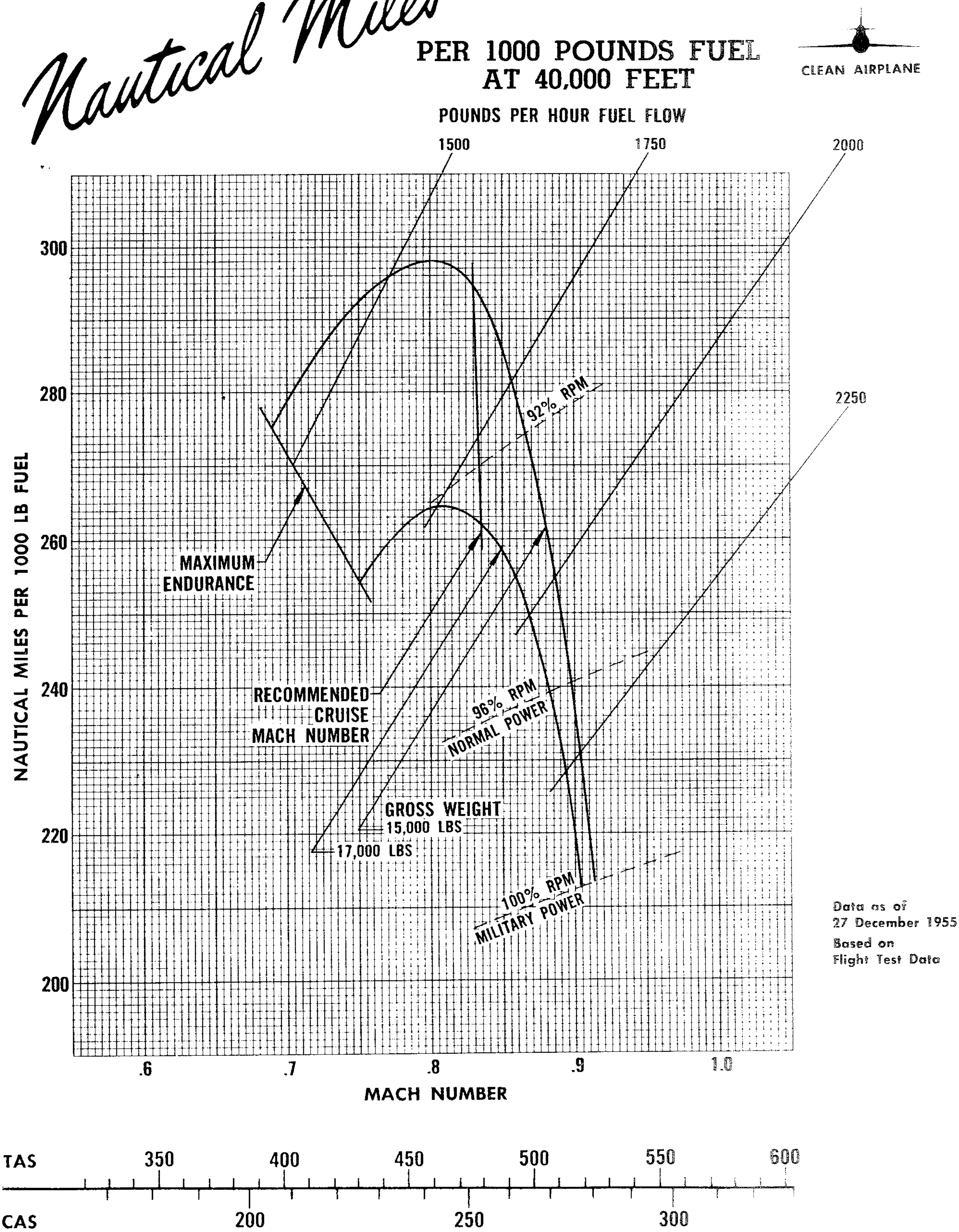


Data as of
27 December 1955
Based on
Flight Test Data

Figure A4—3 (Sheet 4 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 40,000 FEET



Data as of 27 December 1955
Based on Flight Test Data

Figure A4—3 (Sheet 5 of 5)

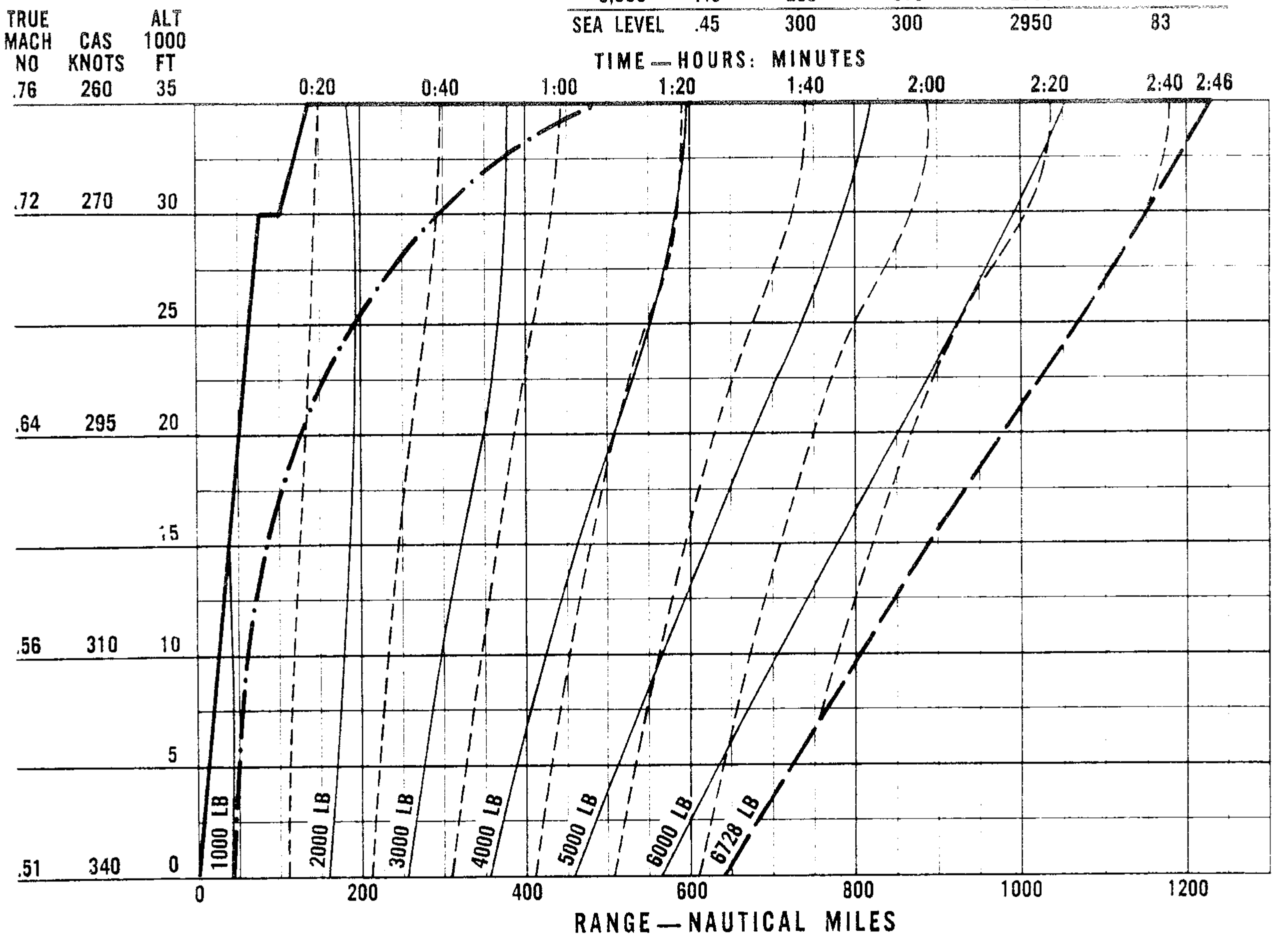
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

Mission Profile

GROSS WEIGHT
15,682 — 22,410 LB

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.79	240	455	1800	95
35,000	.77	260	445	2000	92
30,000	.72	270	425	2100	90
25,000	.68	280	410	2300	89
20,000	.61	285	375	2300	87
15,000	.57	290	355	2450	86
10,000	.53	295	335	2500	85
5,000	.49	295	315	2700	84
SEA LEVEL	.45	300	300	2950	83



REMARKS

1. Fuel allowance for start, taxi and take-off (465 lb) included.
2. No allowance or reserve for loiter, descent or landing.
3. Use military thrust for climb. (see military thrust climb chart for detailed information)
4. Cruise at recommended Mach No.

LEGEND

- ——— Optimum cruise flight path
- — · — · Line of best range for constant altitude
- - - - - Time (start, taxi and take-off not included)
- ——— Fuel consumed
- — — — 6728 lb fuel consumed — Zero fuel remaining

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A4—4

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.79	240	455	1800	95
35,000	.77	260	445	2000	92
30,000	.72	270	425	2100	90
25,000	.68	280	410	2300	89
20,000	.61	285	375	2300	87
15,000	.57	290	355	2450	86
10,000	.53	295	335	2500	85
5,000	.49	295	315	2700	84
SEA LEVEL	.45	300	300	2950	83

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

REMARKS

1. Fuel required at any point below the line of best range for constant altitude flight or below the optimum range flight profile includes military power climb to flight altitude.
2. No allowance made for loiter, descent or landing.
3. Best cruise condition determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach No.

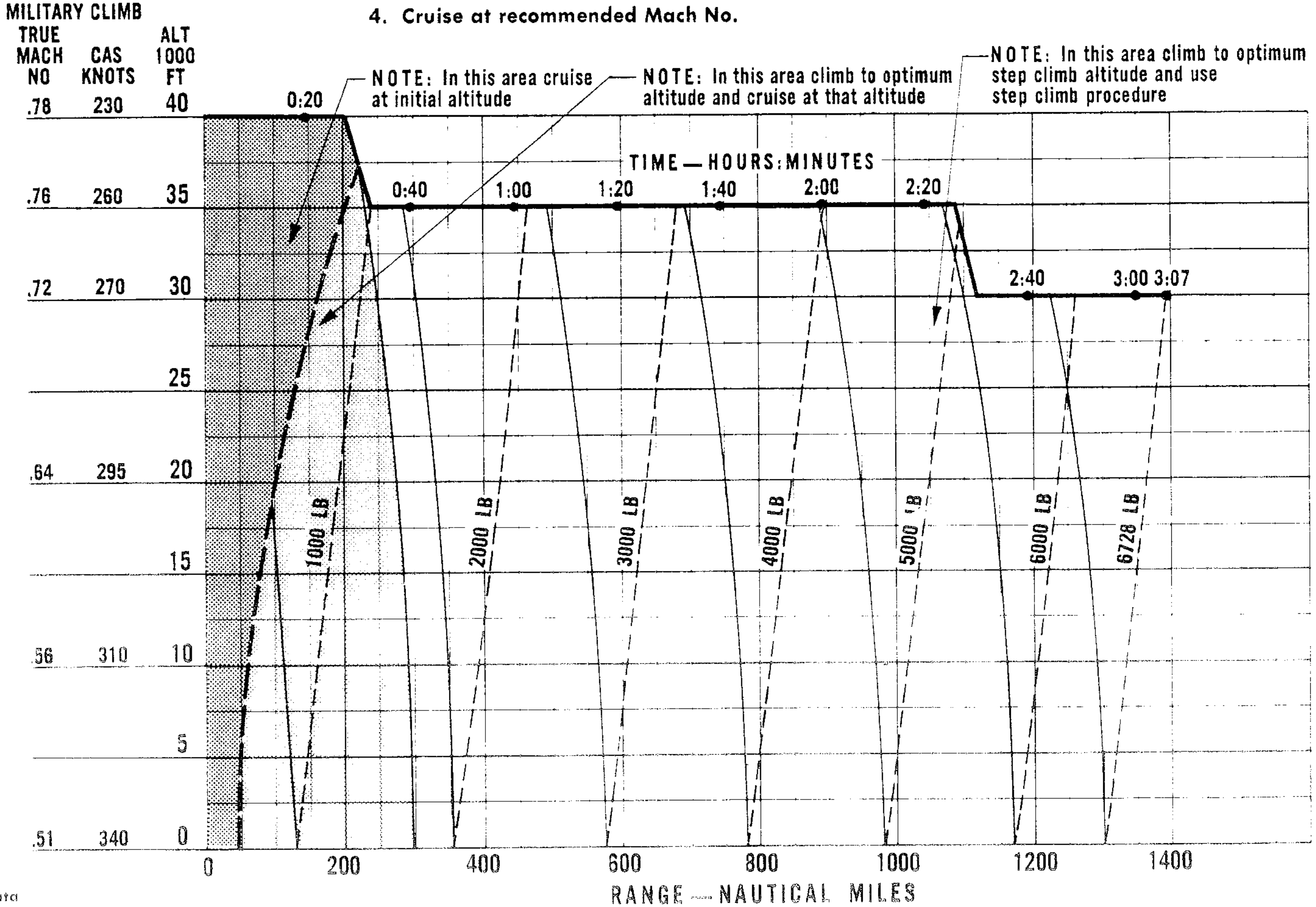
LEGEND

- ———— Optimum range flight profile
- — — — Line of best range for constant altitude
- ———— Climb path guide lines
- - - - - - Fuel required
- • • • Time at cruising altitude

Optimum Return Profile

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

GROSS WEIGHT
15,682 — 22,410 LB



Data as of 27 Dec 1955 — Based on Flight Test Data

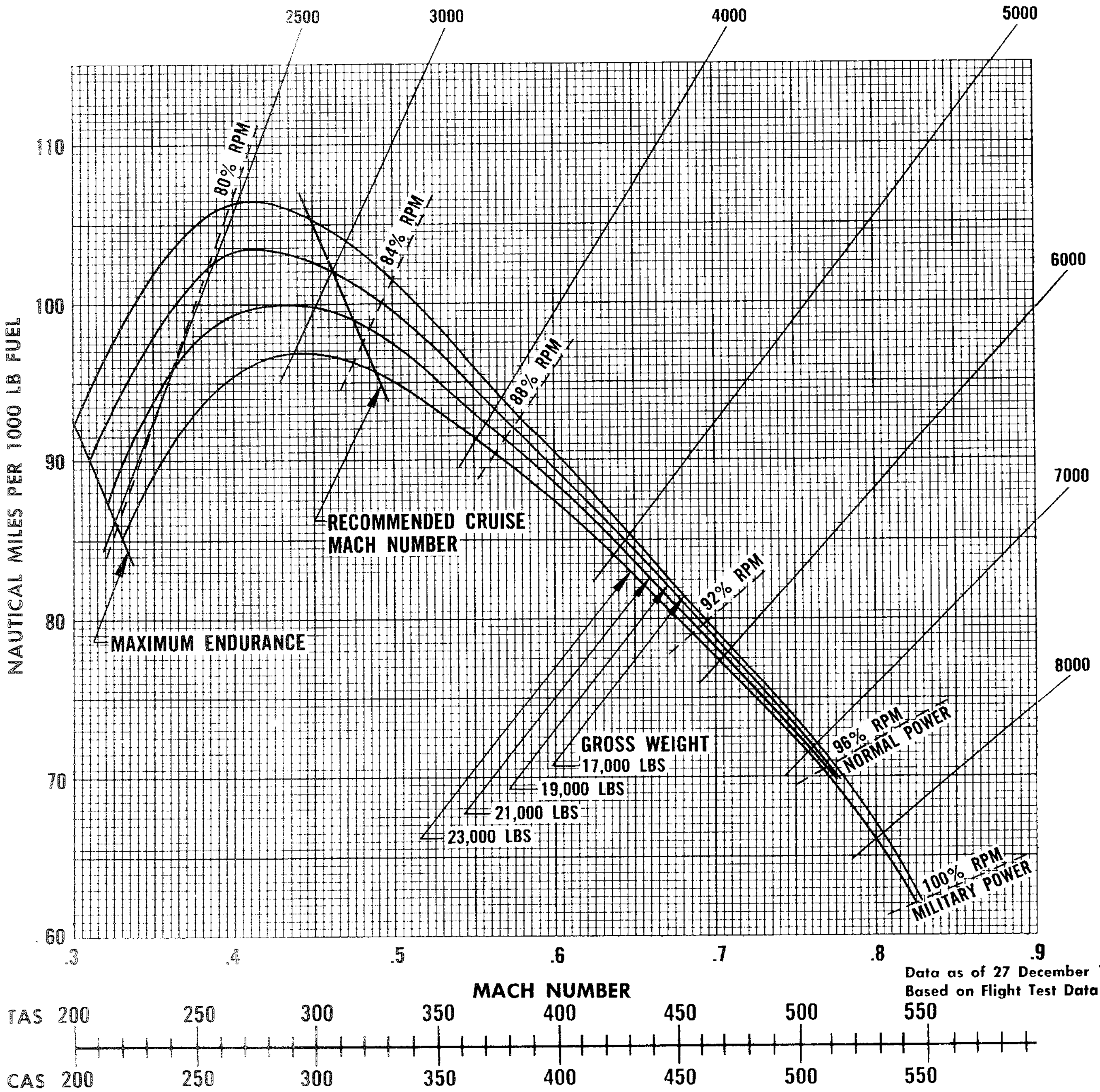
Figure A4—5

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles

CONFIGURATION
CLEAN + TWO 230 GAL CLASS I TANKS

PER 1000 POUNDS FUEL AT SEA LEVEL



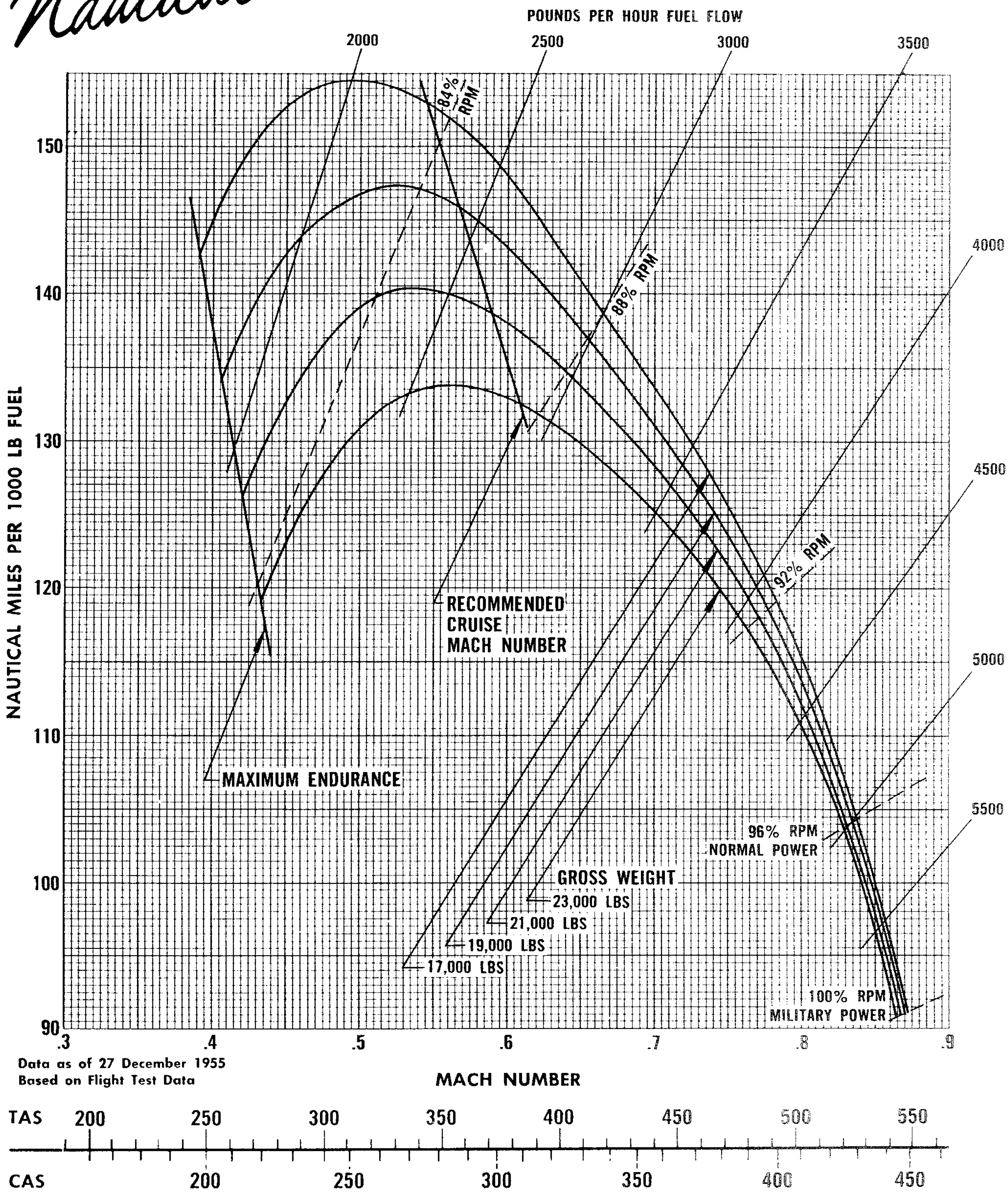
Data as of 27 December 1955
Based on Flight Test Data

Figure A4—6 (Sheet 1 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION
CLEAN + TWO 230
GAL CLASS I TANKS

Nautical Miles PER 1000 POUNDS FUEL AT 15,000 FEET



Data as of 27 December 1955
Based on Flight Test Data

Figure A4-6 (Sheet 2 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 25,000 FEET

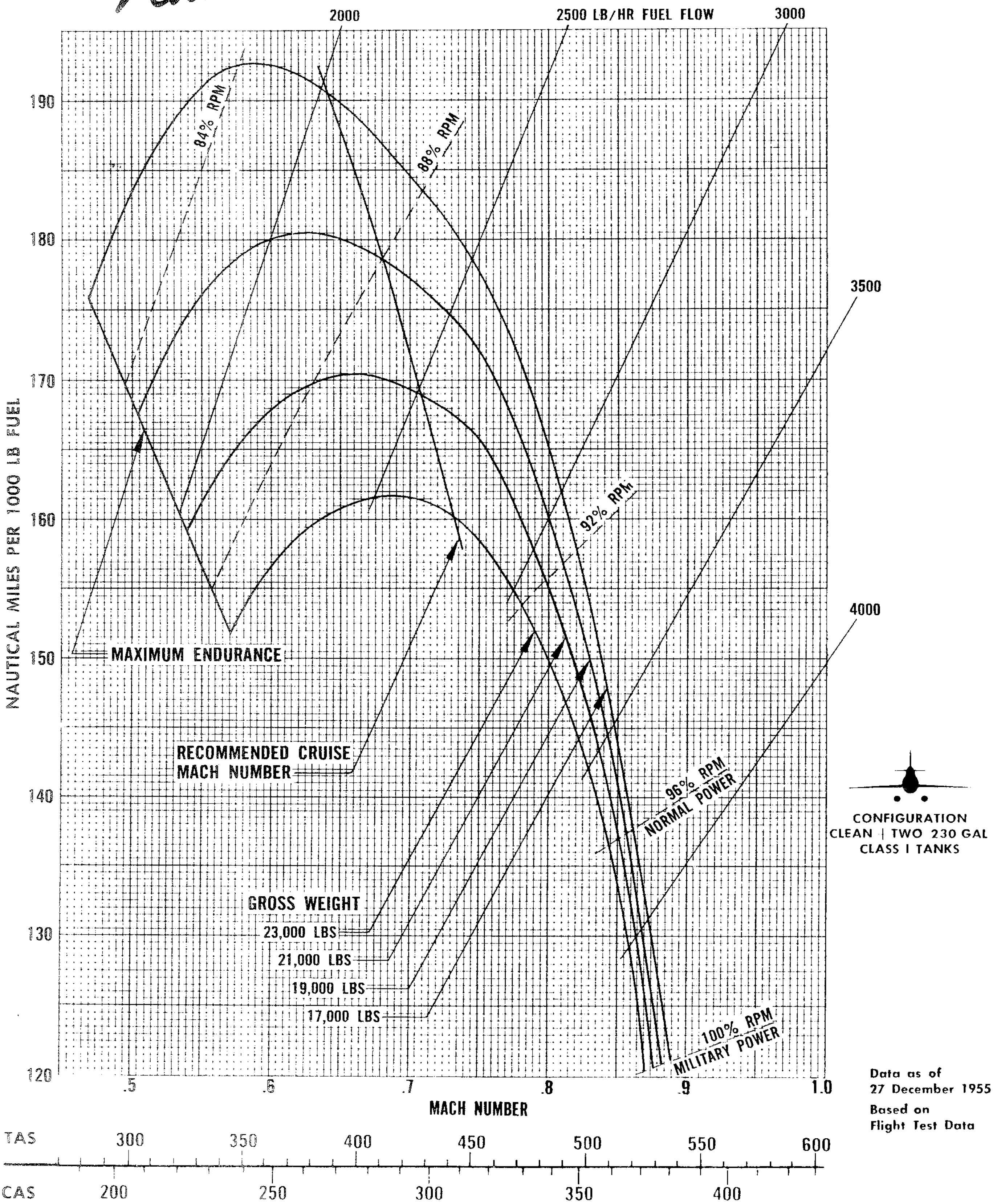


Figure A4—6 (Sheet 3 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 30,000 FEET

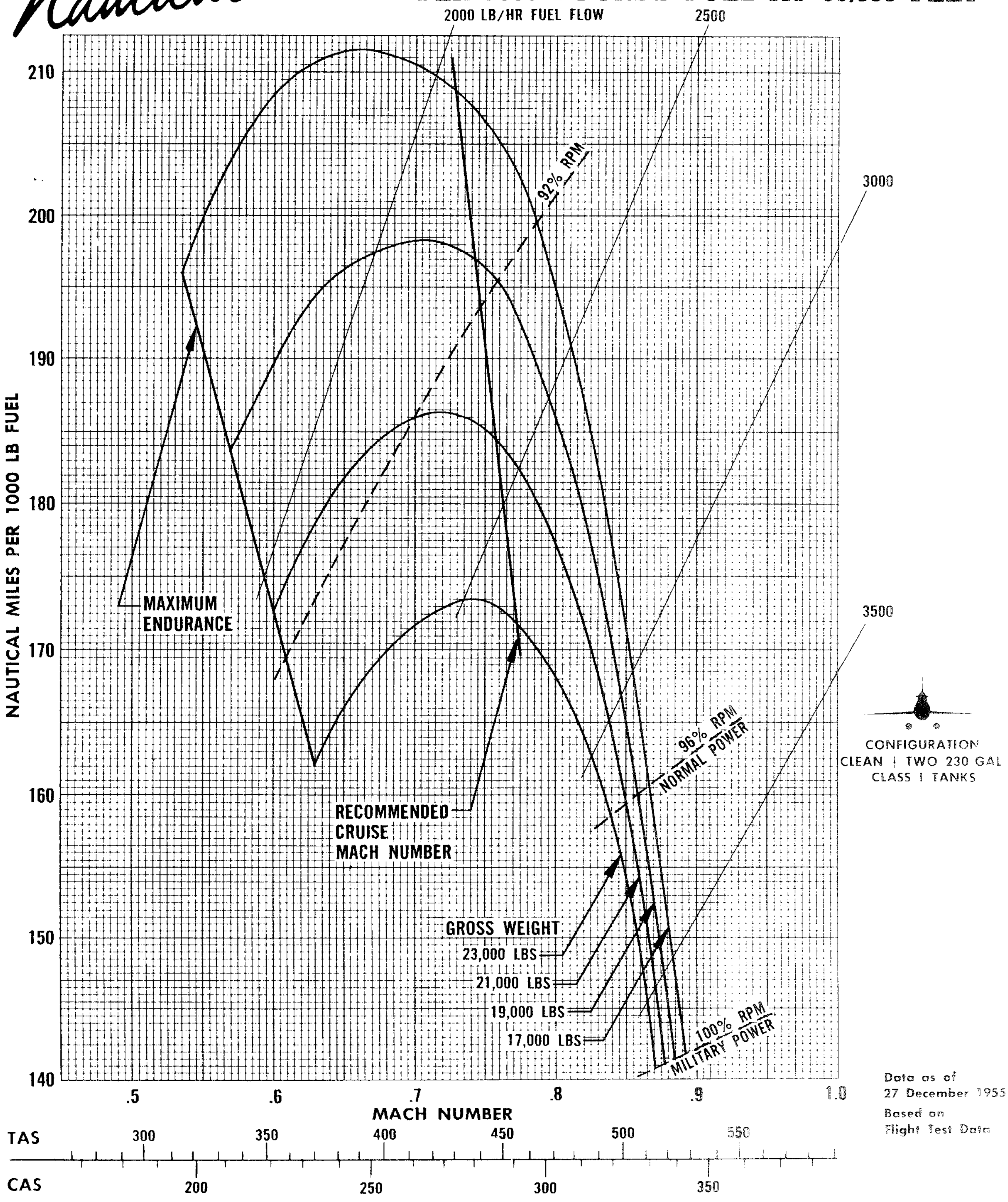


Figure A4—6 (Sheet 4 of 5)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 35,000 FEET

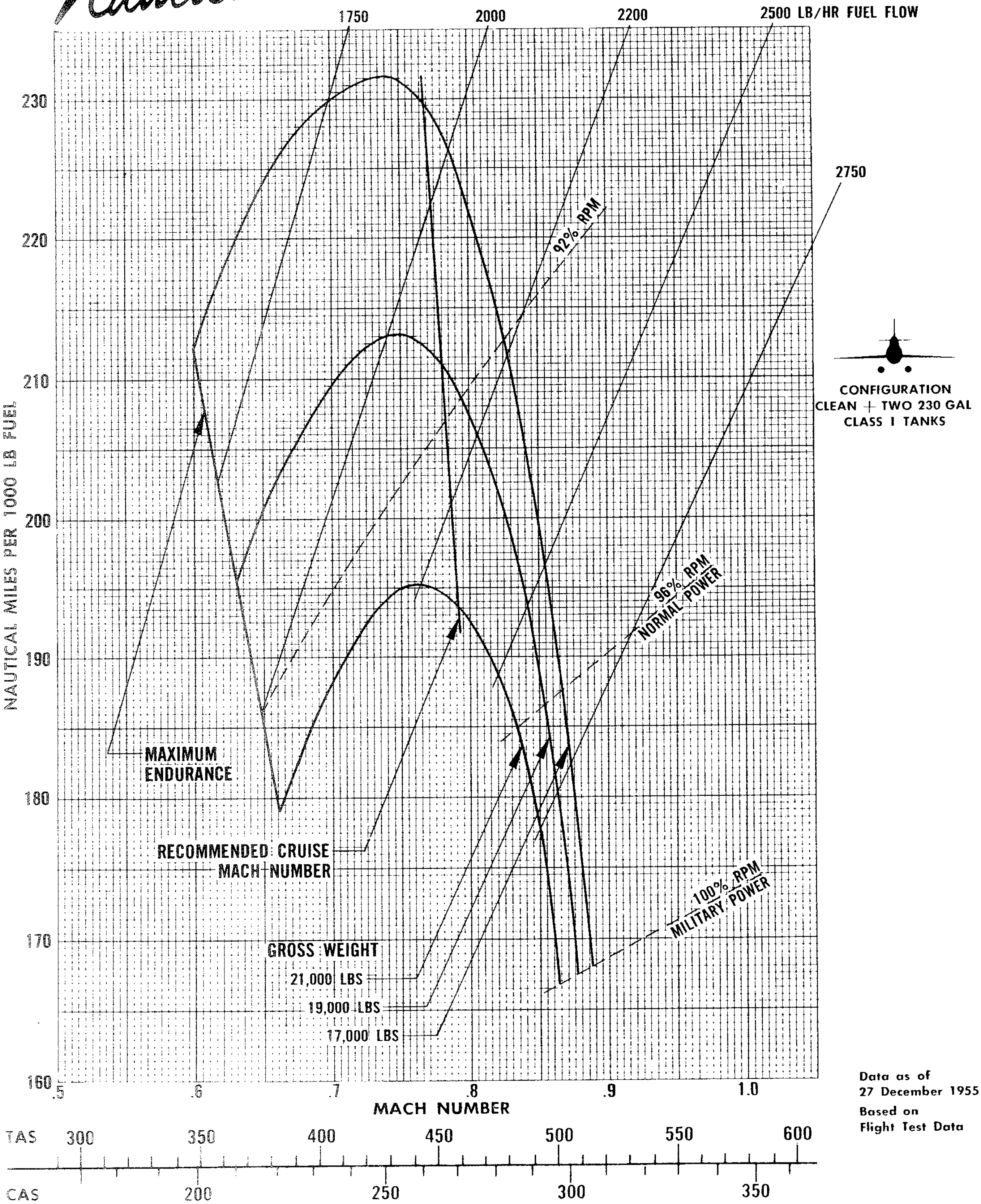


Figure A4— 6 (Sheet 5 of 5)

MODEL: RF-84F
ENGINE: J65-3
STANDARD DAY

Mission Profile

GROSS WEIGHT
15,802 — 25,390 LB

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
35,000	.78	265	450	2350	95
30,000	.76	285	445	2450	93
25,000	.70	290	420	2500	90
20,000	.64	290	390	2550	89
15,000	.58	295	365	2700	88
10,000	.54	300	345	2800	87
5,000	.50	305	325	3000	86
SEA LEVEL	.47	310	310	3150	85

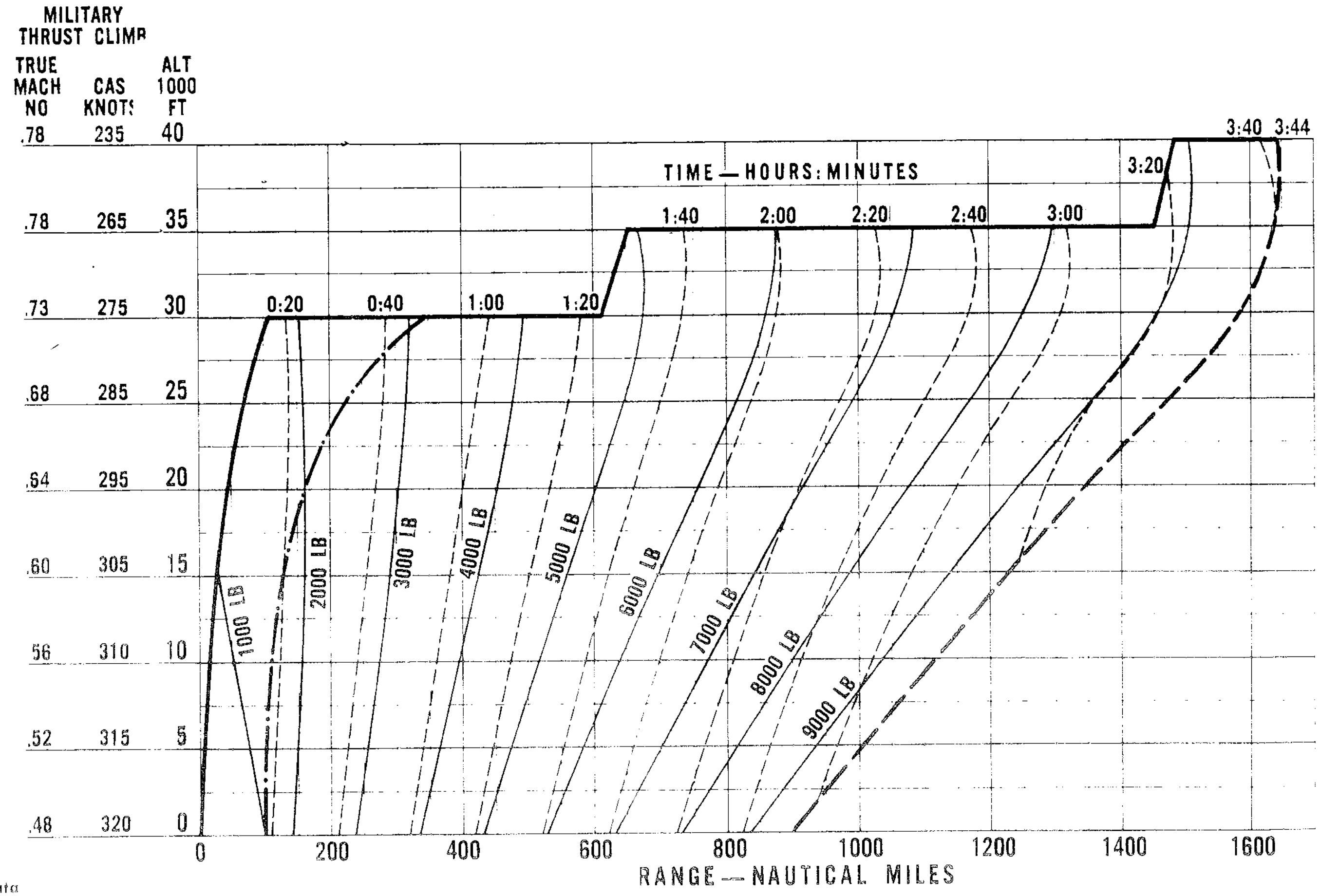
CONFIGURATION
CLEAN + TWO 450 GAL CLASS I TANKS

REMARKS

1. Fuel allowance for start, taxi and take-off (465 lb) included.
2. No allowance or reserve for loiter, descent or landing.
3. Use military thrust for climb. (see military thrust climb chart for detailed information)
4. Cruise at recommended Mach No.

LEGEND

- Optimum cruise flight path
- Line of best range for constant altitude
- Time (start, taxi and take-off not included)
- Fuel consumed
- 9588 lb fuel consumed — Zero fuel remaining



Data as of 27 Dec 1955 -- Based on Flight Test Data

Figure A4—7

CONFIGURATION: CLEAN + TWO 450 GAL CLASS I TANKS

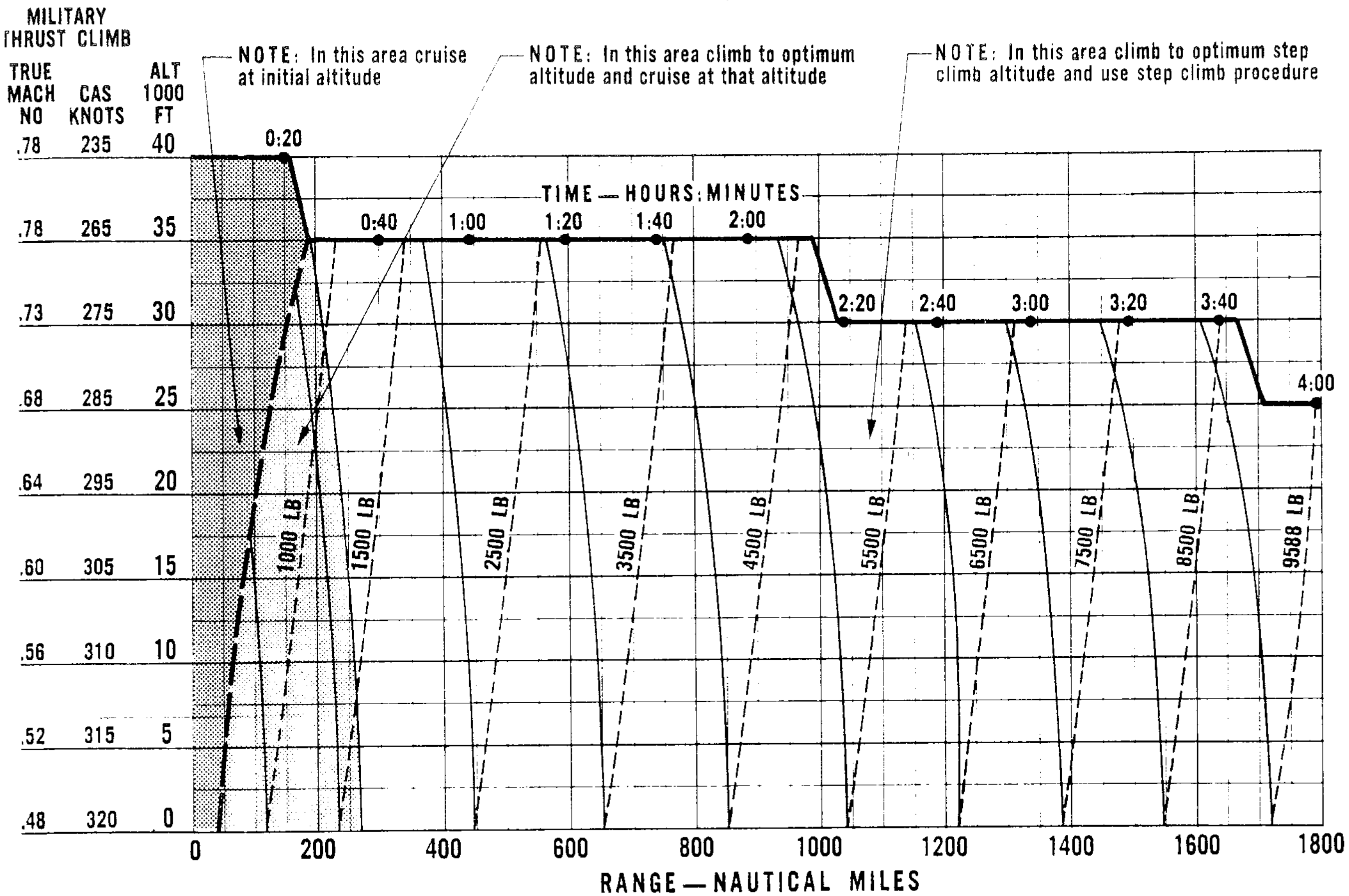
Optimum Return Profile

MODEL: RF-84F

ENGINE: J65-3 — STANDARD DAY

GROSS WEIGHT
15,802 — 25,390 LB

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	APPROXIMATE TAS-KNOTS	APPROXIMATE FUEL-LB/HR	APPROXIMATE % MIL RPM
35,000	.78	265	450	2330	95
30,000	.76	285	445	2450	93
25,000	.70	290	420	2490	90
20,000	.64	290	390	2565	89
15,000	.58	295	365	2675	88
10,000	.54	300	345	2810	87
5,000	.50	305	325	2965	86
SEA LEVEL	.47	310	310	3135	85



REMARKS

1. Fuel required at any point below the line of best range for constant altitude flight or below the optimum range flight profile includes military power climb to flight altitude.
2. No allowance made for loiter, descent or landing.
3. Best cruise condition determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach No.

LEGEND

- Optimum range flight profile
- Line of best range for constant altitude
- Climb path guide lines
- Fuel required
- Time at cruising altitude

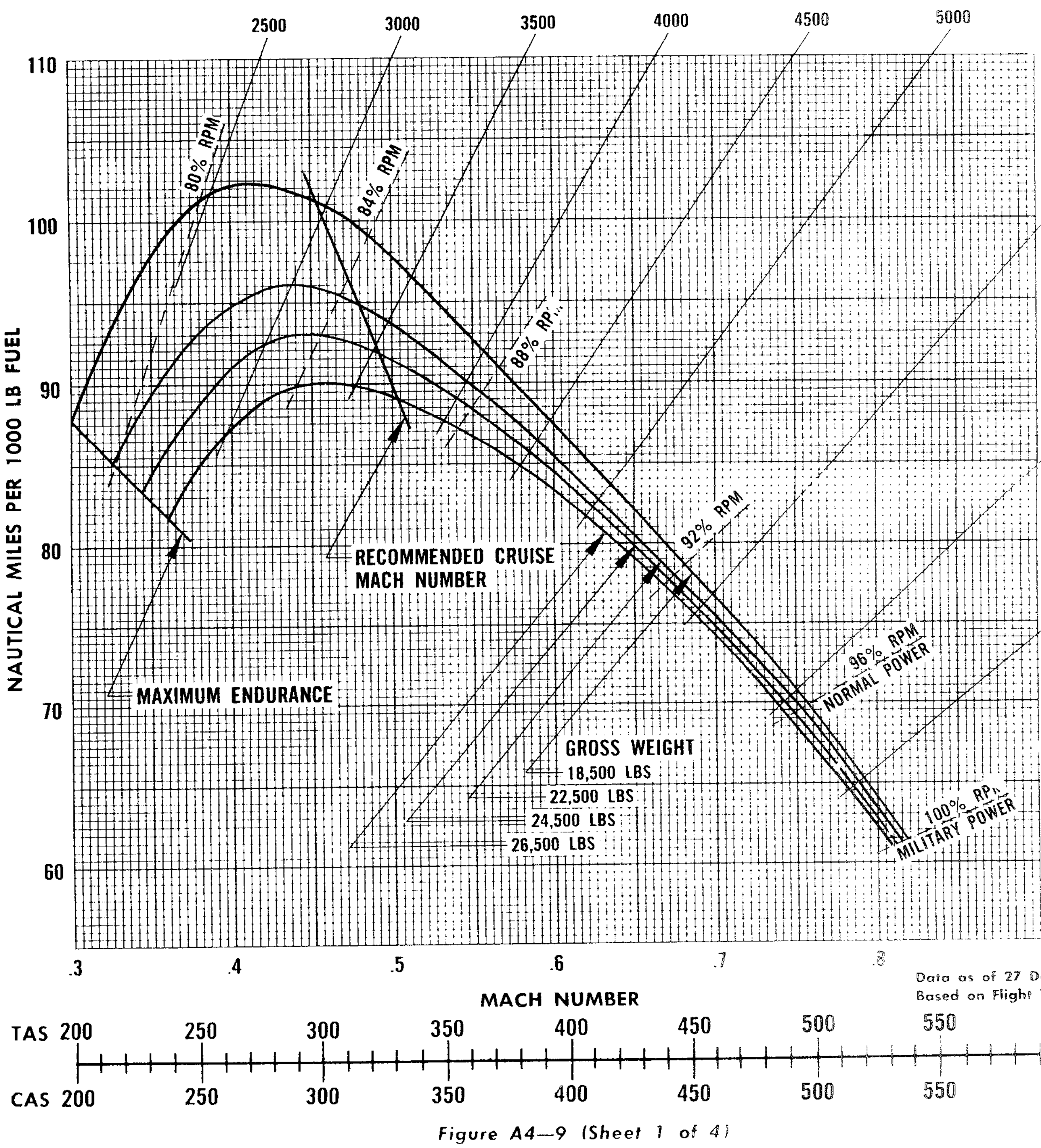
Data as of 27 Dec 1955
Based on Flight Test Data

Figure A4—8

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION
CLEAN | TWO 450 GAL CLASS I TANKS

Nautical Miles PER 1000 POUNDS FUEL AT SEA LEVEL



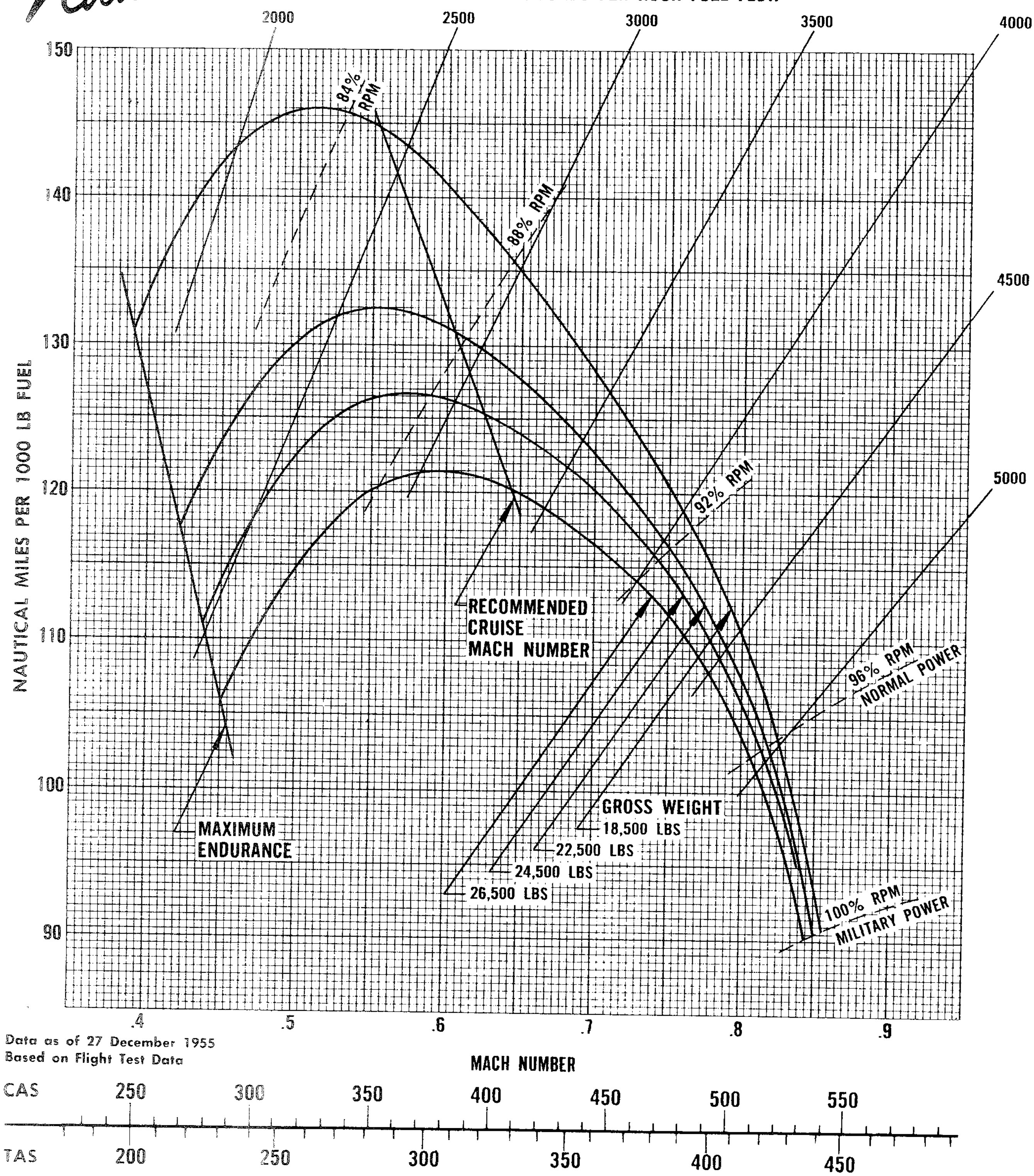
Data as of 27 December 1955
Based on Flight Test Data

Figure A4—9 (Sheet 1 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 15,000 FEET

CONFIGURATION
CLEAN + TWO 450
GAL CLASS I TANKS



Data as of 27 December 1955
Based on Flight Test Data

Figure A4—9 (Sheet 2 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 25,000 FEET

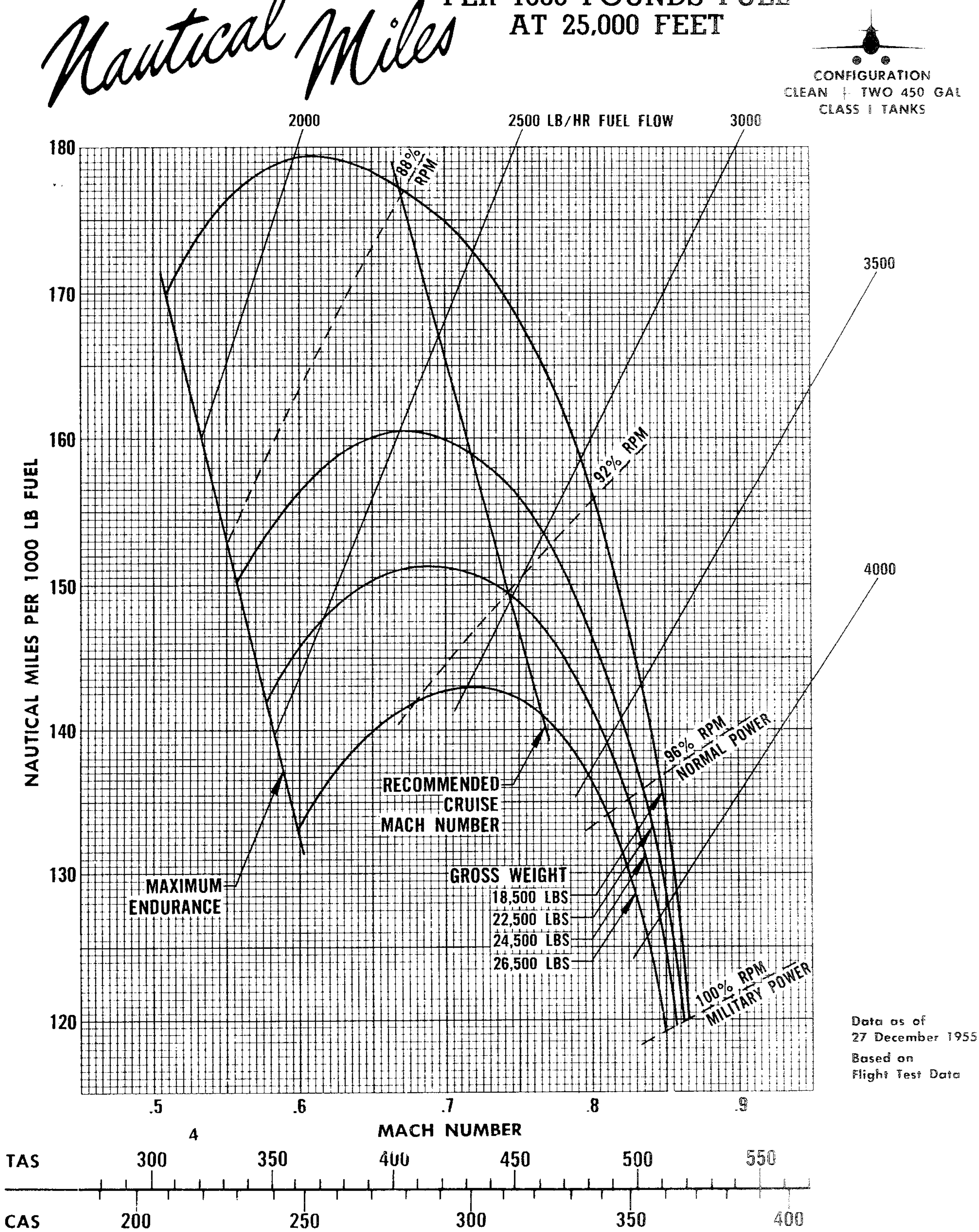
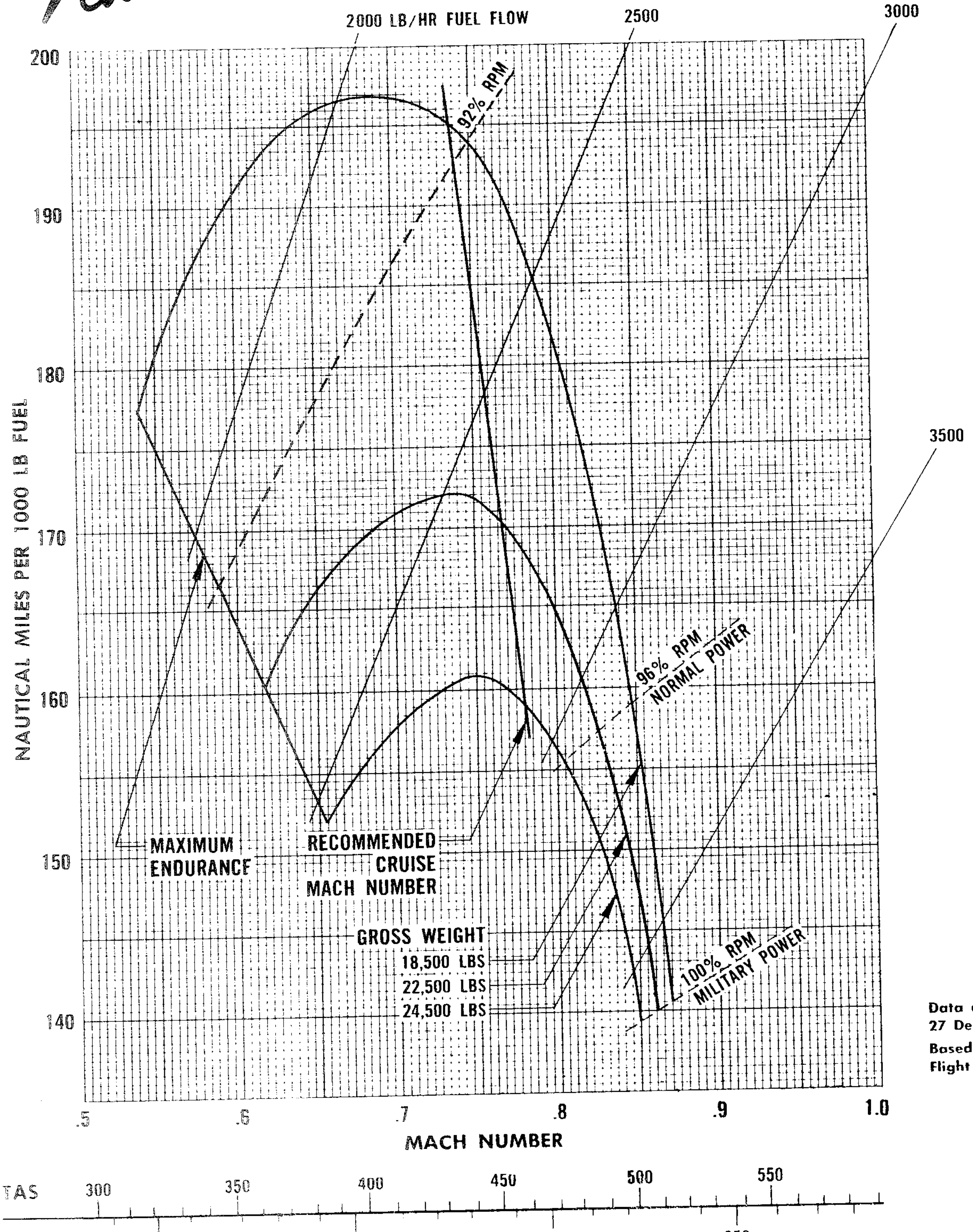


Figure A4—9 (Sheet 3 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 30,000 FEET

CONFIGURATION
CLEAN + TWO 450 GAL
CLASS I TANKS



Data as of
27 December 1955
Based on
Flight Test Data

Figure A4—9 (Sheet 4 of 4)

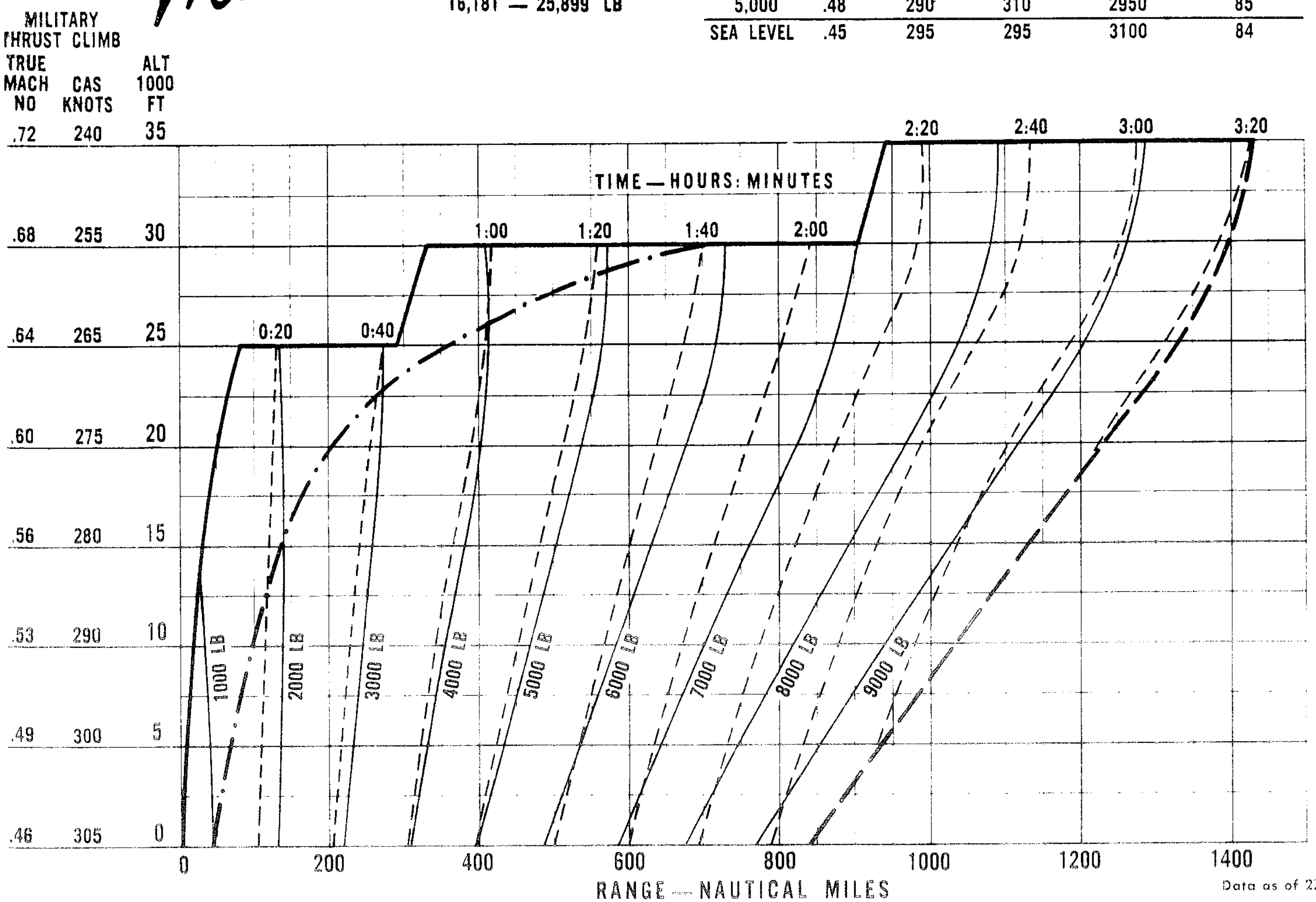
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS + TWO 230 GAL CLASS II TANKS

Mission Profile

GROSS WEIGHT 16,181 — 25,899 LB

FEET ALTITUDE	NO MACH	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
35,000	.75	250	430	2100	94
30,000	.72	270	425	2300	93
25,000	.67	275	400	2500	90
20,000	.61	280	375	2600	89
15,000	.56	280	350	2650	87
10,000	.52	285	330	2800	86
5,000	.48	290	310	2950	85
SEA LEVEL	.45	295	295	3100	84



REMARKS

1. Fuel allowance for start, taxi and take-off (465 lb) included.
2. No allowance or reserve for loiter, descent or landing.
3. Use military thrust for climb. (see military thrust climb chart for detailed information)
4. Cruise at recommended Mach No.

LEGEND

Optimum cruise flight path

Line of best range for constant altitude

Time (start, taxi and take-off not included)

Fuel consumed

9718 lb fuel consumed — Zero fuel remaining

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A4—10

MODEL: RF-84F
ENGINE: J65-3
STANDARD DAY

Optimum Return Profile

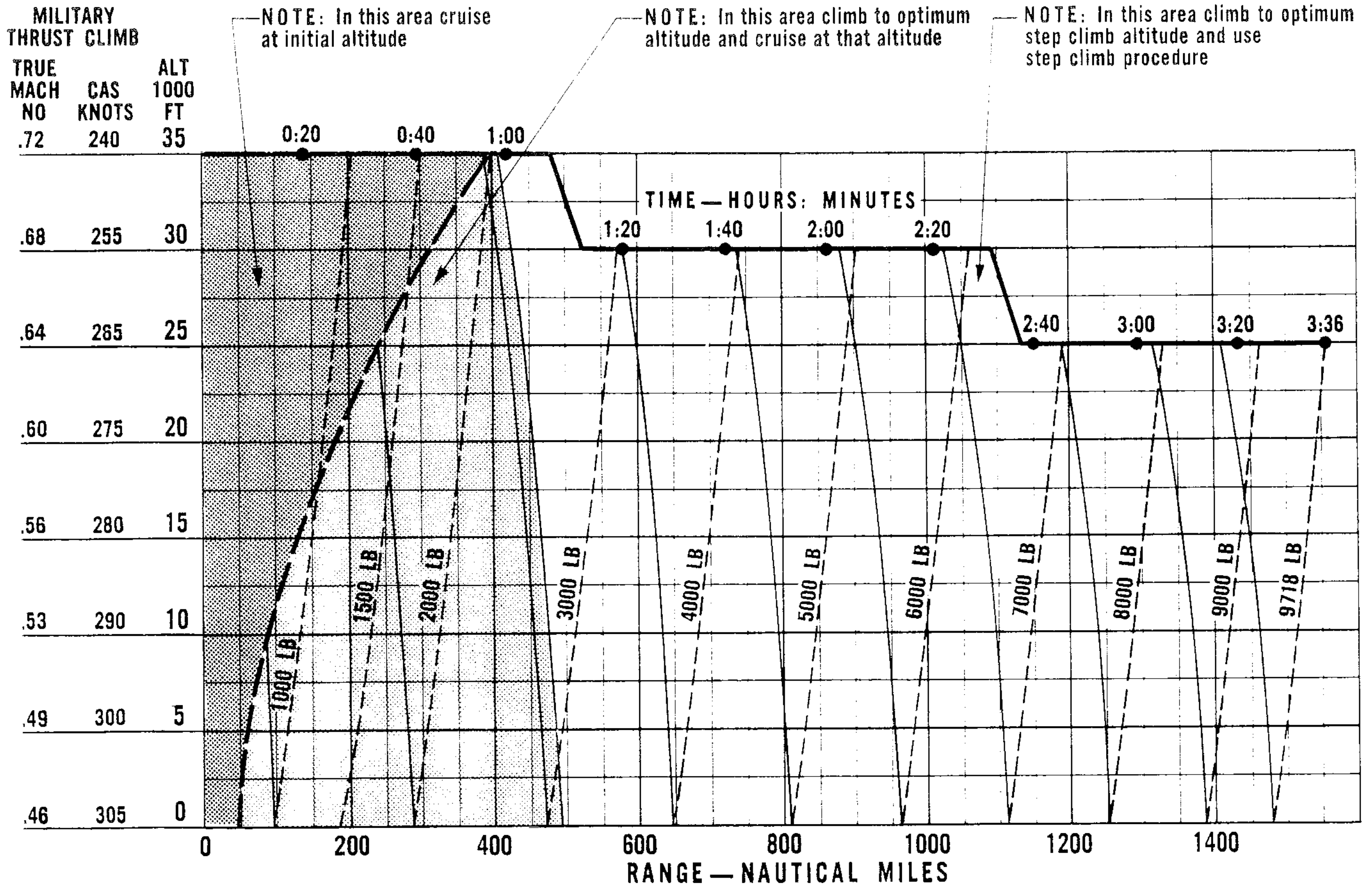
CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS + TWO 230 GAL CLASS II TANKS

GROSS WEIGHT
16,181 — 25,899 LB

FEET ALTITUDE	NO MACH	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HRS	% MIL RPM
35,000	.75	250	430	2100	94
30,000	.72	270	425	2300	93
25,000	.67	275	400	2500	90
20,000	.61	280	375	2600	89
15,000	.56	280	350	2650	87
10,000	.52	285	330	2800	86
5,000	.48	290	310	2950	85
SEA LEVEL	.45	295	295	3100	84

REMARKS

1. Fuel required at any point below the line of best range for constant altitude flight or below the optimum range flight profile includes military power climb to flight altitude.
2. No allowance made for loiter, descent or landing.
3. Best cruise condition determined by intersection of climb path guide lines and lines of best range.
4. Cruise at recommended Mach No.



LEGEND

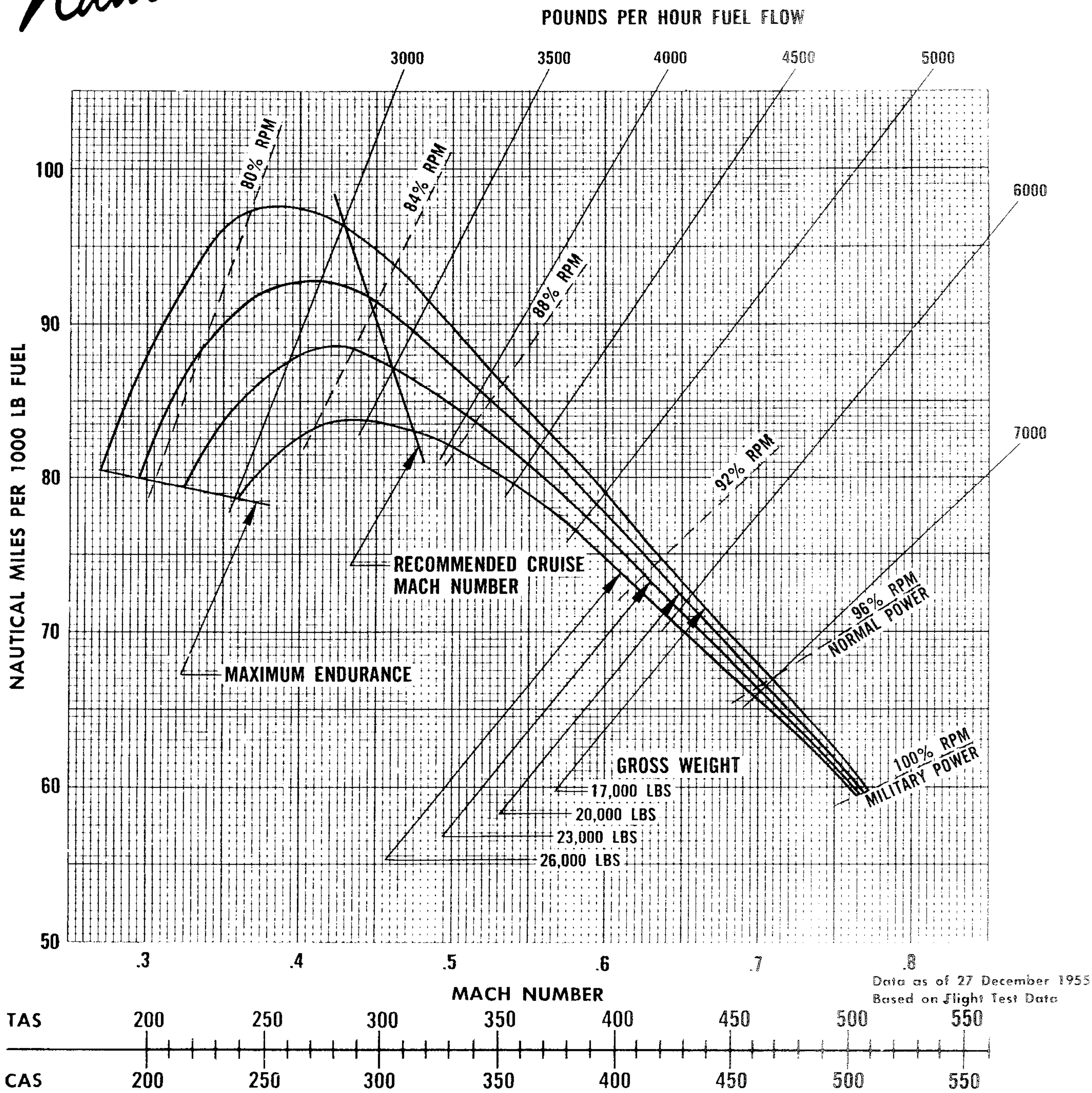
- Optimum range flight profile (solid heavy line)
- Line of best range for constant altitude (heavy dashed line)
- Climb path guide lines (thin solid line)
- Fuel required (thin dashed line)
- • Time at cruising altitude

Data as of 27 Dec 1955 -- Based on Flight Test Data

Figure A4—11

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT SEA LEVEL



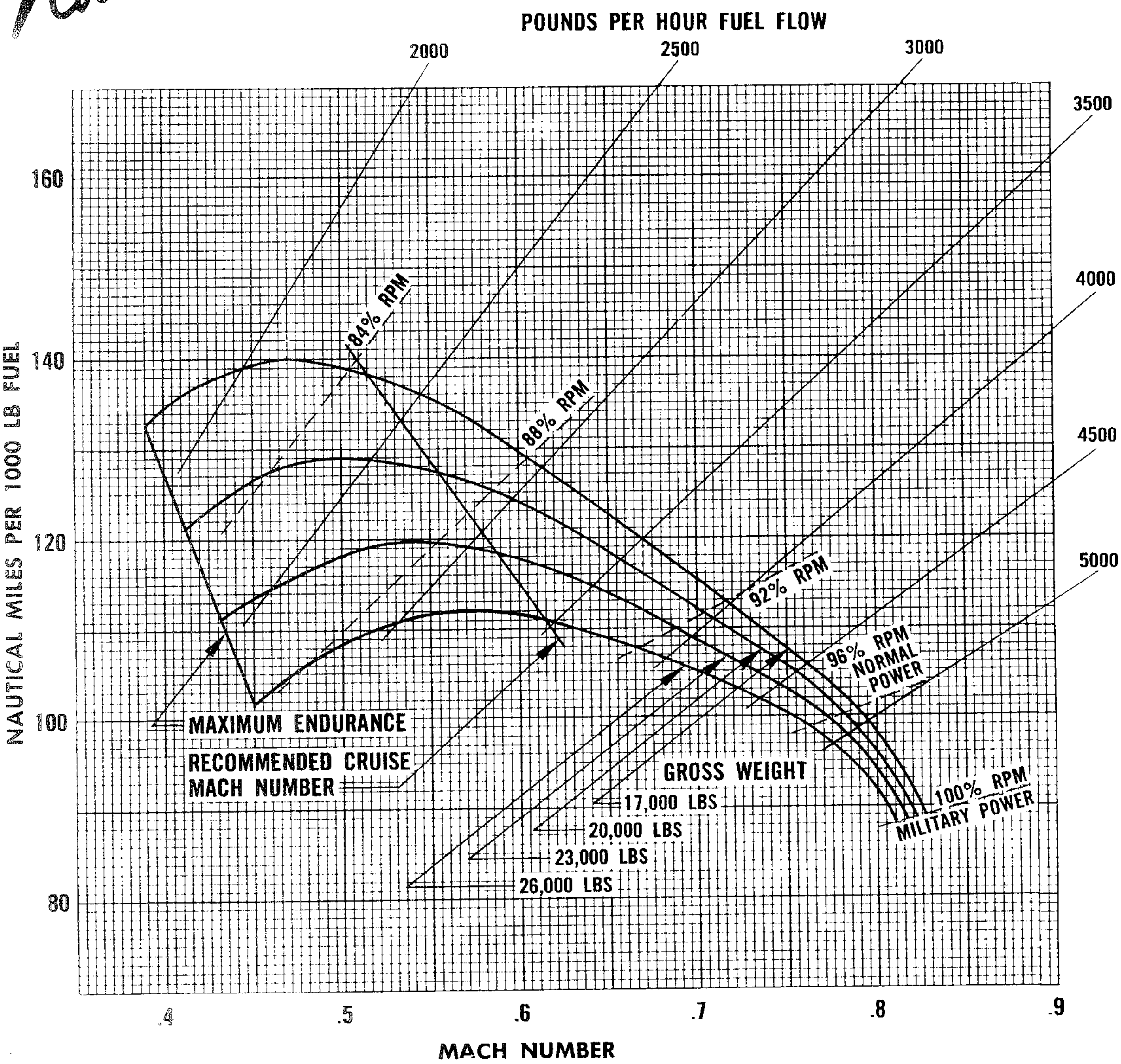
Data as of 27 December 1955
Based on Flight Test Data

Figure A4—12 (Sheet 1 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 15,000 FEET

CONFIGURATION: CLEAN
TWO 230 GAL CLASS I TANKS
TWO 230 GAL CLASS II TANKS



Data as of 27 December 1955
Based on Flight Test Data

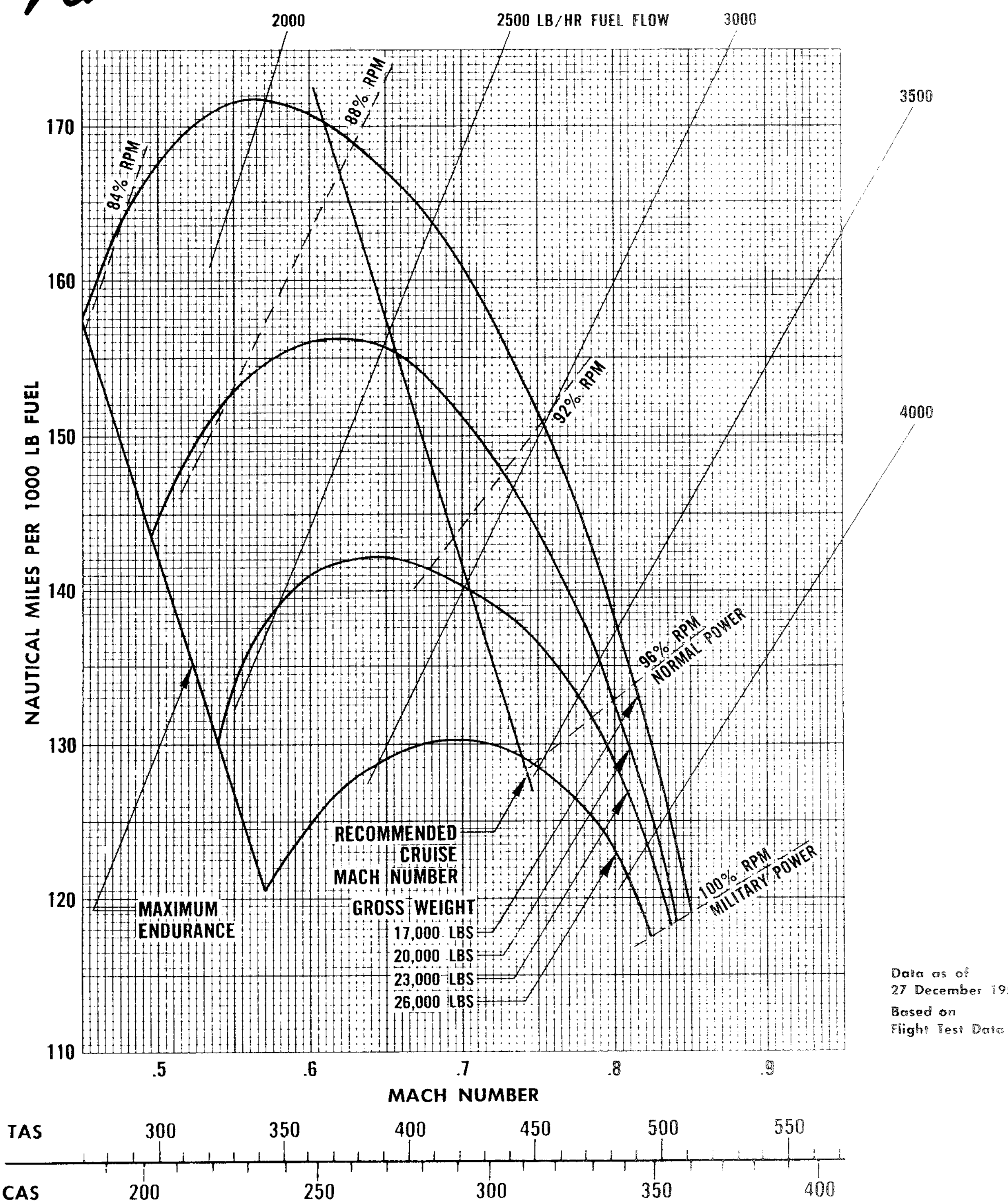
TAS	250	300	350	400	450	500	550
CAS	200	250	300	350	400	450	

Figure A4—12 (Sheet 2 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Nautical Miles PER 1000 POUNDS FUEL AT 25,000 FEET

CONFIGURATION: CLEAN
TWO 230 GAL CLASS I TANKS
TWO 230 GAL CLASS II TANKS



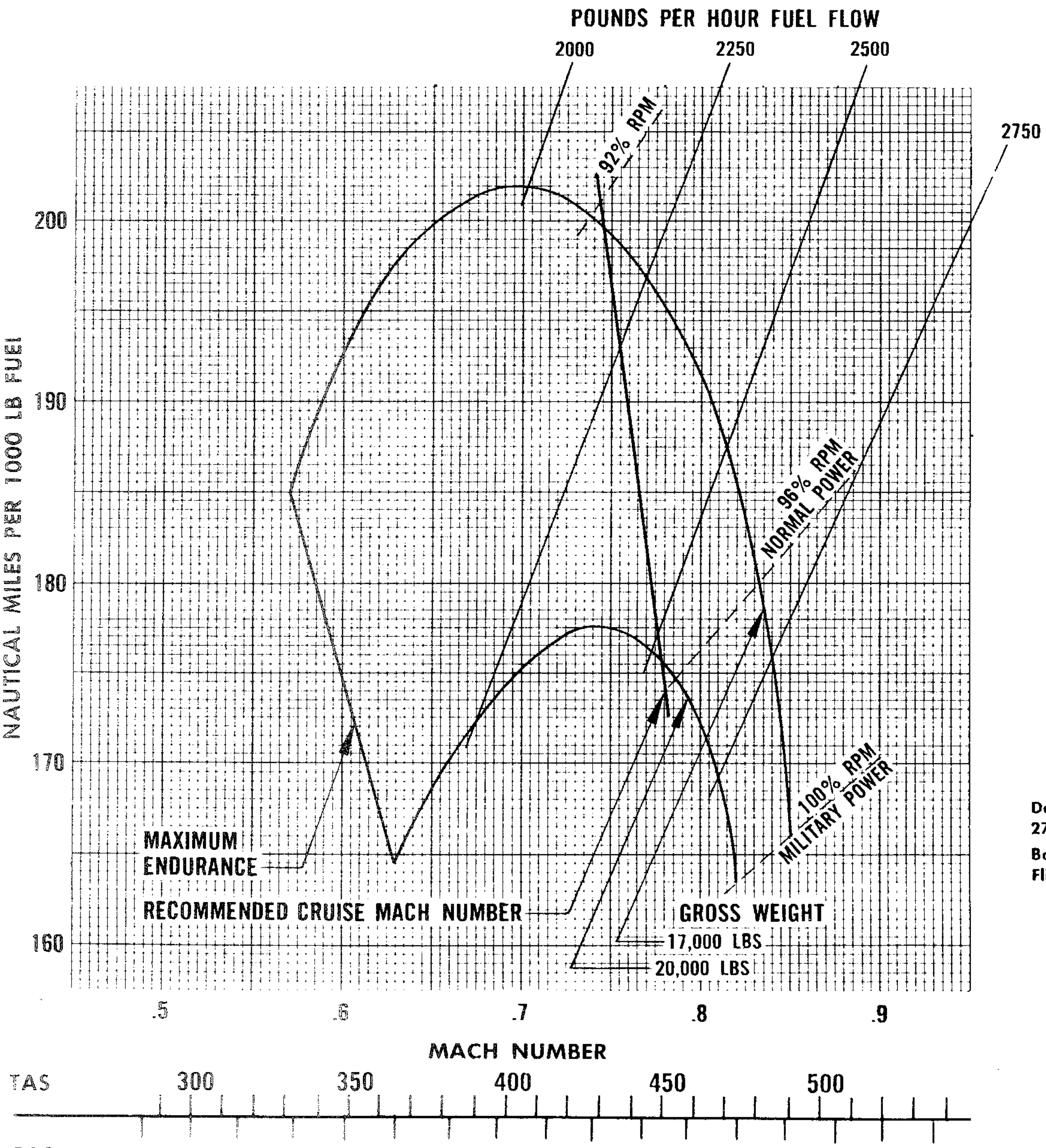
Data as of 27 December 1955
Based on Flight Test Data

Figure A4—12 (Sheet 3 of 4)

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN
TWO 230 GAL CLASS I TANKS
TWO 230 GAL CLASS II TANKS

Nautical Miles PER 1000 POUNDS FUEL AT 35,000 FEET



Data as of
27 December 1955
Based on
Flight Test Data

Figure A4—12 (Sheet 4 of 4)

PART 5 ENDURANCE

TABLE OF CONTENTS

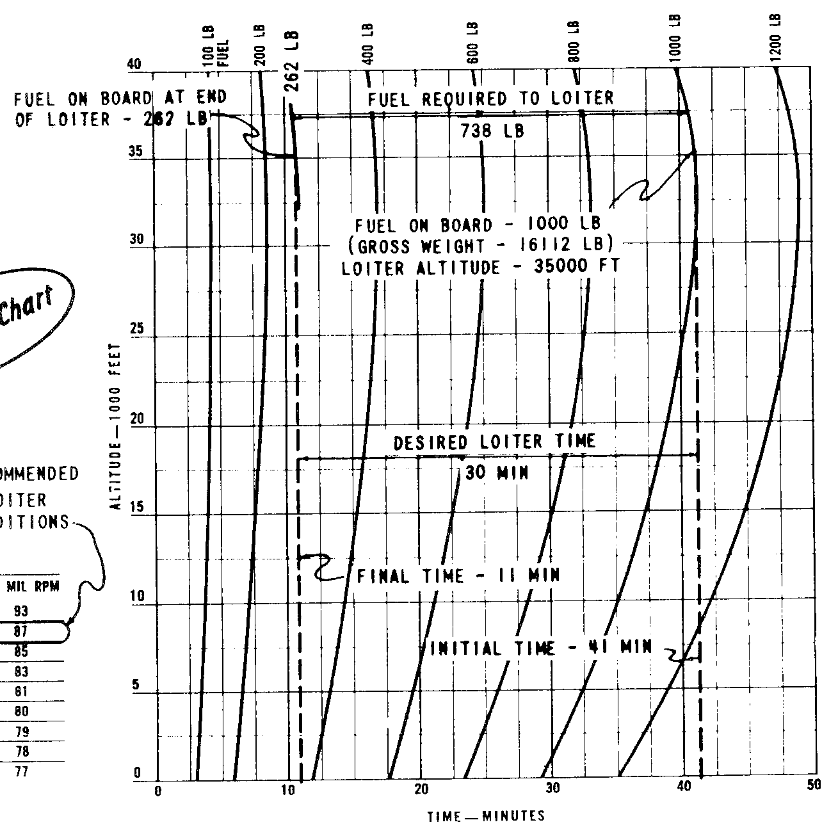
MAXIMUM ENDURANCE

MODEL: RF-84F — — STANDARD DAY

CLEAN AIRPLANE

GROSS WEIGHT
15,112 — 16,312 LB

Sample Chart



REMARKS

1. Loiter at recommended CAS
2. Maintain constant altitude

RECOMMENDED LOITER CONDITIONS

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.74	225	425	1600	93
35,000	.67	225	385	1550	87
30,000	.60	225	355	1550	85
25,000	.54	220	325	1550	83
20,000	.48	220	295	1600	81
15,000	.43	215	270	1650	80
10,000	.39	215	250	1750	79
5,000	.35	210	230	1900	78
SEA LEVEL	.32	210	210	2050	77

Data as of 27 Dec 1955 — Based on Flight Test Data

DESCRIPTION.

These profiles show the maximum time available for the fuel on board when loitering at a constant altitude. The recommended calibrated airspeed (CAS) and the approximate operating conditions are tabulated on each chart for the average gross weight.

USE.

To determine the time available for a given amount of fuel: Enter the chart at the amount of fuel on board at the start of loiter and the flight altitude; note the initial time. Re-enter the chart at the amount of fuel on board at the end of the endurance flight (initial fuel on board less fuel to be used) and read the final time. The difference between the initial and final time is the time available to loiter at constant altitude.

To obtain the fuel required to loiter a given time: Enter the chart at the amount of fuel on board at the start of loiter and flight altitude; note the initial time. Re-enter the chart at the time at the end of loiter (initial time less time to loiter) and read final fuel on board. The difference between the initial and final fuel on board is the fuel required to loiter.

From the example shown, determine the fuel required to loiter at 35,000 feet with no external load for 30 minutes. The fuel on board at start of loiter is 1000 pounds (gross weight = 16,112 pounds).

a. Initial time at 1000 pounds and 35,000 feet — 41 MIN
b. Final time (0:41 — 0:30) — 11 MIN
c. Fuel on board at end of loiter (11 MIN at 35,000 feet) — 262 pounds
d. Fuel required to loiter (1000 — 262) — 738 pounds
e. Recommended loiter CAS — 225 knots

OPTIMUM MAXIMUM ENDURANCE PROFILE

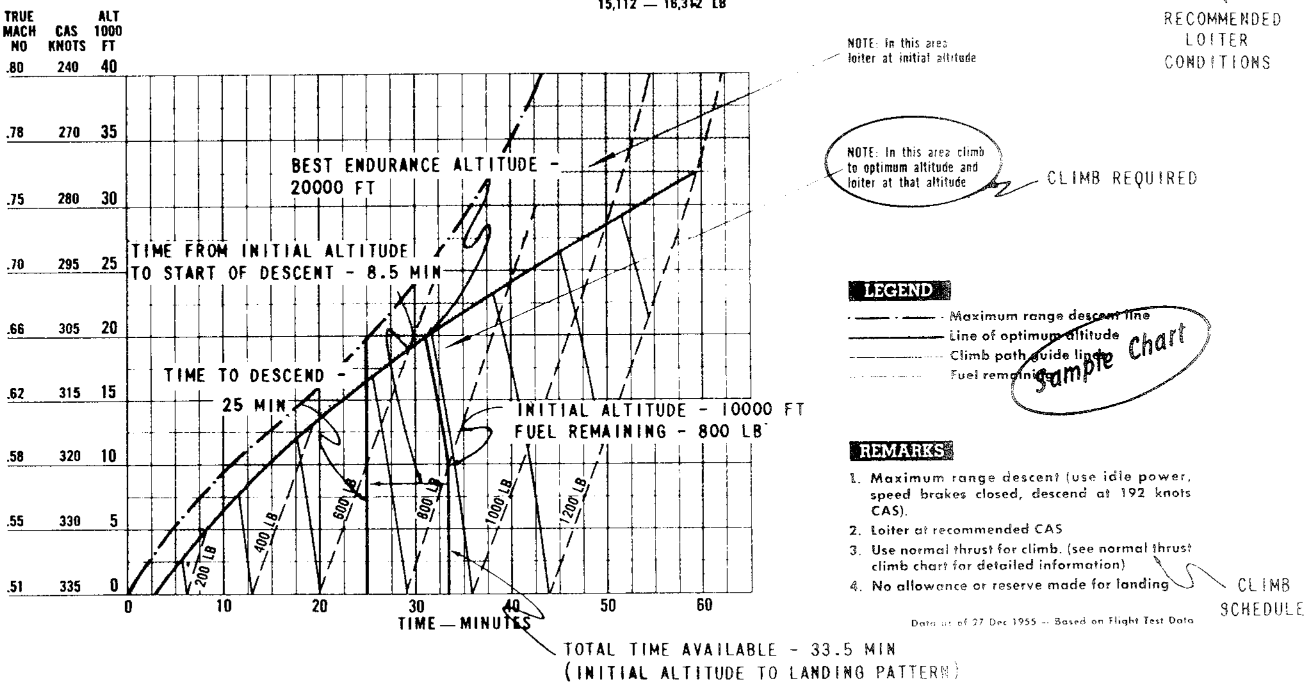
MODEL: RF-84F — ENGINE: — STANDARD DAY

CLEAN AIRPLANE

ALT FEET	MACH NO	APPROXIMATE CAS KNOTS	TAS KNOTS	FUEL LB/HR	% MIL RPM
40,000	.74	225	425	1600	93
35,000	.67	225	385	1550	87
30,000	.60	225	355	1550	85
25,000	.54	220	325	1550	83
20,000	.48	220	295	1600	81
15,000	.43	215	270	1650	80
10,000	.39	215	250	1750	79
5,000	.35	210	230	1900	78
SEA LEVEL	.32	210	210	2050	77

RECOMMENDED LOITER CONDITIONS

GROSS WEIGHT 15,112 — 16,312 LB



DESCRIPTION.

These profiles give the maximum time in the air for the fuel remaining, based on an optimum flight path, from any starting altitude. The flight path required is indicated by the different shaded areas and the notes relative to them. Time and fuel lines shown are based on a normal thrust climb to best endurance altitude, loiter at the altitude, and a maximum-range descent to sea level (no reserve for landing). The loiter speed schedule is tabulated below the chart.

USE.

The chart may be entered at the initial altitude with either the fuel remaining (to determine the time available) or the time desired (to determine the fuel requirement). The shaded area in which the initial point falls establishes the flight path to be used, as stated in the note relative to the area.

From the example shown, determine the time available and necessary flight path to remain aloft with 800 pounds of fuel remaining at 10,000 feet in the clean configuration.

a. Enter profile at 10,000 feet and 800 pounds of fuel remaining to establish starting point. Total time available is 33.5 min.

b. In this area, note that a climb is required.

c. By following the climb guide lines, the best endurance altitude is 20,000 feet.

d. Descent time from 20,000 feet to sea level is 25 minutes.

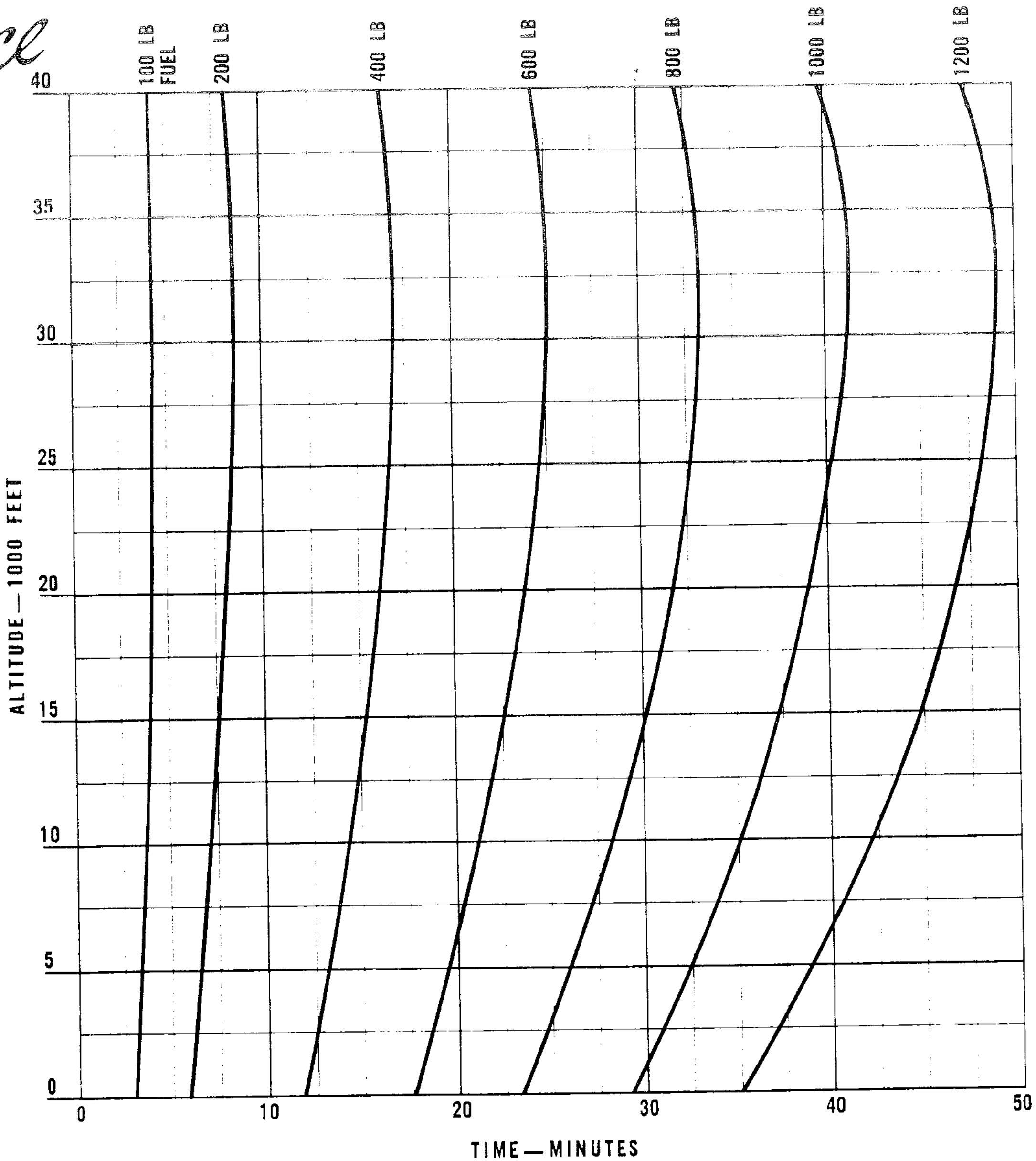
e. Elapsed time from start of climb to start of descent is 8.5 minutes (33.5-25).

MODEL: RF-84F · ENGINE: J65-3 — STANDARD DAY

CLEAN AIRPLANE

Maximum Endurance

GROSS WEIGHT
15,112 — 16,312 LB



REMARKS

1. Loiter at recommended CAS
2. Maintain constant altitude

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	APPROXIMATE TAS-KNOTS	APPROXIMATE FUEL-LB/HR	APPROXIMATE % MIL RPM
40,000	.74	225	425	1600	93
35,000	.67	225	385	1550	87
30,000	.60	225	355	1550	85
25,000	.54	220	325	1550	83
20,000	.48	220	295	1600	81
15,000	.43	215	270	1650	80
10,000	.39	215	250	1750	79
5,000	.35	210	230	1900	78
SEA LEVEL	.32	210	210	2050	77

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5—1

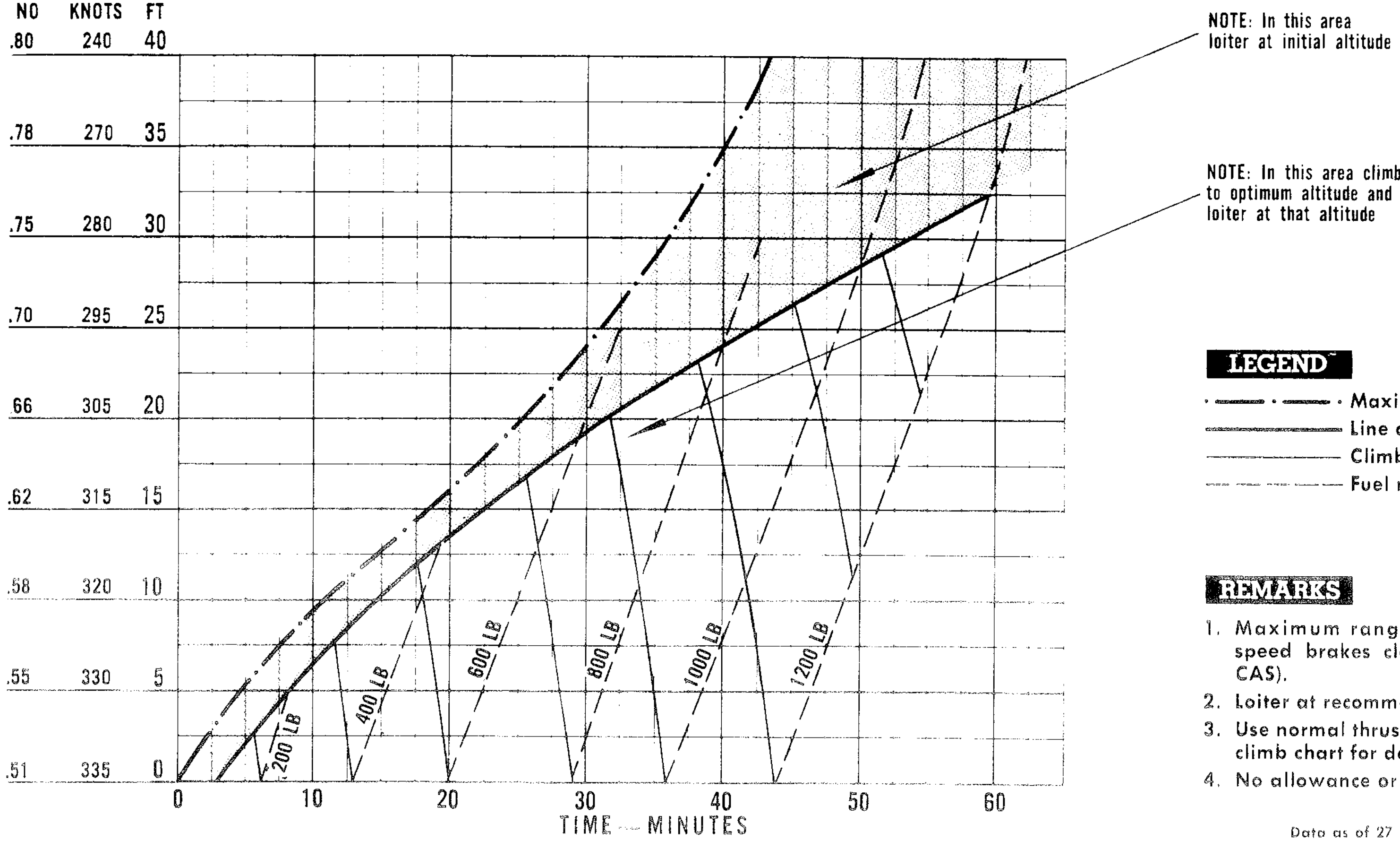
MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CLEAN AIRPLANE

Optimum Profile MAXIMUM ENDURANCE

ALT FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
40,000	.74	225	425	1600	93
35,000	.67	225	385	1550	87
30,000	.60	225	355	1550	85
25,000	.54	220	325	1550	83
20,000	.48	220	295	1600	81
15,000	.43	215	270	1650	80
10,000	.39	215	250	1750	79
5,000	.35	210	230	1900	78
SEA LEVEL	.32	210	210	2050	77

GROSS WEIGHT
15,112 — 16,312 LB



LEGEND

- Maximum range descent line
- Line of optimum altitude
- Climb path guide lines
- Fuel remaining

REMARKS

1. Maximum range descent (use idle power speed brakes closed, descend at 192 knots CAS).
2. Loiter at recommended CAS
3. Use normal thrust for climb. (see normal thrust climb chart for detailed information)
4. No allowance or reserve made for landing

Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5—2

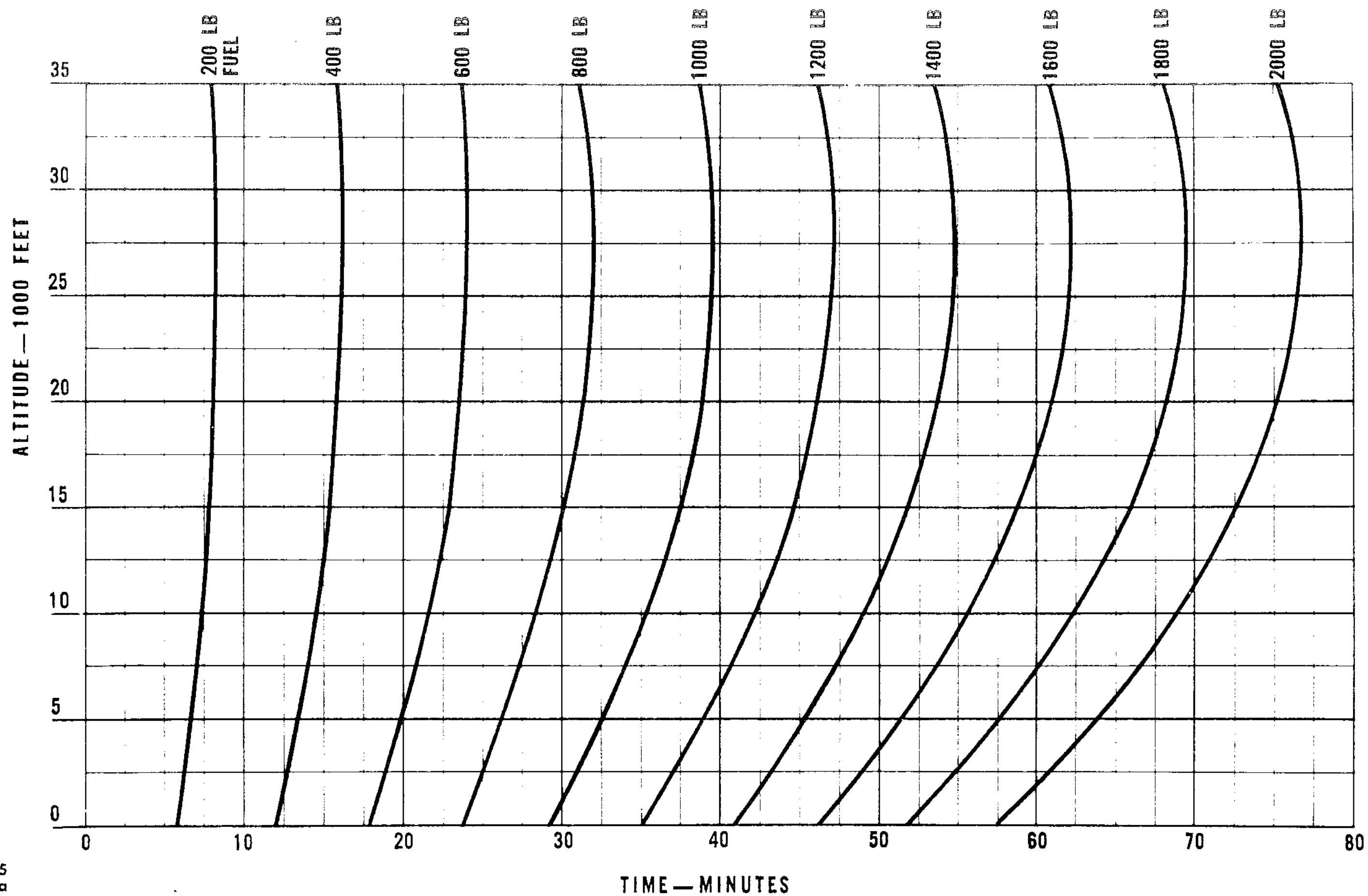
MODEL: RF-84F ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

Maximum Endurance

GROSS WEIGHT
15,682 — 17,682 LB

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	APPROXIMATE TAS-KNOTS	APPROXIMATE FUEL-LB/HR	APPROXIMATE % MIL RPM
35,000	.60	200	345	1500	88
30,000	.54	195	315	1500	85
25,000	.45	190	270	1500	83
20,000	.39	190	240	1500	81
15,000	.37	190	235	1550	80
10,000	.34	190	220	1650	78
5,000	.31	190	205	1800	78
SEA LEVEL	.28	185	185	2000	77



REMARKS

1. Loiter at recommended CAS
2. Maintain constant altitude

Data as of 27 Dec 1955
Based on Flight Test Data

Figure A5—3

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 450 GAL CLASS I TANKS

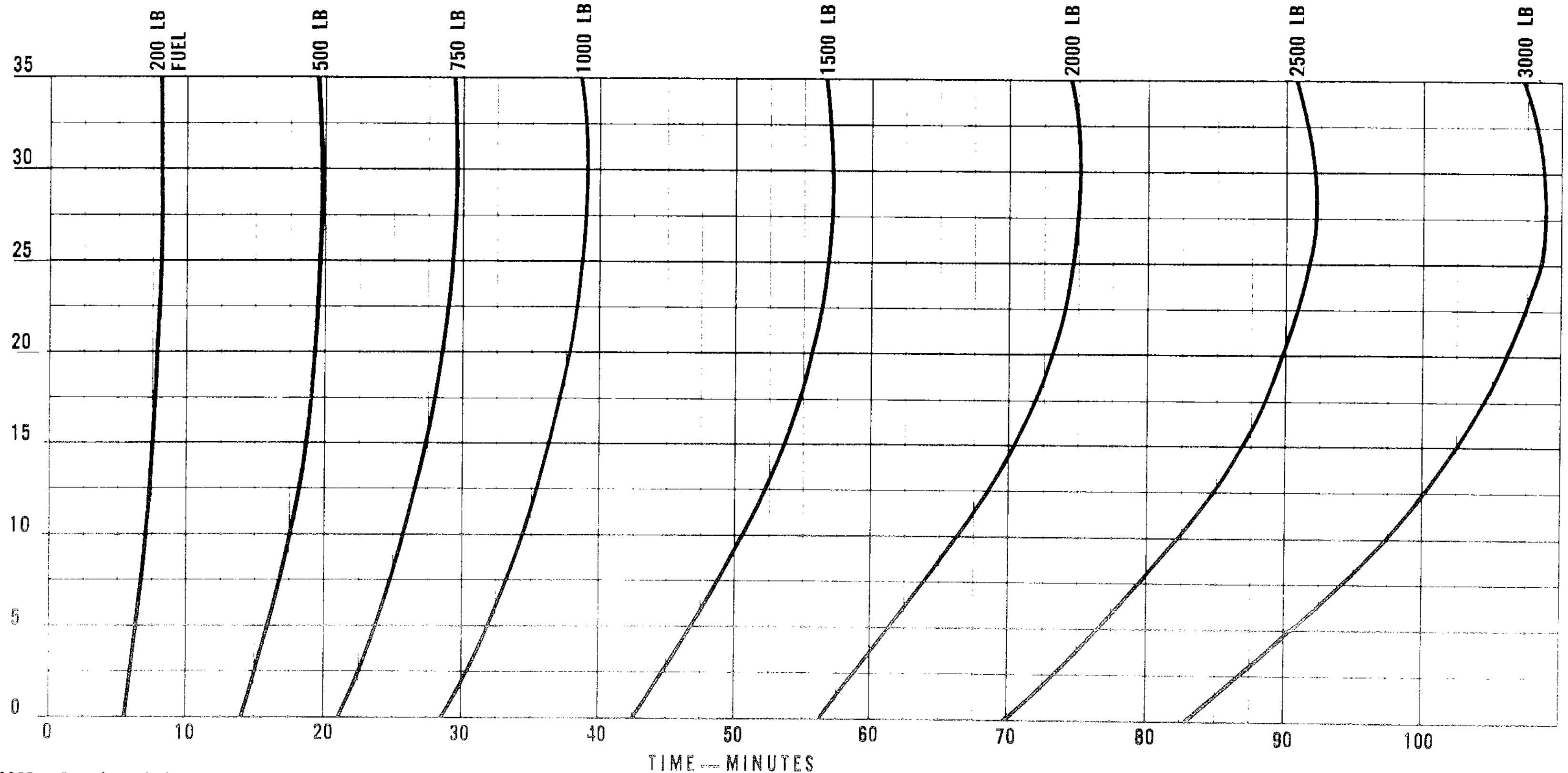
GROSS WEIGHT
15,802 — 18,802 LB

Maximum Endurance

ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	APPROXIMATE TAS-KNOTS	APPROXIMATE FUEL-LB/HR	APPROXIMATE % MIL RPM
35,000	.67	225	385	2000	93
30,000	.58	215	340	1950	92
25,000	.52	215	315	1900	87
20,000	.47	215	290	1950	84
15,000	.42	215	265	2000	83
10,000	.38	215	245	2050	82
5,000	.35	210	225	2150	80
SEA LEVEL	.32	210	210	2300	78

REMARKS

1. Loiter at recommended CAS
2. Maintain constant altitude



Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5—5

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

GROSS WEIGHT
15,682 — 17,682 LB

Optimum MAXIMUM ENDURANCE Profile

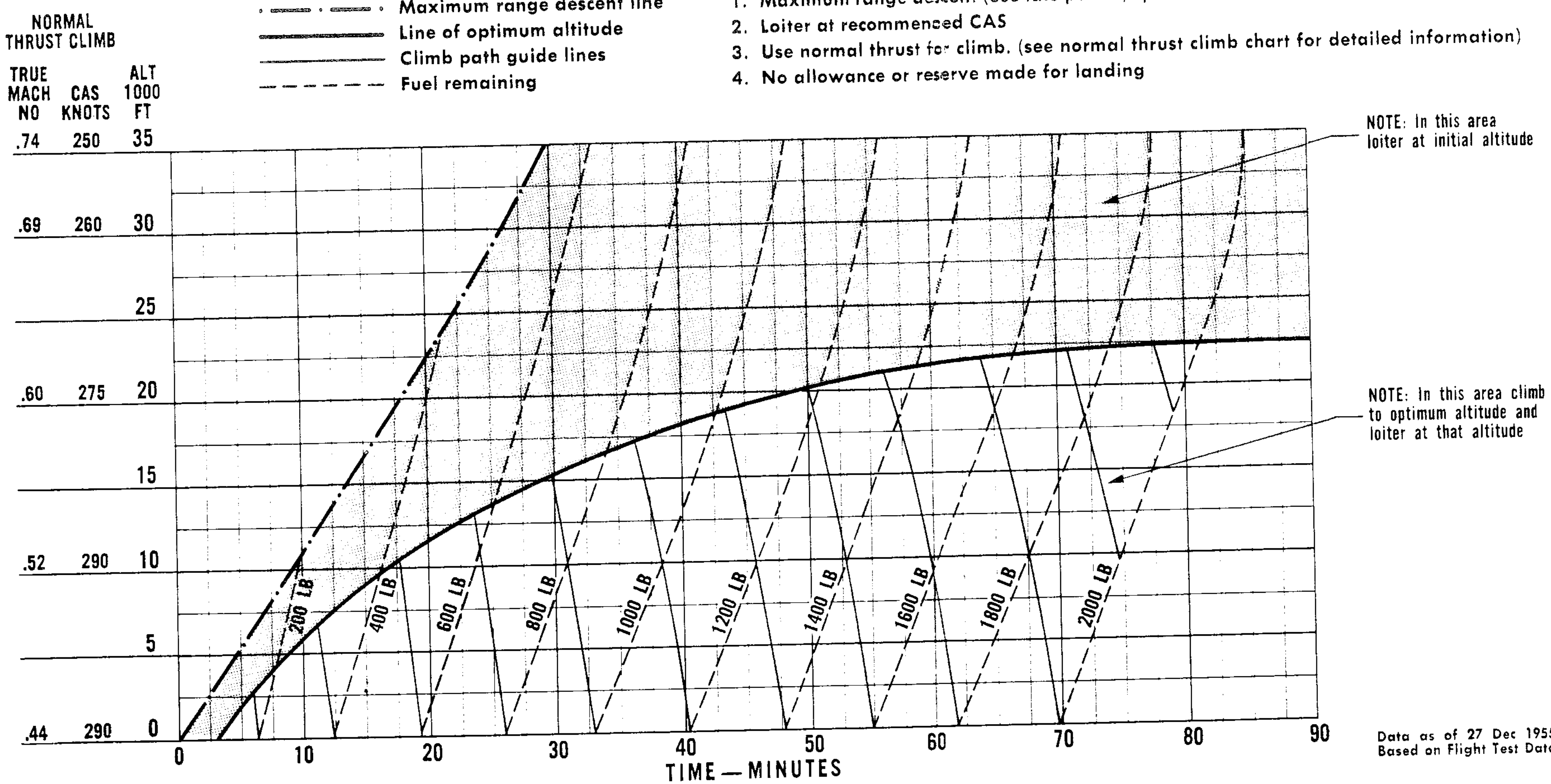
ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	APPROXIMATE TAS-KNOTS	APPROXIMATE FUEL-LB/HR	APPROXIMATE % MIL RPM
35,000	.60	200	345	1500	88
30,000	.54	195	315	1500	85
25,000	.45	190	270	1500	83
20,000	.39	190	240	1500	81
15,000	.37	190	235	1550	80
10,000	.34	190	220	1650	78
5,000	.31	190	205	1800	78
SEA LEVEL	.28	185	185	2000	77

LEGEND

- Maximum range descent line
- Line of optimum altitude
- Climb path guide lines
- Fuel remaining

REMARKS

1. Maximum range descent (use idle power, speed brakes closed, descent at 190 knots CAS)
2. Loiter at recommended CAS
3. Use normal thrust for climb. (see normal thrust climb chart for detailed information)
4. No allowance or reserve made for landing



Data as of 27 Dec 1955
Based on Flight Test Data

Figure A5—4

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN +
TWO 230 GAL CLASS I TANKS +
TWO 230 GAL CLASS II TANKS

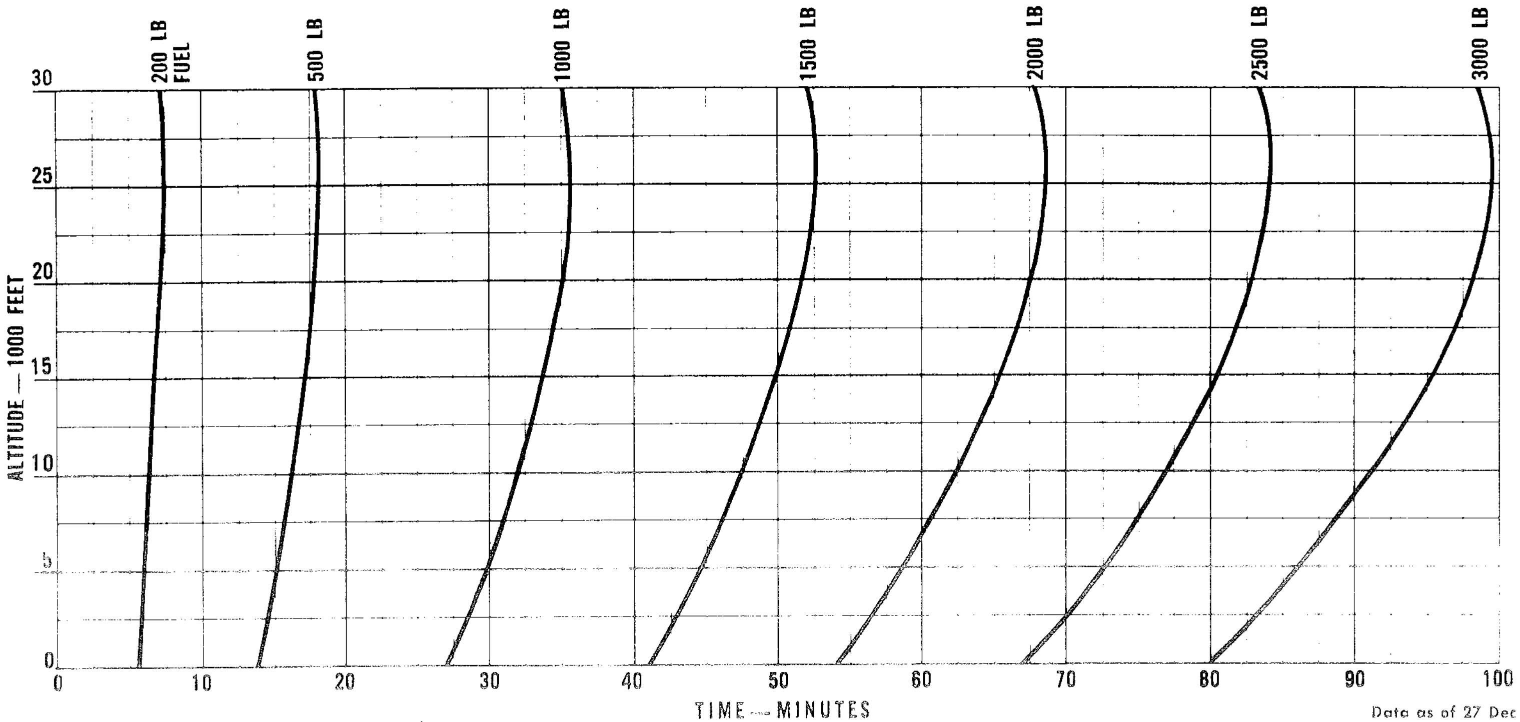
Maximum Endurance

GROSS WEIGHT
16,182 — 19,182 LB

REMARKS

1. Loiter at recommended CAS
2. Maintain constant altitude

ALTITUDE FEET	MACH NO	APPROXIMATE			
		CAS-KNOTS	TAS-KNOTS	FUEL-LB/HRS	% MIL RPM
30,000	.51	190	300	1800	87
25,000	.46	190	275	1800	84
20,000	.42	195	260	1850	83
15,000	.39	195	245	1900	81
10,000	.35	195	225	1950	80
5,000	.31	185	200	2050	78
SEA LEVEL	.27	180	180	2250	77



Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5—7

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 450 GAL CLASS I TANKS

GROSS WEIGHT
15,802 — 18,802 LB

Optimum MAXIMUM ENDURANCE Profile

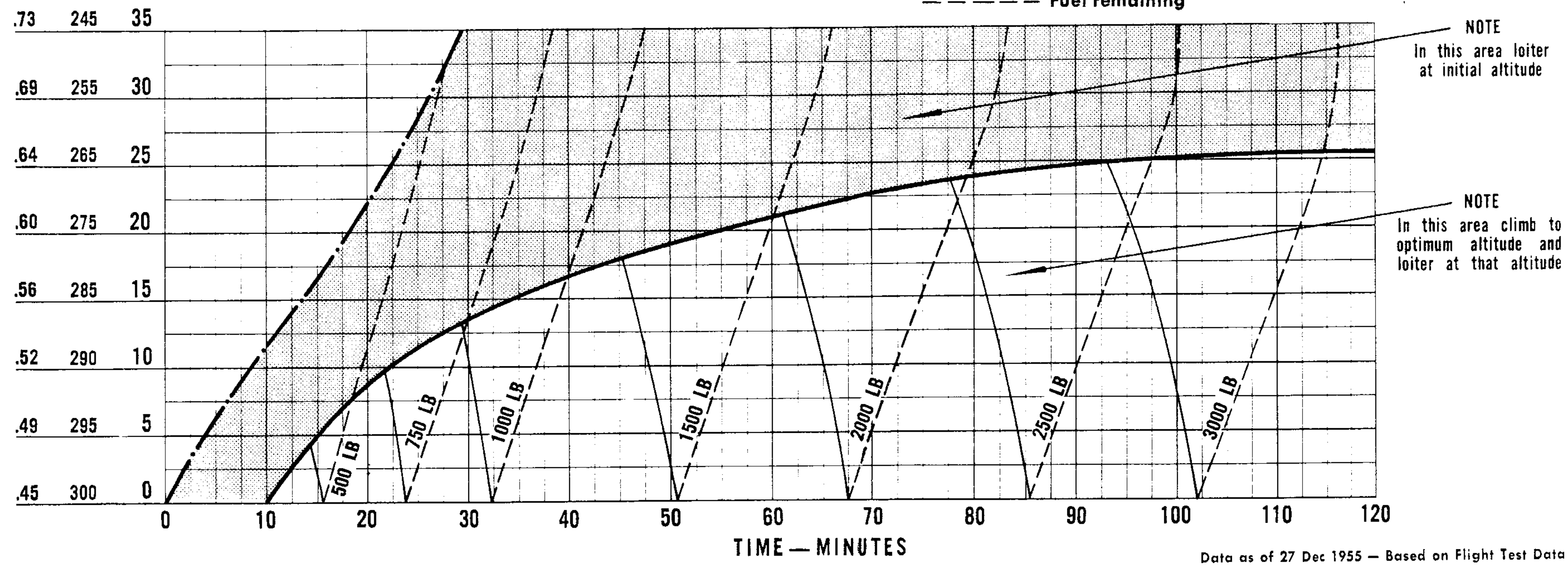
ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
35,000	.67	225	385	2000	93
30,000	.58	215	340	1950	92
25,000	.52	215	315	1900	87
20,000	.47	215	290	1950	84
15,000	.42	215	265	2000	83
10,000	.38	215	245	2050	82
5,000	.35	210	225	2150	80
SEA LEVEL	.32	210	210	2300	78

REMARKS

1. Maximum range descent (use idle power, speed brakes closed, descend at 190 knots CAS).
2. Loiter at recommended CAS
3. Use normal thrust for climb. (see normal thrust climb chart for detailed information)
4. No allowance or reserve made for landing

LEGEND

- ·—·—· Maximum range descent line
- ——— Line of optimum altitude
- ——— Climb path guide lines
- – – – Fuel remaining



Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5–6

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Optimum MAXIMUM ENDURANCE Profile

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS + TWO 230 GAL CLASS II TANKS

GROSS WEIGHT 16,182 — 19,182 LB

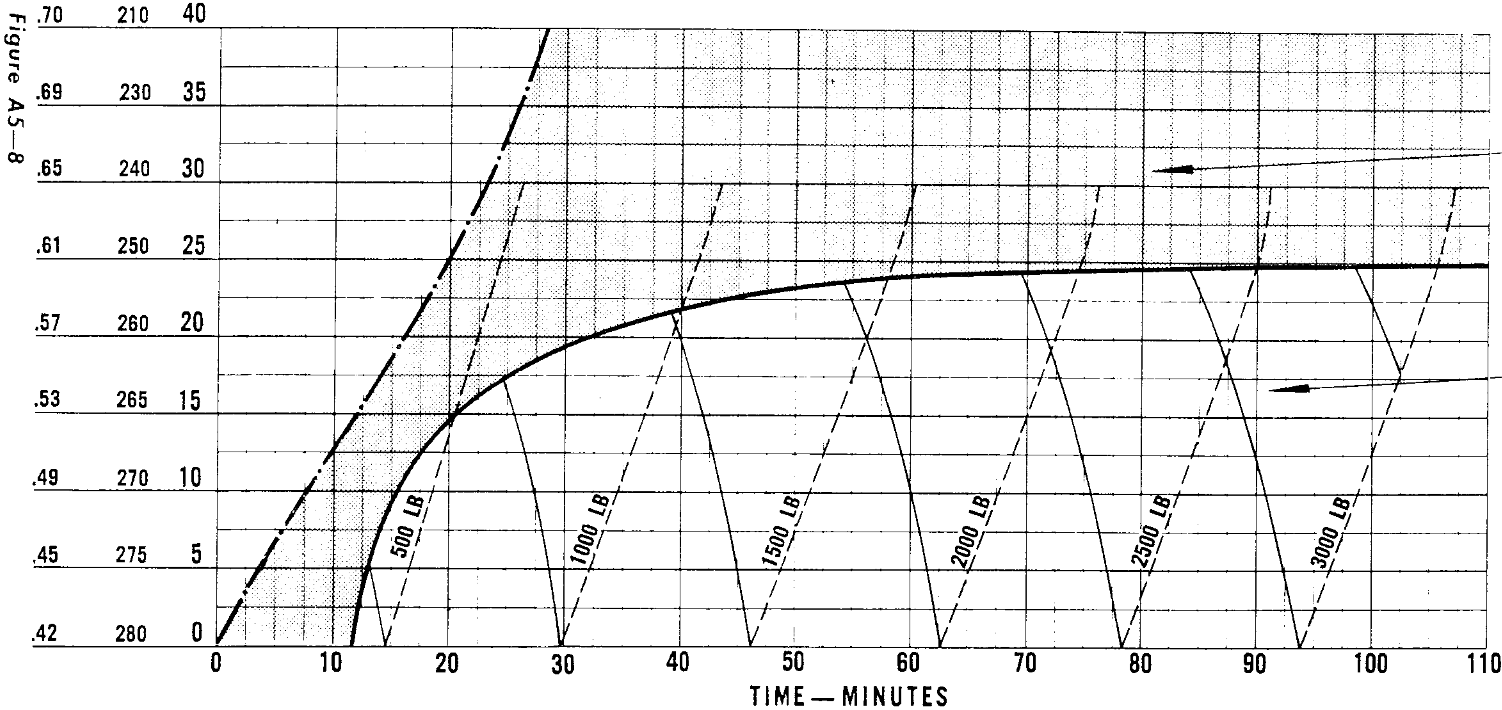
ALTITUDE FEET	MACH NO	APPROXIMATE CAS-KNOTS	TAS-KNOTS	FUEL-LB/HR	% MIL RPM
30,000	.51	190	300	1800	87
25,000	.46	190	275	1800	84
20,000	.42	195	260	1850	83
15,000	.39	195	245	1900	81
10,000	.35	195	225	1950	80
5,000	.31	185	200	2050	78
SEA LEVEL	.27	180	180	2250	77

REMARKS

1. Maximum range descent (use idle power, speed brakes closed, descend at 187 knots CAS).
2. Loiter at recommended CAS
3. Use normal thrust for climb. (see normal thrust climb chart for detailed information)
4. No allowance or reserve made for landing

LEGEND

- Maximum range descent line
- Line of optimum altitude
- Climb path guide lines
- Fuel remaining



Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A5—8

PART 6 DESCENT

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DESCENT.

Two types of descents are shown for all configurations, formation descents and maximum range descents.

Descents

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

CLEAN AIRPLANE

NOTE

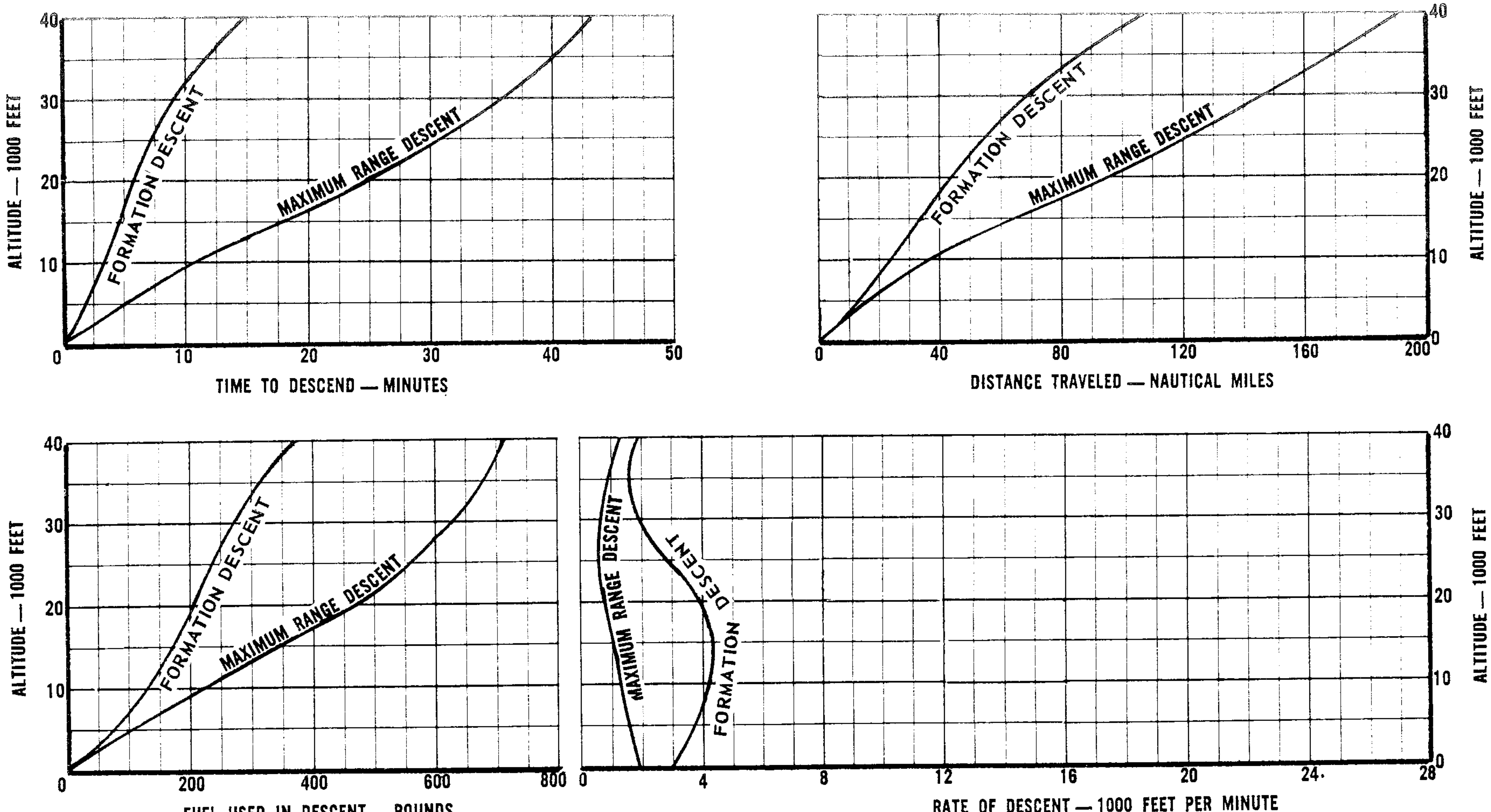
Recommended descent is at Mach .89

Maximum range descent is at constant indicated airspeed of 192 knots

Descend at idle power

Speed brakes closed

Formation descent is at Mach .90 or 400 knots KIAS, whichever is less. Descent at 85% RPM, speed brakes closed.



Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A6—1

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Descents

NOTE

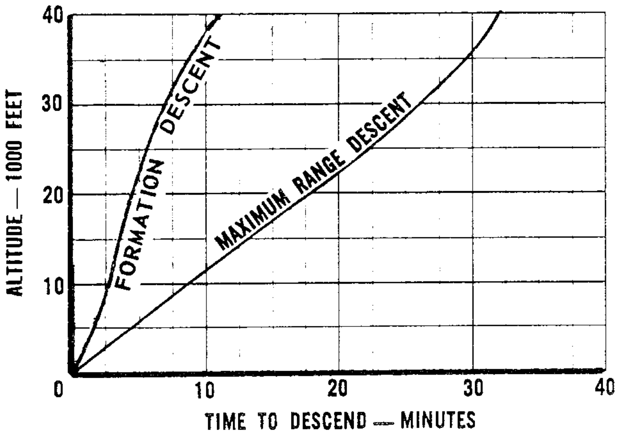
Recommended descent is at Mach .90

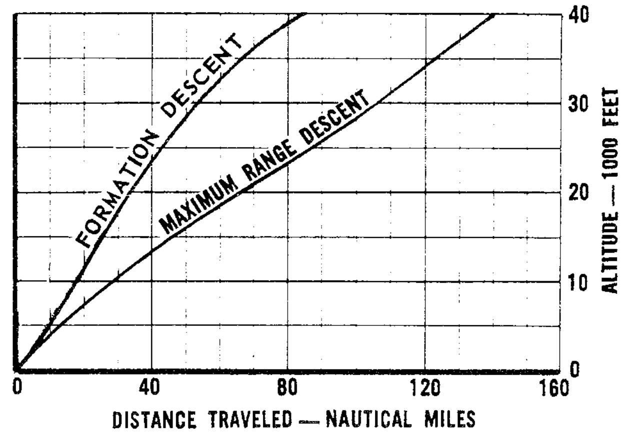
Maximum range descent is at constant indicated airspeed of 190 knots

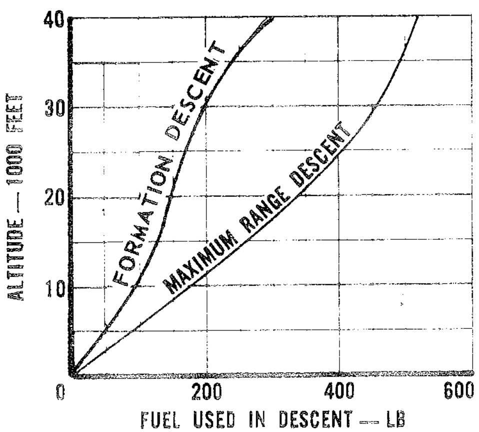
Descend at idle power

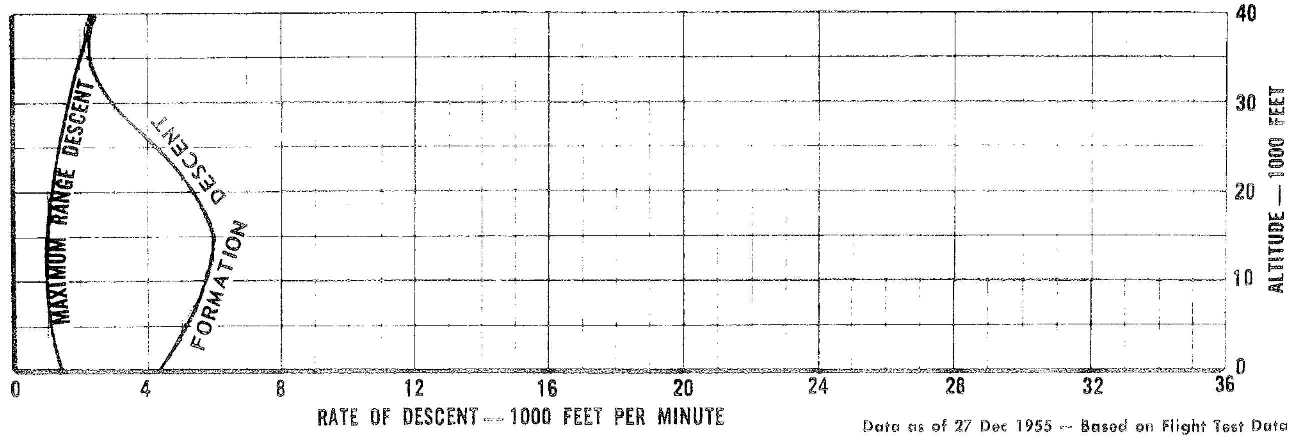
Speed brakes closed

Formation descent is at Mach .90 or 400 knots KIAS, whichever is less. Descent at 85% RPM, speed brakes closed.









Data as of 27 Dec 1955 -- Based on Flight Test Data

Figure A6—2

MODEL: RF-84F ENGINE: J65-3 — STANDARD DAY

Descents

CONFIGURATION: CLEAN + TWO 450 GAL CLASS I TANKS

NOTE

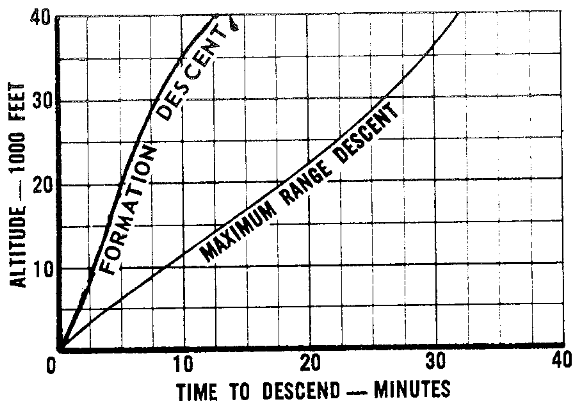
Recommended descent is at Mach .90

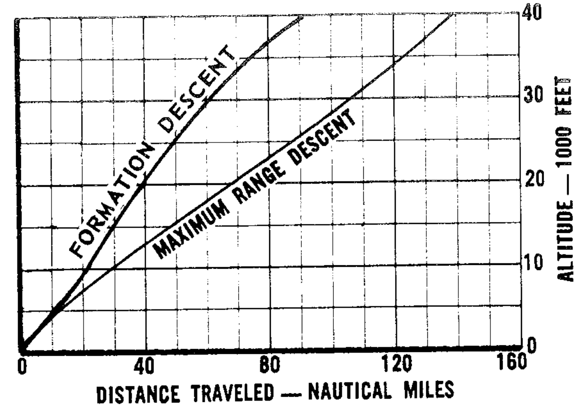
Maximum range descent is at constant indicated airspeed of 190 knots

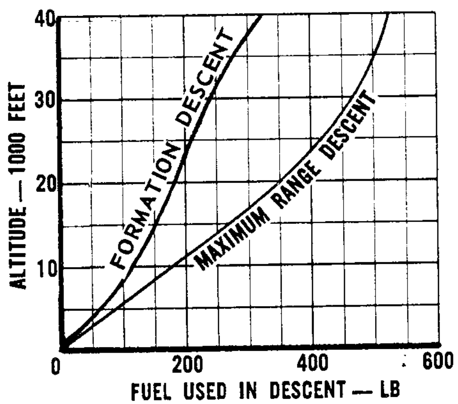
Descend at idle power

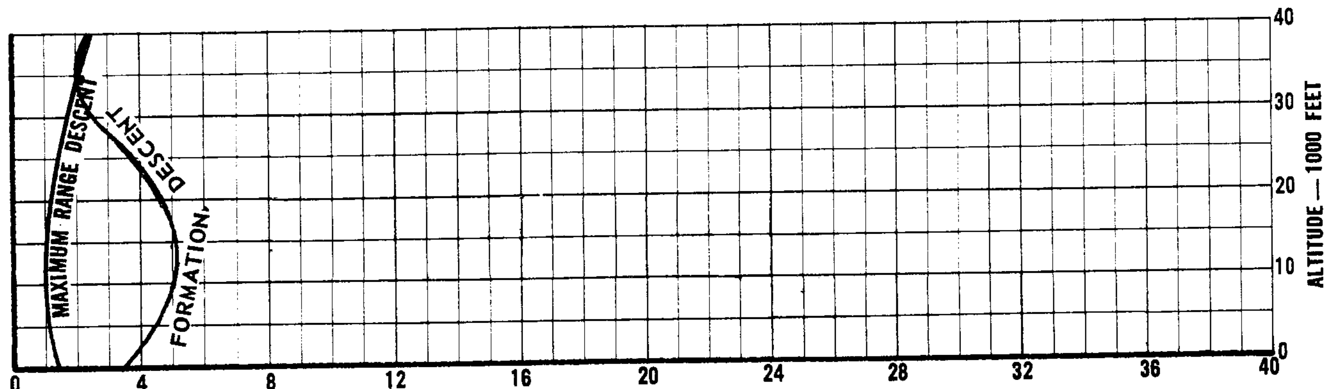
Speed brakes closed

Formation descent is at Mach .90 or 400 knots KIAS, whichever is less. Descent at 85% RPM, speed brakes closed.









Data as of 27 Dec 1955 — Based on Flight Test Data

Figure A6—3

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Descents

CONFIGURATION: CLEAN +
TWO 230 GAL CLASS I TANKS +
TWO 230 GAL CLASS II TANKS

NOTE

Recommended descent is at Mach .88

Maximum range descent is at constant indicated airspeed of 187 knots

Descend at idle power

Speed brakes closed

Formation descent is at Mach .90 or 400 knots KIAS, whichever is less. Descent at 85% RPM, speed brakes closed.

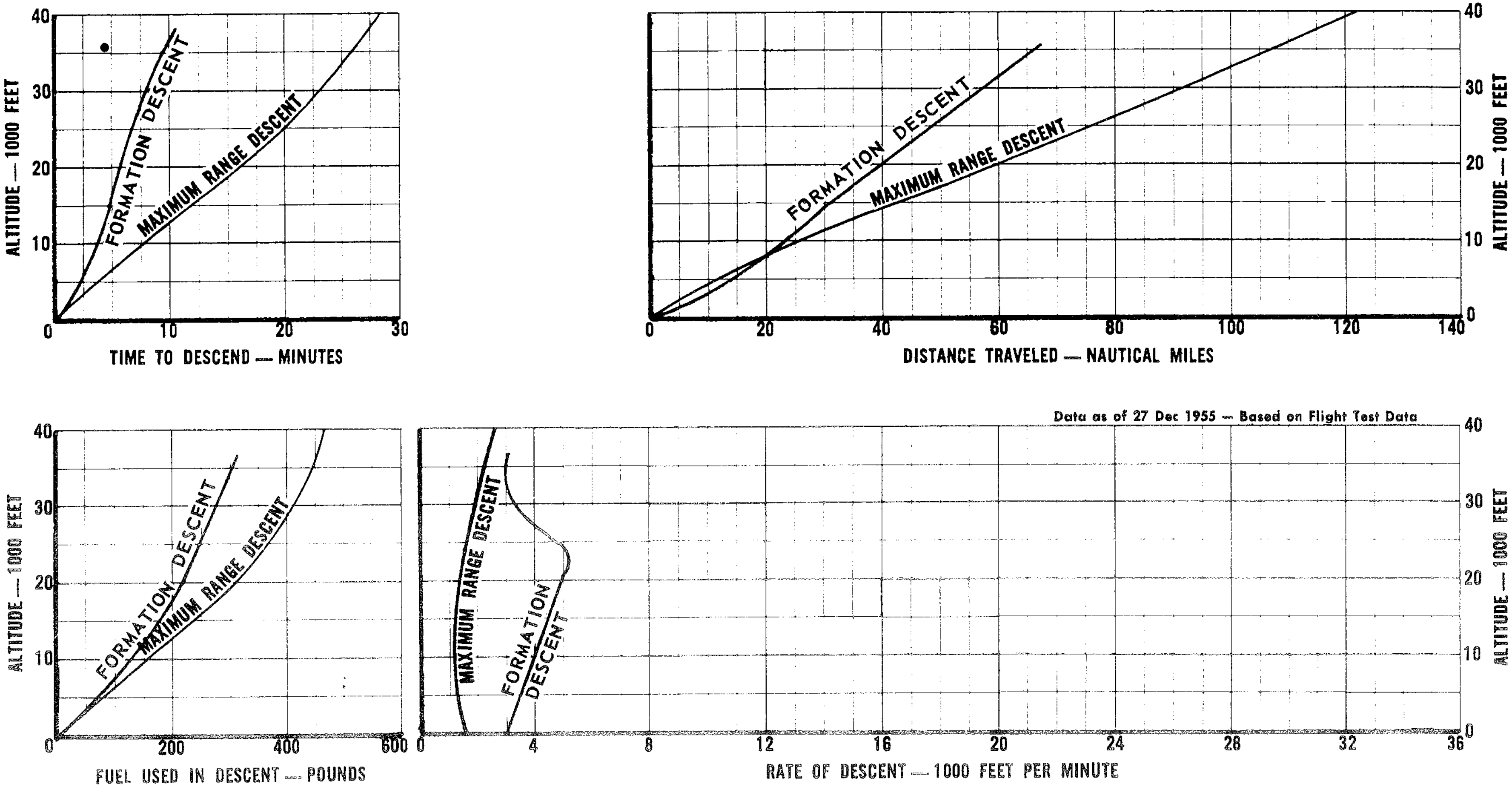


Figure A6—4

PART 7 LANDING

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LANDING DISTANCES.

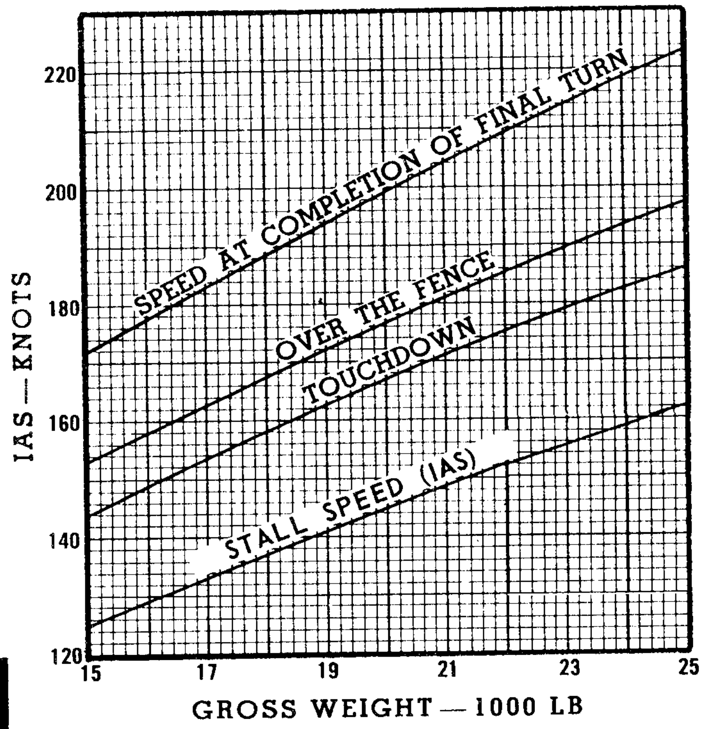
Landing ground roll distances and total distances to clear a 50-foot obstacle are shown in the clean and externally loaded configurations. The distances are computed for a condition with flaps down and speed brakes closed. The following conditions are also considered: dry hard-surface runway; temperature, −20°C thru 60°C (4°F thru 140°F) pressure altitude, sea level through 6000 feet; gross weight, 15,000 through 23,000 pounds; and head winds, 0 through 50 knots. The recommended indicated airspeeds for the approach, over a 50-foot obstacle, and touchdown are listed on the graphs. Following is a sample of the landing data card. For the data card form refer to Section II Abbreviated Check List. Information on this card should be filled out prior to each flight. This data is obtained by referring to the Landing Distance Chart.

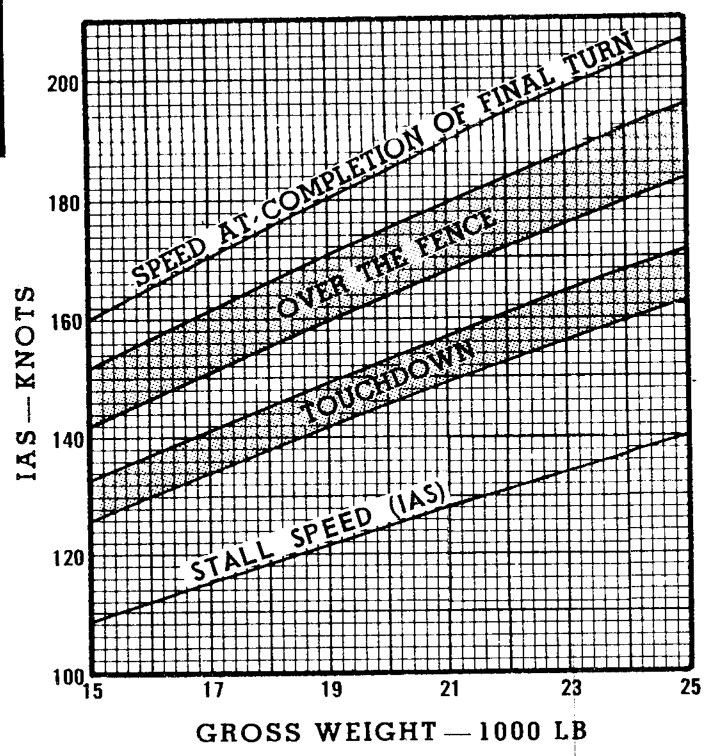
LANDING DATA CARD

Conditions	
Gross Weight	16,500 LB
Runway Length	9,000 FT
OAT	35°C (95°F)
Pressure Altitude	2,000 FT
Runway Gradient	0%
Wind	0 KN
Landing	
Approach Speed	170 KN
Landing Ground Roll (without drag chute)	4,000 FT
Landing Ground Roll (with drag chute)	2,300 FT

APPROACH and LANDING Speeds

FLAPS UP
GEAR DOWN





FLAPS DOWN
GEAR DOWN

Figure A7—1

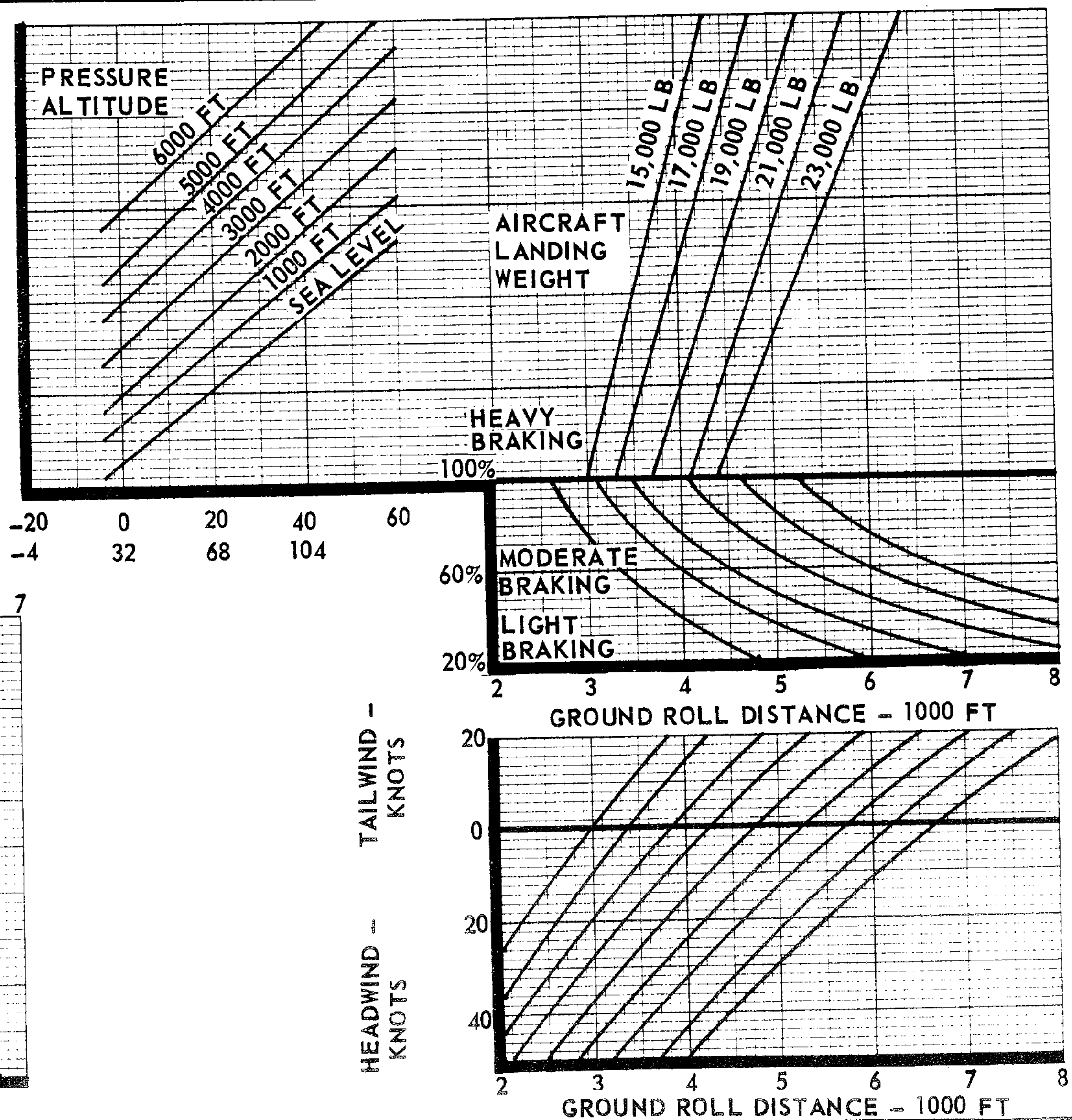
DATA AS OF: 3 JULY 1963
DATA BASIS: FLIGHT TEST – THEORETICAL DATA

LANDING DISTANCES
WITHOUT DRAG CHUTE
SPEED BRAKES CLOSED
AFTER SPEED BRAKES CLOSED, FLAPS UP

DRY: HARD SURFACE RUNWAY RF-84F
J65-W-3, B-3 W-7, B-7 IDLE POWER

LANDING SPEEDS

GROSS WEIGHT LB	IAS – KNOTS FINAL APPROACH	50 FT OBSTACLE	TOUCH-DOWN
15,000	160	140	125
17,000	170	150	135
19,000	180	160	140
21,000	190	170	150
23,000	200	175	155
25,000	210	185	165



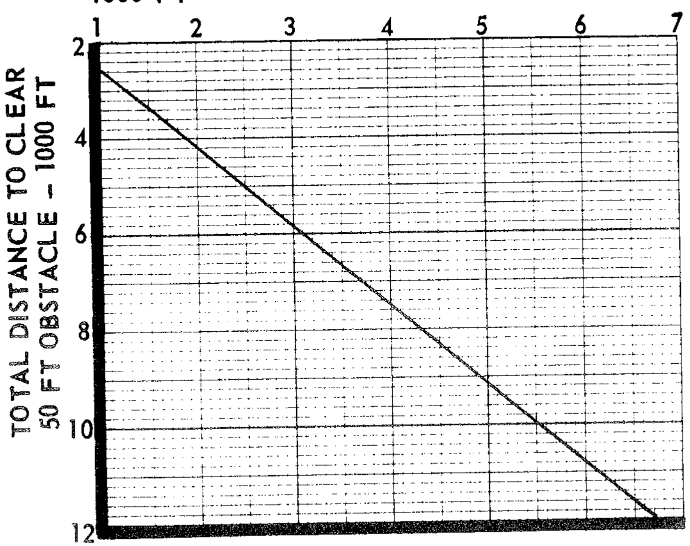


Figure A7-2

DATA OF OF : 3 JULY 1963
DATA BASIS: FLIGHT TEST – THEORETICAL DATA

LANDING DISTANCE
DRAG CHUTE INFLATED AT TOUCHDOWN
FLAPS DOWN – SPEED BRAKES CLOSED

DRY: HARD SURFACE RUNWAY RF-84F
J65-W-3, B-3, W-7, B-7 IDLE POWER

LANDING SPEEDS

GROSS WEIGHT LB	IAS KNOTS FINAL APPROACH	50 FT OBSTACLE	TOUCH-DOWN
15,000	160	140	125
17,000	170	150	135
19,000	180	160	140
21,000	190	170	150
23,000	200	175	155
25,000	210	185	165

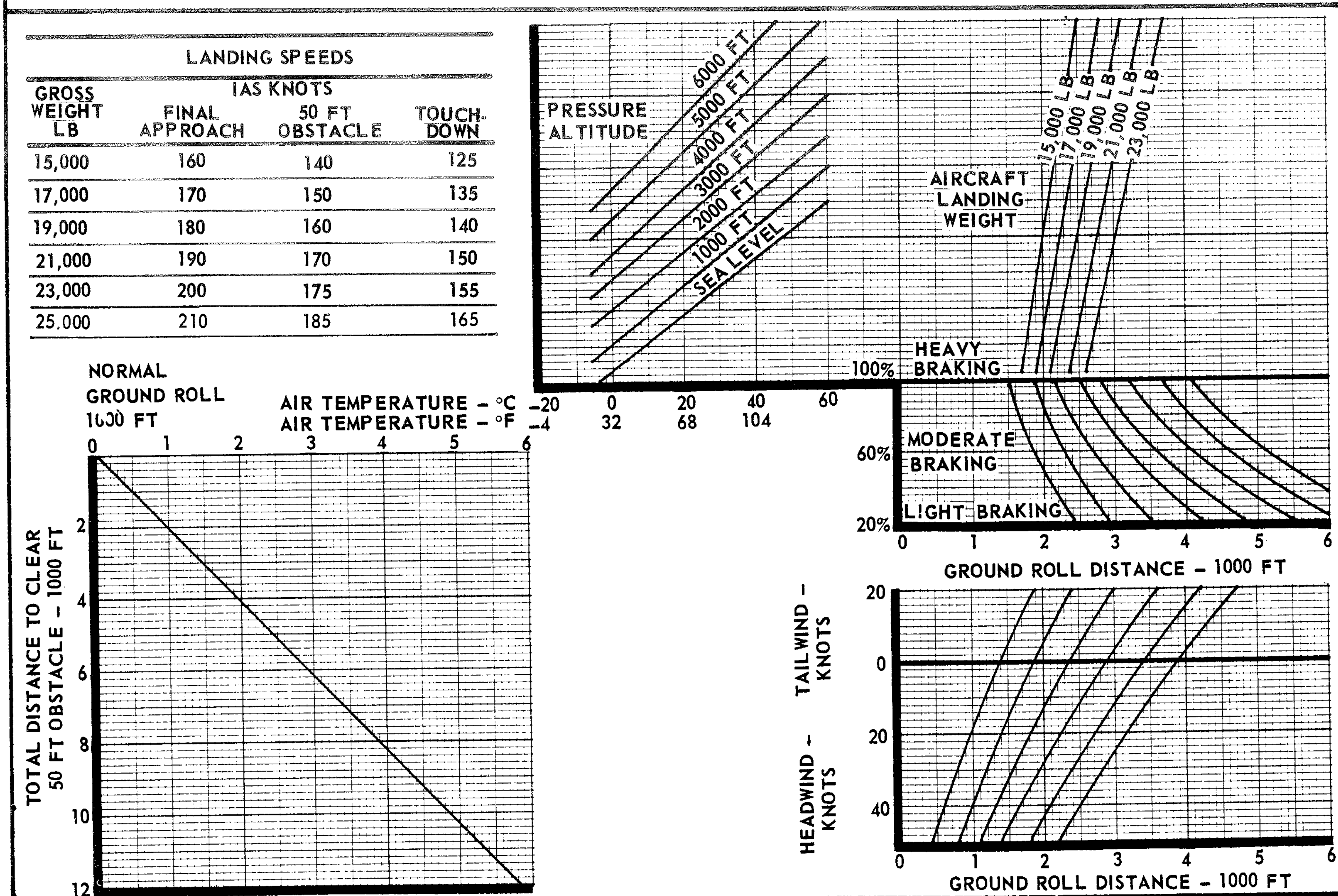


Figure A7-3

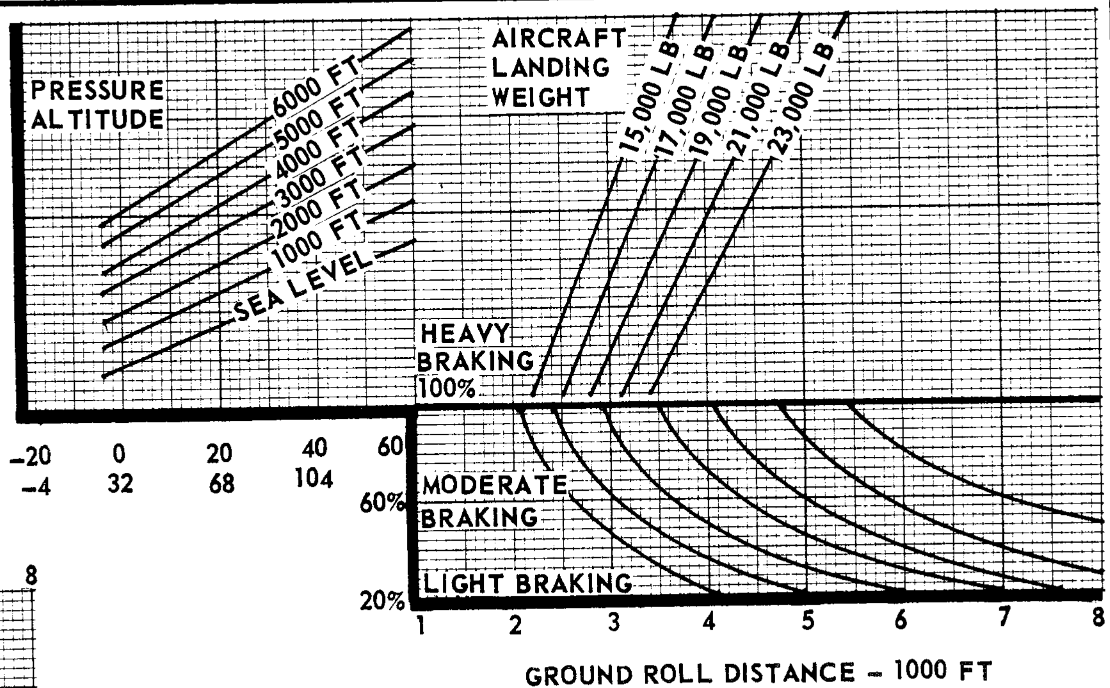
DATA AS OF: 3 JULY 1963
DATA BASIS: THEORETICAL DATA

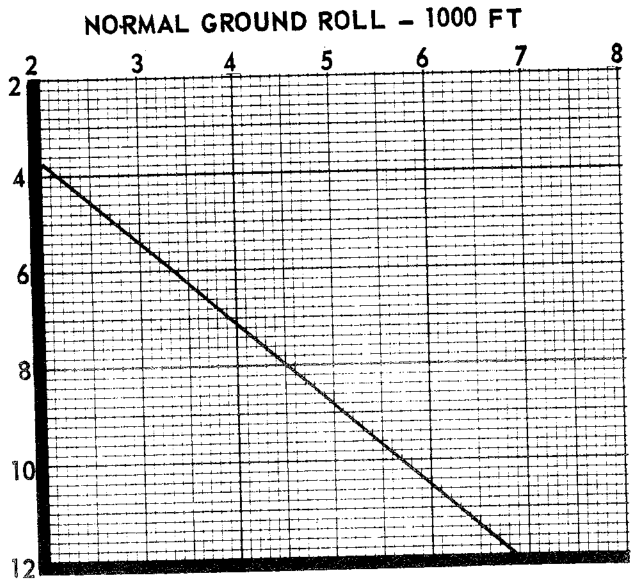
LANDING DISTANCES
WITHOUT DRAG CHUTE
FLAPS DOWN – SPEED BRAKES OPEN

DRY: HARD SURFACE RUNWAY RF-84F
J65-W-3 B-3, W-7 B-7 IDLE POWER

LANDING SPEEDS

GROSS WEIGHT LB	IAS KNOTS FINAL APPROACH	50 FT OBSTACLE	TOUCH-DOWN
15,000	160	140	125
17,000	170	150	135
19,000	180	160	140
21,000	190	170	150
23,000	200	175	155
25,000	210	185	165





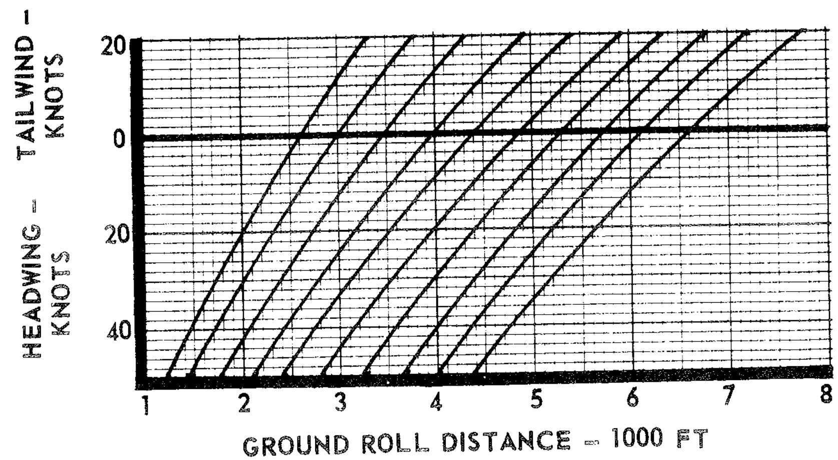


Figure A7-4

DATA AS OF: 3 JULY 1963
DATA BASIS: THEORETICAL DATA

RUNWAY CONDITION
LANDING DISTANCE CHART
FLAPS DOWN
WITHOUT DRAG CHUTE
SPEED BRAKES CLOSED

RF-84F
ENGINE: J65-W-3, B-3, W-7, B-7
IDLE POWER

When other than dry conditions exist on active runways, base operations officers are responsible for determining and relaying to the base weather station the type of runway covering and the relative slickness of the runway as determined by the James Brake Decelerometer. This information is transmitted as part of the teletype weather sequence. This number will be either a one or two digit number and is referred to as the runway condition reading. This number will be followed by the letter "P" if the runway is patchy. A report of SLR14P would indicate slush on the runway, RCR of 14, and patchy conditions.

INSTRUCTIONS: CORRECT YOUR STOPPING DISTANCE BY ENTERING CHART WITH PLANNED GROUND ROLL, MOVE VERTICALLY TO THE LATEST RCR, THEN HORIZONTALLY TO THE LEFT TO OBTAIN THE RCR GROUND ROLL.

EXPLANATION OF TERMS

RCR – RUNWAY CONDITION READING
P – PATCHY
WR – WET RUNWAY
SLR – SLUSH ON RUNWAY
LSR – LOOSE SNOW ON RUNWAY
PSR – PACKED SNOW ON RUNWAY
IR – ICE ON RUNWAY

NOTE

IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS. FOR ICAO REPORT OF GOOD, USE RCR23; FOR MEDIUM USE RCR12; AND FOR POOR RCR5.

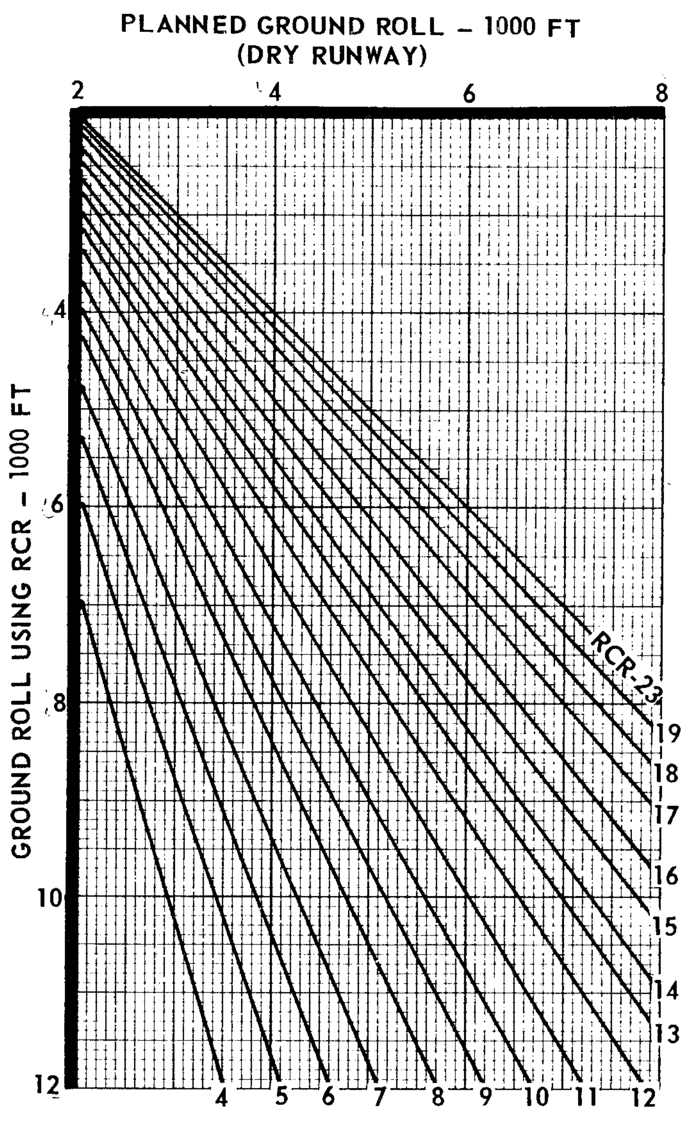


Figure A7-5

DATA AS OF: 3 JULY 1963
DATA BASIS: THEORETICAL DATA

RUNWAY CONDITION
LANDING DISTANCE CHART
SPEED BRAKES CLOSED
WITH DRAG CHUTE

RF-84F
ENGINE: J65-W-3, B-3, W-7, B-7
IDLE POWER

When other than dry conditions exist on active runways, base operations officers are responsible for determining and relaying to the base weather station the type of runway covering and the relative slickness of the runway as determined by the James Brake Decelerometer. This information is transmitted as part of the teletype weather sequence. This number will be either a one or two digit number and is referred to as the runway condition reading. This number will be followed by the letter "P" if the runway is patchy. A report of SLR14P would indicate slush on the runway, RCR of 14, and patchy conditions.

INSTRUCTIONS: CORRECT YOUR STOPPING DISTANCE BY ENTERING CHART WITH PLANNED GROUND ROLL, MOVE VERTICALLY TO THE LATEST REPORTED RCR, THEN HORIZONTALLY TO THE LEFT TO OBTAIN THE RCR GROUND ROLL.

EXPLANATION OF TERMS

RCR – RUNWAY CONDITION READING
P – PATCHY
WR – WET RUNWAY
SLR – SLUSH ON RUNWAY
LSR – LOOSE SNOW ON RUNWAY
PSR – PACKED SNOW ON RUNWAY

NOTE

IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS. FOR ICAO REPORT OF GOOD, USE RCR23; FOR MEDIUM USE RCR12, AND FOR POOR USE RCR5.

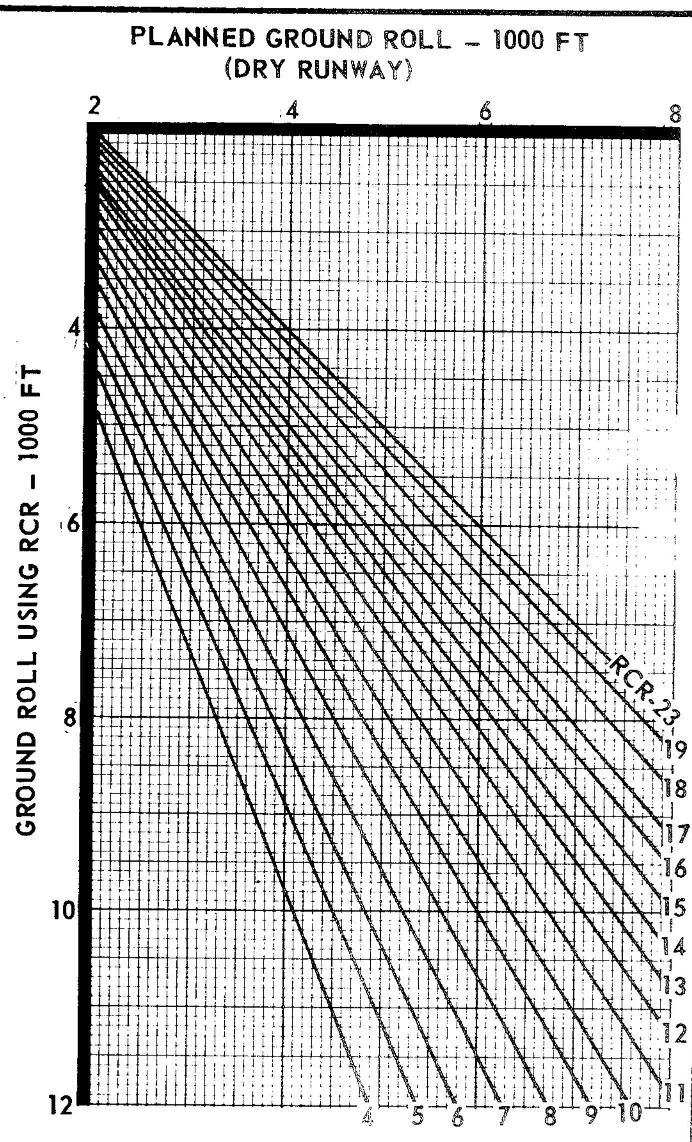


Figure A7-6

DATA AS OF: 3 JULY 1963 DATA BASIS: THEORETICAL DATA	RUNWAY CONDITION LANDING DISTANCE CHART SPEED BRAKES OPEN WITHOUT DRAG CHUTE	RF-84F ENGINE: J65-W-3, B-3, OR W-7, B-7 IDLE POWER

When other than dry conditions exist on active runways, base operations officers are responsible for determining and relaying to the base weather station the type of runway covering and the relative slickness of the runway as determined by the James Brake Decelerometer. This information is transmitted as part of the teletype weather sequence. This number will be either a one or two digit number and is referred to as the runway condition reading. This number will be followed by the letter "P" if the runway is patchy. A report of SLR14P would indicate slush on the runway, RCR of 14, and patchy conditions.

INSTRUCTIONS: CORRECT YOUR STOPPING DISTANCE BY ENTERING CHART WITH PLANNED GROUND ROLL, MOVE VERTICALLY TO THE LATEST RCR, THEN HORIZONTALLY TO THE LEFT TO OBTAIN THE RCR GROUND ROLL.

EXPLANATION OF TERMS

RCR – RUNWAY CONDITION READING
P – PATCHY
WR – WET RUNWAY
SLR – SLUSH ON RUNWAY
LSR – LOOSE SNOW ON RUNWAY
PSR – PACKED SNOW ON RUNWAY
IR – ICE ON RUNWAY

NOTE

IF NO RCR IS AVAILABLE, USE 12 FOR WET RUNWAYS AND 5 FOR ICY RUNWAYS. FOR ICAO REPORT OF GOOD, USE RCR23: FOR MEDIUM USE RCR12, AND FOR POOR USE RCR5.

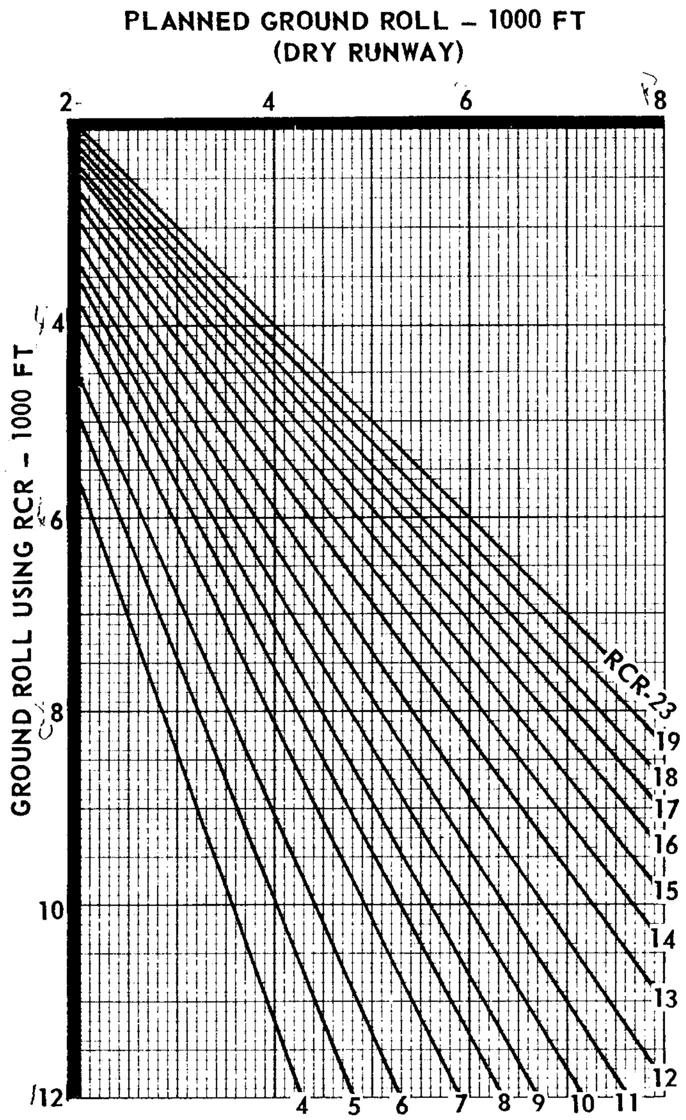


Figure A7-7

PART 8 COMBAT PERFORMANCE

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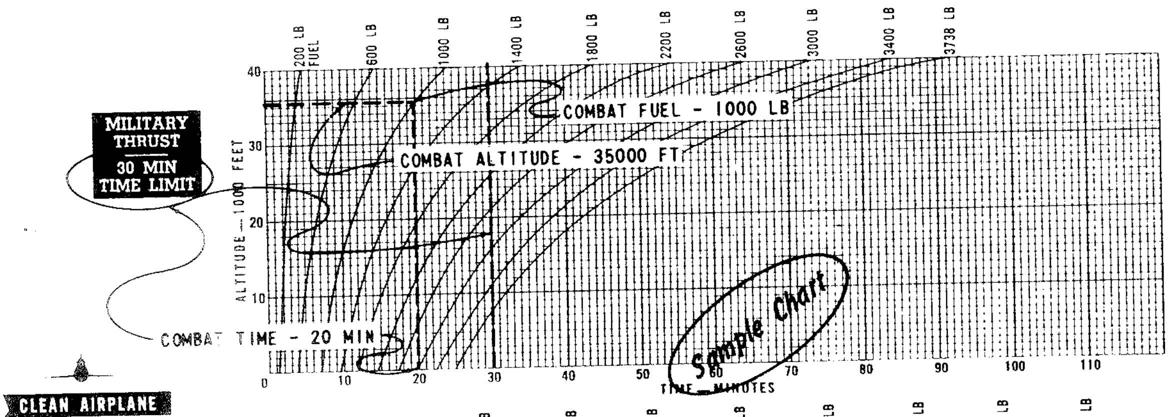
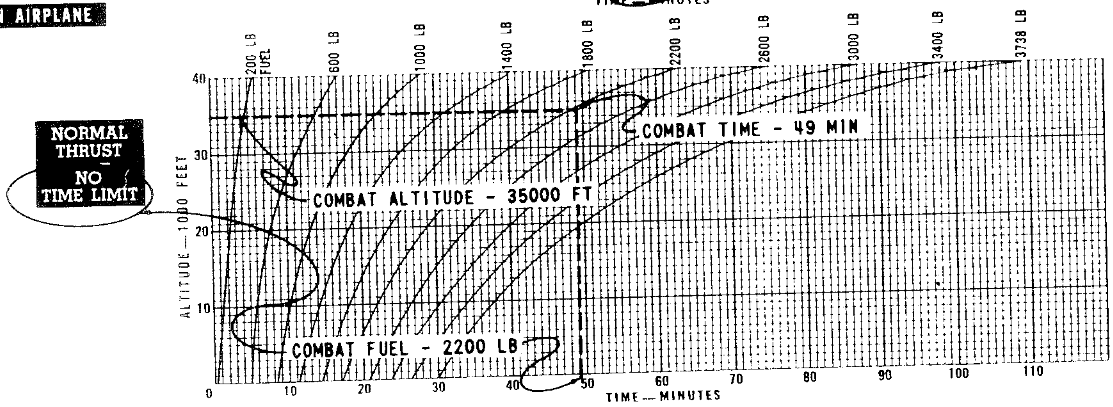
COMBAT ALLOWANCE CHART

MODEL: RF-84F
ENGINE:
STANDARD DAY





DESCRIPTION.

The Combat Allowance chart shows the relationship between time and fuel with changes in altitude at Military and Normal Thrust settings. Combat time or fuel may be determined from this chart for a given thrust setting. The time limitations for Military Thrust operation are shown. Normal thrust does not have a time limitation.

USE.

Enter the chart at the combat altitude and the fuel quantity to be used for combat to obtain the time available. Enter at the altitude and time available for combat to obtain the fuel required.

Using the example shown, obtain the time available for a combat fuel allowance of 2,200 pounds at Normal Thrust and 1,000 pounds at Military Thrust at 35,000 feet.

Military thrust — 20 min
Normal thrust — 49 min

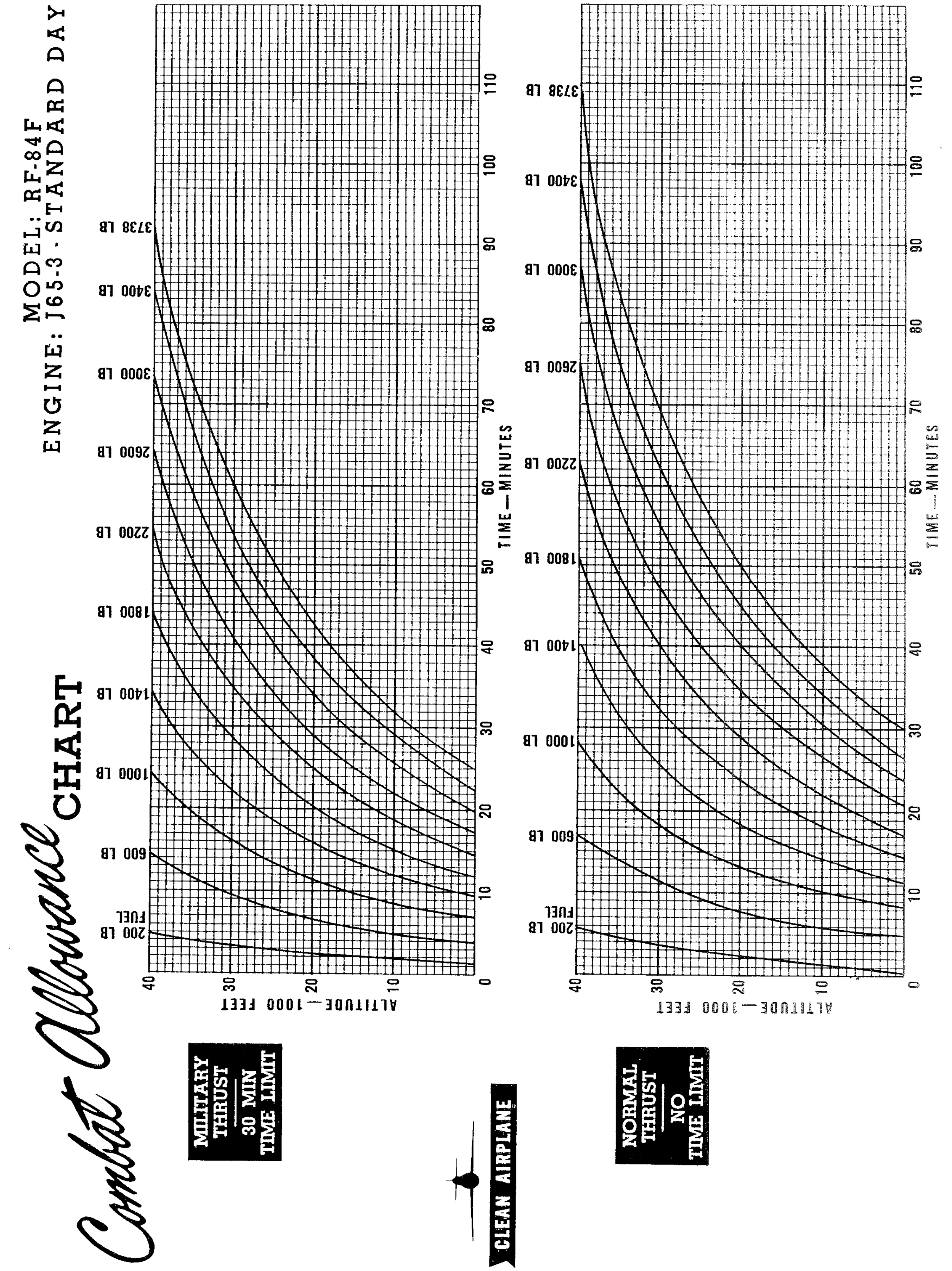

Combat Allowance CHART
MODEL: RF-84F
ENGINE: J65-3 · STANDARD DAY
MILITARY THRUST
30 MIN TIME LIMIT
NORMAL THRUST
NO TIME LIMIT
CLEAN AIRPLANE
200 LB FUEL
600 LB
1000 LB
1400 LB
1800 LB
2200 LB
2600 LB
3000 LB
3400 LB
3738 LB
ALTITUDE—1000 FEET
TIME — MINUTES
0
10
20
30
40
50
60
70
80
90
100
110

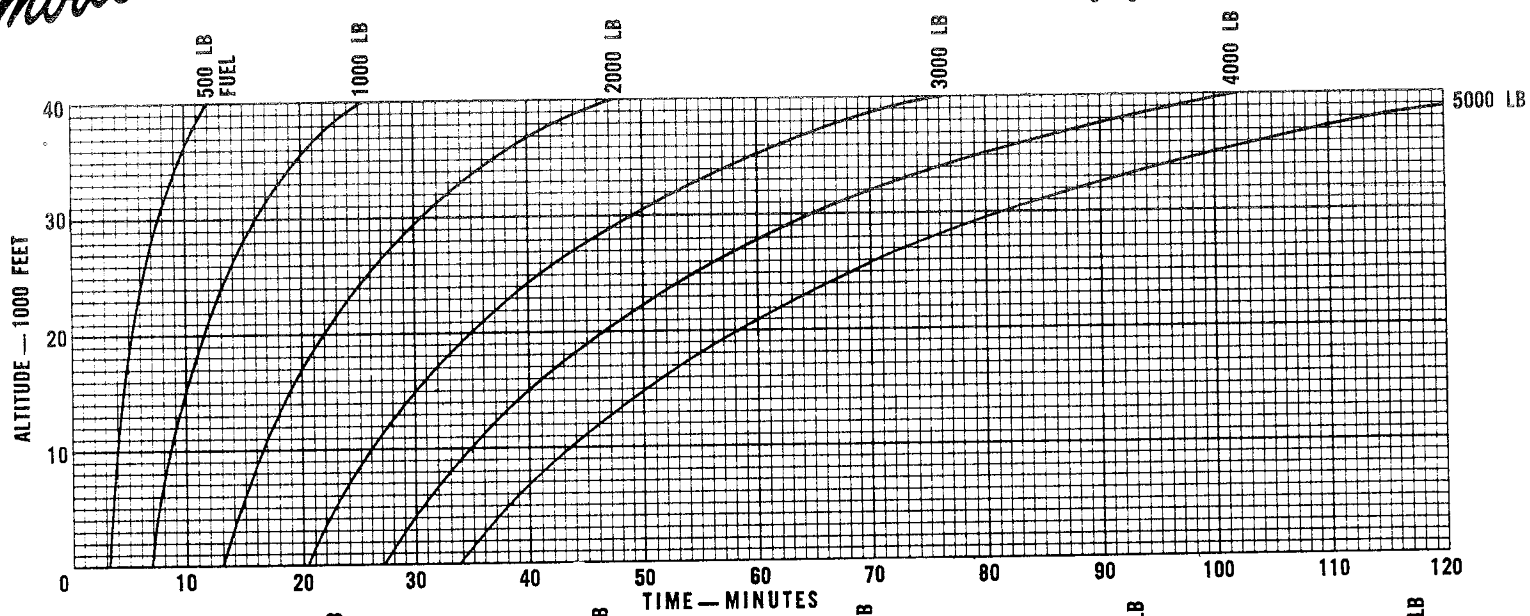
Figure A8—1

Combat Allowance CHART

MODEL: RF-84F ENGINE: J65-3 — STANDARD DAY

CONFIGURATION: CLEAN + TWO 230 GAL CLASS I TANKS

MILITARY THRUST
30 MIN TIME LIMIT



NORMAL THRUST
NO TIME LIMIT

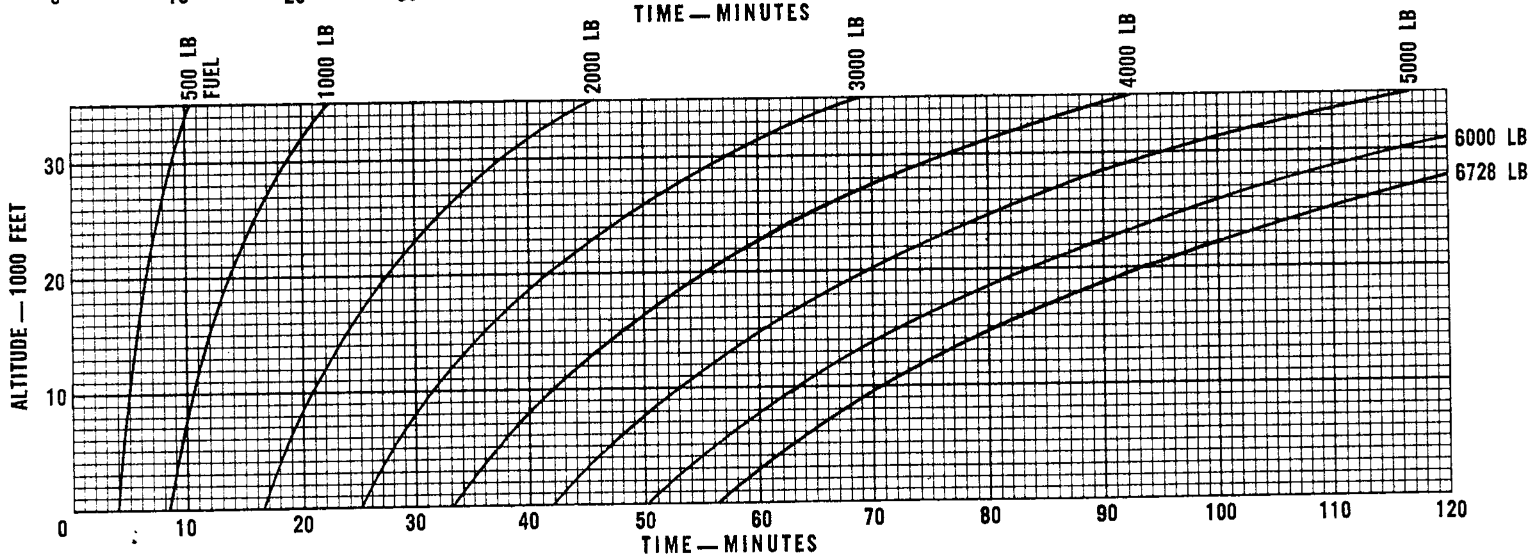


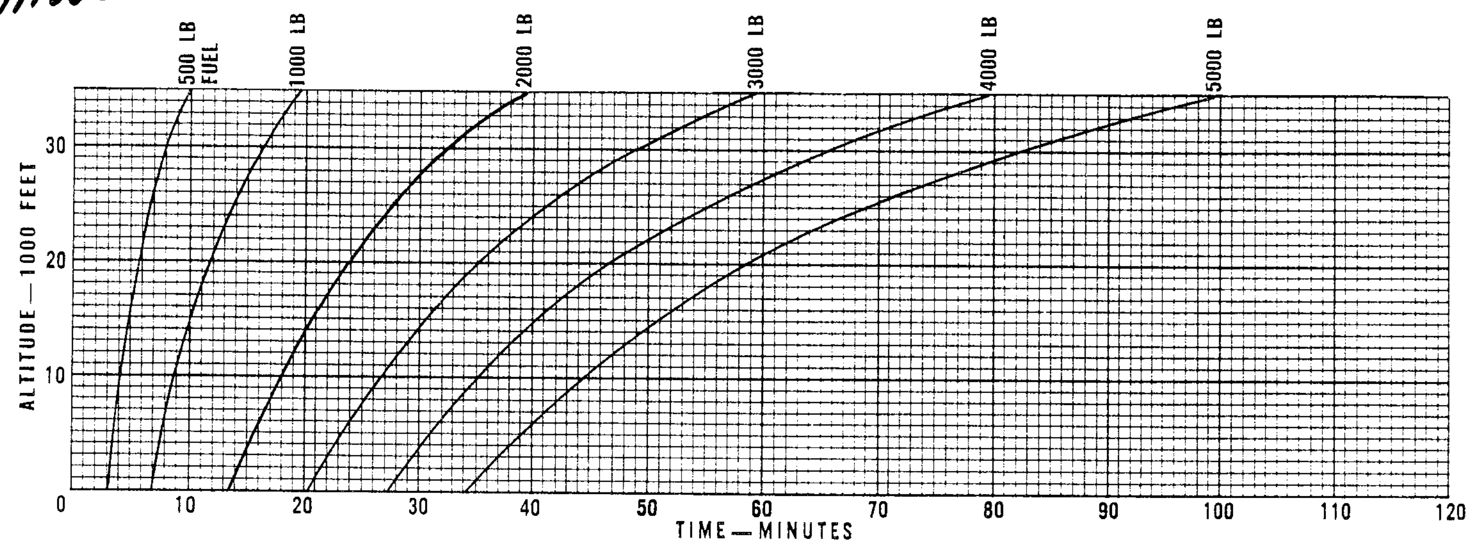
Figure A8—2

MODEL: RF-84F - ENGINE: J65-3 — STANDARD DAY

Combat Allowance CHART

CONFIGURATION: CLEAN + TWO 450 GAL CLASS I TANKS

MILITARY THRUST
30 MIN TIME LIMIT



NORMAL THRUST
NO TIME LIMIT

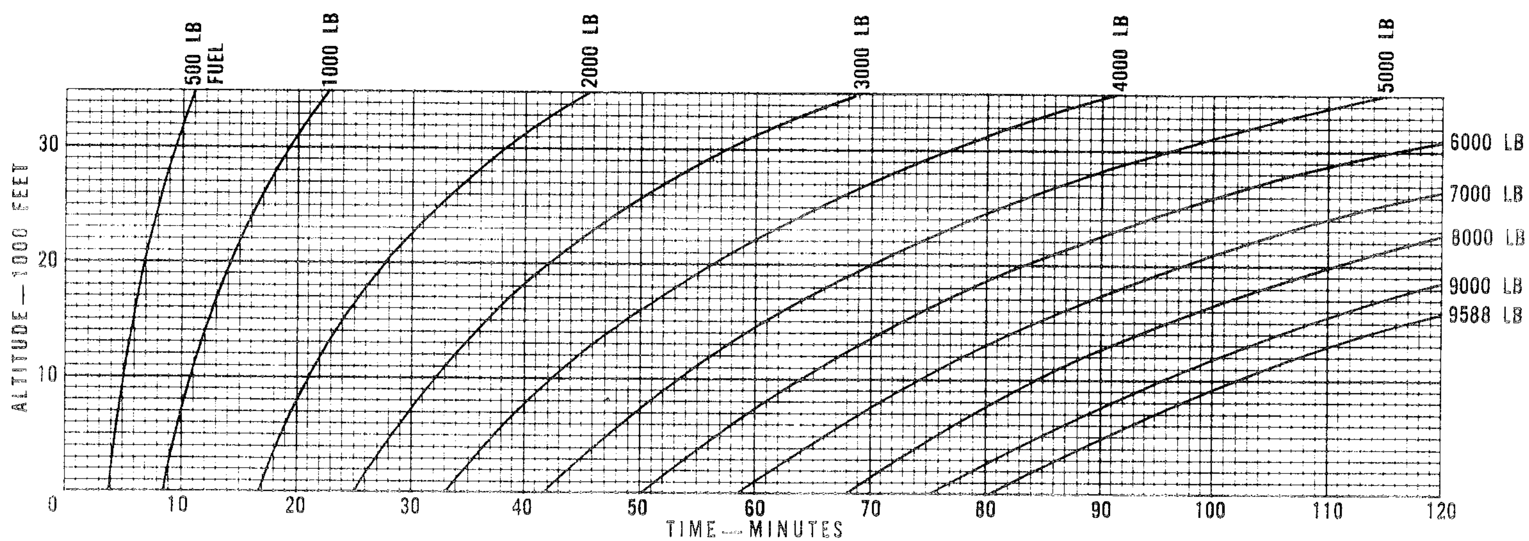


Figure A8—3

PART 9 MISSION PLANNING

SUMMARY.

Check your flight plan during the actual flight to determine whatever deviations exist. These deviations may be applied to the reserve expected at the destination. The most important factors to consider are:

Fuel used during start, taxi, and take-off. (The profile allows 465 pounds for this phase.)

Wind effect.

Deviation from the recommended climb schedule.

Deviation from the recommended cruise settings.

Variation in engine performance.

Navigational errors, formation flight, and fuel actually aboard at take-off.

SAMPLE PROBLEM.

This sample problem combines the use of the charts and graphs in this section to plan a mission.

A combat mission is to be flown using two 230 gallon drop tanks which are to be dropped prior to combat. Prepare a flight plan based on the following data:

Distance to combat area	450 nautical miles
Winds	30 knot head wind to combat area 45 knot tailwind from combat area
Time required at Military Thrust for combat	10 MIN
Total fuel on board	6728 LB

TAKE-OFF.

Obtain the take-off distance from figure A2–1.

Altitude	Sea level
Temperature	15°C (59°F)
Gross weight	22,410 LB
Wind	None
Ground roll distance	5000 FT
Total distance to clear 50-foot obstable	6350 FT
Take-off speed	157 knots IAS

INBOUND TO TARGET.

The inbound leg to the combat area may be determined directly from the Mission Profile chart for two 230 gallon drop tanks. The profile includes a 465 pound fuel allowance for start, taxi, and take-off, as well as fuel required for climb and cruise.

a. Distance	450 nautical miles
b. Cruise altitude	35,000 FT
c. Fuel required (no wind) from profile	3340 LB
d. Time (no wind) from profile	1 HR 1 MIN
e. Average TAS (a ÷ d)	442 KN
f. Ground speed (e — 30 knots)	412 KN
g. Time with wind (a ÷ f)	1 HR 6 MIN
h. Fuel required (with wind) from profile	3500 LB
i. Cruise speed	0.77 Mach NO.
j. Cruise power setting	92% RPM
k. Military Thrust climb speed schedule	(See figure A3–4)

COMBAT ALLOWANCE.

The tanks are dropped during cruise, prior to entering combat. From the Combat Allowance chart (figure A8–1), obtain the fuel required for combat at 35,000 feet.

Combat — Military Thrust (10 minutes)	500 LB

Determine the weight and fuel remaining at the end of combat.

Take-off, climb, and cruise fuel	3500 LB
Combat fuel	500 LB
Total fuel used	4000 LB
Fuel remaining (6728 pounds —4000 pounds)	2728 LB
Two 230 gallon drop tanks	570 LB
Gross weight (22,410 pounds —4000 pounds —570 pounds)	17,840 LB

RETURN.

Assume the return is started 500 nautical miles from base at an altitude of 25,000 feet. Use the Optimum Return Profile for the clean configuration.

a. Distance	500 nautical miles
b. Altitude	25,000 FT
c. Fuel required (no wind) from profile	1950 LB
d. Cruise altitude	40,000 FT
e. Cruise time	56 MIN
f. Time to climb at Military Thrust — clean (25,000 feet to 40,000 feet) gross weight = 17,840 pounds	8 MIN
g. Total time (e + f)	1 HR 4 MIN
h. Average TAS (a ÷ g)	469 KN
i. Ground speed (h + 45 knots)	514 KN
k. Cruise time (j — f)	50.4 MIN
j. Total time with wind (a ÷ i)	50.4 MIN
l. Fuel required (with wind) from profile	1735 LB
m. Cruise speed	0.83 Mach NO.
n. Cruise power setting	95% RPM
o. Fuel remaining over base (2728 pounds —l)	993 LB

DESCENT.

Obtain the fuel required to descend to a sea level base from 40,000 feet. (See figure A—40.)

Recommended descent fuel required	80 LB
Fuel reserve for landing (993 pounds — 80 pounds)	918 LB
Aircraft weight for landing	16,130 LB

LANDING.

Use Landing Distances chart (figure A—45) for clean configuration.

Altitude	Sea level
Temperature	15°C (60°F)
Gross weight	16,130 LB
Ground roll distance	3420 FT
Total distance over 50-foot obstacle	5520 FT
Approach speed	165 KN IAS
Over 50-foot obstacle speed	145 KN IAS
Touchdown speed	130 KN IAS

ALPHABETICAL INDEX

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